R-22

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

MRCET CAMPUS

(AUTONOMOUS INSTITUTION – UGC, GOVT. OF INDIA)

Department of AERONAUTICAL ENGINEERING

SPACE PROPULSION

Prepared by:

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SPACE PROPULSION

B.TECH (R-20 Regulation)
(IV YEAR – I SEM)
(2024-25)

DEPARTMENT AERONAUTCAL ENGINEERING

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

(Autonomous Institution – UGC, Govt. of India)

Recognized under 2(f) and 12 (B) of UGC ACT 1956

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

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

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

- 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.

PROGRAM OUTCOMES

(PO's)

Engineering Graduates will be able to:

- Engineering knowledge: Apply the knowledge of mathematics, science, engineering fundamentals, and an engineering specialization to the solution of complex engineering problems.
- 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.
- 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.
- 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.
- 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.
- 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.
- 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.
- Ethics: Apply ethical principles and commit to professional ethics and

responsibilities and norms of the engineering practice. Individual and team work: Function effectively as an individual, and as a member or leader in diverse teams, and in multidisciplinary settings.

- 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.
- 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.
- 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.

PROGRAM EDUCATIONAL OBJECTIVES – Aeronautical Engineering

- PEO1 (PROFESSIONALISM & CITIZENSHIP): To create and sustain a community of learning in which students acquire knowledge and learn to apply it professionally with due consideration for ethical, ecological and economic issues.
- PEO2 (TECHNICAL ACCOMPLISHMENTS): To provide knowledge based services to satisfy the needs of society and the industry by providing hands on experience in various technologies in core field.
- PEO3 (INVENTION, INNOVATION AND CREATIVITY): To make the students to design, experiment, analyze, and interpret in the core field with the help of other multi disciplinary concepts wherever applicable.
- PEO4 (PROFESSIONAL DEVELOPMENT): To educate the students to disseminate research findings with good soft skills and become a successful entrepreneur.
- PEO5 (HUMAN RESOURCE DEVELOPMENT): To graduate the students in building national capabilities in technology, education and research

PROGRAM SPECIFIC OUTCOMES – Aeronautical Engineering

- To mould students to become a professional with all necessary skills, personality and sound knowledge in basic and advance technological areas.
- To promote understanding of concepts and develop ability in design manufacture and maintenance of aircraft, aerospace vehicles and associated equipment and develop application capability of the concepts sciences to engineering design and processes.
- Understanding the current scenario in the field of aeronautics and acquire ability to apply knowledge of engineering, science and mathematics to design and conduct experiments in the field of Aeronautical Engineering.
- 4. To develop leadership skills in our students necessary to shape the social, intellectual, business and technical worlds.

INDEX

MALLA REDDY COLLEGE OF ENGINEERING AND TECHNOLOGY

IV Year B.Tech. ANE- I Sem **L/T/P/C**

2/1/-/3

(R20A2115) Space Propulsion

Objectives:

1. Students acquire knowledge about the Air-Breathing Propulsion.
2. Students can focus on various chemical-Rocket propulsionand understand the future scenario.
3. To provide an exposure of Nuclear Rocket Propulsion and its utilities.
4. Students get the knowledge of Advanced Rocket Propulsion.
5. Students understand about the Launch vehicles and their space mission operation.

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.

Outcomes:

1. Students can correlate all Air-Breathing propulsion.
2. Students will be able to configure the Chemical Rocket Propulsion.
3. Students will be able to understand about the Nuclear Rocket Propulsion [advantages/disadvantages].
4. Students can predict about the Advanced Rocket Propulsion.
5. Students can able to predict Launch vehicles accordingly their missions.

R17

Code No: R17A2108

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

(Autonomous Institution – UGC, Govt. of India)

III B.Tech I Semester Supplementary Examinations, February 2021

Advanced Propulsion Systems

(AE)

Roll No										

Time: 2 hours 30 min **Max. Marks: 70**

Answer Any **Five** Questions

All Questions carries equal marks.

*****

1 What are the problems of combustion in high-speed flow? Explain the working of scram jet engine. **[14M]**

2 Explain the working and operation of Air turbo-rocket (ATR). **[14M]**

3 What are the materials used for Rocket motor casing? Write the advantages of cryogenic engines. **[14M]**

4 Explain the working principle and applications of Solid propellant rocket motors. **[14M]**

5 Explain the following w.r.t. nuclear fission: **[14M]**

a) Thermal stability

b) Prompt and delayed neutrons.

6 Discuss about the safety issues in nuclear propelled missions. **[14M]**

7 Explain about the generation of thrust in electric propulsion system. **[14M]**

8 Explain the following w.r.t. space launch vehicle: **[14M]**

a) Missions

b) Mission profile

c) Staging employed

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R18

Code No: R18A2115

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

(Autonomous Institution – UGC, Govt. of India)

III B.Tech II Semester Supplementary Examinations, January 2024

Aerospace Propulsion Systems

(AE)

Roll No										

Time: 3 hours **Max. Marks: 70**

Note: This question paper Consists of 5 Sections. Answer **FIVE** Questions, Choosing ONE Question from each SECTION and each Question carries 14 marks.

***

SECTION-I

1	(i) Explain the limitations of turbojets and write the need for supersonic combustion in detail.	**[7M]**
	(ii) Discuss the types of nozzles with neat sketches	**[7M]**
	OR	
2	Describe the need, construction, operation and performance of turbo- ramjet and ejector ramjet with neat sketches.	**[14M]**
	SECTION-II	
3	What is meant by thrust vector control? Explain in detail about TVC mechanisms with neat sketches	**[14M]**
	OR	
4	(i) Explain briefly the selection criteria of solid propellants and list out the important hardware components of solid rockets.	**[8M]**
	(ii) Discuss about the cryogenic engines and engine cooling	**[6M]**
	SECTION-III	
5	Explain the following: (i) Sustainable chain reaction, (ii) Neutron leakage and (iii) Thermal stability	**[14M]**
	OR	
6	(i) Define the terms: Nuclear fission, Power, Thrust and Energy.	**[6M]**
	(ii) Describe the nuclear rocket engine design and performance with neat sketches	**[8M]**
	SECTION-IV	
7	Describe in detail the electrostatic and electro-magnetic thrusters with neat sketches	**[14M]**
	OR	
8	Explain the following: (i) Pulsed plasma thrusters (PPT) for micro-spacecraft, (ii) Photon rocket & (iii) Solar sails	**[14M]**
	SECTION-V	
9	Discuss the staging employed, guidance and control requirements in launch vehicles	**[14M]**
	OR	
10	Describe the space shuttle engine and single-stage-to-orbit (SSTO) rocket with neat sketches	**[14M]**

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R17

Code No: R17A2108

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

(Autonomous Institution – UGC, Govt. of India)

III B.Tech I Semester Supplementary Examinations, June 2022

Advanced Propulsion Systems

(AE)

Roll No										

Time: 3 hours **Max. Marks: 70**

Answer Any **Five** Questions

All Questions carries equal marks.

***

1	Explain the types of nozzles with neat sketches.	**[14M]**
2	Explain the working principle of ATR with neat sketch	**[14M]**
3	Explain about the different types of Propellant grain structures with neat diagrams	**[14M]**
4	Explain about the different types of liquid feed systems with neat sketches	**[14M]**
5	Explain about the safety issues in nuclear propelled missions	**[14M]**
6	Explain the difference between nuclear fusion and fission chain reactions with examples.	**[14M]**
7	Explain the working principle of plasma thrusters (PPT) for micro-spacecraft	**[14M]**
8	Define mission profile and explain the different types of mission profiles for launch vehicles	**[14M]**

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R15

Code No: R15A2103

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

(Autonomous Institution – UGC, Govt. of India)

II B.Tech II Semester supplementary Examinations, April/May 2019

Aerospace Propulsion

(AE)

Roll No										

Time: 3 hours **Max. Marks: 75**

Note: This question paper contains two parts A and B
Part A is compulsory which carriers 25 marks and Answer all questions.
Part B Consists of 5 SECTIONS (One SECTION for each UNIT). Answer FIVE Questions, Choosing ONE Question from each SECTION and each Question carries 10 marks.

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PART-A (25 Marks)

1). a Give the Classification of gas turbine engines. **[2M]**
b Write the advantages and disadvantages of turboprop and turbo-shaft engines **[3M]**
c What are the limits of the compressor **[2M]**
d Discuss about Typical turbine blade profile. **[3M]**
e Explain the nozzle expansion process. **[2M]**
f Describe about after burner. **[3M]**
g What are the limitations of ramjet engines at high speeds? **[2M]**
h Write short notes on distributed fuel injection. **[3M]**
i What are the potential applications of nuclear engines? **[2M]**
j Write short notes on thrusters. **[3M]**

PART-B (50 MARKS)

SECTION-I

2 Explain working principle of turbojet engine and discuss about T-S and P-V diagram with station number. **[10M]**

OR

3 A turbojet-powered aircraft cruises at V_0 = 300 m/s while the engine produces an exhaust speed of 600 m/s. The air mass flow rate is 100 kg/s and the fuel mass flow rate is 2.5 kg/s. The fuel heating value is Q_R = 42,000 kJ/kg. Assuming that the nozzle is perfectly expanded, calculate **[10M]**
(a) engine ram drag in kN
(b) engine gross thrust in kN
(c) engine net thrust in kN

SECTION-II

4 Discuss about the Actual and ideal performance of a single stage axial flow compressor. **[10M]**

OR

5 Discuss about the following **[10M]**
a) Axial flow compressor
b) Centrifugal compressor

SECTION-III

6 What are the types of TVC mechanisms and explain the conceptual basis for vectoring mechanisms. **[10M]**

OR

7 Evaluate the combustion chamber performance based on the following conditions or performance: **[10M]**

a. Pressure loss
b. Combustion efficiency
c. Combustion stability
d. Combustion intensity

SECTION-IV

8 Sketch and explain the working of the air turbo-rocket (ATR). Write down its applications and advantages. **[10M]**

OR

9 Describe about scramjet engine-construction, flow process with the help of neat sketches. **[10M]**

SECTION-V

10 Explain about the nuclear rocket engine with neat sketch. **[10M]**

OR

11 Describe the working principle of electrostatic and electro-magnetic thrusters with neat sketch explaining each part and its function. **[10M]**

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Code No: R18A2115 **R18**

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

(Autonomous Institution – UGC, Govt. of India)

III B.Tech II Semester Supplementary Examinations, December 2022

Aerospace Propulsion Systems

(AE)

Roll No										

Time: 3 hours **Max. Marks: 70**

Note: This question paper Consists of 5 Sections. Answer **FIVE** Questions, Choosing ONE Question from each SECTION and each Question carries 14 marks.

***

SECTION-I

1 Draw a schematic diagram of a scramjet engine, label the parts, describe its performance and discuss their advantages and disadvantages **[14M]**

OR

2 (i) Explain the principle of operation of ejector ramjet with a neat sketch **[7M]**
(ii) Discuss the types of nozzles with neat sketches **[7M]**

SECTION-II

3 Explain the pressure feed system of liquid rocket engine with a neat sketch. What are the advantages and limitations of pressure feed system? **[14M]**

OR

4 Briefly explain the cooling techniques in liquid rockets. Discuss the advantages and disadvantages of solid propellant rockets and liquid propellant rockets **[14M]**

SECTION-III

5 (i) Compare nuclear rocket propulsion and chemical propulsion **[7M]**
(ii) Discuss the fundamentals of fission reactions of nuclear rockets **[7M]**

OR

6 Describe the nuclear rocket engine design and performance with neat sketches **[14M]**

SECTION-IV

7 (i) Explain solar and magnetic sails **[7M]**
(ii) Discuss the application of MEMS in advanced rocket propulsion systems **[7M]**

OR

8 Explain the following:
(i) Propellantless propulsion **[5M]**
(ii) Photon rocket **[5M]**
(iii) Beamed energy propulsion **[4M]**

SECTION-V

9 (i) Explain any three successful launch vehicles **[7M]**
(ii) Discuss the guidance and control requirements in launch vehicles **[7M]**

OR

10 Briefly describe the space shuttle engine and single-stage-to-orbit (SSTO) rocket. **[14M]**

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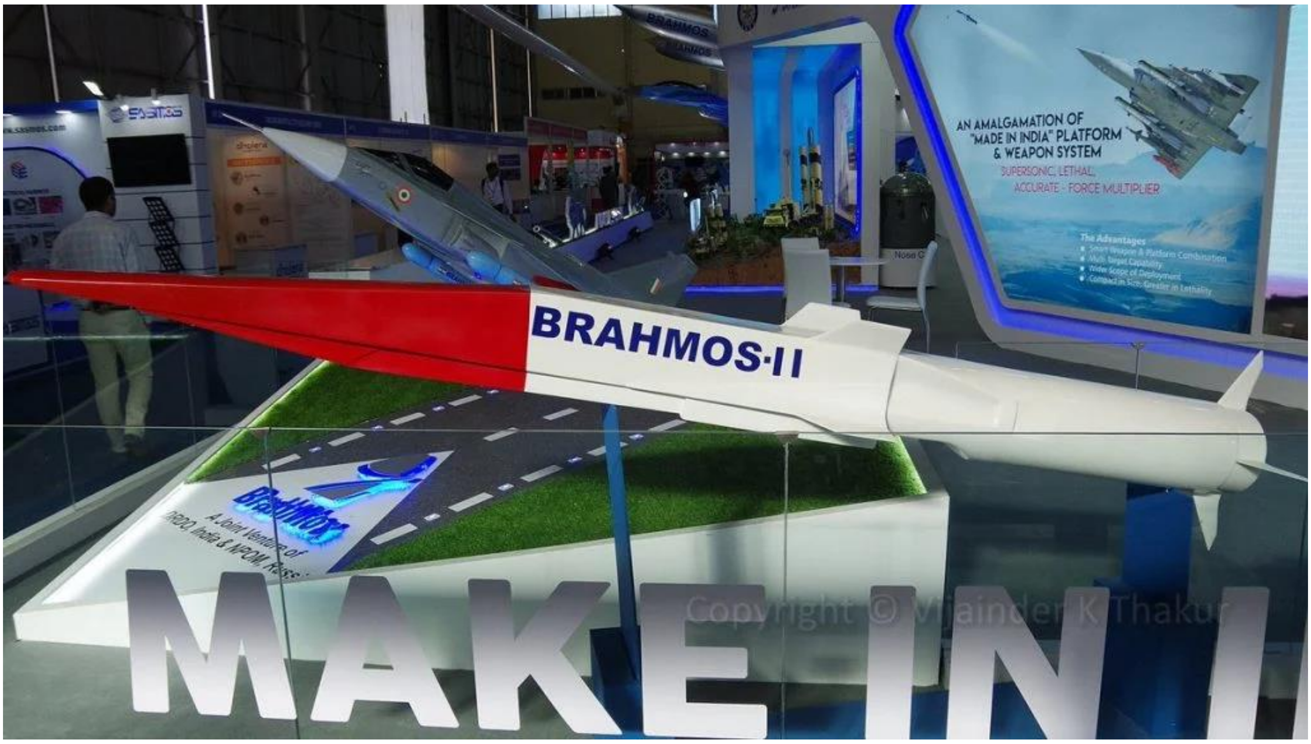
Advanced Propulsion Systems- UNIT 1

HYPERSONIC AIR-BREATHING PROPULSION

Topics:

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.



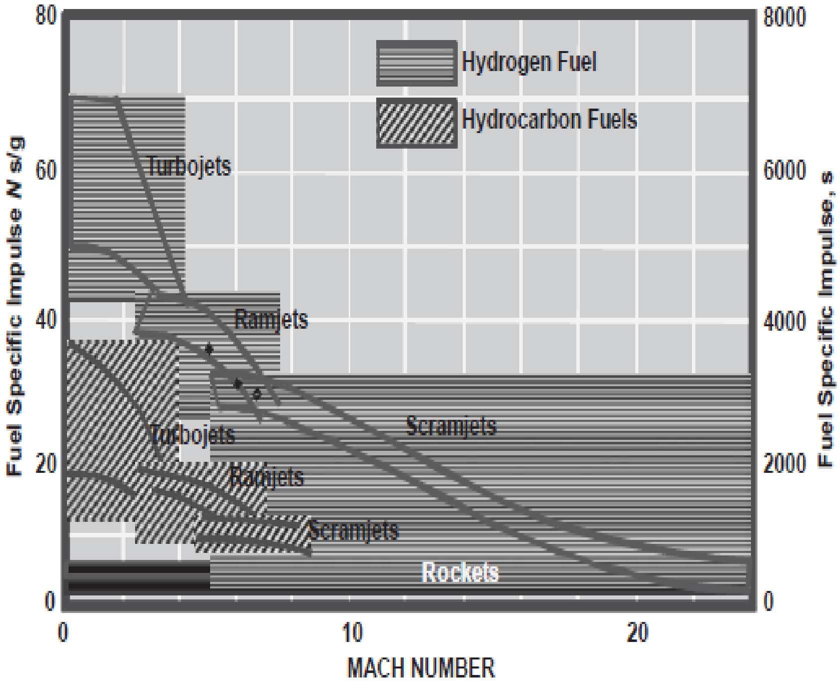
- BrahMos II travels at Mach 7 and has a range of 290 kilometers.
- It can carry conventional, shrapnel, or armor-piercing warheads weighing up to 300 kg and will be launched from a ship in the first phase.
- The missile will replace the Russian naval ships' existing P 800 and Kalibr missiles. The BrahMos P 800 is the entry-level model.

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 comparison 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 turbinebased 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, ie hydrocarbon fuel has adiabatic flame temperature of around 2300 k, so beyond Mach 7.0, heat addition is not possible by burning fuel. Therefore, heat must be added at lower stagnation temperatures ie at supersonic speeds.

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

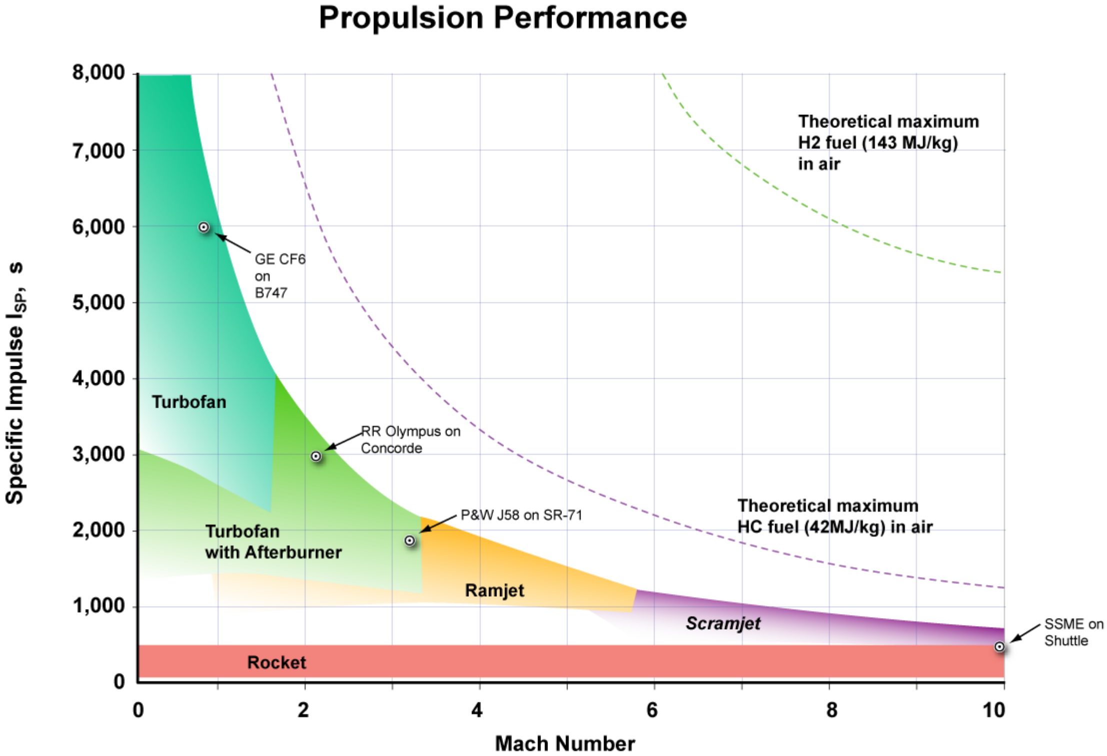
Therefore, all hypersonic transport propulsion systems need supersonic combustion ram jets (Scramjets)

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.



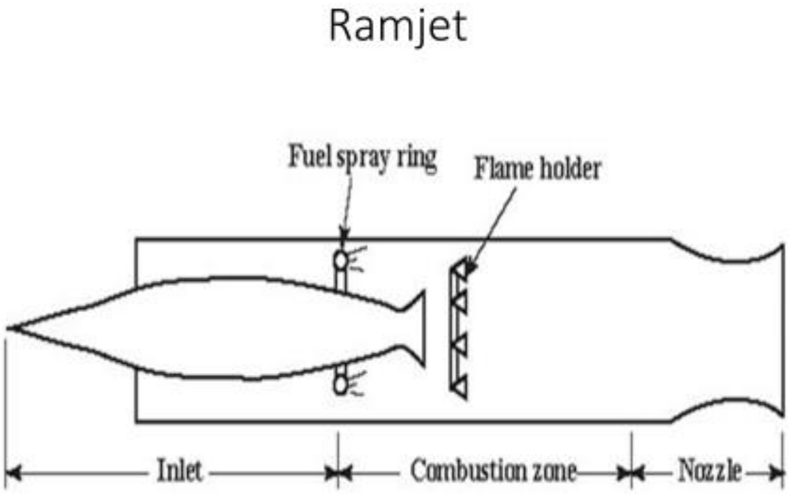
The performance limits of air breathing systems is shown below:

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.
5. As the Mach number continues to increase in the hypersonic regime the scramjet specific impulse approaches that of the rocket engine.

Ramjets:

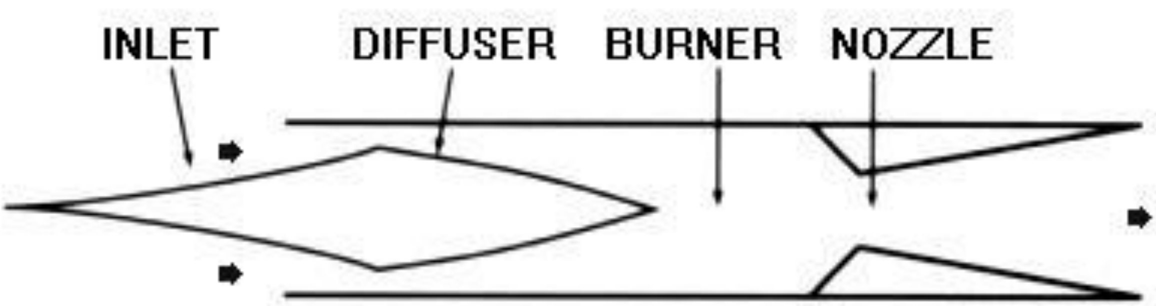


Operating Principle:

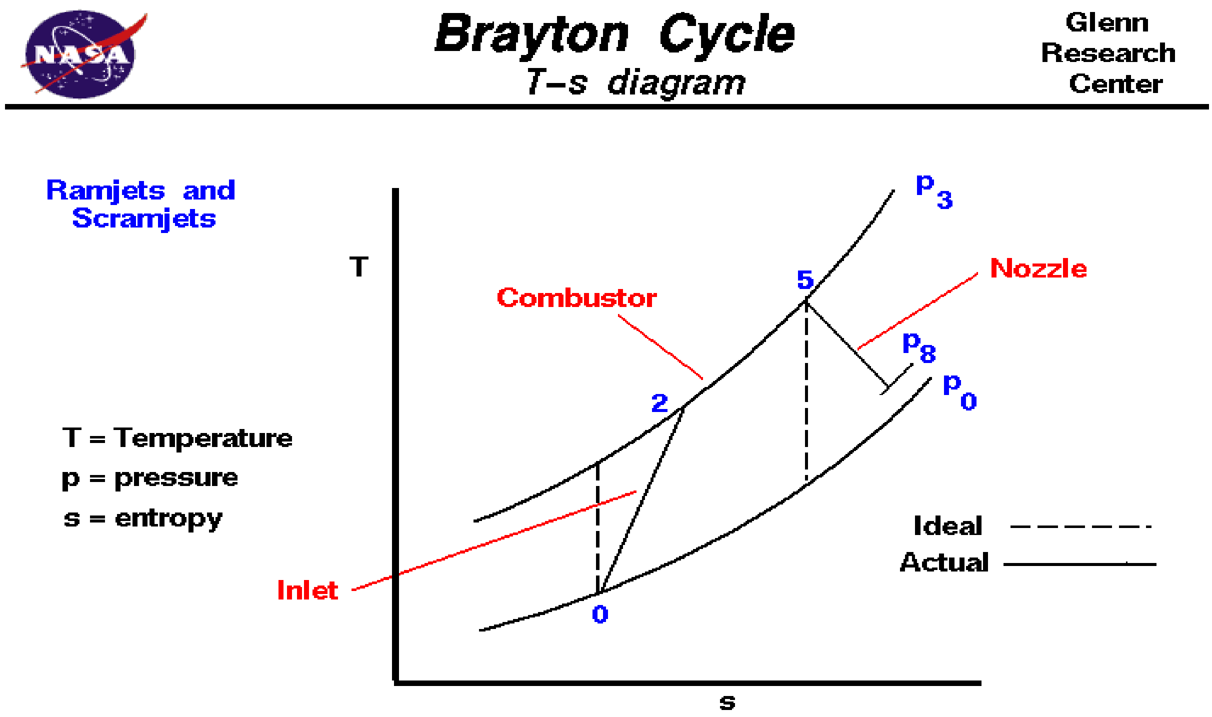
The incoming supersonic airflow is slowed to subsonic speeds by multiple shock waves, created by back-pressuring the engine.

This subsonic flow undergoes combustion andaccelerates 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 M= 6
- Defence applications of scramjets in missiles is also very sought after due to the **very short reaction times** associated with high speed of the missile system
- Subsonic combustion, which technologically is easier to manage with the current knowledge, would be associated, in the hypersonic regime, with high stagnation temperatures that would lead to unacceptable dissociation levels, and hence an inability to materialize the energy rise expected through chemical reactions
- **Combined cycle engines**: No single-engine cycle exists that can efficiently cover the whole range of a flight from takeoff to orbit insertion; therefore, combined cycles are of particular interest for the design of the scramjet cycle

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°). The shock waves also lie close to the body. This leads to merging of shock waves with the boundary layer, which needs to be considered while predicting the pressure distribution over the body.
2. **Entropy Layer:** Theshock wave around the blunt body (a space vehicle)in a hypersonic flow is thin, highly curved and is associated with large velocity gradients across the shock wave. The region behind the shock wave has strong thermodynamic changes and high losses and is called "entropy layer". The entropy layer causes **high aerodynamic heating** of the surface. This requires effective cooling systems.
3. **Viscous Interaction**: The thickness of boundary layer on the surface of the vehicle is directly proportional to the Mach number. As a result the thickness of the boundary layer is very large at high Mach numbers. The thick boundary layer affects the flow outside the boundary layer called viscous interaction, which **increases the drag and aerodynamic heating**.
4. **High-Temperature effects**: The high kinetic energy flow slows down by the effect of boundary layer interaction and results in very high temperatures. Additionally, the region behind the bow shock wave is another reason for rise in temperature. The high temperatures cause chemical reactions in the flow through molecular dissociations, resulting in high zones of aerodynamic heating of the surface.
5. **Low Density Flow**: At very high altitudes beyond 60 km, air is no more a continuous medium, but rarefied and very low density medium. This alters the aerodynamic force coefficients, heat transfer coeffcients vary considerably and need to be factored in predicting vehicle aerodynamic and propulsive behavior.

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

1. **Reduces $\boldsymbol{O_2}$ 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 favor of a rectangular configuration.

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

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

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

Diffusion flame combustion: In the design of diffusion flame for supersonic combustion, the fuel is injected at the inlet parallel to the air flow. (Fuel preinjection 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.

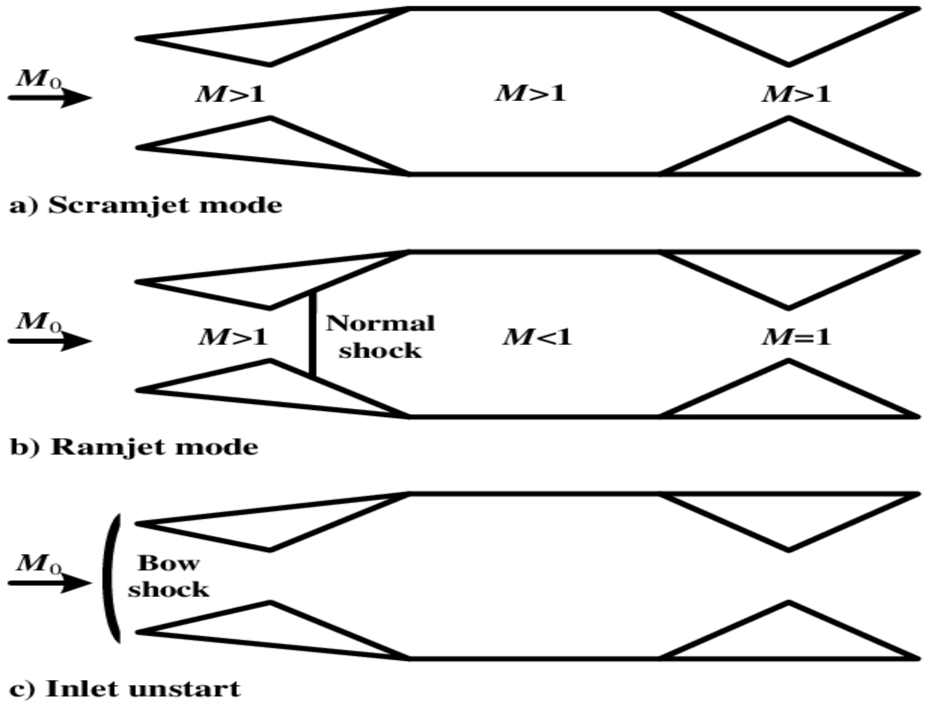
Mixing layers of air at supersonic flows and fuel are characterized by **large-scale eddies** that form due to the **high shear** between both the streams. These eddies entrain fuel and air into the mixing region. Stretching occurs in the interfacial region between the fluids due to compressible shear/mixing 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:



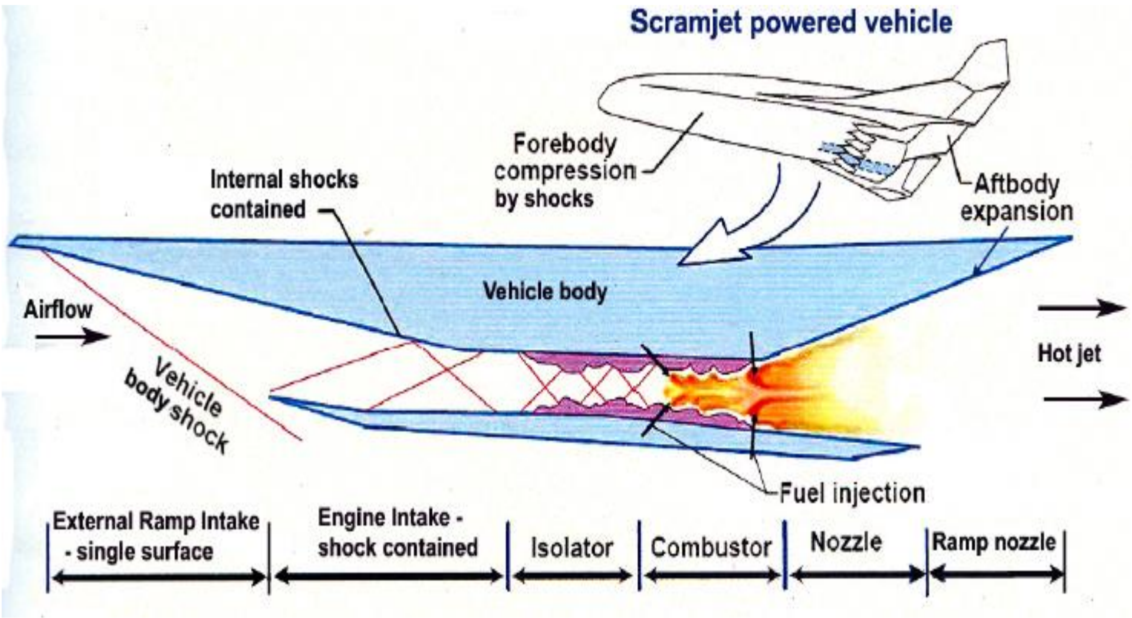
Scramjet Engine- Construction: Scramjet engine is characterized by

- Slow reaction times
- High flow speeds or low residence times in the engine.
- 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 portion of the underside operates as external/internal diffuser, with rear 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. It also 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 external diffusion takes place through a series of oblique shocks along the vehicle front-body.

The engine is designed to take advantage of the compression through shock waves and reduce the load on the diffuser. The air 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.

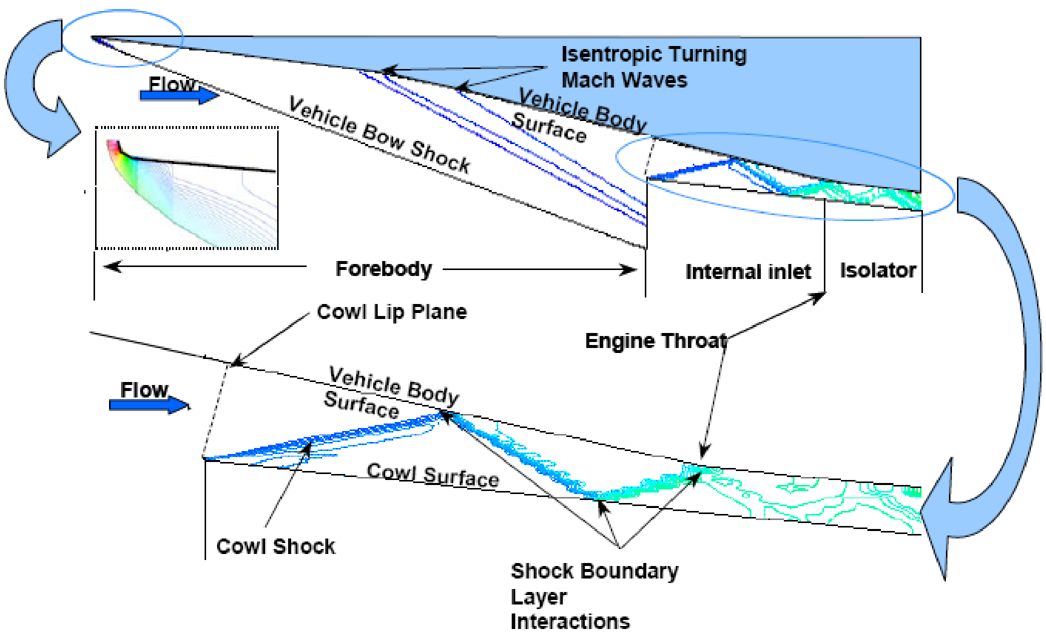
Supercritical Operation of the Inlet: At slow speeds, the inlet will not capture all the free stream air and will result in causing spillage of air, contributing to spillage drag. This condition of inlet is termed as “sub-critical” and should be avoided.
However, as the free stream mach number increases, 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.

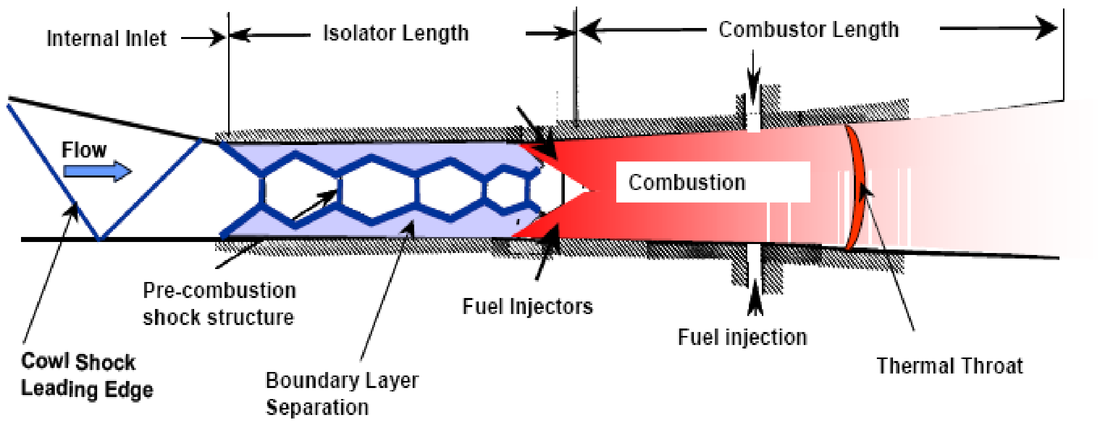


Isolator: Isolator is constant area diffuser containing the internal shock structure, swallowed during supercritical operation of the inlet (or during operation after the inlet "started"). The isolatoris inserted before thecombustor to diffuse the flow further, through a shock train, producing desired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

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

Because a substantial part of the vehicle is dedicated to nozzle expansion, considerable lift and pitch moments are produced by the pressure distribution on this part of the after-body, complicating the nozzle design and vehicle integration.

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

Operation of expansion system is shown below:

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

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

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

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

The shock train provides a mechanism for the incoming supersonic flow to adjust to a static 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, and
- Rocket-based combined cycle (RBCC) engines supplement a traditional rocket's propulsion with a scramjet, allowing for additional oxidizer to be added to the scramjet flow.
- RBCCs offer a possibility to extend a scramjet's operating range to higher speeds.

Working Principle Dual-mode Scramjet:

A pure ramjet engine operates at supersonic speeds, but with subsonic combustion, requires two area restrictions or physical throats. The first throat, at the outlet from the inlet diffuser, is required to stabilize the normal shock formation in order to deliver subsonic flow to burner. The second throat is located downstream of the burner, is required to accelerate the subsonic flow to supersonic velocities. It is important to note that flow is choked (M=1) only in the second throat.
A pure scramjet engine has no physical throat.

The Dual-mode engine uses “no-throat” geometry, capable of switching over from ramjet or scramjet mode. Employing area constrictions mean limiting the mass flow rate at high flight mach numbers.

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

In order that the burner entry flow is subsonic, the flow must be choked (M=1) somewhere downstream, which causes large back pressure 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 (~ √ 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 cause 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.

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:

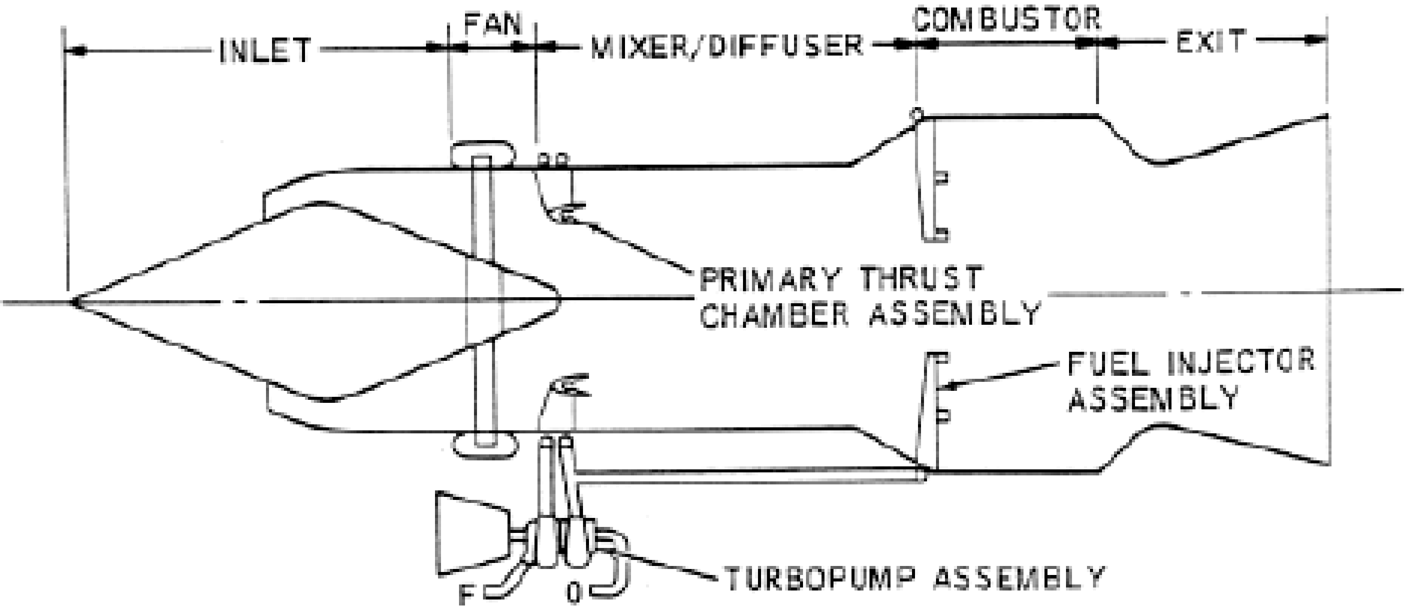
- The ejector portionof the device will have constant area and fixed geometry.
- The inlet primary flow will be supersonic and inlet secondary flow is subsonic. The ejector exit plane is sonic and choked.
- The ejector ramjet has a potential to increase the thrust above the primary flow alone, with a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

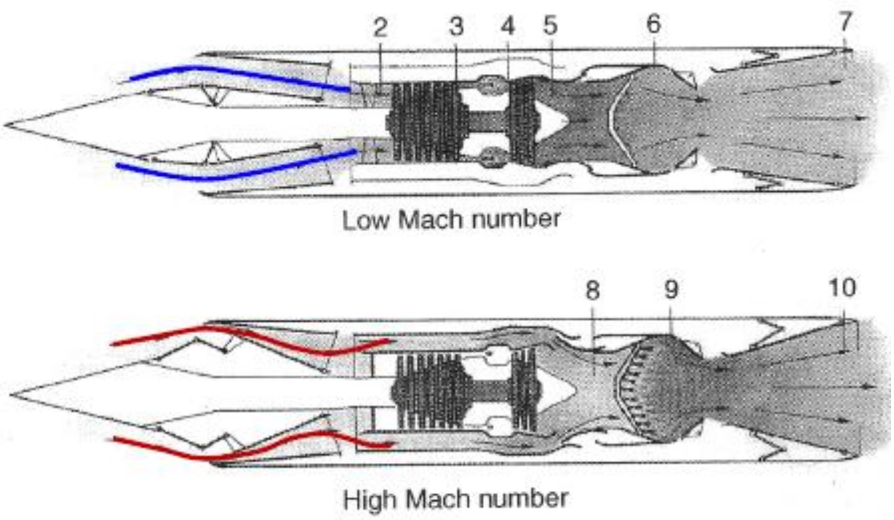
Finally, the cryogenic fuel can be used in air-breathing modes as a **heat sink** to increase the density of the inlet airflow, thus increasing the work output.

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.

Engine General Arrangement and Operating Schematic



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.



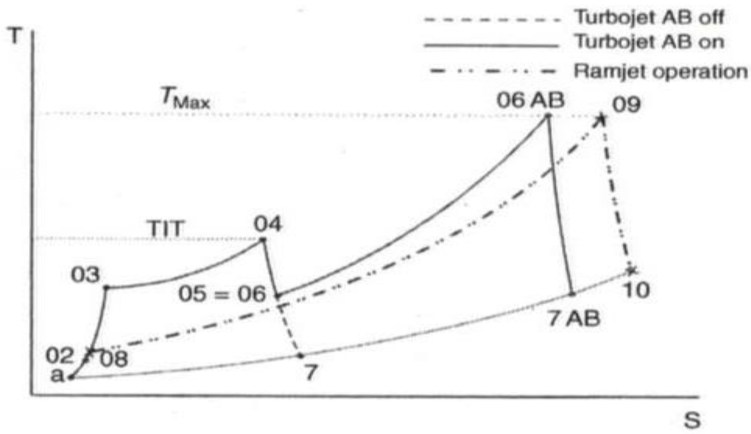
Only Turbojet

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 turboramjet is a hybrid engine that essentially consists of a turbojet mounted inside a ramjet. The turbojet core is mounted inside a duct that contains a combustion chamber downstream of the turbojet nozzle.

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

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

Main components of Air Turbo Ramjet:

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

Turbo ramjet rocket:

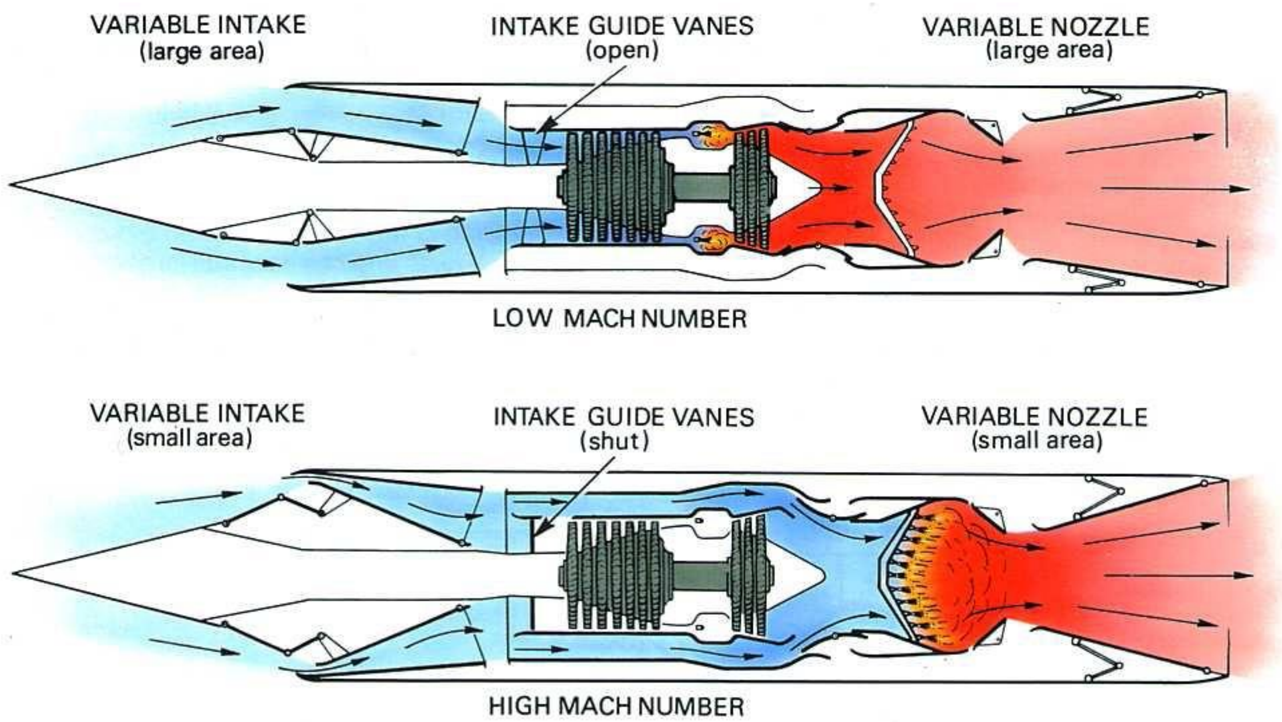
A variation of ATR concept is the addition rocket motor to ATR engine.

The primary reason for adding the internal rocket engine is to supplement thrust available at both lower and higher Mach number range.

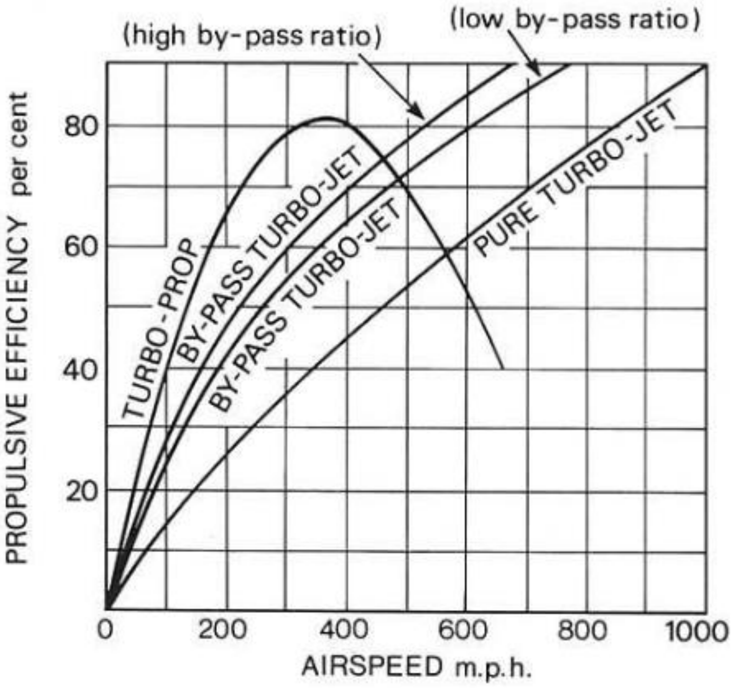
The extra rocket to the core engine integrates with the overall configuration, to augment thrust levels to the core turbojet at lower mach numbers and to the ram/scram engine at higher mach numbers.

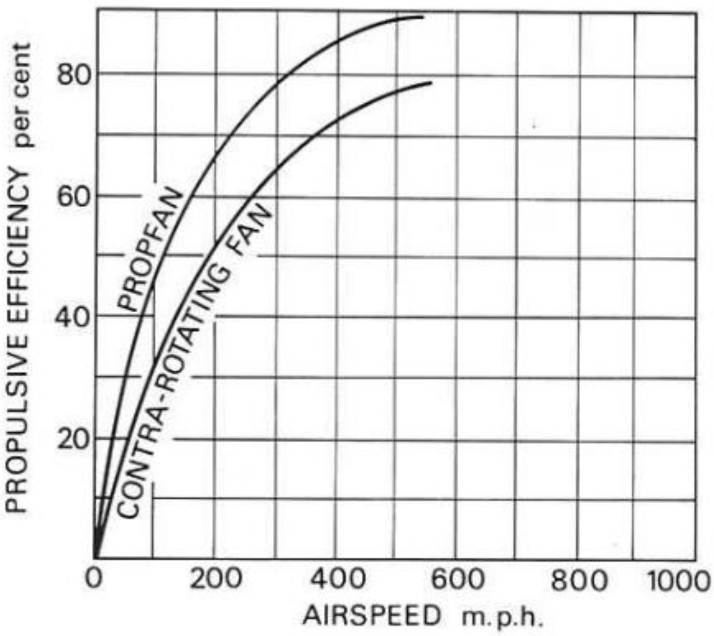
The existing exhaust nozzle is designed to provide the very large area ratios demanded by the combination.

A schematic diagram is given below.



The turbo/ram jet engine (fig. 1-11)





The turbo/ram jet engine (combines the turbo-jet engine (which is used for speeds up to Mach 3) with the ram jet engine, which has good performance at high Mach numbers.

The engine is surrounded by a duct that has a variable intake at the front and an afterburning jet pipe with a variable nozzle at the rear. During take-off and acceleration, the engine functions as a conventional turbo-jet with the afterburner lit; at other flight conditions up to Mach 3, the afterburner is inoperative. As the aircraft accelerates through Mach 3, the turbo-jet is shut down and the intake air is diverted from the compressor, by guide vanes, and ducted straight into the afterburning jet pipe, which becomes a ram jet combustion chamber. This engine is suitable for an aircraft requiring high speed and sustained high Mach number cruise conditions where the engine operates in the ram jet mode.

Liquid Air Cycle Engine (LACE):

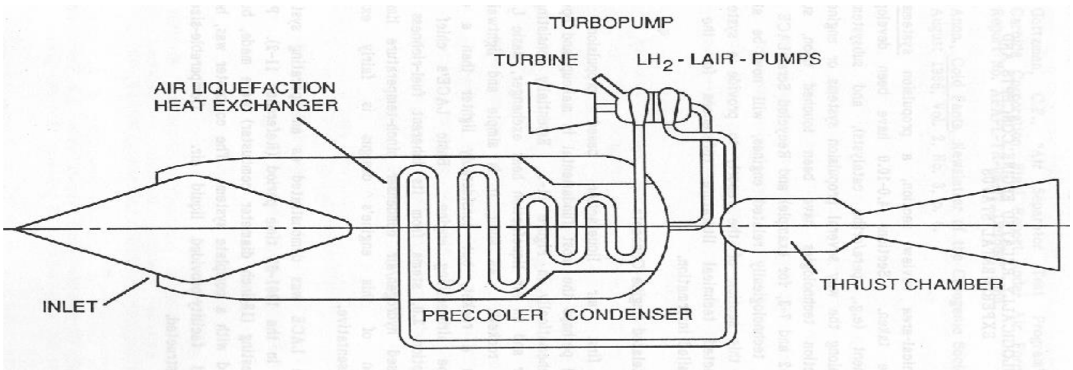
Liquid Air Cycle Engine is a separate class of hypersonic air breathing engine made possible by the availability of very low temperature, cryogenic liquid hydrogen fuel. The liquid hydrogen fuel has high specific energy release (heat of combustion per unit mass), good vehicle cooling capacity and also 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 propellent rocket by gathering part of its oxidizer from the atmosphere.
- It lowers the take-off weight of the spacecraft considerably.

Disadvantages:

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

Fuel injection in Scram Jet Engine:

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

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

Major issues encountered in the scramjet Engine combustion are

- Combustion efficiency in converting chemical energy in to kinetic energy
- Heat transfer at low-pressure conditions in the combustor
- Low residence times in the scramjet

In the design fuel system for supersonic combustion, fuel pre-injection in inlets or isolators holds considerable potential. Pre-injection or distributed injection enhances mixing, flame stability, and combustion efficiency for scramjet engines. The fuel is injected at the inlet parallel to the air flow. Distributed and scheduled fuel injection are adopted in combined cycle engines.

During the operation engines in the lower Mach number range, the flow residence times are relatively large, therefore, fuel injection is considered only in the combustion chamber.

However, as the Mach number increases, the flow is supersonic throughout the combustion chamber with very low residence times. Fuel injection must begin in upstream region, including the inlet.

The large localized heat release in a given section of combustor, gives rise to shock waves which spread the heat release in the flow direction resulting in an advantage of the diffusive mode of supersonic combustion.

Following factors influence the design of combustors:

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

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

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

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

Fuels used for RBCC systems have to satisfy following requirements;

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

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

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

Drag In Scramjet Aircraft:

During hypersonic flight, the engine thrust is only slightly larger
than the vehicle’s drag; hence efficiency of expansion process and thethrust angle relative to the flight direction become critical for the vehicle’sflight 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 DRAG LOSSES)-The main sources of losses in the isolator are caused by the pressure drag and the viscous drag. At hypersonic speeds, relative heat addition to the air progressively decreases with increased flight velocity whereas the drag losses continuously increase until the heat addition can no longer overcome the drag and the air-breathing-based system reaches the extent of its flight envelope.

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

Airflow enters the control volume at the flight conditions, fuel is added to the air in the combustor and the flow exits through the vehicle nozzle. For ease of analysis, the flow exiting the control volume is represented by a one-dimensional average flux of the exhaust plume. The spillage drag and plume drag have been combined into a single force called the additive drag.
Air spillage (and therefore spillage drag) decreases as the vehicle speed approaches the design point of the engine, and the plume drag varies depending on the amount of under-expansion in the nozzle. Both these forces are usually estimated through CFD analysis or through rules-of-thumb based on empirical or experimental databases.

Advanced Propulsion Systems- Unit II

Chemical Rocket Propulsion

1. **Chemical Rocket**: Classification of rocket engine, chemical rocket engine types, working principle, schematic diagram, applications, types, advantages and disadvantages-solid, liquid and hybrid propellant rocket engines,
2. Propellants used, injectors, nozzles, igniters, storage, TVC, combustion instabilities, combustion chamber, pulse detonation engine, rotary rocket engine

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 produceshigh 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.

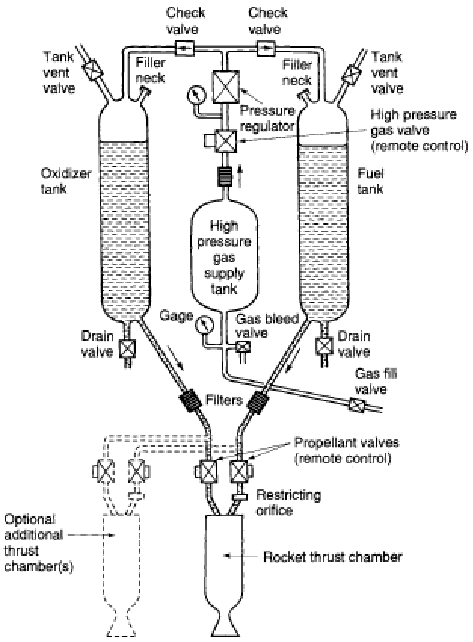
Working Principle:

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.

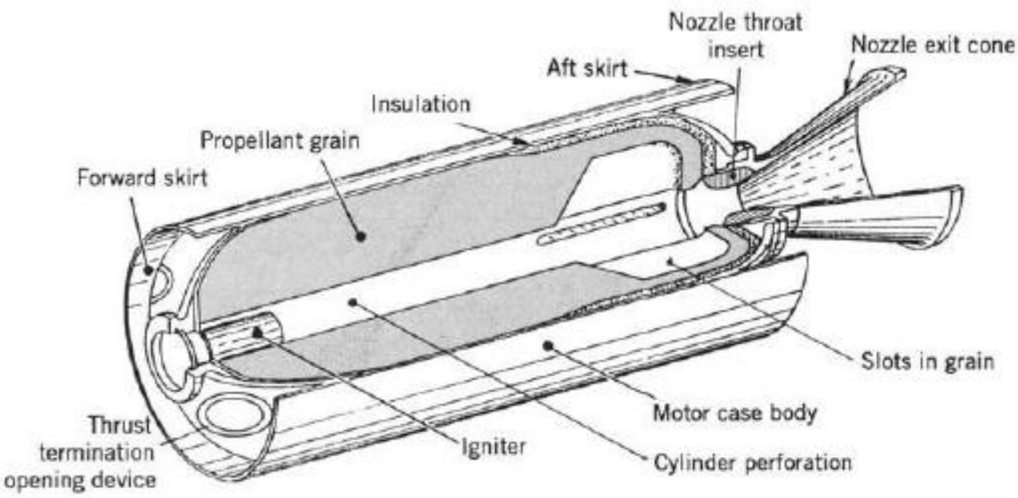
A high pressure gas pressure tank provides pressure feed of oxidizer and fuel through diaphragms. Alternatively, separate pumps may be used to provide pressure feeding of propellants.
The propellants react in the thrust chamber and generate hot combustion gases which are expanded in the supersonic convergent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



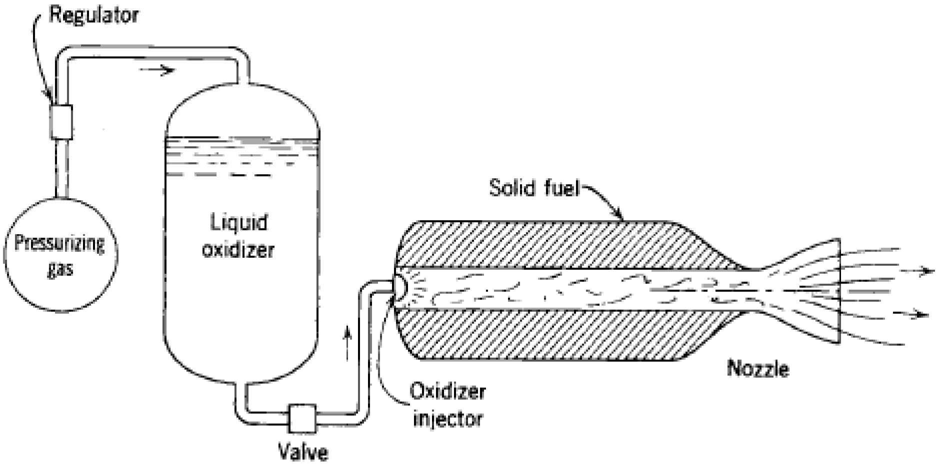
Solid Propellant Rocket Propulsion System: A schematic diagram is shown below:



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.

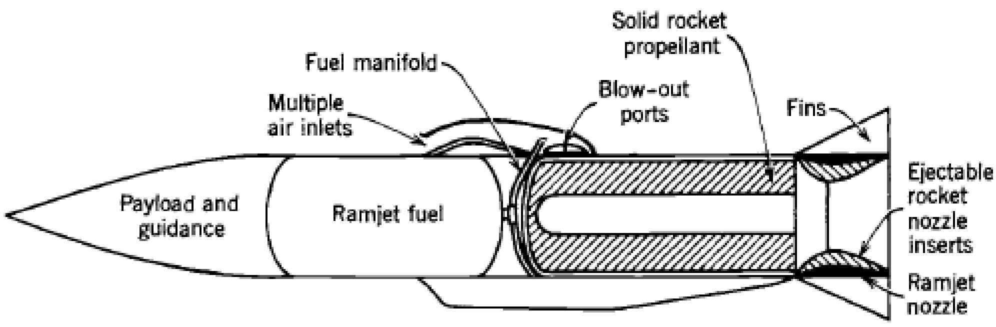
Hybrid Rocket Motors:A schematic diagram is given below:



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.

Combination of Ducted & Rocket-Propulsion System (Dual Cycle Engines): A schematic diagram is as follows:

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



nozzle

The principles of rocket and ramjet can be combined so that the two propulsion systems can operate in sequence, yet utilize a common combustion chamber.

Initially the system operates in rocket mode, and as the solid propellant combustion completes, the air inlet to the combustion chamber opens, for ramjet operation. Ramjet fuel tank supplies the fuel, and the nozzle throat enlarges to accommodate enhanced ram air flow/combustion products.

Applications : Basic application of Rocket propulsion systems are

- Space Launch Vehicles
- Spacecraft
- Missiles
- Other applications

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

Depending on the mission, they can be classified based on

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

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

2. **Spacecraft**:

Depending on the mission, spacecraft can be classified as

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

Majority of spacecraft use liquid propellants engines, with solid propellant boosters.
Electric propulsion systems are used both for primary and secondary propulsion missions on long duration space flights, inter-planetary/inter-stellar missions.

3. **Missiles**:

Missiles can be classified based on

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

4. **Other applications:** Other applications are the secondary applications which include

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

Advantages/Disadvantages of Chemical Rockets:

- **Solid Propellant Rockets:**

Advantages:

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

Disadvantages

1. Explosion and fire potential is larger
2. Most rocket motors cannot withstand bullet impact or being dropped on hard surface
3. Rockets need environmental clearance and safety features for transport on public conveyances
4. Some propellants are very sensitive and can detonate
5. Grain damage occurs through temperature cycling or rough handling-limiting useful life
6. Requires an ignition system; Plumes cause more radio-active attenuation than 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.

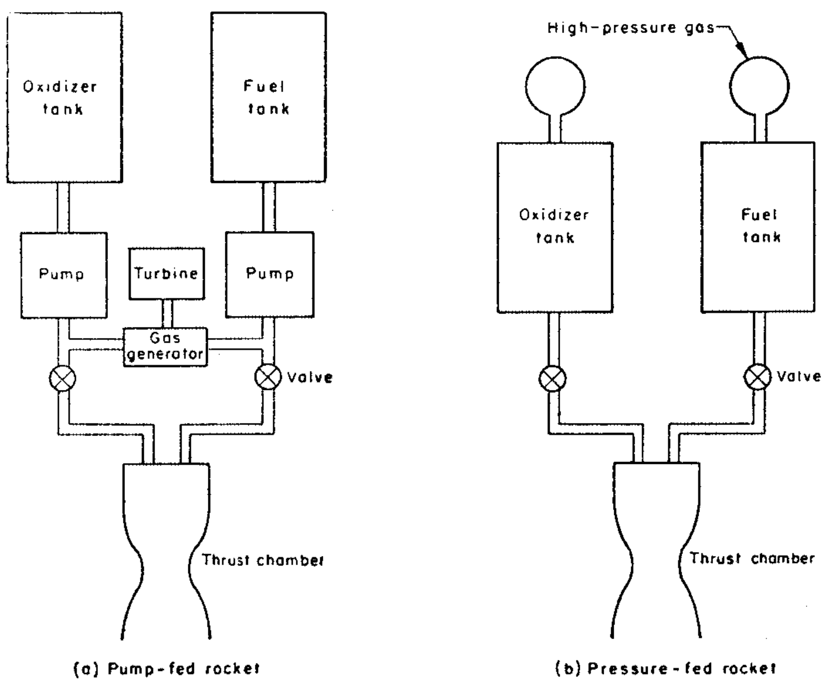
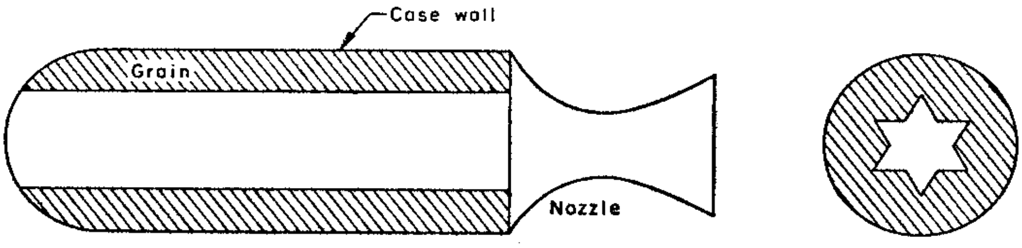


FIG. 2. Schematic of liquid-propellant rocket

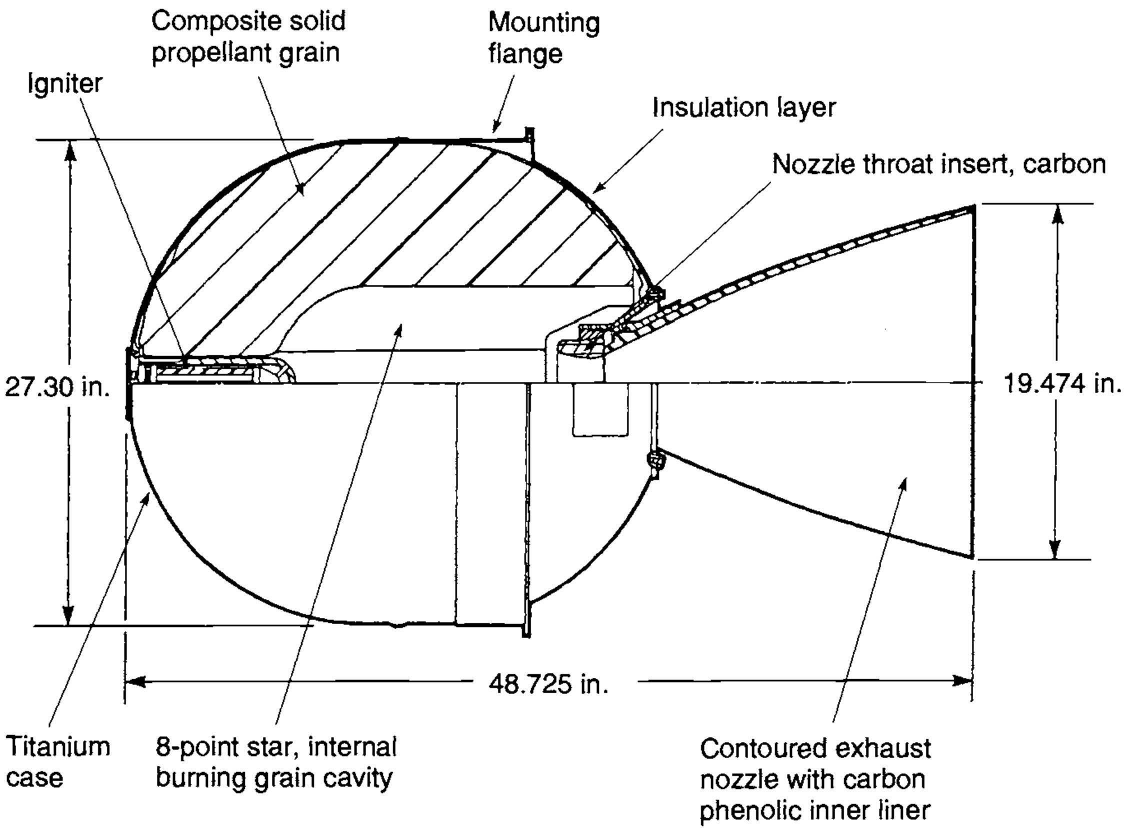
Solid Propellant Rocket Motor:



TVC
Solid Propellant Rocket Motor:

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.

Principle:



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 thethrust 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 = $\boldsymbol{{F}/{\dot{m}}}$

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.

   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:

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

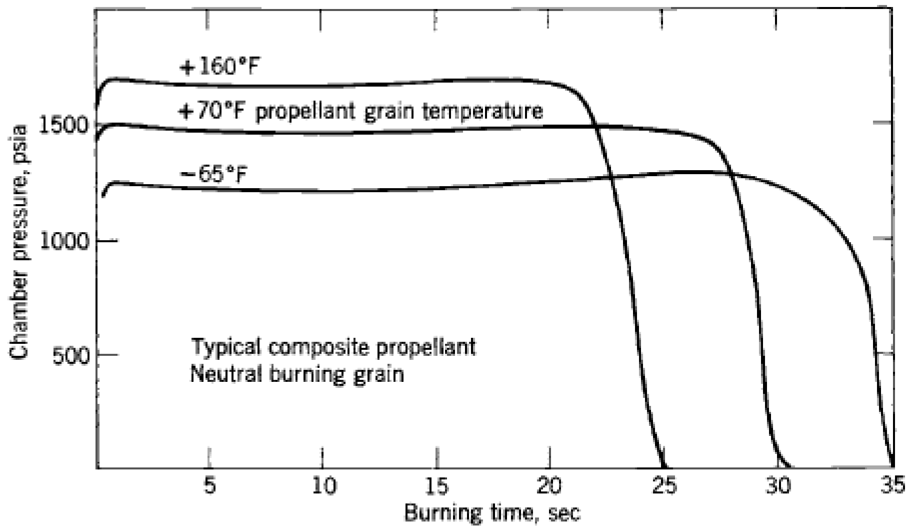
Where A_b is burning area of the propellant grain, r is the burning rate, ρ_b is the solid propellant density and $\dot{m}$ is the mass flow rate of hot combustion gasses.
A_b and r vary with time and chamber pressure.

The relation between burning rate and chamber pressure is

$r = ap_1^n$, 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.

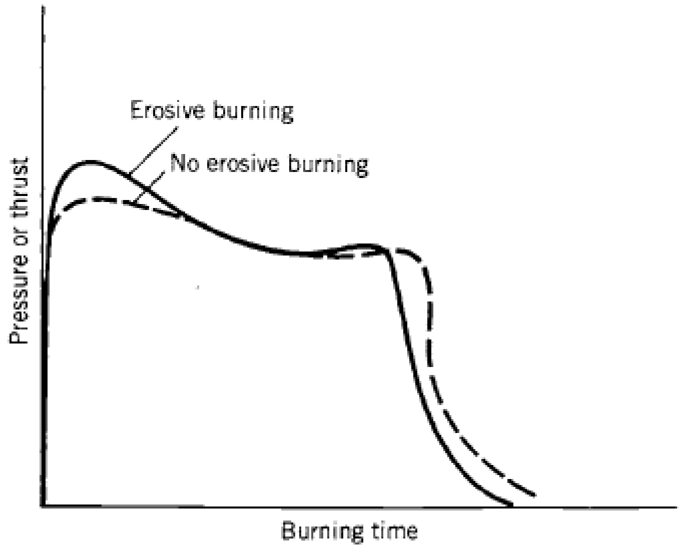
The burning trace of a particular rocket propellant is as given below:



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.

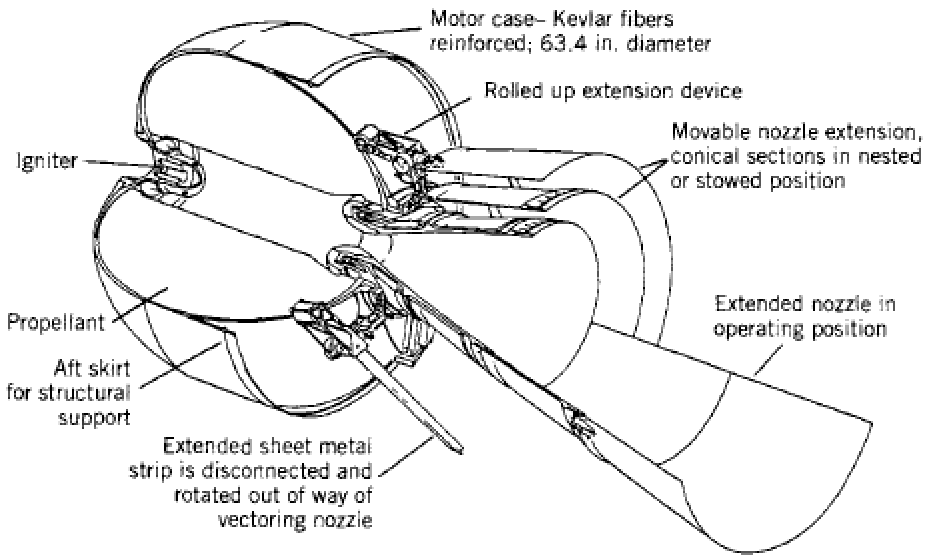
Erosive burning increases the mass flow and thus increases the chamber pressure and the thrust force. This is shown below in the pressure vs time burning trace below:



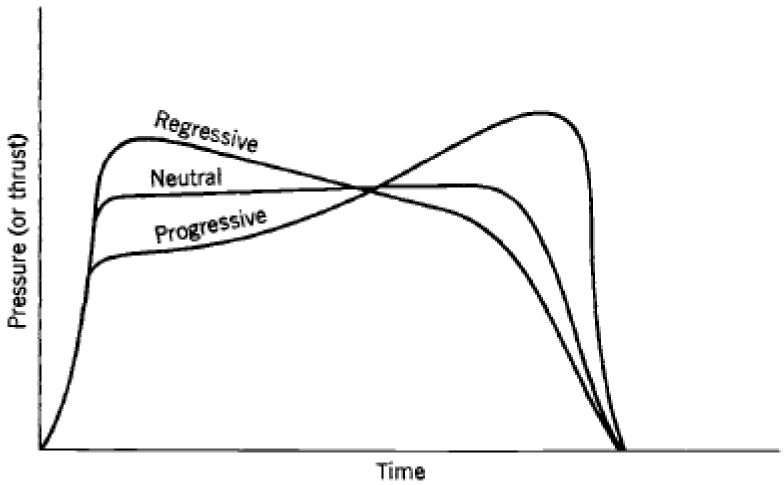
Important definitions and terminology of solid propellants are as follows:

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.



Classification of Grain depending on their pressure-time characteristics: Solid propellant grains ae classified as “neutral, progressive and regressive types” based on pressure-time characteristics as shown below:



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.

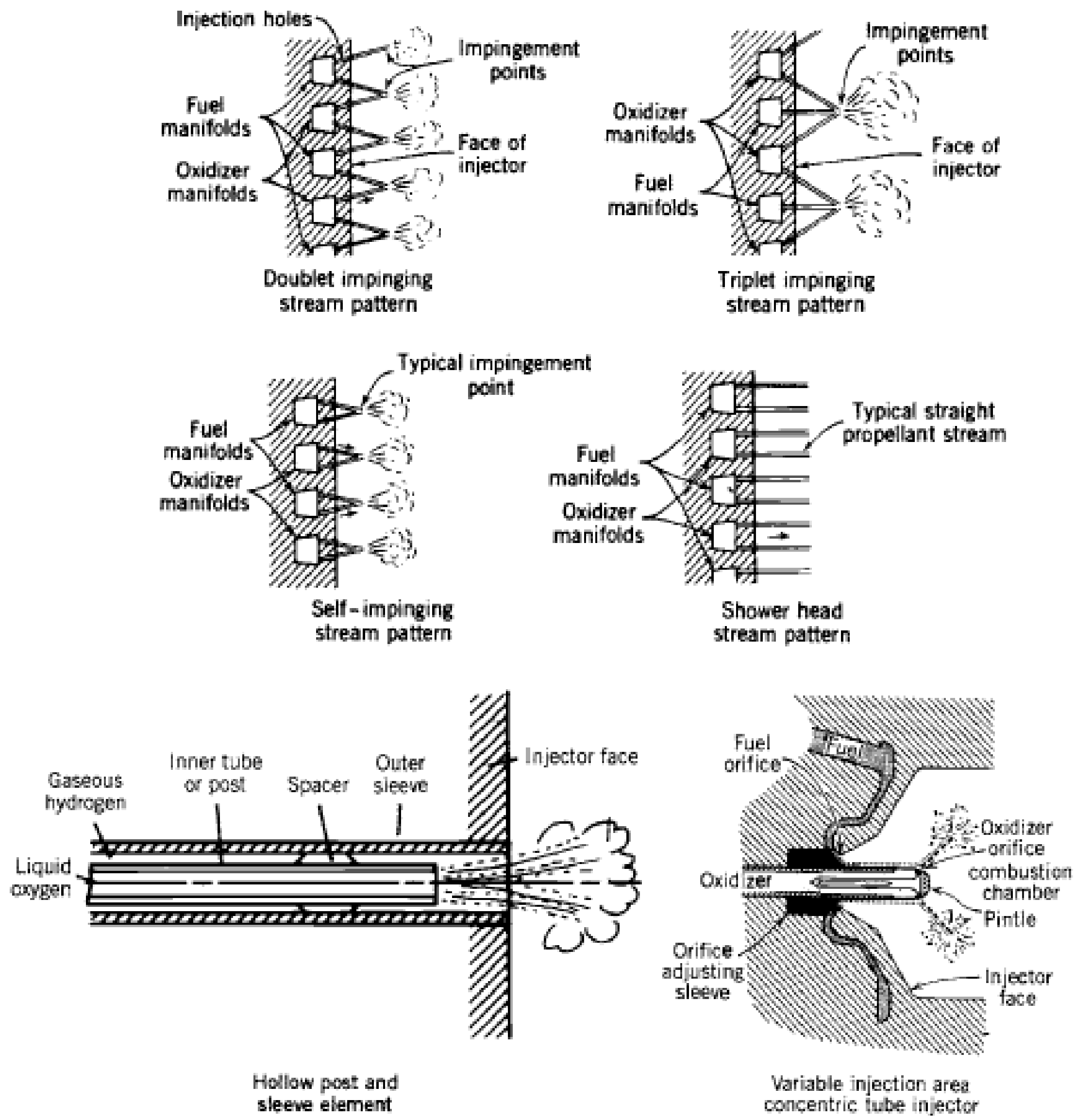
Storable propellants (Nitric acid or gasoline etc) are liquids at ambient temperatures and can be stored in sealed tanks for long periods.

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.

Nozzles:

The function of nozzle is

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

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

Under and Over-expanded Nozzles:

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

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

Alternatively, if the exit pressure is lower than the ambient pressure, the flow is considered to be **over expanded**.

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

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

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

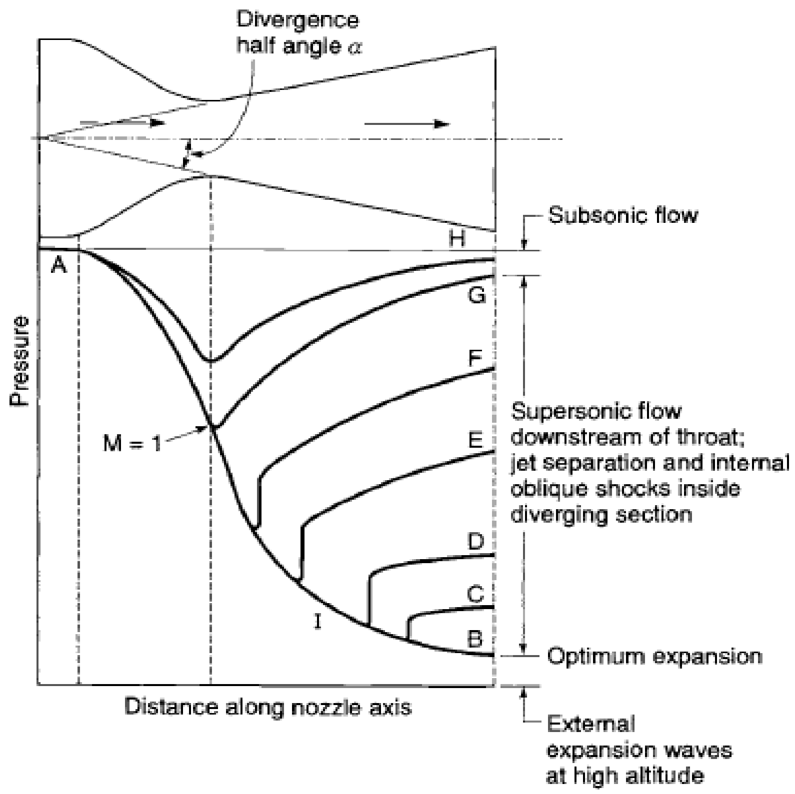


FIGURE 3–9. Distribution of pressures in a converging–diverging nozzle for different flow conditions. Inlet pressure is the same, but exit pressure changes. Based on experimental data from A. Stodala.

- Curve AB shows variation of pressure with optimum back pressure at the design area ratio. (with M = 1 at throat)

- Curves AC and AD show variation of pressure along the axis for increasingly higher external pressure (over-expansion). At point I, on curve AD, the pressure is lower than the exit pressure and a sudden rise in pressure takes place accompanied by separation of flow from the walls. (Condition when aircraft flies at altitude lower than design altitude)

- The sudden pressure rise in the curve AD is a compression discontinuity accompanied by a compression wave.

- Expansion waves occur in cases where external pressure is lower than the exit pressure, ie below point B. (Under-expansion – condition when aircraft flies at an altitude higher than design altitude)

- When external pressure p_3 is below the nozzle exit pressure p_2, it signifies under-expansion. The expansion of gas inside the nozzle is incomplete and the value of C_F and I_s will be less than at optimum expansion.
- For external pressure p_3 is slightly higher than the nozzle exit pressure, the nozzle continues to flow full. This continues till p_2 reaches a value between about 25 and 40% of p_3. The expansion is inefficient, C_F and I_s values are lower than optimum.

- For higher external pressures, separation of flow takes place inside the divergent portion of the nozzle. The axial location of separation depends on the local pressure and wall contour. With steady flow, separation is also axially symmetric.
- On separation, at the nozzle exit plane, the center portion remains supersonic while the surrounding annular flow is subsonic.
- **Divergence of the Flow**: The divergence of flow at the nozzle exit section causes losses which varies as a function of cosine of the divergence angle.

  Correction factor λ = $\frac{1}{2}$(1+cos α)

  The losses can be reduced by using a bell-shaped nozzle contour
- Small chamber or port areas relative to the throat area, ie **low nozzle-contraction ratios** $\boldsymbol{A_1/A_t}$causes pressure losses in the chamber and cause reduction of thrust andexhaust velocity slightly. For A_1/A_t of 3.5, thrust reduction is around 1.5%, whereas for A_1/A_t of 2.0 (smaller chamber), thrust reduction is around5.0%.
- **Lower flow velocity i**n the boundary layer or wall friction can reduce the effective exhaust velocity by 0.5 to 1.5%
- **Solid particles or liquid droplets** in the exhaust gas can cause losses up to 5%
- **Chemical reactions in nozzle flow** change gas properties and gas temperatures and cause losses of around 0.5%.
- Transient pressure operation during start or stop operations cause losses.
- **Gradual erosion** of throat region increases the throat area and decrease chamber pressure and thrust by 1 to 6%
- Using real gas properties cause loss of performance by 0.2 to 0.7%
- Operation at off design altitudes cause a loss of up to 15%.This can be reduced by using nozzles with altitude compensation.

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

Flow Conditions in Nozzle:

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

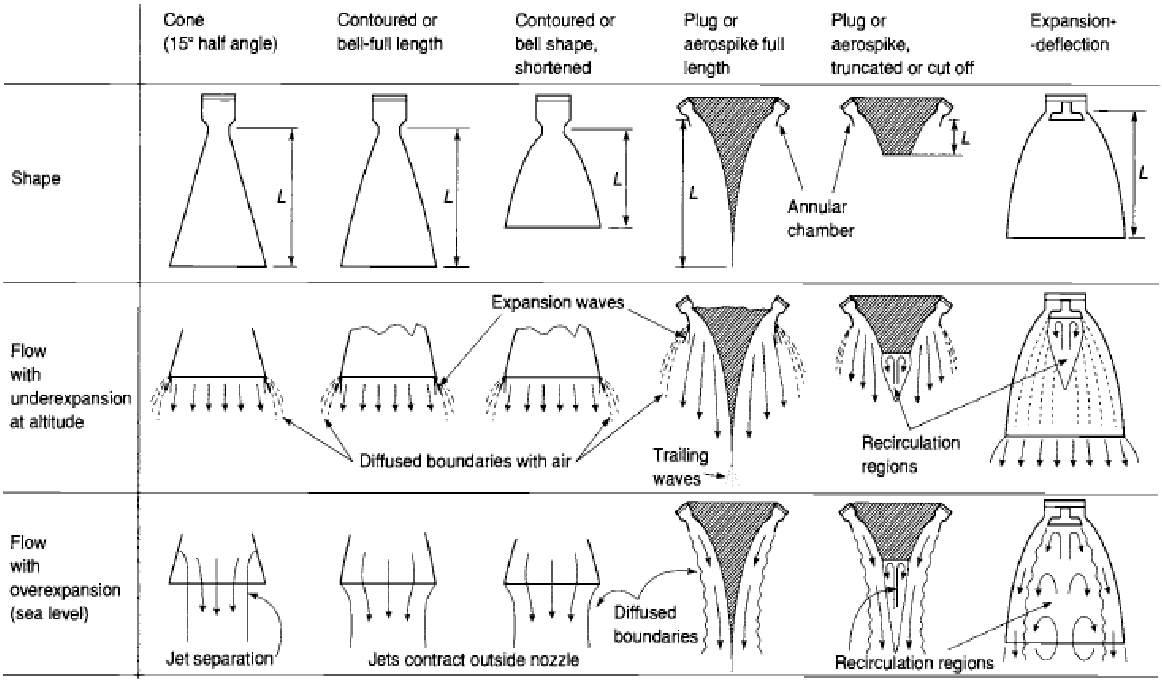
- In some rockets, the gaseous working fluid contains many small liquid droplets and/or solid particles that must be accelerated by the gas.
- This occurs in with solid propellants and some gelled liquid propellants which contain aluminum powder that forms small oxide particles in the exhaust.
- It can also occur with ion-oxide catalysts, or propellants containing beryllium, boron or zirconium.

- In general, if the particles are very small, with diameters of 0.005mm or less, they will have almost same velocity as the gas and will be in thermal equilibrium with the nozzle gas flow
- The solid/liquid particles give up heat to the gas during expansion in a nozzle.
- As the gases give up kinetic energy to accelerate the particles, they gain thermal energy from the particles.
- As the particle diameter become bigger, the larger particles do not move as fast as the gas and do not give up heat as readily as the small particles.
- The larger particles have a lower momentum and they reach nozzle exit at a higher temperature than the smaller particles.
- For larger particles, over 0.015 mm diameter, the specific impulse can be 10 to 20 % less than the specific impulse value without flow lag.

Relations in nozzle:

- In the convergent section, flow is subsonic. The chamber contraction area ratio A_1/A_t is small, in the range of 3 to 6.
- In solid propellant rocket chamber, A_1 refers to flow passage or port cavity of the grain.
- The divergent portion handles supersonic flow, the area ratio becomes large very quickly. Value of k also varies significantly.
- The area ratio in the divergent section, A_2/A_t ranges between 15 to 20; at M=4
- T_0 = T [1 + $\frac{1}{2}$(k--1)M^2]
- M = $\sqrt{\frac{2}{\gamma-1}[\frac{T_0}{T} - 1]}$
- The exhaust velocityv_2 is calculated as v_2 = $\sqrt{\frac{2\gamma}{\gamma+1} RT_1[1 - (\frac{p_2}{p_1})^{\frac{\gamma-1}{\gamma}}]}$
- For optimum expansion p_2=p_3, v_2=c_{2opt}; the optimum expansion occurs only at design altitude. At all other altitudes, the nozzle is either under expansion or over expansion condition.
- The throat condition is defined by pressure ratio p_t/p_1 =$(\frac{2}{\gamma+1})^{\frac{\gamma}{\gamma-1}}$
- The throat temperature T_t = (2 T_1)/(γ+1)
- The throat velocity v_t = $\sqrt{\frac{2\gamma}{\gamma+1} RT_1}$

Types of Nozzles



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 = Thrust/propellant mass flow rate

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

Rocket Equation:

- Tsiolkovsky's Rocket Equation calculates the acceleration of the rocket vehicle with mass decreasing continuously due to burning of propellant. The incremental velocity of the rocket vehicle **ΔV** is calculated as equal to:

$$\Delta V = V_e \ln \left[\frac{M_i}{M_f}\right]$$

Where **ΔV** is the velocity increment calculated from the rocket equation, and is a measure of the energy expended by the rocket;

V_e is the average velocity of exhaust gas

M_i is the initial mass if the spacecraft and

M_f is the final mass of the spacecraft. (Final mass of the spacecraft = Initial mass – propellant mass)

Limitations of Chemical Rockets:

- Chemical rockets can achieve up to 4.5 km/s exhaust velocities, thereby limiting maximum velocity increment that is needed for deep space missions
- Chemical propulsion systems are also limited by the energy stored in their propellants
- This limitation can be overcome by using electrical power available on board, from nuclear/solar sources, and couple it with propellant carried on board

Propulsion Systems-Comparision

Types of propulsion systems along with maximum are given below

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

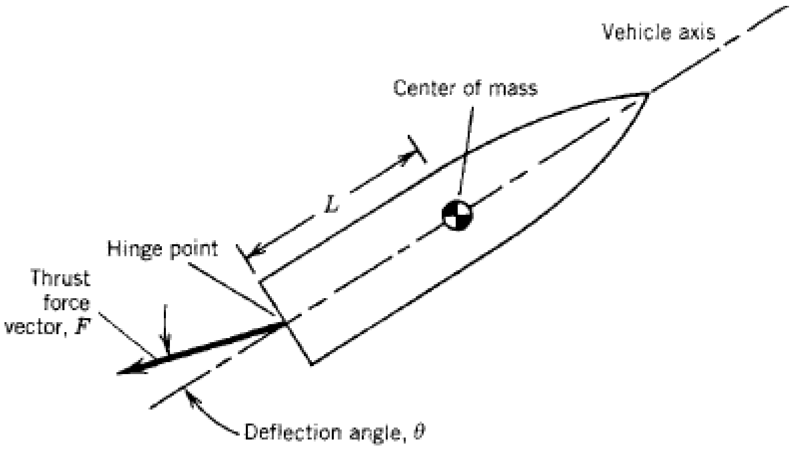
Thrust Vector Control in aircraft: Thrust vectoring nozzles are used in combat aircraft for take-off and landing. Thrust vectoring is also deployed to improve maneuvering and augment lift in flight during combat
Concorde aircraft used two convergent nozzles, primary and secondary nozzles
The secondary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

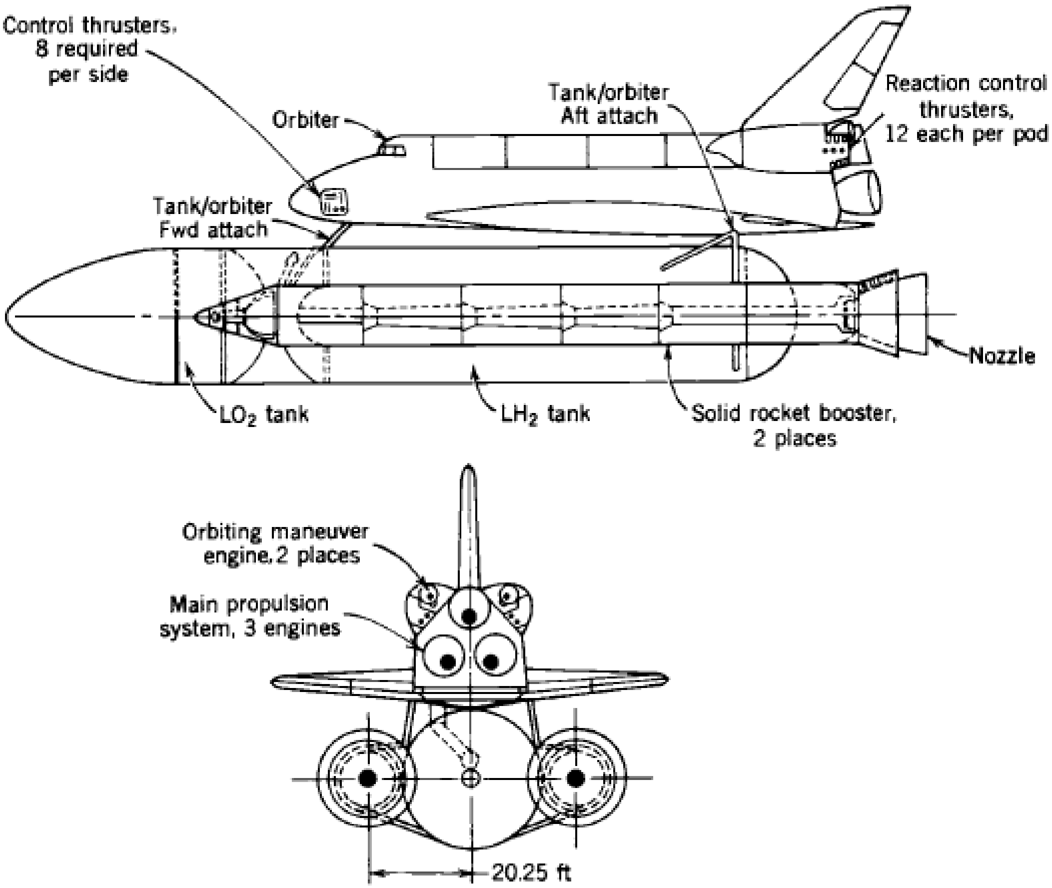
1. To willfully change flight path or trajectory of the vehicle
2. To rotate the vehicle or change its attitude during flight
3. To correct deviation between from intended trajectory
4. To correct for thrust misalignment of a fixed nozzle during operation



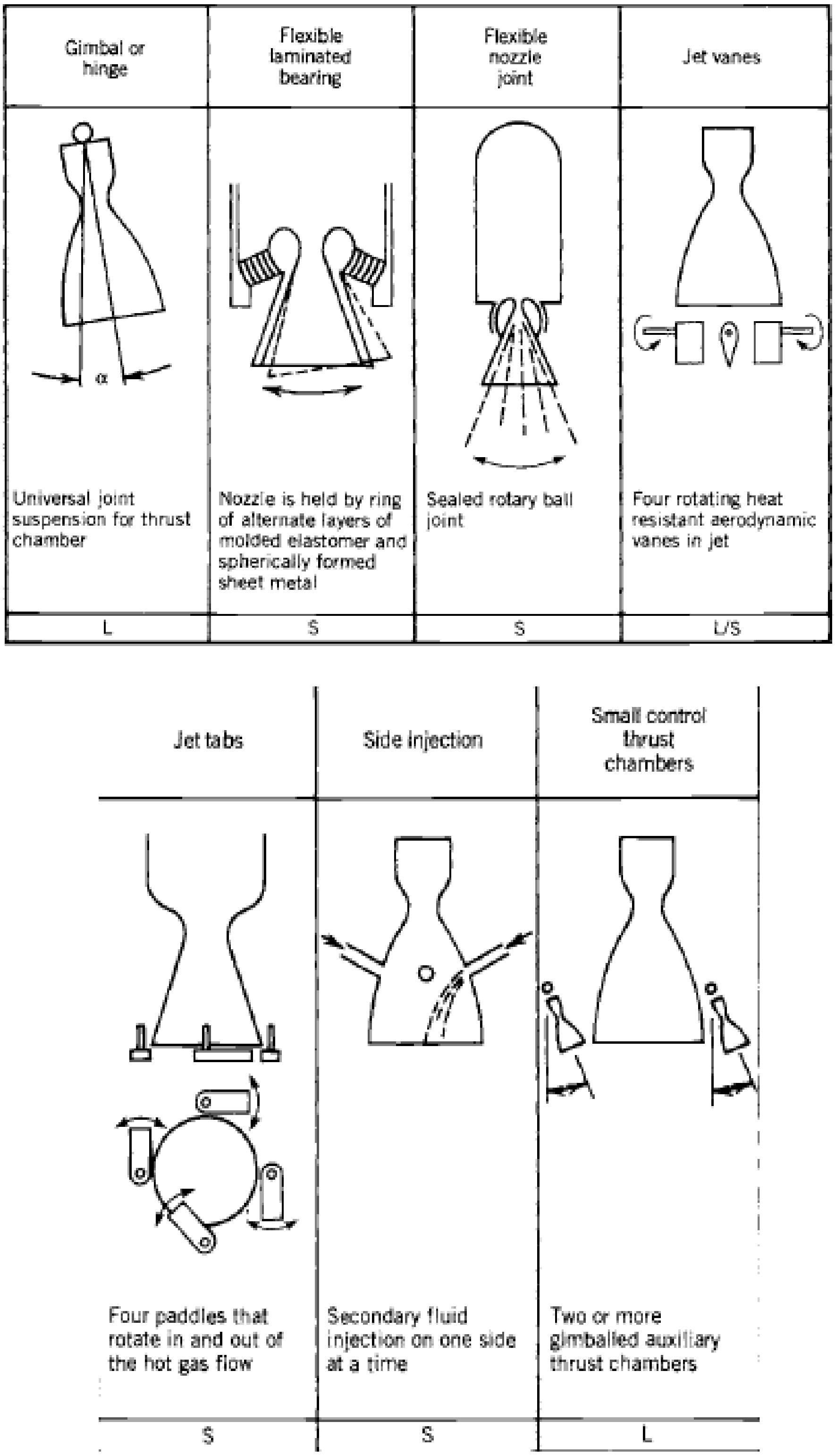
TVC mechanisms with Single Nozzle: Many different mechanisms are used on rocket systems having single nozzle. They are:

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.]



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

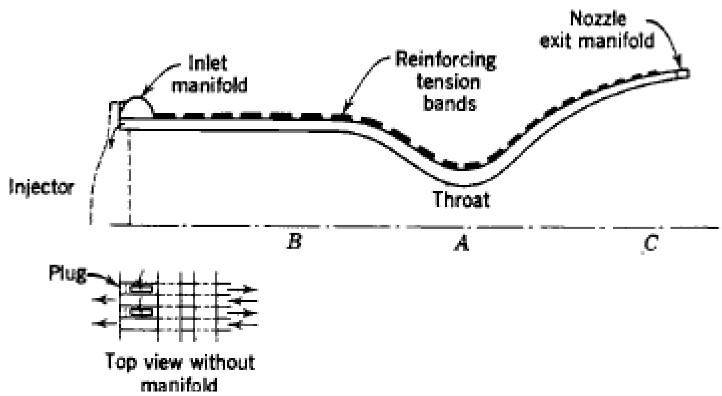
- Regenerative Cooling
- Radiation cooling.

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.

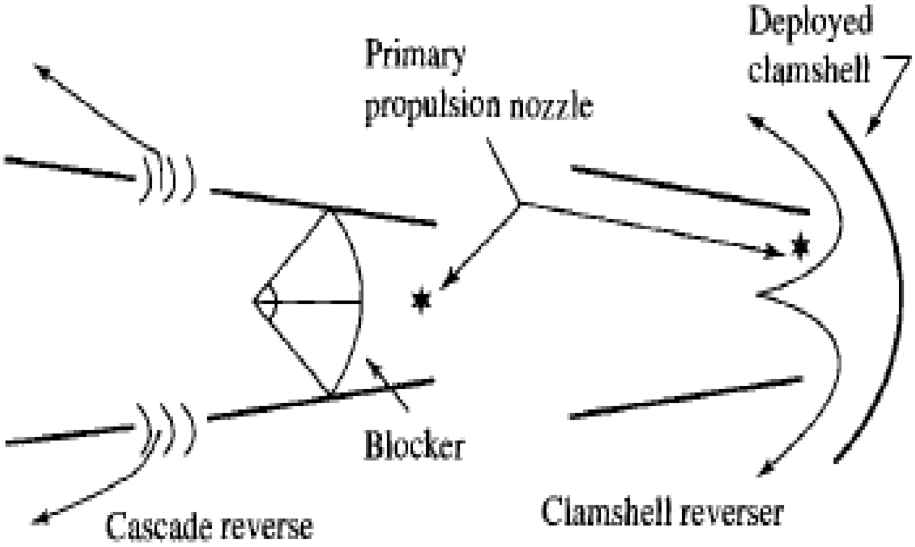
Ablative materials are usually used in nozzles, but also as insulation materials. They are usually composite materials like high silica glass, Kevlar etc.

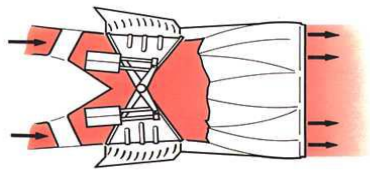
The ablation process is a combination of surface melting, sublimation, evaporation and decomposition in depth. Progressive layers of ablative material undergo endothermic degradation, that is, physical and chemical changes that absorb heat. While some of the ablative material evaporates, remaining becomes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

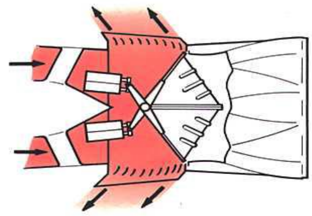
Film Cooling and Special Insulation are supplementary techniques used to locally augment their cooling capabilities

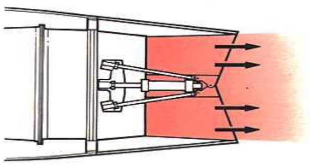
Thrust Reversers:

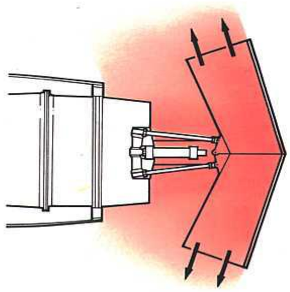
- Thrust Reversers are used in commercial transport aircraft to supplement brakes
- In-flight thrust reversal has been shown to enhance combat effectiveness of fighter aircraft
- Two types of thrust reversers are used
- Cascade-Blocker type and
- Clamshell type

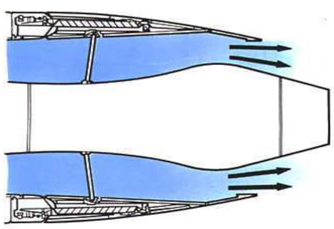


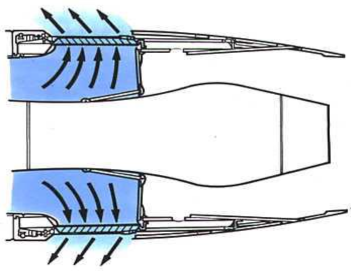

CLAMSHELL DOORS IN FORWARD THRUST POSITION


CLAMSHELL DOORS IN REVERSE THRUST POSITION


ACTUATOR EXTENDED AND BUCKET DOORS IN FORWARD THRUST POSITION


ACTUATOR AND BUCKET DOORS IN REVERSE THRUST POSITION


COLD STREAM REVERSER IN FORWARD THRUST POSITION


COLD STREAM REVERSER IN REVERSE THRUST POSITION

Cascade-Blocker Type:

- Primary nozzle exit is blocked off
- Cascades are opened ahead of the blocker in the upstream of nozzle duct to reverse the flow

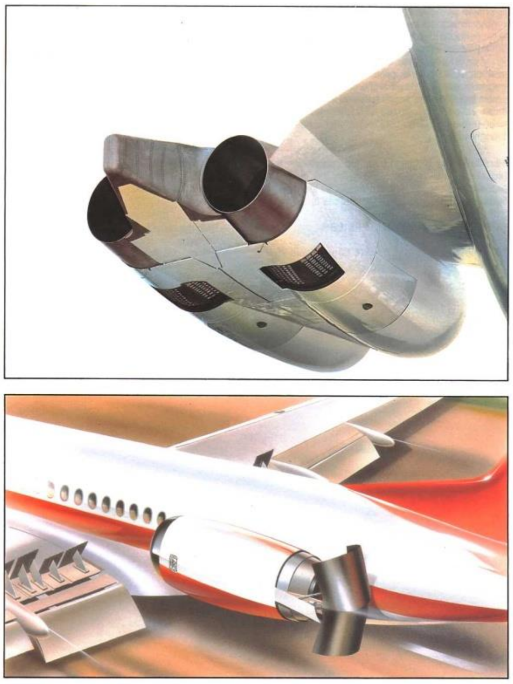
Clamshell type:

- Clamshell slides back blocking exhaust path partially
- Exhaust jet is split and reversed by clamshell

In both types, the effective nozzle throat area increases during the brief transitory period, to prevent compressor stall

- Bypass engines use cascade-blocker type in bypass/fan nozzle

Clamshell Doors-Thrust Reversing



Liquid Propellant Feed Systems:

There are two types of *feed systems* used for liquid propellant rocket engines: those that use pumps for moving the propellants from their flight

vehicle tanks to the thrust chamber, and those that use high-pressure gas for expelling or displacing their propellants from their tanks. They are discussed further in Chapter 10 and in Section 6.2 of this chapter.

Provide Control

Valves to control pressurization and flow to the thrust chambers (start/stop/throttle)
Sensors to measure temperatures, pressures, valve positions, thrust, etc., and computers to monitor/analyze system status, issue command signals, and correct if sensed condition is outside predetermined limits
Manned vehicle can require system status display and command signal override

Enhance Reliability

Fewest practical number of components/subassemblies
Ability to provide emergency mode engine operation, such as return of Space Shuttle vehicle to landing
Filters to catch dirt in propellant lines, which could prevent valve from closing or small injector

Provide for Reusability

Provisions to drain remaining propellants or pressurants
Provision for cleaning, purging, flushing, and drying the feed system and refilling propellants and pressurizing gas in field
Devices to check functioning of key components prior to next operation
Features to allow checking of engine calibration and leak testing after operation
Features for access of inspection devices for visual inspection at internal surfaces or components

Enable Effective Propellant Utilization

High tank expulsion efficiency with minimum residual, unavailable propellant
Lowest possible ambient temperature variation or matched propellant property variation with

the pressure regulator at a constant pressure to the propellant tanks. The check valves prevent mixing of the oxidizer with the fuel when the unit is not in an upright position. The propellants are fed to the thrust chamber by opening valves. When the propellants are completely consumed, the pressurizing gas can also scavenge and clean lines and valves of much of the liquid propellant residue. The variations in this system, such as the combination of several valves into one or the elimination and addition of certain components, depend to a large extent on the application. If a unit is to be used over and over, such as space-maneuver rocket, it will include several additional features such as, possibly, a thrust-regulating device and a tank level gauge; they will not be found in an expendable, single-shot unit, which may not even have a tank-drainage

6.6. TURBOPUMP FEED SYSTEMS AND ENGINE CYCLES

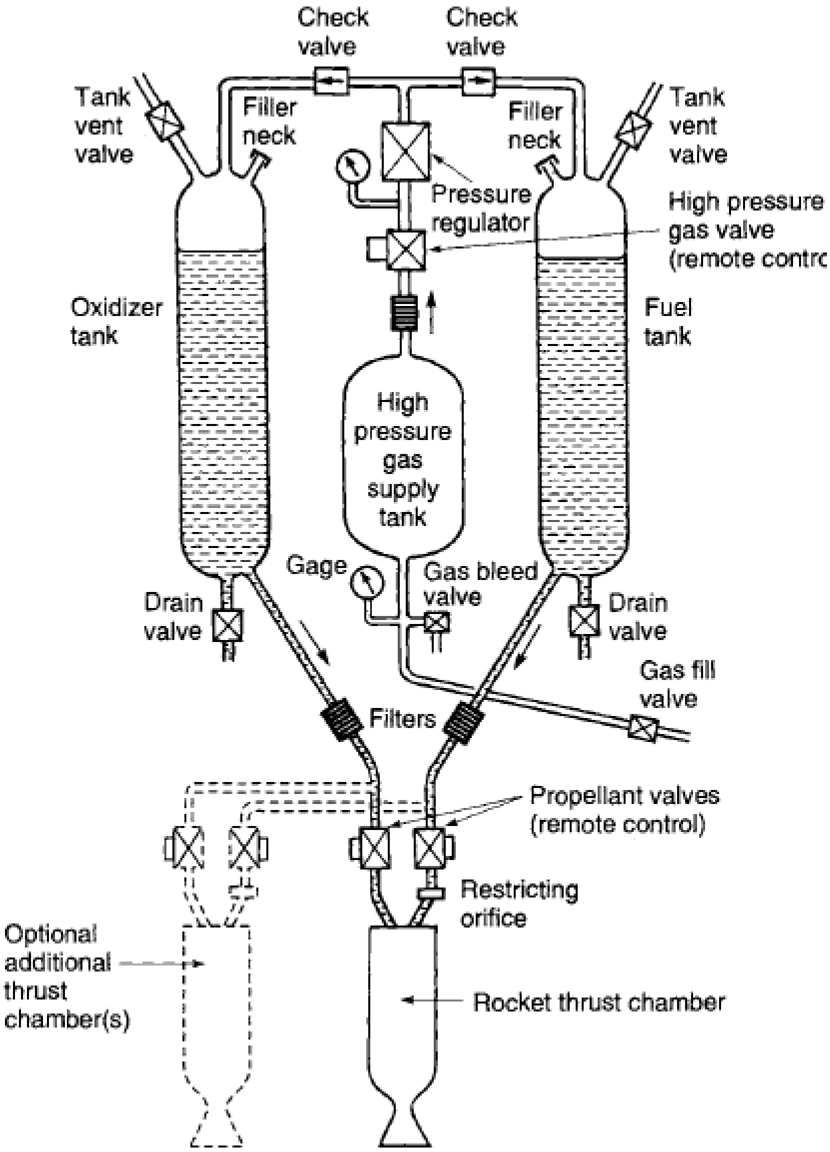
The principal components of a rocket engine with one type of turbopump system are shown in the simplified diagram of Fig. 1–4. Here the propellants are pressurized by means of *pumps*, which in turn are driven by *turbines*. These turbines derive their power from the expansion of hot gases. Engines with turbopumps are preferred for booster and sustainer stages of space launch vehicles, long-range missiles, and in the past also for aircraft performance augmentation. They are usually lighter than other types for these high thrust, long duration applications. The inert hardware mass of the rocket engine (without tanks) is essentially independent of duration. Examples can be seen In Figs. 6–1 and 6–9 and also in Refs. 6–1, 6–2, and 6–6. For aircraft performance augmentation the rocket pump can be driven directly by the jet engine, as in Ref. 6–12. From the turbopump feed system options depicted in Fig. 6–2, the designer can select the most suitable concept for a particular application.

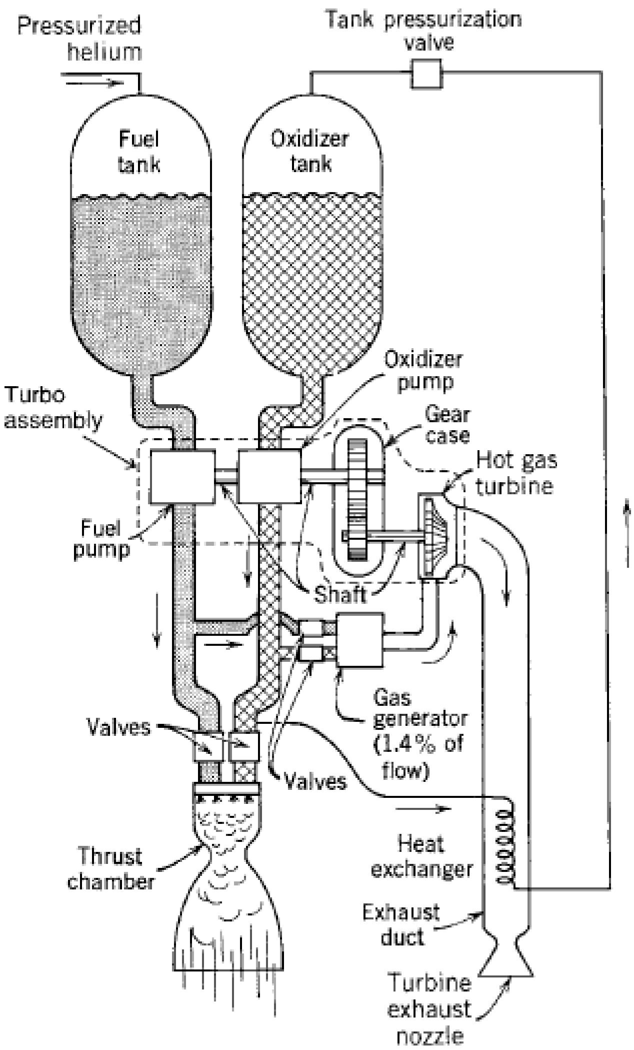
6.3. GAS PRESSURE FEED SYSTEMS

One of the simplest and most common means of pressurizing the propellants is to force them out of their respective tanks by displacing them with high-pressure gas. This gas is fed into the propellant tanks at a controlled pressure, thereby giving a controlled propellant discharge. Because of their relative simplicity, the rocket engines with pressurized feed systems can be very reliable.

Materials and Fabrication:

The choice of the materials for inner walls of the thrust chamber and throat region is influenced by hot gases resulting from the propellant combination, maximum wall temperature and the duty cycle. For high performance, regeneratively cooled thrust chambers, copper alloy with small additives of zirconium, silver or silicon, is preferred. Copper has good thermal conductivity and will not oxidize, preventing corrosion and also facilitate thin wall design.





Additional Study Material:

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

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

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

F = $V_e \frac{dm}{dt}$
where V_e is the exhaust gas velocity.

The rocket equation for the velocity increment ΔV is

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

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

Velocity increment needed for launch:

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

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

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

Deductions fromTsiolkovsky's Rocket equation:

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

Types of propulsion systems along with maximum ΔV are given below

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

Mission Velocity/Delta V budget:

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

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

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

Where ΔV_g is required to overcome the gravitational potential

ΔV_{drag} is required to overcome the drag encountered by the vehicle while in earth's atmosphere

ΔV_{orbit} is the required velocity increment for the vehicle to reach given orbit

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

To Calculate ΔV required for launching a satellite in to LEO at 100 km:

- To calculate ΔV_g: See work out. ΔV_g(100km LEO)= 1.4 km/s
- To calculate ΔV_{orbit} : See work out. ΔV_{orbit}(100km LEO)=7.76 km/s
- ΔV_{drag}= 0.1 km/s; $\Delta V_{initial}$=0.4 km/s
- ΔV(100km LEO) = (1.4+0.1+7.76-0.4) = 8.9 km/s

NOZZLES:

Flow Losses:

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

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

- In some rockets, the gaseous working fluid contains many small liquid droplets and/or solid particles that must be accelerated by the gas.
- This occurs in with solid propellants and some gelled liquid propellants which contain aluminum powder that forms small oxide particles in the exhaust.
- It can also occur with ion oxide catalysts, or propellants containing beryllium, boron or zirconium.
- In general, if the particles are very small, with diameters of 0.005mm or less, they will have almost same velocity as the gas and will be in thermal equilibrium with the nozzle gas flow
- The solid/liquid particles give up heat to the gas during expansion in a nozzle.
- As the gases give up kinetic energy to accelerate the particles, they gain thermal energy from the particles.
- As the particle diameter become bigger, the larger particles do not move as fast as the gas and do not give up heat as readily as the small particles.
- The larger particles have a lower momentum and they reach nozzle exit at a higher temperature than the smaller particles.
- For larger particles, over 0.015 mm diameter, the specific impulse can be 10 to 20 % less than the specific impulse value without flow lag.

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

- **Frozen Equilibrium:** The composition of the combustion products is invariant, that is, no change in gas composition. There are no chemical reactions or phase changes in the nozzle flow. The product composition remains same from nozzle inlet to exit. This method is usually simple, but underestimates the performance by 1 to 4%.
- In the frozen flow case, no chemical change occurs during expansion, there are no rate processes at all occurring, the molecules preserving their identity all the way.

Shifting Equilibrium:

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

- In the shifting equilibrium flow case, reactions do occur, their rate is so high (compared to the expansion rate) that conditions adjust continuously to maintain equilibrium at the local pressure and enthalpy level.
- With the result, the whole process can be regarded as reversible (and hence isentropic)
- The actual expansion process in a rocket or ramjet nozzle is intermediate between the extremes of frozen and shifting equilibrium flow.
- The equilibrium flow produces higher performance due to recovery of some of the chemical energy tied up in the decomposition of complex molecular species in the chamber - a kind of afterburning effect.

mass of useful propellant. Values of F/w are given in Table 2–1. The *thrust to weight ratio* is useful to compare different types of rocket systems.

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

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

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

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

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

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

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

The thrust (Eq. 2–5) is

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

The thrust-to-weight ratio of the vehicle is

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

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

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

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

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

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

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

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

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

The *effective exhaust velocity* as defined by Eq. 2–6 applies to all rockets that thermodynamically expand hot gas in a nozzle and, indeed, to all mass expulsion systems. From Eq. 2–14 and for constant propellant mass flow this can be modified to

$$c = v_2 + (p_2 - p_3)A_2/\dot{m} \tag{2–16}$$

Equation 2–6 shows that c can be determined from thrust and propellant flow measurements. When $p_2 = p_3$, the effective exhaust velocity c is equal to the average actual exhaust velocity of the propellant gases v_2. When $p_2 \neq p_3$ then $c \neq v_2$. The second term of the right-hand side of Eq. 2–16 is usually small in relation to v_2; thus the effective exhaust velocity is usually close in value to the actual exhaust velocity. When $c = v_2$ the thrust (from Eq. 2–14) can be rewritten as

$$F = (\dot{w}/g_0)v_2 = \dot{m}c \tag{2–17}$$

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

$$c^* = p_1 A_t/\dot{m} \tag{2–18}$$

The characteristic velocity c^* is used in comparing the relative performance of different chemical rocket propulsion system designs and propellants; it is easily determined from measured data of $\dot{m}$, p_1, and A_t. It relates to the efficiency of the combustion and is essentially independent of nozzle characteristics.

Example 2–2. The following measurements were made in a sea level test of a solid propellant rocket motor:

Burn duration	40 sec
Initial mass before test	1210 kg
Mass of rocket motor after test	215 kg
Average thrust	62,250 N
Chamber pressure	7.00 MPa
Nozzle exit pressure	0.070 MPa
Nozzle throat diameter	0.0855 m
Nozzle exit diameter	0.2703 m

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

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

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

The nozzle areas at the throat and exit are

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

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

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

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

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

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

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

The propellant *mixture ratio* for a bipropellant is the ratio at which the oxidizer and fuel are mixed and react to give hot gases. The mixture ratio r is defined as the ratio of the oxidizer mass flow rate $\dot{m}_o$ and the fuel mass flow rate $\dot{m}_f$ or

$$r = \dot{m}_o/\dot{m}_f \tag{6–1}$$

The mixture ratio defines the composition of the reaction products. It is usually chosen to give a maximum value of specific impulse or $T_1/\mathfrak{M}$, where T_1 is the combustion temperature and $\mathfrak{M}$ is the average molecular mass of the reaction gases (see Eq. 3–16 or Fig. 3–2). For a given thrust F and a given effective exhaust velocity c, the total propellant flow is given by Eq. 2–6; namely, $\dot{m} = \dot{w}/g_0 = F/c$. The relationships between $r, \dot{m}, \dot{m}_o$, and $\dot{m}_f$ are

$$\dot{m}_o + \dot{m}_f = \dot{m} \tag{6–2}$$

$$\dot{m}_o = r\dot{m}/(r+1) \tag{6–3}$$

$$\dot{m}_f = \dot{m}/(r+1) \tag{6–4}$$

These same four equations are valid when w and $\dot{w}$ (weight) are substituted for m and $\dot{m}$. Calculated performance values for a number of different propellant combinations are given for specific mixture ratios in Table 5–5. Physical properties and a discussion of several common liquid propellants and their safety concerns are described in Chapter 7.

Example 6–1. A liquid oxygen–liquid hydrogen rocket thrust chamber of 10,000-lbf thrust operates at a chamber pressure of 1000 psia, a mixture ratio of 3.40, has exhaust products with a mean molecular mass of 8.9 lbm/lb-mol, a combustion temperature of 4380°F, and a specific heat ratio of 1.26. Determine the nozzle area, exit area for optimum operation at an altitude where $p_3 = p_2 = 1.58$ psia, the propellant weight and volume flow rates, and the total propellant requirements for 2 min of operation. Assume that the actual specific impulse is 97% of the theoretical value.

SOLUTION. The exhaust velocity for an optimum nozzle is determined from Eq. 3–16, but with a correction factor of g_0 for the foot-pound system.

$$\begin{aligned} v_2 &= \sqrt{\frac{2g_0k}{k-1}\frac{R'T_1}{\mathfrak{M}}\left[1-\left(\frac{p_2}{p_1}\right)^{(k-1)/k}\right]} \\ &= \sqrt{\frac{2\times 32.2\times 1.26}{0.26}\frac{1544\times 4840}{8.9}(1-0.00158^{0.205})} = 13{,}900 \text{ ft/sec} \end{aligned}$$

The theoretical specific impulse is c/g_0, or in this case v_2/g_0 or $13{,}900/32.2 = 431$ sec. The actual specific impulse is $0.97 \times 431 = 418$ sec. The theoretical or ideal thrust coefficient can be found from Eq. 3–30 or from Fig. 3–6 ($p_2 = p_3$) to be $C_F = 1.76$. The actual thrust coefficient is slightly less, say 98% or $C_F = 1.72$. The throat area required is found from Eq. 3–31.

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$$A_t = F/(C_F p_1) = 10{,}000/(1.72 \times 1000) = 5.80 \text{ in.}^2 \text{ (2.71 in. diameter)}$$

The optimum area ratio can be found from Eq. 3–25 or Fig. 3–5 to be 42. The exit area is $5.80 \times 42 = 244$ in.^{2} (17.6 in. diameter). The weight density of oxygen is 71.1 lbf/ft^3 and of hydrogen is 4.4 lbf/ft^3. The propellant weight flow rate is (Equation 2–5)

$$\dot{w} = F/I_s = 10{,}000/418 = 24.0 \text{ lbf/sec}$$

The oxygen and fuel weight flow rates are, from Eqs. 6–3 and 6–4,

$$\dot{w}_o = \dot{w}r/(r+1) = 24.0 \times 3.40/4.40 = 18.55 \text{ lbf/sec}$$
$$\dot{w}_f = \dot{w}/(r+1) = 24/4.40 = 5.45 \text{ lbf/sec}$$

The volume flow rates are determined from the densities and the weight flow rates.

$$\dot{V}_o = \dot{w}_o/\rho_o = 18.55/71.1 = 0.261 \text{ ft}^3\text{/sec}$$
$$\dot{V}_f = \dot{w}_f/\rho_f = 5.45/4.4 = 1.24 \text{ ft}^3\text{/sec}$$

For 120 sec of operations (arbitrarily allow the equivalent of two additional seconds for start and stop transients and unavailable propellant), the weight and volume of required propellant are

$$w_o = 18.55 \times 122 = 2260 \text{ lbf of oxygen}$$
$$w_f = 5.45 \times 122 = 665 \text{ lbf of hydrogen}$$
$$V_o = 0.261 \times 122 = 31.8 \text{ ft}^3 \text{ of oxygen}$$
$$V_f = 1.24 \times 122 = 151 \text{ ft}^3 \text{ of hydrogen}$$

Note that, with the low-density fuel, the volume flow rate and therefore the tank volume of hydrogen are large compared to that of the oxidizer.

Advanced Propulsion Systems-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 vehiclemass** 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:

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

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

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

For $K_\infty< 1$, no chain reaction is possible

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

$K_\infty = 1$ is the critical level and K_∞ will needs to be controlled at 1 for steady production of heat in the reactor.

The subscript$_\infty$ 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 Ƞ for U^{235} is 2.07, available for furthur fission process.

The value of Ƞ 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, with in a few milliseconds. This would make the control mechanism of moving control rods in/out of the core to regulate neutron flux very difficult.

However, the control process is helped/made effective due to presence of "delayed neutrons". The movement of about 1% of the neutrons is delayed because formation of unstable intermediate nuclei of isotopes like iodine and bromine which undergo decay during the nuclear process, but will cause induce time lag between prompt neutrons and delayed neutrons.

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

Thermal Stability:

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 raises. This is due to the fact that density of core materials increases causing them to expand, increasing the mean distance between collision and increases the probability of fission.

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.

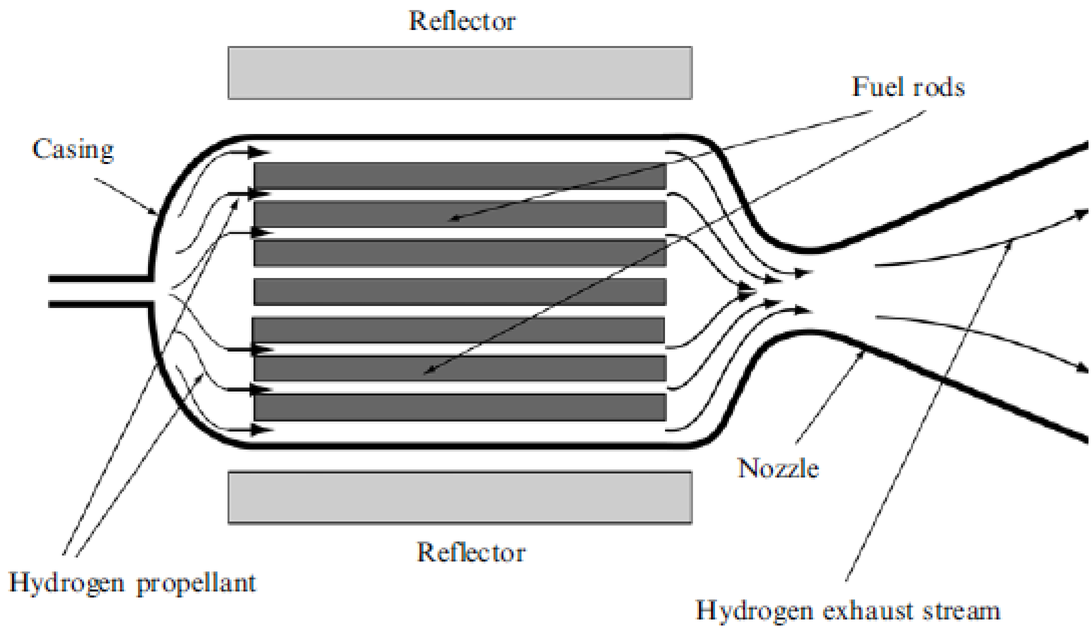
To increase the power level, value of k is allowed to become greater than 1. Once the desired power level is reached, k is returned to value of 1, and the reactor contimues to produce power at the new level. A decrease of power is also established in a similar way.

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

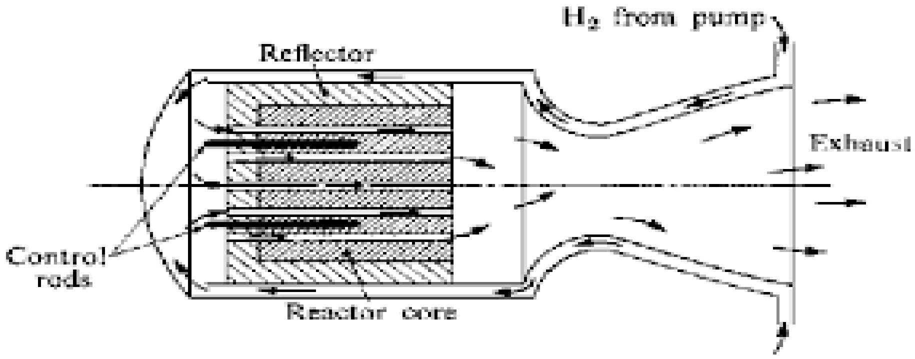
Once, the engine is in stable operating mode, the **thrust can be varied by positioning the control rods**.The **power ouput is a function of neutron flux**.

About 1% of the neutrons produced by fission are delayed.

When the reactor is shut down, the fission process and hence the power output continues to be produced. So fission heating will go on for several moments.

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.

ELECTRIC THRUSTERS-MISSION APPLICATIONS TO SPACE FLIGHT

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

1.1 **Limitations of Chemical Rocket Engines**:

- **Explosion & Fire Potential (SPR &LPR)**: Explosion and fire potential is larger failure can be catastrophic.
- **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 plume 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 Pressurization (LPR):** Tanks need to be pressurized by separate system. This needs high pressure inert gas storage for long periods of time.

Electric 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

Electric propulsion

Chemical rockets use the energy stored in the propellants to create a hot gas, which then becomes the working fluid in a heat engine, and is expelled, generating thrust. There is an elegant simplicity in this triple function of the propellant and its combustion products, which is reflected in the simple nature of the rocket engine. There is, however, a fundamental limitation which results from combining the functions of working fluid and energy source: no more energy can be put into the rocket than is contained in the propellant flowing into the engine. This means that the power output of the rocket is rigidly defined by the chemical energy and flow rate of the propellant. The exhaust velocity and thrust are defined by the thermodynamic relationships in Chapter 2, and there is no possibility of exceeding these values. As has previously been pointed out, the arrival of the space age was dependent on stretching the ability of chemical rockets to the limit, through multi-staging, and on engines that perform very close to their theoretical best. More ambitious space programmes—manned missions to Mars, for example—could be achieved with the same technology, but would require a very large effective mass ratio because of the velocity increment involved. Moreover, all the necessary propellant would need to be raised to Earth orbit. It would be preferable if somehow more propulsive power could be extracted from the propellant, and the exhaust velocity could be increased beyond the 4.5 km/s that is the best available from chemical rockets.

1.2 Electric Propulsion Systems:

1.2.1 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.

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

1 **Electrothermal**: In this type, the propellant is heated electrically and expanded thermodynamically where the gas is accelerated to supersonic speeds through a nozzle, as in chemical rockets, to produce thrust.

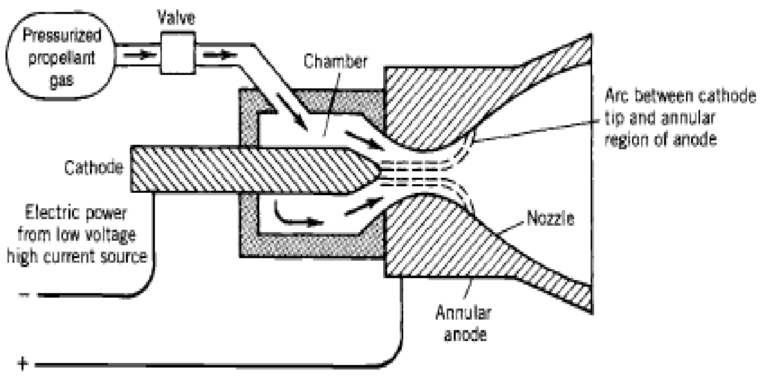
2 **Electrostatic or Ion propulsion engine:** In this type, **a**cceleration is achieved by the interaction of electrostatic fields on non-neutral or charged propellant particles such as atomic ions, droplets or colloids.

3 **Electromagnetic or Magnetoplasma engine:** In this type, the acceleration is achieved by the interaction of electric and magnetic within a plasma. The plasmas are moderately dense, high temperature gases which are electrically neutral but good conductors of electricity.

Electrothermal Thruster: Electrothermal thrusters use the simplest way to heat the propellant with a hot wire coil, through which an electric current passes. 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.

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

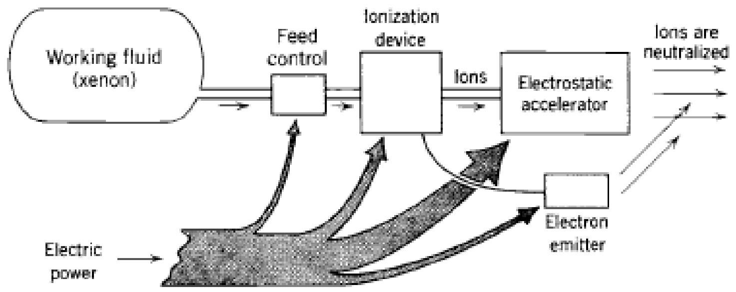
A schematic diagram ofarc-heatingelectric propulsion system is shown below. The arc plasma temperature is around 15,000 K.



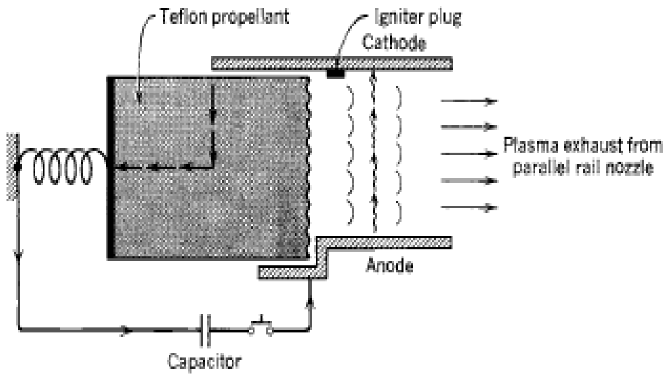
Electrostatic and Electromagnetic thrusters accomplish propulsion through different means. They do not use thermodynamic expansion of gas in the nozzle.

Both Electrostatic and electromagnetic thrusters work only in vacuum.

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



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



The thruster uses a parallel rail accelerator for self-induced magnetic acceleration of a current carrying plasma. When the capacitor is discharged, an arc is struck at the left side of the rails. The high current in the plasma arc induces a magnetic field. The action of current and the magnetic field causes the plasma to be accelerated at right angles to both the current and the magnetic field, ie in the direction of the rails.
Each time an arc is created, a small amount of propellant (Teflon), is vaporized and converted in to a small plasma cloud. The plasma is then ejected giving a small pulse of thrust. The thruster can operate with many pulses per second.
The magnetoplasma rocket is used as spacecraft attitude control engine.

1.2.3 Performance of ElectricThrusters:

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

Electric Vehicle Performance: Thepropulsive forcedeveloped 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 = $\frac{mv_e^2}{2P_E}$, 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 = mv_e = $\sqrt{2m\eta\xi M_E}$ = $\sqrt{\frac{2\eta\xi M_E M_P}{t}}$

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

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 $\times 10^6$ sec ; Pay load mass = 100 kg;

ξ = 100 W/kg; ɳ = 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 propellantM_P = $\dot{m}$t = 1.02× 10^{-5} ×2.42× 10^6 = 24.69 kg

The electrical power required is P_E= ($\frac{1}{2}\dot{m}v_e^2$)/ɳ = $\frac{1}{2}$ (1.02× 10^{-5} × 2000^2 × 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 × 9.8 ln (163.9/139.2) = 3200 m/sec

2.1 System Parameters-Interrelations:

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

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

The exhaust velocity is given by the relation

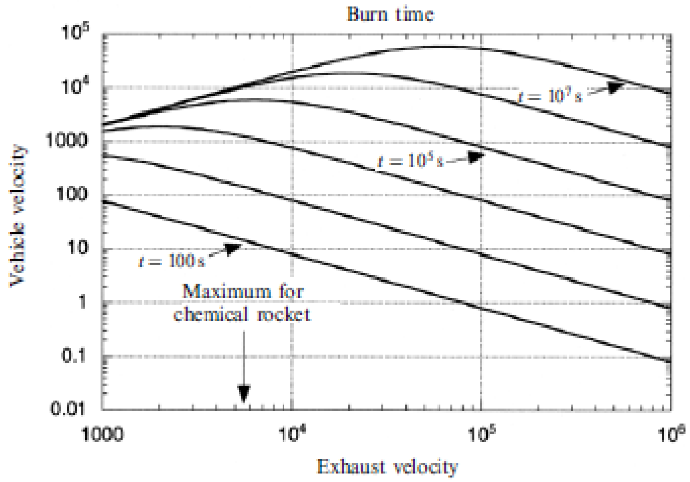
$$\frac{M_E}{M_P}=\frac{v_e^2}{2\eta\xi t}$$

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.

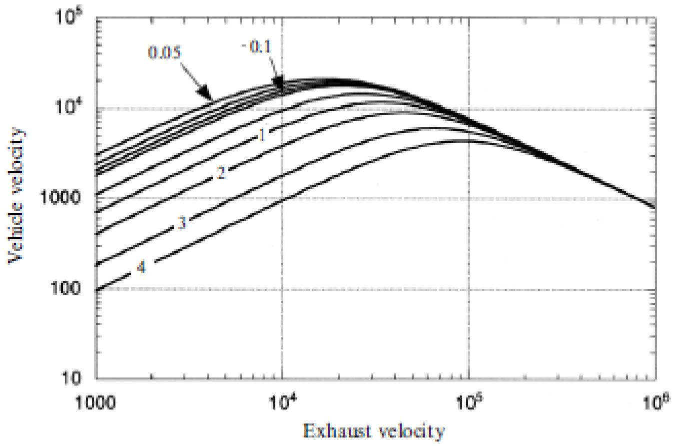
- Figure below shows vehicle velocity as a function of exhaust velocity and burn time t



- 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 aremeant for bringing saving of propellant mass. Relation between vehicle velocity as a function of the ratio of payload(structural) mass topropellant mass is indicated below:

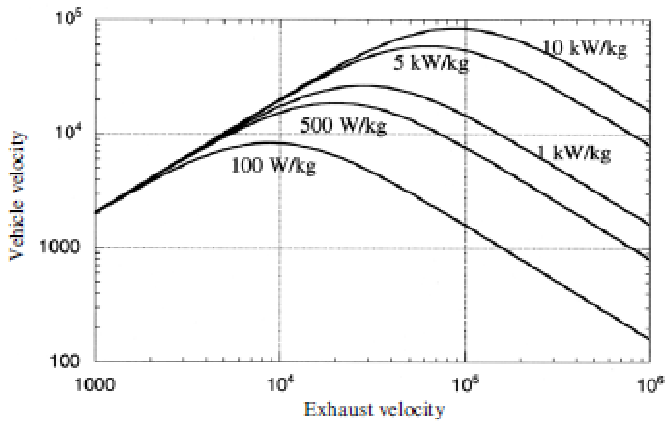
- In the interrelation below, the burn time is fixed at 1 million seconds, and the power-to-mass ratio, ξ is fixed at 500 W/kg.
- The ratio$\frac{M_S}{M_P}$ is shown as a parameter.



It is evident from the interrelation that

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

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



The above interrelation shows that

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

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

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

2.2 Electric Thrusters : Operation: Operation of

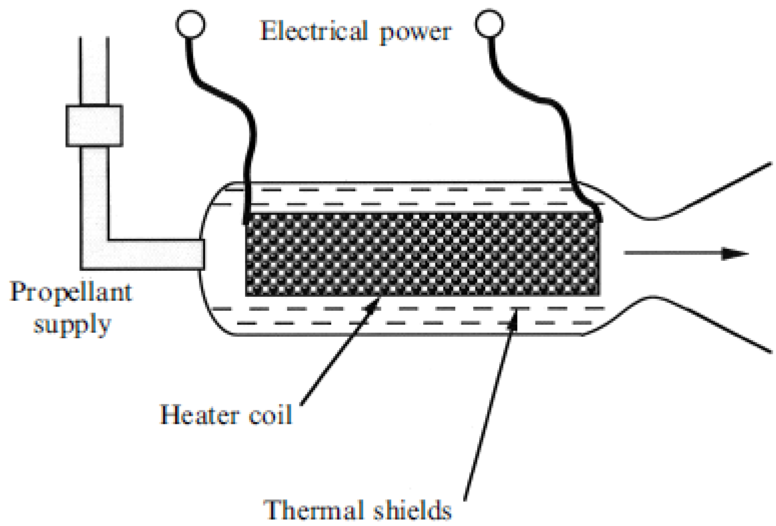
Operation of Electrical thrusters: Electrical thrusters can bedivided into two broad categories; those that use electricity to heat up the propellant, which emerges as a neutral gas; and those that use electric and magnetic fields to accelerate the ions. The functional form and analysis of these two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

2.2.1 Resiso-jet:

Operating Principle & Components: The basic electrothermal thruster, resisto-jet, consist ofa 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:



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

$v_e = C_F C^*$

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

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

- The nozzle exit temperature in chemical rockets depends on type of propellant, where as in electric thrusters, the nozzle exit temperature is an inverse of mass flow rate.
- The melting temperature of heating element limits the maximum temperature levels in the thruster.

Example: Consider following data:

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

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

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

- Electric thrusters can attain very high exhaust velocities,

The mass flow rate m is calculated from

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

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

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

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

2.2.2 Arc-Jet Thruster:

Operating Principle: In the Arc-Jet thruster, the propellant gas is heated by passing an electric arc through the flow. Temperatures in the order 30,000-50,000 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 a electrothermal thruster. The temperature limit can be much higher.

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

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

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

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

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

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

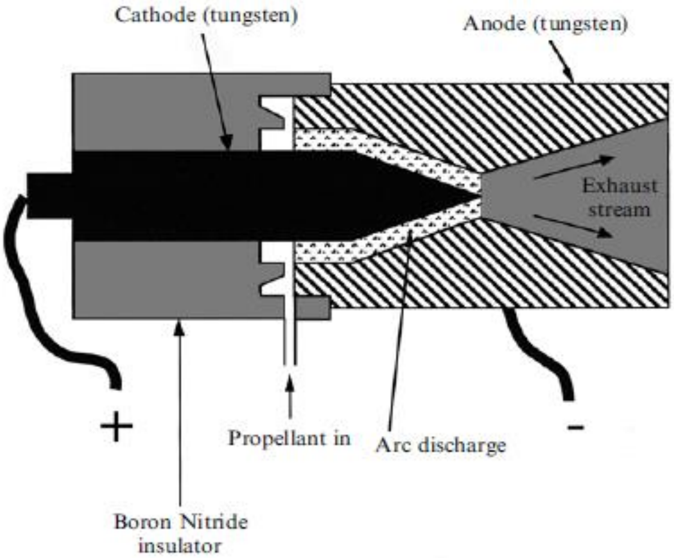


Figure 6.6. Schematic of an arc-jet thruster.

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

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

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

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

3.1 Electrostatic Thrusters:

Performance Parameters:

- 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.

3.1.2 Ion Thruster:

Working Principle: The propellant is ionized, and then enters a region of strong electric field, where the positive ions are accelerated. The ions are accelerated passing through the grid and leave the engine as a high 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.

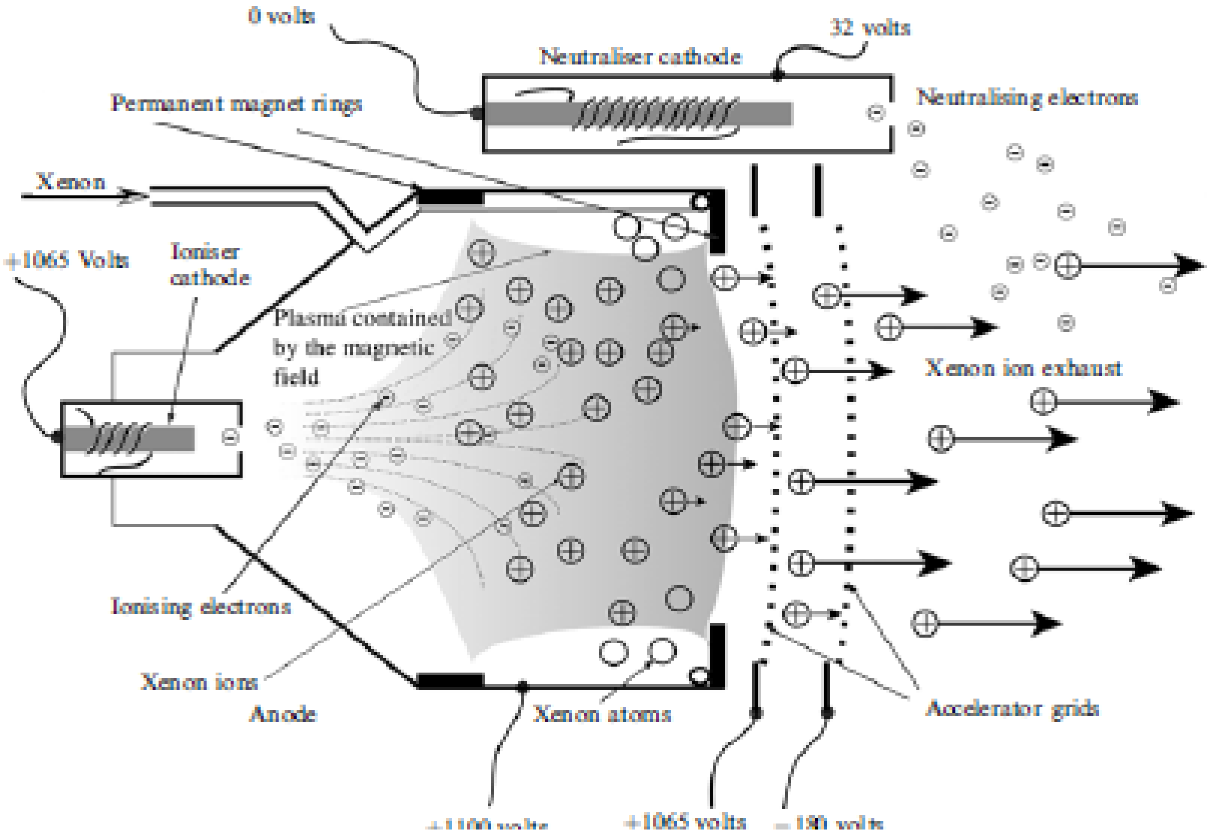
The thruster is divided into two chambers. Propellant, (usually Xenon gas) enters ionisation chamber in the form of neutral gas molecules.

The cathode at the center, emits electrons, which are accelerated by the electric field. These electrons ionisepropellant 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 missionslike inter-planetary missions and station keeping.

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

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

The accelerating grids have an electric field between them, which gets partially blocked as the ions start 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 Thrusters: The low thrust-high exhaust velocity ion thrusters are limited by space-charge limit. Plasma thrusters (electromagnetic thrusters) offer higher thrust values.

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

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

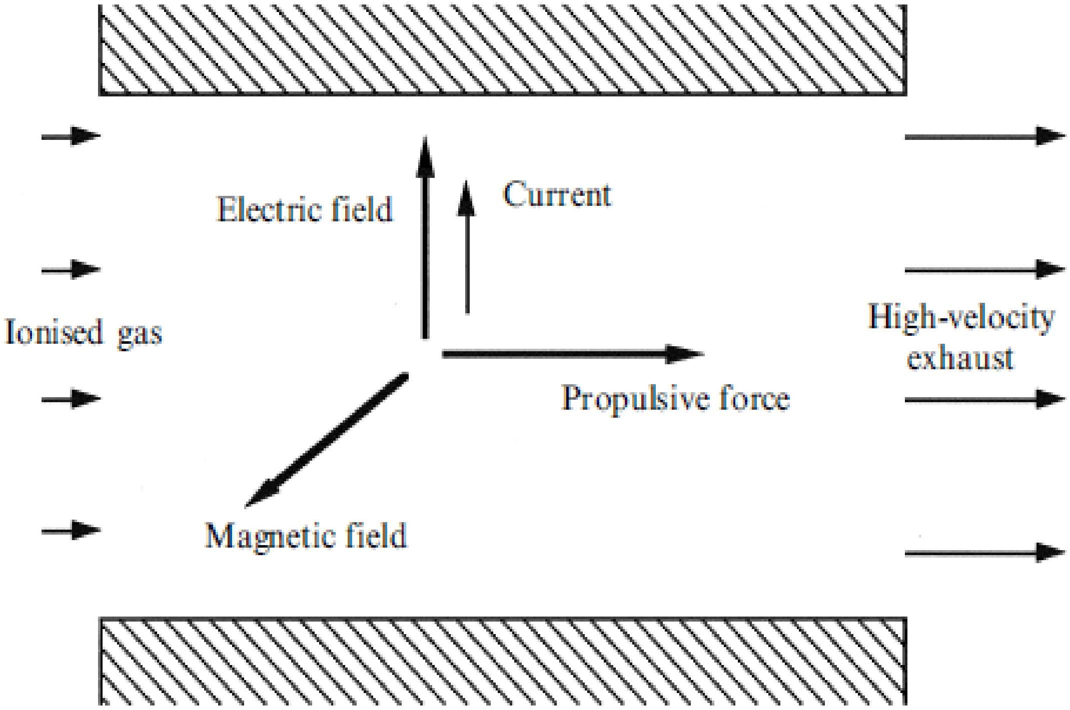
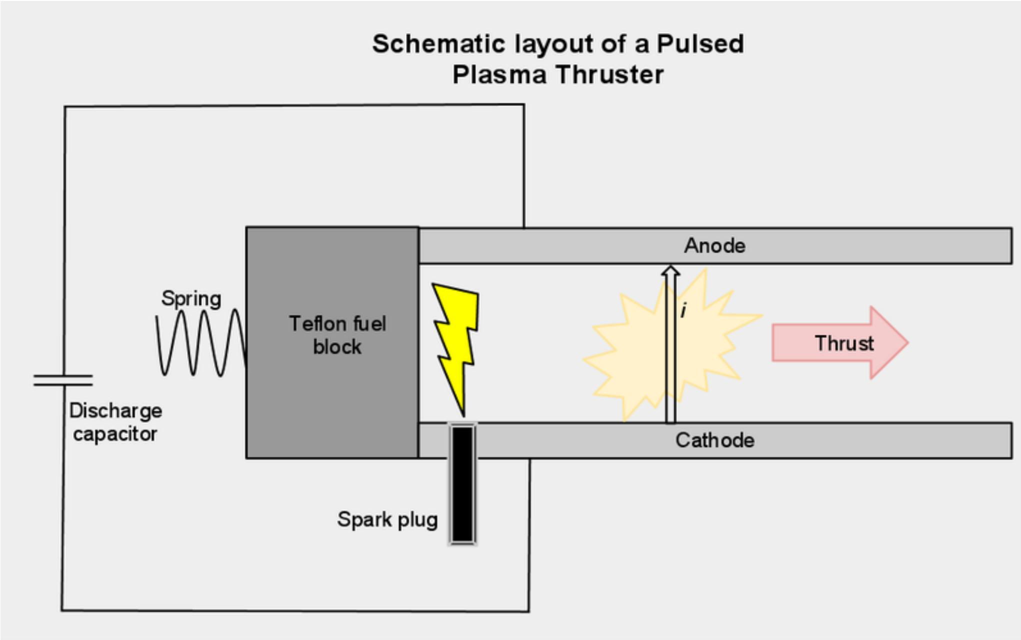


Figure 6.14. Principle of the plasma thruster.

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

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

The operation of MPD thrusters is similar to pulsed thrusters.

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

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

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

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

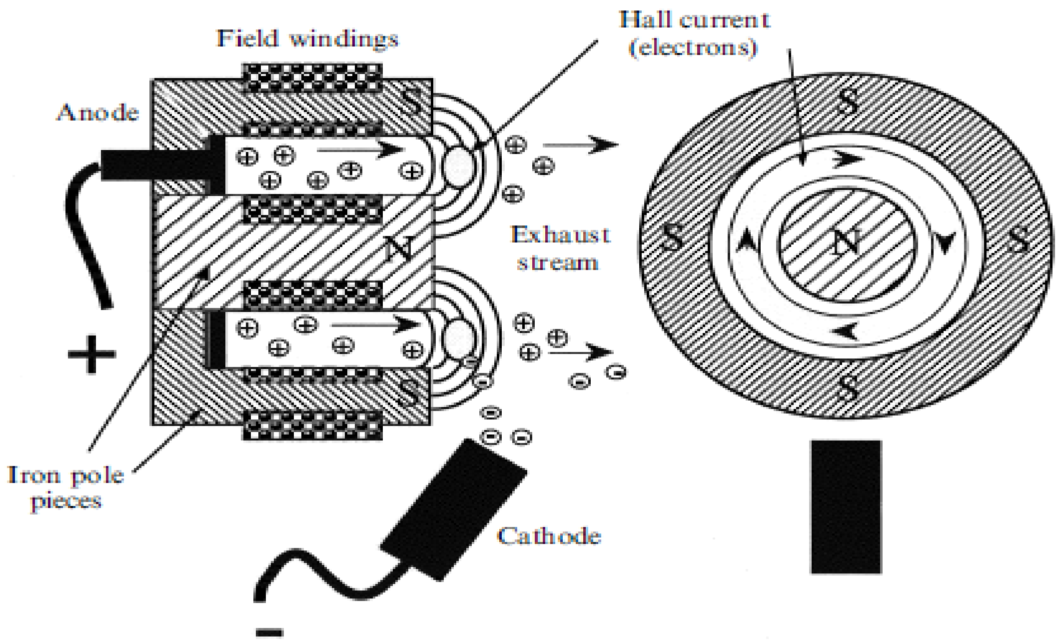


Figure 6.16. Schematic of the Hall thruster.

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.

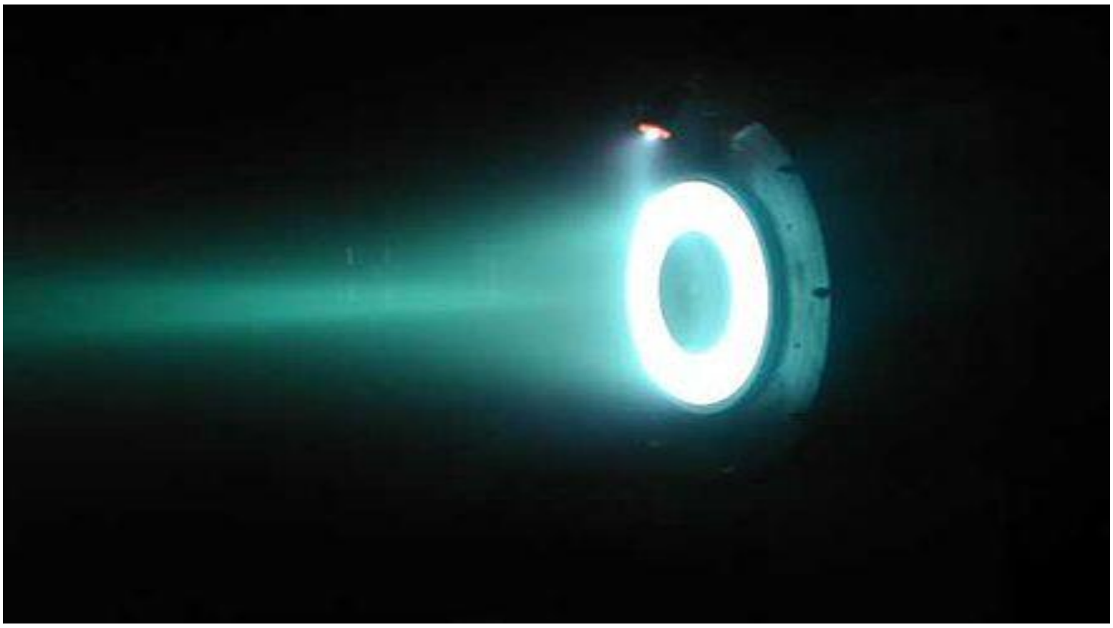
Thrust levels are very small, around 80mN for a 300 V, 1.5 W thruster.

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

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

The applications of Hall-effect thrusters include control of orientation & position of orbiting satellites and to power the main propulsion engine for medium-size robotic space vehicles.

2 KW Hall Thruster in Operation



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

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

Magnetoplasmadynamic (**MPD) thruster** is a form of electromagnetic propulsive vehicle which uses Lorentz force (the force on the charged particle by an electromagnetic field) to generate thrust. It is also referred to as Lorentz Force Accelerator (LFA) or MPD arcjet.

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.

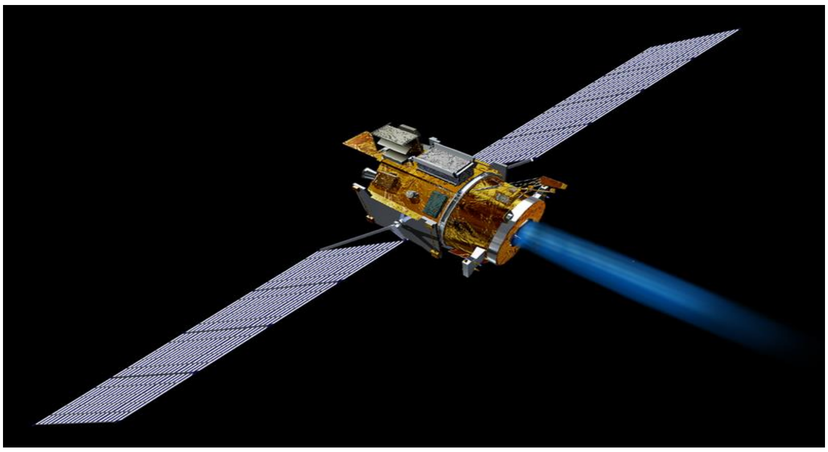
Solar Electric Propulsion (SEP):

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.

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 10,000 km^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.

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.
- Since 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.
- The other concept is the Mini-magnetosphere (M2P2); where a large scale magnetic field would be produced by a magnetic dipole on board a spacecraft and a plasma generated by a plasma generator would be injected in to the magnetic field. The magnetic field will then expand a few kilometers in diameter producing a magnetic sail.

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.

Launch Assist Technologies: A completely reusable, single-stage-to-orbit (SSTO) launch vehicle with good payload capacity is difficult to achieve with purely on-board chemical propulsion systems.

Launch assist technologies are under study which include use of electromagnetic energy or use of external energy sources.

Reduction of Required Δv: A small decrease of the required Δv will increase payload capability. This can be achieved by

- Launching the spacecraft from an aircraft with an initial velocity
- Providing an initial boost with a chemical/electromagnetic catapult
- Launching outside the atmosphere or from am an altitude.