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SCHOOL OF MECHANICAL ENGINEERING

DEPARTMENT OF AERONAUTICAL ENGINEEING

SAE1307- Aerospace Propulsion

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.

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

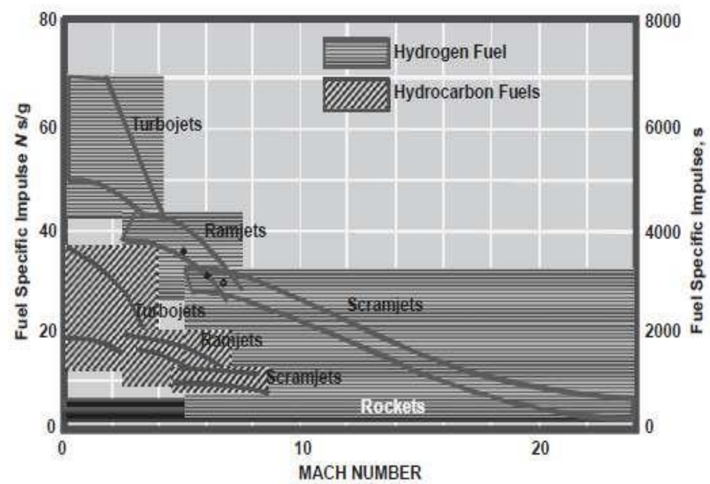


Fig:1.1 Mach 3 flight :

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,

$\frac{P_0}{P}$ = $[1+ \frac{\gamma+1}{2} M^2$] 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).2.5The 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

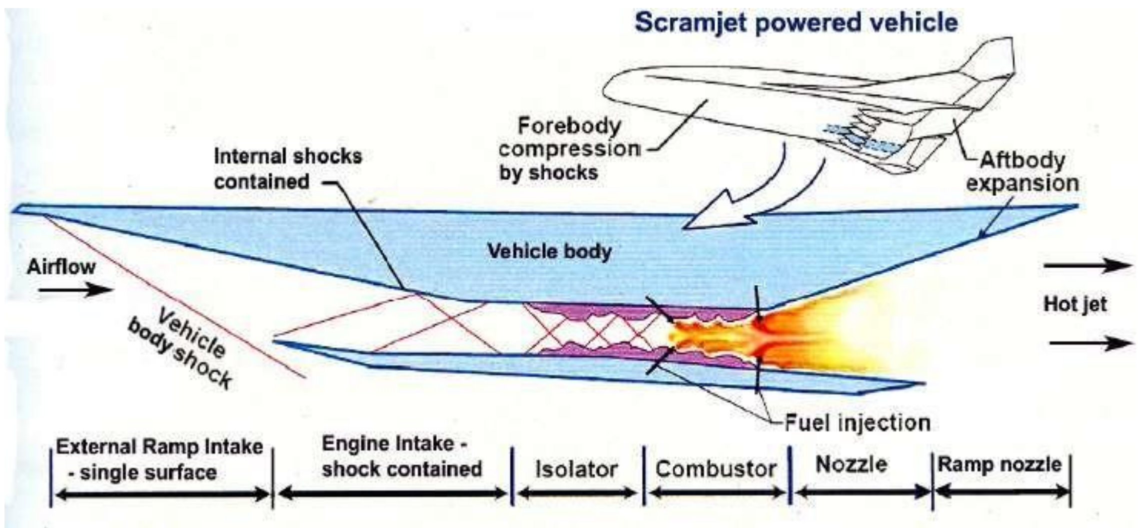
- Isolator
- Supersonic combustor
- Exhaust nozzle or aft body expansion component
- 

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

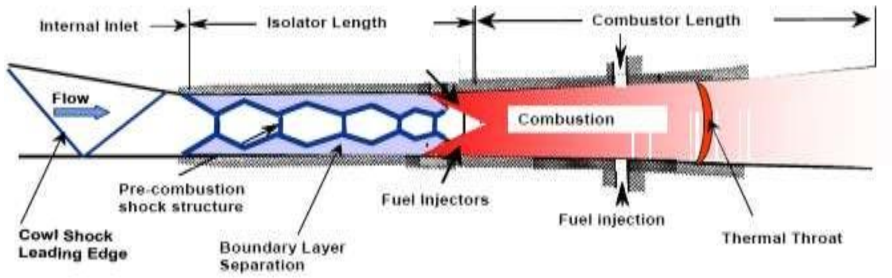


Fig: 1.4 combustor

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 of switching 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 ($\sim \sqrt{T}$) 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:

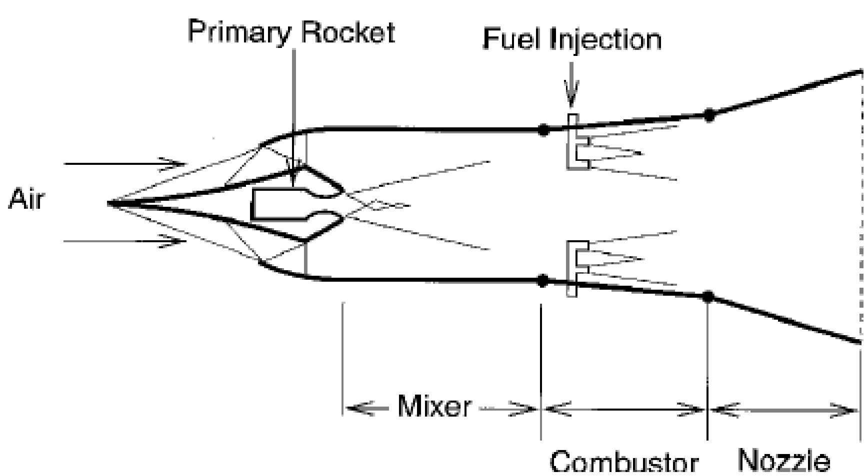


Fig: 1.5 rocket subsystem

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.

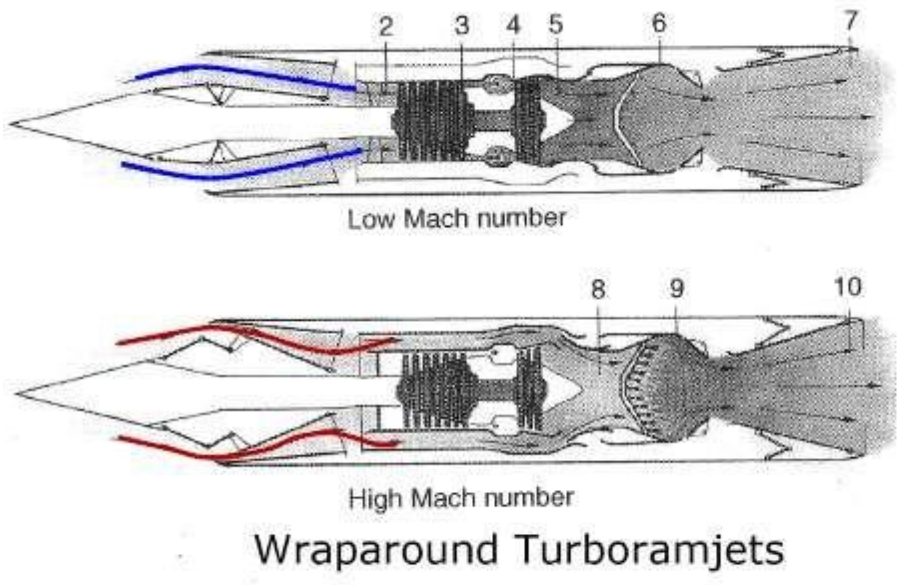
Higher I_{sp} in rocket mode: In the rocket-only mode, the use of the engine duct as a highly expanded nozzle at high altitudes increases the specific impulse of that mode of operation.

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.

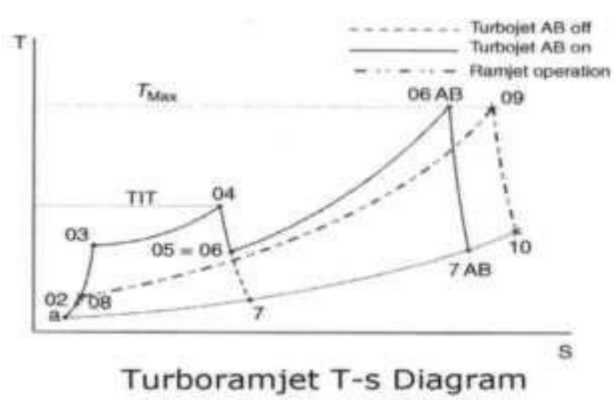


Fig: 1.7 Turboramjet t-s diagram

Air Turbo Ram Jet

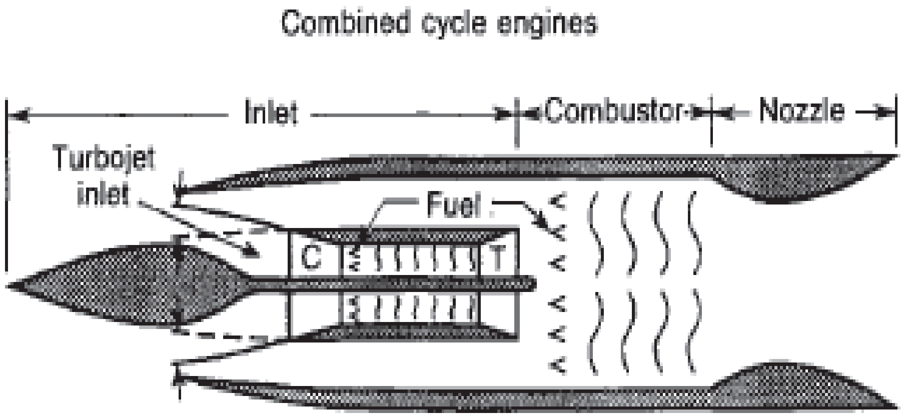


Fig: 1.8 turbo ramjet

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:

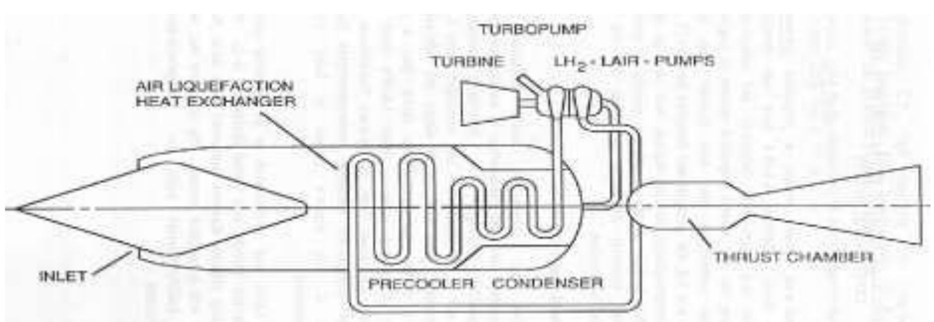


Fig: 1.9 Liquid Air Cycle Engine

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.

Rocket Engine

Chemical Rocket:

Classification of Rocket Engine Propulsion Systems:

All classical propulsive systems depend on conservation of momentum. This principle is used in many ways to create thrust. Majority of systems 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 do not need even any propellant at all. The propellant also need not be carried with the spacecraft.

The performance of the propulsive system depends on total mass of the spacecraft and on the speed of the propellant.

Hence, all propulsion systems which reduce the need of propellant making the spacecraft 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 (solid propellant, 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/Internal Energy: Nuclear, Air breathing propellant-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 reaction of propellant chemicals usually fuel and an oxidizing chemical. The reaction product gases are at very high temperatures (2500to4100°C). These gases are subsequently expanded in a nozzle and accelerated to high velocities (100 to 4300 m/sec). Since theses gas temperatures are about twice the melting point of steel, it is necessary to cool or insulate all the surfaces that are exposed to hot gases.

Engine Types: According to different physical state of the propellants, the types of chemical rocket propulsion devices is as follows:

1. **Liquid Propellant Rocket Engines:** Use liquid oxidizer and fuel, which are fed under pressure from tanks into a thrust chamber.

- **Liquid Bi-propellant engine** uses liquid oxidizer and a liquid fuel (eg: liquid oxygen and kerosene)
- **Liquid Monopropellant** uses a single liquid that contains both oxidizing and fuel species; which decomposes into hot gas during combustion.

The liquid propellant rockets are also classified based on type 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), while pump fed systems are used in high thrust applications such as space launch vehicles etc.

2. **Solid Propellant Rocket Engine**: The solid propellant rocket engines burn a grain of solid propellant within the combustion chamber or case. The solid propellant charge, called grain contains all chemical elements including oxidizer and fuel for complete burning. The resulting hot gases expand through a supersonic nozzle and impart thrust. There are no feed systems or valves.

3. **Gaseous Propellant Rocket Engines**: They use stored high pressure gas such as hydrozen, helium etc as propellant. These are usually cold gas engine systems used for attitude control systems for space vehicles. Heating of the gas by electrical energy or by

combustion of a monopropellant improves the performance and such systems 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**: A ducted rocket or air-augmented rocket combines principles of rocket and ramjet engines. The ducted propulsion system provides better performance (specific impulse) than the chemical rocket, while operating within the earth'satmosphere.

Working Principle:

Liquid Propellant Rocket System: Liquid propellants are used in this system, which are fed into the combustion chamber under pressure. The liquid oxidizer and liquid fuel are stored in separate tanks.

A high pressure gas pressure tank provides pressure feed of oxidizer and fuel through diaphragms. Alternatively, separate pumps may be used to provide pressure feeding of propellants. The propellants react in the thrust chamber and generate hot combustion gases which are expanded in the supersonic convergent divergent nozzle. The system permits repetitive use and can be started and shut off, as required. It is possible to operate the rocket for long durations, exceeding 1 hour by providing adequate cooling of the thrust chamber and C-D nozzle.

A liquid propellant rocket propulsion system requires several precision valves, complex feed mechanism including pumpsetc.

A schematic diagram is as follows:

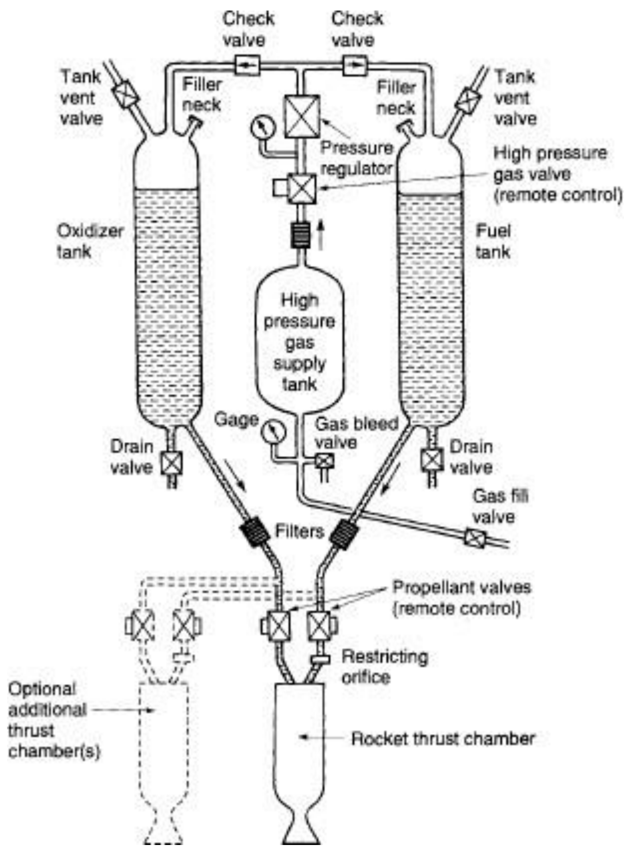


Fig: 2.1 **Solid Propellant Rocket Propulsion System**: A schematic diagram is shown below:

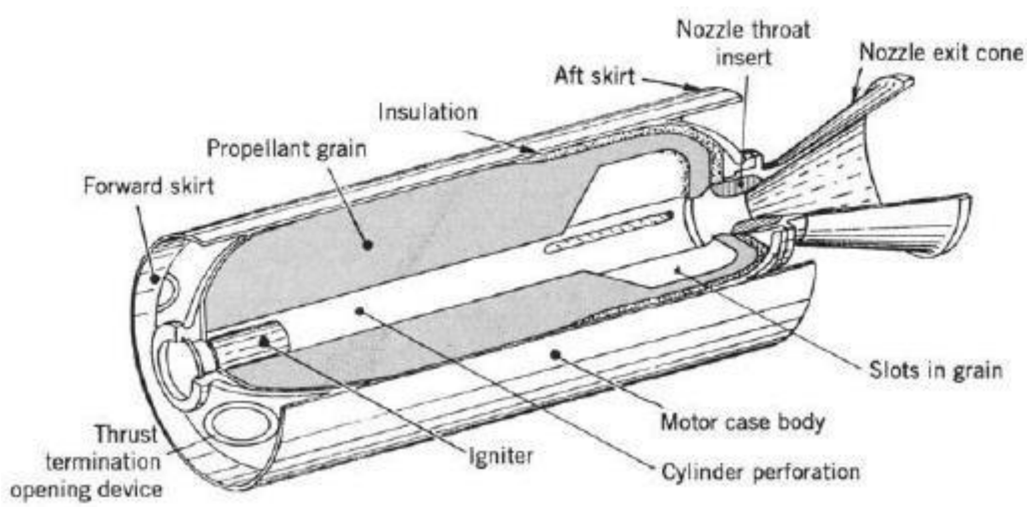


Fig: 2.2 solid propellant

The solid propellant is contained in the combustion chamber or case. The solid propellant charge is called the grain and contains all chemical elements required for complete burning. An igniter is needed to initiate the burning process. Once ignited, the burning proceeds at a predetermined rate on all exposed internal surfaces of the grain, till the complete propellant is consumed. Slots are provided in the grain structure based on variation of burning rate. The resulting hot combustion gases are expanded through a supersonic convergent divergent nozzle to provide the thrust.

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

Hybrid Rocket Motors: A schematic diagram is given below:

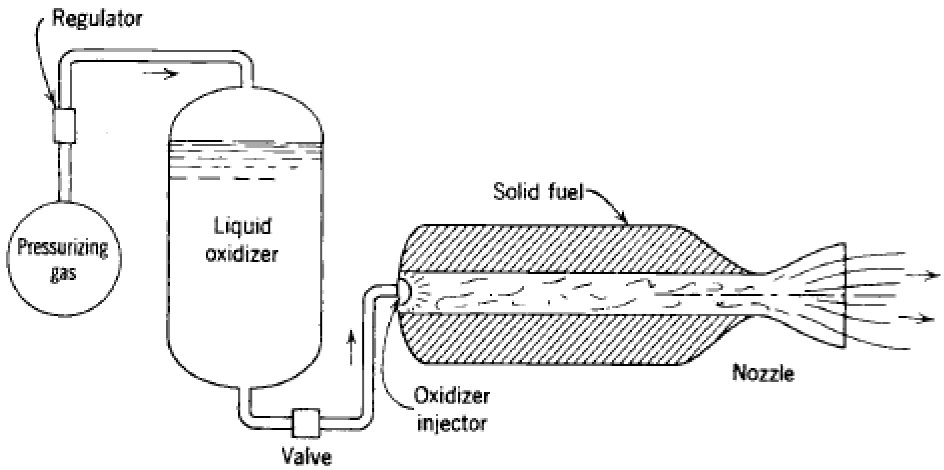
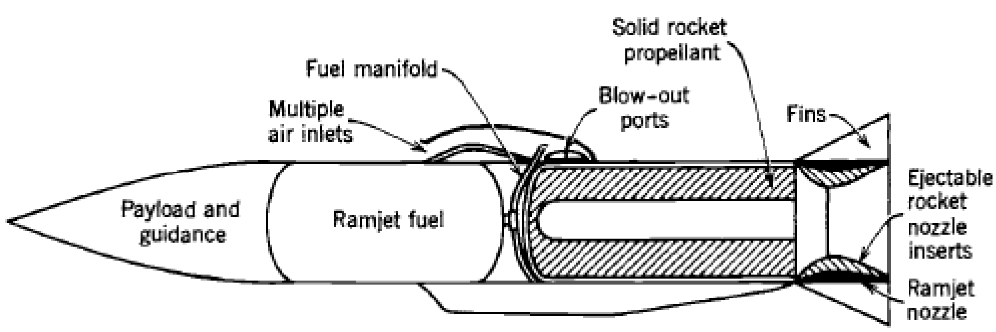


Fig: 2.2 Hybrid Rocket Motors

Hybrid rocket propulsion systems use both solid and liquid propellants. In the above diagram, a liquid oxidizer is held in tank, and is injected, under pressure, into the combustion chamber filled with solid propellant fuel. The hot combustion gases are expanded in the supersonic convergent divergent nozzle.

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



The principles of rocket and ramjet can be combined so that the two propulsion systems can operate in sequence, yet utilize a common combustion chamber.
Initially the system operates in rocket mode, and as the solid propellant combustion completes, the air inlet to the combustion chamber opens, for ramjet operation. Ramjet fuel tank supplies

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**:Space Launch Vehicles or Space boosters are used to place spacecraft from earth in to outer space depending on the mission. Space launch vehicles are usually multistage vehicles using chemical rocket propulsion systems.

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 or used for initial stages whereas liquid propellants are used higher stages. Gaseous propellants are used for rocket 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. Electric propulsion systems are used both for primary and secondary propulsion missions on long duration space flights, inter-planetary/inter-stellar missions.

3. **Missiles**:

Missiles can be classified based on

- **Range: Strategic**(Long range ballistic missiles); **Tactical** (short range targets 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:** Other applications are the secondary applications which include

- 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. Higher overall density of propellant leading to compact size
8. Some propellants have non-toxic, clean exhaust gases
9. Grain design allows use of several nozzles
10. Thrust termination devices allow control over total impulse
11. Can provide TVC
12. Some tactical rocket motors can be produced in large quantities
13. Rocket motors can be designed for recovery, re-use (space shuttle rocket motor)
14. Can be throttled, or stopped and re-started few times, if pre-programmed

Disadvantages

1. Explosion and fire potential is larger
2. Most rocket motors cannot withstand bullet impact or being dropped on hard surface
3. Rockets need environmental clearance and safety features for transport on public conveyances
4. Some propellants are very sensitive and can detonate
5. Grain damage occurs through temperature cycling or rough handling-limiting useful life
6. Requires an ignition system Plumes cause more radio-active attenuation than LPRsExhaust gases are toxic in case of composite propellants with ammonium perchlorate
7. Some propellants can deteriorate (self-decompose) during storage
8. Only some motors can be stopped, but motor becomes disabled
9. Once ignited, difficult to change pre-determined thrust levels

10. Grain integrity (internal cracks, unbounded areas etc) difficult to examine
11. Initial grain temperature effects the thrust levels and flight duration this needs to be carefully factored

- **Liquid Propellant Rockets**

Advantages:

1. Provides higher impulse for given propellant density; increases attainable vehicle velocity increment and mission velocity
2. Can be randomly throttles and stopped and restarted
3. Provides for pulsed (repetitive) operation. Some small thrust rockets allow over 250,000 times usage.
4. Better control over mission terminal velocity, with precise thrust termination devices
5. Can be largely checked prior to operation ie can be tested for full thrust operation on ground
6. Thrust chamber smaller, can be cooled
7. Thrust chamber can be designed for re-use after check ups
8. Thrust chamber has thinner walls and light weight
9. With pumped propellant feed system, inert system weight (including tanks) is lower allowing high propellant mass fraction
10. Liquid propellants are storable in the vehicle for more than 20 years and engine can be ready for use quickly
11. Propellant feed system can be designed to feed multiple thrust chambers
12. Plume radiation and smoke are usually low
13. Propellant tanks can be located such that vehicle stability is high

Disadvantages:

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

Criteria Used for Selecting of Rocket Propulsion System:

1. **Mission Definition**: The purpose and final objective of the system will decide the payload, flight regime and the type of vehicle propulsion system
2. **Affordability (cost):** The cost of R&D, production, operation, facility cost must be within budgetary guidelines.
3. **System Performance**: The propulsion system should be designed to optimize the performance.
4. **Survivability (Safety)**: All hazards must be known in advance. In case any failure, the damage to personnel, equipment, facilities and environment must be minimum.
5. **Reliability**: Technical risks, manufacturing risks and failure risks must be low. Complex systems 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 downtime minimum.
8. **Geometric Constraints**: Propulsion system should fit in to the vehicle within available length and diameter. It is preferable to have a propulsion system 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 propulsion system should be capable of completing the mission in given time frame.

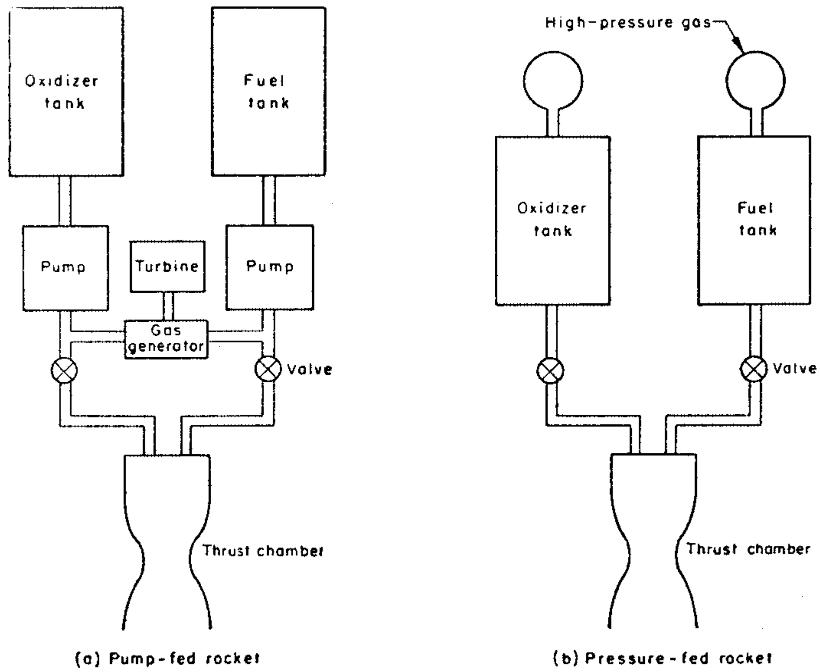


FIG. 2. Schematic of liquid-propellant rocket

Fig: 3.1 liquid propellant rocket

Solid Propellant Rocket Motor:

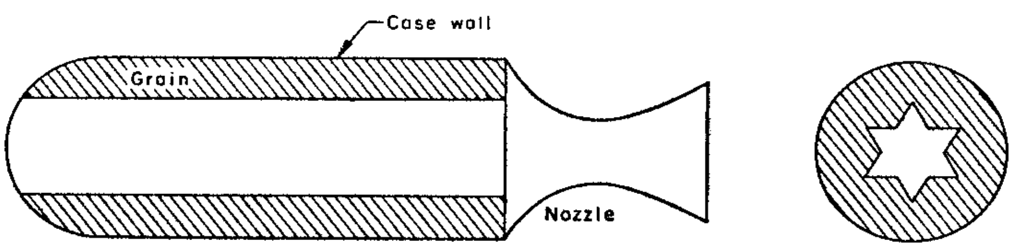


Fig: 3.2 Solid Propellant Rocket Motor

Performance:

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

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

For constant thrust, this reduces to I_t = Ft

2. **Specific Impulse**: The specific impulse is the total impulse per unit weight of propellant. It is an important figure of merit of performance of the rocket system.
For constant thrust and propellant flow, specific impulse is

$$I_s = \frac{I_t}{w} = \frac{F}{w}$$

The performance of rocket is determined largely by the rocket-propellant combination and the total amount of usable propellant. The performance of propellants is characterized by the specific impulse, a measure of thrust produced per unit of propellant consumed per second. The unit of specific impulse is sec.

The velocity that can be achieved by a rocket is directly proportional to the specific impulse of its propellants.

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

$$c = \frac{F}{\dot{m}}$$

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

4. **Mass Ratio MR**: The mass ratio of a vehicle is defined to be the final mass m_f (after the rocket has consumed all usable propellant) divided by mass m_0 (before rocket operation).

$$\text{Mass ratio MR} = \frac{m_f}{m_0}$$

- The final mass m_f is the mass of the vehicle after the rocket has ceased to operate when all the useful propellant mass m_p is consumed and ejected.
- The final mass m_f includes mass of guidance devices, navigational gear, payload, flight control system, vehicle structure tanks, control surfaces, communication equipment and unusable propellant etc
- Value of MR ranges between around 10 % for large vehicles to around 60 % for tactical missiles

5. **Propellant mass fraction ζ**: The propellant mass fraction ζ indicates the fraction of propellant mass m_p in an initial mass m_0. It can be applied to the vehicle or a stage.

$$\zeta = \frac{m_p}{m_0}$$

$$m_0 = m_f + m_p$$

6. **The Impulse-to-weight Ratio:** The impulse to weight ratio of the propulsion system is defined as the total impulse I_t divided by the initial vehicle weight w_0.

A high value indicates an efficient design.

Impulse-to-weight ratio = $\frac{I_t}{w} = \frac{I_t}{(m_f + m_p)g_0} = \frac{I_s}{\{m_f / m_p\}+1}$

Solid Propellant Types:

Three types solid propellants are in use:

- Double-Base
- Composite
- Composite modified double-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 powdered metal (fuel), crystalline oxidizer and polymer binder.

The oxidizer is usually ammonium nitrate, potassium chlorate or ammonium chlorate. The fuels used are often hydrocarbons such as 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-propellant rocket** unit has two separate liquid propellants, an oxidizer and a fuel. They are stored separately and are not mixed outside the combustion chamber. Majority of liquid propellant rockets use bi-propellants.

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

A **Cold gas propellant rocket** stores cold gas (eg: nitrogen) at high pressure, gives low performance and is a very simple system. It is used for roll control and attitude control.

A **cryogenic propellant rocket** stores liquid propellant at very low temperature. Cryogenic propellant is a liquefied gas at very temperature, such as liquid oxygen (at -183° C) or liquid hydrogen (at -253° C). Provision for venting the storage tank and minimizing the vaporizing loses are essential for this type of rockets.

Storable propellants (eg. Nitric acid or gasoline) are liquids at ambient temperatures and can be stored in sealed tanks for long periods. **Space storable propellants** (eg. Ammonia) are liquids at space environment. Their storage tanks need specific 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 or any substance that releases stored energy. Even cold gas at high pressure is a propellant.

Requirements of a propellant: In a rocket vehicle, the aim is to get a high V_j, which in turn need high chamber temperature and pressure T_c and P_c. We also need low value of molecular mass of gasses $\mathfrak{M}$.

The parameter that determines the quality of chamber performance is C^*, the characteristic velocity.

$$C^* = (R\,T_c)/Ѓ$$

The transfer function C^* indicates the capacity of the chamber to generate high pressure gasses must be high. This is a property of the propellant.

T_c must be very high

$\mathfrak{M}$ must be low and

γ, specific heat ratio, must be low

High value of Q, heat release/unit mass and

C_p must be small, which increases the enthalpy content

Molecular Mass $\mathfrak{M}$:Achemicalpropellantsgeneratehigh $T_c$throughchemicalreactionsieheat 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 $\mathfrak{M}$ will be low.

The atomic mass of hydrogen is 1(molecular mass is 2); helium is 4; carbon is 12, aluminum is 27 etc. Oxygen is a powerful oxidizer with 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 at constant pressure C_p: T_c=Q/C_p.Therefore the specific heat at constant pressure must besmall.

Units of specific heat is Joules/mole Kelvin

A single atom of hydrogen or oxygen H or O is mono-atomic and has a C_p of 20 J/ mole K

O_2, H_2 or OH, diatomic combinations, C_p increases to 35 J/mole K,

For higher order tri-atomic combinations like CO_2 etc, it increases to 65 J/mole K. Thisis

because,theenergyabsorbedbymono-atomicsubstancesis less.

We can therefore deduce that if the product combustion gases are mostly mono-atomic, the available heat energy is higher.

The specific heat at constant pressure of elements increases with temperature.

Therefore, if product gases dissociate to mono-atomic or diatomic combustion products, the specific heat at constant pressure is small and the temperature will be high.

Specific Heat Ratio γ: More complex molecules havelower value of γ, because the degrees of freedom are more. Therefore, we need combustion gas products to be more complex for low 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.

Since the effect of specific heat ratio on the jet velocity is less pronounced than that of the specific heat, we prefer propellants with combustion products having lower value of specific heat at constant pressure.

Low $\boldsymbol{\gamma}$ requirement is contrary to lower C_p and lower Molecular Mass $\mathfrak{M}$.

Rocket Equation: Tsiolkovsky's equation calculates the acceleration of the rocket vehicle with mass decreasing continuously due to burning of propellant.

The equation is derived for a spacecraft being accelerated by an unbalanced force ie the **thrust.** Thrust is the only unbalanced force on a spacecraft. Drag is considered zero and the weight is considered balanced by the centrifugal forces.

The mass of the spacecraft is decreasing at the propellant mass flow rate of $\frac{dm}{dt}$.
The thrust force on the spacecraft is equal to the momentum change of the exhaust gas, that is

$$F = V_e \frac{dm}{dt}$$

where V_e is the exhaust gas velocity.

The rocket equation for the velocity increment ΔV is

$$\Delta V = V_e \ln \left({}^{M_i} / {}_{M_f} \right) = I_s g_c \ln \left({}^{M_i} / {}_{M_f} \right)$$

Where M_i is the initial mass of the spacecraft; M_f is the final mass; I_s is the specific impulse of the rocket.

Velocity increment needed for launch:

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

The velocity increment is the velocity calculated from the rocket equation, and is a measure of the energy expended by the rocket.

The vehicle velocity is less than this, because of gravity loss, and the energy needed to reach orbital altitude. The difference represents the energy expended against gravity loss and potential energy.

Deductions from Tsiolkovsky's Rocket equation:

- Ratio of initial and current mass of vehicle defines current velocity.
- It is applicable to any velocity increment, when initial and final masses are known.

- The exhaust velocity V_e is assumed to be constant, and is valid for most of the real cases.
- The velocity of rocket vehicle, at any instant of burn time, is dependent only on exhaust velocity and the instantaneous mass ratio.
- Use of multi-stage vehicles enable achieving higher velocity increment for deep space missions

Types of propulsion systems along with maximum ΔV are given below

- Chemical: Solid, liquid, hybrid-max ΔV=Solid-5.7-7.1 km/s; liquid-6.9-11.5 km/s
- Magneto-hydrodynamic (MHD)Propulsion: max ΔV-4.6 km/s
- Nuclear: Fission-max ΔV=11.5-20.7km/s; Fusion-max ΔV-230-2300km/s; Antimatter-max ΔV- 1380 km/s
- Electric: Electro-thermal- max ΔV-3.5-27.6 km/s; Electrostatic- max ΔV-27.6-230 km/s; Electromagnetic-max ΔV-16.1-115 km/s
- Propellant-less: Photon Rocket - max ΔV-unlimited; Solar sails; Magnetic sails

Mission Velocity/Delta V budget:

A convenient way to work out the magnitude of total energy requirement of space mission is to use the concept of mission velocity. It is the sum of all flight velocity increments needed to achieve the mission objective.

- The performance required from the propulsion system is calculated based on required change in velocity ie ΔV, which is comprised of several components, as indicated below:

$$\Delta \mathrm{V} = \Delta \mathrm{V}_g + \Delta \mathrm{V}_{drag} + \Delta \mathrm{V}_{orbit} - \Delta \mathrm{V}_{initial}$$

Where $\Delta \mathrm{V}_g$ is required to overcome the gravitational potential

$\Delta \mathrm{V}_{drag}$ is required to overcome the drag encountered by the vehicle while in earth's atmosphere

$\Delta \mathrm{V}_{orbit}$ is the required velocity increment for the vehicle to reach given orbit

$\Delta \mathrm{V}_{initial}$ is the initial velocity of the vehicle by virtue of earth's rotational speed. The rotational speed of earth at equator is 0.464 km/sec and varies with latitude of the launch station, for example, it is 0.408 km/sec at a latitude of 28.5° latitude (cape kennedy)

Combustion Instability: Combustion instability refers to fluctuations in pressure build up in combustion chamber which may turn severe if the combustion process is not controlled.

Unstable combustion, or combustion instability results in oscillations occurring at regular intervals, which may increase or may die out.

Principle types of combustion instabilities are

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

Combustion instability can be avoided by avoiding resonance in the feed systems, providing acostic damping (absorbers by providing small cavities) in the chamber and provision of injector baffleschanges in the injector geometry etc.

Reaction Control Systems (RCS):

Propulsion systems are of two types:

1. **Primary Propulsion function** – High thrust engines, used for launch along flight path, for orbit injection, interplanetary missions etc). Some missions require four to six rocket units while more complex manned spacecraft need 40 to 80 rocket units in all stages.
2. **Secondary Propulsion function** - Low thrust applications like attitude control, spin control, momentum wheel, stage separation or for settling of fluids. The small thrust rockets must give pulses of small bursts of thrust, necessitating thousands of restarts. The small attitude control rockets must give pulses or short bursts of thrust, necessitating thousands of restarts.

Higher thrust engines are used for primary propulsion systems for inter-planetary spacecraft.

Majority of spacecraft used liquid propellant engines, with solid propellant boosters. Several spacecraft have operated successfully with electrical propulsion for attitude control. Electrical propulsion systems will probably be used for primary and secondary propulsion missions on long duration spaceflights.

A reaction Control System (RCS) is needed to provide trajectory corrections (small Δv corrections) and correcting rotational or attitude position of the spacecraft and launch vehicles. RCS is also called auxilliary rocket propulsion system.

If only rotational maneuvers are made, RCS is termed as attitude control system. Maneuvers

Conducted by RCS:

1. **Velocity Vector Adjustment and Minor In-flight Correction Maneuvers**: These are performed with low thrust, short duration and intermittent (pulsing) operations. RCS

uses multiple small liquid propellant thrusters, both for translation and rotation. The **reaction control rocket systems** in a space launch vehicle will allow accurate orbit injection after it is placed in the orbit by another less accurate propulsion system. The **vernier rockets** placed on a ballistic missile are used to accurately calibrate the terminal velocity vector for improved target accuracy. Mid-course guidance correction maneuvers also fall in this category. Propulsion systems for orbit maintenance maneuvers are called station keeping maneuvers. They keep the spacecraft in the intended orbit overcoming the perturbing forces.

2. **Rendezvous and Docking Maneuvers**: The relative positions of the launch planet and the target planet are critical for planetary transfer mission. The spacecraft has to meet or **rendezvous** with the target planet when it arrives at the target orbit. There is a specific **time-window** for a launch of a spacecraft that will make a successful rendezvous. Docking is the linking up of two spacecraft and requires a gradual gentle approach (low thrust, pulsing mode thrusters) so as not to damage the spacecraft. These maneuvers involve both rotational and translational maneuvers conducted by small reaction control thrusters.

3. **Simple rotational maneuvers:** These maneuvers rotate the spacecraft on command into specific angular position so as to orient or point a telescope, instrument, solar panel or antenna for the purpose of navigation, communication or solar power reception. Such a maneuver is also intended to keep the orientation of the satellite in a specific direction, for example if the antenna needs to be continuously pointed towards the center of earth. Then the satellite needs to be rotated around its own axis once every satellite revolution. These maneuvers can also provide flight stability or correcting angular oscillations. If the rotation needs to be performed quickly, then a chemical multi-thruster reaction control system is used.

4. **Change of Plane of Flight Trajectory:** This maneuver requires application of thrust force in a direction normal to the original plane. This maneuver is performed by a propulsion system that has been rotated by the RCS in to proper orientation.

5. **Transfer of orbits:** Sometimes, transfer orbits can be achieved with very low thrust levels (0.001 to 1 N), using electric propulsion systems

An RCS can be incorporated in the payload stage and each of the multi-stage vehicle. RCS operates throughout the flight and provides control torques and forces. Liquid propellant rocket engines with multiple thrusters are most commonly used. Electric propulsion systems are used on spacecraft.

Life of RCS may be short or it may be used throughout the mission duration (may be more than 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)

In order to apply torque, it is necessary to use two thrust chambers with equal and opposite start/stop times. There is a minimum of 12 thrusters in a torque 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**

RCS systems are characterized by the magnitude of force, quantity (number) and duty cycles. A duty cycle refers to number of thrust pulses, operating time and time between pulses

RCS systems can be mass expulsion types (rockets) or non-mass expulsion types. Reaction wheels or flywheels and momentum storage devices are examples for non-mass expulsion types. The vehicle angular momentum can be changed by accelerating or decelerating a reaction wheel.

The propellants for RCS fall into three categories-cold gas jets (also called inert gas jets), warm or heated gas jets and chemical combustion rockets. Hydrazine is most commonly used monopropellant in RCS.

Nozzles:

The function of nozzle is

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

Because of the high temperatures that the nozzle experiences, materials used in nozzle construction are usually nickel-based alloys, titanium alloys or ceramic composites.

Under-expanded Nozzles:

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

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

Over-Expanded Nozzle: If the exit pressure is lower than the ambient pressure, the flow is considered to be **over-expanded**.

- In an **over-expanded nozzle**, the fluid attains a lower exit pressure than the atmospheric pressure.
- The exit area at this condition is too large than the optimum area.
- This condition occurs when the nozzle operates at altitudes lower than the design altitude.
- Since the pressure inside the nozzle is lower than the outside pressure, there is possibility of flow separation due to adverse pressure gradient.

Governing equations of flow: For analyzing the flow through nozzle, the flow is assumed to be friction-less and adiabatic, and the exit pressure is assumed equal to the ambient pressure.

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

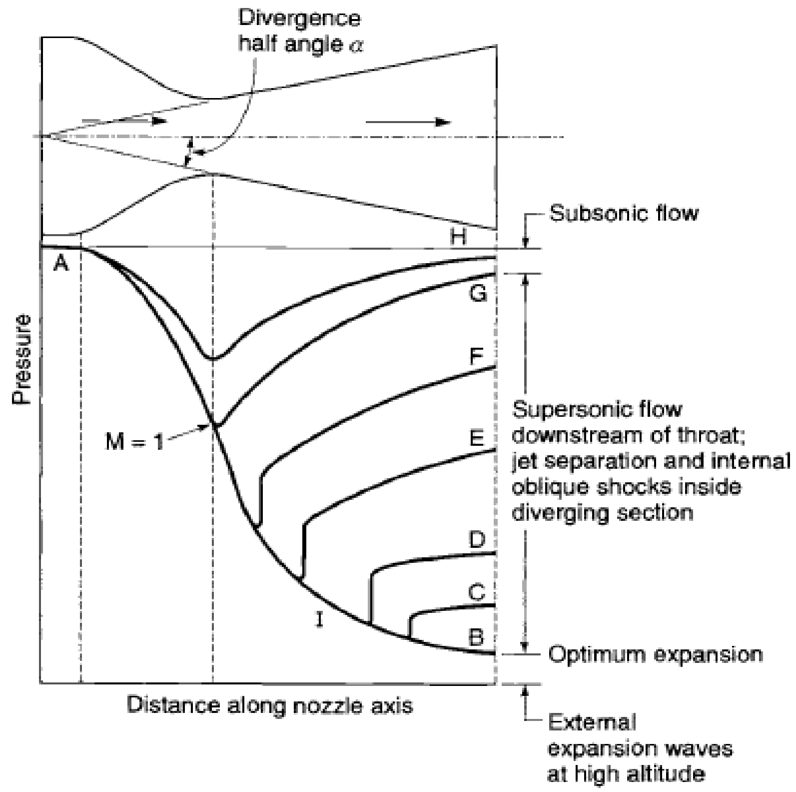


FIGURE 3–9. Distribution of pressures in a converging–diverging nozzle for different flow conditions. Inlet pressure is the same, but exit pressure changes. Based on experimental data from A. Stodala.

- Curve AB shows variation of pressure with optimum back pressure at the design area ratio. (with M = 1 at throat)
- Curves AC and AD show variation of pressure along the axis for increasingly higher external pressure (over-expansion). At point I, on curve AD, the pressure is lower than the exit pressure and a sudden rise in pressure takes place accompanied by separation of flow from the walls. (Condition when aircraft flies at altitude lower than design altitude)
- The sudden pressure rise in the curve AD is a compression discontinuity accompanied by a compression wave.
- Expansion waves occur in cases where external pressure is lower than the exit pressure, ie below point B. (Under-expansion – condition when aircraft flies at an altitude higher than design altitude)

Relations in nozzle:

- In the convergent section, flow is subsonic. The chamber contraction area ratio A_1 / A_t is small, in the range of 3 to 6.
- In solid propellant rocket chamber, A_1 refers to flow passage or port cavity of the grain.
- The divergent portion handles supersonic flow, the area ratio becomes large very quickly. Value of k also varies significantly.
- The area ratio in the divergent section, A_2 / A_t ranges between 15 to 20; at M=4
- $T_0 = T\,[\,1 + \frac{1}{2}(k-1)M^2\,]$
- $M = \sqrt{\frac{2}{\gamma-1}\left[\frac{T_0}{T} - 1\right]}$
- The exhaust velocity v_2 is calculated as $v_2 = \sqrt{\frac{2\gamma}{\gamma+1} RT_1\left[1 - \left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma}}\right]}$
- For optimum expansion p_2=p_3, v_2=c_{2opt}; the optimum expansion occurs only at design altitude. At all other altitudes, the nozzle is either under expansion or over expansion condition.
- The throat condition is defined by pressure ratio $p_t / p_1 = \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma}{\gamma-1}}$
- The throat temperature T_t = (2 T_1)/(γ+1)
- The throat velocity $v_t = \sqrt{\frac{2\gamma}{\gamma+1} RT_1}$

Losses in Nozzle: In actual case, the flow is non-isentropic. The entropy increases due losses caused by **friction in the boundary layer, flow turbulence, secondary flows due to 3-D flows, shocks and flow separations**. However, the flow remains adiabatic and the total enthalpy remains constant.

Types of Nozzles:

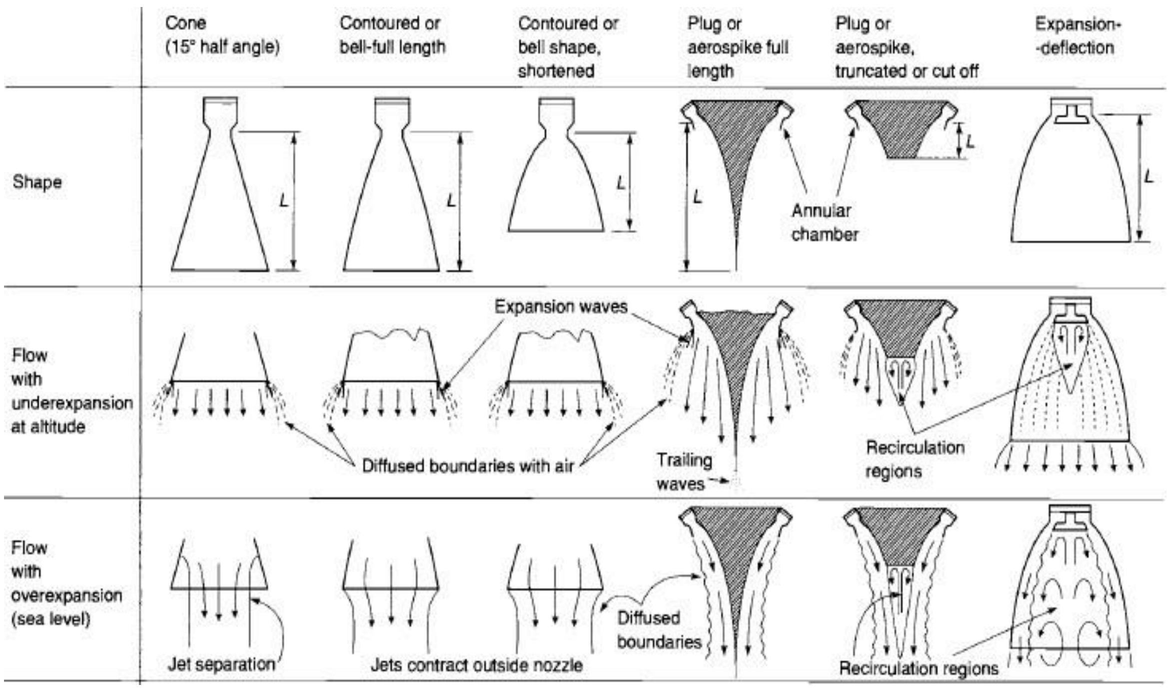


Fig: 3.3 Flow Conditions in Nozzle:

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

- In some rockets, the gaseous working fluid contains many small liquid droplets and/or solid particles that must be accelerated by the gas.
- This occurs in with solid propellants and some gelled liquid propellants which contain aluminum powder that forms small oxide particles in the exhaust.
- It can also occur with ion oxide catalysts, or propellants containing beryllium, boron or zirconium.
- In general, if the particles are very small, with diameters of 0.005mm or less, they will have almost same velocity as the gas and will be in thermal equilibrium with the nozzle gas flow
- The solid/liquid particles give up heat to the gas during expansion in a nozzle.

- As the gases give up kinetic energy to accelerate the particles, they gain thermal energy from the particles.
- As the particle diameter become bigger, the larger particles do not move as fast as the gas and do not give up heat as readily as the small particles.
- The larger particles have a lower momentum and they reach nozzle exit at a higher temperature than the smaller particles.
- For larger particles, over 0.015 mm diameter, the specific impulse can be 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:** The composition of the combustion products is invariant, that is, no change in gas composition. There are no chemical reactions or phase changes in the nozzle flow. The product composition remains same from nozzle inlet to exit. This method is usually simple, but underestimates the performance by 1 to 4%.
- In the frozen flow case, no chemical change occurs during expansion, there are no rate processes at all occurring, the molecules preserving their identity all the way.

Shifting Equilibrium:

- Instantaneous chemical reactions, phase changes occur between gaseous and condensed phases of all species in the exhaust gases.
- Thus, the product composition shifts as the flow proceeds through nozzle
- The results calculated are called shifting equilibrium performance.
- The gas composition and mass percentages are different in the chamber and nozzle exit.
- This analysis is more complex and the values of the performance parameters, are overstated by to 4%.

- In the shifting equilibrium flow case, reactions do occur, their rate is so high (compared to the expansion rate) that conditions adjust continuously to maintain equilibrium at the local pressure and enthalpy level.
- With the result, the whole process can be regarded as reversible (and hence isentropic)
- The actual expansion process in a rocket or ramjet nozzle is intermediate between the 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.

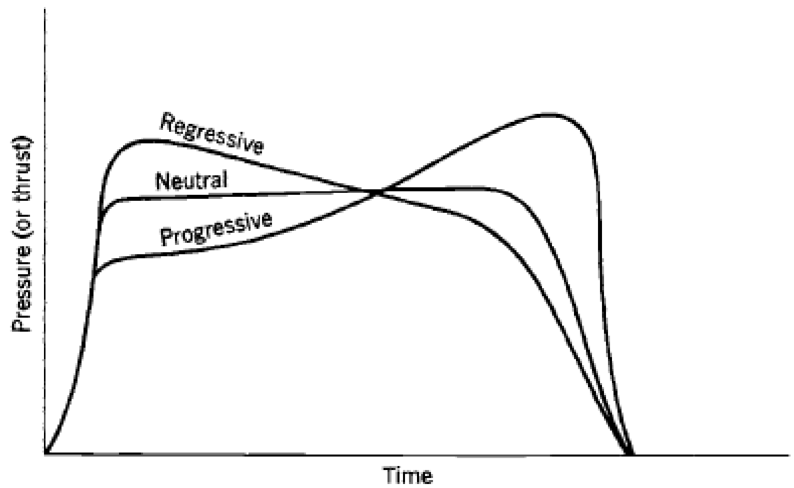


Fig: 3.4 pressure vs Time contour

- **Progressive Burning**-The thrust, pressure and burning surface area increases with burn time.
- **Regressive Burning**-The thrust, pressure and burning surface area decreases with burn time
- **Neutral Burning**-The thrust, pressure and burning surface area remains constant through motor burn time.
- **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 oxidiser percentage
- 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 grain temperature

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

Injectors:

The functions of injector are similar to those of a carburetor of an I.C engine. The functions are

- Injector has to introduce and meter the flow of liquid propellants into the combustion chamber
- It has to atomize the fuel, ie cause the liquid to be broken up into small droplets in the combustion chamber
- It has to cause distribution and mixing up of propellants such that a correct proportion of mixture of fuel and oxidizer will result
- It has to ensure uniform propellant mass flow and composition over the cross section of the combustion chamber.

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

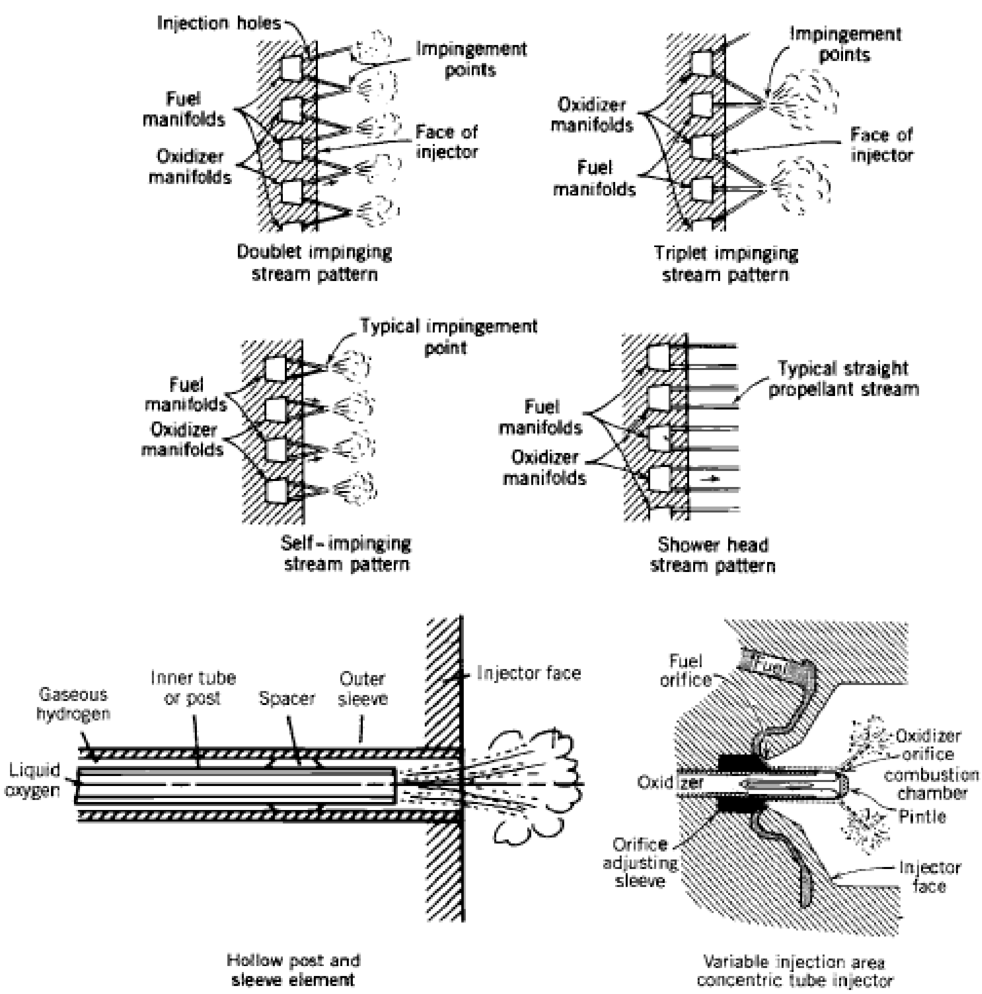


Fig: 3.5 types of injector

The injection hole pattern on the face of the injector is closely related to the internal manifolds or feed passages within the injector. These provide for the distribution of the propellant from the injector inlet to all injection holes.

A large manifold volume allows low passage velocities and good distribution of flow over the cross section of the camber. A small manifold volume allows low for a lighter weight injector and reduces the amount of “dribble” flow after the main valves are shut. However, the higher passage velocities cause a more uneven flow through different injection holes and thus poor 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 minimized as much as possible.

Impinging-stream type, multiple-hole injectors are commonly used with oxygen-hydrocarbon and storable propellants. The propellants are injected trough a number of small holes in such a manner that the fuel and oxidizer streams impinge on each other. Impinging patterns can also be fuel-on-fuel or oxidizer-on-oxidizer types.

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.

Sheet or Spray type injectors give cylindrical, conical or other types of spray sheets, with sprays generally intersect to promote mixing and atomization. The width of the sheet can be varies by using axially movable sleeve, it is possible to throttle the propellant over a wide range. This type of **variable area concentric tube injector** is used in lunar module.

The **Co-axial hollow post injector** is used for liquid oxidizer and gaseous hydrogen injectors (shown on lower left of above diagram). The liquid hydrogen gets gasified in the outer sleeve by absorbing heat from the cooling jackets. The gasified hydrogen flows at high velocity (around 330 m/sec) while the liquid oxygen flows slowly (around 33 m/sec). This differential velocity causes a shear action, which helps in breaking up the oxygen stream in to small droplets.

The injector assembly shown below, used space shuttle, has 600 concentric sleeve injection elements, of which 75 of them are lengthened beyond injector face to form cooling baffles, which reduces combustion instabilities.

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 and operating characteristics of injectors are given below:

- **Propellant Combination**: The particular combination of fuel and oxidizer affects the characteristics such as chemical reactivity, speed of vaporization, ignition temperature, diffusion of hot gasses, volatility and the surface tension. Hypergolic (self-igniting) propellants generally require different designs from those required by propellants that must be ignited. Each combination requires own design injector design.
- **Injector Orifice Pattern and Orifice Size**: With individual holes in the injector plate, there is a optimum performance and heat transfer condition for parameters like orifice size, angle of impingement, distance of the impingement from the injector face, number of 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 earlier designs.
- **Transient Conditions**: Starting and stopping the rocket motor operation require special provisions like temporary plugging of holes, accurate valve timing, insertion of paper cups over holes to prevent entry of one propellant in to manifold of other propellant etc.
- **Structural Design**: The injector is highly loaded by pressure forces from the combustion chamber and the propellant manifolds. During transients (starting and stopping), these pressure conditions case severe stresses. The faces of injector are usually flat and need reinforcements. Also the structure of the injector must be flexible enough to withstand the thermal deformations caused by heating by hot combustion gases and cold cryogenic propellants.

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

Combustion Chamber:

Liquid Propellant Rocket: The combustion chamber or thrust chamber is a combustion device where the liquid propellants are metered, injected, atomized, mixed and burned to form hot gaseous reaction products. These hot gases are accelerated and ejected from the nozzle at a high 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 jackets to absorb the heat that is transferred from the hot combustion gases to the chamber walls. There are uncooled thrust chambers, which use ablative materials to withstand high temperatures.

The combustion chamber where the burning takes place, must be designed to withstand the heat generated. The volume of the chamber must be large enough for adequate mixing, evaporation and complete combustion of the propellants. The chamber diameter and volume influence the cooling requirement.

The characteristic chamber length is defined as the length that a chamber of same volume would have if it were a straight tube and had no converging nozzle section.

$$L^* = V_C/A_t$$

Solid Propellant Rocket: In solid propellant rocket motors, the propellant is contained and stored in the combustion chamber. The storage may be for long periods of around 15 years, so the motor casing s sometimes hermitically sealed.

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

Desirable Properties of Solid Propellants:

- High specific impulse
- Low molecular weight to provide high exhaust velocity
- High heat of formation to result in high temperatures
- Combustion products must contain simple light elements
- High density to result in better specific energy and low size
- Simple to manufacture with few moving parts
- Re-usability of components
- Smoke-less, non-toxic exhaust

Nuclear Rocket

Power-Thrust-Energy: The high specific energy of nuclear fuel is the reason which makes nuclear propulsion ideal for deep space missions including manned missions to other planets.

For voyages to planets, a spacecraft needs to be given a very high velocity of 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.

The maximum exhaust velocity of a **LOX/LH** engine is about 4.5km/s and it works out to **energy per kg as 10.4 MJ/kg**. So about 6 kg of propellant is needed to be burnt for every 1 kg of vehicle mass, in order to provide enough energy to set a 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.

The high specific energy of nuclear fuel is a major advantage for high energy interplanetary missions.

The energy stored in nuclear propellants is 10^7 - 10^9 times higher than chemical propellants. A propulsion system using nuclear energy can achieve any specific impulse comparable to the speed of light.

Nuclear Fuel Basics:

Nuclear processes (Fission or Fusion) use very small quantities of matter. A working fluid, usually is coupled with nuclear reaction products.

After fusion or fission of atoms, the end product will have smaller mass than the initial atoms**. This mass defect is directly transferred in to energy based on Einstein's equation E = mc^2**

Fission Propulsion: Nuclear Fission is a process in which a large nucleus of an atom splits into two smaller nuclei (lighter nuclei) with release of energy. The fission process often produces free neutrons and releases a very large amount of energy. The splitting of nucleus is as a result of neutron bombardment.

The mass changes and associated energy changes in nuclear reactions are significant. For example, the **energy released from the nuclear reaction of 1 kg of uranium is equivalent to** the energy released during the combustion of **about four billion kilograms of coal**.

Nuclear Fission:

Nuclear Fission is used in high thrust applications. Fission is a process where a neutron is absorbed a uranium nucleus, which causes the nucleus to split into two nuclei (of mass about half that of uranium). The mass defect causes release of energy, in the form of kinetic energy of the two fission fragments. The splitting process is also associated with release of two or more neutrons are emitted at the same time as the fission of the nucleus occurs. These neutrons go to interact with another nucleus and cause to split, thereby, setting up a chain reaction. Since rate at which energy is released depends only on the neutron flux, the power output of a fission system is controlled by inserting materials that absorb neutrons.

In a controlled nuclear fission, the uranium becomes very hot, leading to melting of Uranium. Hence, to continue with the energy release, it is essential to cool the uranium extracting heat. The cooling of Uranium is accomplished using a propellant, which passes through the reactor and then expelled out of the nozzle.

Two isotopes of Uranium, U^{238} and U^{235} are available, of which 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.
- Fusion is more complex than fission, since in fusion, in order to bring the two positively charged nuclear cores close together, the energy of electrostatic repulsion has to be overcome and maintained
- The energy released in nuclear propulsion is governed by Einstein's equation $E=mc^2$
- Nearly all gained energy through the mass defect is released as heat.
- While Fission and Fusion transfer only part of the nuclear binding energy in to heat, the Matter-Antimatter annihilation (eg. Proton-antiproton or hydrogen-antihydrogen etc) can release all the nuclear energy.

Sizing of the Reactor/Ensuring Sustainable Chain Reaction:

There are two approaches that will improve the chances of sustainable chain reaction. They are

- **Enrichment of U^{235}**: It involves increasing the percentage of U^{235} in the natural uranium to a level that highly increases the probability. The process of enrichment is complicated and costly. Natural Uranium contains very low % of U^{235} (0.72%). Although U^{238} also participates in fission process, the probability of fission initiation is very low since U^{238} requires collisions with neutrons with high energy levels, thus reducing the probability very low.

- **Use of moderator**: The second approach is to slow the neutrons quickly and reduce absorption of neutrons by U^{238} nuclei by using a moderator, usually carbon or water. The moderator is mixed with the uranium atoms in a **homogeneous reactor**, or the moderator and uranium can be in separate blocks, as a **heterogeneous reactor**.

The heterogeneous reactor which uses cylindrical rods of Uranium separated by blocks of moderator, improves the probability of sustained reaction high and permits use of more natural Uranium. However, this increases the size of the reactor, as more moderator is required.

For space applications, the need to keep size low, requires use of enriched Uranium. Plutonium can also be used in the same way as enriched Uranium, but the material is poisonous and highly radioactive. Safety issues are complex to 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 deciding the size of space reactor:

- In a fission reactor using moderator, sufficient travel distance must be 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 probability of sustained fission. Neutron leakage must be low.
- Larger reactors will have lesser leakage than the smaller ones.
- Heat generated by fission must be efficiently removed preventing reactor core from overheating.
- Propellant flow through channels passing through the reactor 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_∞ 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 $K_\infty < 1$, no chain reaction is possible For

$K_\infty > 1$, the chain reaction is possible

$K_\infty = 1$ is the critical level and K_∞ will needs to be controlled at 1 for steady 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:

∞ is the number of neutrons that emerge from fission of the nucleus, per incident neutron. U^{235} nucleus produces 2.44 neutrons on an average per incident. The value of η for U^{235} is 2.07, available for furthur fission process.

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

ε is the fast fission factor, indicates the probability that a neutron is available for furthur fission process. Value of ε should be 1.

P is the "resonance escape probability", which indicates chances of absorption by U^{238} nuclei before causing furtur fission process. Value of p depends on fraction of U^{238} in the fuel and its distribution. If the moderator slows down the neutrons quickly,theirchancesofcapturearereduced,withvalueofphigh.Valueofp ranges from 0.6 to 0.8.

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

Reactor Dimensions/Neutron Leakage:

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

The key factors 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 thermal energies (escaping absorption).

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 or super-critical.

The critical link between geometry of the reactor and the criticality is given by the "**buckling factor**".

The buckling factor is calculated based on neutron diffusion in a reactor of different shapes. It is found to be inversely proporional to the the length L and radius R of the reactor.

Control:

Control of neutron flux and hence the power output is essential for the reactor. Control is maintained by using number of control rods with high absorption in the core. The control rods move in a channel and be inserted or withdrawn from the core.

When fully inserted, they absorb the neutrons so that the reactor goes sub-critical and the fission stops. At an intermediate position, the neutrons are absorbed just enough to retain the criticality.

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

At the start up, the rods are withdrawn so that k is greater than one and netron flux and power output increases. Once desired critical level is reached, the rods are inserted in to the intermediate position. Shut down is achieved by fully inserting the rods in to the core.

Reflection:

In normal operation, the neutrons diffusing out of the nuclear core will be lost in fission process or get absorbed. Smaller reactors can be designed to cause the neutrons to diffuse back again in to the reactor, after leaving the core, spending some time scattering off the nuclei in the external moderator. Some neutrons diffusing out of reactor core will participate in the fission process and the remaining could be made to diffuse back. A core fitted with an external moderator, called "reflector" can be advantageous, in that smaller quantities of U^{235} is needed to achieve criticality.

For space based reactors, ability to control neutron reflection provides a contro element. This reduces the need for internal control rods which are inconvenient in a space reactor.

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
- Can avoid use of internal control rods for regulating neutron flux in the space reactors.

Prompt and Delayed Neutrons:

The fission process inside the nuclear core involves neutrons being released and travelling to the next nuclei/moderator along path. With-in the nuclear dimensions, the travel time is almost instantaneous, with in a few milliseconds. This would make the control mechanism of moving control rods in/out of the core to regulate neutron flux very difficult.

However, the control process 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 which undergo decay during the nuclear process, but will cause induce time lag between prompt neutrons and delayed neutrons.

The delayed neutrons makes the control process though movement of control rods more effective.

Thermal Stability:
The multiplication factor k is sensitive to temperature. K decreases when the temperature raises. This is due to the fact that density of core materials increases causing them to expand, increasing the mean distance between collision and increases the probability of fission.

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

As k gets more than 1, the increased release of energy due to neutron flux being more, increases the temperature, which in turn, reduces the value of k. Thus thermal stability is established.

There are two factors at work, which govern the power output. For a stable state of the core, value of k is one. The power level depends on the neutron flux, which is stable only when k equals one.

To increase the power level, value of k is allowed to become greater than 1. Once the desired power level is reached, k is returned to value of 1, and the reactor 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.

The core contains highly enriched Uranium, mixed with a quantity of moderator. Higher the level of enrichment, difficlt is to control the engine and cost is also high. However, lowering the enrichment increases the size of the reactor.

Nuclear Thermal Rocket Engine:

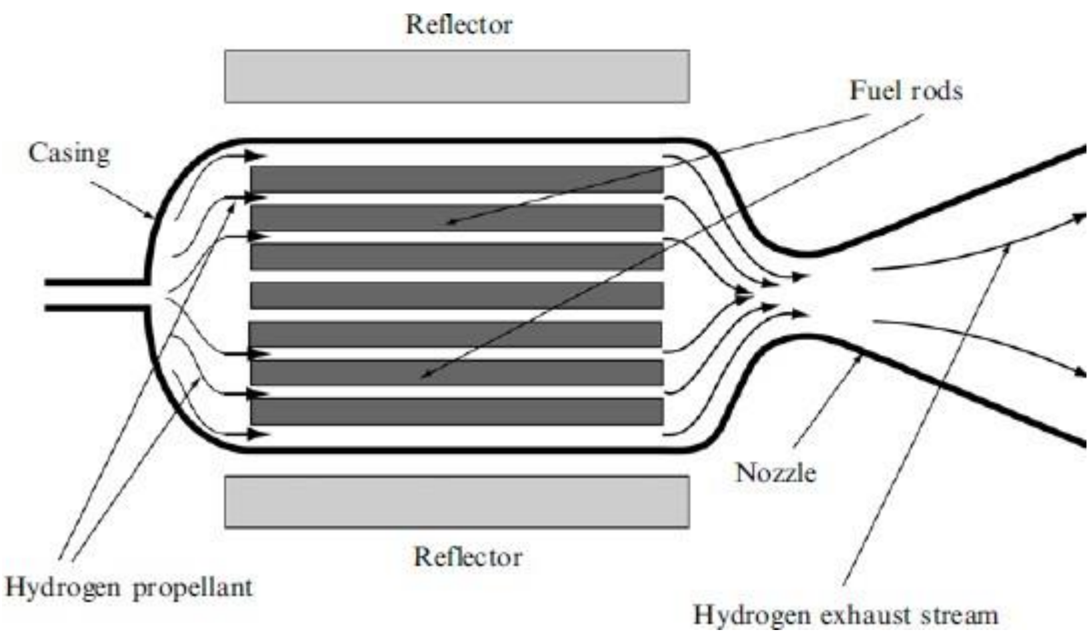


Fig: 5.1 Nuclear rocket

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 smilar 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 during the operation and after use. Pure uranium by itself is safe to handle, since its 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. Beyond the casing, there is a high flux of both neutrons and gamma rays, 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 on the forward end of the engine. Any humans can be safely in the cabin well 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.

Other than the forward side, the radiation shield is not provided anywhere else on the spacecraft.

Propellant Flow & Cooling:

The propellant flow is similar to chemical liquid engines except that there are 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 heat is extracted by the propellant and exhausted down the nozzle.

The reflector and the casing needs 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:

The start up of the nuclear thermal rocket is similar to a cryogenic chemical engine. The whole distribution system has to be cooled down so that the cold hydrogen does not cause thermal shock in the components. Once started, the power output of the reacor will raise very quicky, in matter of seconds. The cooling of the core casing by the propellant must keep pace with rapid heating.

Initially the pressure in the chamber is not adequate to drive the propellant turbo-pumps. Initially, during starting phase, electrical power must drive the turbo- pumps.

Once, the engine is in stable operating mode, the thrust can be varied by positioning the control rods. The power ouput is a function of neutron flux.

About 1% of the neutrons produced by fission are delayed.. When the reactor is shut down, the fission process and hence the power output continues 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. The specific energy of nuclear propellant is far greater than chemical propellant.
2. High Δv values can be obtained by nuclear propulsion.
3. Large increments of Δv are possible with low usage of propellant in nuclear propulsion.
4. The advantage of nuclear rocket is intermediate between chemical and electrical propulsion when onle exhaust velocity is considered.
5. An ion engine can only generate thrust of fraction of a Newton, but nuclear engine can produce thrust in hundreads of Newtons.
6. Nuclear Engines can provide the high delta velocity required for interplanetary missions to Mars, Venus and beyond.
7. Use of nuclear engines for space journeys can shorten the time of journey to a great extent.

Development Status of Nuclear Thermal Rocket:

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

The ground testing of nuclear thermal rocket has been stopped since 1970 due to restrictions placed on release of nuclear contaminated exhaust from the rocket.

There is renewed interest in the need for a nuclear thermal rocket engine as the main booster for the manned mission to Mars.

One proposal that is feasible, but costly is to test nuclear core in space. And activation and safe disposal of the core needs to be sorted out. The safety issues also need to be addressed since nuclear core for space applications need to use enriched Uranium.

It is likely that a nuclear propelled mission will be mounted in the next decade. The proposal under consideration is that a fission reactor will provide the electricity necessary for an electric propulsion.
If the safety aspects and political acceptance can be obtained, then the nuclear thermal engine will take its place in the propulsion systems for space exploration.

Electrical Rocket:

Limitations of Chemical Rocket Engines:

1. **Explosion & Fire Potential**: Explosion and fire potential is larger, failure can be catastrophic.
2. **Storage Difficulty**: Some propellants deteriorate (self-decompose) in storage. Cryogenic propellants cannot be stored for long periods except when tanks are well insulated. A few propellants like Red Fuming Nitric Acid (RFNA) give toxic vapors and fumes. Under certain conditions, some propellants and grains can detonate.
3. **Loading/Transportation Difficulty**: Liquid Propellant loading occurs at the launch stand and storage facility is needed. Many propellants require environmental permit and safety features for transport on public conveyance.
4. **Separate Ignition System:** All propellants , except liquid hypergolic propellants, need ignition system. Each restart 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:** Thermal insulation is required in all rocket motors.

ELECTRIC THRUSTERS-MISSION APPLICATIONS TO SPACE FLIGHT

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

1.1 Limitations of Chemical Rocket Engines:

- **Explosion & Fire Potential (SPR &LPR)**: Explosion and fire potential is larger, failure can be catastrophic.
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- **Smoky Plume (SPR & LPR):** 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.
- **Need For Thermal Insulation (SPR & LPR):** Thermal insulation is required in almost all motors.

- **Difficult to detect grain integrity(SPR):** Cracks in the grain and unbounded areas are difficult to detect.
- **Toxic Exhaust Gases (SPR)**: Exhaust gases are usually toxic for composite propellants containing Ammonium Perchlorate.
- **Difficult to Re-use (SPR):** If designed for reuse, the motor requires extensive rework and new propellants.
- **Difficult to change thrust ratings(SPR):** Once ignited, the predetermined thrust and duration cannot be changed.

- **Complex Design (LPR) :** Relatively complex design, more parts and hence more probability for malfunction.
- **Sloshing in Tanks (LPR):** Sloshing in tanks can cause flight stability problem**.** Baffles are needed 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 inert gas storage for long periods of time.

1.2 Electric Propulsion Systems:

1.2.1 Structure: The basic subsystems of a electric propulsion thruster are

1. **Energy Source**: Energy source that can be solar or nuclear energy with auxiliary components like pumps, heat conductors, radiators and controls. The energy source is different from the propellant;
2. **Conversion Devices**: The conversion devices transform the energy from above source in to electrical form at proper voltage, frequency and current suitable for electric propulsion system;
3. **Propellant System**: The propellant system stores, meters and delivers the propellant to the thruster;
4. **Thruster**: One or more thruster to convert the electric energy into kinetic energy exhaust. The term thruster is commonly used to mean the thrust chamber.

1.2.2 Types of Electric Thrusters: Three fundamental types of electric thrusters are available;

1. **Electrothermal**: In this type, the propellant is heated electrically and expanded thermodynamically where the gas is accelerated to supersonic speeds through a nozzle, as in chemical rockets, to produce thrust.
2. **Electrostatic or Ion propulsion engine:** In this type, **a**cceleration is achieved by the 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 the interaction of electric and magnetic within a plasma. The plasmas are moderately dense, high temperature gases which are electrically neutral but good conductors of electricity.

Electrothermal Thruster: Electrothermal thrusters use the simplest way to heat the propellant with a hot wire coil, through which an electric current pass. More energy can be delivered from electric current if an arc is struck through the propellant, which generates higher temperature than the resistive approach and therefore produces a higher exhaust velocity.

The propellant is heated electrically by heated resistors or electric arcs and the hot gas is thermodynamically expanded in a nozzle and accelerated to supersonic speeds. The electrothermal units have thrust ranges of 0.01 to 0.5 N, with exhaust velocities of 1000 to 5000 m/sec. Ammonium, hydrogen, nitrogen or hydrazine are used as propellants.

A schematic diagram of arc heating electric propulsion system is shown below. The arc plasma temperature is around 15,000 K.

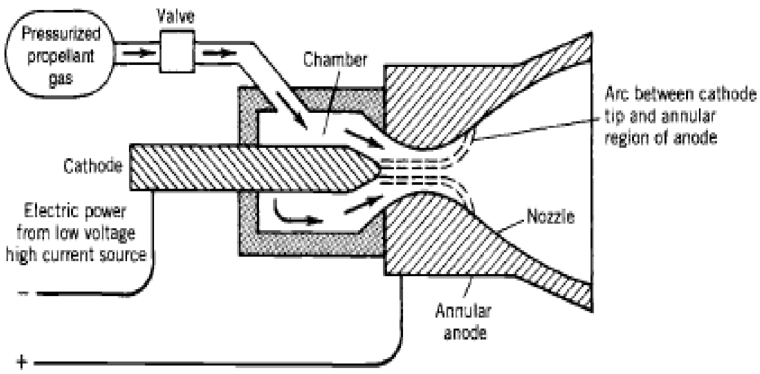


Fig: 5.2 Electromagnetic thrusters

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): In an ion rocket engine, a working fluid, like xenon, is ionized by stripping off electrons. The electrically charged heavy ions are then accelerated to very high velocities (2000 to 60,000 m/sec) by means of electrostatic fields. The ions are subsequently electrically neutralized by combining them with electrons to prevent building up of a space charge on the vehicle. A simplified schematic diagram of an Ion Rocket is shown below:

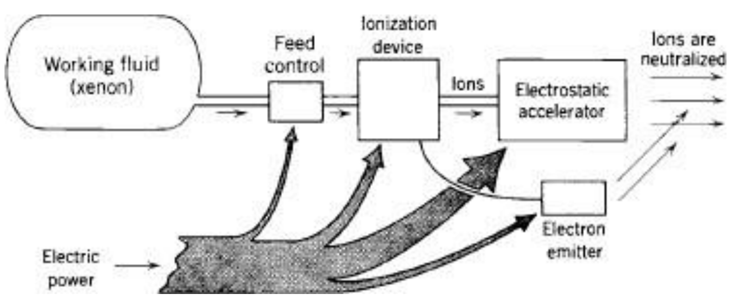


Fig: 5.3 Magneto plasma Rocket

Magneto plasma Rocket (Electromagnetic Thruster): Electrical plasma is an energized hot gas containing ions, electrons and neutral particles. In the magneto plasma rocket, an electrical plasma is accelerated by the interaction between electric currents and magnetic fields and ejected at high velocity (1000 to 50,000 m/sec).
A simple pulsed (not continuously operating) unit with a solid propellant is shown below:

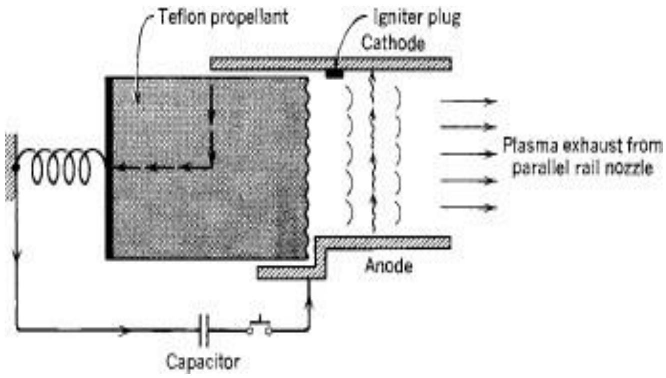


Fig: 5.4 thruster uses a parallel rail accelerator

The thruster uses a parallel rail accelerator for self-induced magnetic acceleration of a current carrying plasma. When the capacitor is discharged, an arc is struck at the left side of the rails. The high current in the plasma arc induces a magnetic field. The action of current and the magnetic field causes the plasma to be accelerated at right angles to both the current and the magnetic field, ie in the direction of the rails.

Each time an arc is created, a small amount of propellant (Teflon), is vaporized and converted in to a small plasma cloud. The plasma is then ejected giving a small pulse of thrust. The thruster can operate with many pulses per second.

The magnetoplasma rocket is used as spacecraft attitude control engine.

1.2.3 Performance of Electric Thrusters:

- The thrust levels of Electric thrusters are small relative to chemical and nuclear rockets. They have
- substantially higher specific impulse which results in longer operational life for satellites whose life is limited by quantity of propellant they carry.
- Electric thrusters give accelerations too low to overcome the high gravity field earth launches. 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 does not diminish with progress through the flight, while the mass of propellant in a chemical rocket decreases as the vehicle accelerates. This is the key difference between Electrical and chemical rockets.

Current Technology: The electrical thrusters need substantial quantities of power on board. All types of present day electrical thrusters depend on vehicle-borne power source-based on solar, chemical or nuclear energy.

- The mass of electric generating equipment, power conversion and conditioning equipment can become much higher increasing the mass of thrusters.
- This causes high increase of inert vehicle mass.

Application of Electric thrusters: The application falls into of our broad categories:

1. Overcoming translational and rotational perturbations: These would include
   - Station keeping for satellites in geosynchronous orbits (GEO),
   - Aligning telescopes or antennas in Low Earth Orbits (LEO) and Medium Earth Orbits (MEO)
   - Drag compensation for satellites in LEO and MEOs
2. Increasing satellite speed to overcome weak gravitational field, for Orbit raising from LEO to a higher orbit even up to GEO. Circularizing an elliptical orbit; This would require velocity increments of 2000m/sec to 6000 m/sec
3. Potential missions as Inter-planetary travel or Deep spaceprobes.

Electric Vehicle Performance: The propulsive force developed by an electric thruster is the momentum transferred to the propellant. The Rocket equation applies to electric thrusters;

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

R = $\frac{M_0}{M}$; M_0 is the mass of rocket at ignition (initial mass) and M is mass of vehicle (final mass) R can also be expressed as R= $\frac{M_S + M_p + M_E,}{M_S + M_E}$

where M_s is mass of structure including payload, propellant tanks and thrusters, M_p is mass of 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 ƞ, in converting electric power to thrust, which is expressed as

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

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

The exhaust velocity v_e can be expressed as

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

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

The thrust developed by the thruster F can then be written as,

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

The exhaust velocity v_e is not a free parameter. It is decided by the power P_E and the mass flow rate m. The mass flow rate m, in turn depends on burn time t and mass of propellant M_P

The energy carried away per second by the exhaust is $\frac{1}{2}$ mv_e^2, this is governed by the power converted in the thruster.

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

Higher mass flow rate also implies shorter burn time t. The

rocket equation 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} is equal to $\frac{1}{2}\dot{m}v_e^2$. The power-to-thrust ratio, $\frac{P}{F}$ can be written as

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

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; burn time (duration)=4 weeks=2.42 $\times 10^6$ sec; Payload mass=100kg; ξ =

100 W/kg; η =0.5

The flight characteristic parameters are $\dot{m}$, M_P, P_E, M_E and Velocity increment Δ V

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

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

The mass of propellant M_P = $\dot{m}$t = 1.02 $\times 10^{-5}$ ×2.42 $\times 10^6$ = 24.69 kg

The electrical power required is 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

The mass of electrical power system, M_E will be

M_E = P_E/ξ = 3.92/0.1 = 39.2 kg ;

The final vehicle mass after burnout = mass of power system + mass of payload = 39.2 + 100 kg

The initial vehicle mass M_0 = final vehicle mass + propellant mass = 39.2 + 100 + 24.69 = 163.9 kg The

velocity increment ΔV= $v_e log_e$ R = 2000 × 9.8 ln (163.9/139.2) = 3200 m/sec

2.1 System Parameters-Interrelations:

1. **Vehicle Velocity V as a function of Exhaust velocity v_e**: The relation between V and v_e is given by the equation

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

The exhaust velocity is given by the relation

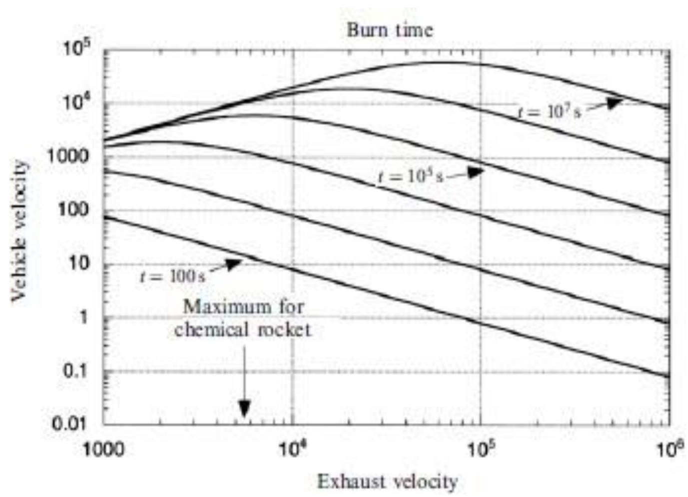
$$\frac{M_E}{M_P} = \frac{v_e^2}{2\xi t}$$

We can write the mass ratio as, R= $\frac{M_s + M_p + M_E,}{M_S+M_E}$ which can be written as

Above equations indicate that

- The mass ratio R (dry vehicle weight divided by propellant weight) for a given dry vehicle weight, decreases as the exhaust velocity increases. This is because higher exhaust velocity needs higher power supply mass.
- This means that for the electrical thrusters, an increase in v_e requires an increase mass of power source, or dry vehicle mass, thereby resulting in no improvement of vehicle velocity.

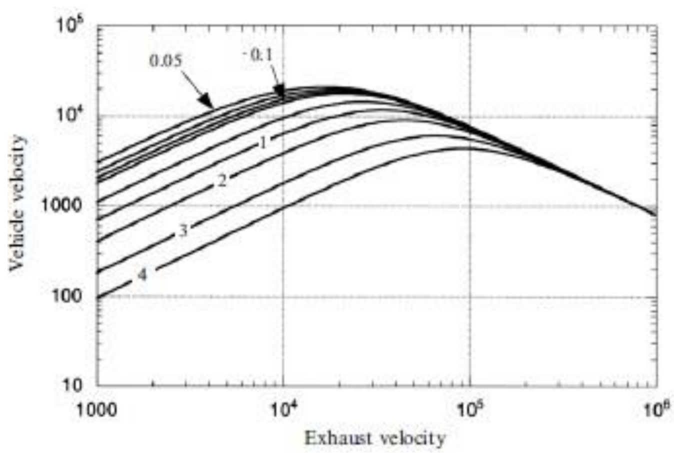
- Figure below shows vehicle velocity as a function of exhaust velocity and burn time t



- Above graph assumes a fixed relationship between exhaust velocity and power supply mass, with burn time as a parameter.
- The ratio of structural mass to propellant mass is also fixed at 0.15, equivalent to a mass ratio of 6.6.
- It is 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 to increasing mass of power supply, and hence reduction in mass ratio.
- With the mass ratio fixed for the rocket, changes in burn time indicate changes in mass flow rate. The exhaust velocity for a given power depends inversely on the mass flow rate. So low mass flow rates or long burn times are beneficial. Also, thrust is inversely proportional to the burn time, and so long burn times and high exhaust velocities imply low thrust.
- 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 are meant for bringing saving of propellant mass. Relation between vehicle velocity as a function of the ratio of payload(structural) mass to propellant mass is indicated below:

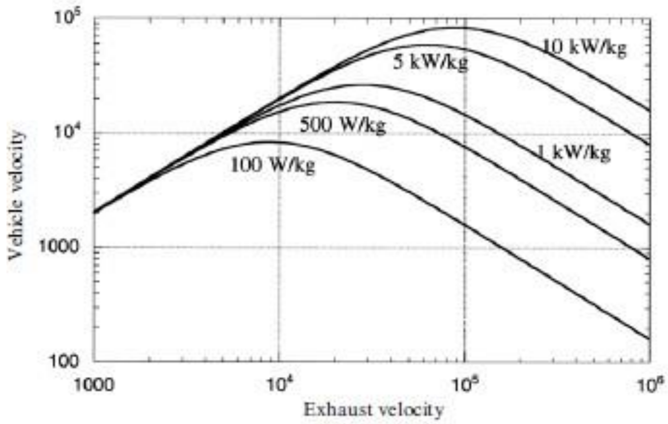
- In the interrelation below, the burn time is fixed at 1 million seconds, and the power-to-mass ratio, ξ is fixed at 500 W/kg.
- The ratio $\frac{M_S}{M_P}$ is shown as a parameter.



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 also shifts to the right, ie the peak occurs at higher exhaust velocity as the power-to-mass ratio increases.

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

- High exhaust velocities allows much higher payload-to-propellant mass ratios High
- power-to-mass ratio allows crucial in obtaining the best performance.
- The basic characteristics of electric thrusters are **high exhaust velocity, low thrust levels and long burn times**

2.2 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.

2.2.1 Resiso-jet:

Operating Principle & Components: The basic electrothermal thruster, resisto-jet, consist of a nozzle with a high expansion ratio, connected to a chamber in which the propellant is heated by a hot wire through which an electric current passes. The hot gases generated by the heated propellant passes through a nozzle and are expanded thermodynamically. The expansion in the nozzle results in a high velocity exhaust at the end of nozzle. For high exhaust velocity, the temperature and pressure of gases entering the nozzle should be high. This needs efficient heating of propellant.

To maximize heat transfer to the gas, a multichannel heat exchanger is used to bring as much of gas volume as possible in contact with the heater.

The resistor-jet thruster is illustrated below:

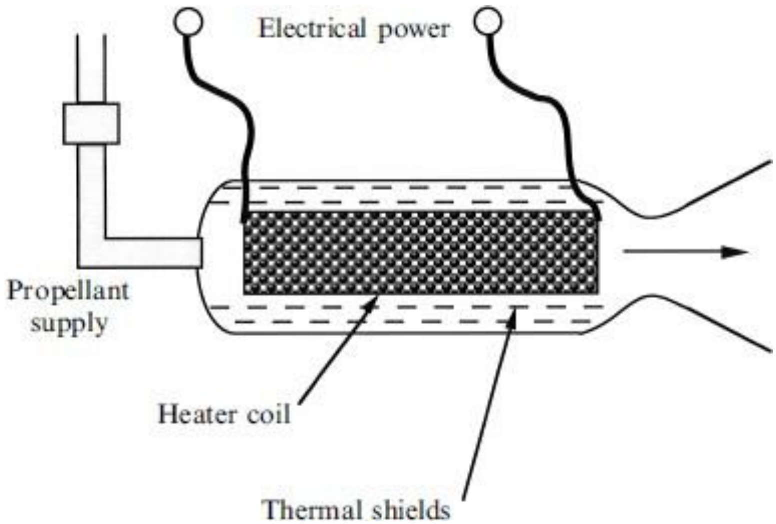


Fig: 5.5 resistor-jet thruster

System Parameters & Performance: The exhaust velocity is calculated using the thrust coefficient and characteristic velocity

$$v_e = C_F C^*$$

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

- The thrust is a function nozzle exit and chamber pressure (p_e & p_c)
- Since these thrusters are used in vacuum, high nozzle expansion ratios are used, (around 2.25 for γ = 1.2)
- While in the chemical rockets C^* depends on $\mathfrak{M}$ & T_c for the electric thrusters, C^* mainly depends on $\mathfrak{M}$. (since there is no combustion and the nozzle exit temperature depends on power input and mass flow rate)
- The nozzle exit temperature in chemical rockets depends on type of propellant, whereas in 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; Propellant is hydrogen with $\mathfrak{M}$ = 2.

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

v_e = $C_F C^*$ = 2.25 × 4659 = 10,483 m/sec

- Electric thrusters can attain very high exhaust velocities,

The mass flow rate m is calculated from

$\frac{1}{2}$ mv_e^2 = η P_E or m = $\frac{2\eta P_E}{v_e^2}$ = $\frac{2\times\eta\times1000}{10{,}483^2}$ = 1.8× 10^{-5} 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 that the vehicle can achieve very high exhaust velocities, but at low thrust values, the time taken to accelerate to such high velocities is very long
- This is the fundamental difference between chemical rockets and electric thrusters. The
- electrical efficiency can be very high at 90%

Propellants used could be hydrogen, helium, water (even waste water can be used) or hydrazene.

Disadvantages: Higher exhaust velocities and power are difficult to achieve since transfer of heat from filament to gas is difficult.

2.2.2 Arc-Jet Thruster:

Operating Principle: In the Arc-Jet thruster, the propellant gas is heated by passing an electric arc through the flow. Temperatures in the order 30,000-50,000k are achieved at the centerline which fully ionizes the propellant.

The anode and cathode are made of tungsten, which has high melting point. The cathode rod is pointed and is supported in an insulator. The insulator also holds the anode. The anode is shaped to create a gap with the pointed cathode, across which the arc is struck. The propellant flows through this gap and gets 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.

The power that can be applied across an arc-jet is up to 100 times higher than the filament of a electrothermal thruster. The temperature limit can be much higher.

While the propellant is ionized, the electrons and positive ions move towards anode and cathode. The cathode is struck at high speeds, causing vaporization of the cathode material, thereby limiting it's life.

The arcs cause concentration of energy and cause hot spots leading to erosion of the electrodes. Heat

losses due to ionization and dissociation are higher than electrothermal thrusters.

Maximum exhaust velocities are around 20 km/s. Hydrogen, ammonia and hydrazine are used as propellants.

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

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

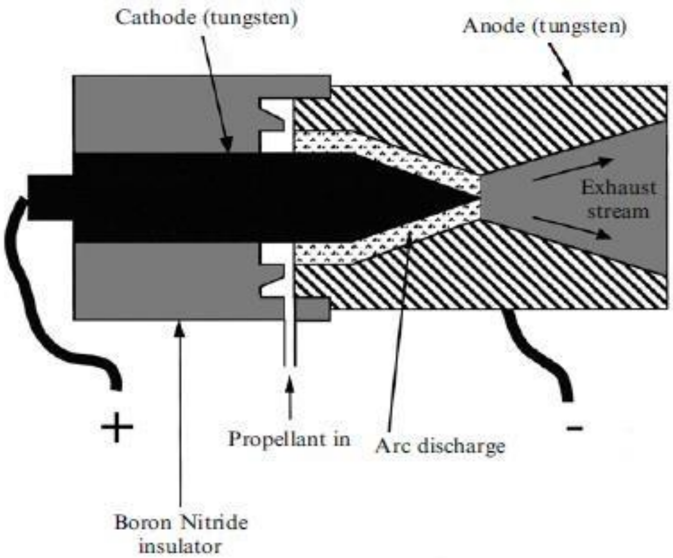


Figure 6.6. Schematic of an arc-jet thruster.

Fig: 5.6 Thermal Propulsion

2.2.3 Solar/Laser/Microwave Thermal Propulsion: Beamed energy, for example, a laser can be used for heating instead of on board energy source. Solar/laser/microwave energy source, external to the vehicle is used to heat up the propellant. The external beamed energy may be from an earth or space based infrastructure. The energy is then concentrated on a heat exchanger or directly on the propellant, which is then heated up and expelled through a conventional nozzle. Specific impulses of 800-1200 sec and thrust levels of several hundred mN are possible using sunlight and hydrogen as propellant.

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.

This concept is under development for using solar thermal propulsion to raise the communications satellite from LEO to GEO in about 20 days. This concept uses very little propellant, saving launch costs significantly.

3.1 Electrostatic Thrusters:

Performance Parameters:

- Ifthepropellantisionized,itcanbeacceleratedveryeffectivelybyelectrostaticfields.The velocity gainedforan **ion mass m** and **charge q due to the electric potential difference U is 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: The propellant is ionized, and then enters a region of strong electric field, where the positive ions are accelerated. The ions are accelerated passing through the grid and leave the engine as a high velocityexhauststream.Highestexhaustvelocities(morethan32,000m/s)areachievedby accelerating positiveionsinanelectricfieldcreatedbytwogridshavinglargepotentialdifference.

The electrons do not leave, therefore the electron current is discharged through a neutralising cathode, in to the exhaust.This would neutralise the spacecraft. The electrons discharged carry little momentum, therefore do not affect the thrust.

The thruster is divided into two chambers. Propellant, (usually Xenon gas) enters ionisation chamber in 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.

Thrust is exerted by the departing ion stream on the accelerating grids and is transferred through the body of thethrustertothespacecraft. Theexhaustvelocityisgovernedbythepotentialdifference betweenthe

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

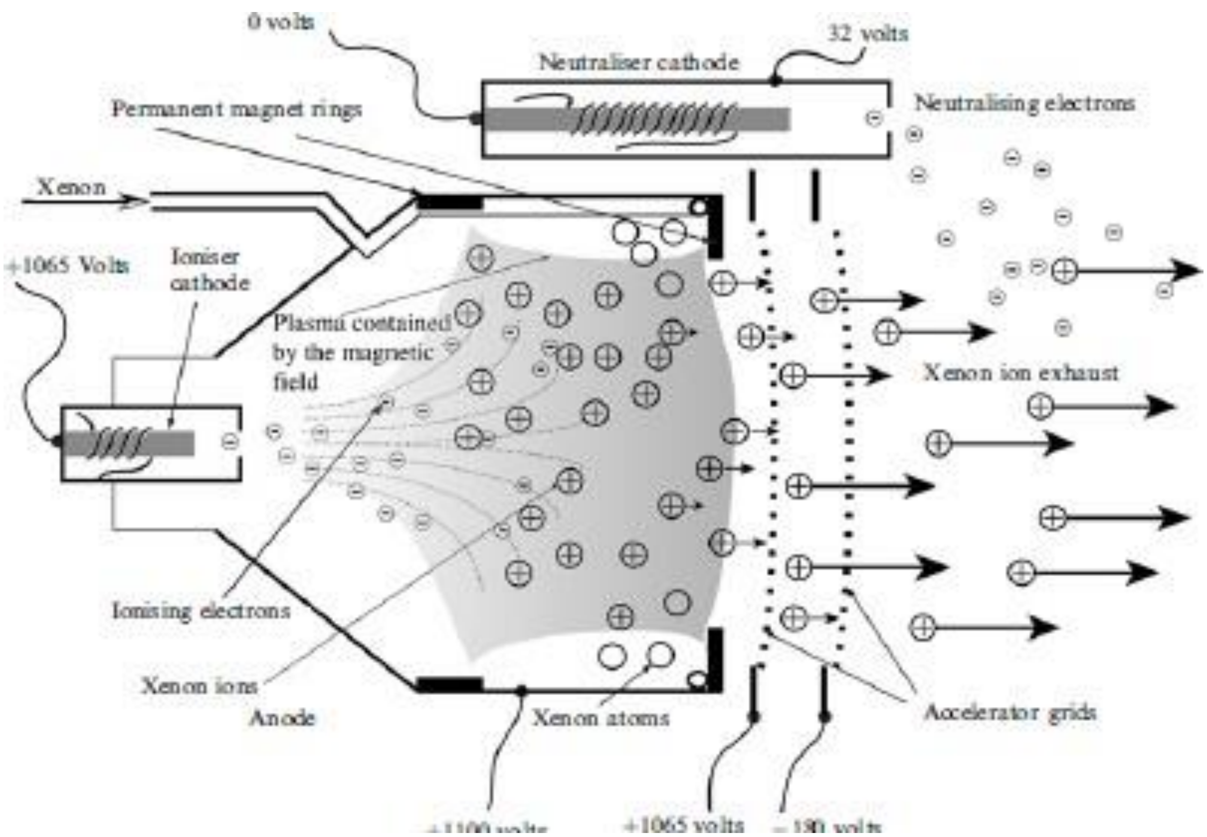


Fig: 5.7 Applications of Ion Engines:

Ion engines are best used for very high velocity increment missionslike inter-planetary missions 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 electric field between them, which gets partially blocked as the ions start acceleratingalongthegrids.Asthedensityofflowofionsincreases,apointwillreachwhenthe 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: The low thrust-high exhaust velocity ion thrusters are limited by space-charge limit. Plasma thrusters (electromagnetic thrusters) offer higher thrust values.

In plasma thrusters, an ionised gas passes through a channel across which orthogonal electric and magnetic fields are maintained. The current carried by the plasma (electrons and ions) along the electric field vector interacts with the magnetic vector, generating a high propulsive force. The plasma accelerates without the need for area change

Magnetoplasma Dynamic (MPD) thrusters and Pulsed Plasma thrusters (PPT) are conventional type of electromagnetic thrusters. The Hall Effect thruster is another variant of the electromagnetic thruster.

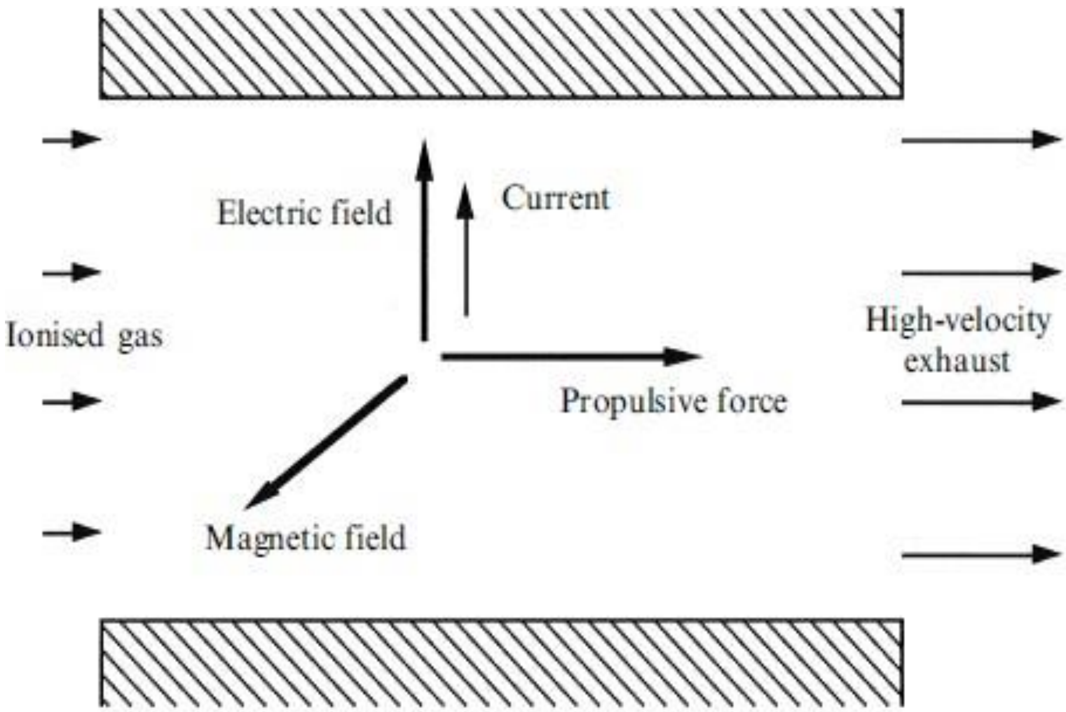


Figure 6.14. Principle of the plasma thruster.

Fig: 5.8 Pulsed plasma thruster

Pulsed plasma thruster (PPT): Plasma thrusters do not use high voltage grids or anodes/cathodes to accelerate the charged particles in the plasma, but rather uses currents and potentials which are generated internally in the plasma to accelerate the plasma ions.

While this results in lower exhaust velocities by virtue of the lack of high accelerating voltages, this type of thruster has a number of advantages.

In the PPT operation, an electric arc is passed through the fuel, causing ablation and sublimation of the fuel. The heat generated by this arc causes the resultant gas to turn into plasma, thereby creating a charged gas cloud. Due to the force of the ablation, the plasma is propelled at low speed between two charged plates (anode and cathode).

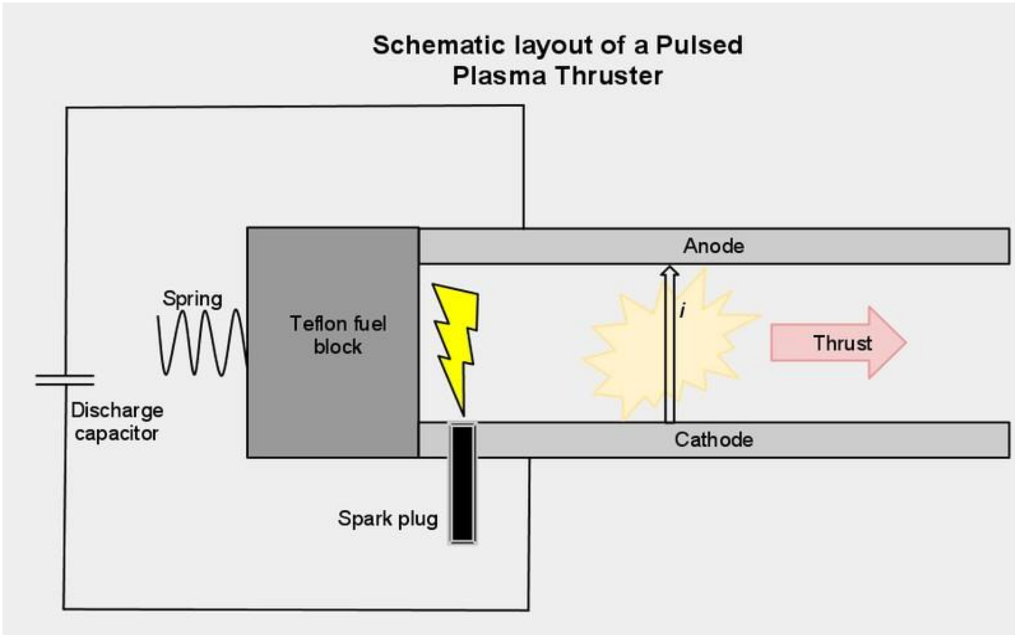


Fig: 5.9 Pulsed plasma thruster

Since the plasma is charged, the fuel effectively completes the circuit between the two plates, allowing a current to flow through the plasma. This flow of electrons generates a strong electromagnetic field which then exerts a Lorentz force on the plasma, accelerating the plasma out of the PPT exhaust at high velocity.

The time needed to recharge the plates following each burst of fuel, and the time between each arc causes pulsing. The frequency of pulsing is normally very high and so it generates an almost continuous and smooth thrust.

While the thrust generated by PPT is very low, it can operate continuously for extended periods of time, yielding a large final speed.

A solid material, teflon (PTFE) is commonly used propellant. Few PPTs use liquid or gaseous propellants also.

Magnetoplasmadynamic (MPD) thrusters: MPD thrusters, also referred as Lorentz Accelerators, use the Lorentz force (a force resulting from the interaction between a magnetic field and an electric current) to generate thrust

The electric charge flowing through the plasma in the presence of a magnetic field causing the plasma to accelerate due to the generated magnetic force.

The operation of MPD thrusters is similar to pulsed thrusters.

Hall Thrusters: Hall Effect Thrusters combine a strong magnetic field perpendicular to the electric field 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 magnetic field and these electrons confined to the field are used to ionise the propellant.

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

Operation of Hall Thruster: An electric potential between 150 and 800 volts is applied between the anode and cathode. Electrons from a hollow cathode enter a ring shaped anode with a potential difference of around 300 V.

The central spike forms one pole of the magnet, and around the inner pole, an outer circular pole forms an annular radial magnetic field in between. The propellant, usually xenon gas is fed through anode where the neutral xenon atoms diffuse into the channel, and ionised by colliding with the circulating high energy electrons.

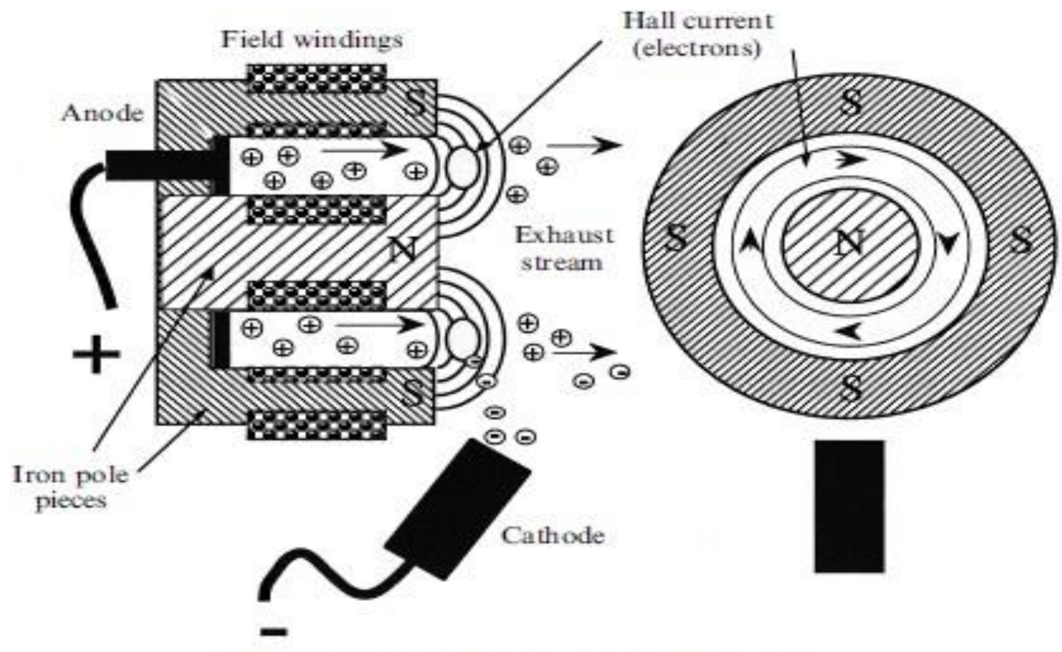


Figure 6.16. Schematic of the Hall thruster.

Fig: 5.10 Hall Thruster

The xenon ions are then accelerated by the electric field between anode and cathode. Ions reach speeds of around 15 km/sec with specific impulse of 1500 sec.

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

The accelerating ions also pull some electrons forming a plume. The remaining electrons are stuck orbiting the region, forming a circulating hall current. This circulating electrons of hall current ionise almost all the propellant.

Hall thrusters can provide exhaust velocities of 10-80 km/s and specific impulse 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 orbiting satellites and to power the main propulsion engine for medium-size robotic space vehicles.

2 KW Hall Thruster in Operation

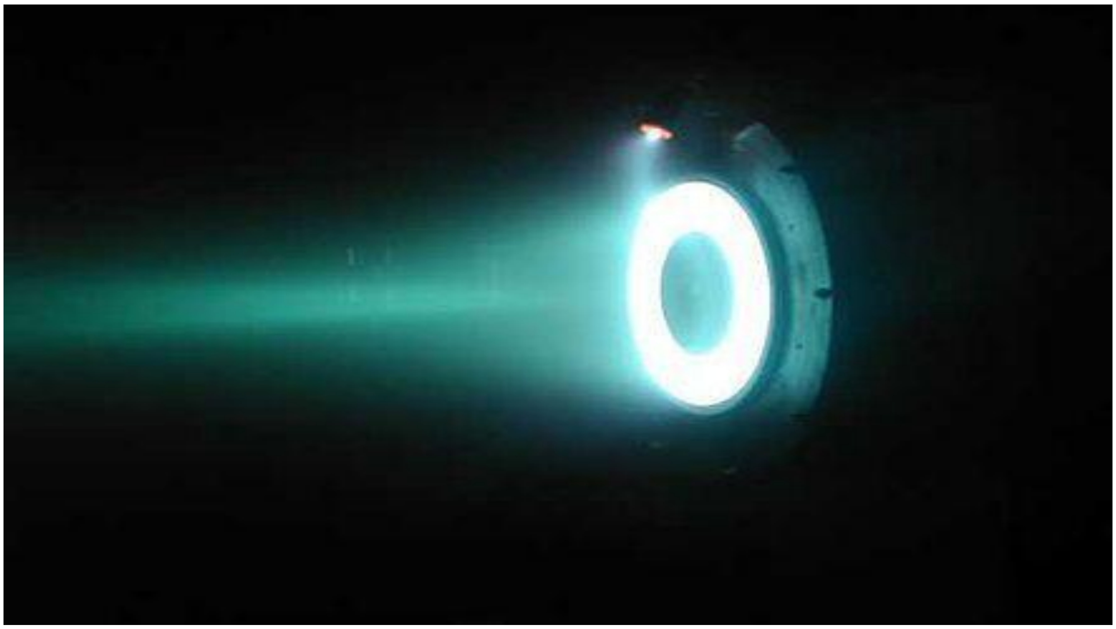


Fig: 5.11 Electric Thrusters

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 perturbations in orbits; **Drag compensation** for satellites in Low Earth Orbits; **Aligning** telescopes or antennas. **Electro-thermal (resisto-jets)** are preferred using low cost propellant like cold gas or waste water. MPD thrusters are also being considered for attitude control of space vehicles.
2. **Station Keeping**: For station keeping purpose, savings in propellant mass is very significant. Synchronous and GEO satellites have long life periods need extensive station keeping requirement. **Electro-thermal (Arc-jets) thrusters** have been widely used for this task. Hall thrusters and Ion engines are most suitable.
3. **Raising Orbits**: From low earth to higher orbits (up to Geostationary orbits), circularizing an elliptical orbit Inter-planetary travel and deep space probes. They all require relatively high thrust and power in the range of around 100kW, much higher velocity increments than those needed for station keeping. Also these corrections need to carried out in reasonable length of time. **Hall thrusters and Ion engines** are again preferred here.

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

CHAPTER II

INTE.RNAL BALLISTICS OF ROCKET PROPELLANTS

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There is a broad usage of solid rocket propellants in propulsion systems and gas generators. In such systems, there are challenging problems involving many complicated aerothermochemical processes. Solutions of such complicated systems require a good understanding of many processes like heat transfer, turbulence combustion, diffusion, chemical reactions etc. The combustion process in a solid rocket chamber helps to visualize the burning rate behaviour of solid propellant. Because the reaction rates during combustion are much faster than those of commonly observed chemical reactions and because physical changes such as velocity, temperature and pressure during combustion , a knowledge of aerotheromocheinistry is needed to understand combustion phenomena. It is the purpose here to present advances in combustion phenomenon of different types of propellants and to present the aerothermocliemical equations applicable to solid propellants

1. PERFORMANCE OF A SOLID ROCKET' PROPELLANT'

COMBUSTION :- It is an exothermic reaction involving rapid oxidising reactions. T'he combustion phenomenon of solid propellants are much more complex than those of gas because of the phase changes involved in going from a solid to a liquid to a gas.

PROPELLANT :- It is defined as the material which generates a large number of gaseous molecules at high temperature during combustion which can self-sustain without the presence of ambient oxidises combustion In other words propellant is a material which burns rapidly but does not explode and is self sustaining during burning Major applications of propellants are in rockets and guns both of which use gas at hig*' r ressure and temperature to Generate propulsive forces. Rockets consists of propellant and combustion chamber within their projectiles, whereas gun projectiles usually do not. The solid propellant rockets work at operating pressures between 10 and 300 atm.

A rocket motor is a typical example of an energy transfer system which can be directly explained by thermodynamics and Newton's third law a pressurized high temperature gas generated in the system is expanded adiabatically and the sensible energy of the gas is converted to kinetic energy. Thus the system produces a reaction force. Therefore, the thermodynamic requirement for rocket motor is to get a lugh pressure and high temperature gas in the motor, and to convert the sensible energy effectively into the kinetic energy through physicocheinical processes. Thus, temperature in the chamber, molecular weight of combustion gases and ratio of specific heats are the performance parameters '"2. However, these parameters cannot be measured. Hence these parameters are converted to the following parameters.

CHARACTERISTICS VELOCITY: It is defined as

(21)

Experimentally, (?* = $\frac{gA, PJl}{M}$ (2.2)

It is the figure of merit of the propellant. It depends upon the combustion flame temperature and molecular weight of combustion gases. It represents the combustion efficiency of propellant.

TDRU$T COEFFICIENT: It is defined as,

$$\frac{}{A_t P_c} \quad (2.3)$$

for non viscous comiessible gas ,

$$C_F = \Gamma\sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} \quad (2.4)$$

It depends upon the expansion ratio of the nozzle as seen from equation (2.4).

SPECIFIC IMPUL$E: It is defined as the thrust per unit weight flow rate of propellant.

$$I_{sp} = \frac{}{,,,} \quad (2.3)$$

and it is

$$I_{sp} = \frac{1}{g}\sqrt{\frac{2\gamma}{\gamma-1}\frac{RT_c}{M}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} \quad (2.6)$$

for radially expanding gases.

BUItNlNG ItA"I“E : It is a distance, normal to the burning surface of the propellant, travelled by a flame fTont per unit time. It is a key factor in determining the performance of solid propellant. In general, burning rate, called Vielle’s law, 1s given by

$$r = aP' \quad (2.7)$$

is assumed for all types of propellants.

Since the gas generation in the chamber is a function of the propellant burning and mass discharge through the nozzle, the mass balance in the rocket motor can be written as

$$\frac{d}{dt}\left(\rho_g V_c\right) = \dot{m}_g - \dot{m}_d \quad (2.8)$$

Under steady state condition, RHS of equation (2.8) vanishes. However, this term plays a dominant role in ignition transients, oscillary burning, and combustion tCrmination. The rate of mass discharge is given by

$$\tilde{m} = \frac{A,1'.g}{} \quad (2.9)$$

which is purely a fluid dynamical relation.

’the rate of gas generation in the combustion chamber is equal to the mass burnins rate of the propellant.

$$n \qquad pApr \quad (2.10)$$

Under the steady state,

$$A_b r = \frac{A_t P_c g}{*} \quad (2.11)$$

$$P_c^{1-n} = \quad (2.12)$$

The combustion index n can never exceed one and it is desirable to have low or even a negative value. If n is smaller than one, alter a small increase in pressure the mass flow through the nozzle becomes greater than the mass burning rate and the pressure returns towards equilibrium. The smaller the value of n, the quicker the return. On the other hand, if n is greater than unity, the mass burning rate exceeds the discharge rate and pressure increases until an explosion occurs. Inversely following a negative differential, the pressure continues to drop until combustion stops. Therefore stable burning can only be achieved if the combustion index n is smaller than one.

2. COMBUSTION PROCL$$ OF VARIOUS TYPES OF SOLID PROPELLANTS

It is of prime importance for solid propellants to satisfy the various combustion characteristics described in the previous section to obtain stable and high performance combustion in rocket motors. However, the physical and chemical properties of these propellants are entirely different from each other, and the combustion processes are also different. 4“he following paragraphs describe the flame structure and burning rate characteristics of various types of solid propellants. The propellants are double base propellants, ammonium perchlorate based composite propellants, composite modified double base propellants and nitramine composite propellants.

2.1 DOUBLk BA$E PROPELLANTS.- Since the physical structure of a double base propellants is homogeneous, thC Combustion flame structure appears to be homogeneous and one dimensional along the burning direction. The decomposed gas produced at the burning surface is premixed with oxidizer and fuel on a molecular scale. The luminous flame stands some distance above the burning surface and approaches the burning surface when the pressure is raised. Between the burning surface of the propellant and luminous flame, a transparent region exists and is called the dark zone, which is equivalent to the flame stand off clistance, decreases with inci‘easing pressure, or in other words with inci casing (our ning mites

1 ollowing ttiblc cx|3liiiiis this :ispccl

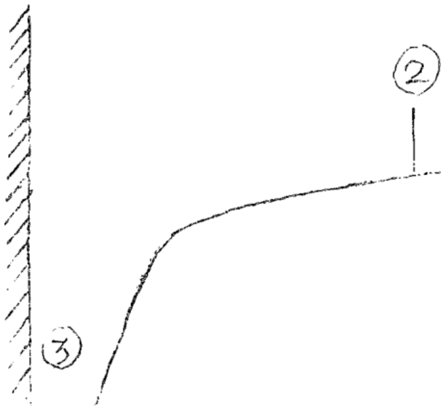
p (at ni)	r (cm/s)	thickness of fl‹itiic (nun)
10	0.22	1.30
20	0.31	0.33
30	0.40	0.14

The flame structure for double base propellant is shown in fig.1

2.2 AMMONIUM PERCHLORATE COMPOSITE PROPELLAN"£.- Since the physical structure of composite propellant is heterogeneous, the combustion wave structure appears to be heterogeneous also. The luminous flame is attached to the burning surface even at low pressures. At the burning surface, the decomposed gases from ammonium perchlorate particles and the fuel binder are interdiffused and produced diffusion frame structure. 4“hus, the flame structure of AP composite propellants is complex and locally three dimensional in shape. The luminous flame stand off distance above the burning surface of AP- composite propellants is less dependent on pi essure compared with double base propellants. It is evident that the diffusion process plays a significant role on the burning rate of this type of propellants. The combustion wave structure is illustrated in fig 2

2.3 COMPOSITE MOl3IFlkD DOUBLE BA$E PROPELLANTS. ’1 he specific impulse of double base propellants is less than that of AP- composite propellants. Thus, oiten oxidizing or lugh energy crystalline particles are added to double base propellants in order to increase the specific im|iulse. These types of propellants are called composite modified double base propellants (CMDB) propellants. In general, double base propellants contain less oxidizing fragments. Thus, the addition of AP shifts the equivalence ratio of decomposed gases to stoichinietric and T_f and Ip increase.

The AP-CMDB propellants docs not have the dark zone of the double base pi opcllanl s. which is i iscrl ,is ,i lease wi‹i i i ix "I“his is a siynifica nl ‹lilTci ence when ct›uqi:ii c ‹1 with the double base propellants. as the concentration of All increases, the flame structure



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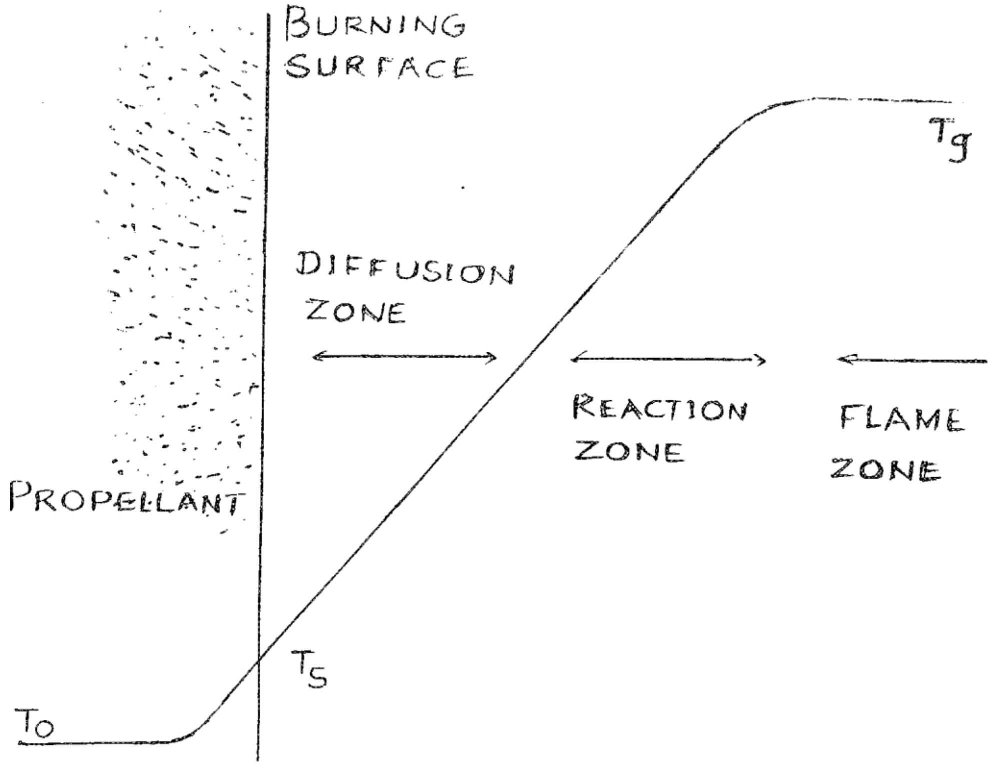
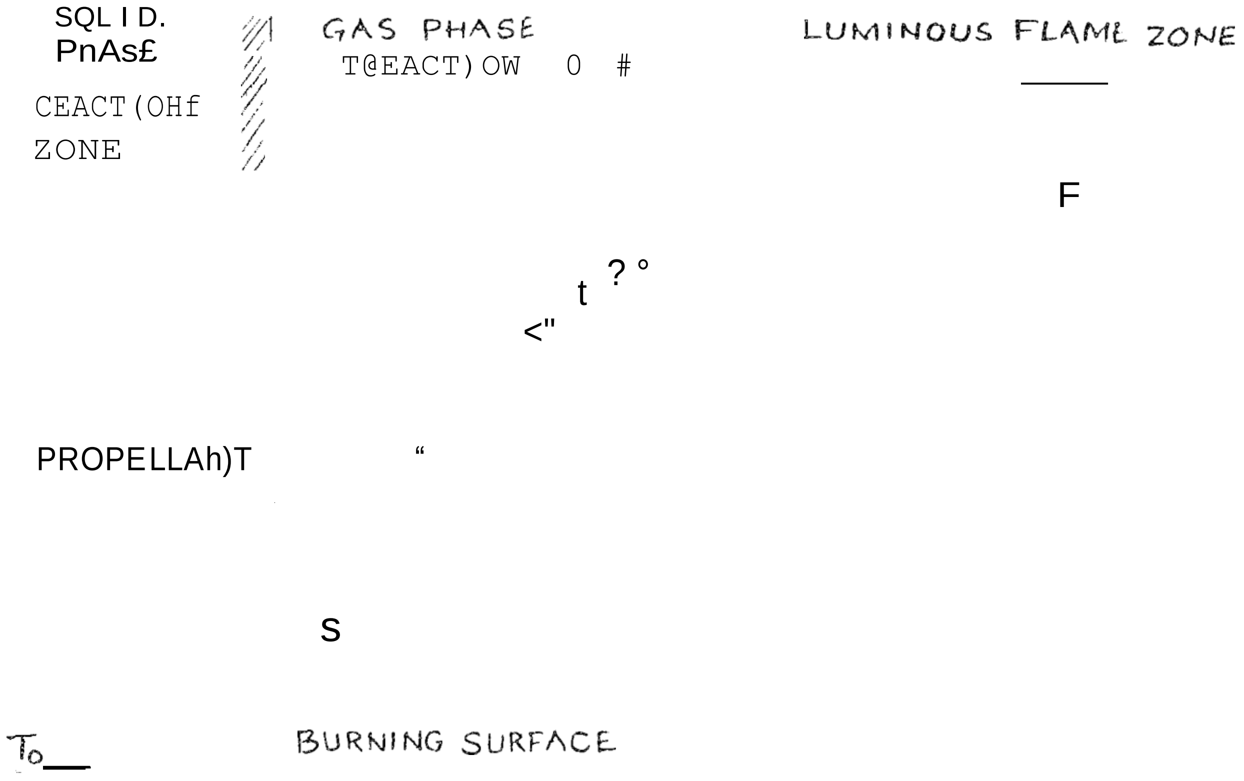


FIG 2: COMBUSTION WAVE STRUCTURE OF AP-PROPELLANT

and the burning rate characteristics approach that of AP composite pi‘opellaiits, and the combustion wave becomes highly heterogeneous. ’the decomposecl fuel-rich gas, produced from the double base propellants, and the decomposed oxidizes -rich gas, produced from AP produce diffusion flames above the particles of AP. Diffusion flames are produced around the AP monopropellant flame streams which are produced bv each particle of AP. As the number of AP particles increases, the number of diffusion flamelets increases in the gas phase An extreme case, the flame structure of double base propellants which is used as a base matrix, is eliminated. completely, and the burning rate of the propellant becomes similar to that of’ AP-composite prope[lants. "these characteristics are dependent on the concentration and particle size of AP. The buriñg rate increases as the AP concentration increases for the same particle size of AP, it also increases when the size of AB decreases. The wa e structure for such type of propellant is depicted in fig 3

When RDX is mixecl with a double base propellant, the specific impulse increases because of the increased flame ternperatuFC. Since the combustion products of RDX and double base propellants are fuel-rich, exothertnic chemical reactions are possible between their combustion products "I“hus, the flame temperature and the specific impulse of the RD *X -G MI9 D* propellants is similar to that of double base propellants and is significantly different fi’om that of AP-(iMI3B propellants. T’he combustion wave consists of three successive reaction zones fizz zone and luminous flame zones. ’I’he luminous flame stands some distance above the burnins surface. 3“he flame standoff distance decreases with increasing pressure. Increasing the pressure results in the luminous flame approaching the burning surface

the crystalline RBX particles mixed with the base matrix of double base propellants riielt, decompose and gasify at the burning surface of the propellants. the decomposed RI)X gas diffuses into the dec‹xnposed gas of the base matrix at and just above the burning surface 3 his diffusion pi‘ocess occurs before the RDX particles produced their inonopropellant flarnclets ahove the burnitig surface Thus, a hon' 8eneous



t ; /OfdLUSTIOU \^/ AVE 3TRUCTU RE OF CMD& PROPL 1LANY reactive gas of the mixture of the RDX and the base matrix decomposed gases is formed at the burning surface and produces the luminous flame at some distance above the burning surface. 4 hus the combustion wave is one-dimensional in the burning direction. "I"he burning rate of kDX-CMDB propellant decreases when the concentration of RDX inci eases and is nearly independent of particle size

2.4 NITWJX COMPO$I"fk PltOPELLANT$: The physical structure of RBX composite prope1l‹ints is very heterogeneous and is siiTlllar to the physical structure of AP-composite propellants. However, the combustion wave structure is homogeneous and is similar to the combustion wave structure of double base propellants. When the concentration of RBX increases, the burning rate of RDX-composite propellants increases with the burning rate relatively independent of CDX particle six.e.

3. GOVERNING EQUATIONS :

3•1 THk PROCJ$S OF PRO PkLLANT BURNIN G: As discussed earlier, combustion performance of solid propellant rocket motors can be improved by the following physico-chemical parameters

i) Increasing specific impulse

ii) obtaining a spectrum of propellant burnin b rate

iii) reducing the pressure exponent of burning rate and

iv) reducing the temperature sensitivity of burning rate.

All of these items are related to the phenomena of propellant combustion mechanism Various combustion phenomenon occurring during rocket motor operation, such as ignition, erosive burning, oscillatory burning and combustion determination must be studied using the principles of heat transfer, fluid mechanics, and chemical kinetics.

"I'he burning process is largely dependent on the combustion wave structure. This wave structure is a function tif the propellant composition, pressure and various operating conditions. Propellan ts produce heat and high temperature gas by the phenomenon of combustion The heat 1"eed back from the high temperature gas to the unburnt part of the propellants raises this portion of the propellants to the decomposition temperature. As a result, the unburnt portion gasifies and produces the heat by an exothermic chemical reaction. This successive heat feed back process makes the propellant burning occur continuously to sustain a steady state burning.

3.2 COM B **UST ION WAVE STRUCTURE** : The fig 4 is a schematic drawing of a simplified combustion wave structure of solid propellant. The temperature increases rapidly from the initial temperature to the burning surface temperature Ts just beneath the burning surface. At the burning surface, either an endothermic or exothermic decomposition reaction produces reactive gases. This reaction zone is called the solid phase reaction or condensed phase reaction zone. The reactive gases emitted from the burning surface, are the gas phase above the burning surface and generate heat to form a luminous flame zone. Thus, the temperature increases in the gas phase reaction zone and reaches a maximum in the luminous flame zone. In the luminous flame zone, the exothermic reaction is completed and equilibrium combustion products are formed. The combustion products flow away continuously downstream of the combustion zone and burning surface redresses in the opposite direction of the combustion product flow. The heat balance at the burning surface is

$$\lambda_P \left(\frac{dI'}{dx}\right)_s = z_g \left(\frac{dI''}{dx}\right)_{s} + \rho_P r Q_s + I_g \qquad (2.13)$$

a.z **CONSERVATIO N EQUATIONS:** The conservation equations of mass, momentum, energy and the chemical species in the combustion wave for ideal gas can be written as

MASS 9 (pv) = 0 (2.14)

MOMENTUM pv V v = - 9 p (2.15)

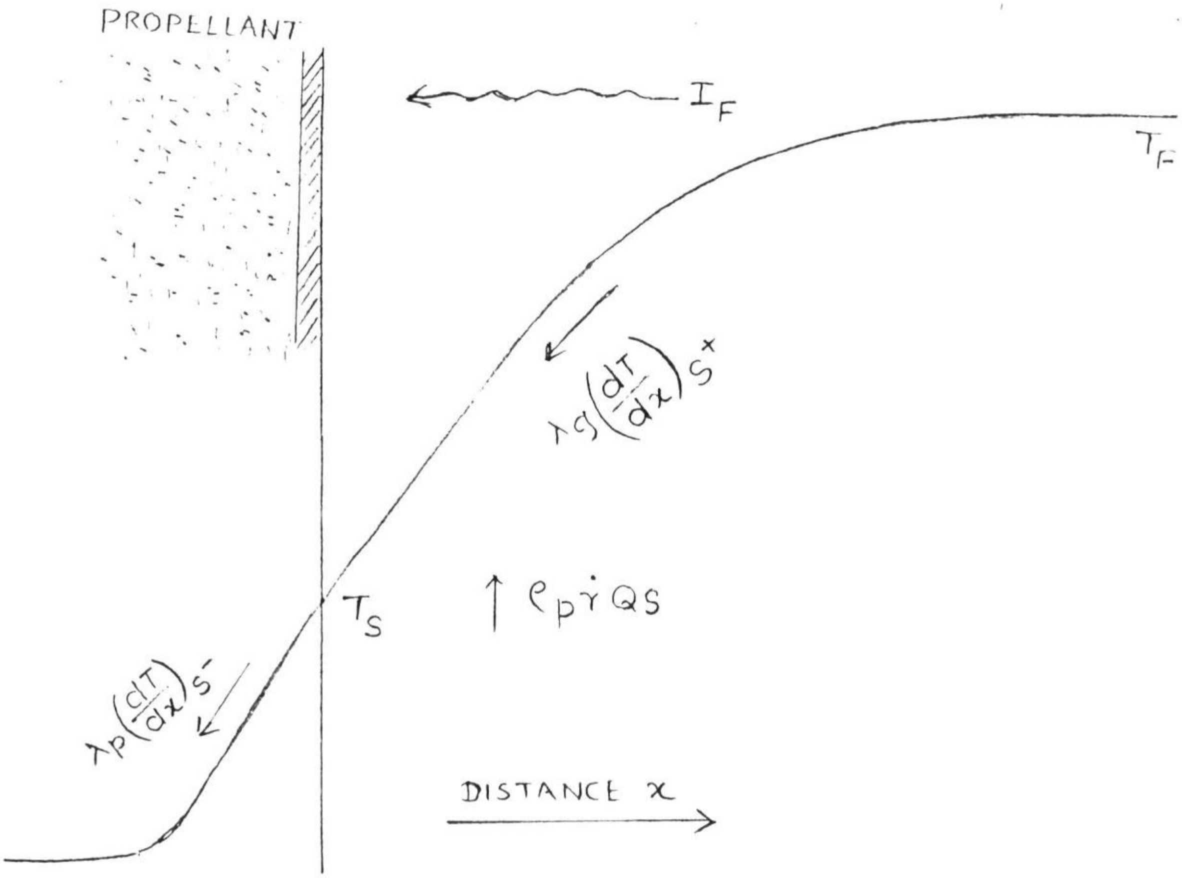


FIG 4 : HEAT BALANCE AT BURNING SURFACE

$$\text{ENERGY} \quad : \nabla.(\rho c_c T - \lambda \Delta T) = - \Sigma \omega_i R_i \qquad (21G$$

$$\text{SPECIES} \quad : \nabla.(\rho v Y_i - \rho D \nabla Y_i) = \omega_i \qquad (2.17)$$

:AT TRANSFER IN A SOLID PHASE .

Assuming that the chemical reaction does not occur in the solid phase, the energy equation in t)ie solid phase can be written as

$$2 \frac{\hat{a}\, 1}{} - \text{puc} \frac{cl1}{} - 0 \qquad (2.18)$$

Using the mass balance equation in the solid and gas phase, this energy equation can be written as

$$\lambda_P \frac{d^2 T}{2} - \rho_P r c_P \frac{dT}{dx} = 0 \qquad (2.19)$$

where ñp and c are assumed to be independent of temperature. Integrating this equation between T = To at x = - and T = T, at x = 0 .We get the heat feed back from the burning surface to the solid phase at x = 0 as

$$\lambda_P \left(\frac{dT}{dx} \right)_s = \rho_p r c_p (T_s - T_0) \qquad (2.20)$$

hence the temperature profile is

$$T(x) \quad F(o)' \quad T \quad o) \exp \left(\quad '' \; '' \; 'I\bullet \quad \right) \qquad (2.21)$$

3.5 HEAT TRANSFER IN A GAS **PHA$E** .- In the gas phase reaction zone, the reaction process is very complicated with a large number of chemical species and elementary reactions occurring. 'I“o understand the reaction processes occurring in the gas phase, the reaction rates for each elementary reaction must be determined and the conservation equations must be solvent. However, in general, the exact solutions of the conservation equations are ditllcult to obtain because of the lack of the kinetic parameters in the gas phase. In order to determine the rate of heat transfer from the gas phase to the burning surface and to determine the burning rate of the propellant, simpliéed reaction rate models are given below.

The equation of energy can be written as

$$\lambda_g \frac{d^2T}{dx^2} - \rho_g u_g c_{pg} \frac{dT}{dx} + \dot{\omega}_g Q_g = 0 \qquad (2,22)$$

.siiicc ‹»p is ii liiiicl i‹›ii t›I“ I‹.iiq›c”t i\I cii I hix c‹|ii1lI it›tt in II ii‹›it- liitcf\i im‹l 1Imtlyl ic:tIl x‹›I\ii i‹›i1 is hut t›ussil›lc.

Assuming o, is constant initially and integrating this equation we get,

$$\lambda_g \left. \frac{dT}{dx} \right|_{s^+} = Q_g \int_0^{\infty} \exp\left(\frac{-\rho_g u_g c_{pg} x}{\lambda_g} \right) \dot{\omega}_g \, dx \qquad (2.23)$$

There will be three cases as shown in fig 5.

i) "I"he chemical reactions occur in a limited region only and reaction rate is constant throughout the region between x = xr and x = x, shown in the figure above. In this case the reaction rate in the gas phase can be expressed as a step function. Flence (2.23) becomes

$$\lambda_g \left[\frac{dT}{dx} \right]_{s^+} = \frac{\lambda_g}{\rho_g u_g c_{pg}} \dot{\omega}_g Q_g \left\{ \exp\left(\frac{-\rho_g u_g c_{pg}}{\lambda_g} x_i \right) - \exp\left(\frac{\rho_g u_g c_{pg}}{\lambda_g} \right) x_F \right\} \qquad (2.24)$$

For xi = xt, the thickness of the reaction zone is extremely thin and the reaction rate is assumed to be infinite. Hence in this case we have,

$$\lambda_g \left[\frac{dT}{dx} \right]_{s^+} = \rho_g u_g Q_g \exp\left[\frac{-\rho_g u_g c_{pg}}{\lambda_g} x_F \right] \qquad (2.25)$$

For xi = 0, the reaction starts at the burning surface and we get.

$$\lambda_g \left. \frac{dT}{dx} \right]_{s^+} = \frac{\lambda_g}{\rho_g u_g c_{pg}} \dot{\omega}_g Q_g = \frac{\lambda_g}{\rho_p r c_{pg}} \dot{\omega}_g Q_g \qquad (2\ 2G)$$

3.6 LAMINAR BOUNDARY LAYkR kQUATIONS .- The governing partial differential equations for chemically reacting boundary layer flow are deduced from the conservation equations for multicomponent reacting system. The order of magnitude analysis has been used to simplify the generalized momentum, eneryy and species equations for two dimensional laminar reacting boundary layer flow. The equations for the flow under these conditions are

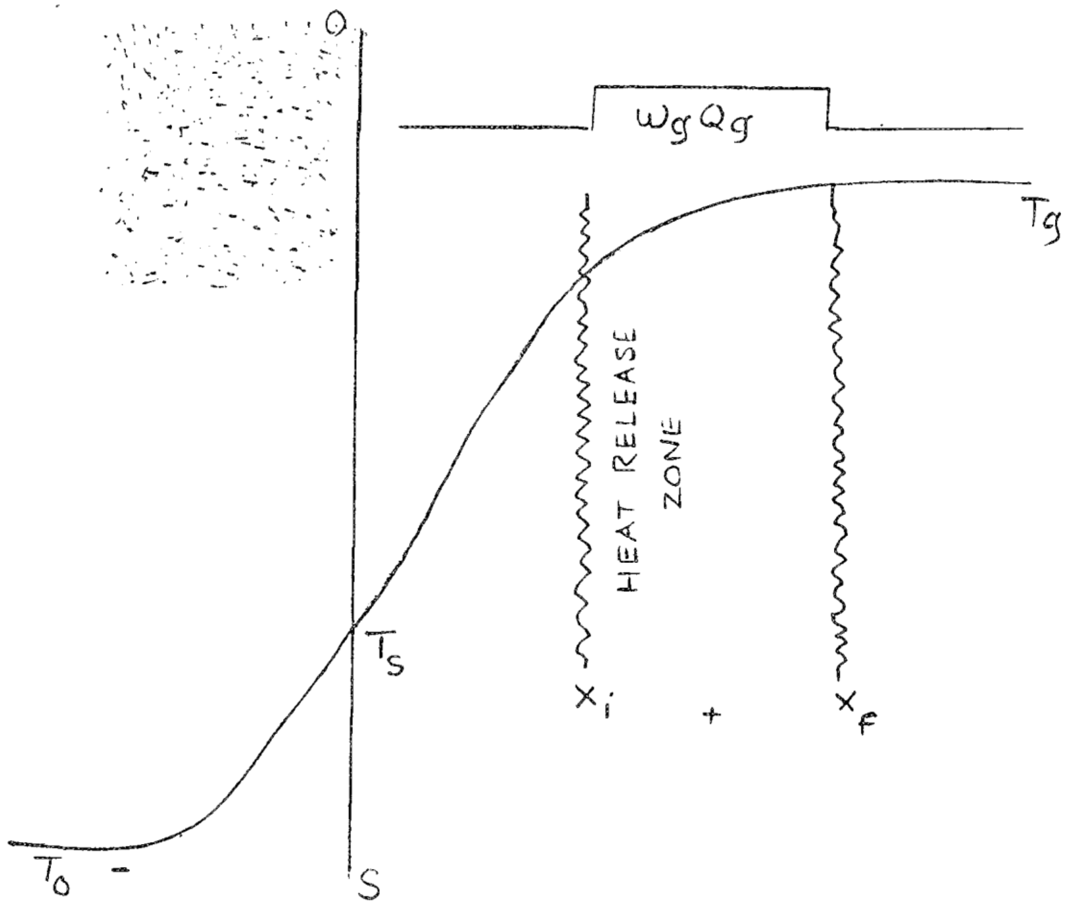


FIG 5 : GAS PHASE HEAT TRANSFER

$$\frac{\partial(\rho u)}{\partial x} + \frac{\partial(\rho v)}{\partial y} = 0$$

$$u\frac{\partial u}{\partial x} + v\frac{\partial u}{\partial y} = -\frac{1}{\rho}\frac{\partial P}{\partial x} + \frac{1}{\rho}\frac{\partial}{\partial y}\left(\mu\frac{\partial u}{\partial y}\right) \qquad (2.28)$$

$$4 \qquad (2.29)$$

$$\rho\left(u\frac{}{\partial x} + v\frac{}{\partial y}\right) \quad \frac{}{\partial y}\left\{\frac{\mu}{\Pr}\left[\frac{\partial h}{\partial y}\right]\right\} \qquad (2.30)$$

$$\rho u\frac{\partial Y_i}{\partial x} + \rho v\frac{\partial Y_i}{\partial y} - \frac{\partial}{\partial y}\left(\rho D_i\frac{\partial Y_i}{\partial y}\right) = \dot{\omega}_i \qquad (2.31)$$

$$\frac{r}{\quad} RT \qquad (2.32)$$

3.7 GOvrRNlNG x9tia"i“ioxs or russULEN"I’ BOUNDARY LAYra ›a/i"rii

CHEMICAL REACTIONS : This section outlines the theoretical model for a reacting turbulent boundary layer. The theoretical model comprises of the conservation equations of mass, momentum, species, enthalpy, turbulent kinetic energy and turbulent dissipation rate and the equations of state.

"l“he physical model considered in the theoretical analysis consists of flow of gases (a) over a flat plate or’ (b) inside a cylinder.

In both cases, the surface could be subjected to blowing or suction or could be a burning surface. However in the present approach, the surface is the burning surface of a solid propellant. A set of general conservation equation for a reacting turbulent compressible fluid flow are given below The boundary layer can be assumed to be quasi-steady planar or axi-symmetric and chemically reacting. Major assumptions are

l. The average flow properties are steady

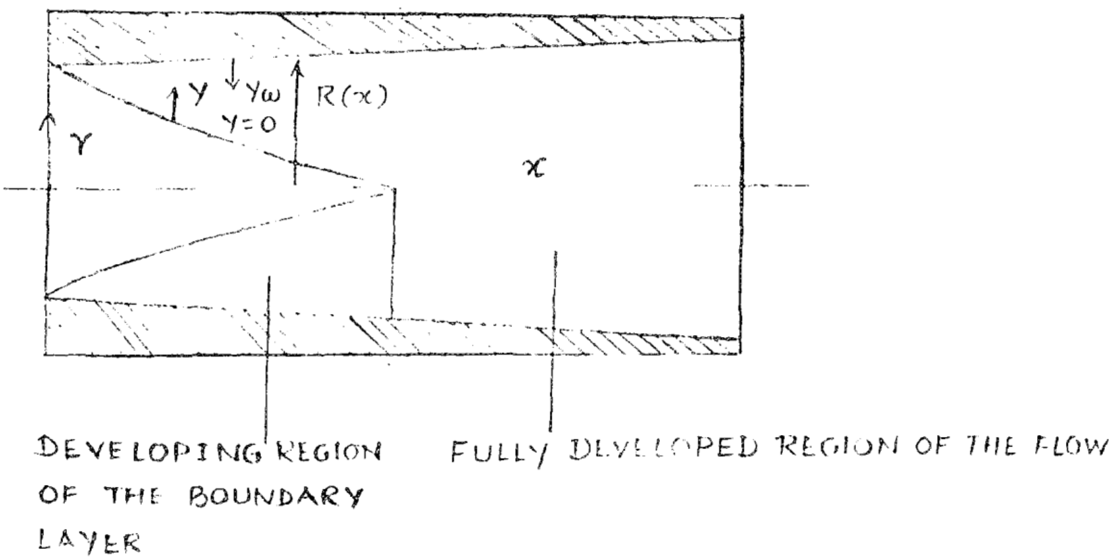


FIG 6 : PHYSICAL MODEL

2. II ody to i‘CCS ill ti él Still(

3. There is no radiative heat ttansl’er

4. "1“he 1 cwis number is unity.

5. Fick's law of diffusion is valid

A second order two etJuation K - e turbulence model is used to achieve closure of turbulent How problem. Using these assumptions and following order of magnitude analysis, consemation equations for mass, momentum, species, enthalpy, turbulent kinetic energy and tuTbulent diffusion and the equation of state are obtained as

$$\frac{\partial}{\partial x}\left(r^m \bar{\rho}\bar{u}\right)+\frac{\partial}{\partial y}\left(r^m \bar{\rho}\bar{v}\right)=0 \qquad (2.33)$$

$$\rho\bar{u}\frac{}{\partial x}+\bar{\rho}v\frac{}{\partial y}=\frac{}{r^m}\frac{\partial}{\partial y}\left[r^m \mu_{eff}\frac{\partial\bar{u}}{\partial y}\right]-\frac{\partial\bar{P}}{\partial x} \qquad (2.34)$$

$$\bar{\rho}\bar{u}\frac{}{\partial y}+\bar{\rho}\bar{v}\frac{\partial Y_i}{\partial y}=\frac{1}{r^m}\frac{\partial}{\partial y}\left[r^m\left(\frac{\mu}{Sc}\right)_{eff}\frac{\partial Y_i}{\partial y}\right]+\bar{\dot{\omega}} \qquad (2.35)$$

$$\frac{\partial\bar{H}}{\partial x}+\bar{\rho}\bar{v}\frac{\partial\bar{H}}{\partial y}=\frac{1}{r^m}\frac{\partial}{\partial y}\left[r^m\left(\frac{\mu}{\Pr}\right)_{eff}\frac{\partial\bar{H}}{\partial y}+\left\{\mu_{eff}\quad\left(\frac{\mu}{\Pr}\right)_{eff}\right\}\frac{\partial\left(\bar{u}^2/2\right)}{\partial y}\right]$$

(2.36)

$$\bar{\rho}\bar{u}\frac{}{\partial x}+\bar{\rho}\bar{v}\frac{}{\partial y}=\frac{1}{r^m}\frac{}{\partial y}\left[r^m\left(\mu+\frac{\mu_t}{c_1}\right)\frac{\partial K}{g}\right]+\mu_t\left(\frac{}{\partial y}\right)^2-\rho\varepsilon$$

(2.37)

$$y‹\bullet\frac{\partial\varepsilon}{\partial x}+\bar{\rho}\bar{v}\frac{\partial\varepsilon}{\partial y}\quad\frac{1}{r^m}\frac{\partial}{\partial y}\left[r'''\left(u+\frac{\mu_f}{c_2}\right)\frac{\partial\varepsilon}{\partial y}\right]+c_3\mu_t\left(\frac{\partial\bar{u}}{\partial y}\right)^2\frac{\varepsilon}{K}-c_4\rho\frac{\varepsilon^2}{K}$$

(2.38)

$$\bar{P}=\frac{\bar{\rho}RuT}{} \qquad (2.39)$$

4. PROCESS OF DIFFUSION

A diffusion flame may be defined as any flame in which the fuel and oxidizer are initially separated (non-premixed). Diffusion flames are either laminar or turbulent.

4.1 LAMINAR DIFFUSION : In this type of flame, fuel and air flow with the same linear velocity. Molecules behave in an orderly manner and the flow is almost stagnant

4.2 TURBULENT DIFFUSION : When the reactants enter in a separate streams, oxidizer and fuel being non-premixed, the resulting flames are called the turbulent diffusion flames. In this type of flow the molecules behave in disorderly manner, there will be eddies formation.

4.3 F1CK'S LAW OF DIFFUSION'[6]: The mass diffusion velocity v, in the y-direction is given by

$$v_i = -\frac{D_i}{Y_i}\frac{\partial Y_i}{\partial y} \qquad (2.40)$$

These equations have been obtained under the assumption that the body force and pressure gradient diffusion are neglected.

5. DIMENSIONAL ANALYSIS [4]'

Dimensional Analysis is a mathematical technique which makes use of the study of dimensions as an aid to the solution of several physical problems. Each physical phenomenon can be expressed by an equation composed of variables (or physical quantities) which may be dimensional or non-dimensional quantities. Dimensional analysis helps in determining a systematic arrangement of variables in the physical relationship and combining dimensional variables to form non-dimensional parameters. In the study of fluid mechanics the dimensional analysis has been found to be useful in both analytical and experimental investigations. This technique is useful in

(1) Testing the dimensional homogeneity of any equation of fluid motion.

(2) Deriving the equation expressed in terms of non- dimensional parameters to show the relative importance of each paraiiieter

(3) Planning model terms and presenting experimental results in a systematic manner in terms of non- dimensional parameters.

5.1 JYIETHODS OF DIMENSIONAL ANALYSIS. Following are the two methods of dimensional analysis generally used.

(a) RAYLEIGlI METHOD : In this method, a functional relationship of inore variables is expressed in the form of an exponential equation which must be dimensionally homogeneous. Thus ifI is some function of variables J/, J2, ..., Jo, the functional equation can be written as

$$X = f(X1, X2, \ldots\ldots, Xn), \tag{2.41}$$

whereI is a dependent variable, J7, J2, ..., Jii are independent variables. 'this equation may be expressed as

$$X — t''X1 \quad X2^{b} \quad flirt \tag{2.42}$$

where is a dimensionless constant which may be determined either from the physical characterization of the problem or from the experimental measurements. The exponentials *a, h,* c, ... are then evaluated on the basis that the equation is dimensionally homogeneous. The dimensionless parameters are formed by grouping together the variables with like powers.

(b) BUCKINGAM THEORF.M: The theorem makes use of the following assumptions

(i) It is always possible to select m independent fundamental units in a physical phenomenon. (In dynamics of viscous compressible fluids m = 4, i.e., mass, length, time and temperature).

(ii) 4"liere exist n quantities, say *Q1, Q2, ..., Qn* involved in a physical phenomenon whose diriiensional formulae may be exprcssetl in terms *v to* fundamental units.

(iii) there exists a functional relationship between the n dimensional quantities *Q1, Q2, ... ,* (7ii, say

$$f(Q1, Q2, \ldots\ldots, Qn) = 0, \tag{2.43}$$

and this equation is independent of the types of units chosen and is dimensionally homogeneous.

The II- theorem may now be stated as follows

An equation in physical variables, which is dimensionally homogeneous, can be reduced to a relationship among a complete set of dimensionless products.

5.2 IETHOD OF FINDING OUT - PRODUCTS: The following steps may be adopted in finding out the Al -products in a complete set

(i) Write down the dimensional matrix of n physical quantities, involving in a physical phenomenon, having m independent fundamental units.

(ii) Determine the rank of the dimensional matrix. If the rank is r(say) then the number of

II 's will be (n-r).

(iii) Select r quantities out of the n physical quantities as base quantities, keeping in view that these r quantities should have different dimensions and the dimension of any of the fundamental unit should not be zero in all of them.

(iv) Express H›, 2. . ... , ., as power products of these r quantities raised to arbitrary integer exponents and one of the excluded, but different in different Fl 's, (n-r) quantities.

(v) Equate to zero the total sum of dimensions of each fundamental unit in each Q-product to get the integer exponents.

This chapter forms a basis for the work presented in the thesis. With a short introduction in this chapter, basic concepts in the field of internal ballistics of solid rocket propellants have been discussed. The perfolmance parameters like characteristic velocity, thrust coefficient, and specific impulse have been defined. The combustion process of composite, double base, composite modified double base propellant has been presented in detail. The governing equations of laminar now and turbulent flow have been presented The expressions for heat ti'ansfer in solitl and gas phase have also becti tievclope‹1, These expressitins along with the structure ot' combustion wave tiave been utilised to develop a generalized burn rate model for solid rocket propellant using tl e technique of dimensional analysis.