AEROSPACE PROPULSION SYSTEMS (R18A2115)

COURSE FILE

III B. Tech II Semester

(2021-2022)

Prepared By

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Department of Aeronautical Engineering

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

(Autonomous Institution – UGC, Govt. of India)

Affiliated to JNTU, Hyderabad, Approved by AICTE - Accredited by NBA & NAAC – 'A' Grade - ISO 9001:2015 Certified)

Maisammaguda, Dhulapally (Post Via. Kompally), Secunderabad – 500100, Telangana State, India.

MRCET VISION

- To become a model institution in the fields of Engineering, Technology and Management.
- To have a perfect synchronization of the ideologies of MRCET with challenging demands of International Pioneering Organizations.

MRCET MISSION

To establish a pedestal for the integral innovation, team spirit, originality and competence in the students, expose them to face the global challenges and become pioneers of Indian vision of modern society.

MRCET QUALITY POLICY

- To pursue continual improvement of teaching learning process of Undergraduate and Post Graduate programs in Engineering & Management vigorously.
- To provide state of art infrastructure and expertise to impart the quality education.

PROGRAM OUTCOMES

Engineering Graduates will be able to:

1. **Engineering knowledge**: Apply the knowledge of mathematics, science, engineering fundamentals, and an engineering specialization to the solution of complex engineering problems.

2. **Problem analysis**: Identify, formulate, review research literature, and analyze complex engineering problems reaching substantiated conclusions using first principles of mathematics, natural sciences, and engineering sciences.

3. **Design / development of solutions**: Design solutions for complex engineering problems and design system components or processes that meet the specified needs with appropriate consideration for the public health and safety, and the cultural, societal, and environmental considerations.

4. **Conduct investigations of complex problems**: Use research-based knowledge and research methods including design of experiments, analysis and interpretation of data, and synthesis of the information to provide valid conclusions.

5. **Modern tool usage**: Create, select, and apply appropriate techniques, resources, and modern engineering and IT tools including prediction and modeling to complex engineering activities with an understanding of the limitations.

6. **The engineer and society**: Apply reasoning informed by the contextual knowledge to assess societal, health, safety, legal and cultural issues and the consequent responsibilities relevant to the professional engineering practice.

7. **Environment and sustainability**: Understand the impact of the professional engineering solutions in societal and environmental contexts, and demonstrate the knowledge of, and need for sustainable development.

8. **Ethics**: Apply ethical principles and commit to professional ethics and responsibilities and norms of the engineering practice.

9. **Individual and team work**: Function effectively as an individual, and as a member or leader in diverse teams, and in multidisciplinary settings.

10. **Communication**: Communicate effectively on complex engineering activities with the engineering community and with society at large, such as, being able to comprehend and write effective reports and design documentation, make effective presentations, and give and receive clear instructions.

11. **Project management and finance**: Demonstrate knowledge and understanding of the engineering and management principles and apply these to one's own work, as a member and leader in a team, to manage projects and in multi-disciplinary environments.

12. **Life-long learning**: Recognize the need for, and have the preparation and ability to engage in independent and life-long learning in the broadest context of technological change.

DEPARTMENT OF AERONAUTICAL ENGINEERING

VISION

Department of Aeronautical Engineering aims to be indispensable source in Aeronautical Engineering which has a zeal to provide the value driven platform for the students to acquire knowledge and empower themselves to shoulder higher responsibility in building a strong nation.

MISSION

The primary mission of the department is to promote engineering education and research. To strive consistently to provide quality education, keeping in pace with time and technology. Department passions to integrate the intellectual, spiritual, ethical and social development of the students for shaping them into dynamic engineers.

QUALITY POLICY STATEMENT

Impart up-to-date knowledge to the students in Aeronautical area to make them quality engineers. Make the students experience the applications on quality equipment and tools. Provide systems, resources and training opportunities to achieve continuous improvement. Maintain global standards in education, training and services.

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

III Year B. Tech, ANE-II Sem

	L	T/P/D	C
(R18A2115) Aerospace Propulsion Systems	**4**	**1/-/-**	**4**

Objective:

- Students acquire knowledge about the present space equipment.
- Students can focus on various launch systems available in aerospace industry and also understand the future scenario.
- To provide an exposure with testing and design limitations.

UNIT-I
HYPERSONIC AIR-BREATHING PROPULSION

Ramjets at high speeds and limitations of turbojets. Need for supersonic combustion, Implications criticality of efficient diffusion and acceleration, problems of combustion in high speed flow, scramjet engine- construction, flow process- description, spill-over drag, plume drag. Isolator, combustor, thermal protection, thermal throat, scheduled &distributed fuel injection. Types of nozzles and nozzle flow. Scramjet performance- numericals.

Combined cycle engines- turbo-ramjet, Air turbo-rocket (ATR), ejector ramjet, Liquid-air collection engine (LACE) - need, principle, construction, operation, performance.

UNIT-II
CHEMICAL ROCKET PROPULSION

Classification of rocket engine, chemical rocket engine types, working principle, schematic diagram, applications, types, advantages and disadvantages- solid, liquid and hybrid propellant rocket engine, TVC.

Solid propellant rocket motors, principle, applications, Solid propellant types, composition, properties, Propellant grain, properties, structural design, insulators and inhibitors- functions, requirements, Rocket motor casing- materials. Igniters, types, construction, Liquid propellants- types, composition, properties, performance, Propellant, feed systems- pressurisation, injectors, starting and ignition, cryogenic engines, Engine cooling.

UNIT-III NUCLEAR ROCKET PROPULSION: Nuclear propulsion history, Power, thrust, energy. Nuclear fission- basics, sustainable chain reaction, neutron leakage, control, reflection, prompt and delayed neutrons, thermal stability. Principles and fuel elements. The nuclear thermal rocket engine, start-up and shutdown. Development status of nuclear engines, alternative reactor types, safety issues in nuclear propelled missions.

UNIT-IV ADVANCED ROCKET PROPULSION

ELECTRICAL: Limitations of chemical rocket engines. Electric propulsion systems- structure, types, generation of thrust. Electrostatic thrusters, electro-magnetic thrusters, applications to space missions, pulsed plasma thrusters (PPT) for micro-spacecraft, solar electric propulsion.

ADVANCED SYSTEMS: Micro-propulsion, application of MEMS, chemical, electric micro-thrusters, principle, description, Propellantless propulsion, teethers, momentum exchange, Photon rocket, beamed energy propulsion, solar, magnetic sails.

UNIT V LAUNCH VEHICLES

Role and military functions of space launch vehicle, Types, missions, mission profile, staging employed in the vehicle, guidance and control requirements. Some successful launch vehicles, Description of space shuttle engine, Propellant slosh - Propellant hammer, geysering effect in cryogenic rocket engines, SSTO.

Text Books:

1. Cornelisse, J. W., Schoyer H.F.R. and Wakker, K.F., Rocket propulsion and space flight Dynamics, Pitman, 1979.
2. Turner, M.J.L., Rocket and Spacecraft Propulsion, Springer, 2001.

Outcome:

- Students can correlate with the different launch vehicles and missiles available.
- Students will be able to configure the launch vehicle or missile required for specific purpose.
- Students will be able to design the conceptual requirements.

Unit 1

HYPERSONIC AIR-BREATHING PROPULSION

Ramjets at high speeds and limitations of turbojets. Need for supersonic combustion, Implications criticality of efficient diffusion and acceleration, problems of combustion in high speed flow, scramjet engine- construction, flow process- description, spill-over drag, plume drag. Isolator, combustor, thermal protection, thermal throat, scheduled &distributed fuel injection. Types of nozzles and nozzle flow. Scramjet performance- numericals.

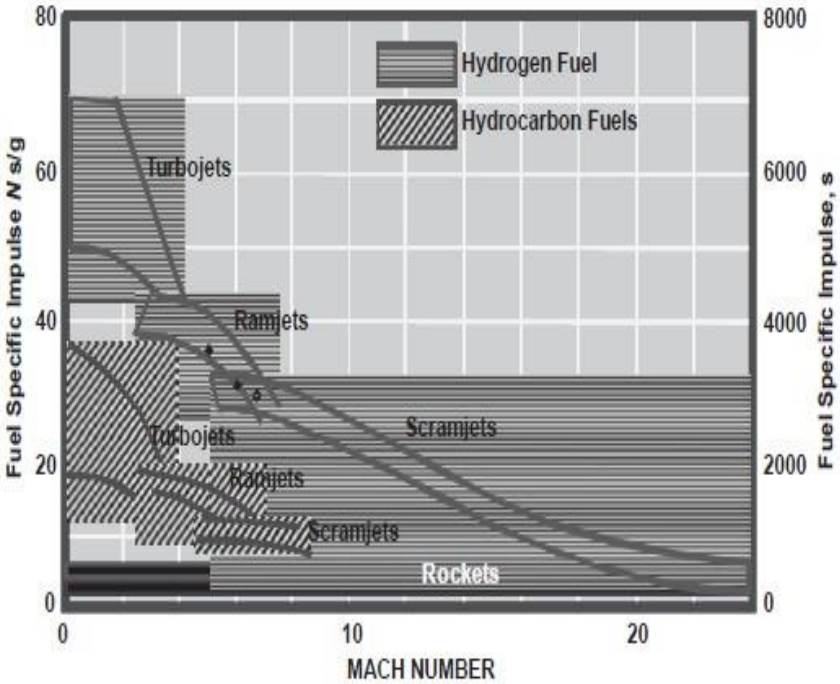
Combined cycle engines- turbo-ramjet, Air turbo-rocket (ATR), ejector ramjet, Liquid-air collection engine (LACE) - need, principle, construction, operation, performance.

Limitations of Turbojets/Turbofans and Ramjets at High Speeds:

In order to compare performance of various air-breathing propulsion systems, we define a parameter, specific impulse, as follows:

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

Performance based differences between the different engine cycles are clearly illustrated in the fuel specific impulse, I_{sp}vs Mach number diagram shown.



Around Mach 2.25, stagnation pressure captured by the intake is around 11.2,

$\frac{P_0}{P}$ = **[** 1+ $\frac{\gamma-1}{2}M^2$ **]** to the power $\frac{\gamma}{\gamma-1}$ = 11.2

Assuming intake efficiency of around 60-70%, this ram pressure capture by the intake works out to 7.0. So, beyond 2.25 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 ($^{T}/_{W}$).

Specific Thrust T/W comparision at 2.5 M
Ramjet – 160-170 N/Kg
Turbojet – 40-70 N/Kg
Turbofan – 30-60 N/Kg

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).

Ramjets are used in military missiles like Akash, Brahmos missiles.

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, (i.e) 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 at supersonic speeds.

Also, the oxygen content in the air reduces with increase in temperature.

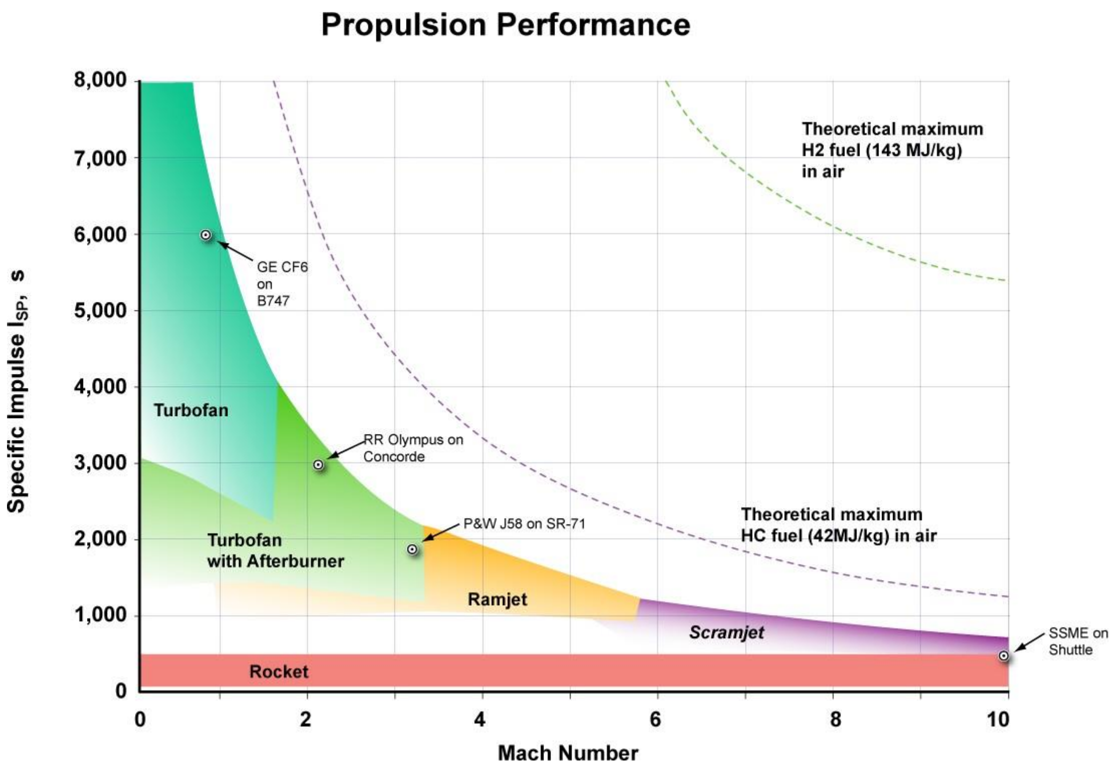
Therefore, all hypersonic transport propulsion systems need supersonic combustion ram jets (Scramjets)

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.

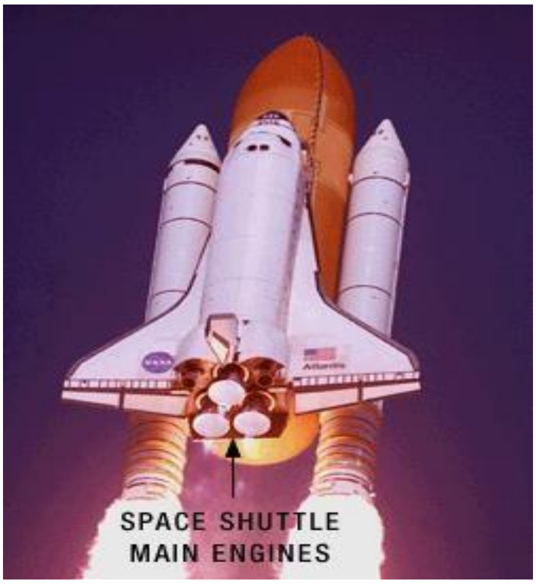
The low specific impulse of rockets, in comparison with the other propulsion systems clearly eliminates the rocket from consideration for long range cruise but as the Mach number continues to increase in the hypersonic regime the scramjet specific impulse approaches that of the rocket engine.

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.



The performance limits of air breathing systems are shown below:

Space Shuttle Main Engine (SSME):



Air-Breathing Engines-Operating Speeds

1. Around Mach 3 flight regime, Ramjet becomes more efficient as a propulsive system in comparison with the turbine based engines (turbojets of turbofans).
2. Ramjet is preferred in the speed range of 2-4 Mach, due to its higher specific thrust($^{T}/_{W}$).
3. Beyond Mach 5, specific impulse of ramjet decays rapidly and the Scramjet delivers a higher specific impulse at higher speeds.
4. The rocket's specific impulse is considerably lower than the other propulsion systems but it offers operation capabilities from sea-level static to beyond the atmosphere which no other propulsion system mentioned here can do.
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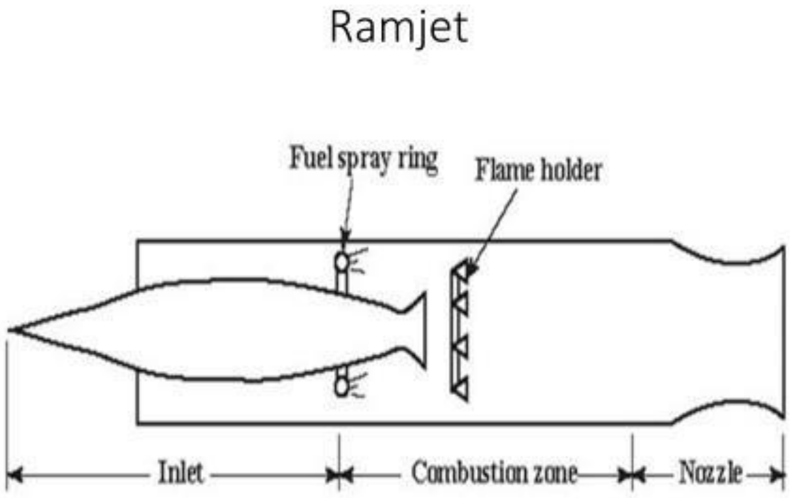
Ramjets:

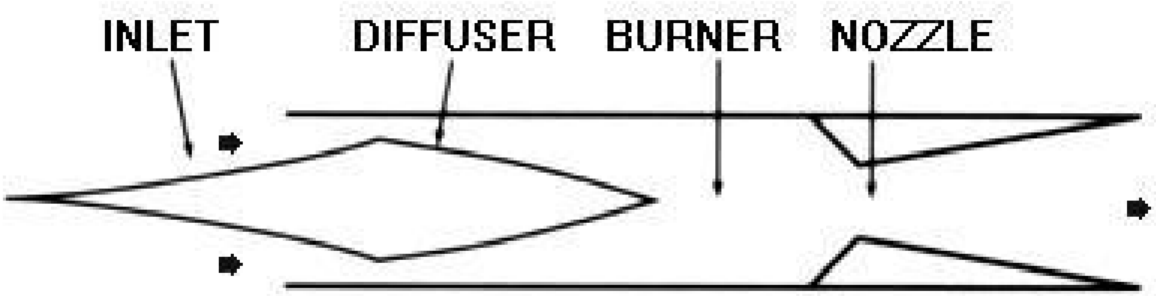


Operating Principle:

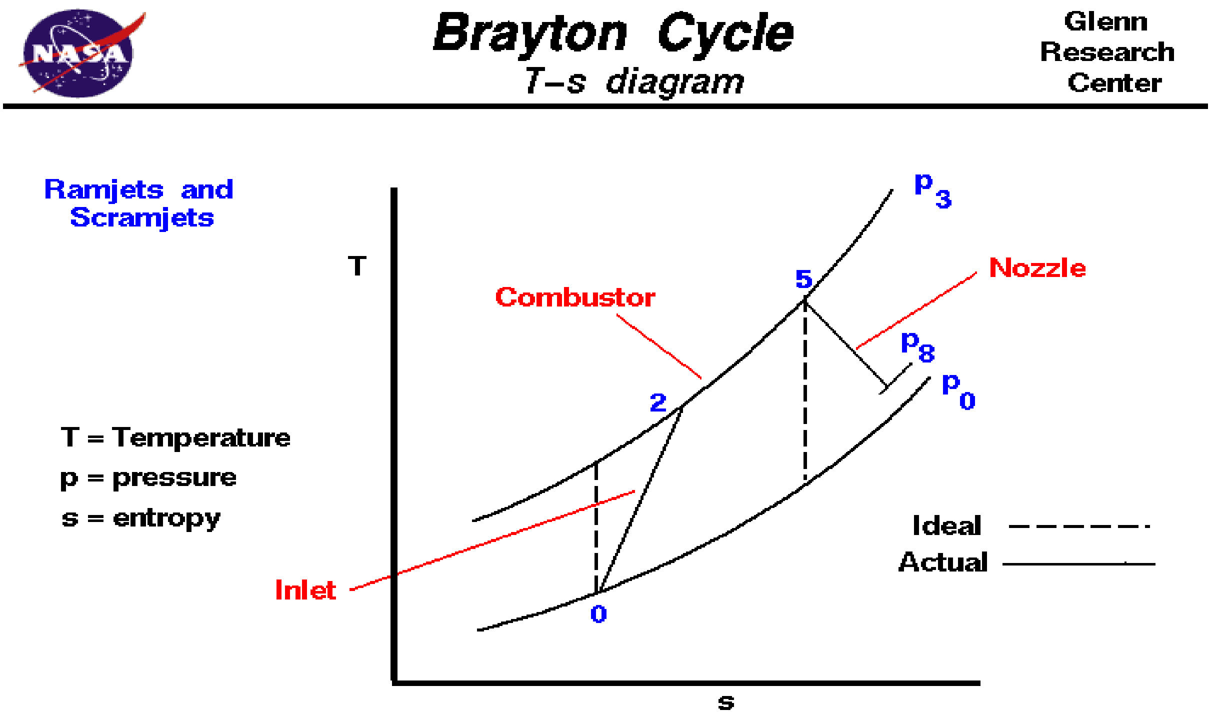
The incoming supersonic airflow is slowed to subsonic speeds by multiple shock waves, created by back-pressuring the engine.

This subsonic flow undergoes combustion and accelerates through a narrow throat, to supersonic speeds.

Ramjet-Schematic:



Brayton Cycle- Ramjet:



Operation-Ramjet:

1. Ramjet Engine consists of **supersonic diffuser, subsonic diffuser, combustion chamber and nozzle section.**
2. Air from atmosphere enters the supersonic diffuser at a very high speed. Diffusion in the inlet occurs due the geometric shaping and shock waves
3. The air velocity gets reduced in the supersonic diffuser through normal and oblique shock waves.
4. Velocity is further reduced in the subsonic diffuser.
5. The diffuser converts the kinetic energy of the entering air into **static pressure and temperature rise** which is achieved by ram effect.
6. Fuel is injected into the combustor through suitable injectors causing mixing of fuel with the air and the mixture is burnt.
7. Combustion gases attain a temperature of around 1500-2000 k by continuous combustion of fuel air mixture.
8. Gases pass through the tail pipe and nozzle, which expands the gases completely.
9. The gases leave the engine with a speed much higher than the air entering the engine

Advantages of Ramjet Aircraft:

1. Ramjet is a simple machine and does not have any moving parts.
2. Since turbine is not used, maximum temperature allowed is very high, around 2000 C, as compared to around 900 C in turbojets.
3. We can burn air/fuel ratios of 13:1which gives greater thrust.
4. Specific fuel consumption is much better than other gas turbine engines, at high speeds and altitudes.
5. Wide range of fuels can be used.
6. It is very cheap to produce; adoptable for mass production

Disadvantages of Ramjet Aircraft:

1. It is not possible to start a ramjet engine without an external launching device
2. The engine heavily relies on the diffuser and it is very difficult to design a diffuser which gives good pressure recovery over a wide range of speeds
3. Due to high air speed, the combustion chamber requires flame holders to stabilize the combustion
4. At very high temperatures of about 2000 C, dissociation of combustion products take place, reducing the efficiency of the plant
5. High fuel consumption at low speeds

Hypersonic Aircraft:

- 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. (SSTO Concept)
- The hypersonic flight regime is commonly considered to begin when velocities exceed Mach 6
- Defense 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

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.

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

A Hypersonic Vehicle is a vehicle that travels at least 4 times faster than the speed-of-sound, or greater than Mach 4. A hypersonic vehicle can be an airplane, missile, or spacecraft. Some hypersonic vehicles have a special type of jet engine called a Supersonic Combustion Ramjet (scramjet) to fly through the atmosphere. Sometimes, a hypersonic plane uses a rocket engine. A Re-entry Vehicle is another type of Hypersonic Vehicle. A re-entry vehicle is a spacecraft that travels through space and re-enters the atmosphere of a planet.

Types of Hypersonic Vehicles:

- Turbojet or turbo-ramjet; Wraparound Turbo-ram jet engine
- Ram-Scram jet combination
- Turbo-Rocket
- Liquid-air collection (LACE) engine.
- Air turbo-rocket (ATR)
- Ejector Ramjet

Some examples of hypersonic vehicles already flown successfully are:

- German V-2 rocket
- Intercontinental Ballistic Missiles (ICBM), flying at around Mach 25
- X-15 hypersonic test vehicle

These vehicles are Propulsion vehicles, based on air breathing/rocket engine concepts, capable of reaching speeds of presently under development in at least five countries.   In US, the vehicle is designated as aerospace plane.

ISRO has been conducting experiments for developing Scramjet and has been successful in conducting supersonic combustion for very short periods.

Physical Aerodynamic Aspects of hypersonic vehicles:

1. **Thin Shock Layers**: The oblique shock wave formed at the vehicle body is very thin and makes a much smaller angle (around 25°º). The shock waves also lie close to the body. This leads to merging of shock waves with the boundary layer, which needs to be considered while predicting the pressure distribution over the body.
2. **Entropy Layer:** The shock 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.

Problems of Combustion in High Speed Flow: **S**upersonic combustion poses following problems

1. **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.
2. **Slow Reaction rate (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)
3. **Low residence time (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.
4. **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.

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

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

6. **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
7. **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.

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 favour 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.

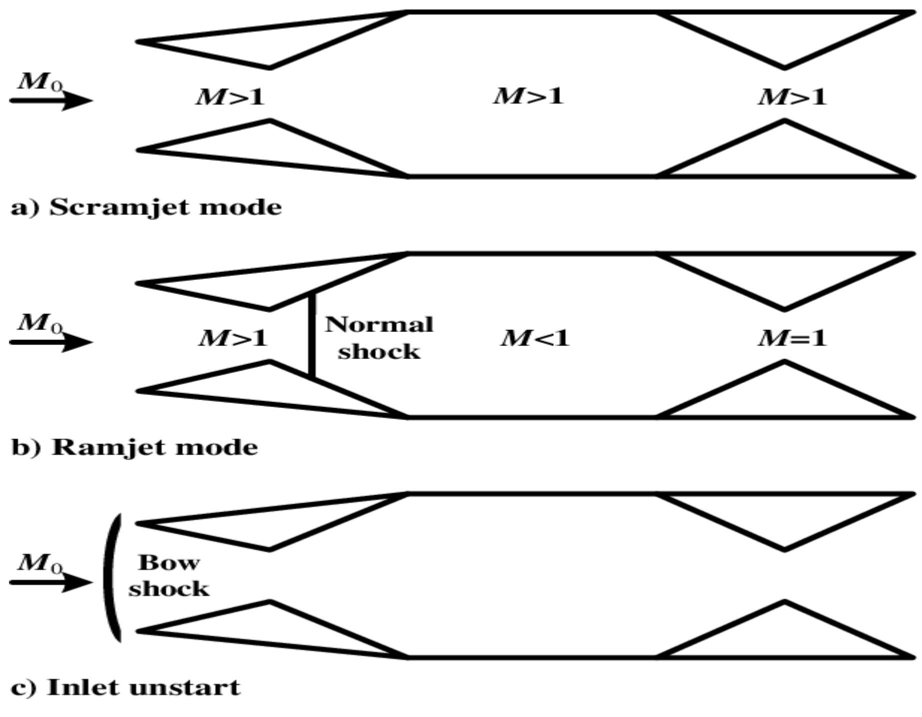
Design of scramjet combustor must take into account the requirement that the fuel be well mixed with the air within a few microseconds. The criticality of timing must be such that the ignition delay time plus the time to complete the reaction are less than the **residence time** of flow through the combustor. 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".

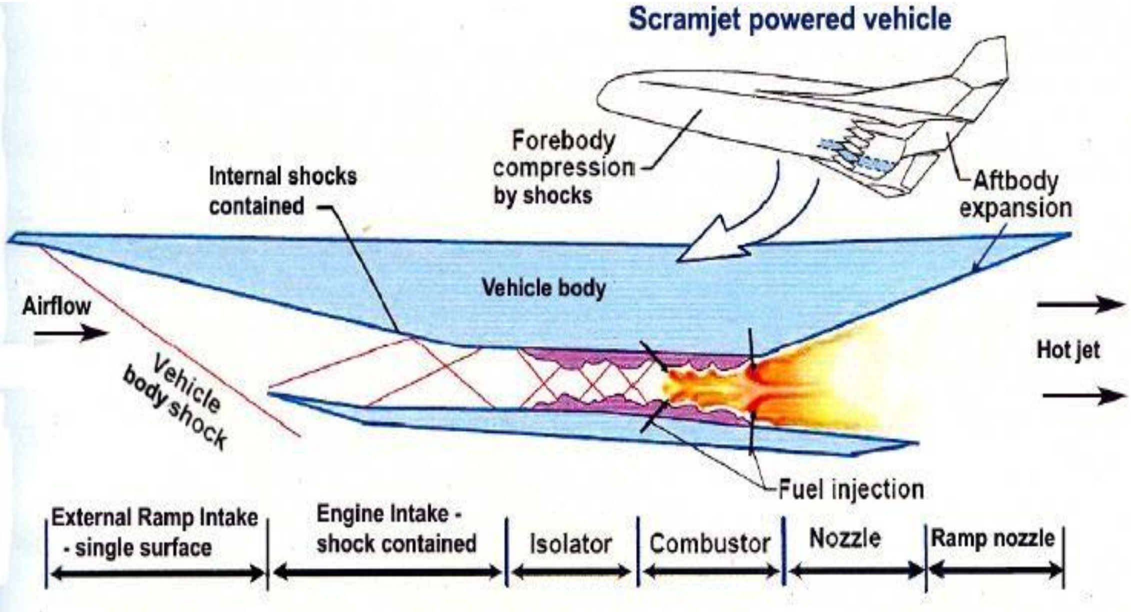
Non-equilibrium effects in diffusion flames: The local heat release leads to enhanced local temperatures. Similarly, there would be non-uniform temperature distribution since the fuel sprays are introduced over parts of the cross section. This leads to non-uniformity in other quantities as well. The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

Flow Conditions-Ramjet & Scramjet:



Scramjet Engine- Construction: Scramjet engine is characterized by slow reaction times and high flow speeds i.e 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
- 

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.

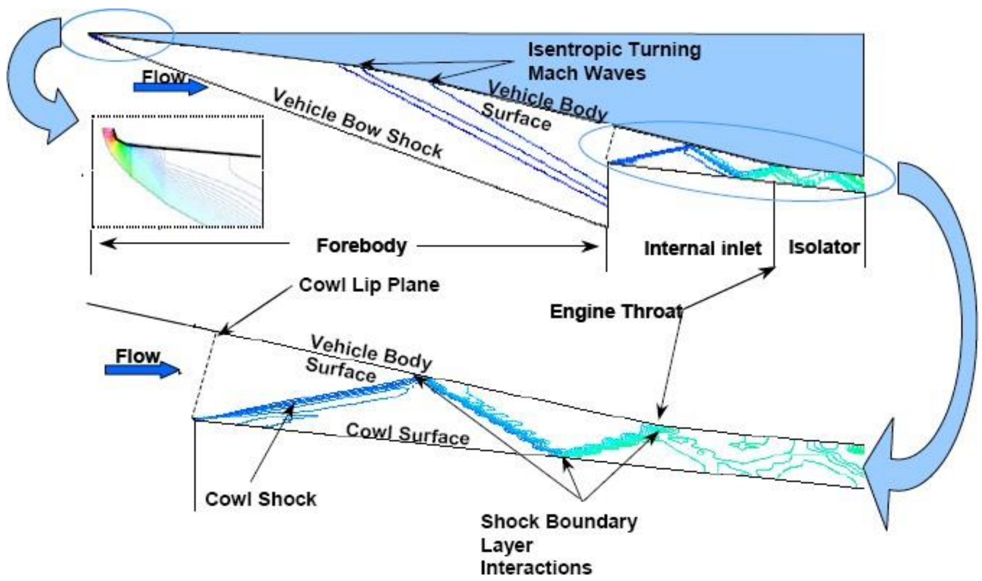
Two methods in design of air induction system are the positioning of oblique shocks avoiding interference with each other and providing blow holes to remove laminar layer turning it turbulent.

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.



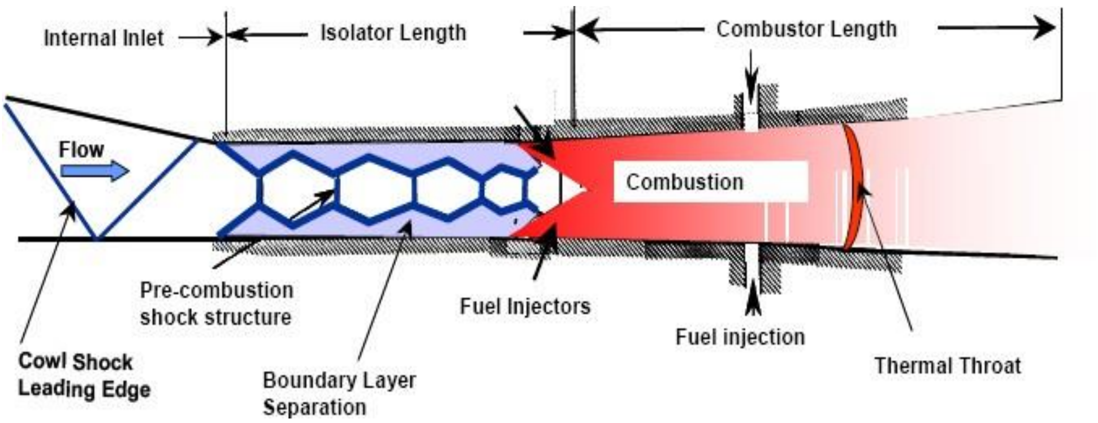
Isolator: Isolator is constant area diffuser containing the internal shock structure, swallowed during supercritical operation of the inlet (or during operation after the inlet "started"). The isolatoris inserted before thecombustor to diffuse the flow further, through a shock train, producing desired 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



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 opentype, 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 Nozzlephysical 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.

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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 back pressure 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.
- 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 p_3 at burner entry. This back pressure causes a normal shock train to form in the isolator. As long as the back pressure does not exceed isolator's ability to maintain the normal shock train, the isolator will perform as a variable area diffuser to enable subsonic flow in the burner.

- The function of the second ramjet throat, to choke the flow and accelerate the subsonic flow to supersonic speeds in the nozzle is provided by the means of a "choked thermal throat". The thermal throat is brought about by choosing the right combination of area distribution and fuel-air mixing/combustion.
- The heating of the gas by combustion causes the speed of sound in the gas to increase (through increase of $\sqrt{t}$and hence cause Mach number to decrease) even though the gas is still travelling at the same speed.
- Forcing the speed of air flow in the combustion chamber under Mach 1 in this way is called "thermal choking".
- A thermal throat results when the flow is slowed through tailored heat for causing dual-mode operation.
- Local heat release leads to enhanced temperatures. This increase causes increased acoustic velocity (~ $\sqrt{}$ T) and reduction in Mach number even if the local speed is unaltered.

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

Distributed 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 injection is 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 favour 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 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:

1. Air-fuel inter-action occurs over entire length of inlet-isolator-combustor resulting in better mixing.
2. Complete combustion in shorter isolator/combustor lengths, thereby reducing engine weight and cooling loads.
3. We can use combination of liquid and gaseous fuels through different sets of injectors
4. 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 afterbody, 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 freestream 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 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.

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 around 2010, 2015, 2025, respectively

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:

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.

In the same process, the total pressure of the secondary flow is raised to a value between that of ambient and primary flow. The ejector is crudely analogous to a mixed exhaust flow bypass turbofan engine, although energy transfer efficiency is low. Because it is accomplished by viscous shear forces rather than rotating turbo-machinery.

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.

Since the ramjets produce little or no thrust during take-off, a rocket engine or its equivalent must be part of the vehicle. This device could either operate independently or act as the primary of an ejector for which an existing ramjet or scramjet passage could serve as a shroud. In the latter case, the pressurized flow leaving the ejector can be decelerated, mixed with fuel, and burned in a combustor. The combustion products are then accelerated through a nozzle to produce thrust.

The net effect of the ejector is to supply the burner with a flow of pressurized air that would be roughly equivalent to the ram conditions of a much higher forward speed. Thus, the ejector–burner-nozzle combination is referred to as the “ejector ramjet”.

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:

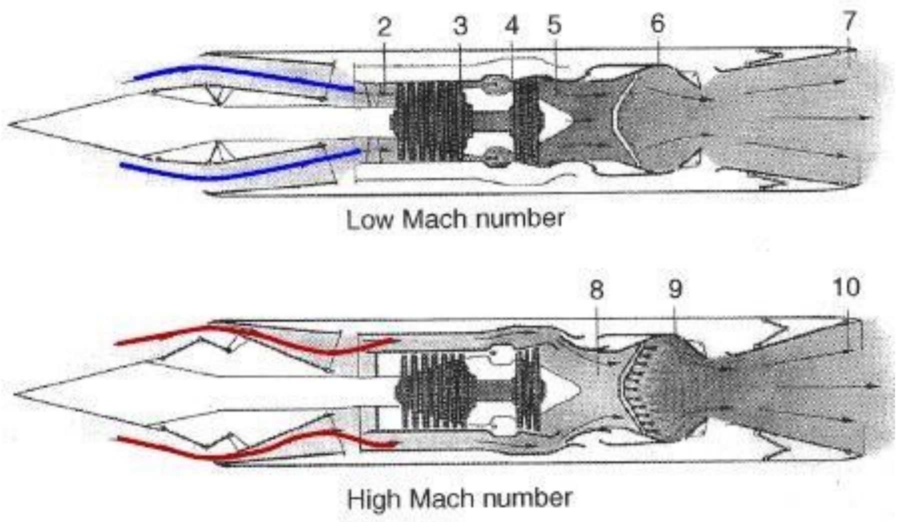
- The ejector portion of the device will have constant area and fixed geometry.
- The inlet primary flow will be supersonic and inlet secondary flow is subsonic. The ejector exit plane is sonic and choked.
- The ejector ramjet has a potential to increase the thrust above the primary flow alone, with 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.



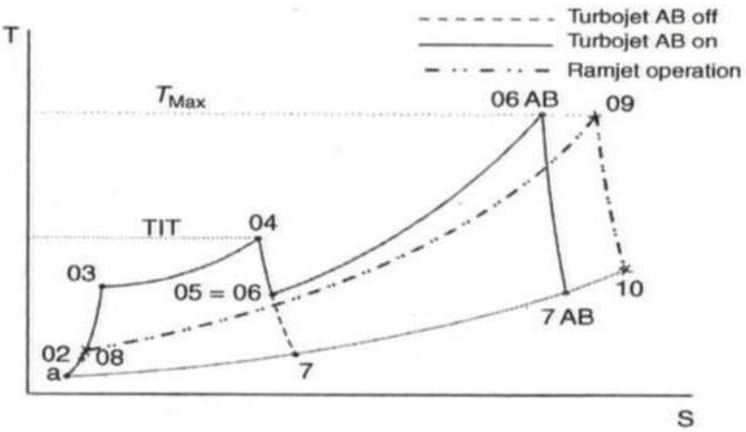
Only Turbojet

Only Ramjet

Wraparound Turboramjets

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.



Turboramjet T-s Diagram

The Air 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.

Air 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 below.

Liquid Air Cycle Engine (LACE):

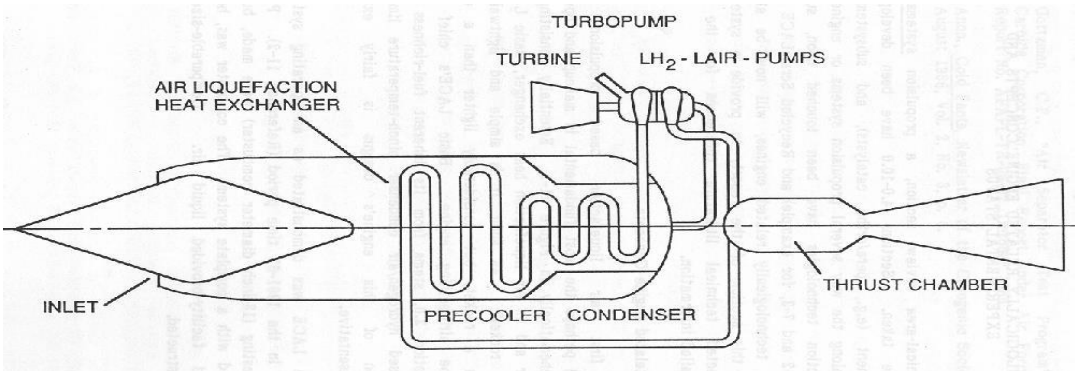
Liquid Air Cycle Engine is a separate class of hypersonic air breathing engine made possible by the availability of very low temperature, cryogenic liquid hydrogen fuel. The liquid hydrogen fuel has high specific energy release (heat of combustion per unit mass), good vehicle cooling capacity and also 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 is 78.9 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:



compressed

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.

UNIT-II

CHEMICAL ROCKET PROPULSION

Classification of rocket engine, chemical rocket engine types, working principle, schematic diagram, applications, types, advantages and disadvantages- solid, liquid and hybrid propellant rocket engine, TVC.

Solid propellant rocket motors, principle, applications, Solid propellant types, composition, properties, Propellant grain, properties, structural design, insulators and inhibitors- functions, requirements, Rocket motor casing- materials. Igniters, types, construction, Liquid propellants- types, composition, properties, performance, Propellant, feed systems- pressurisation, injectors, starting and ignition, cryogenic engines, Engine cooling.

Classification of Rocket Engine Propulsion Systems:

1. All classical propulsive systems create thrust based on conservation of momentum. majority of systems expel mass.
2. Rocket Engines can be classified based on how they are accelerated
3. The propulsive energy comes from the onboard propellant itself **(internal energy**) as in chemical reaction, or they can be accelerated using **external energy source**.
4. The performance of the propulsive system depends on total mass of the spacecraft and on the speed of the propellant.

Classification of Propulsion System:

- **Source of Energy** (Internal, External)
- **Type of Energy Source or Type of propellant used**(Chemical, Nuclear, Solar, Electrical 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)

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 Propulsion Systems
- 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 (2500 to 4100° 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 are 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's atmosphere.

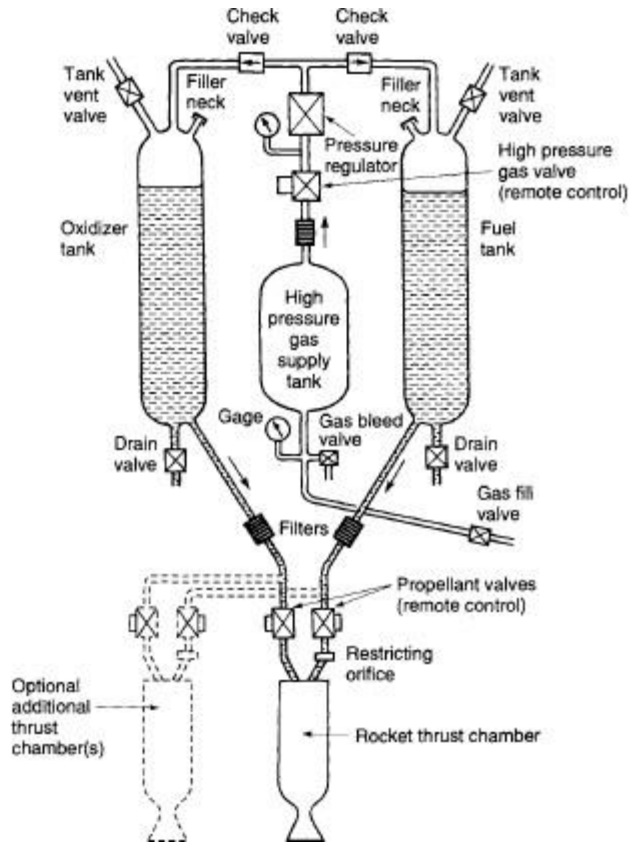
Working Principle:

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

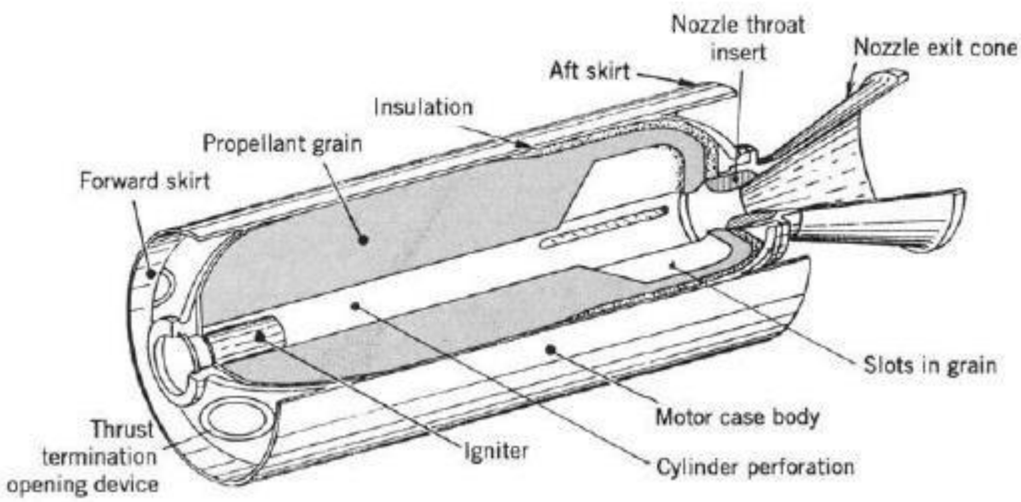
A high pressure gas pressure tank provides pressure feed of oxidizer and fuel through diaphragms. Alternatively, separate pumps may be used to provide pressure feeding of propellants.
The propellants react in the thrust chamber and generate hot combustion gases which are expanded in the supersonic convergent 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 pumps etc.

A schematic diagram is as follows:



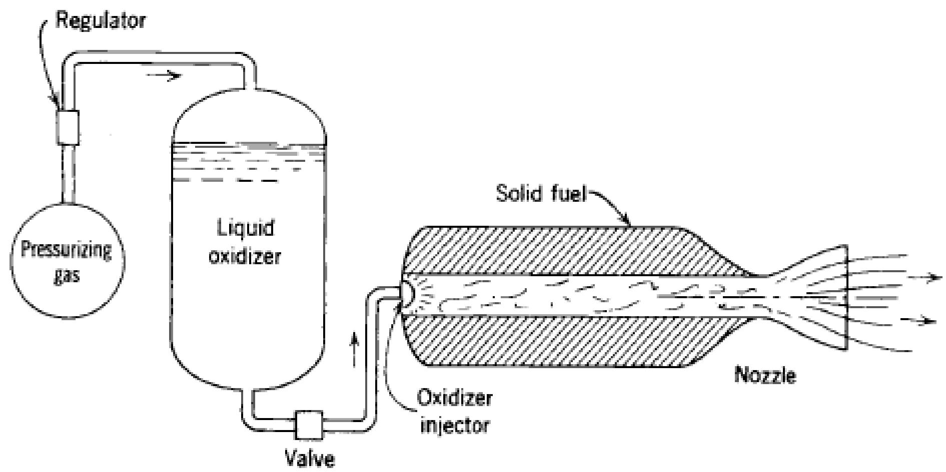
Solid Propellant Rocket Propulsion System: A schematic diagram is shown below:



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:



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.

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

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 with in 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. **Producebility**: Easy to manufacture, inspect and assemble
12. **Schedule**: The propulsion system should be capable of completing the mission in given time frame.

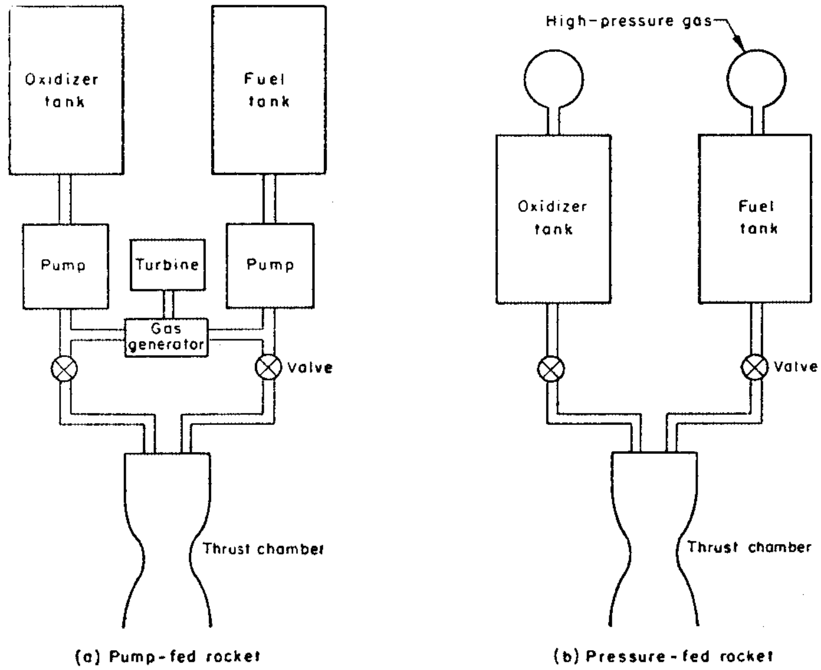
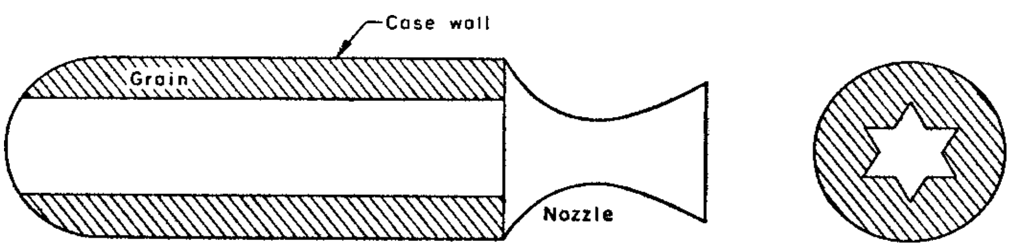
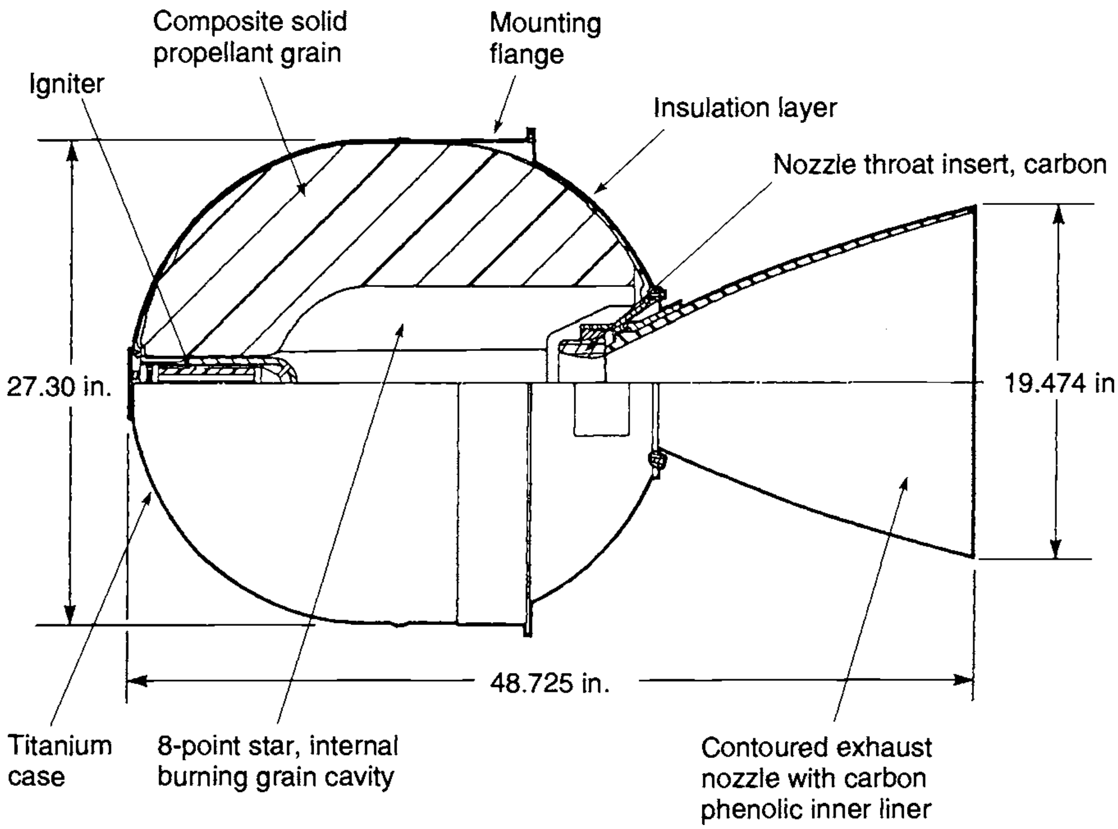


FIG. 2. Schematic of liquid-propellant rocket

Solid Propellant Rocket Motor:





Operation: In solid propellant rocket motors (SPR), the motor is commonly used to mean the "engine" for liquid propellant rockets (LPR). Both LPRs and LPRs are extensively used in rocket applications.

The propellant is stored and contained directly in the motor, which also acts as combustion chamber. Solid Propellant Rocket Motors can deliver varying thrust from 2 N to over 4 X 10^6N.

SPR has no moving parts, in comparison with LPRs and are easy to manufacture and maintain. However, they cannot be fully checked before use and thrust also cannot be varied randomly during flight.

The grain is the solid body of hardened propellant that is about 90% of the total motor mass. The igniter provides the energy to start combustion process and is electrically activated. Once ignited, the grain starts burning on its exposed surfaces. Many grains have internal geometric features to control the burning process which determines the mass flow and thrust.

The inner surfaces of the case have an insulation layer to keep the case from becoming too hot. The case is made of steel/aluminum or titanium.

The nozzle accelerates the hot gases generated by burning the propellant, producing the necessary thrust. Majority of SPRs have a simple fixed CD nozzle. Some nozzles have provision to rotate slightly so as alter the thrust axis for steering the vehicle in desired direction.

Nozzle is made of high temperature materials, usually graphite or ablative material to absorb the heat.
The SPR is attached to the vehicle with a flange and skirt provided on the SPR.

Almost all SPRs are used only once. The hardware that remains after the propellant is burnt, the motor case, nozzle case is discarded. In rare cases, the hardware like nozzle case is recovered and reused after refurbishing.

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 is 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.

Performance:

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

   $$\boldsymbol{I_t = \int_0^t F\,dt}$$

   For constant thrust, this reduces to $\boldsymbol{I_t}$ = 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

   $$I_S = {}^{I_t}/_{w} = {}^{F}/_{\dot{w}}$$

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

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

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).

Mass ratio MR = 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.

$$\zeta = m_p/m_0$$

$$m_0 = m_f + m_p$$

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.

$$\text{Impulse-to-weight ratio} = \frac{I_t}{w_0} = \frac{I_t}{(m_f+m_p)g_0} = \frac{I_s}{\{m_f/m_p\}+1}$$

SPR Performance:

Burning Rate: The burning surface of a propellant grain recedes in a direction perpendicular to the surface. The rate of regression, usually expressed in cm/sec or in/sec, is called the burning rate.
The burning rate is a function of propellant composition and motor operating conditions.

For composite propellants, burning rate can be increased by

- Adding a burning rate catalyst called "burning rate modifier"
- Decreasing the oxidizer particle size
- Increasing oxidizer percentage
- Increasing the heat of combustion of the binder or plasticizer
- Imbed wires or metal staples in the propellant

Apart from propellant composition, burning rate can also be increased by

- Increasing the combustion chamber pressure
- Initial temperature of the propellant prior to start of burning
- Combustion gas temperature
- Velocity of gasses flowing parallel to the burning surface
- Motor motion (acceleration or spin induced grain stress)

The burning rate r governs the mass flow rate of hot gasses generated $\dot{m}$ as given below:

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

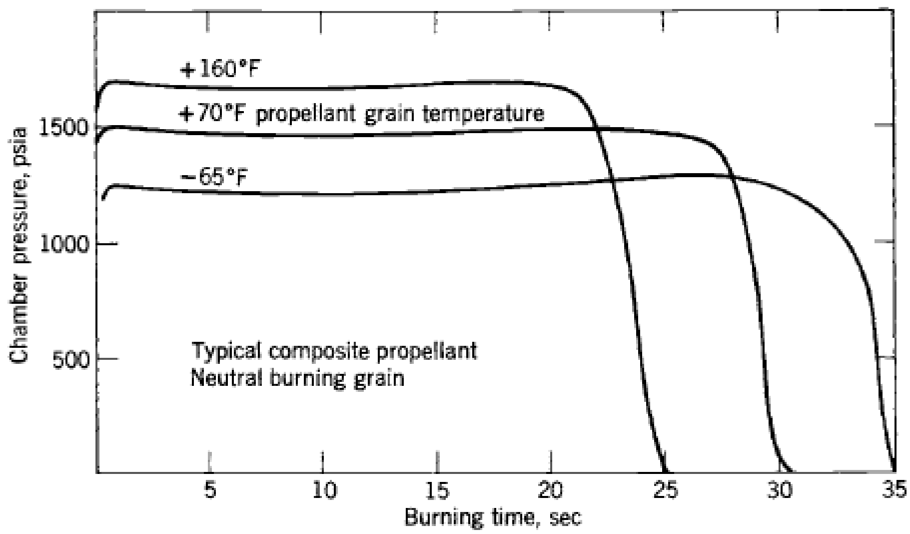
Where A_b is burning area of the propellant grain, r is the burning rate, ρ_b is the solid propellant density and $\dot{m}$ is the mass flow rate of hot combustion gasses.
A_b and r vary with time and chamber pressure.

The relation between burning rate and chamber pressure is

$r = a p_1^n$, where p_1 is the chamber pressure in MPa or psia, a is constant dependent on grain temperture r is the burning rate, usually in cm/sec or in/sec. "n" is burning rate exponent, called combustion index, describing the influence of chamber pressure on burning rate. For stable operation, n has values greater than 0, but less than 1. High values of n give rapid change in burning rate with pressure. Most propellants have burning rate exponent between 0.2 and 0.6.

Burning Rate relation with Temperature: Temperature affects the chemical reactions. The initial ambiant (grain) temperature of a propellant grain prior to combustion influences the burning rate.
For air launched missile motors, the extremes are usually 219 K (--65^0 F) and 344 K (160^0 F).
In large rocket motors, an uneven heating of grain (for example, Sun heating one side of the grain) can cause sufficiently large difference in burning rates, thereby resulting in thrust mis-alignment.

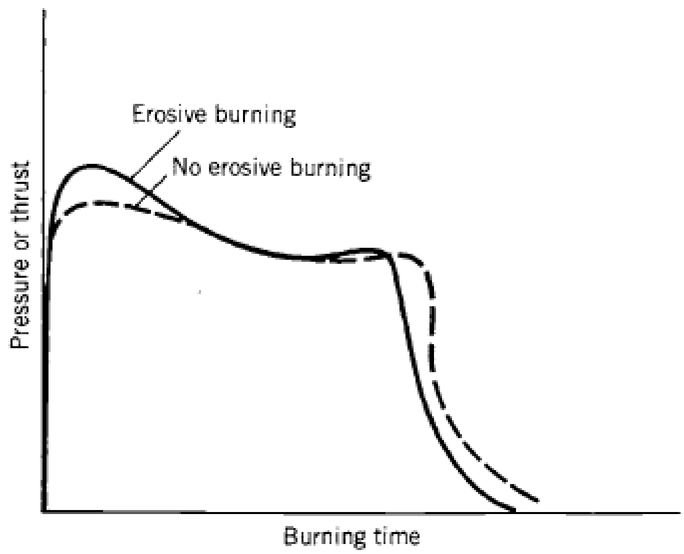
The burning trace of a particular rocket propellant is as given below:



Burning enhancement with Erosion: Erosive burning refers to the increase in the propellant burning rate caused by the high-velocity flow of combustion gasses over the burning propellant surface. It can seriolsy effect the performance of the rocket motor. It occurs primarily in the port passages or perforations of the grain as combustion gases flow towards the nozzle.

The hgh velocity near the burning surface and the turbulant mixing in the boundary layer increases the heat transfer to the solid propellant and thus increases the burning rate.

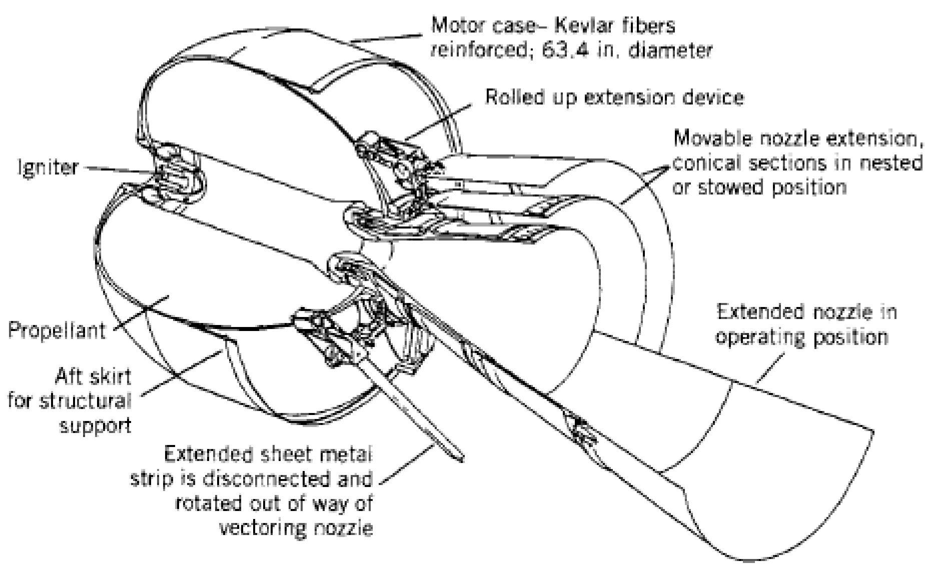
Erosive burning increases the mass flow and thus increases the chamber pressure and the thrust force. This is shown below in the pressure vs time burning trace below:



Important definitions and terminology of solid propellants are as follows:

1. **Configuration**: The shape or geometry of the initial burning surfaces of a grain as it is intended to operate in a motor.
2. **Cylindrical Grain**: A grain in which the internal cross-section is constant along the axis regardless of perforation shape. (refer to diagram below)
3. **Perforation**: The central cavity port or flow passage of a propellant grain. The cross section of perforation may be cylindrical or star-shape, etc.
4. **Sliver**: Unburnt propellant remaining in the motor at the time of burnout. (refer diagram below)
5. **Neutral burning**: Motor burn time during which the thrust pressure and the burning surface area remain approxiamately constant. Many grains are neutral burning types.
6. **Progressive Burning**: Burn time during which, the thrust, pressure and burning surfce area, increase.
7. **Regressive Burning**: Burn time during which the thrust, pressure and burn surface area decrease.

The Inertial Upper Stage (IUS) rocket motor with Extendable Exit Cone (EEC) nozzle (shown below): This motor is used in propelling the upper launch vehicle stages of spacecraft. The grain is internal tube perforation. When lauched, with lower vehicle stages are operating, the two conical movable nozzlesare stowed around the inner conical nozzle. The movable nozzle segments are moved in to their operating position by light weight electrical motors. The nozzle area ratio improves from 49 to 181.



Propellant Grain & Grain Configuration: Th grain is shaped mass of solid propellant inside the rocket motor. The propellant material and the geometrical configuration of the grain determine the motor performance.

The propellant grain is a cast, molded or extruded body and its appearance and texture is similar to hard rubber or plastic. Once ignited, it will burn on all exposed surfaces to form hot gases that are then exhausted through the nozzle.

Methods of holding the Grain in the Case:

- **Cartridge-loaded or Free-standing Grain**: The grain is manufactured seperately from the case and then loaded in to or assembled in to the case.Freestanding grains can easily replaced in case of aging. These are used in small tactical missiles and cost less.

- **Case-bonded Grains**: The case is used as a mold and the propellant is cast directly in the case. The grain is bonded to the case insulation. Case bonded grains give better performance, but are more expensive to manufacture. Case bonded grains are used in almost all large sized motors.

Inhibitors:A layor or coating of slow or non-burning material applied to a part of grain's propellant surface to prevent burning on that surface. The inhibitors are glued, painted, dipped or sprayed on to he surface. Usually, polymetric rubber type with filler materials is used as inhibitor. By preventing burning on inhibited surfaces, the burning area can be controlled and reduced. Inhibitors are also called “Restrictors”.

Liner: A sticky non self-burning thin layer of polymer layer that is applied to the interior of the casing prior to casting the propellant in order to promote good bonding between the propellant grain and casing/insulator.

Internal Insulator: An internal layer between the casing and the propellant grain made of an adhesive, thermally insulating material that will not readily burn. The purpose of the insulator is to limit the heat transfer to and the temperature rise of the casing during the rocket operation.

Solid Propellant Types:

Three types of solid propellants are in use:

- Double-Base
- Composite
- Composite modified double-base

Double-Base (DB) Propellants: Consists of nitrocellulose and nitroglycerine plus additives in small quantity. It is a homogeneous mixture of two explosives. DB solid propellants contain solid ingredient nitrocellulose NC, which absorbs liquid nitroglycerine NG plus minor percentage of additives. Both ingredients are explosives and function as a combined fuel and oxidizer.

DB propellants are made as extrusions or cast and are extensively used in small rocket/missiles or as lower stages/boosters in large rocket launchers.
Additives like crystalline nitramines (RDX) improve performance and density. Additives like binders (rubber like polybutadiene), improve physical properties. Most of the DB propellants have smokeless exhaust.

Additives like solid aluminum perchlorate (AP) or aluminum (Al) increase performance, but the exhaust becomes smoky. Such propellants with additives like AP or Al are called Composite Modified Double Base (CMDB) propellants.

Composite Propellants: Composite propellants are heterogeneous propellant grain with the oxidizer crystals and powdered fuel (usually Al) held together in a matrix of synthetic rubber (or plastic) binder. Composite propellants are the most commonly used solid propellants.

The oxidizer is usually ammonium nitrate, potassium chlorate or ammonium chlorate. The fuels used are often hydrocarbons such as asphaltic-type compounds or plastics.

Conventional composite propellants contain between 60-72% of AP as crystalline oxidizer and up to 22% of Al powder as a metal fuel and 8-10% of binder (organic polymer).

Modified composite propellants use plasticizers like NC to add to performance.

Composite propellants give higher densities, specific impulse and wide range of burning rates.

Composite-modified double-base (CMDB): Combines additives in composite and double-base propellants.

Additives like solid aluminum perchlorate (AP) or aluminum (Al) increase performance, but the exhaust becomes smoky. Such propellants with additives like AP or Al are called Composite Modified Double Base (CMDB) propellants.

Solid Propellants-Desirable Properties/Characteristics: Desirable characteristics of solid propellants are given below:

1. High performance or high specific impulse. This means high gas temperature and low molecular mass.

   $$v_2 = \sqrt{\frac{2g_0 k}{k-1}\frac{R'T_1}{\mathfrak{M}}\left[1-\left(\frac{p_2}{p_1}\right)^{(k-1)/k}\right]}$$

   Hydrogen is the lightest propellant, followed by carbon, oxygen and fluorine. Among metals, aluminum, beryllium and lithium are lighter.
2. Predictable, reproducible and adjustable burning rate
3. The variation of burning rate exponent n and temperature coefficient a, with changes in motor thrust or chamber pressure must be small.
4. Adequate physical properties (including bond strength) over intended operating temperature range.
5. High density (allows a small chamber volume). Liquid hydrogen is energetic with low molecular weight. But it is a bulky substance requiring large tanks.
6. Predictable ignition qualities like reaction time etc
7. Good aging characteristics and long life. Aging and life predictions depend on propellant's chemical and physical properties.
8. Low absorption of moisture, which causes chemical deterioration.
9. Simple, low-cost and low-hazard manufacturing ease.
10. Guarenteed availability of raw materials
11. Low technical risk in handling, storage and transportation
12. Relative insensitivity of the propellants
13. Non-toxic exhaust gases.
14. Not prone to combustion instabilities.

Liquid Propellants

A liquid propellant rocket propulsion system is commonly called "rocket engine". It has all the hardware components and propellants required to produce thrust. It consists of one or more thrust chambers, one or more tanks to store propellants, a feed mechanism to force the propellants in to the thrust chamber, ignition mechanism where required and thrust vector control (TVC) systems.

The propellants in an LPR propulsion system constitute:

1. Oxidizer (Liquid oxygen, Hydrogen peroxide, Nitric acid-also called red fuming nitric acid),
2. Liquid fuel (gasoline, alcohol, liquid hydrogen etc)
3. Chemical compounds as ingredients (catalysts or gelling agents etc)

Today, we commonly use three LPR systems. They are:

1. Cryogenic oxygen-hydrogen system
2. Liquid oxygen-hydrocarbon fuel combination (used for booster systems mainly)
3. Several storable propellant combination systems used in ballistic missile systems (where long storage life is necessary). For example, Russia prefers using Nitric acid-hydrocarbon fuel combination. US uses Nitrogen tetroxide and hydrazine fuel.

Liquid Propellants-Properties: Selection of liquid propellants is a compromise of various factors.

Economic Factors:

1. Low cost propellants should be available in desired quantities. Production process must be simple.
2. Propellants must be storable, non-toxic.

Performance Factors:

3. Must provide high specific impulse and exhaust velocity. The liquid propellant combination of liquid fluorine oxidizer and liquid hydrogen has highest specific impulse of 480 secs and 1000 psia at sea level condition.

Physical Hazard factors:

4. **Corrosion effects** on performance should be low and predictable. Propellants like hydrogen peroxide and nitrogen tetroxide must be handled in containers and pipelines of specific materials.
5. **Explosion Hazard**: Propellants must be stable and safe to handle. Some propellants like hydrogen peroxide and nitro-methane are unstable and tend to detonate under certain conditions of impurities, temperature and shock.
6. **Fire Hazard**: Many oxidizers like nitric acid, fluorine, nitrogen tetroxide start chemical reactions with large variety of organic compounds. Most of the liquid fuels are readily ignitable when exposed to air and heat.
7. **Accidental Spills**: Unforeseen mishaps during transportation, loading/unloading of liquid propellants cause spills exposing people unexpected fires or health hazards.
8. Health Hazards: Many propellants are toxic and poisonous and need special precautions while handling.

Physical Properties:

9. **Low Freezing Point**: Low freezing point of propellants permits operation of rockets in very cold weather. Additives are available to depress the freezing point of propellants.
10. **High Specific Gravity**: In order to accommodate large mass of propellants in a given tank volume, dense liquid propellants are preferable.
11. **High Stability**: Propellants should not deteriorate or decompose with long term (over 15 years) storage. Propellants must also have good chemical stability with no decomposition during operation at elevated temperatures.
12. **Good Pumping Characteristics**: A low vapor pressure permits easy handling during pumping for loading and transportation purposes. If the viscosity is too high, pumping becomes difficult. Propellants like liquid oxygen, liquid hydrogen have high vapor pressure and need special design provisions for handling.
13. **Ignition, Combustion Properties:** Some liquid propellants combinations are spontaneously ignitable. They are called Hypergolic Propellants. All rocket propellants must be readily ignitable with small ignition delays.

Liquid Propellant-Types:

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 is essential for this type of rockets.

Storable propellants (Nitric acid or gasoline etc) are liquids at ambient temperatures and can be stored in sealed tanks for long periods.

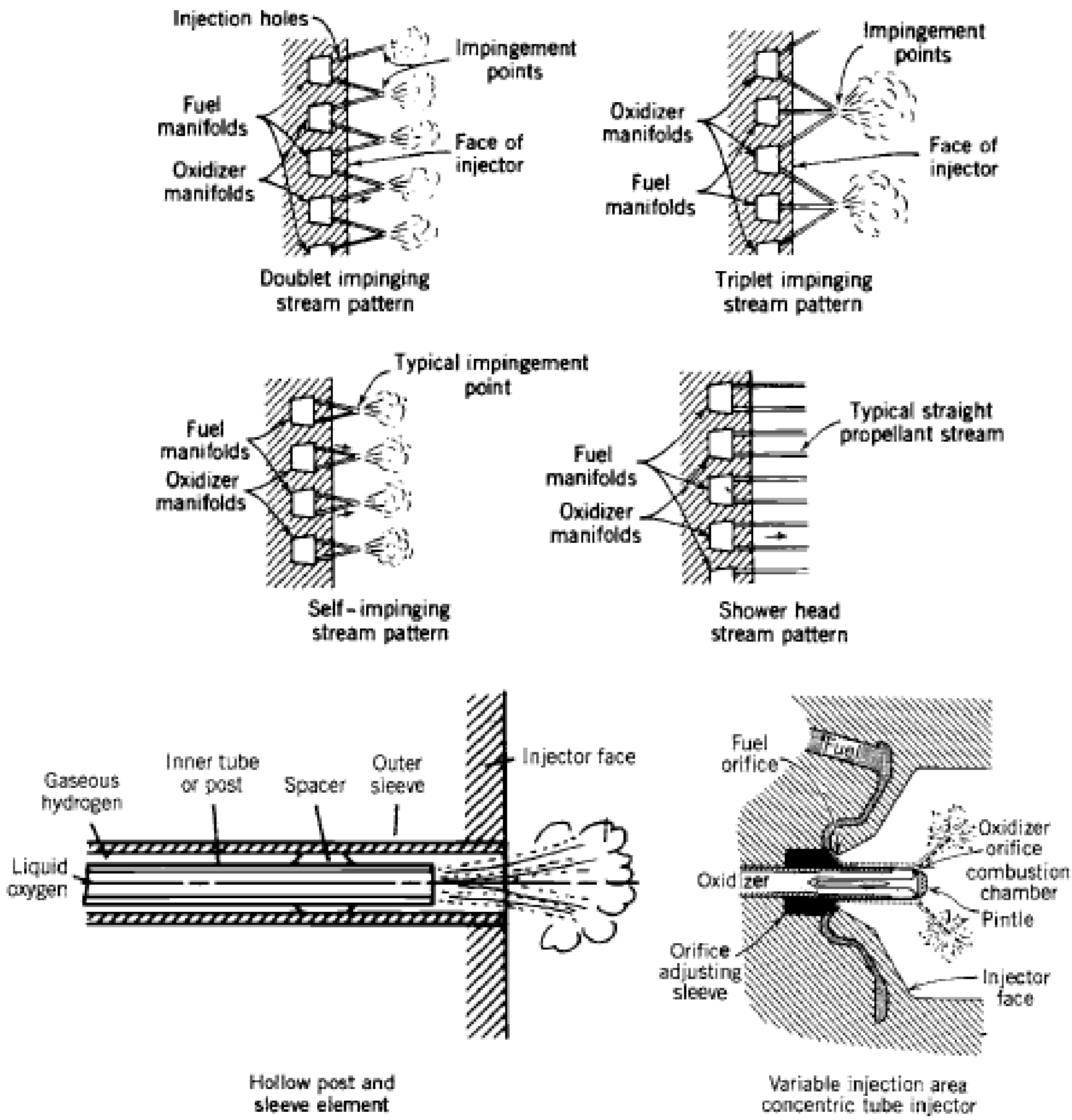
Space storable propellants (Ammonia) are liquids at space environment. Their storage tanks need specific design, specific thermal conditions and pressure.

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

1. Injector has to introduce and meter the flow of liquid propellants in to the combustion chamber
2. It has to atomize the fuel, that is cause the liquid to be broken up in to small droplets in the combustion chamber

3. It has to cause distribution and mixing up of propellants such that a correct proportion of mixture of fuel and oxidizer will result
4. 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:



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 afterburning, after valve closing leading to irregular combustion.

For applications needing very accurate terminal velocity requirements, the cut-off impulse has to be very small, with passage volume 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 outer sleeve by absorbing heat from the cooling jackets. The gasified hydrogen flows at high velocity (around 330 mt/sec) while the liquid oxygen flows slowly (around 33 mt/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.

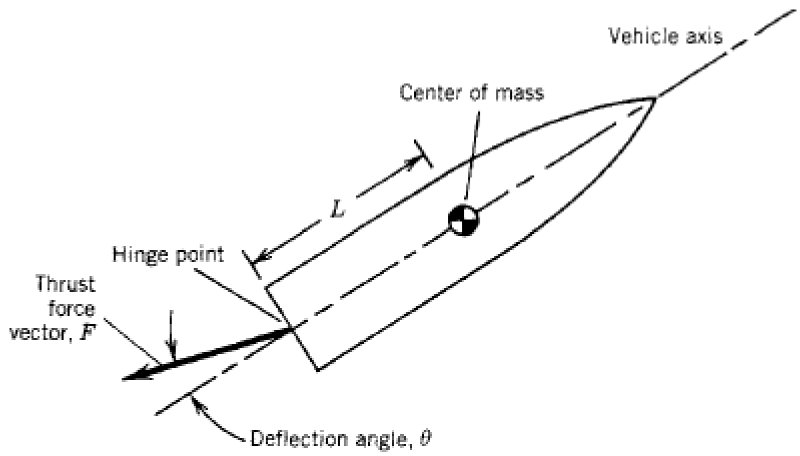
Thrust Vector Control in aircraft: Thrust vectoring nozzles are used in combat aircraft for take-off and landing. Thrust vectoring is also deployed to improve maneuvering and augment lift in flight during combat
Concorde aircraft used two convergent nozzles, primary and secondary nozzles
The secondary nozzle is convergent during take-off and used as divergent section during supersonic cruise

Thrust Vector Control in Rockets: In addition to providing thrust force to the flying vehicle, rocket propulsion system can provide moments to rotate the flying vehicle and thus provide control of vehicle’s attitude and flight path. The pitch, roll and yaw motions of the vehicle are controlled by varying the direction of thrust vectors.

All chemical systems provide several types of thrust vector control (TVC). TVC is effective only when propulsion system is operating and creating an exhaust jet. If the rocket engine is not firing, TVC is inoperative and the vehicle needs to be provided some other means of control.

Aerodynamic fins provided on the vehicle body are operative only in earth’s (or any other celestial body) environment.
The purposes of TVC are:

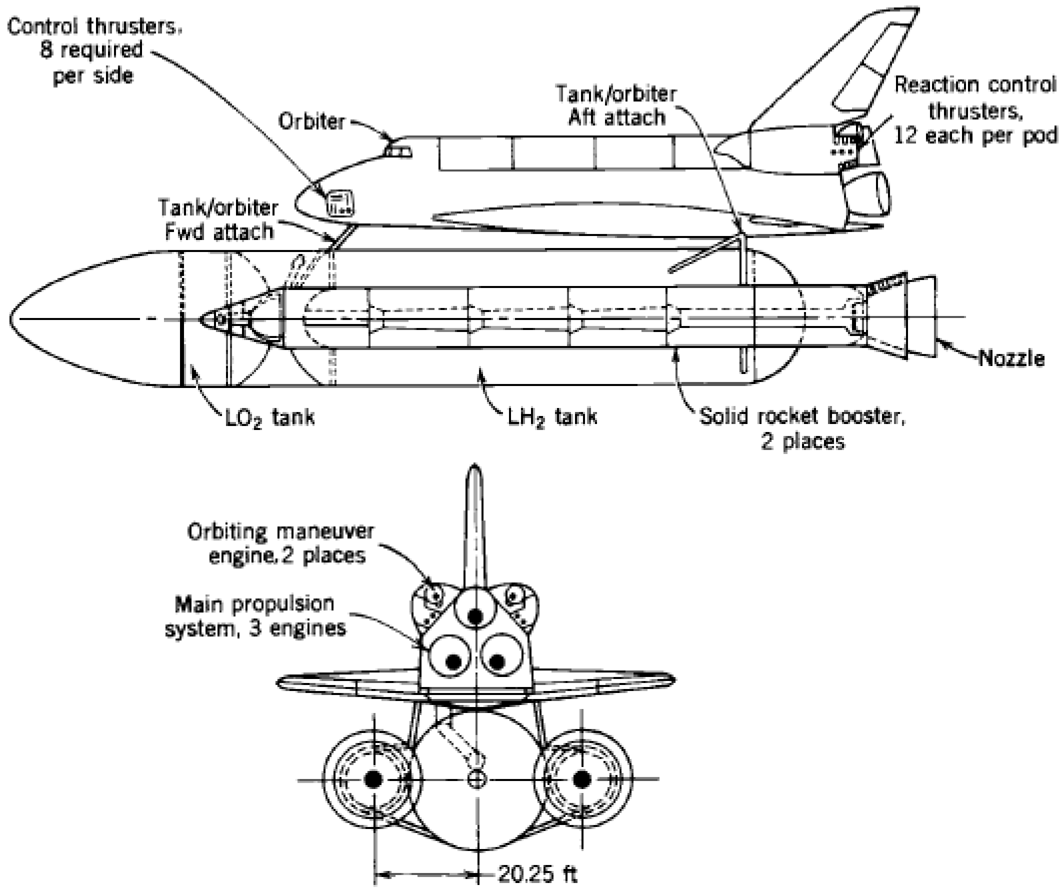
1. To willfully change flight path or trajectory of the vehicle
2. To rotate the vehicle or change its attitude during flight
3. To correct deviation between from intended trajectory
4. To correct for thrust misalignment of a fixed nozzle during operation



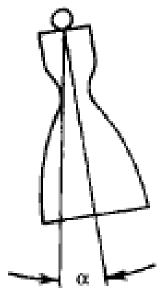
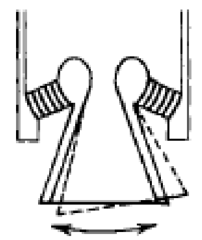
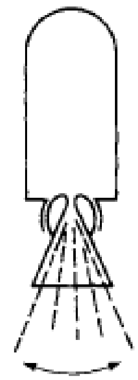
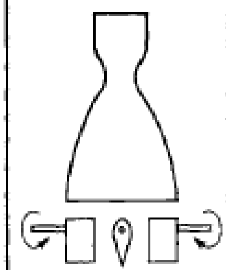
TVC mechanisms with Single Nozzle: Many different mechanisms are used on rocket systems having single nozzle. They are:

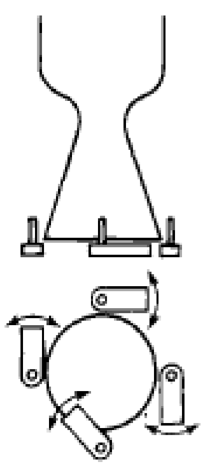
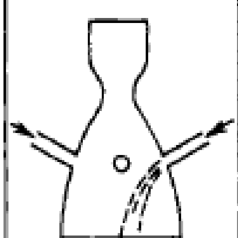
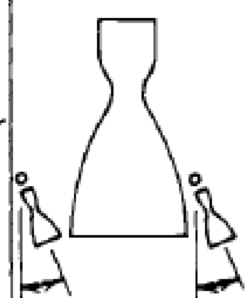
1. **Mechanical deflection of nozzle or thrust chamber**: The nozzle may be hinged, which permits rotation about one axis only or may be on a gimbal which is a universal joint allowing rotation on a 360^0. The whole engine is pivoted on a bearing rotating the thrust vector. Hinged or gimbal nozzle/engine needs flexible piping for propellant. This arrangement is commonly used. The space shuttle has two gimballed orbit maneuver engines and three gimbaled main engines.
2. **Insertion of heat-resistant movable body in to the exhaust jet**. Deflection the bodies inserted into the exhaust stream generates aerodynamic forces and cause deflection of a part of the exhaust gas flow. **Jet Vanes** are pairs of heat-resistant, aerodynamic wing-shaped surfaces, submerged in the exhaust jet flow of the fixed nozzle. Graphite jet vanes were first used in German V-2 missile in WW II and later in the Scud missile. Although jet vanes cause extra drag, they provide good roll or pitch control.
3. **Injection of fluid in to the side of diverging nozzle section**, causing asymmetrical distortion of supersonic exhaust flow. Injection of secondary fluid through the wall of the nozzle into the main gas stream has the effect of forming oblique shocks in the nozzle diverging section, thus causing an unsymmetrical distribution of the main gas flow producing a side force. This scheme has found application in few large solid propellant rockets, such as Titan III and Minuteman missiles. The injected fluid could be hot gas bled off from the main thrust chamber of SPR or LPR or generated through a separate gas generator.
4. **Separate thrust producing devices** that are not part of the main flow through nozzle.

Of all the mechanical deflection types, movable nozzles are most efficient. They are low weight TVC systems.
The jet-tab TVC system a simple low-torque system for SPR motors. They are successfully used in small tactical missiles. (Tomahawk cruise missile). Usually four tabs, independently actuated are used, rotated in or out of the exhaust stream.



[Simplified sketch of the Space Shuttle Vehicle. The shuttle orbitor is a delta winged vehicle about the sized of medium range jetliner. It is reusable, cargo carrying spacecraft-airplane combination that takes off vertically and lands horizontally like an aircraft. Each shuttle orbitor is designed for minimum 100 missions, and can carry up to 65,000 lbs of payload to a LEO and a crew of 4 members and 10 passengers. It can return up to 25,000 lbs of payload back to earth.]

Gimbal or hinge	Flexible laminated bearing	Flexible nozzle joint	Jet vanes
			
Universal joint suspension for thrust chamber	Nozzle is held by ring of alternate layers of molded elastomer and spherically formed sheet metal	Sealed rotary ball joint	Four rotating heat resistant aerodynamic vanes in jet
L	S	S	L/S

Jet tabs	Side injection	Small control thrust chambers
		
Four paddles that rotate in and out of the hot gas flow	Secondary fluid injection on one side at a time	Two or more gimballed auxiliary thrust chambers
S	S	L

TVC with multiple thrust chambers or Nozzles: Several rocket systems use two or more rocket engines or a single rocket motor with two or more nozzles. With such systems, two gimbaled nozzles or thrust chambers provide roll control which need very small deflections. For pitch or yaw control, deflections would be larger and in the same direction. This can be achieved four hinged or gimbaled nozzles or thrust chambers.

Differential throttling concept can be used on multiple nozzle or thrust chamber systems. It has four thrust chambers, which are selectively throttled producing differing thrust levels of between 2 to 15%.

Cooling of Thrust Chambers/Nozzles: The primary objective of cooling the chamber and nozzle walls from becoming too hot or enable them to withstand high loads or stresses. Most wall materials lose strength as temperature is increased.

Basically, there are two types of cooling methods employed in rocket vehicles. They are

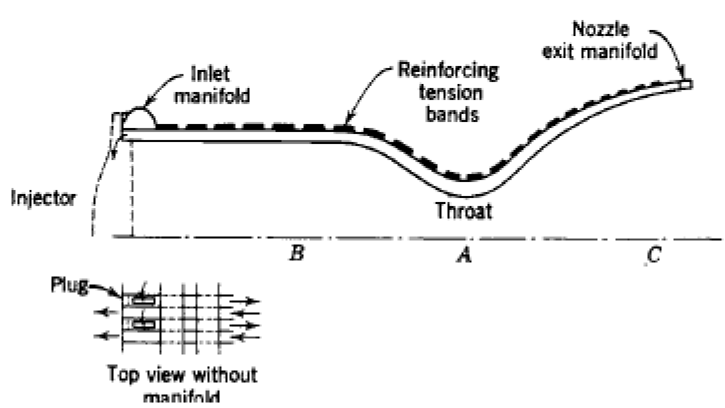
- Regenerative Cooling
- Radiation cooling.

Both above methods are called as **Steady State Methods**, in which the heat transfer rate and the temperature of the chamber reach thermal equilibrium.

Regenerative Cooling: Regenerative cooling is done by building a cooling jacket around the thrust chamber and circulating one of the liquid propellants (usually, fuel) through it before it is fed to the injector. This cooling technique is used primarily with liquid bipropellants thrust chambers of medium to large thrust. It can be used to cool the injector also.

Internal cooled passages, cooling jackets or cooling coils permit the circulation of the coolant. Nozzle throat is usually the location where highest temperatures are attained. The coolant passages are designed such that the coolant flow velocity is highest at the critical places like throat regions.

Coolant enters through the inner manifold in to every other tube and proceeds axially towards nozzle exit.



Radiation Cooling: In radiation cooling, the chamber or the nozzle has only one single wall made of high temperature material. When it reaches thermal equilibrium, this wall usually glows red and radiates heat away to the surroundings or to empty space.

Radiation cooling is used for LPR thrust chambers, divergent nozzles with area ratio of about 6-10. Small thrust propellant chambers suit the radiation cooling method.

The heat absorbed by the coolant is not wasted. It augments the initial energy content of the propellant before injection. This has the effect of increasing the exhaust velocity by 0.1 to 1.5%.

For divergent nozzles with high area ratios, the temperatures in the exit section of the divergent nozzle are relatively low. Hence, un-cooled high temperature materials like niobium or carbon fibers are used.

Transient heat Transfer or Unsteady Heat Transfer: It is also called “**Heat Sink Cooling**”

Thrust chambers with unsteady heat transfer are basically two types. One is a simple metal chamber (steel, copper etc) made with walls sufficiently thick to absorb the heat energy.
The thrust chamber does not reach thermal equilibrium. The temperature keeps increasing with operating duration. The heat absorbing capacity of the hardware determines the maximum operating duration. The rocket combustion operation must be stopped before any of the walls reach critical temperature at which it could fail.
This method is mostly used in motors with small camber pressure and temperatures.

The other method of heat sink cooling of thrust chambers can be done by absorbing heat in an inner liner of ablative material, such as fiber-reinforced plastics. Ablative materials are also used for short duration transient operation. Ablative materials consist of series of strong, oriented fibers like glass, Kevlar or carbon, arranged in a matrix using epoxy resins.

Ablative materials are usually used in nozzles, but also as insulation materials. They are usually composite materials like high silica glass, Kevlar etc.

The ablation process is a combination of surface melting, sublimation, evaporation and decomposition in depth. Progressive layers of ablative material undergo endothermic degradation, that is, physical and chemical changes that absorb heat. While some of the ablative material evaporates, remaining becomes charred and porous layer on the cooled surface.

Film Cooling and Special Insulation are supplementary techniques used to locally augment their cooling capabilities

Unit III

Nuclear Rocket Propulsion

Limitations of chemical propulsion systems for deep space missions:

- **Chemical propulsion systems cannot provide high ΔV required for deep space missions**.

Types of propulsion systems along with maximum delta V, are given below:

- **Chemical**: Max ΔV=Solid-5.7-7.1 km/s; **liquid-6.9-11.5 km/s**
- **Nuclear**: Max ΔV=Fission-**11.5-20.7km/s**; Fusion-**230-2300 km/s**;
- **Electric**: Max ΔV=Electro-thermal-3.5-27.6 km/s; Electrostatic- **27.6-230 km/s**; Electromagnetic-max- 16.1-115 km/s

- **Chemical propulsion systems are limited by the energy stored in their propellants.**
- **The energy source to the engine and the propellant flow must be separate, in order to provide much higher exhaust velocities.**
- **Nuclear or electric energy sources can be utilized to speed up the propellant exhaust to very high speeds.**
- **This limitation can be overcome by using power available on board, from nuclear/electrical/solar sources, and couple it with propellant carried on board**

Nuclear Rockets-Power/Thrust/Energy

1. Interplanatory/deep space and manned missions need high exhaust velocities (beyond 11 km/sec)
2. Nuclear fuels have high specific energy and are ideal for above missions
3. The **specific energy(power per unit mass)** required for interplanetary missions works out to 60.5 MJ/kg.
4. The specific energy of a **LOX/LH** engine is 10.4 MJ/kg.

5. So about **6 kg of propellant** is needed to be burnt for every **1 kg of vehicle mass** for interplanetary mission.
6. In comparison, the **energy contained in a kg of pure uranium 235 is 79.3 x 10^6 MJ**.
7. A **single kg of uranium 235 can provide energy to place a 1000 t vehicle for interplanetary mission.**

Low Molecular Weight Exhaust:

- **The chemical engines on the Space Shuttle Main Engine (SSME) have a much higher temperature than a solid core nuclear thermal rocket (NTR) (4,000K as opposed to 2,000K).**

- **But the NTR has a higher exhaust velocity because it uses low molecular weight hydrogen as propellant, instead of that high molecular weight combustion products that comes out of the SSME.**

- **Why cannot chemical engines use low molecular weight propellant? Because in chemical engines, the fuel and the propellant are one and the same, but in an NTR the fuel is the uranium and the propellant is an inert gas.**

    - **Chemical rockets** expel a ton of mass at a relatively low velocity. ***(high propellant mass flow but low exhaust velocity)***

    - **Ion drives** expel a tiny amount of mass, a low 0.0001g. ***(low propellant mass flow but high exhaust velocity)***

Nuclear Rocket-Basic Concept:

1. Nuclear process (fission or fusion) uses very small quantities of fuel
2. The end product of fission or fusion is smaller resulting in "**mass defect**".
3. This mass defect releases energy based on Einstein's $E=mC^2$
4. Nearly all gained energy through the mass defect is released as heat.
5. A working fluid, usually an inert gas, is coupled with the nuclear reaction products, generating the propellant expelled out of the space vehicle

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

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

Whereas the thrust produced, F is given by

$$F = m v_e$$

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

where m is the mass flow rate and v_e is the exhaust velocity

This is the relation beween power and thrust prodeced of a rocket vehicle.

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 Fission& Fusion-Basics:

Nuclear Fission: 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 splitting of nucleus is result of neutron bombardment.

The neutron is absorbed a uranium nucleus, which causes the nucleus to split into two nuclei (of mass about half that of uranium). The sum of the two smaller nuclei is smaller than the original un-split nucleus. This difference is called the "mass defect".

The mass defect causes release of very large amount 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**.

Nuclear Fission is used in high thrust applications.

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.

Control of Nuclear Fission: 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 since the reduced kinetic energy converts to heat, 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. This is the principle of generating thrust in a nuclear powered rocket engine.

The fission process, splitting of uranium nucleus results in release of two or more neutrons with high kinetic energy of around 200 MeV of energy. In principle, these scattered neutrons travelling randomly, can cause furthur uranium nuclei to split and release more neutrons, thus setting up a chain reaction.

Inability of Natural Uranium to sustain chain reaction: Two isotopes of Uranium, U^{238} and U^{235} are available, of which U^{235} has high probability of initiating fission process.

Uranium is a natural material in which the major constituent is U^{238} and the other constituent U^{235} is only 0.72%. Although U^{238}undergoes fission, but the probability is very low. Since the U^{238} nucleus will not split if the incident neutron has energy levels less than 150 MeV. The neutrons which emerge out of fission process with energy levels of around 200 MeV, quickly lose their energy as they scatter and thereafter cannot cause fission in an U^{238} atom. Low energy neutrons are called “thermal” since their motion is similar to other atoms in the uranium matrix with low energy levels. On the other hand, U^{235} atoms readily participate in fission process even at low energy levels.

Very few neutrons released by fission process cause further atoms to split in a uranium matrix dominated by U^{235}. However, the neutrons with low energy levels get absorbed by U^{235}atoms and will not participate in further fission process. Therefore, it is not possible to sustain chain reaction in pure natural uranium with low U^{235} content.

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 of interaction between a cooling neutron and a U^{235} atom. The process of enrichment is complicated and costly. Uranium with enough quantity of U^{235} is called “enriched” and depending on the intended use, it could be 2%, 20%, 50% or even 90%. Enrichment methods are based on diffusion process through filters or centrifuge action.

- **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**.

Sizing of Reactor: 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.

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 even more natural Uranium, without enrichment. However, this increases the size of the reactor, as more moderator is required.

The need to keep the size low for space applications, will require use of enriched uranium, which is highly poisonous and radio-active. Safety issues add to the complexities of building a small sized reactor for space applications.

Neutron Leakage, Control and reflection

Calculating Criticality: Criticality factor relates to calculating the space/size that the fission reactor needs to 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.

To arrive at optimum size of the reactor, criticality factor is considered which is defined by the “four-factor formula”, as given below:

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

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_∞< 1, no chain reaction is possible

For K_∞ > 1, the chain reaction is possible

K_∞ = 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:

η 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, their chances of capture are reduced, with value of p high. Value of p ranges from 0.6 to 0.8.

The fourth parameter f is the "thermal utilization factor", indicating probability of capture of low energy neutrons after slowing down by moderator.

Reactor Dimensions/Neutron Leakage:

As the size of the reactor decreases, the neutron leakage increases, less space is available for moderator. Therefore, more neutrons need to be provided which requires enrichment of natural Uranium. For very small reactors, almost 90% 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 neutron flux, as the distance from source of neutrons increase. It is about 52 cm in graphite.

The slowing-down length expresses the mean distance travelled by neutrons, 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

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

Where N & N_0 are the number of neutrons crossing a unit volume of materialin at the source and as the distance increases, situated at a distance r from the is the 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, within 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:

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

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

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.

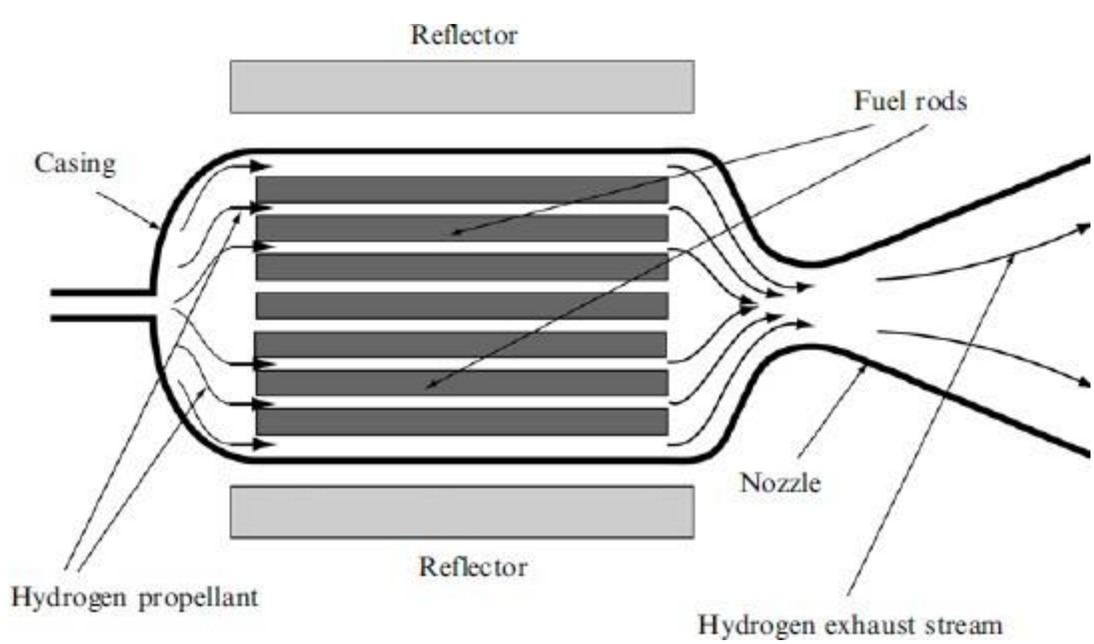
To increase the power level, value of k is allowed to become greater than 1. Once the desired power level is reached, k is returned to value of 1, and the reactor contimues to produce power at the new level. A decrease of power is also established in a similar way.

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

Nuclear Thermal Rocket Engine:



Operation: Hydrogen propellant enters the engine core from the left, and is heated as it passes down the channels in the fuel rods. The hot gas then expands down the nozzle to generate high velocity exhaust stream. The rate of fission and the heat production is controlled by the reflector.

The exhaust velocity of a nuclear thermal rocket, with a conservative design, (power output of 970 MW) produced 8.7 Km/s which is nearly double than that of LOX/LH LPR engine. This shows that using separate energy source and propellant allows much higher exhaust gas velocities.

A nuclear engine can produce high thrust levels comparable to chemical engines and high exhaust gas velocities of electrical engines.

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 specific engineering details are unique to the fission engine. They are:

1. Optimum use of enrichment and moderator
2. High temperature properties of nuclear fuel elements and moderator material.
3. Radiation and its management
4. Propellant Flow & Cooling
5. Start-up and Shut-down

Optimum use of enrichment & moderator: The space reactor will use enriched uranium containing between 50-90% U^{235}. As the percentage of U^{235} increases, k_{∞} also increases. And more moderator is needed to avoid too many cooled neutrons being absorbed without taking part in fission process. An **optimized mixture of U^{238}, U^{235} and moderator provides the core size small enough to be used in space.**

High temerature properties of fuel elements/moderator: The **thrust developed** by the nuclear thermal engine depends on the **temperature of the propellant**

gasses and their molecular weight. Considering that we use prpellants like hydrogen with low molecular weight, the temperature attained by the propellant from the nuclear core limits the thrust developed. The temperature which the nuclear core fuel elements can withstand without losing their mechanical properties, is the limiting factor. **High temperature properties of the nuclear elents and moderator material sets the limit to the propellant maximum temperature.**

Uranium metal itsaelf melts at 1400k. Comparing with typical chemical rocket combustion temperatures of 3200k, Uranium metal is not fit for use as nuclear fuel. The most common compound of uranium used as nuclear fuel is uranium dioxide, UO_2, which can withstand temperatures of up to 3075 k. It is also stable with hydrogen. Uranium carbide is another compound with a melting point of 2670 k.

Graphite is commonly used as moderator with high material integrity up to 3990 k and good structural and dimensional properties. However, at high temperatures, it chemically reacts with hydrogen forming hydrocarbons, causing erosion of fuel/moderator elements. For prolonged use, the fission fragments in the exhaust stream are highly radioactive. This has dangerous consequences during ground testing and low altitude operations. A protective coating on fuel elements (graphite – uranium oxide matrix), using neutral elements like **Niobium or Zirconium carbides prevents the erosion of fuel/moderator core**.

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

Start-up and Shut-down:

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 propellantturbo-pumps. **Initially, during starting phase, electrical power must drive the turbo-pumps.**

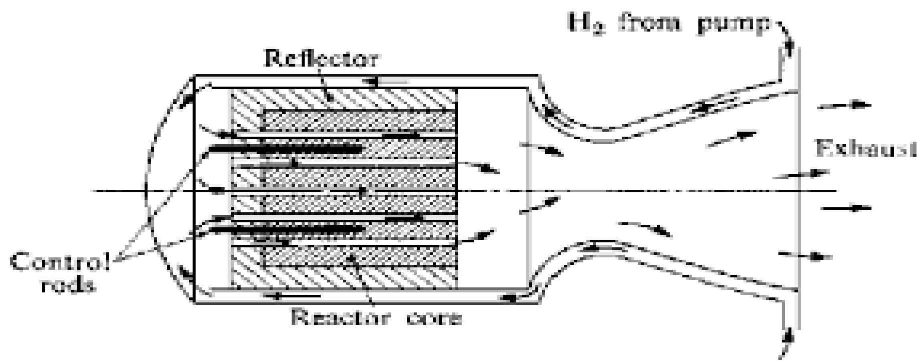
Once, the engine is in stable operating mode, the **thrust can be varied by positioning the control rods**.The **power ouput is a function of neutron flux**.

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.

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

Solid Core Nuclear Rocket Engine



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 only exhaust velocity is considered.
5. An ion engine can only generate thrust of fraction of a newton or just a few newtons, 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.

Operational issues with nuclear engine:

1. Start-up and shut down opeations for a nuclear engine are more complicated than for a cemical engine.
2. Major operational issues associated with use of nuclear engines are associated with radiation and danger to human life. Right now testing of nuclear engines within earth's atmosphere. Although hydrogen ued as propellant is not hazardous, there are fuel rod fragments caused by erosion in the exhaust which is radioactive. This limits use of nuclear engines to earth's orbit and beyond.
3. Firing of nuclear engine is considered only for high earth orbits and for inter-planetory transfer manoeuvres.

APS Unit IV: Electric Rocket Propulsion & Advanced Systems

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

Limitations of Chemical Rocket Engines:

- **Explosion & Fire Potential (SPR &LPR)**: Explosion and fire potential is larger, failure can be catastrophic.
- **Storage Difficulty (SPR &LPR)**: 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.
- **Loading/Transportation Difficulty (SPR & LPR)**: 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.
- **Separate Ignition System (SPR &LPR):** All propellants , except liquid hypergolic propellants, need ignition system. Each restart requires separate ignition system.
- **Smoky Plume (SPR & LPR):** Smoky exhaust plumes can 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.

Limitation of Chemical Rocket Propulsion: Chemical rockets use the energy stored in the propellants to create a hot gas, which then becomes the working fluid in the heat engine and is expelled through the nozzle, generating thrust. The chemical propellant is performing double function of energy source and ejected working fluid (combustion products).

The fundamental limitation is that no more energy can be put in to the rocket than that contained in the propellant.

Multi staging can improve the vehicle velocities achieved by chemical rockets, but these systems can not be used for long range, deep space missions, inter-planetary missions.

Electric Propulsion Systems:

Structure: The basic subsystems of an **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 in to kinetic energy exhaust. The term thruster is commonly used to mean the thrust chamber.

Operation of Electrical thrusters:
Electrical thrusters can be divided into two broad categories;

- those that use electricity to heat up the propellant, which emerges as a neutral gas these are called electrothermal thrusters)
- those that use electric and magnetic fields to accelerate the ions. The functional form and analysis of these two classes differ. (These are called electrostatic and electromagnetic thrusters)

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 Magnetoplasma engine:** In this type, the acceleration is achieved by the interaction of electric and magnetic within the plasma. The plasmas are moderately

dense, high temperature gases which are electrically neutral but good conductors of electricity.

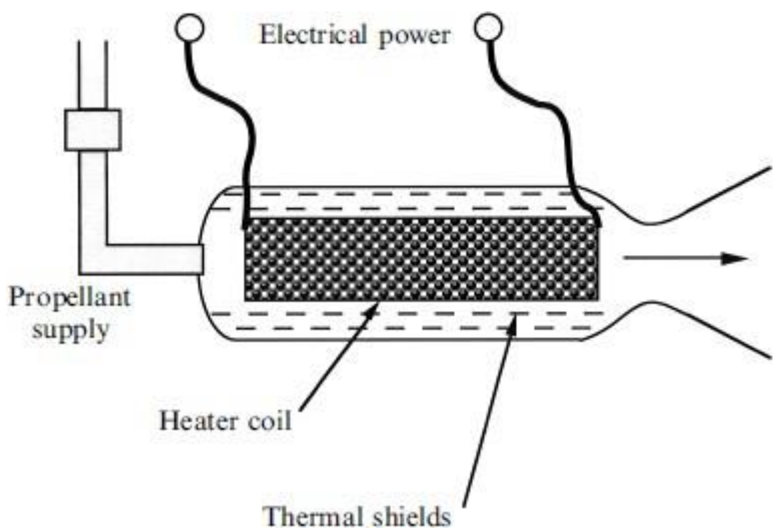
Electrothermal Thrusters: Two types are available-Resisto-jet and Arc-jet thrusters.

Resisto-jet thruster:

Operating Principle & Components: Resisto-jet, consists 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 resisto-jet thruster is illustrated below:



The ressistojet units have thrust ranges of 0.01 to 0.5 N, with exhaust velocities of 1000 to 5000 m/sec. Propellants used could be hydrogen, helium, water (even waste water can be used) or hydrazene.

Characteristics of Electrothermal thrusters:

- Electric thrusters can attain very high exhaust velocities.
- 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%

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

Arc-jet thruster:

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. Resisto-jet thrusters use the simplest way to heat the propellant with a hot wire coil, through which an electric current passes.

In an Arc-jet thruster, 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 propellant gas is heated by passing an electric arc through the flow. Temperatures in the order 30,000-50,000 k 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 resistojet thruster. The temperature limit can be much higher.

Arcjet units can attain maximum exhaust velocities are around 20 km/s. Hydrogen, ammonia and hydrazine are used as propellants.

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

Disadvantages: 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 resistojet thrusters.

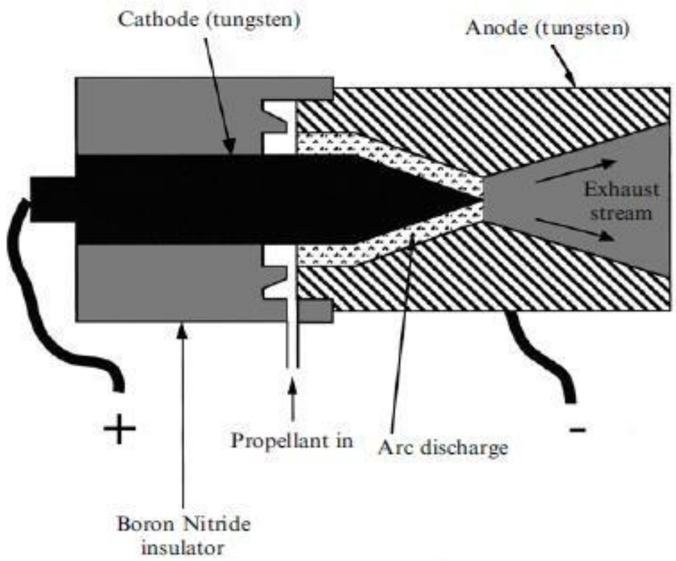
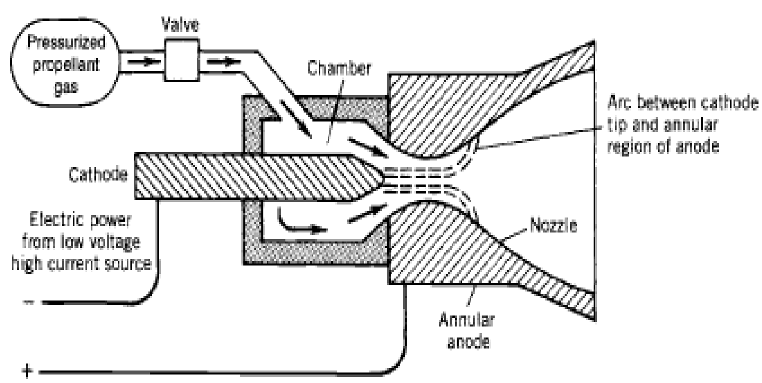


Figure 6.6. Schematic of an arc-jet thruster.

(Another diagram)

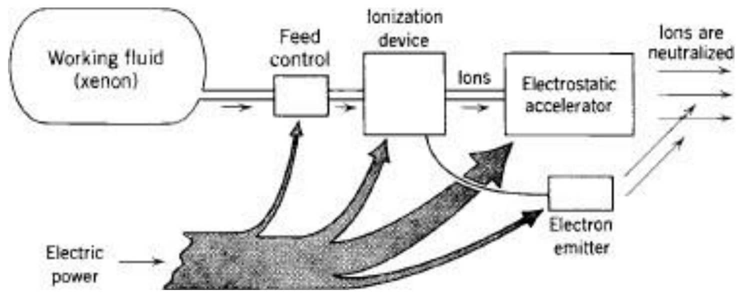


Electrostatic and Electromagnetic thrusters accomplish propulsion through different means. They do not use thermodynamic expansion of gas in the nozzle.

Both Electrostatic and electromagnetic thrusters work only in vacuum.

Electrostatic Thrusters: **Ion Rocket Engine**

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:



Performance Parameters:

If the propellant is ionized, it can be accelerated very effectively by electrostatic fields. The velocity gained for an **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

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

And the force generated F, can be expressed as

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

For obtaining very high specific impulse, a multi-ionised, light ion would be ideal. However, since the thruster should produce high thrust, propellant with heavy ions is preferred.

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 velocity exhaust stream. Highest exhaust velocities (more than 32,000 m/s) are achieved by accelerating positive ions in an electric field created by two grids having large potential difference.

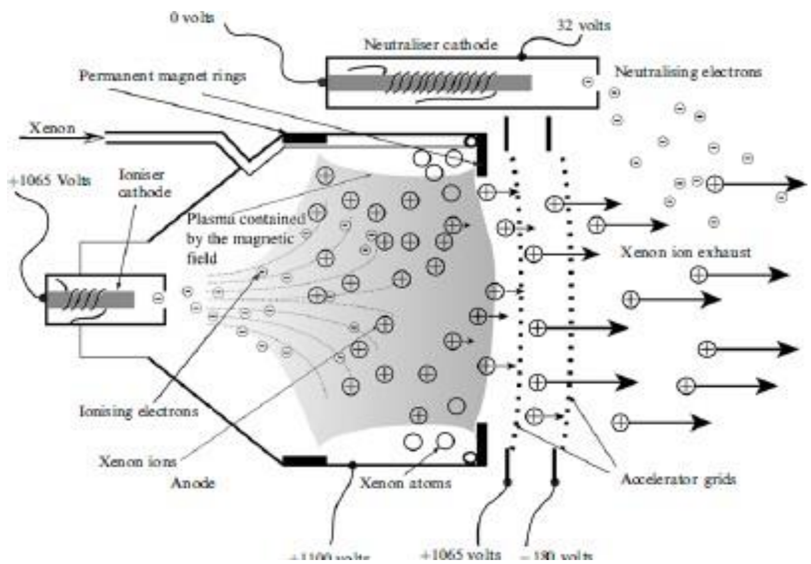
The electrons do not leave, therefore the electron current is discharged through a neutralising cathode, in to the exhaust. This would neutralise the spacecraft. The electrons discharged carry little momentum, therefore do not affect the thrust.

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 ionize propellant through electron collision. The ionised propellant drift through the grids with high 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 the thruster to the spacecraft. **The exhaust velocity is governed by the potential difference between the grids and the mass flow rate is directly related to the current flowing between the grids**.

There is no need for a nozzle to generate thrust .



Applications of Ion Engines:

Ion engines are best used for very high velocity increment missions like inter-planetary missions and station keeping.

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

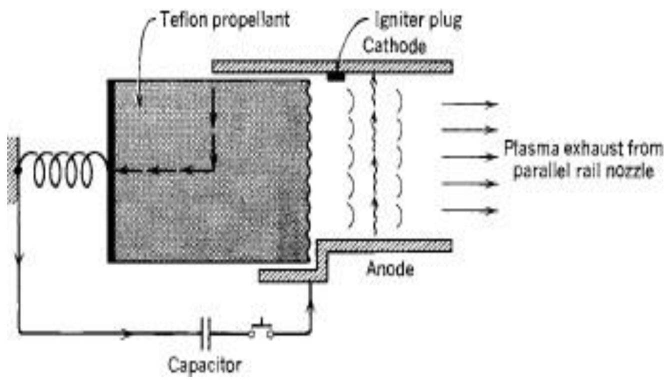
Limitation of Ion Thrusters- The space-charge limit**:**

The accelerating grids have an electric field between them, which gets partially blocked as the ions start accelerating along the grids. As the density of flow of ions increases, a point will reach when the 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 Thruster: (**Magnetoplasma Rocket)** Electrical plasma is an energized hot gas containing ions, electrons and neutral particles. In the magnetoplasma rocket, 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:



The thruster uses a parallel rail accelerator for self-induced magnetic acceleration of the 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.

Operating Principle: 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

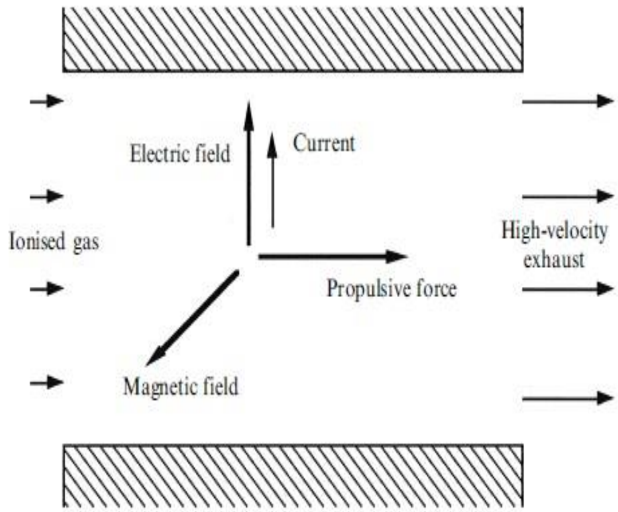


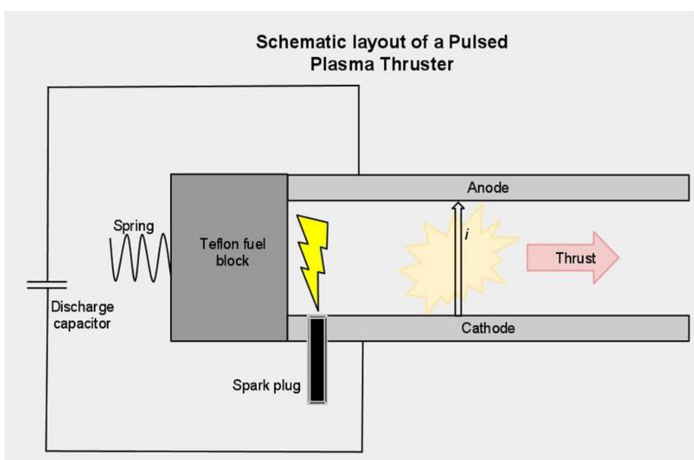
Figure 6.14. Principle of the plasma thruster.

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.

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).



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. It is also called MPD arcjet or Lorentz force accelerator.

In MPD thruster, a gaseous propellant is ionized and fed into an acceleration chamber where magnetic and electric fields are created using a power source (usually solar array charger and convertors).The magnetic field may be externally applied or induced.

The particles are then propelled by the Lorentz force resulting from the interaction between current flowing through the plasma and magnetic field. The thrust and specific impulse of the vehicle depend on the power input. MPD thrusters usually generate exhaust velocity of 15-60 km/s and thrust between 2-25 N. They operate with an efficiency of 40-60%

Various propellants such as xenon, neon, argon, hydrogen, hydrazine and lithium are used.

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 in to the channel, and ionised by colliding with the circulating high energy electrons.

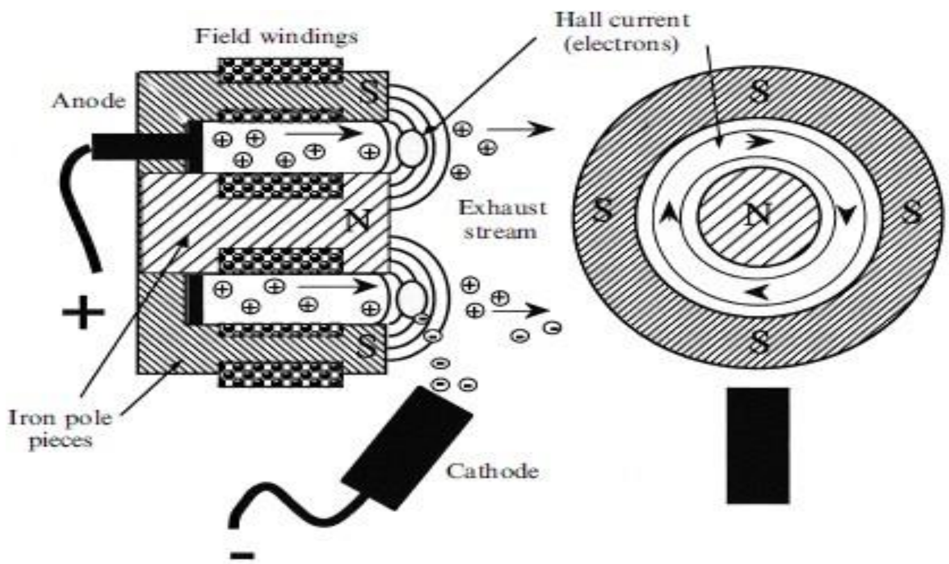


Figure 6.16. Schematic of the Hall thruster.

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 80mN 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. These 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.

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 four broad categories:

1. **Attitude Correction (Space Station/Spacecraft)**: Overcoming translational and rotational perturbations: These would include
    - **Station keeping for satellites.** Especially, in geosynchronous orbits (GEO), satellites have long life and need extensive station keeping.
    - **Aligning telescopes** or **antennas** in Low Earth Orbits(LEO) and Medium Earth Orbits (MEO)
    - **Drag compensation for satellites** in LEO and MEOs

    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. Hall thruters and ion engines are also used.

2. **Raising Orbits**: 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. All inter-planetary travel and deep space missions need orbit raising that need relatively high thrust and power for high velocity increments. Hall thrusters and ion engines are mostly used.
3. **Interplanetary missions**: Potential missions as Inter-planetary travel or Deep space probes. Ion engines are most suitable.

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, 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 η, in converting electric power to thrust, which is expressed as

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

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

The exhaust velocityv_e can be expressed as

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

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

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

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

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,

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

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 = 2000 sec; F = 0.20 N; burn time (duration) = 4 weeks = 2.42×10^6 sec ; Pay load mass = 100 kg;

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× 10^{-5}

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

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

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 burn out = mass of power system + mass of pay load = 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 Δ V = $v_e log_e$ R = 2000 9X8 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

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

The exhaust velocity is given by the relation

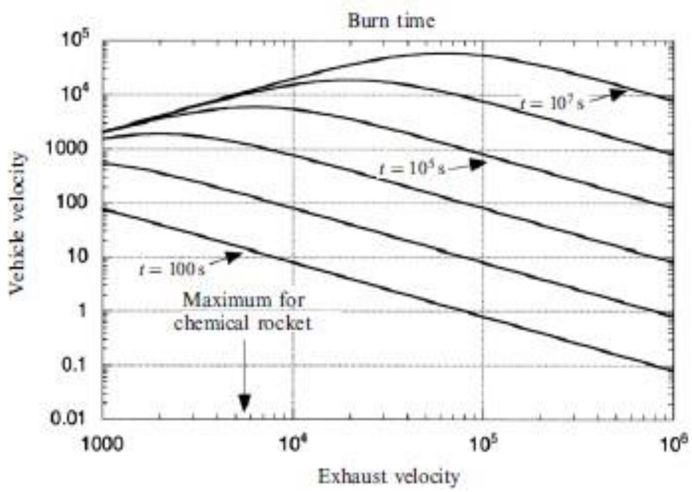
$\frac{M_E}{M_P} = \frac{v_e^2}{2\eta\xi t}$

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

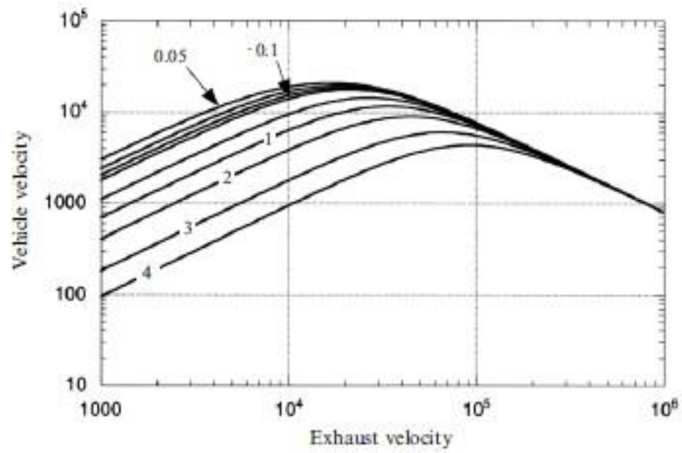


- Above graph assumes a fixed relationship between exhaust velocity and power supply mass, with burn time as a parameter.
- The ratio of structural mass to propellant mass is also fixed at 0.15, equivalent to a mass ratio of 6.6.
- It is evident that vehicle velocity does not always increase with exhaustvelocity., 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:

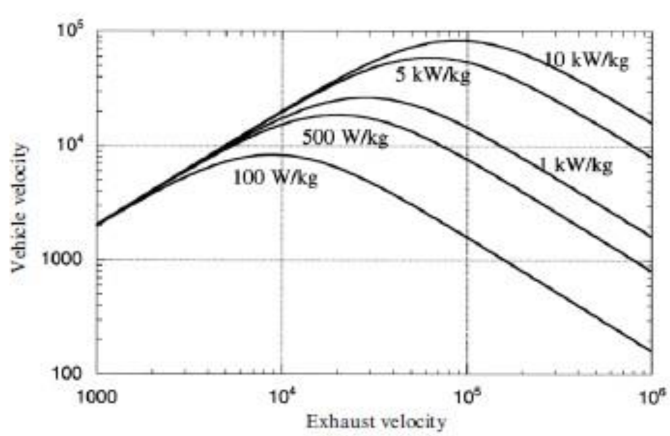
- In the interrelation below, the burn time is fixed at 1 million seconds, and the power-to-mass ratio, ξ is fixed at 500 W/kg.
- The ratio $\frac{M_S}{M_P}$ is shown as a parameter.



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**

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.

Solar Electric Propulsion (SEP):

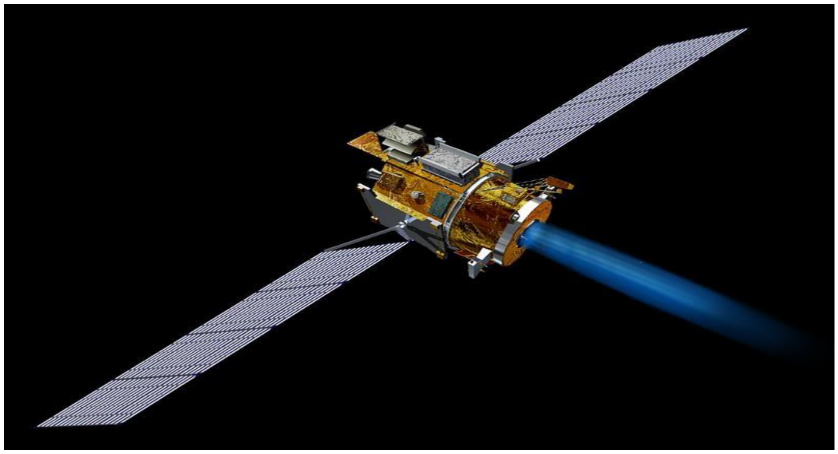
Solar Electric Propulsion (SEP) refers to the combination of solar cells and one or more electric thrusters used to propel the spacecraft through outer space. SEP has significantly higher specific impulse than normal chemical rockets, thus requiring less propellant mass making it suitable for missions to Mars and beyond into deep space.

Solar cells have been used for space applications since 1958. Solar cells are made of p-n junctions, convert photon energy to electric energy. Silicon cells and gallium arsenide cells are space qualified foruse. Typically, solar cell arrays are designed for 20% over-capacity to allow for material degradation during operation. Performance degradation of solar cells is due to radiation and particle impact damage, particularly in the radiation belts around earth.

Orientation to Sun is a very critical factor when using solar cells. Solar cell panels can be

- Fixed and body mounted to the spacecraft

- Rigid and deployable (protected during launch and positioned in space)
- Flexible panels that are deployed (rolled out or unfolded)



Solar electric propulsion takes advantage of magnetism and electricity to propel the vehicle in space. Electricity produced by the solar cells and the magnetic field in the vehicle provide the thrust force through ejecting ions/plasma.

Energised by the electric power from onboard solar arrays, the SEP system uses ten times less propellant than a comparable conventional chemical propulsion system. SEP is capable of manned missions in to solar system and beyond.

SEP project usually uses electrostatic Hall thrusters which generates and traps electrons in the magnetic field, using them to ionize the onboard propellant, usually inert xenon gas. The ions are ejected by the Lorentz force orthogonal to the electric and magnetic field, generating thrust. The system is able to generate more than 20 km/sec to enable all orbital transfers required for deep space missions.

ADVANCED SYSTEMS: Micro-propulsion, application of MEMS, chemical, electric micro-thrusters, principle, description, Propellantless propulsion, teethers, momentum exchange, Photon rocket, be energy propulsion, solar, magnetic sails.

Micro-Propulsion:

Micro-Propulsion system provides extremely small and precise thrust for a variety of satellite missions. Formation flying and precise attitude control are examples where thrust levels in the micro- to milli-Newton range are required.

The micro-propulsion system contains the thruster module which is a silicon wafer stack with four complete rocket engines with integrated flow control valves, filters, and heaters. Extremely small heaters are located inside the thrust chamber to improve the specific impulse and hence efficient use of the propellant.

Micro-propulsion systems use Micro Electro-Mechanical Systems (MEMS) technology. MEMS technology involves creating very small components, fabricated in the form of silicon chips. Chips can be bonded together, allowing nozzles, heaters, valves, filters, and controls to be sandwiched into very compact units.

MEMs types:

Chemical MEMS include:

- MEMS based cold gas propulsion
- Micro-monopropellant rocket engine

MEMS Electric Propulsion Units include:

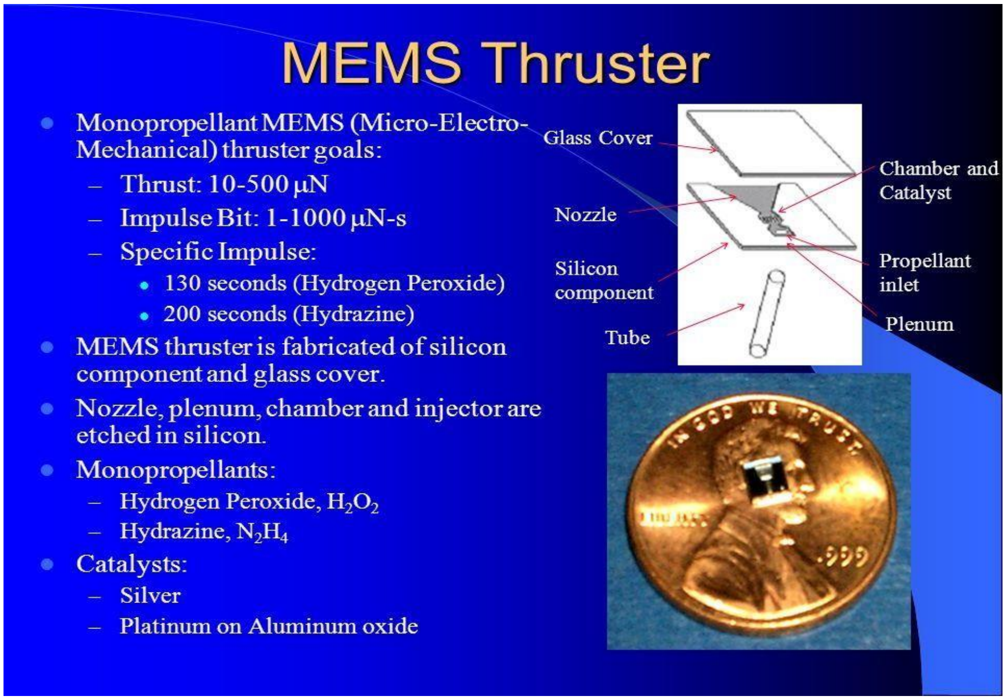
- Ion Engines
- Hall Thrusters
- Pulsed Plasma Thrusters (PPT)
- Field Emitted Electric Propulsion (FEEP)

Propellants used in MEMs:

- Hydrogen peroxide is commonly used mono-propellant
- Cold gas provides thrust in MEMS
- Teflan is popularly considered for PPT thrusters
- Hall and the ion propulsion systems commonly use xenon

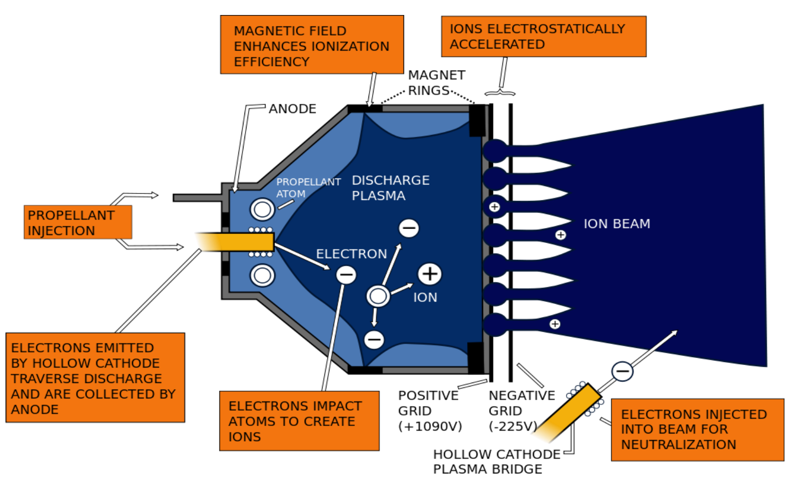
MEMs Resistojet:

- One type of MEMS thruster is a resisto-jet which works by heating gas molecules to increase their energy before expelling them through a nozzle
- The MEMS resisto-jet incorporates **three silicon chips** mounted on top of one another.
- The bottom chip is covered in heating elements, the middle chip has a long, winding channel carved in it, and the top chip features a small nozzle etched above the end of the channel
- Gas flows through the winding channel, gaining kinetic energy as it contacts the heating elements.
- The energy added to the gas molecules causes their speed to increase as they reach the end of the channel and exit through the nozzle



Electric MEMs:

- TU Dresden is developing a highly miniaturized Field Emission Electric Propulsion thruster, called NanoFEEP with the aim to enable attitude and orbit control of small satellites like CubeSats.
- The highly miniaturized thruster has a volume of less than 3 cubic cms, weighs less than 6 g and is capable of generating thrusts around 8 μN continuously with peaks up to 22 μN.
- Porous tungsten needle emitters and the metal Gallium as propellant are used.
- The thruster design is used in the electric propulsion system on a CubeSat to enable orbit and two axis attitude control.



Application of MEMS:

- **Satellite station keeping operations**
- **Drag Compensation of large spacecraft**
- **Small Δv corrections**
- **orbit manouvers**
- **Attitude/orientation changes**

Propellantless Propulsion:

Electric and nuclear propulsion mostly use hydrogen or xenon gas stored on board. However, the propellant gas need not be carried on board. The propellantless propulsion vehicle design collects neutral gas from atmosphere close to a planet (like earth or Mars) and then utilizes it as propellant. In case of LEO orbit, oxygen is predominantly available.

Interstellar Ramjet: Interstellar ramjet features propellantless propulsion design. The concept is to collect the interstellar hydrogen gas available in galaxies, using it as fuel for fusion reactor and create thrust. It is estimated that collecting area needs to be about 10,000km^2 for an acceleration of about 10m/s^2 (equal to earth's gravitational acceleration).

The ramjet variant of a fusion rocket is capable of reasonable interstellar travel, using enormous electromagnetic fields (ranging from kilometers to many thousands of kilometers in diameter) as a ram scoop to collect and compress hydrogen from the interstellar medium. High speeds force the reactive mass into a progressively constricted magnetic field, compressing it until thermonuclear fusion occurs. The magnetic field then directs the energy as rocket exhaust opposite to the intended direction of travel, thereby accelerating the vessel.

The interstellar vehicle must be a propellantless design and can feature any of the following concepts:

Photon Rocket:

- A very simple concept is to directly convert electrical energy in to kinetic energy via the use of a laser. Photons are then used as a propellant producing thrust.
- Since the energy stored in a laser is proportional to the frequency of emitted light f and plank's constant h, W = h.f
- Photon velocity is speed of light c, thrust generated by photons of the laser beam will be F=P/c or F= (h.f.R)/c, where P is the power and R is repetition rate.
- Specific impulse I_{sp} =c/g
- This is a propulsion concept with no limits on achievable Δv and very high thrust/weight ratio.

Solar Sails:It is a very popular propellantless propulsion concept. The pressure from solar photons is collected using a very large very sail to produce thrust. The solar pressure is around

9 N/km^2 near Eart's LEO and decreases with distance from Sun. Therefore, for interplanetary missions, very large sail is required. Some design features of solar sail are

- In addition to main sail, smaller steering sails are needed for attitude control
- Deployment mechanism and mass of sails are very important.
- Solar sails can be used for interplanetary missions or for going out of solar system
- Solar sail concept can be combined with photon rocket concept

Space Tethers:

Space tethers are long cables which can be used for propulsion, momentum exchange, stabilization and altitude control, or maintaining the relative positions of components of large dispersed satellite. A space tether is a long cable used to couple spacecraft to each other or to other masses, such as a spent booster rocket, space station, or an asteroid. Space tether cables are usually made of thin strands of high-strength fibers or conducting wires.

The tether can provide a mechanical connection between two space objects that enables the **transfer of energy and momentum from one object to the other**, and as a result they can be used to provide space propulsion without consuming propellant. Additionally, conductive space tethers can interact with the Earth's magnetic field and ionospheric plasma to generate thrust or drag forces without expending propellant.

Four main techniques for space tethers are in development:

- **Momentum exchange Tethers**: Momentum exchange tethers allow momentum and energy to be transferred between objects in space, enabling a tether system to toss spacecraft from one orbit to another. They can be used for orbital maneuvering, or as part of a planetary-surface-to-orbit / orbit-to-escape-velocity space transportation system. These can be either rotating tethers, or non-rotating tethers, that capture an arriving spacecraft and then release it at a later time into a different orbit with a different velocity.

- **Tethered formation flying:**This is typically a non-conductive tether formation that accurately maintains a set distance between multiple space vehicles flying in formation.

- **Electro-dynamic tethers**: The tethers interact with the Earth's magnetosphere to generate power or propulsion without consuming propellant.

- **Electric Sail**: A form of solar wind sail with electrically charged tethers that will be pushed by the momentum of solar wind ions.

Electric Sail consists of number of long thin and conducting tethers which are kept in a **high positive potential** by an onboard electric gun.

The positively charged tethers deflect solar wind protons, extracting momentum from them. (The charged particles from Sun are moving at speeds ranging between 250-750 Km/sec.)

They also attract electrons from solar wind plasma producing electron current.

A full sized sail would have 50-100 straightened tethers with a length of 20 km each. Applications of Electric Sail:

- **Fast missions of > 50 km/s or 10 AU/year with modest payload**
- As inter-stellar mission power source
- Missions to study Sun at closer distance
- Two-way missions to inner solar system

Magnetic Sails:

Solar wind travels at high speeds of 300-800km/s. Since solar wind consists of charged particles, a magnetic dipole can deflect the solar wind and create thrust.

Solar wind is a continuous stream of plasma that flows outwards from the Sun, containing several million protons and electrons flowing at high speeds. Solar wind has low densities, very high magnetic fields are required.

Presently, two concepts are under development; Magnetic Sail, where a super conductor ring of several km diameter creates a large magnetic field. This would require cooling of this superconductor and the structure will also be heavy.

A magnetic sail is method of spacecraft propulsion which uses a static magnetic field to deflect charged particles radiated by Sun as a solar wind. The deflected charged particles impart momentum to accelerate the spacecraft.

Magnetic sail can also be deployed in planetary and solar magnetospheres.

Breakthrough Propulsion: Available concepts of propulsion systems are not adequate to achieve space travel to other solar systems in reasonable time frames. With speeds of known concepts, it will take about 4.3 light years to reach nearest star to solar system, Alpha Centauri.

The fundamental limitations governing the technology are the inability to create energy and the inability to go beyond speed of light. The program seeks to link gravitation with electro magnetism.

Breakthrough propulsion research seeks to explore avenues for creating a propulsion system that require no propellant, needs as little power as possible and be able reach fastest speeds.

Beamed Energy Propulsion:

Rockets are momentum machines; they use mass ejected from the rocket to provide momentum to the rocket. So rockets generally attempt to put as much velocity into their working mass as possible

In order to accelerate the working mass, energy is required. In a conventional rocket, the fuel is chemically combined to provide the energy, and the resulting fuel products, the exhaust, are used as the working mass. There is no particular reason why the same fuel has to be used for both energy and momentum.

Rockets can, however, carry their working mass and use some other source of energy.

Electrical thrusters (rockets) carry separate energy source and working mass.

Conventional chemical rockets use **propellant containing both source of energy and working mass** creating momentum change. **Electrical and nuclear rockets use separate source of energy** (electromagnetic fields or nuclear reactor) **and working mass** (inert gas or solid fuel block like teflon)

Beamed energy propulsion uses energy beamed through laser from external source and working mass carried in the vehicle. The propellant-less propulsion vehicles are designed to **collect neutral gas from atmosphere close to a planet** (like earth or Mars) and then utilize it as propellant.

Beamed energy propulsion uses energy from a source external to the spacecraft. If a nuclear reactor is left on the ground and its energy transmitted to the spacecraft, the weight of the reactor is removed as well. The issue then is to get the energy into the spacecraft. **This is the idea behind beamed power**.

With beamed propulsion, the power-source is stationary on the ground, directly heats propellant on the spacecraft with a laser beam from a fixed installation. This permits the spacecraft to obtain energy from external power-source, saving significant amounts of mass, greatly improving performance.

Solar/Laser/Microwave Thermal Propulsion: Beamed energy, external to the vehicle, is used for heating. The external beamed energy may be earth-based or space-based.

The energy is concentrated directly heating the propellant, which is expelled through a conventional nozzle. Alternatively, a reflector is used to collect and concentrate sunlight/laser/microwave energy on to the propellant held in the chamber of the thruster.

Beamed energy propulsion uses Sunlight or hydrogen as propellant. **Specific impulses of 800-1200 sec** and **thrust levels of several hundred mN** are possible.

This concept is under development for raising the communications satellite from LEO to GEO in about 20 days.

Advanced Propulsion Concepts (summary):

Micro-Propulsion system provides extremely small and precise thrust for a variety of satellite missions.
Formation flying and precise attitude control are examples where thrust levels in the micro- to **milli-Newton range** are required.
The micro-propulsion system contains the thruster module which is a silicon wafer stack with four complete rocket engines with integrated flow control valves, filters, and heaters.
Extremely small heaters are located inside the thrust chamber to improve the specific impulse and hence efficient use of the propellant.

Micro-propulsion systems use Micro Electro-Mechanical Systems(MEMS) technology.MEMS technology involves creating very small components, fabricated in the form of silicon chips.Chips can be bonded together, allowing nozzles, heaters, valves, filters, and controls to be sandwiched into very compact units.

MEMS Types: Chemical MEMS include, MEMS based cold gas propulsion and Micro-monopropellant rocket engine

MEMS **Electrics Propulsion Units** include, Ion Engines, Hall Thrusters, Pulsed Plasma Thrusters (PPT) and Field Emitted Electric Propulsion (FEEP)

Chemical MEMS: A cold gas system, either a conventional one or one using MEMS components, comprises a gas storage tank, an isolation valve, a filter, a pressure transducer, a fill and drain valve, a pressure regulator, a pressure relief valve, a flow control valve and nozzles.

The simplest chemical propulsion system is the cold gas system, where cold gas is expelled through a nozzle. The gas velocity, and hence the I_{sp}, depend on the propellant used, the temperature of the propellant, and the shape of the nozzle. In a cold gas system an inert gas is commonly used, e.g. nitrogen.

Chemical MEMS use:

- MEMS based cold gas propulsion
- Micro-monopropellant rocket engine: The propellant in this engine is hydrogen peroxide. As the propellant passes through the catalytic reaction chamber, it is reduced to oxygen and water, causing a large temperature increase, and increased flow velocity. Force is obtained when the reactants are lead through a nozzle. The I_{sp} of this engine should be around 150 s.

MEMS Electric propulsion system: Electric propulsion is the collective name of several different propulsion methods. In electric propulsion thrust is obtained through positively charged ions expelled from the spacecraft. (Negatively charged ions, or electrons, must also be expelled, to avoid charging of the spacecraft.).

The propellant in PPT devices is commonly Teflon. A local plasma is achieved by a small electric discharge. The resulting ions are accelerated to increase the thrust.

Both the Hall and the ion propulsion system commonly use xenon as a propellant.

MEMS Resisto-jet Engine:

Description: One type of MEMS thruster is a resisto-jet which works by heating gas molecules to increase their energy before expelling them through a nozzle.
The MEMS resisto-jet incorporates three silicon chips mounted on top of one another. The bottom chip is covered in heating elements, the middle chip has a long, winding channel carved in it, and the top chip features a small nozzle etched above the end of the channel .
Gas flows through the winding channel, gaining kinetic energy as it contacts the heating elements. The energy added to the gas molecules causes their speed to increase as they reach the end of the channel and exit through the nozzle.

Application of MEMS:

- Satellite station keeping operations
- Drag Compensation of large spacecraft
- Small Δv corrections
- orbit maneuvers
- Attitude/Inclination changes

UNIT V LAUNCH VEHICLES

Geysering Effect & Water Hammering in Cryogenic Engines: Geysering problem arises in the design of propellant feed systems using long lines to connect the propellant tank to the engine. Cryogenic propellants are always stored in large tanks under specified low temperature conditions and are transferred to the rocket engines prior to the launch. Geysering can be described as the rapid expulsion of a boiling liquid and its vapor from the feed line. The atmosphere heats up the propellant in the feed line, establishing a fluid circulation within the feed line. The heated fluid rises and cool fluid descends in the tube. This circulation increases the bubble formation of evaporated fumes and the bubbles combine to become bigger in size.

Formation of larger bubbles cause low pressure regions below the bubbles, causing more evaporation. This process becomes self-sustaining and the vapor formation rate increases faster than it can be removed from the feed line. This leads to explosive expulsion of liquid from the tube. This effect is called geysering effect of cryogenic propellants.

As the refueling continues, the tube is refilled with propellant from the reservoir, interaction between the vapor bubbles causes high impact loads or "water hammer" forces that may cause structural damage to the feed system. This will create safety hazard to the launch vehicle and launch site.

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 with in 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.

Applications of Rocket Propulsion Systems: Basic application of Rocket propulsion systems are

- Space Launch Vehicles
- Spacecraft
- Missiles
- Other applications

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.

Role: A launch vehicle or carrier rocket is a rocket used to carry a payload from the Earth's surface into outer surface. A launch system includes the launch vehicle, payload, the launch pad and other infrastructure.

It provides the spacecraft with payload, with adequate velocity to overcome the gravitational forces, atmospheric drag and reach the intended orbit or flight path to reach destination in space.

It consists of a series of rocket/stages that ignite, burnout providing the required thrust. The stages are jettisoned after burnout, in a multi stage launch vehicle.

Usually, the first or lowest stage, often called booster stage, is the largest and it requires large thrust and largest total impulse. All stages need chemical propulsion to achieve the desired thrust-to weight ratio.

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

The thrust levels become smaller with subsequent stages and the vehicle weight becomes progressively lighter. The thrust magnitudes depend on the mass of the vehicle, which in turn depend on the mass of the payload and mission.

Many heavy launch vehicles have two to six strap-on solid propellant motor boosters mounted on the first stage of the vehicle. Boosters operate simultaneously with the first stage and after burn-out, separate from the first stage and dropped off before completion of first stage propulsive operation or burn-out.

For certain missions, the launch vehicle can't deliver a spacecraft to its final orbit by itself. Instead, when the launch vehicle finishes its job, it leaves the spacecraft in a parking orbit.

- A **parking orbit** is a temporary orbit where the spacecraft stays until transferring to its final mission orbit. After the spacecraft is in its parking orbit, a final "kick" sends it into a transfer orbit.
- A **transfer orbit** is an intermediate orbit that takes the spacecraft from its parking orbit to its final, mission orbit

Types of Launch Vehicles(Classification): They are classified based on

1. **Usage**:

- Expendable launch vehicles are designed for one-time use. They usually separate from their payload, and may break up during atmospheric re-entry.
- Reusable launch vehicle, on the other hand, are designed to be recovered intact and used again for subsequent launches.
- For orbital space flights, the Space Shuttle was the only launch vehicle with components which have been used for multiple flights.
- Non Rocket Space launch alternatives are at the planning stage.

2. **Type of Launch Platform**: Launch vehicle can be launched from land (ICBM), sea (mobile platform, from submarine SLBMs); or air (AAMs- Pegasus)
3. **Type of Propellant used:** Storable, cryogenic, solid, liquid or hybrid propellants
4. **Number of Stages**: Single stage, multi-stage
5. **Size/mass of payload; By size**

   - A Sounding rocket cannot reach orbit and is only capable of sub-orbital spaceflight
   - A Small lift launch vehicle is capable of lifting up to 2,000kg *(4,400lbs)* of payload into low earth orbit (LEO)
   - A Medium lift launch vehicle is capable of lofting between 2,000 to 20,000kg *(4,400 to 44,000lbs)* of payload into LEO.
   - A Heavy lift launch vehicle is capable of lofting between 20,000 to 50,000kg *(44,000 to 110,200lbs)* of payload into LEO.
   - A Super heavy lift vehicle is capable of lofting more than 50,000kg *(110,200lbs+)* of payload into LEO.

6. **Manned or unmanned vehicles**
7. **Specific space objective:** Earth-orbit/moon-landing/inter-planetary/inter-stellar or Deep space missions

Indian Space Launch Vehicles: These include SLV, ASLV, PSLV & GSLV. All launched from Sriharikota, AP.

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.

Missiles/Rockets: Military applications of launch vehicles: Launch vehicles are used for military purposes as

1. **Rockets**: A launch vehicle with an explosive warhead, usually utilized for short ranges.
2. **Missiles**: Missiles are rockets with guidance & control systems, usually used for long range applications.

   Missiles can be classified based on:

1. **Range: Strategic**(Long range ballistic missiles); **Tactical** (short range targets as local support to ground forces)
2. **Launch Platform: Ground/surface launched; ocean/ship launched; Underneath Sea (submarine) launched**
3. **Type of propellant used: Solid/Liquid** or **Combined Cycle Engines Type of Usage: Surface-to-air; Air-to surface; Air-to-air**

Other applications of Rocket Propulsion systems: : Other applications are the secondary applications which include

- Attitude control
- Stage separation
- Orbital changes
- Spin control

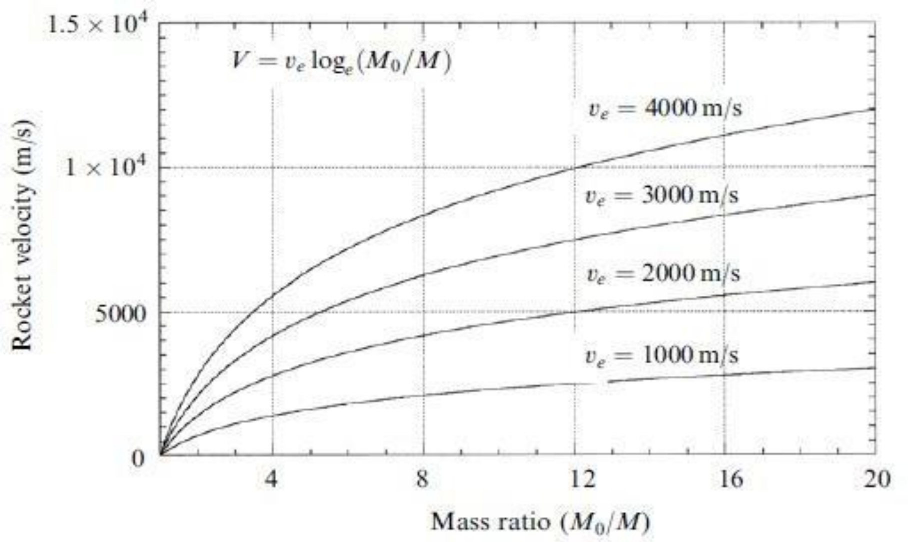
- Settling of liquids in tanks
- Target drones
- Underwater rockets like torpedoes
- Research Rockets

Multi-staging:

The rocket velocity V (called incremental velocity) is given by

$V = v_e \ln \left(\frac{M_o}{M}\right)$, where M_o is initial mass of the rocket, M is the current mass; v_e is the effective exhaust velocity of the propellant combustion prosducts.

The vehicle velocity is therefore dependent on natural logarithm of the ratio of initial mass to current mass and the exhaust velocity. The mass ratio is defined as the ratio of initial mass to current mass.



the rocket velocity is plotted against the mass ratio in the above graph.

The rocket equation shows that the rocket velocity depends only on two factors, the mass ratio and the exhaust velocity. The size of the rocket, time of burn or any other factor has no influence on the rocket velocity.

The exhaust velocity of chemical propellants is limited to 4500 m/s. The only way to increase the vehicle velocity is to increase the mass ratio.

To achieve a high rocket velocity, the mass ratio has to be high. For example, a mass ratio of 5 implies thar 80% of initial mass of rocket is the fuel. So most of the weight of a rocket is fuel leaving little scope for othe structure, control and guidence equipment and payload.

A Boeing 747 aircraft weighs 400 T with 70 T fuel; mass ratio of plus 1. A Saturn V rocket used for taking 8 men for 8 day space trip, weighs 3000 T, most of the vehicle carries fuel.

The volocity increment needed for launch:

There is a distinction between the velocity increment and the actual velocity of the vhicle. The velocity increment calculated from the rocket equation is a measure of energy expended by the rocket and it includes the velocity to overcome gravity, velocity needed to reach the orbital altitude and the vehicle velocity. Therefore, the vehicle velocity is less than the incremental velocity ΔV. The actual vehicle velocity in 500 km circular Earth orbit is 7.6 km/s, whereas the velocity increment is 8.7 km/s. the difference is the energy expendd by the vehicle in overcoming the gravitational and drag forces.

It can be seen from the graph above that the rocket vehicle can travel faster than the speed of its exhaust. The constant accelerating force on the walls of the combustion chamber and inside the nozzle and the high mass ratios make it possible to achieve higher veocity increments.

A single stage chemical rocket using a liquid propellant combination of pure hydrogen and oxygen with exhaust velocity of 4 km/sec, can achieve the escape velocity of 11 km/sec with a high mass ration of 14, which is very difficult.

The escape velocity of chemical rockets can be increased by multi-staging, dividing the propellant mass M_f between the stages. The exhaust velocity may be treated as constant. Multi-staging effectively increases the mass ratio of overall vehicle.

In a two stage rocket, the first rocket is ignited and all propellant burnt, and the first stage structure gets discarded. The second stage starts burning after discarding the first stage. To compare the performance of single stag and two stage rockets, the single stage rocket will have incremental velocity of

$V_0 = v_e \ln R_0$, where R_0 is the mass ratio of single stage vehicle.

The two stage rocket will have

$V = v_e \ln R_1 + v_e \ln R_2$ where R_1 and R_2 are the mass ratios of fist and second stages. It can be shown as

$$V > V_0$$

If we assume rocket total mass of 100 T include 10 T structure mass and 1 T payload. Exhaust velocity v_e is assumed as 2700 m/s.

The velocity of single stage vehicle will be

$$V_0 = 2700 \ln \frac{10+89+1}{10+1} = 5.959 \text{ km/s}$$

In case the above rocket is split in to two stages carrying equal amount of propellant and structure mass, then the total velocity of both stages will be,

$$V_1 = 2700 \ln \frac{10+89+1}{10+44.5+1} = 1.590 \text{ km/s}$$

The first stage is discarded with half propellant burnt and half structure mass. Velocity of second stage is

$$V_2 = 2700 \ln \frac{5+44.5+1}{5+1} = 5.752 \text{ km/s}$$

Total velocity achieved by the two stage vehicle is V = 1.590 + 5.752 = 7.342 which is higher than the single stage vehicle. This is achieved by not burning more fuel.

Similarly, if the same rocket is built into three equal stages, a velocity of 8.092 km/s can be achieved.

The multi-stage concept is depicted below:

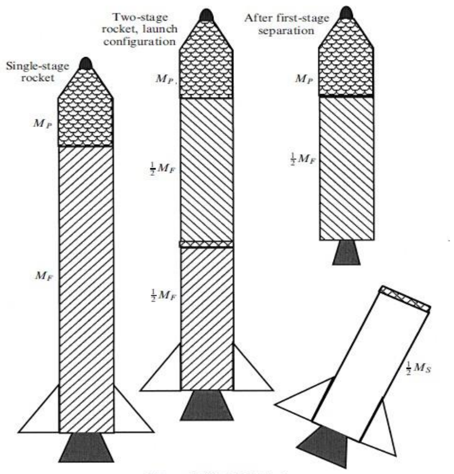
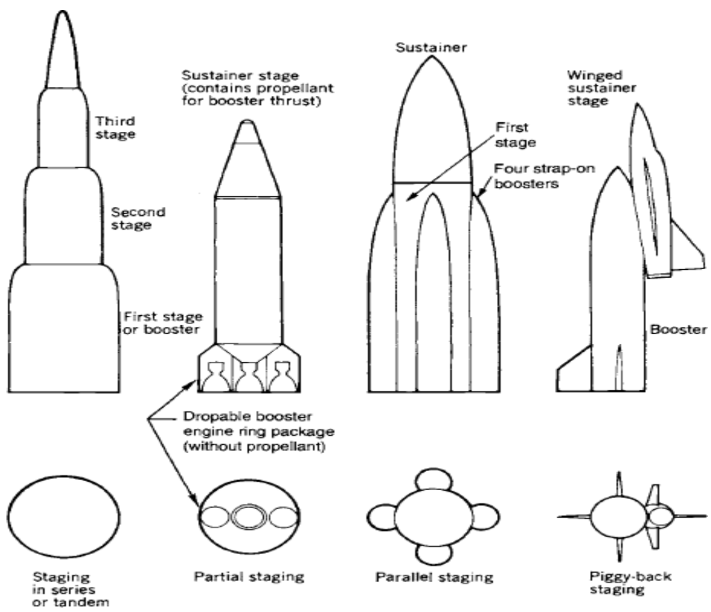


Figure 1.10. Multistaging.

Geometric configurations of multi-stage launch vehicles:



The first common configuration is the stages stacked vertically on top of each other, as in Minuteman long range missile or the Delta launch vehicle.

The second version is the “Partial staging”, which allows all engines to be started simultaneously, on ground. This avoids starting higher stages in air.

The third configuration allows separate strap-on booster stages are attached to the first stage rocket in a vertical configuration. This allows increased initial performance.

The last configuration is the piggy-back version used in space shuttle. The booster is expended after use, while the winged sustainer, space shuttle, is recovered and re-used.

Single Stage To Orbit (SSTO):

- Trans-atmospheric Vehicles (TAV) are single-stage, manned, air-breathing, winged crafts. TAVs are meant as **single stage to orbit(SSTO)** craft.
- The **SSTO** vehicle reaches orbit without jettisoning hardware, expending only propellants and fluids.
- These vehicles flying at trans-atmospheric altitudes and very high speeds encounter extremely high temperatures and pressures.

Advantages:

1. Avoids the costs and complexities of multi-staging
2. Improved reliability due to simple structures and fewer components
3. Most SSTOs are recoverable and reusable
4. Allows more extensive use of space environment

Limitations:

1. Smaller payloads
2. Needs high performance propellants
3. Lower structural mass needs modern materials
4. Key engineering challenges of high mass ratio and high exhaust velocity

Practical Approaches to SSTO: Key Challenges - High exhaust velocities and High mass ratios

- **High Mass Ratio**: Involves reducing structural mass for given mass of propellant.
- **OneEnginefor all altitudes**: Reusable vehicle need advanced features like variable shape of nozzles with altitude correction; use of plug nozzle; use of aero-spike nozzle; rotary rocket engine etc
- **Air-breathing SSTO- Use of Scramjets:** Ramjets using supersonic combustion forms the air-breathing constituent of an integrated SSTO vehicle.
- **Combined cycle engines** like integral Ram/scram/rockets are used

High Mass Ratio-**Reducing mass of structure**:

1. **Propellant tanks**: For lowest mass of tanks, materials with ratio of Young's modulus to density are preferred;
   - Double skinned honeycomb walls result in weight reduction;
   - Composites like carbon fiber composites are preferable
2. **Liquid Oxygen**: Preferable for use, but LOX need metallic tanks like aluminum-lithium alloys. Carbon fiber reinforced carbon compatibility with LOX is being studied
3. **Composite Tanks**: Leading to substantial mass reduction and favourable mass ratios

Engines: Few improvements are:

- **Use of Bell-shaped nozzles**: We have high divergence angle immediately behind throat section; and the contour reverses causing low divergence at nozzle exit. Use of extendable bell sections is also a solution
- **Plug Nozzle**: The performance of this nozzle is independent of altitude. The nozzle has a conical plug at the center. Thrust is developed as the exhaust stream flows along the outer surface of the conical plug. The exhaust stream emerges from an annular aperture between the plug and the combustion chamber wall. The outer boundary of the exhaust stream is the slipstream.

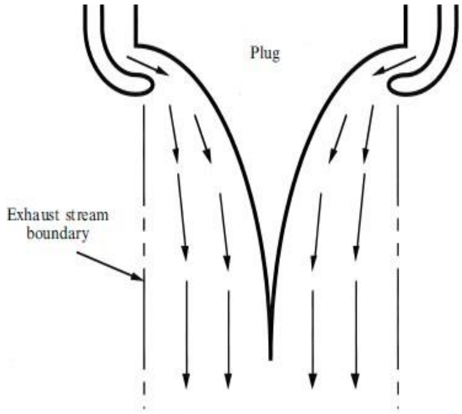


Figure 8.8. Principle of the plug nozzle.

- **Altitude Adapting Designs**: One possible solution would be to use an aerospike engine, which can be effective in a wide range of ambient pressures
- **The Aerospike Nozzle:** The aerospike nozzle replaces the long central plug with a truncated cone. Cool gas is circulated in to the exhaust stream from the face of the cone.
- **Use of Tri-propellants**: Uses conventional hydrocarbon as fuel and additionally hydrogen is used as a working fluid. This arrangement reduces the molecular weight of the exhaust gases.
- **SCRAMjet Engine**: Use of supersonic combustion ram engine as an air-breathing engine in combination of a combined cycle arrangement with rocket engine.

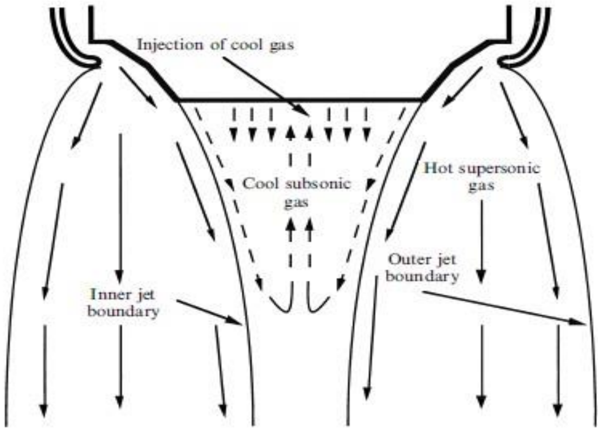


Figure 8.10. Principle of the aerospike nozzle.

Engines improvements-

The Rotary Rocket Engine:

- Concept merges a launch vehicle with a helicopter
- Spinning rotor blades, powered by tip jets, would lift the vehicle in the earliest stage of launch. Once the air density thinned to the point that helicopter flight was impractical, the vehicle would continue its ascent on pure rocket power, with the rotor acting as a giant turbo-pump.

Rotary Rocket Engine

- The rocket engine comprises of a number of combustion chambers mounted around the rim of a wheel.
- Radial pipes connect the chambers to a central propellant tank.
- Propellant is delivered to the combustion chambers as the wheel rotates; under action of centrifugal force.
- Achieves high propellant delivery rates are achieved with no pumps
- The rotary rocket engine is a fully reusable single-stage-to-orbit (SSTO) manned spacecraft

Missions & Mission profile:

Each space launch vehicle has a specific space flight objective or mission, such as an earth-orbit or a moon landing or Mars orbit. The design of the propulsion system of the launch vehicle is tailored to fit a specific application, which is termed as “mission requirement”. The mission requirement is usually stated in terms of

- application (anti-aircraft rocket, upper stage of launch vehicle, deep-space probe or crew transfer for space station)
- mission velocity needed to reach desired orbit

- flight trajectories required (surface launch, orbit transfer)
- minimum life of flight vehicle (during orbit)

Mission requirements are translated in to rocket engine requirements in terms of thrust required, propellants used, number of re-starts, number of thrust chambers, number of stages and flight envelope. Key design features like chamber pressure, type of feed system, method of cooling, thrust variation also depend on mission requirements. Thrust sizes vary between fraction of a N to several KNs.

Missions for space launch vehicles can be many, like

- **Military missions** (launching reconnaissance satellites, Command & control satellites)
- Non-military (launching weather observation satellites, Geo-positioning satellites)
- Space exploration (Space environment, planetary missions)
- Commercial (Communication, scientific experimental or adventure)

Mission applications of solid/liquid rocket propulsion systems in launch vehicles include large booster and higher stage motors, strategic/tactical rockets and missiles (ballistic/cruise).

Mission applications of electrical propulsion systems include precise low-thrust station keeping, attitude control, orbital manoeuvres and deep space missions with high specific impulse.

Nuclear propulsion systems are best suited for manned planetary exploration missions.

Launch vehicle mission profile: The launch vehicle goes through five distict mission phases. They are

- Launch phase: From lift-off to spacecraft separation
- Transfer orbit phase: From spacecraft separation till orbit maneuver firing
- Phasing orbit phase: From the first orbit maneuver firing till the final orbit maneuver firing
- Trim orbit phase: From final orbit maneuver firing till station acquisition
- On-station orbit phase: From station acquisition till completion of spacecraft life.

Some successful launch vehicles:

USA:

1. Atlas V: It is the fifth major version of the Atlas rocket family currently in use. A joint venture by Lockheed and Boeing. Used to carry lunar orbiters. Vehicle mass 590 T; payload: 4,750-8,900 kg to GTO (Geosynchronous transfer orbit). Two stages with five booster strap-on rockets.
2. Delta IV: A heavy lift launch vehicle, world's second highest capacity launch vehicle. Developed by McDonnel Douglas. It is an all liquid fuelled vehicle, consisting of one upper stage, one booster stage and two strap on boosters. Payload of 14,220 kg to GTO.
3. Titan IV: A heavy lift launch vehicle, used for placing satellites in low and medium earth orbits. developed by Lockheed Martin, recently retired.
4. Peagasus: Belongs to the "air launch to orbit" family. It is an air launched rocket system developed by Northrop Grumman Corp, capable of carrying small payloads of up to 450 kg, in to LEO. Consists of three solid propellant stages. Can be launched at a height of 12 km from B-52 aircraft.

Russia:

1. Soyaz: Soyaz is a family of expendable launch vehicles developed by Progress Rocket Space Center, Russia. With over 1700 flights since 1966, Soyaz is the most frequently used launch vehicle in the world. It is three stage vehicle, first stage has four liquid propellant strap on rockets, second is a single liquid propellant rocket. The third stage houses the Soyaz spacecraft designed to carry passengers dock with International Space Station (ISS).
   With the ending of US Space Shutle program, Soyaz rockets become the only launch vehicles able to transport astronauts to the International Space Station.
2. Proton-M: A heavy lift vehicle, with a payload capacity of 6800 kg to GTO. Consists of three stages, all powered by liquid propellant (hypergolic combination). The first stage is unique employing engines that can swivel tangentially up to 7° for providing TVC.
3. Rus-M: Human-rated orbital launch vehicle under development, to replace Soyaz.

China:

1. Long March Rocket family: Developed by China Academy of Launch Vehicle Technology, PRC, has a payload capability of 5,500 kg till GTO. Long March 4 version is a three stage rocket, using liquid propellants. Has launched earth-orbiting spacecraft and also lunar orbiters.
2. Hyperbola 3: Launch vehicle (3 stages using liquid fuels) with up to 500 kg payload in to LEO.

India:

1. GSLV Mk III: Lift off weight 640 t. Uses three stages, with solid strap-ons, liquid fuelled second stage and cryogenic fuel foe upper stage with payload capacity of 4000 kg till GTO. Can launch 4 to class communications satellites.

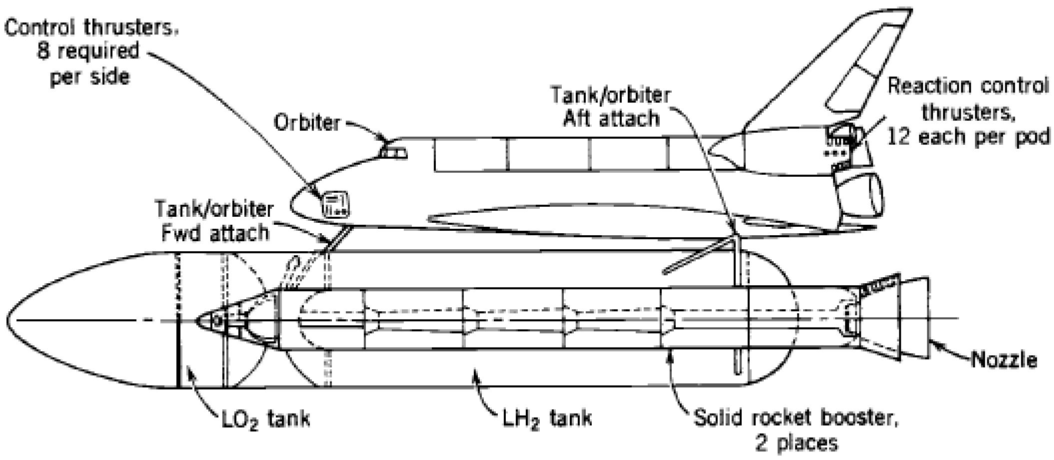
2. RLV-TD: Reusable launch vehicle-Technology demonstrator is under development. It is a fully re-usable launch vehicle-winged aircraft combination. A hypersonic vehicle, using two stage launch vehicle.
3. Scramjet-TD: Air breathing hypersonic scramjet technology demonstrator flight trials are in progress.

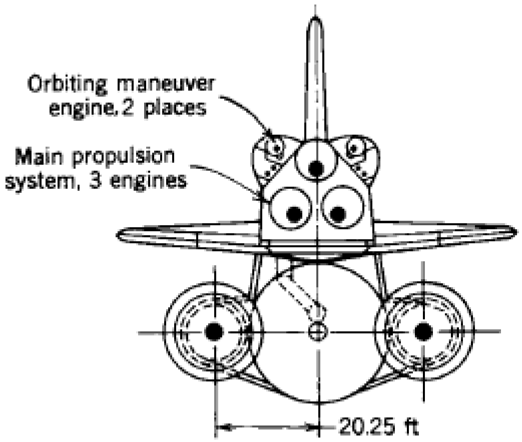
Space shuttle Main Engine (SSME): The space shuttle main engine is a liquid fuel cryogenic rocket engine. The engine is named RS-25. NASA intends to continue using the engine on the space shuttle successor, the Space launch system (SLS), which is a US super heavy lift expendable launch vehicle, being developed for lunar crew flights and in future, human missions to Mars.

The SSME is manufactured by Rocketdyne and Pratt & Whitney. It uses LOX/LH combination. Developing 1,860 kN thrust at SL.

The space shuttle uses a cluster of three RS-25 engines and two solid rocket boosters.

The shuttle components include the Orbiter Vehicle (OV) with three RS-25 main engines, a pair of recoverable solid rocket boosters (SRBs) and the expendable external tank (ET) containing LOX and LH. The sopace is launched vertically like a rocket, with the two SRBs operating in parallel with the OV's three MSMEs. The ET provides fuel to the MSMEs. The SRBs are jettisoned before the vehicle reaches LEO and the ET is jettisoned just before orbit insertion. The orbiter is provided with two orbital maneuvering system (OMS) engines, which are used to de-orbit and re-enter the atmosphere. The orbiter then glides like a spaceplane to the runway landing facility.





The space orbiter is a delta-winged reusable cargo space vehicle that takes off vertically and lands horizontally like a glider. Each shuttle orbiter was designed for minimum 100 missions; carrying up to 30,000 kg load to LEO and can return up to 11000 kg load to earth.

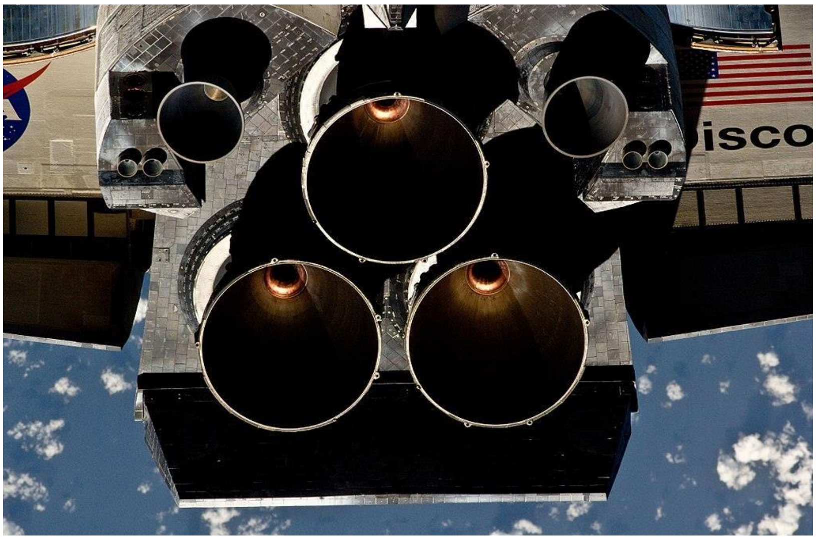
SSME uses staged combustion cycle (or pre-burner cycle) which is power cycle used with bi-propellants. In the staged combustion cycle, the propellant flows through multiple combustion chambers, and is thus combusted in stages. The main advantage is that it provides high fuel efficiency measured through increased specific impulse, while it also increases the engineering complexity.

Typically, the propellant flows through two combustion chambers, the first is called preburner and the second is called the main combustion chamber. In the preburner, a small portion is propellant is combusted (burnt), and the over-pressure produced is used to drive the turbo pumps, that feed the engine with propellant. In the main combustion chamber, the propellants are combusted completely to produce thrust.

The SSME uses two separate preburners, each mounted on a separate turbopump. One preburner burns all fuel with part of oxidizer while the other burns all oxidizer with part of the fuel. Complete combustion is achieved in the main combustion chamber.

SSME uses LOX/LH liquid propellant combination.

SSME with two orbit maneuvering pods:



Guidance and control requirements:

Primary function of guidance in a LV is to steer the vehicle along an optimal path, satisfying the constraints on the trajectory. A variety of guidance schemes are adopted to cater for different launch vehicle missions.

Launch Vehicles (LVs) could be conventional expendable launch vehicles (ELVs), advanced launch vehicles like reusable launch vehicles (RLVs) or air breathing hypersonic launch vehicles (ABLVs). These vehicles have typical flight paths depending on their mission objectives.

Role of guidance in launch missions is to generate steering commands for guiding the vehicle along an optimal path satisfying the trajectory. The guidance technique must know the present position, direction and magnitude of motion with respect to reference points for determining the changes in motion required to achieve desired result. The steering commands are generated to correct deviations.

Basin inputs for guidance design are:

- Mission requirements: Typical requirement will be injecting payload into low earth orbit (LEO) or geo-stationary earth orbit (GEO)
- Constraints to be met: Typically specifications like the trajectory, mission and vehicle structural integrity etc
- Guidence specifications: Typical specifications will be orbital parameters

Control refers to the manipulation of forces, by way of steering controls, thrusters etc, needed to execute guidance commands, while maintaining the vehicle stability. Control mechanisms use sensors and detectors to measure the output performance of the process being controlled and use these measurements to provide corrective feedback helping to achieve the desired performance.