Aerospace Propulsion II

III YEAR II SEMISTER

DEPARTMENT OF AERONAUTICAL ENGINEERING

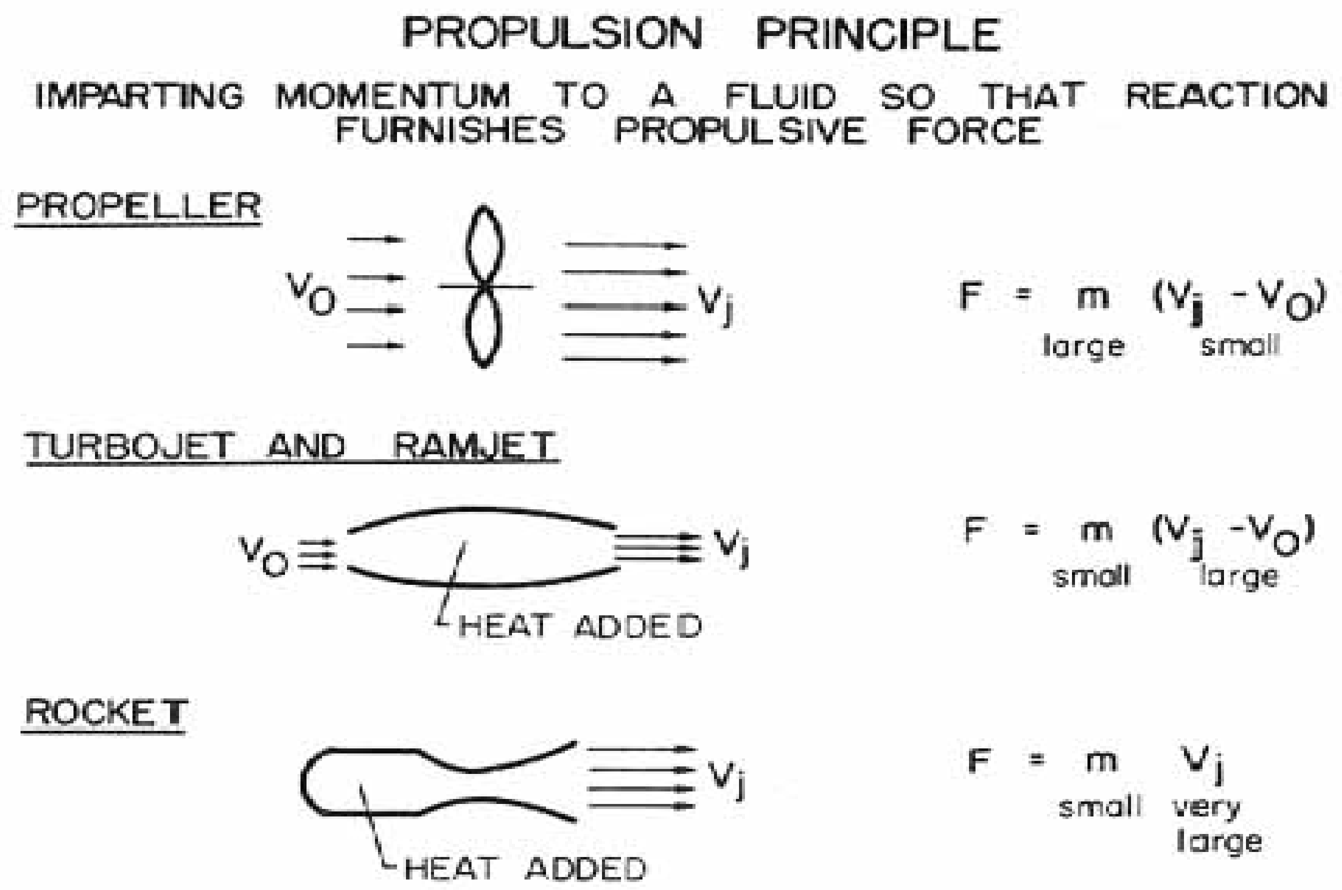
INSTITUTE OF AERONAUTICAL ENGINEERING

Prepared by
C.Sandeep
Assistant professor

Propulsion

Propulsion: The act of changing a body's motion from mechanisms providing force to that body

Jet Propulsion: Reaction force imparted to device by momentum of ejected matter



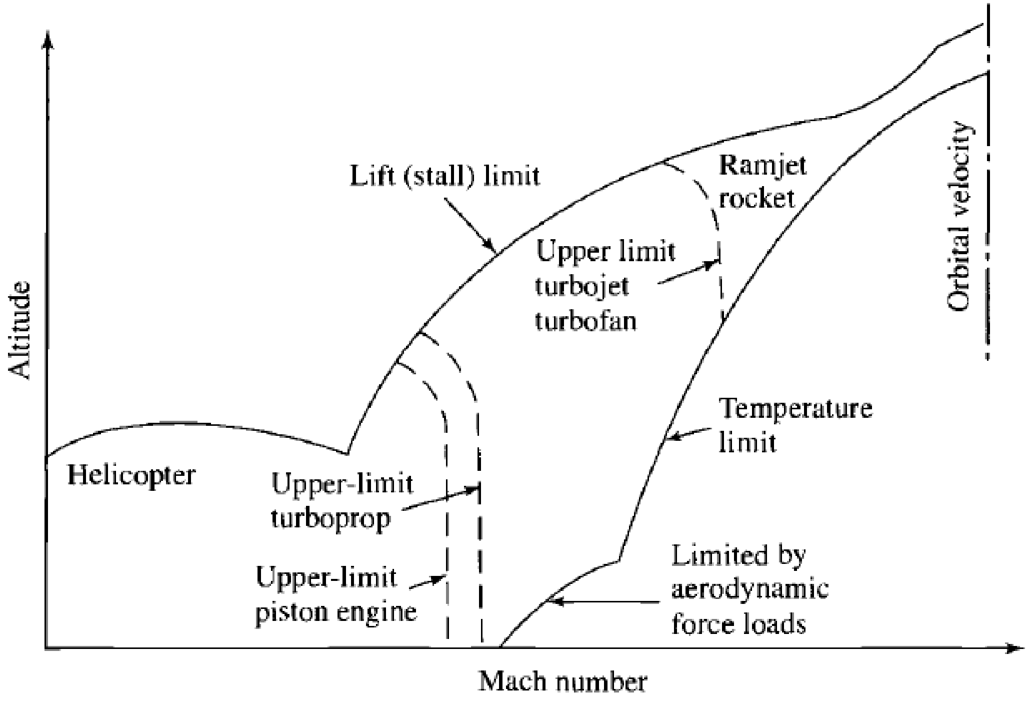
Air-Breathing (Ducted) Propulsion: Devices that use surrounding medium as the "working fluid" along with some amount of stored fuel

Rocket Propulsion: Produces thrust by ejecting stored matter

Operational Envelopes

Each engine type will operate only within a certain range of altitudes and Mach numbers (velocities).

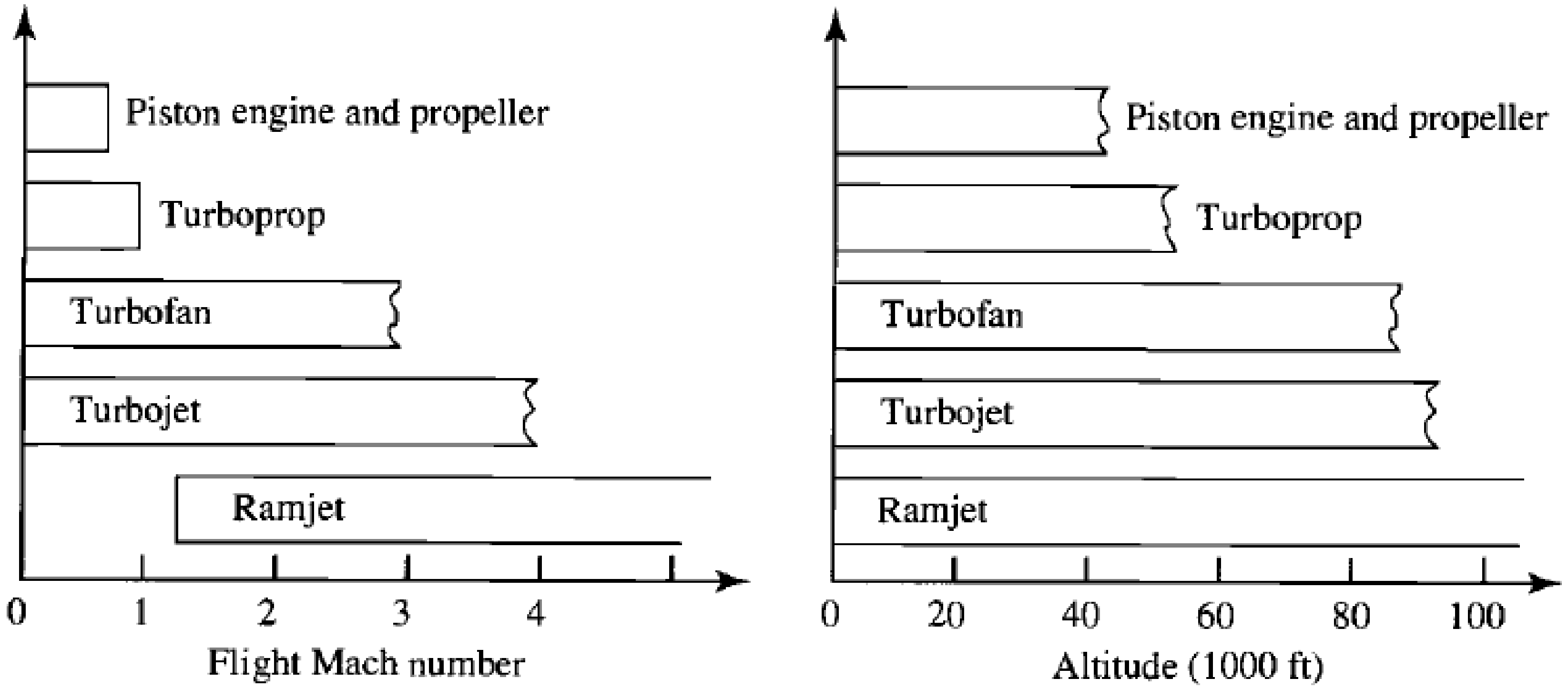
The approximate velocity and altitude limits, or *corridor of flight, within which airlift vehicles can operate. The corridor is* bounded by a *lift limit, a temperature limit, and an aerodynamic force limit.*



Flight limits.

Operational Envelopes

The lift limit is determined by the maximum level-flight altitude at a given velocity. The temperature limit is set by the structural thermal limits of the material used in construction of the aircraft. At any given altitude, the maximum velocity attained is temperature-limited by aerodynamic heating effects. At lower altitudes, velocity is limited by aerodynamic force loads rather than by temperatu

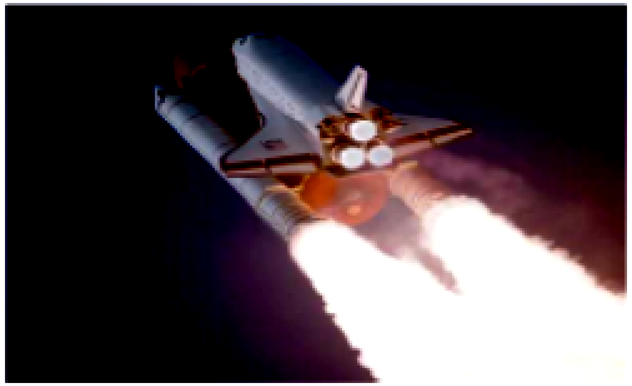


Engine operational limits.

Propulsion Systems

Jet Propulsion

- **Air-Breathing Propulsion**
  - **Turbomachinery Based**
    - Turbojet
    - Turbofan
    - Turboprop
  - **Pure-duct Based**
    - RAMJET
    - SCRAMJET
- **Rocket Propulsion**
  - **Liquid**
    - Saturn V
    - NASA SLS
  - **Solid**
    - Shuttle Side Boosters



Combinations of both ducted and rocket systems may be attractive for some applications

Propulsion

Energy sources in propulsion systems can vary:

- Chemical Energy
- Solar Radiation Energy
- Nuclear Energy
- Electrical Energy

Regardless of energy source, all basically rely on adding energy to a mass of **propellant** thereby accelerating it to generate thrust (force)

Propellant: Ejected stored matter that causes thrust

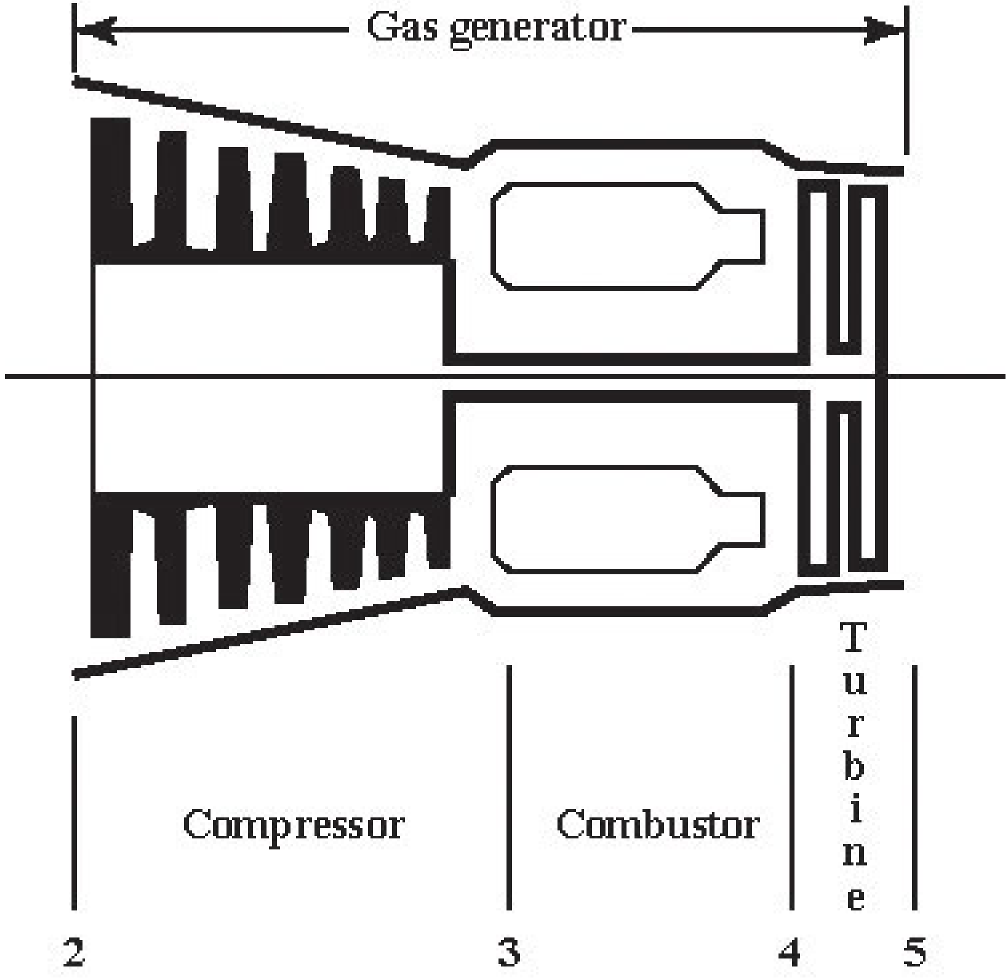
Air Breathing Propulsion

- Air breathing propulsion systems use oxygen in atmospheric air to burn fuel stored on the vehicle
- Turbojet
- Turbofan (High BR, Low BR, Afterburning)
- Turboprop
- RAMJETS
- SCRAMJETS

Air Breathing Propulsion: Gas Turbine Systems

Gas Generator

- The basis of turbojet, turbofan, and turboprop propulsion is the gas generator
- Supplies high-temperature, high-pressure gas
- Stand alone, most of the energy of this device is used to drive turbines
- Turbine rotational energy is converted into electricity



Air Breathing Propulsion

Turbojet

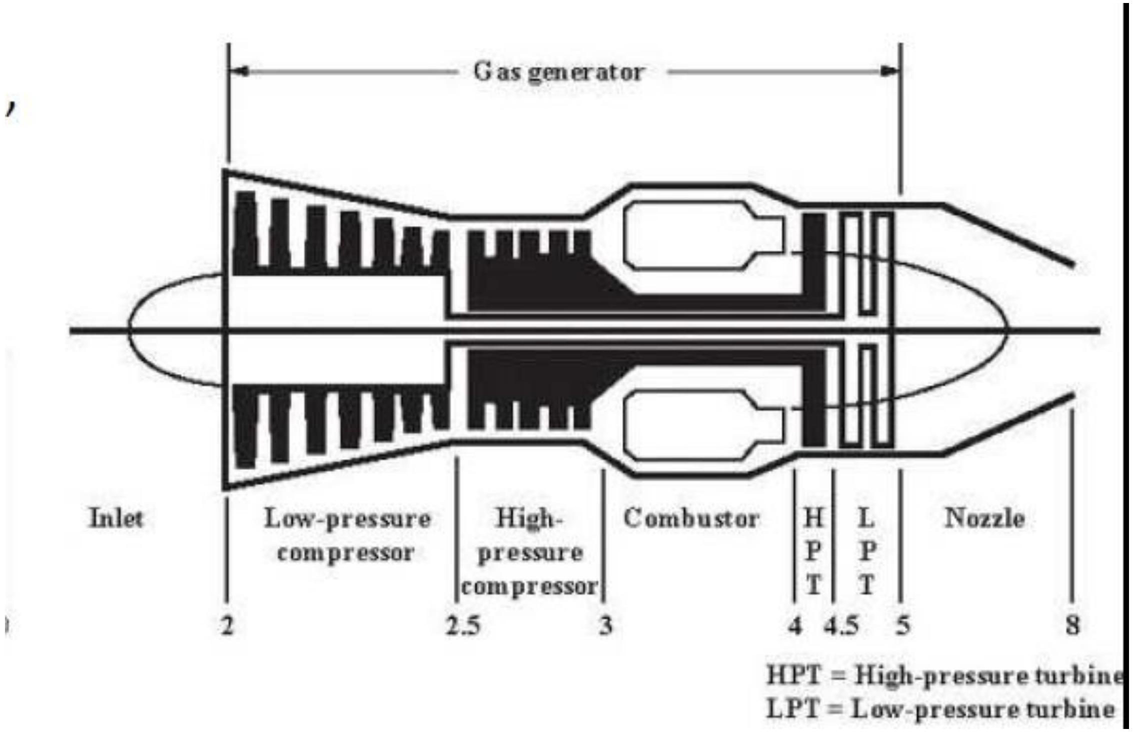
By adding an inlet and a nozzle a turbojet can be constructed

• Gas generator still supplies high-temperature, high-pressure gas

• Some of the energy of this device is used to drive turbines and auxiliary systems

• Most of the energy in the high-temperature, high-pressure gas is allowed to flow to the nozzle

• Nozzle accelerates flow to high velocity to impart thrust



Propulsive Efficiency

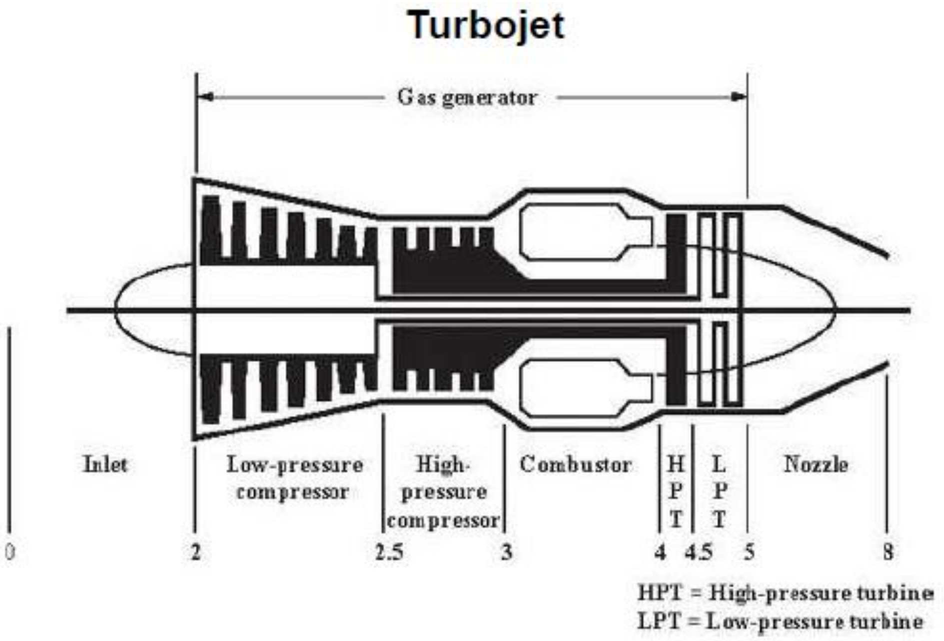
A measure of how effectively engine power is used to propel aircraft

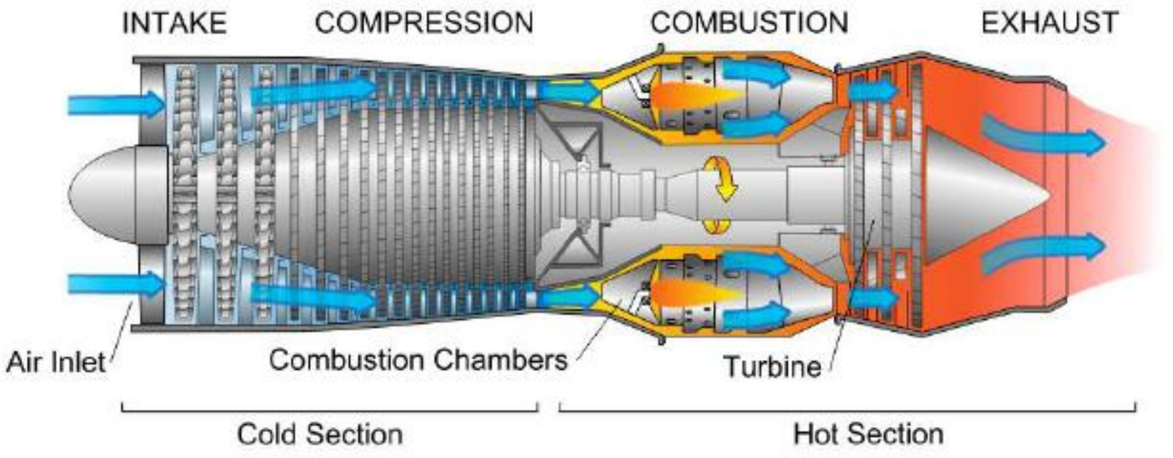
Propulsive Efficiency

- A measure of how effectively engine power is used to propel aircraft

$$\eta_P = \frac{TV_o}{\dot{W}_{out}} = \frac{TV_o}{\frac{1}{2g_c}\left[(\dot{m}_o + \dot{m}_f)V_e{}^2 - \dot{m}_o V_o{}^2\right]}$$

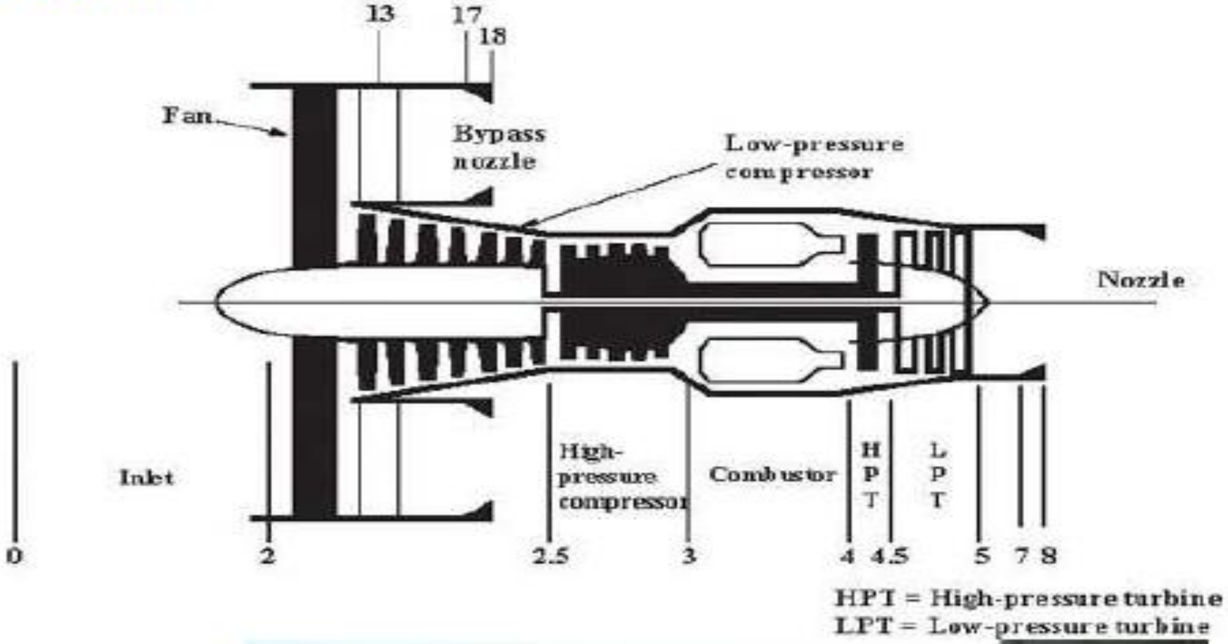
Air Breathing Propulsion: Gas Turbine Systems



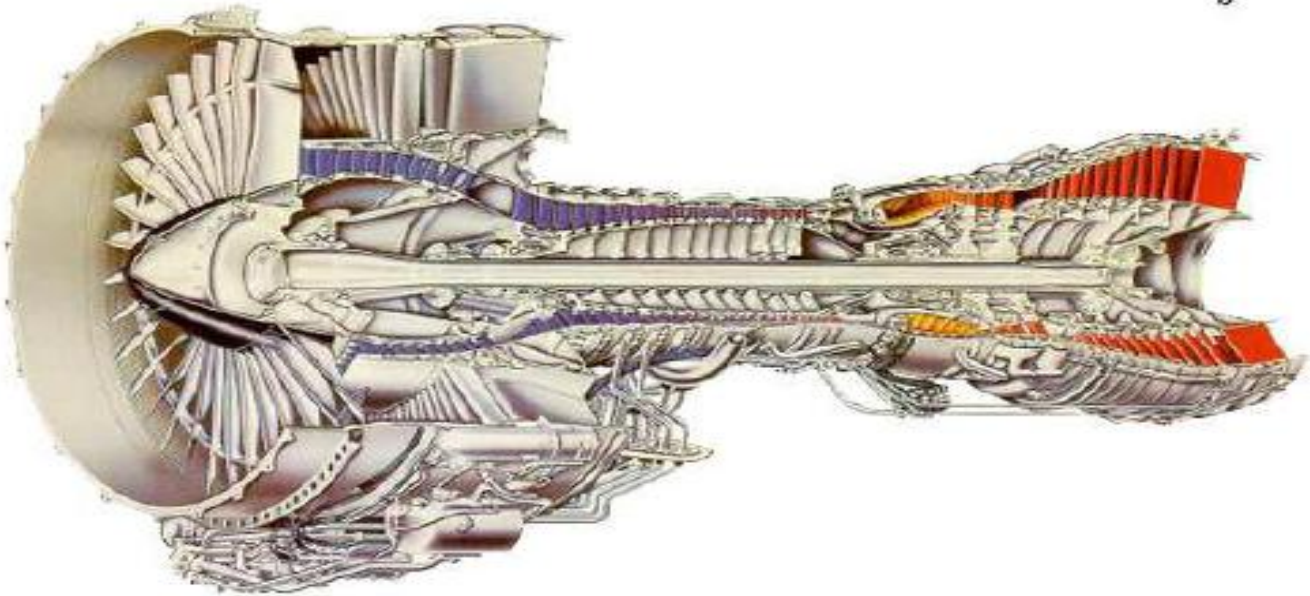


Air Breathing Propulsion: Gas Turbine Systems

Turbofan



Turbofans can be High Bypass Ratio which are normally used for commercial transport…

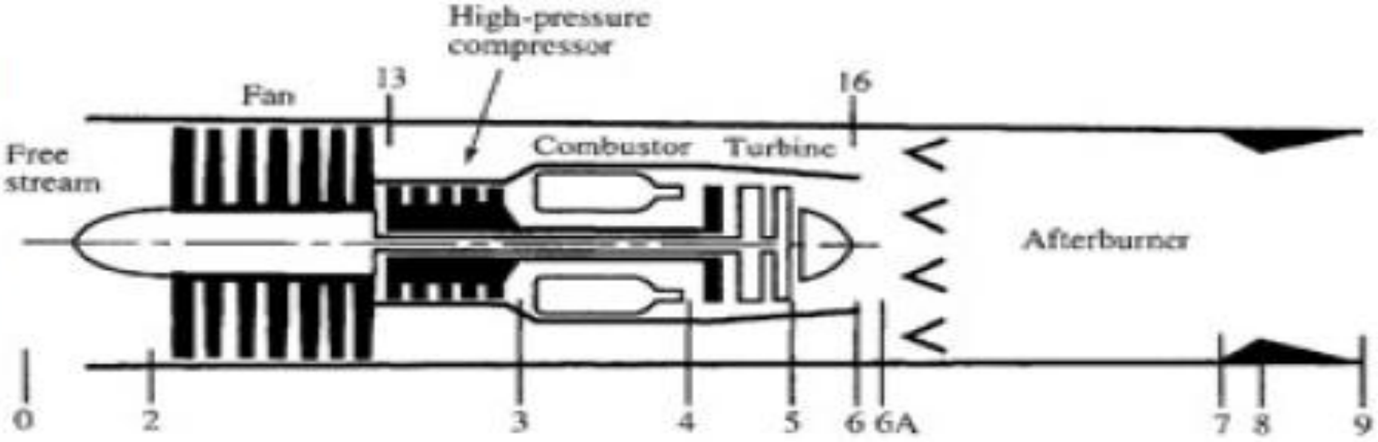


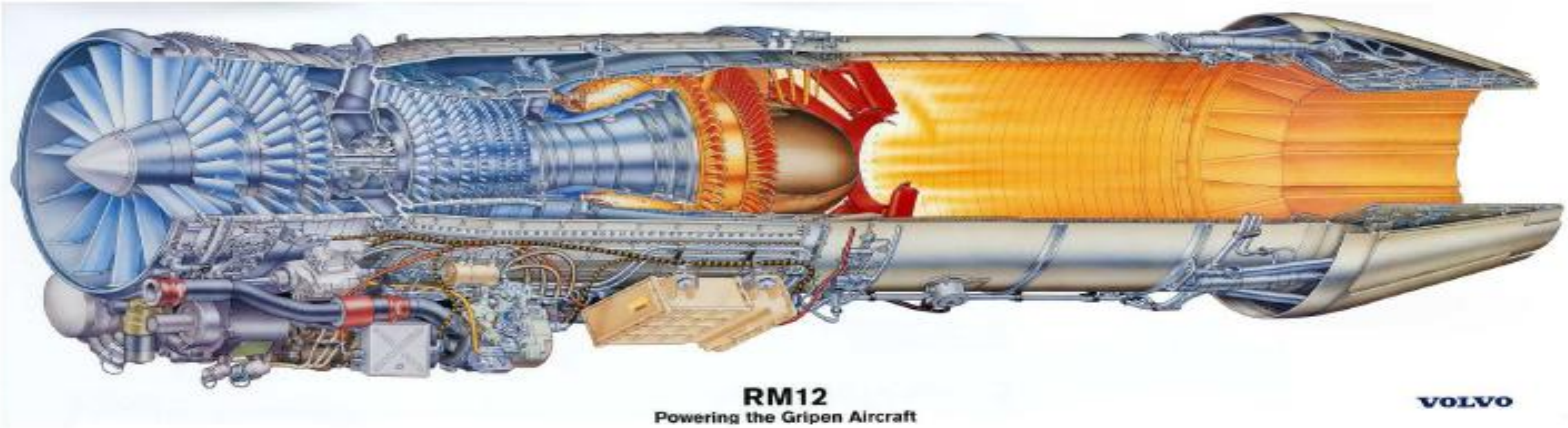
Air Breathing Propulsion: Gas Turbine Systems

Turbofan

… or they can be of Low Bypass Ratio which are normally used for higher-speed military aircraft.

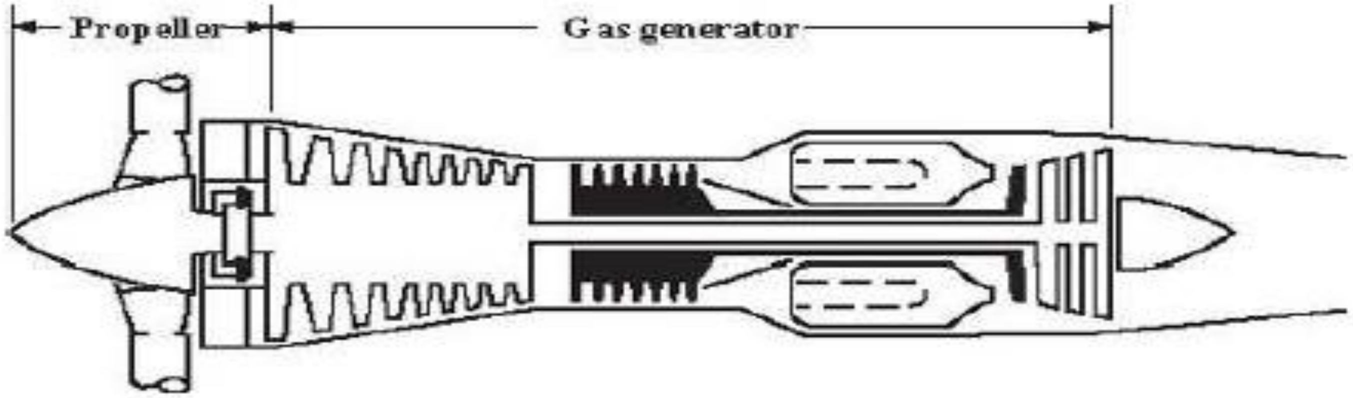
When even more aircraft velocity is needed an afterburner can be outfitted at a cost of significantly lowering engine efficiencies

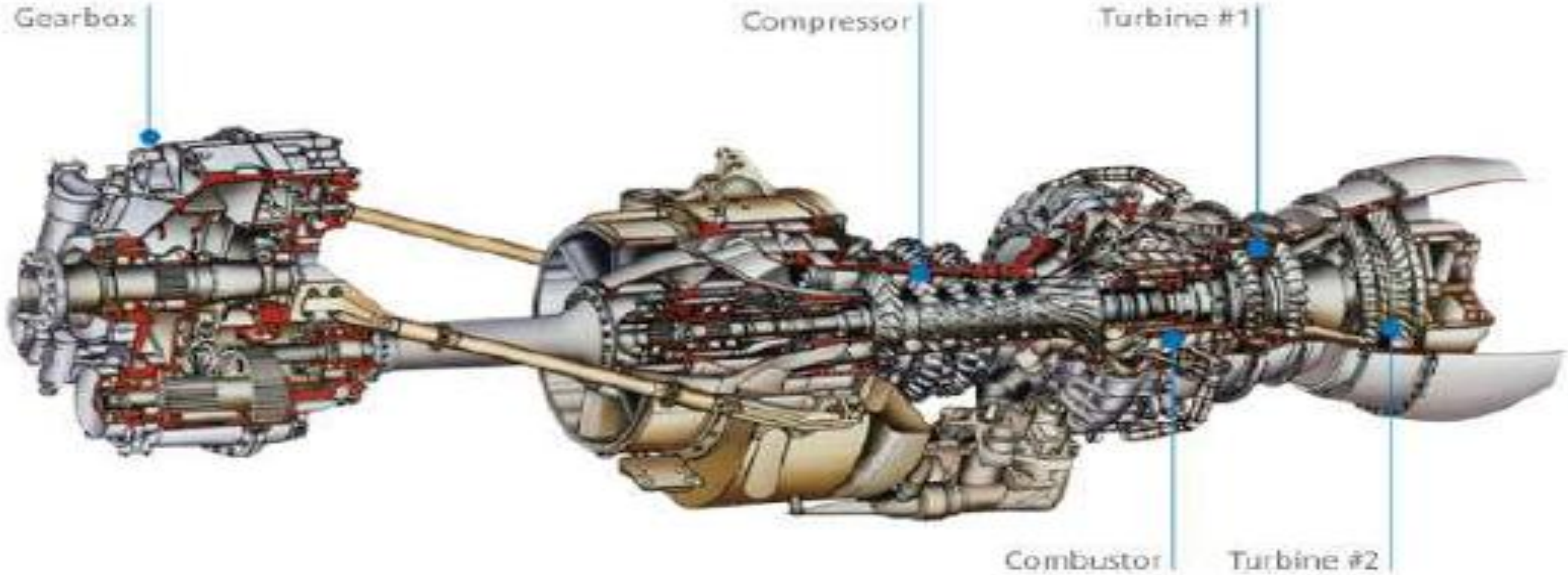




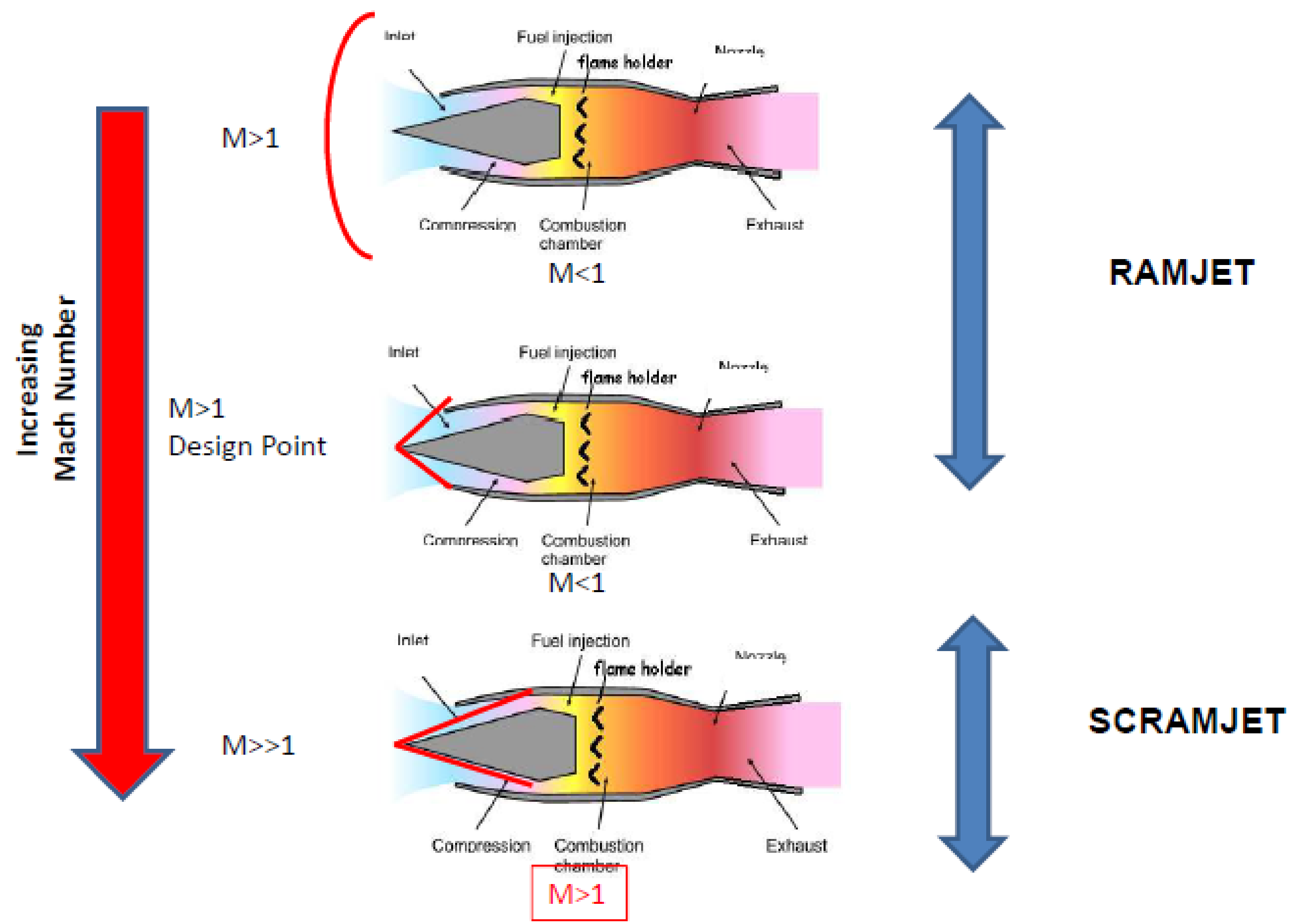
Air Breathing Propulsion: Gas Turbine Systems

Turboprop





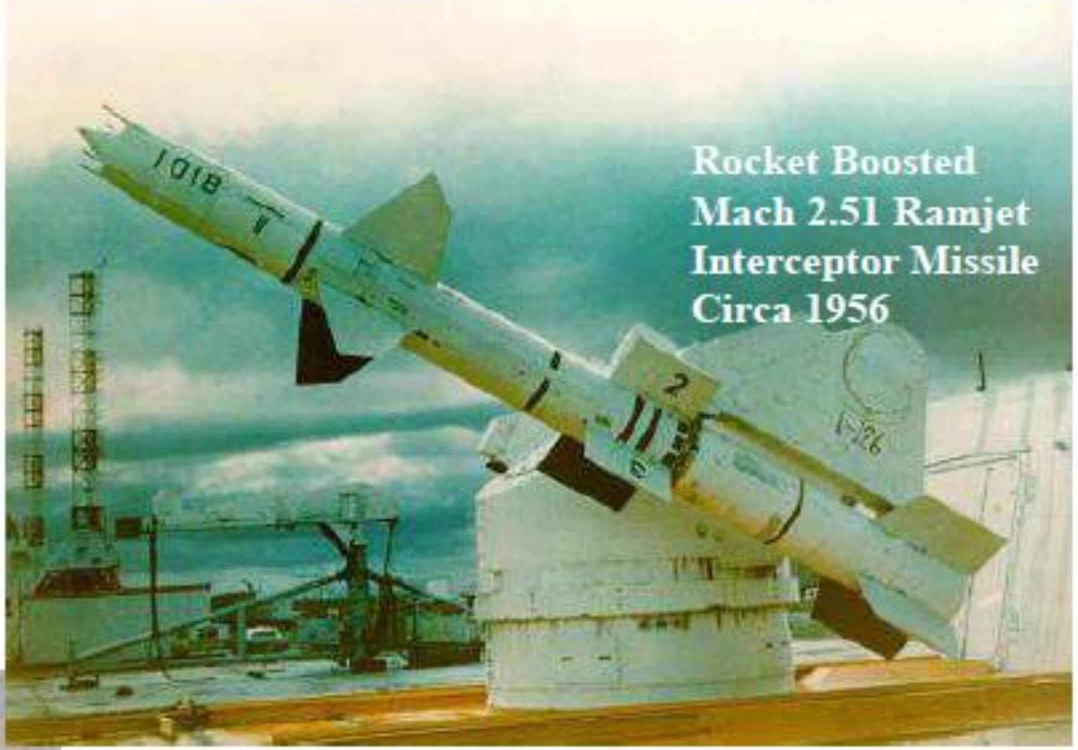
Air Breathing Propulsion: Ducted Systems

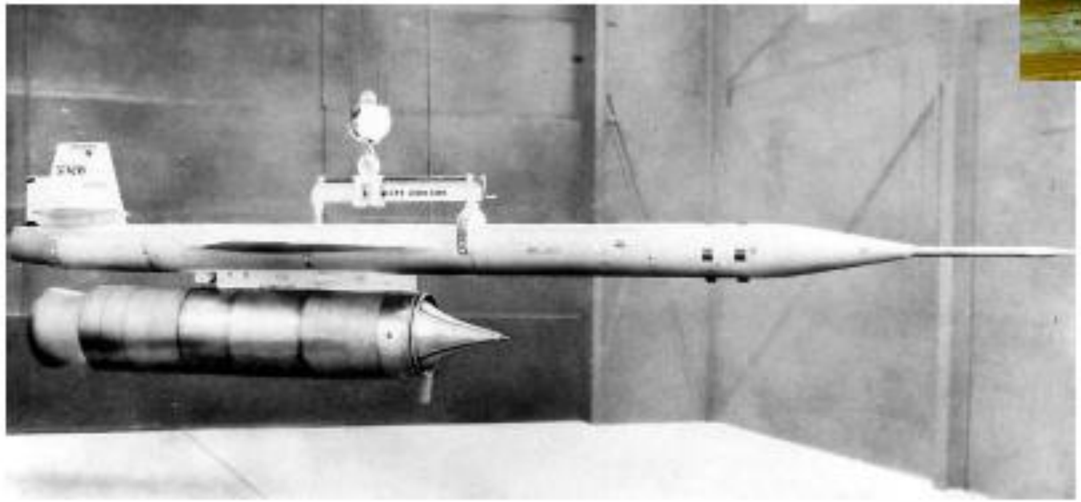


Air Breathing Propulsion: Ducted Systems

RAMJET

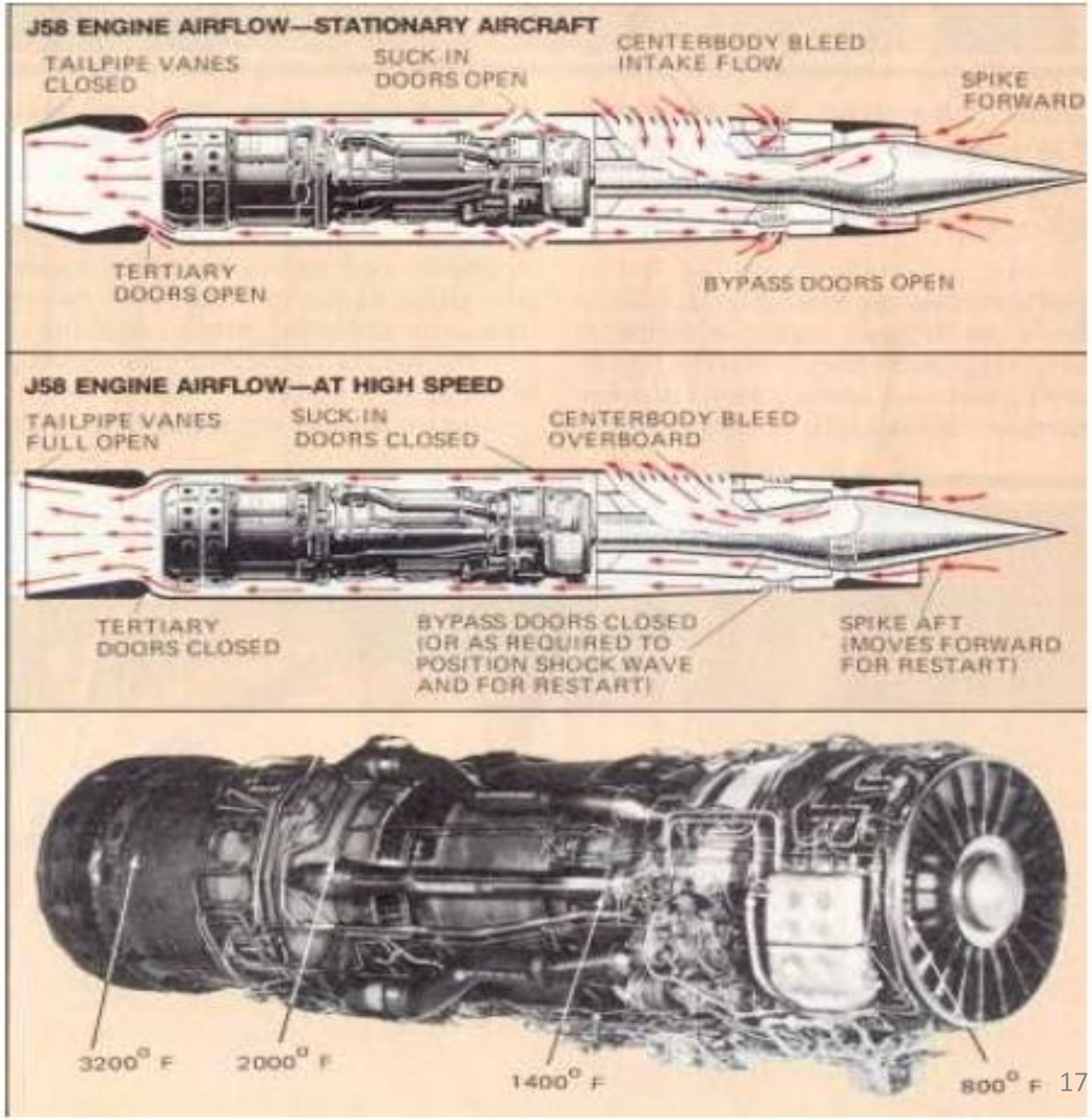
RAMJET propulsion is not a new concept





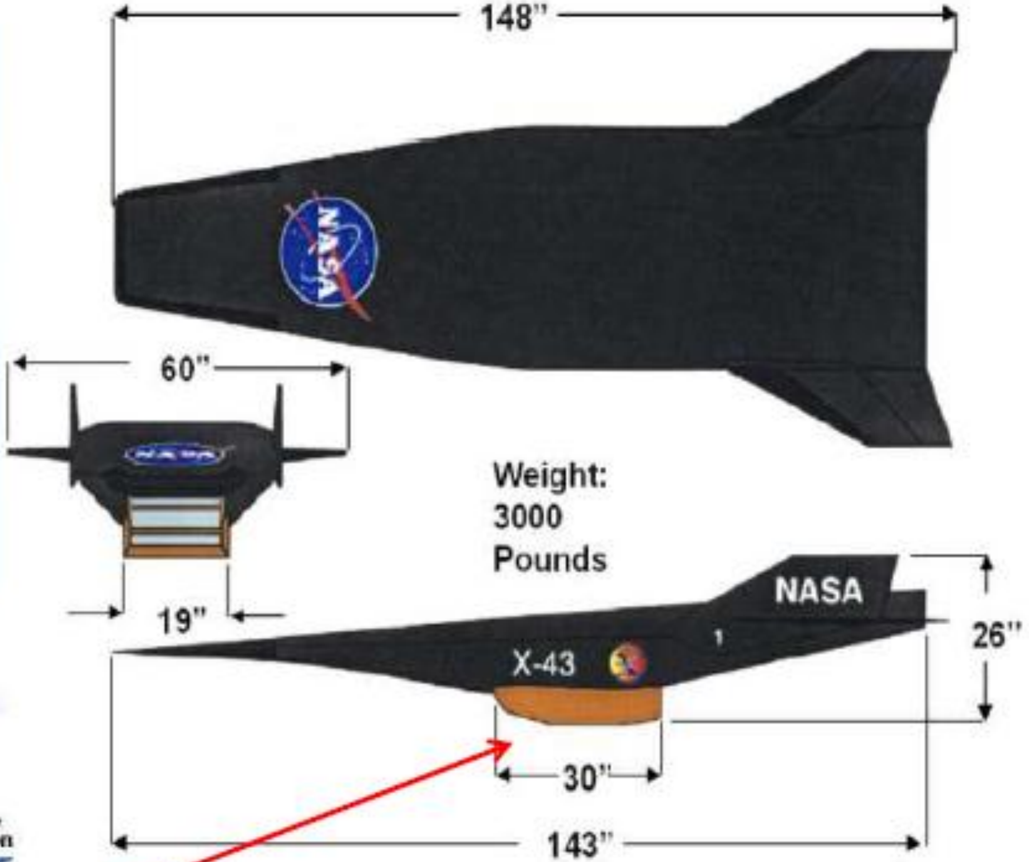
Air Breathing Propulsion: Ducted Systems

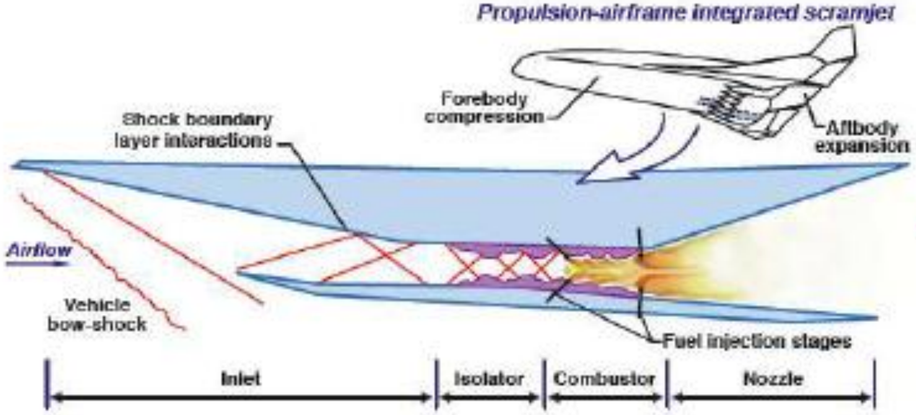
- SR-71 Blackbird used a combined Turbojet-Ramjet propulsion system
- Top speed around Mach 3.2



Air Breathing Propulsion: Ducted Systems

SCRAMJET





Rocket Propulsion

Many ways to classify rocket propulsion

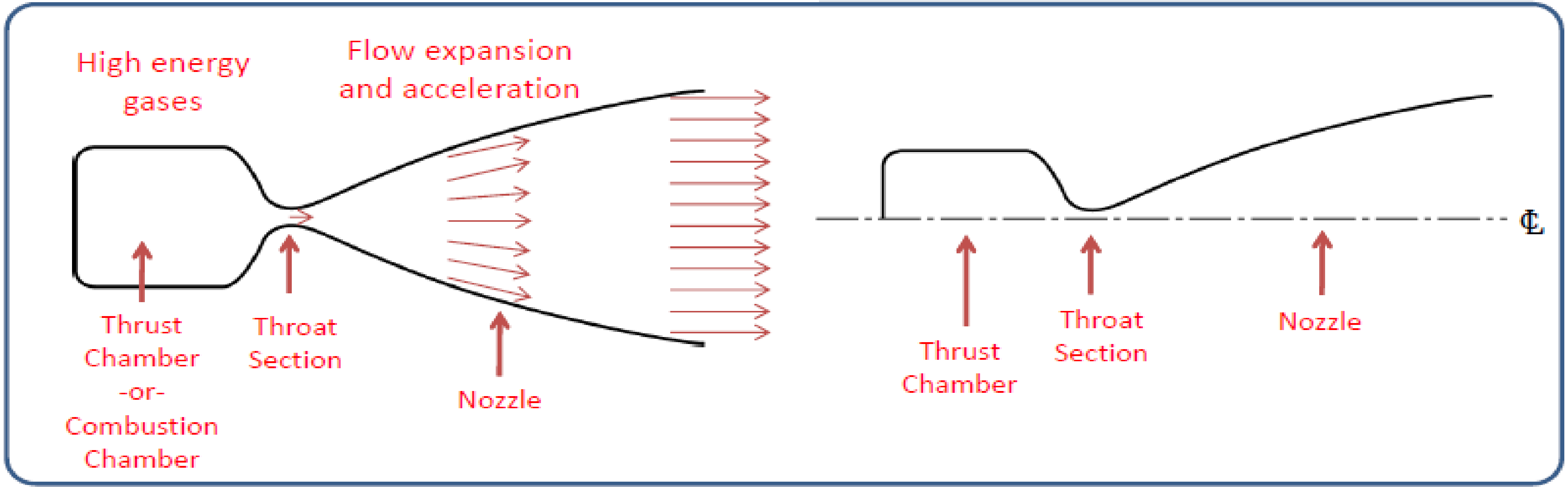
Energy Source
- Chemical
- Nuclear
- Solar

Function
- Booster Stage
- Sustainer/Upper Stage
- Attitude Control
- Orbit Station Keeping

Type of Vehicle
- Aircraft
- Missile
- Assisted Take-off
- Space Vehicle

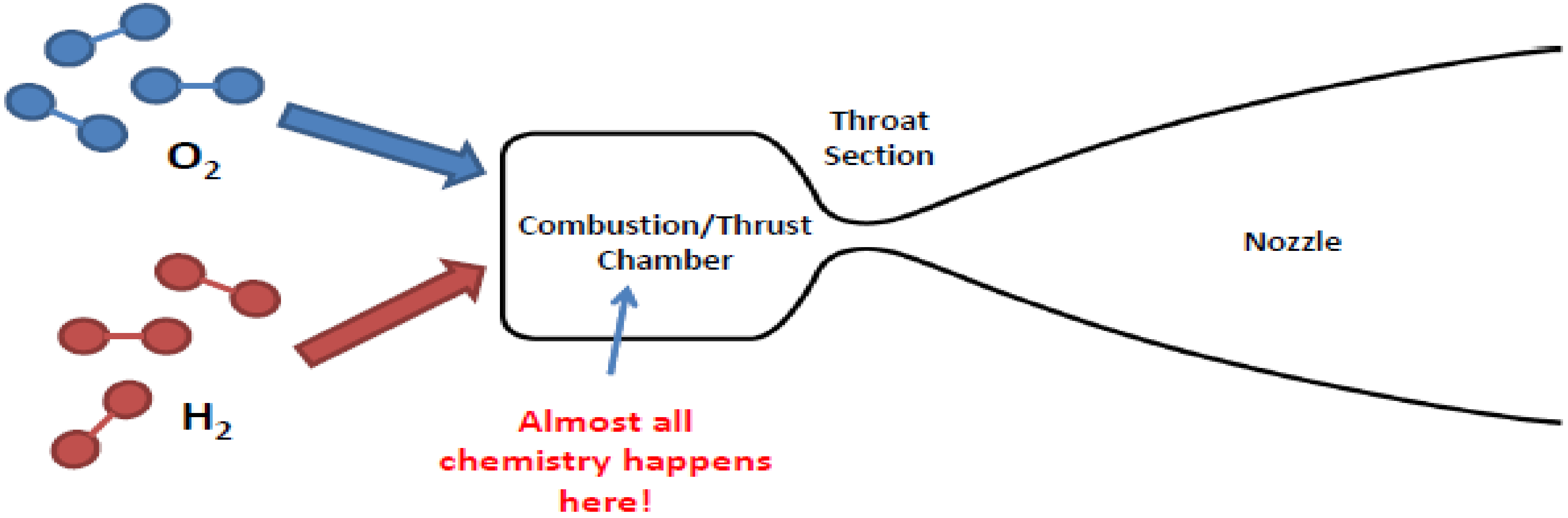
Rocket Propulsion

- Another Classification is method of producing thrust and focus of this course
- A majority of systems utilize the thermodynamic expansion of a gas
- Here, the internal energy of gas is converted into kinetic energy
- This thermodynamic approach uses the same generic equipment:



Other methods for producing thrust exist but will receive minor coverage in this class

Rocket Propulsion: Chemical Rocket Propulsion



An understanding of how to make use of chemical energy from various fuel/oxidizer sources is important.

It is also useful to grasp the concept of equilibrium chemistry and frozen flow analysis which will be covered as well.

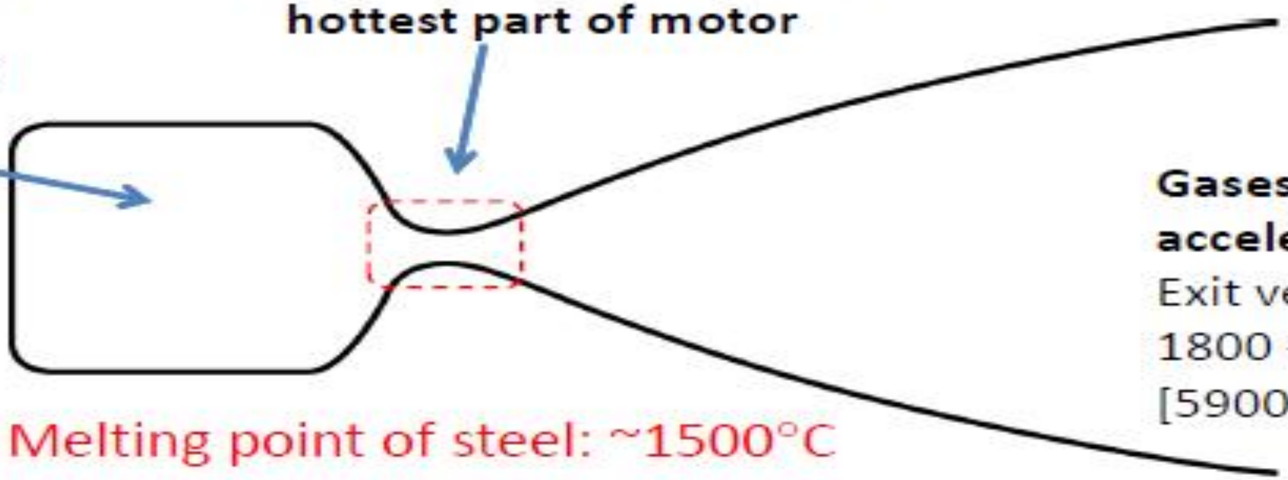
We normally assume <u>**frozen flow**</u> conditions

- Whatever chemical equilibrium chemistry occurs in the thrust chamber maintains through the nozzle expansion process
- No chemical reactions through nozzle
- This is a valid assumption as characteristic flow time much shorter than the chemical reaction time

Rocket Propulsion: Chemical Rocket Propulsion

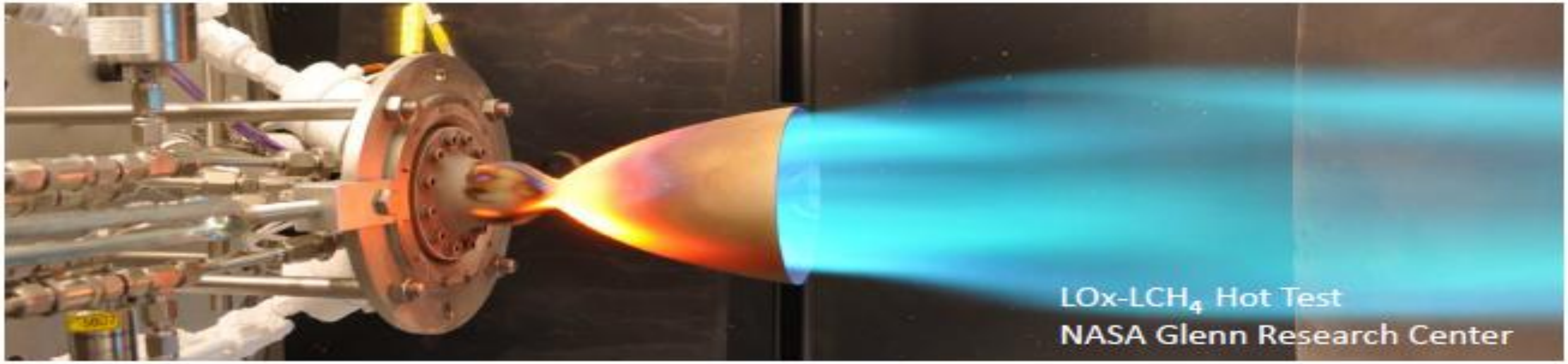
Energy form high pressure combustion
Reacting gases reach high temperatures
2500°C → 4100°C
[4500°F → 7400°F]

Throat section is generally hottest part of motor



Gases expanded and accelerated to high velocities
Exit velocity, v_e
1800 → 4300 m/s
[5900 ft/s → 14,100 ft/s]

Melting point of steel: ~1500°C



- Obviously, it is necessary to appropriately insulate/cool surfaces exposed to high-temperature gases
- Exotic materials (ceramics) and cooling methods (fuel nozzle cooling) often necessary

Rocket Propulsion: Chemical Rocket Propulsion

Classes of Chemical Rocket Propulsion Devices

Liquid Propellant Rocket Engines: liquid propellants fed under pressure from stored tanks into thrust chamber

Bipropellant System: consists of a liquid oxidizer (i.e. LOx) and a liquid fuel (liq. hydrogen, kerosene, etc.)

Monopropellant system: uses a liquid that contains both oxidizing and fuel species

Orbital Sciences Anteres launch using two Aerojet AJ26-62 liquid fuel motors

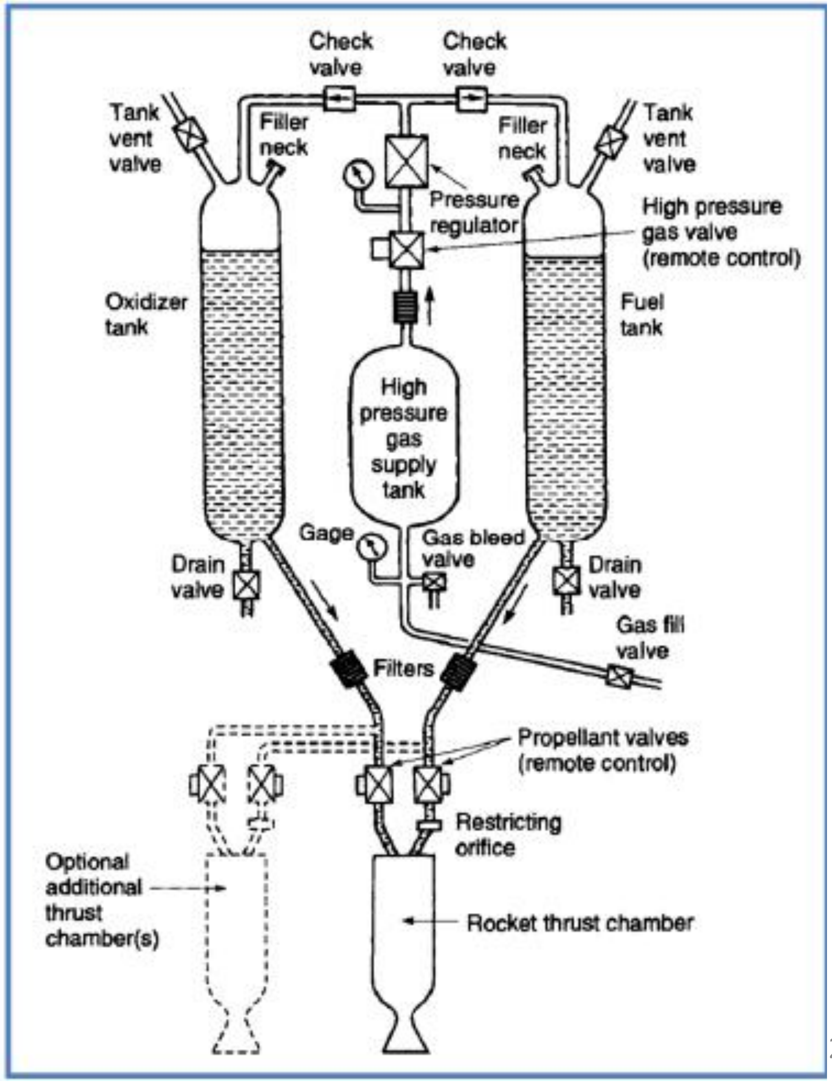
Rocket Propulsion: Chemical Rocket Propulsion

Liquid Fuel Rocket Propulsion Devices

The system here depicts a *gas pressure fed* liquid bi-propellant system

These are adequate and quite common on low-thrust systems:

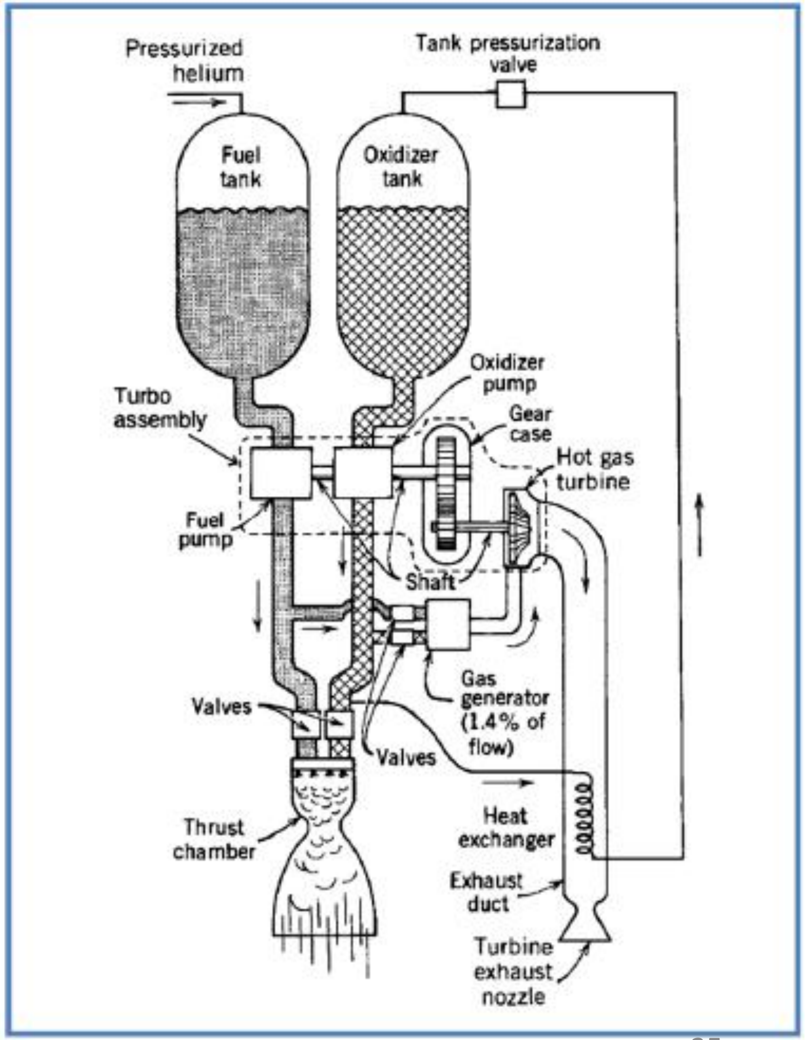
- Attitude control thrusters
- Micro-thrusters



Rocket Propulsion: Chemical Rocket Propulsion

Liquid Fuel Rocket Propulsion Devices

- Launch systems, however, consume massive amounts of fuel/oxidizer
- **High mass flow delivery to thrust chamber required**
- Turbo-pumps are normally required for these high-thrust systems



Rocket Propulsion: Chemical Rocket Propulsion

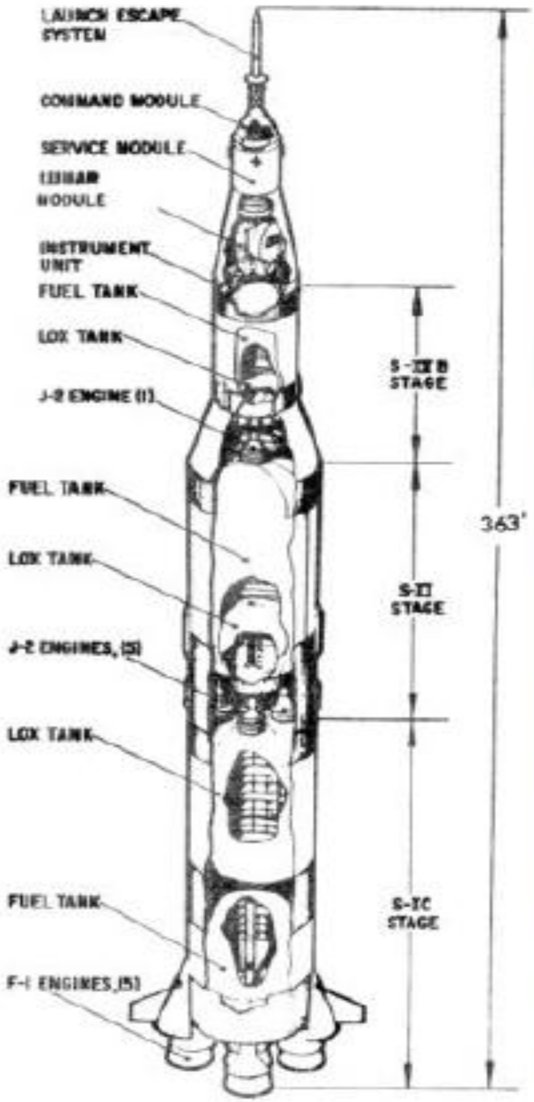
Solid Fuel Rocket Propulsion Devices

Saturn V used five F1 bi-propellant liquid rocket motors for boost

F-1 engine - the most powerful single-nozzle, liquid-fueled rocket engine ever developed

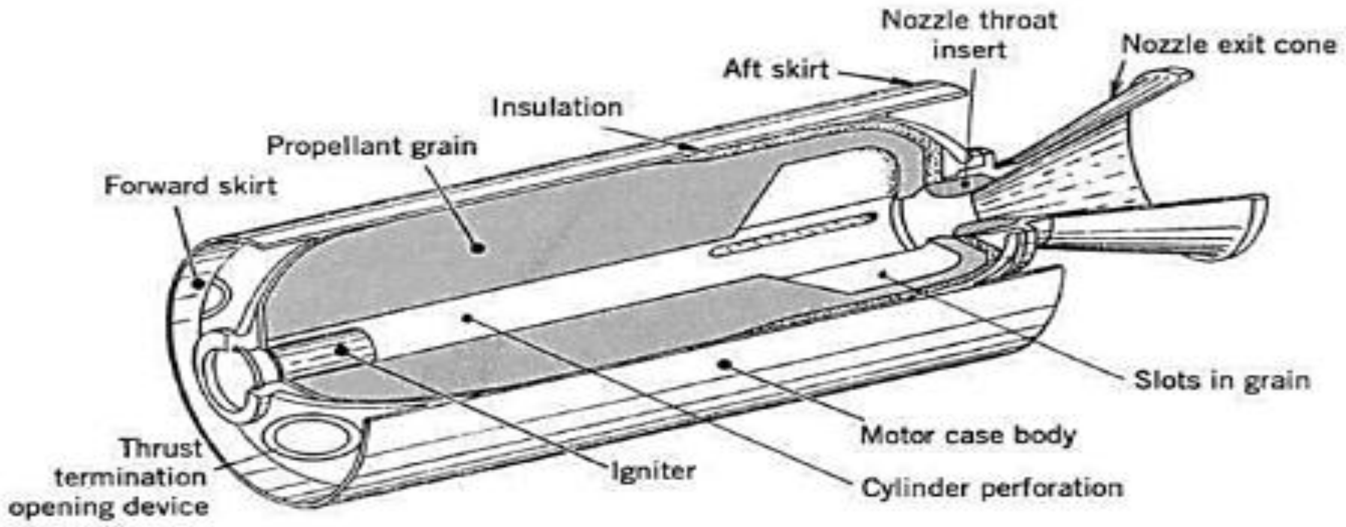
Each F-1 engine consumed 3945 lb/sec of O_2 oxidizer and 1738 lb/sec of kerosene fuel

Doing the math, the five total thrusters consumed about 13 metric tons of propellant per second during boost.



Rocket Propulsion: Chemical Rocket Propulsion

Liquid Fuel Rocket Propulsion Devices



Propellant to be burned is in combustion chamber (also known as the **case**)

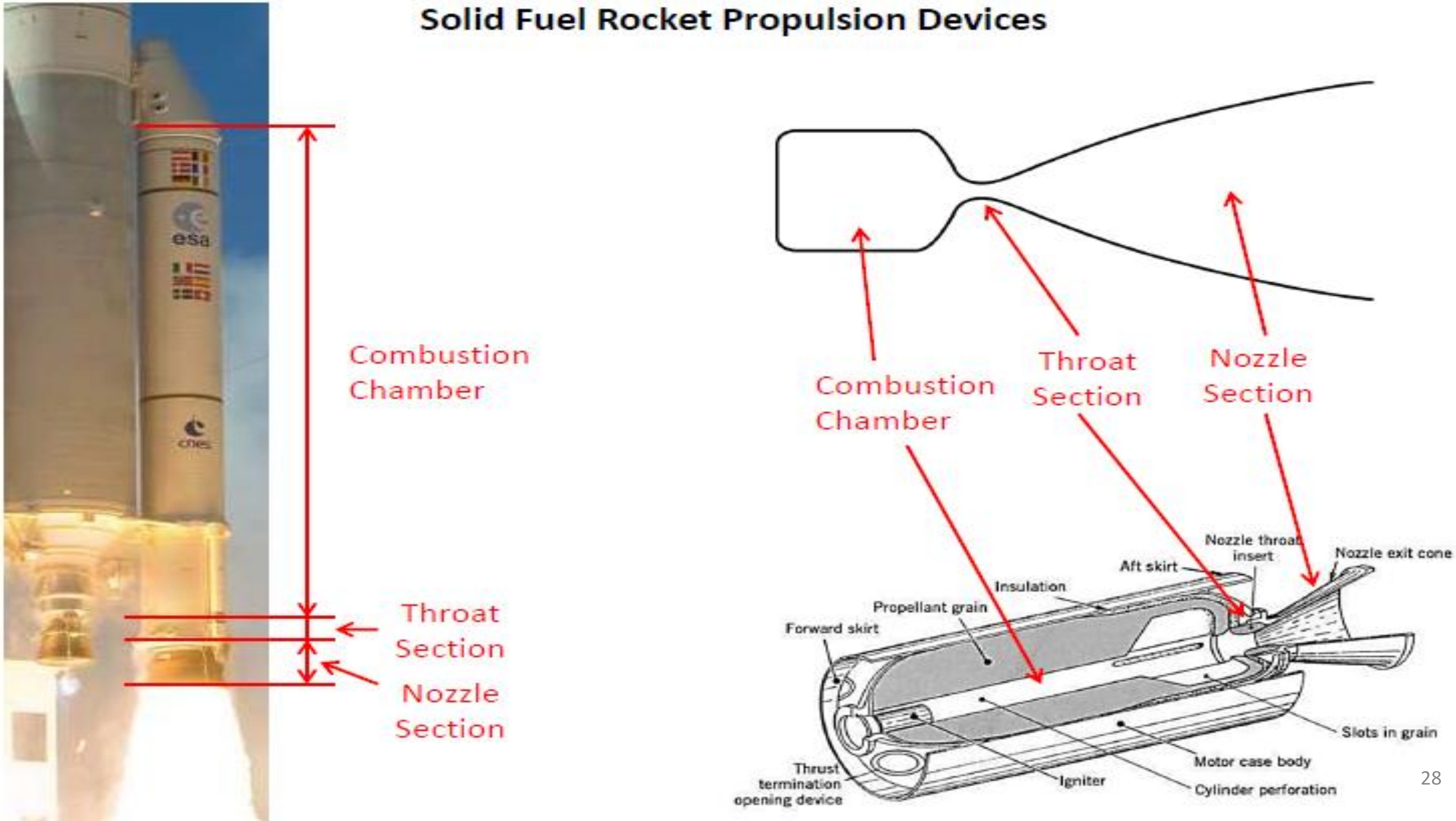
Grain: solid propellant charge containing all chemical elements for complete burning

Burning occurs on exposed internal surfaces of the grain within the **perforation**.

Solid propellant systems are generally more simple than liquid systems as there are no fuel feed systems of valves

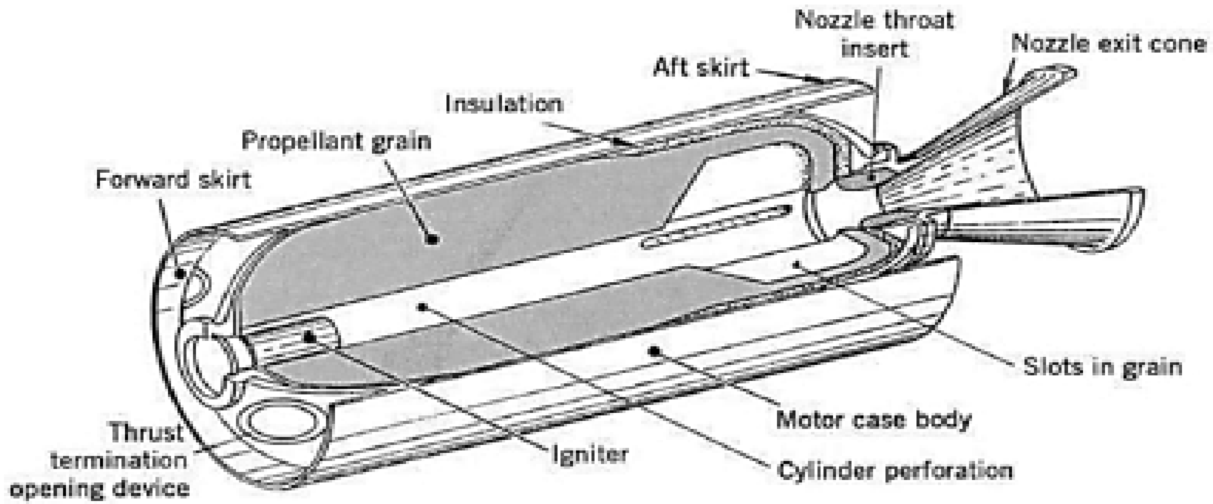
ESA Ariane 5 launch using two solid fuel motors and one liquid fuel motor for boost

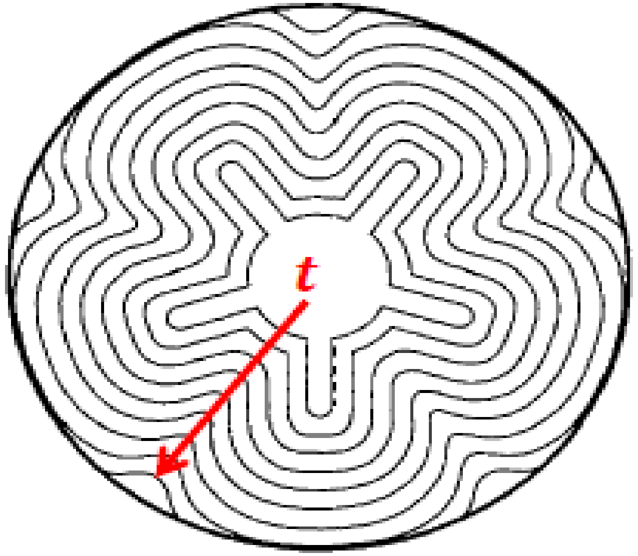
Rocket Propulsion: Chemical Rocket Propulsion



Rocket Propulsion: Chemical Rocket Propulsion

Solid Fuel Rocket Propulsion Devices



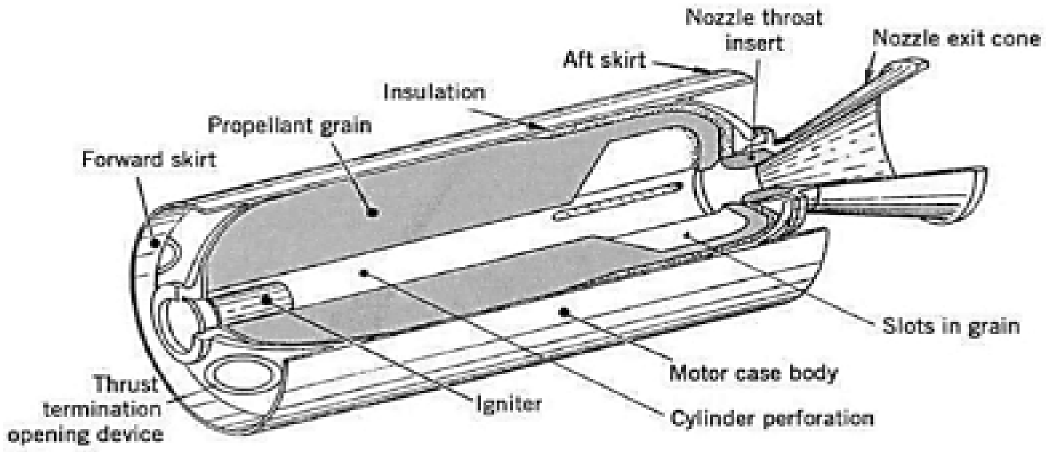


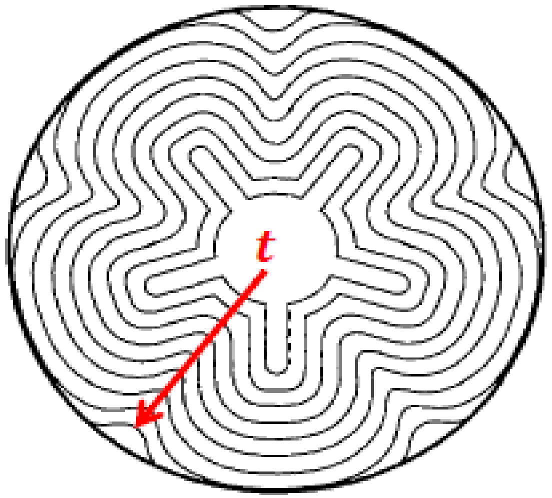
But, as grain is consumed over time of burn the exposed surface area changes

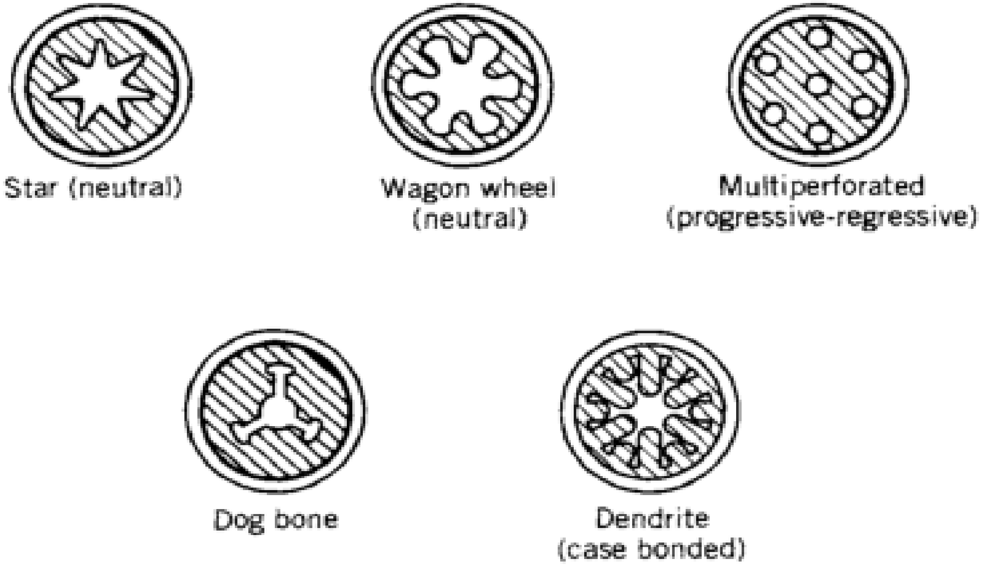
Different grain geometries can be designed to meet mission objectives

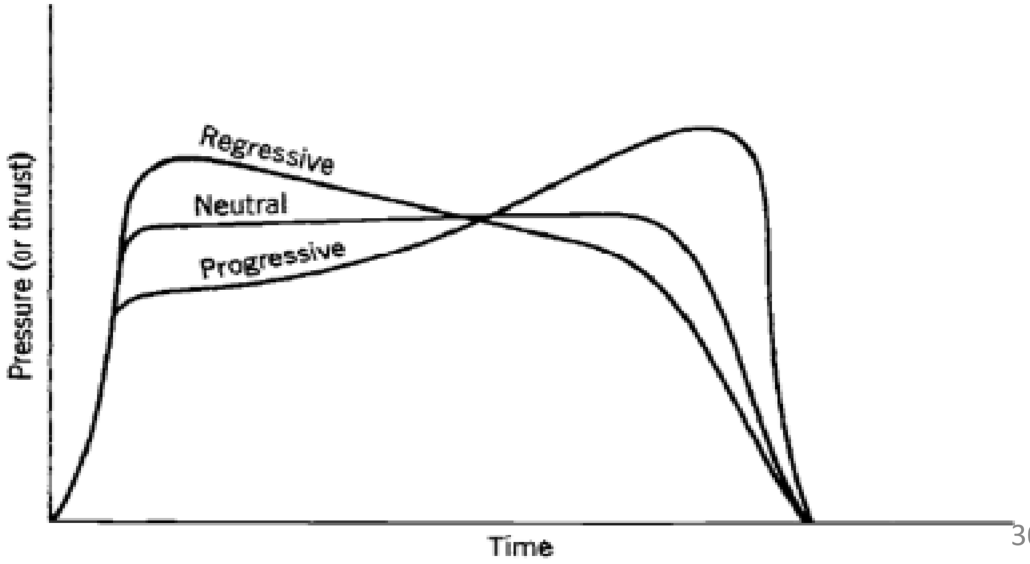
Rocket Propulsion: Chemical Rocket Propulsion

Solid Fuel Rocket Propulsion Devices



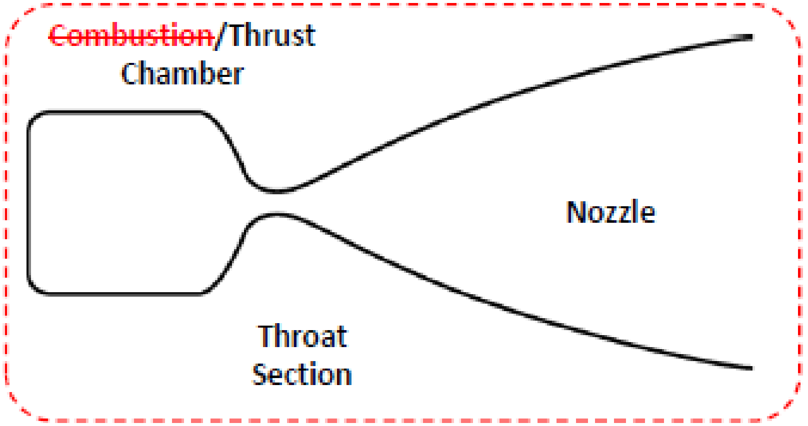


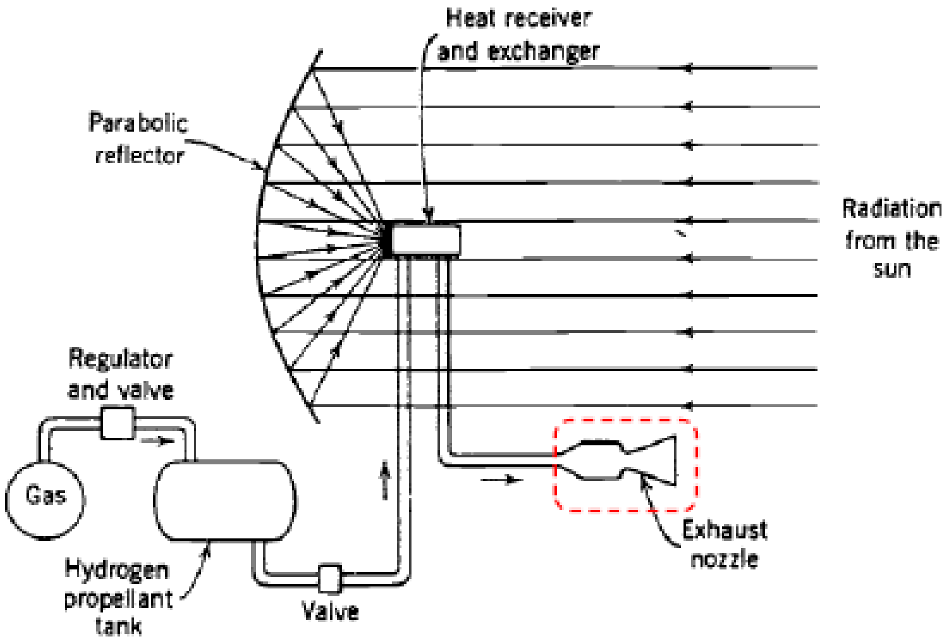


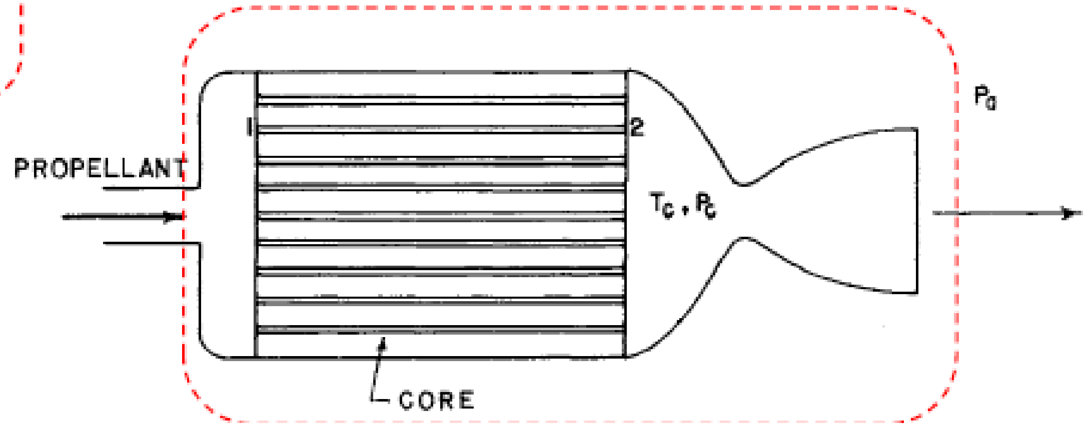


Rocket Propulsion: Non-chemical Rockets

For this course we will mainly focus on rocket propulsion with chemical reactions as energy sources (some various energy source concepts will be presented near semester's end)







TRANS-ATMOSPHERIC AND SPACE FLIGHT MISSION PROPULSION REQUIREMENTSPROPULSION SYSTEMS- CLASSIFICATION, PERFORMANCE CHARACTERISTICS

Unit 1

Syllabus

- Hypersonic transport vehicles, military missiles, space launch vehicles, spacecraft- role, types, missions profile, trajectories, operating conditions- gravity, atmosphere. Incremental flight velocity budget for climb out and acceleration, orbital injection- Breguet equation for cruise– mission propulsion requirements- thrust levels, burnig time, economy.
- High speed propulsion systems- types, construction, operating principles- sources of energy, generation of power, momentum, propellants,- applications, performance parameters- specific thrust, specific impulse, internal efficiency, propulsive efficiency- typical values. Reaction control systems- applications

Hypersonic Transport Vehicles

- 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 or 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, and most of the time, does not have an engine.

Chapter 2

AIR BREATHING ENINES FOR HYPERSONIC TRANSPORT PLANES AND MILITARYMISSILES- SUPERSONIC COMBUSTION- THE SCRAM-JET ENGINE

Performance of turbojets, ramjets at high speeds- limitations. Need for supersonic combustion- implications criticality of efficient diffusion and acceleration, problems of combustion in high speed flow.

The scramjet engine- construction, flow process- description, control volume analysis- spill-over drag, plume drag. Component performance analysis- isolator, combustor- flow detachment and reattachment, thermal throat, scheduled, distributed fuel injection. Nozzle flow, losses- failure to recombination, viscous losses,plume losses. Scramjet performance, applications.

Combined cycle engines- turbo-ramjet, air turbo-rocket (ATR), ejector ramjet- Liquid-air collection engine(LACE)- need, principle, construction, operation, performance, applications to hypersonic transport plane and missile propulsion

AIR BREATHING PROPULSION

Propulsive device

- generates the net thrust to overcome inertia and gains speed

Air breathing propulsive device

- uses oxygen in air to burn fuel and to generate thrust
- limited to dense atmosphere up to a height of about 100 km
- only jet propulsion presented in this seminar

Turbojet

- Engines with rotating machinery
- Used by most civil and military aircrafts operating at subsonic speeds (Mach <1)
- General Electric , Pratt & Whitney , Rolls Royce

Ramjet

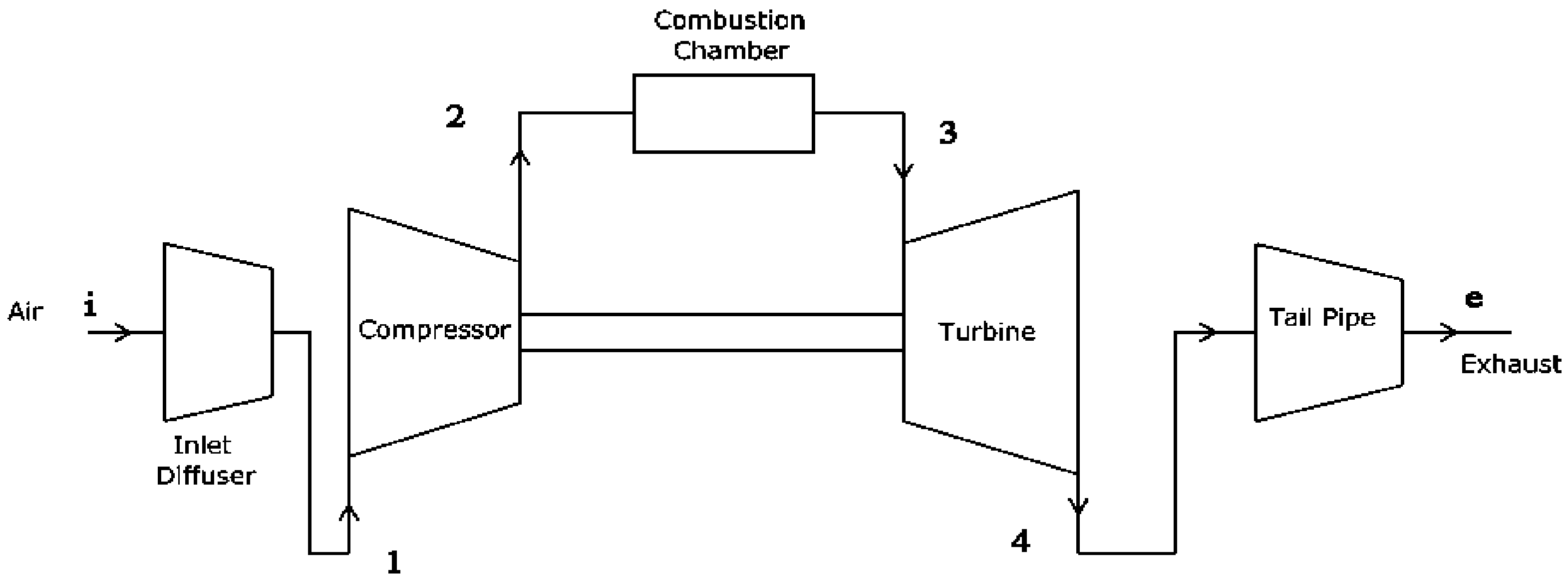
- Engines without rotating machinery
- Used for supersonic missiles up to Mach 5
- Soviet SA-6 ,British Sea Dart , French Snecma M-88

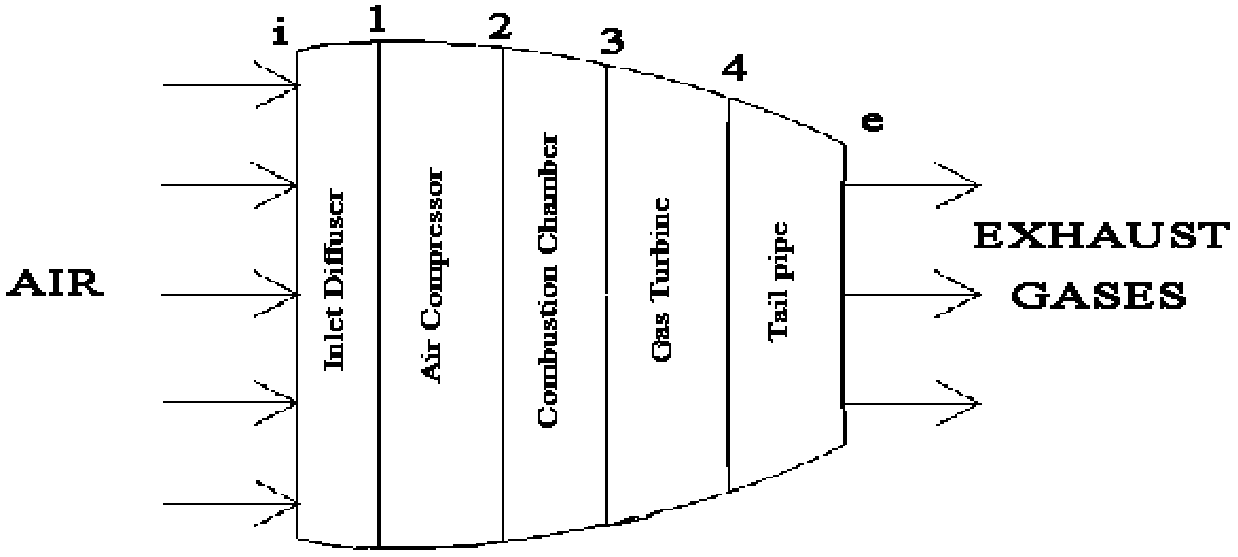
Scramjet

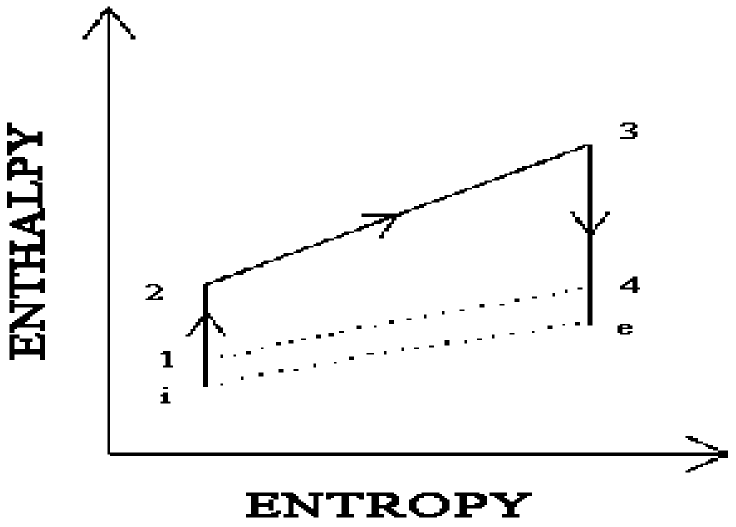
- Engines devoid of rotating machinery
- Will be used for hypersonic missiles above Mach 5
- Nasa X-43 , British HOTOL , German Sanger

TURBOJET

- Air sucked in through the inlet diffuser
- Compressed and used to burn the fuel in the combustor
- Combustion products used to drive the turbine
- Exhaust through the nozzle to generate jet propulsion




i
1
2
3
4
e
AIR
Inlet Diffuser
Air Compressor
Combustion Chamber
Gas Turbine
Tail pipe
EXHAUST
GASES


ENTHALPY
ENTROPY
3
2
4
e
1
i

LIMITATIONS

At higher Mach numbers

- Fuel consumption increases
- Moving parts do not contribute to engine power
    - Share of compressor at Mach 1 = 50 %

      Mach 2 = 15 %

      Mach 3 = 04 %
- Moving parts causes losses
- High temperatures (around 3000 K) are produced
- Compressor blades cannot withstand that temperature
- No such high temperature withstanding blade material exists
- Compression created by speed is enough to keep engine process

RAMJET

- At speeds above Mach 3 a passive intake can compress the air due to ramming effect (without use of compressor) for subsonic combustion in the combustion chamber

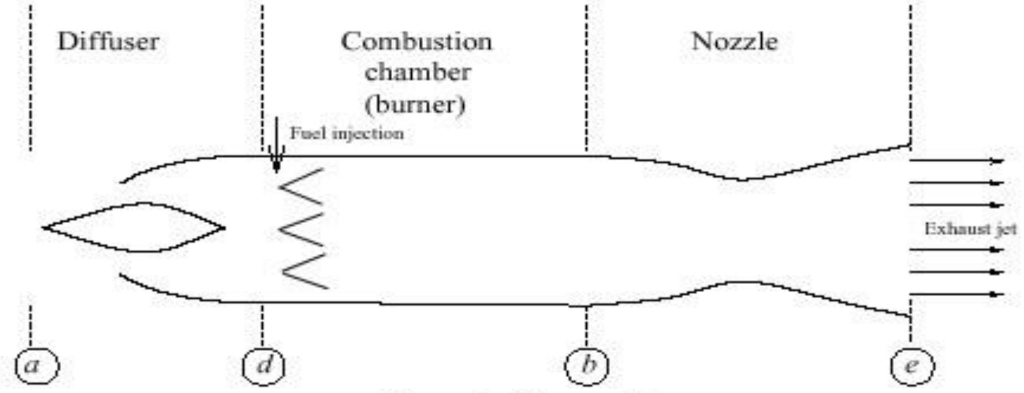


Figure 1: The ramjet.

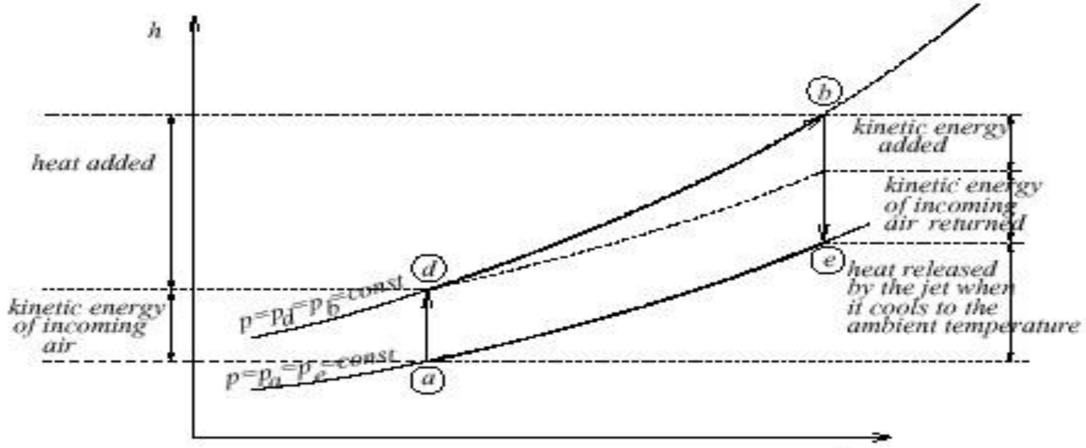


Figure 2: Enthalpy-entropy diagram for an ideal ramjet

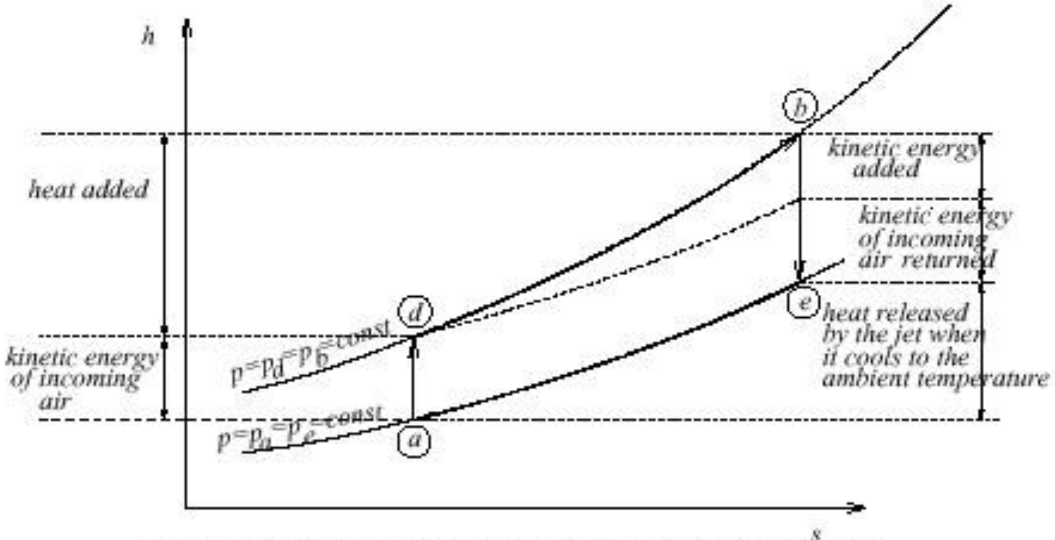
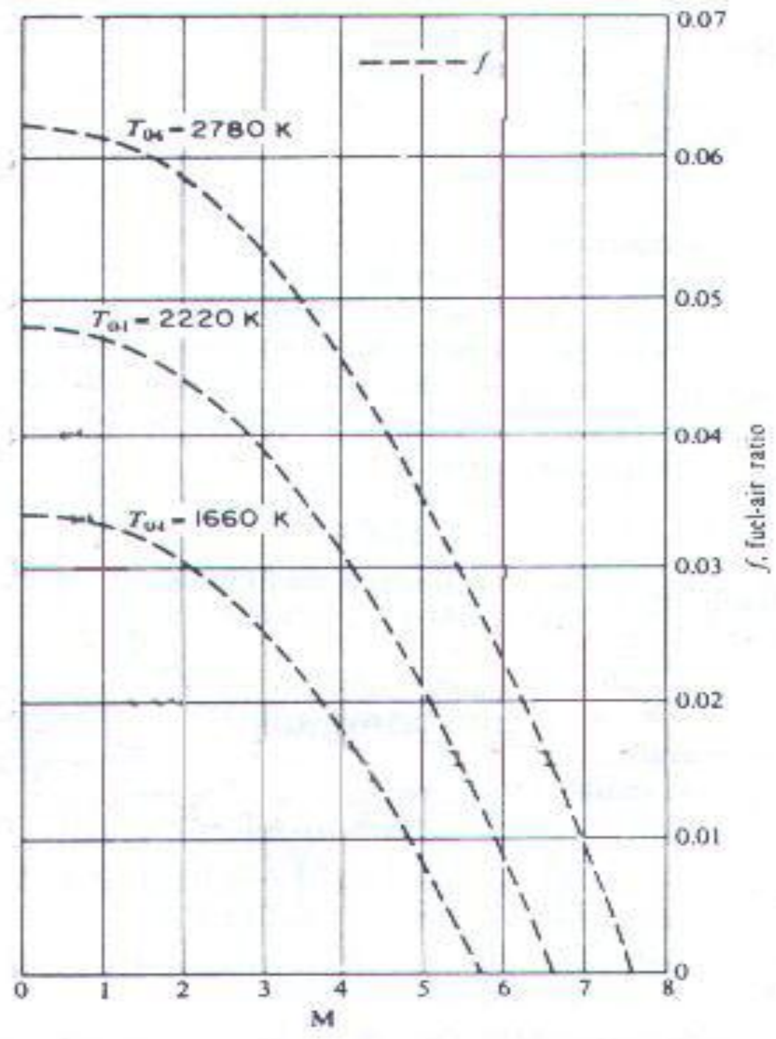


Figure 2: Enthalpy-entropy diagram for an ideal ramjet

- Mach number decreased and point b kept constant
  - TSFC becomes high
  - Larger size of engine
- heat added increased and point d kept constant
  - increase in maximum temperature
  - material properties of engine walls

PERFORMANCE



Experimental Conditions

Inlet temp = 220 K

Cp = 0.24 kcal/kg-k

γ = 1.4

FUELS USED

- Gaseous Fuel Ramjet

  * eg. hydrogen

- Liquid Fuel Ramjet

  * kerosene , synthetic hydrocarbon fuel

  eg. US made RJ1, RJ4 ; French CSD07T , CSD15T

- Solid Fuel Ramjet

  * polymers loaded with metal particles like Mg ,Al or B

  eg. Polyether , polyester , polyurethane

ADVANTAGES

- Able to attain high speeds up to mach 5
- No moving parts so less wear & tear and minimum losses
- Reduced weight and smaller engine
- Lighter and simpler than turbojet
- Higher temperatures can be employed

DISADVANTAGES

- Bad performance at lower speeds
- Needs booster to accelerate it to a speed where ramjet begins to produce thrust
- Higher fuel consumption
- Maximum operating altitude is limited
- High temperature material required

APPLICATIONS

USSR

- SA -4 Ganef surface to air missile

  propelled by 4 booster rockets & one liquid fuel ramjet

  Range 50 km Mach 2.5 at altitude 1.1 to 24 km

- SA –6 Gainful surface to air missile

  solid propellant booster propels it to Mach 1.5

  ramjet propels it to Mach 2.8 . Range 60 km

USA

- ASALM

  liquid fuel ramjet ; reaches Mach 4 at 30 km height

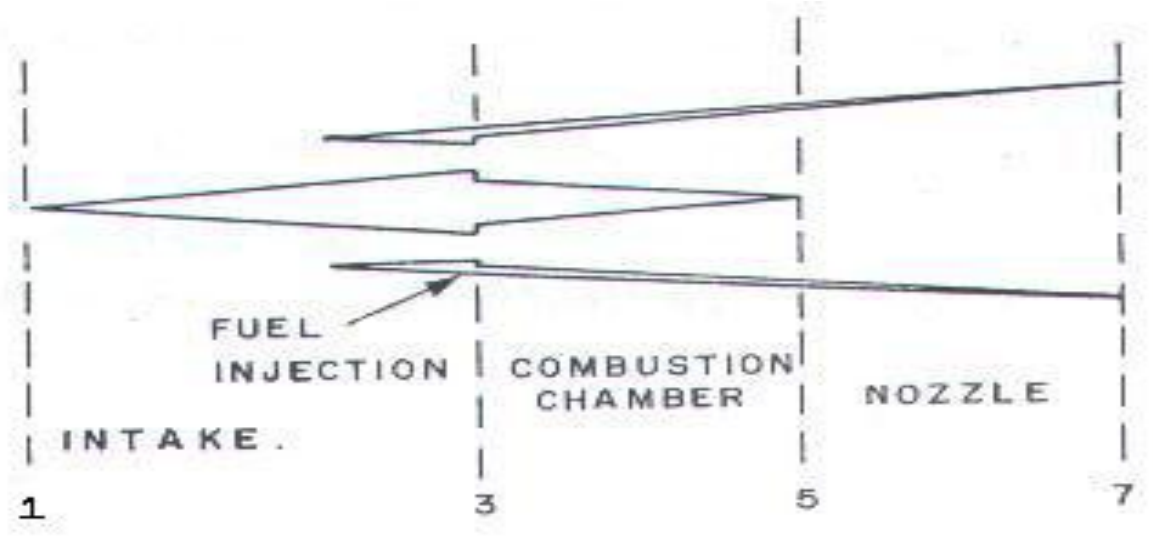
- YAQM127A

  used by Navy ; Mach 2.5 capability Range of 100 km

SCRAMJET

- **S**upersonic **C**ombustion **RAMJET**
- In ramjet supersonic speed of air is reduced to subsonic speeds in combustion chamber thereby causing high temperature rise.
- If combustion is done at supersonic speed temperature rise could be avoided.
- Achieving supersonic combustion is the ultimate challenge

  Dwell time in the combustor is low



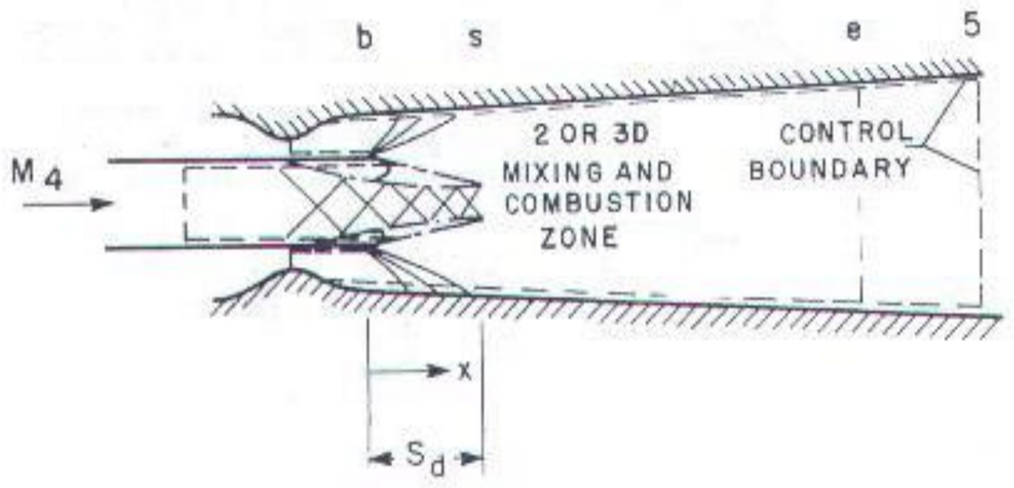
SUPERSONIC COMBUSTION

Major Issues

Proper mixing

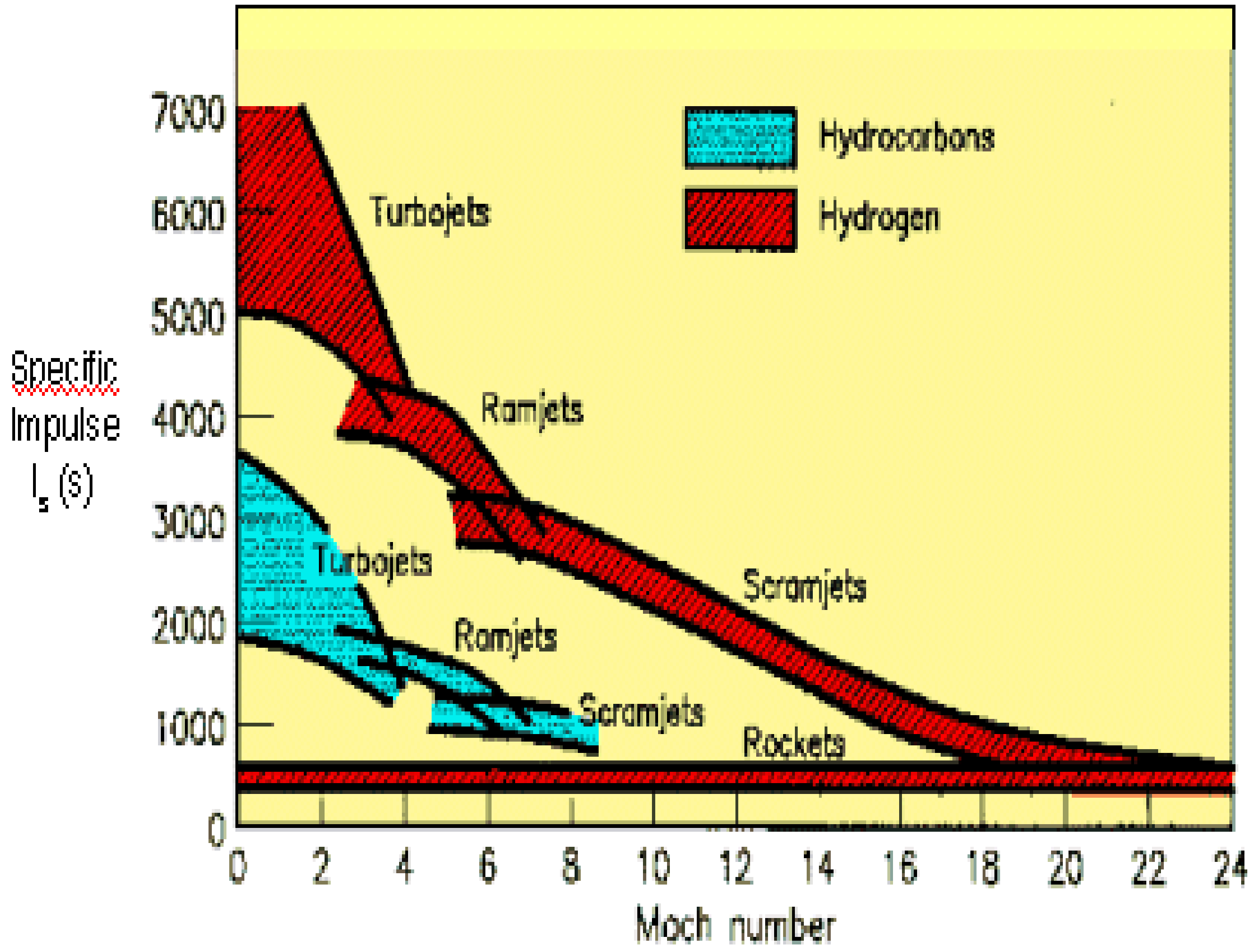
Ignition

Stable combustion



- Flight Mach no. is 6 to 10   Inlet Mach no. is 2 to 4
- Blockage caused by injection and heat release generates a "shock train"
- intense mixing and combustion with large gradients in flow properties & chemical composition in the axial,radial and circumferential directions
- Divergent combustor adds to proper mixing and ignition and to compensate for the pressure rise

Comparison between turbojet,ramjet,scramjet and rocket



HYPERSONIC COMBUSTION

- Occurs when flight Mach nos. are 20 and above and when combustor inlet Mach nos. become greater than 5
- Kinetic energy of free stream air entering scramjet is large compared to the energy released by fuel combustion
- At Mach 25 heat release from combustion is 10 % of enthalpy of gases while at Mach 8 it is 50 %.
- Flow deflections due to heat release is small and it eliminates the possibility of strong shock formation

* Effect of turbulence generation and mixing at hypersonic speeds

* Behavior of fuel-air mixing when air velocity exceeds fuel injection velocity

Jet Engine Performance

- It is seen that engine thrust is proportional to the mass flow rate through the engine and to the excess of the jet velocity over the flight velocity.
- The specific thrust of an engine is defined as the ratio of the engine thrust to its mass flow rate. The specific thrust is

$$m(V_5 - V_a) = F - (p_5 - p_a)A_5 \qquad [\mathrm{lb_f} \mid \mathrm{kN}]$$

$$F = m(V_5 - V_a) + (p_5 - p_a)A_5 \qquad [\mathrm{lb_f} \mid \mathrm{kN}]$$

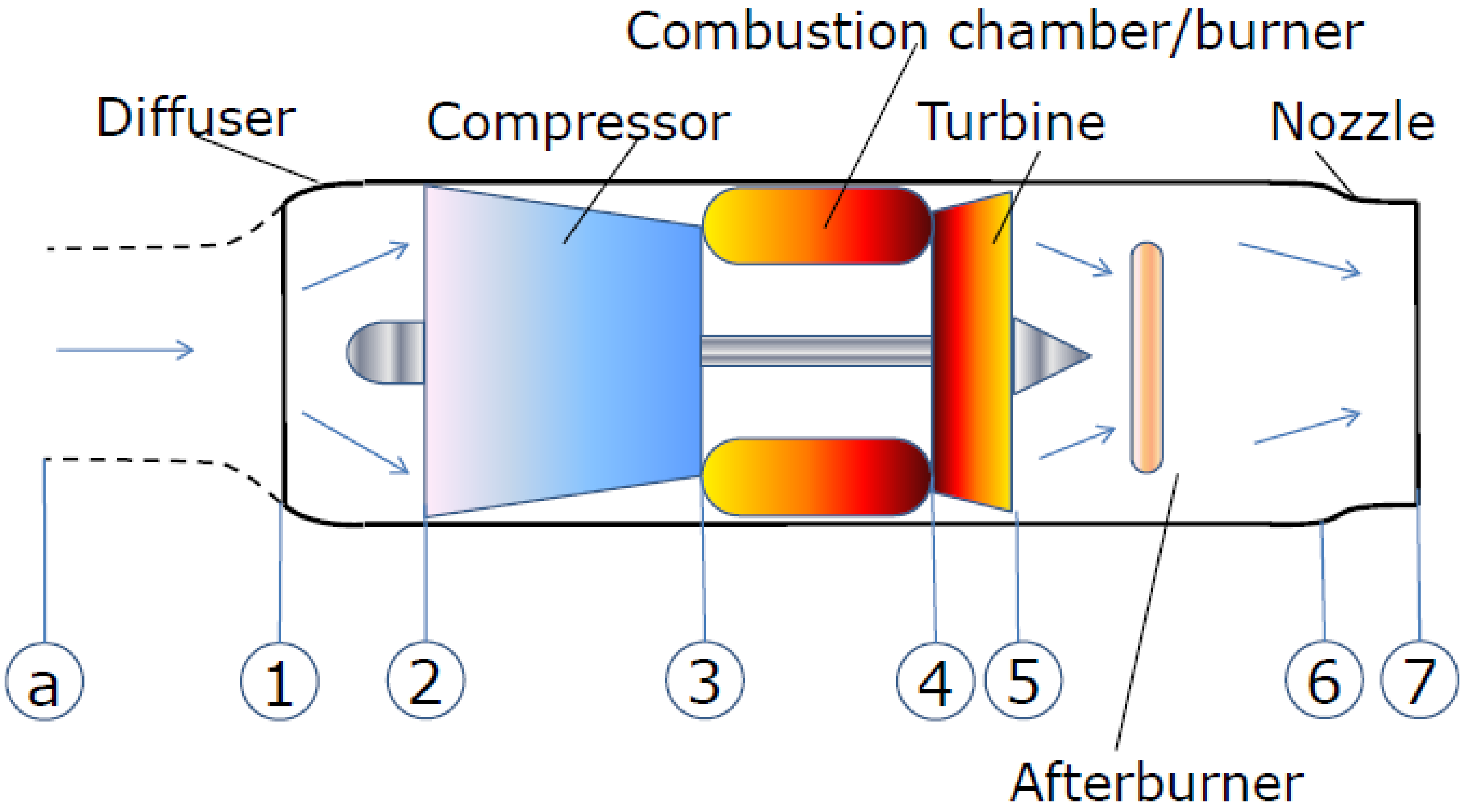
$$F/m = (V_5 - V_a) + (p_5 - p_a)A_5/m \qquad [\mathrm{lb_f\text{-}s/lb_m} \mid \mathrm{kN\text{-}s/kg}]$$

- Because the engine mass flow rate is proportional to its exit area, A_5/m depends only on design nozzle exit conditions.
- As a consequence, F/m is independent of mass flow rate and depends only on flight velocity and altitude.
- Assigning an engine design thrust then determines the required engine-mass flow rate and nozzle exit area and thus the engine diameter. Thus the specific thrust, F/m, is an important engine design parameter for scaling engine size with required thrust at given flight conditions.
- Another important engine design parameter is the thrust specific fuel consumption, TSFC, the ratio of the mass rate of fuel consumption to the engine thrust

$$\mathrm{TSFC} = m_f/F \qquad [\mathrm{lb_m} \,/\, \mathrm{lb_f\text{-}s} \mid \mathrm{kg} \,/\, \mathrm{kN\text{-}s}]$$

- Low values of TSFC, of course, are favorable. The distance an aircraft can fly without refueling, called its range, is inversely proportional to the TSFC of its engines.

Real cycle for turbojet engines



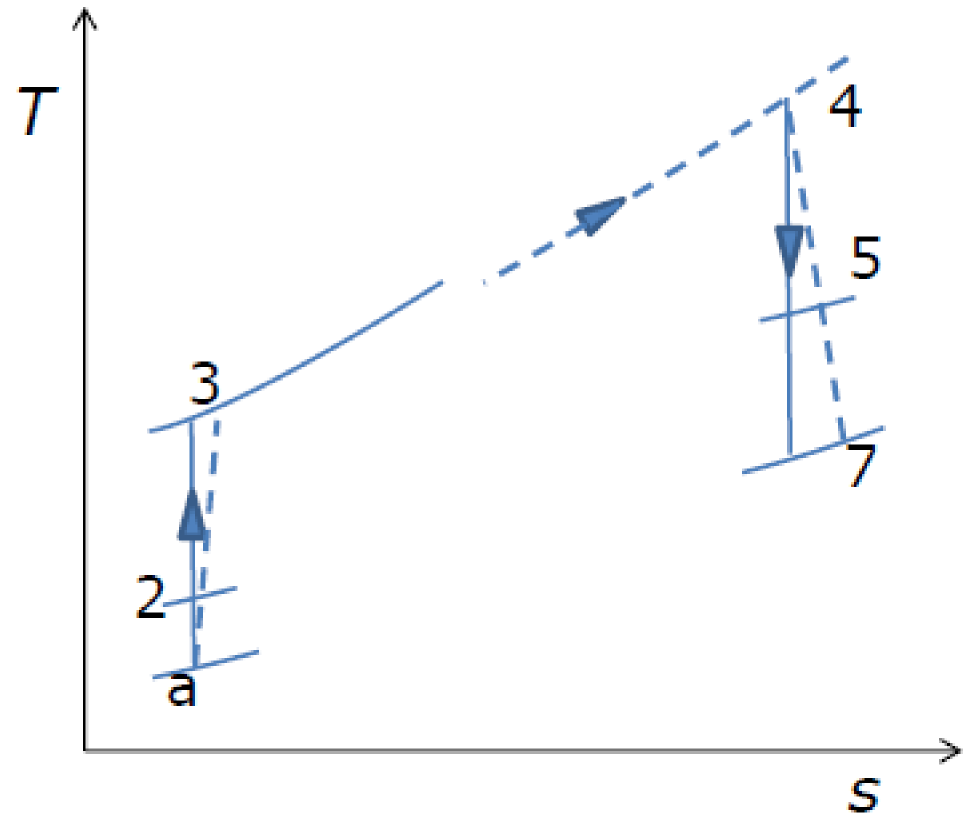
Schematic of a turbojet engine and station numbering scheme

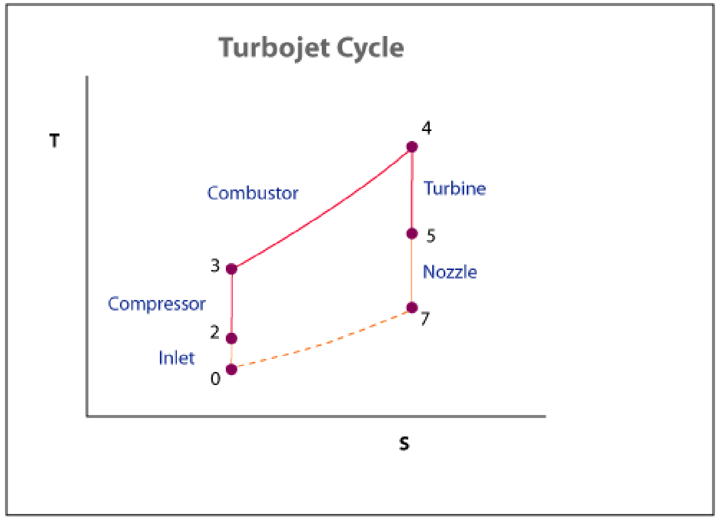
Real cycle for turbojet engines

- The different processes in a turbojet cycle are the following:
- a-1: Air from far upstream is brought to the air intake (diffuser) with some acceleration/deceleration
- 1-2: Air is decelerated as is passes through the diffuser
- 2-3: Air is compressed in a compressor (axial or centrifugal)
- 3-4 The air is heated using a combustion chamber/burner

- 4-5: The air is expanded in a turbine to obtain power to drive the compressor
- 5-6: The air may or may not be further heated in an afterburner by adding further fuel
- 6-7: The air is accelerated and exhausted through the nozzle.

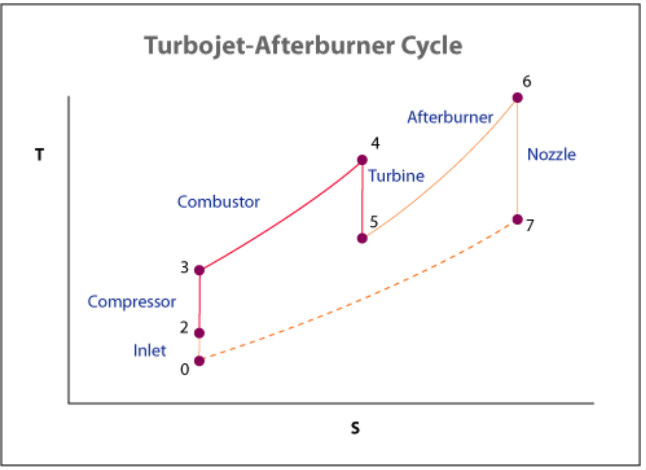
Real cycle for turbojet engines

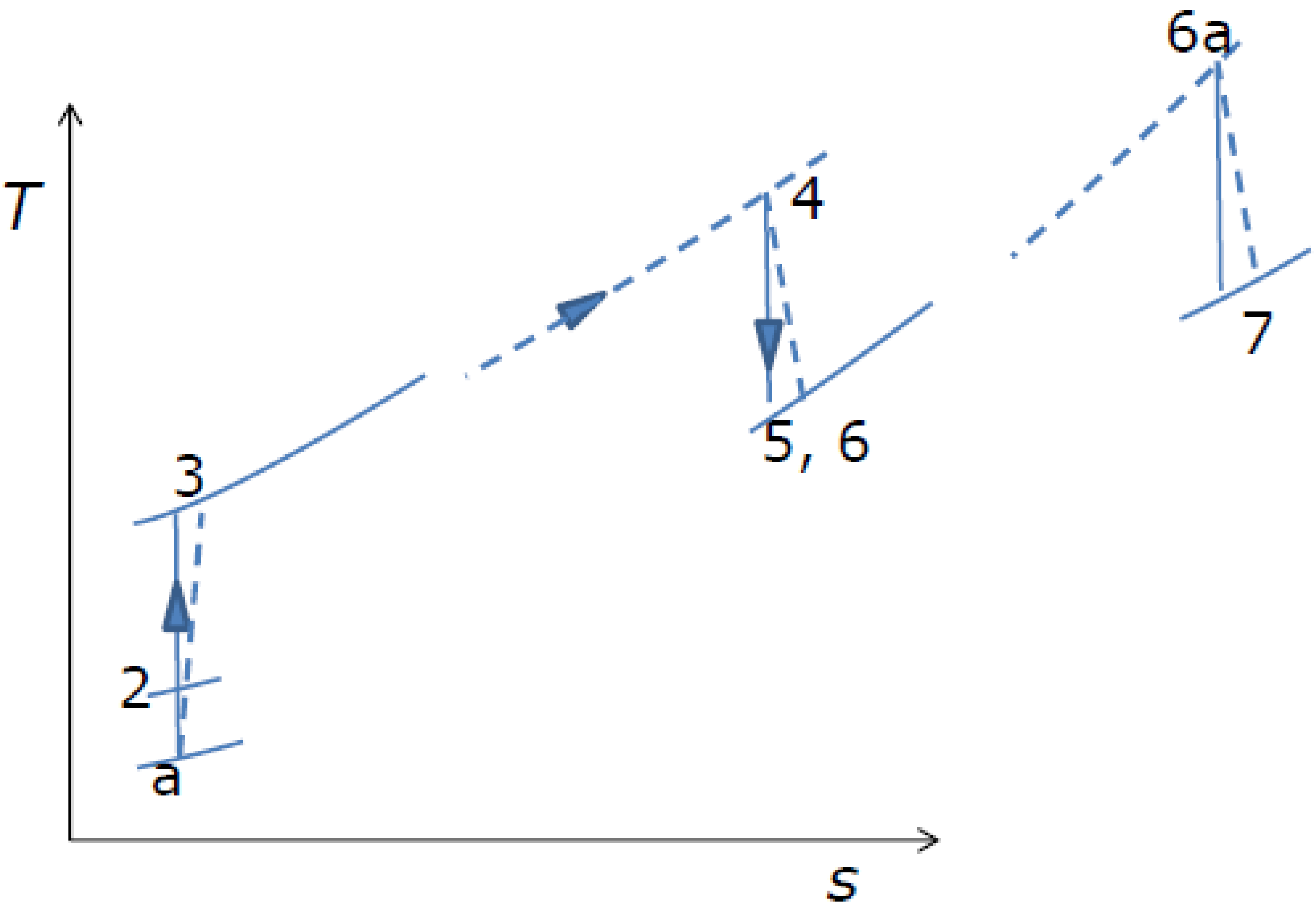




Real turbojet cycle (without afterburning) on a T-s diagram

Real cycle for turbojet engines

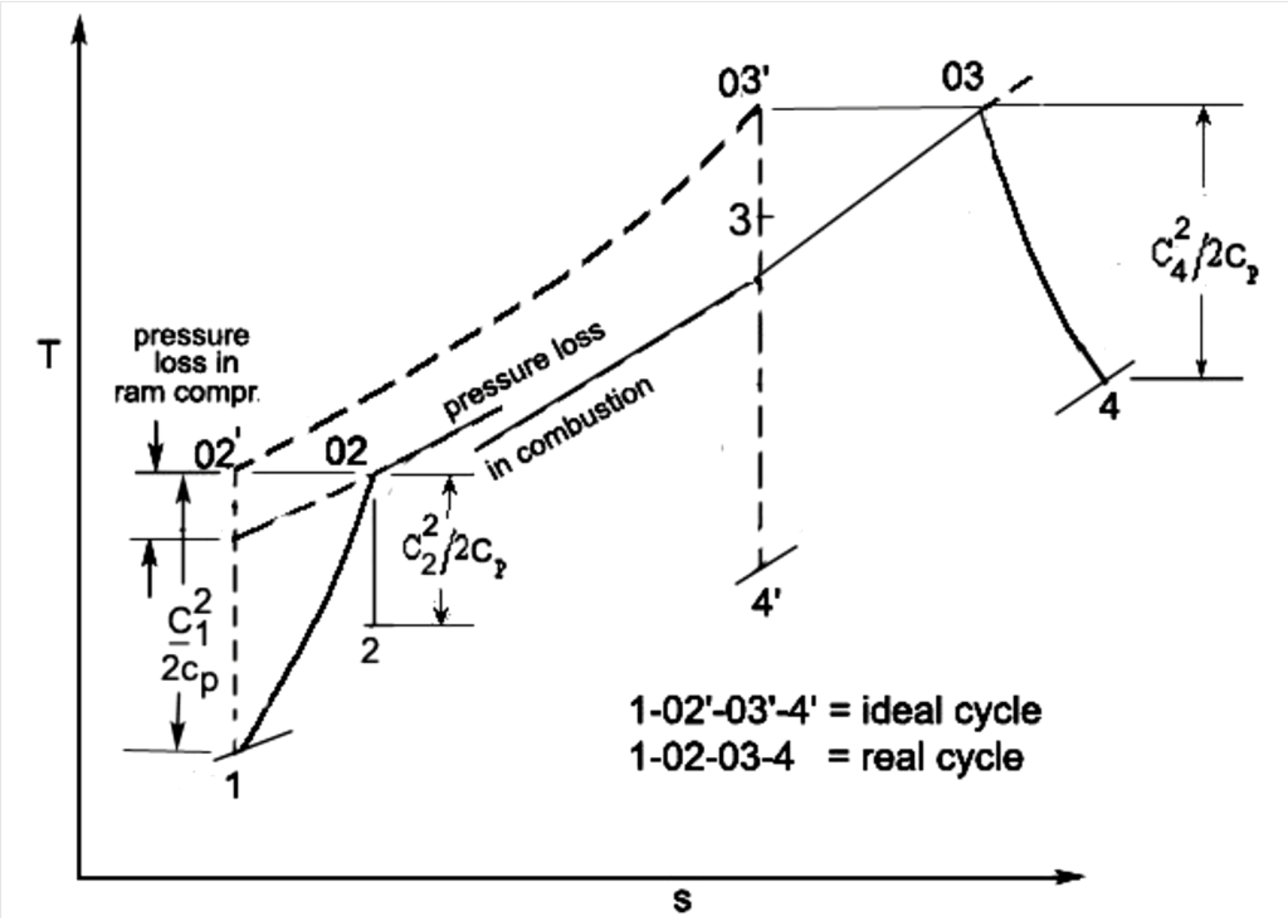




Real turbojet cycle (with afterburning) on a T-s diagram

Performance of Ramjet Engines

Ramjet Engine Thermodynamic cycle



Performance Parameters

- **Parameters**
- Thrust: Ramjet engine thrust is defined as the net change in total momentum as the working fluid passes through the engine. The general expression for thrust,

$$F = \dot{m}(C_4 - C_1) + A_e.(P_e - P_a)$$

where P_e and P_a are the engine exit and ambient static pressures respectively

- The general thrust equation may be modified to include fuel mass flow and then may be rearranged by substituting the mass flow term from the continuity condition,

$$F = \rho . V_a . A_1 \left(\overline{m} . \frac{V_e}{V_a} - 1 \right) + A_e p_a \left(\frac{p_e}{p_a} - 1 \right)$$

- Where, $\overline{m} = 1 + f$, *f*=fuel/air ratio, and A1= Area of free stream air entering the engine

Specific thrust may be written as :

$$\frac{F}{\dot{m}} = V_a.\left(\bar{m}\frac{V_e}{V_a} - 1\right) + \frac{A_e}{A_1}.\frac{p_a}{\rho_a V_a}.\left(\frac{p_e}{p_a} - 1\right)$$

- For a reasonable positive value of specific thrust to be achieved, Either $V_e > V_a$ i.e. substantial acceleration through the engine needs to be accomplished,
- or $p_e > p_a$ i.e a substantial pressure (static) residual (at exit face) inside the engine are required to be achieved.

Design of a Ramjet

- Design of a ramjet engine and performance prediction involves estimation of pressures, temperatures, velocities and flow areas at the critical stations through the engine.
- Even though various analytical CFD techniques, including those incorporating reactive flow, have now come into use, it is still practical to start with an one-dimensional (constant flow properties across any passage area at any station) fluid flow theories.
- CFD techniques require a first cut geometry.
- Deviations from the one-dimensional flow may be corrected for with empirical correlations.

(1) Varying specific heat method

(2) Average specific method

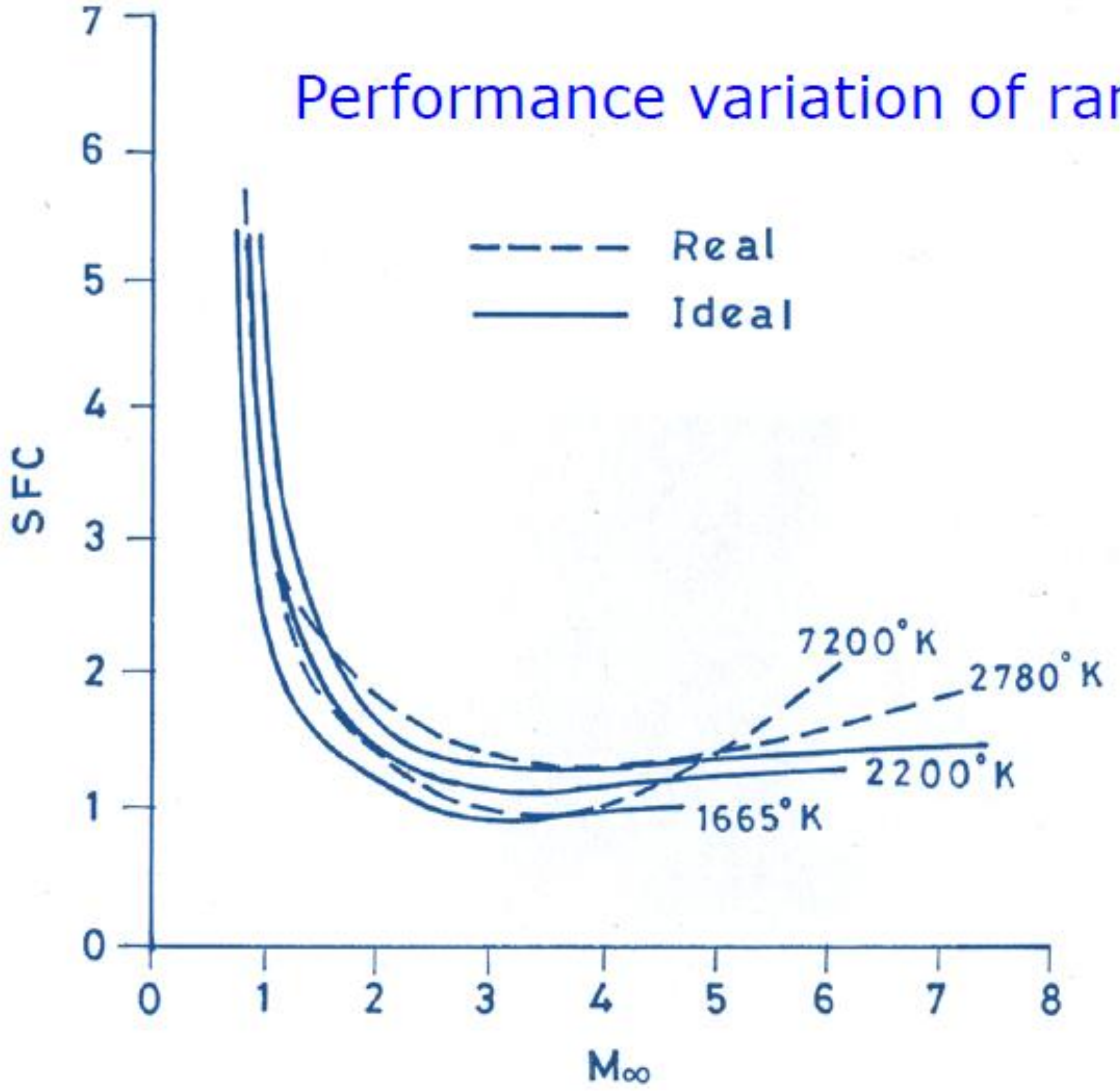
(3) Arbitrary and/or constant specific heat method

Comparative discussion for design of an engine

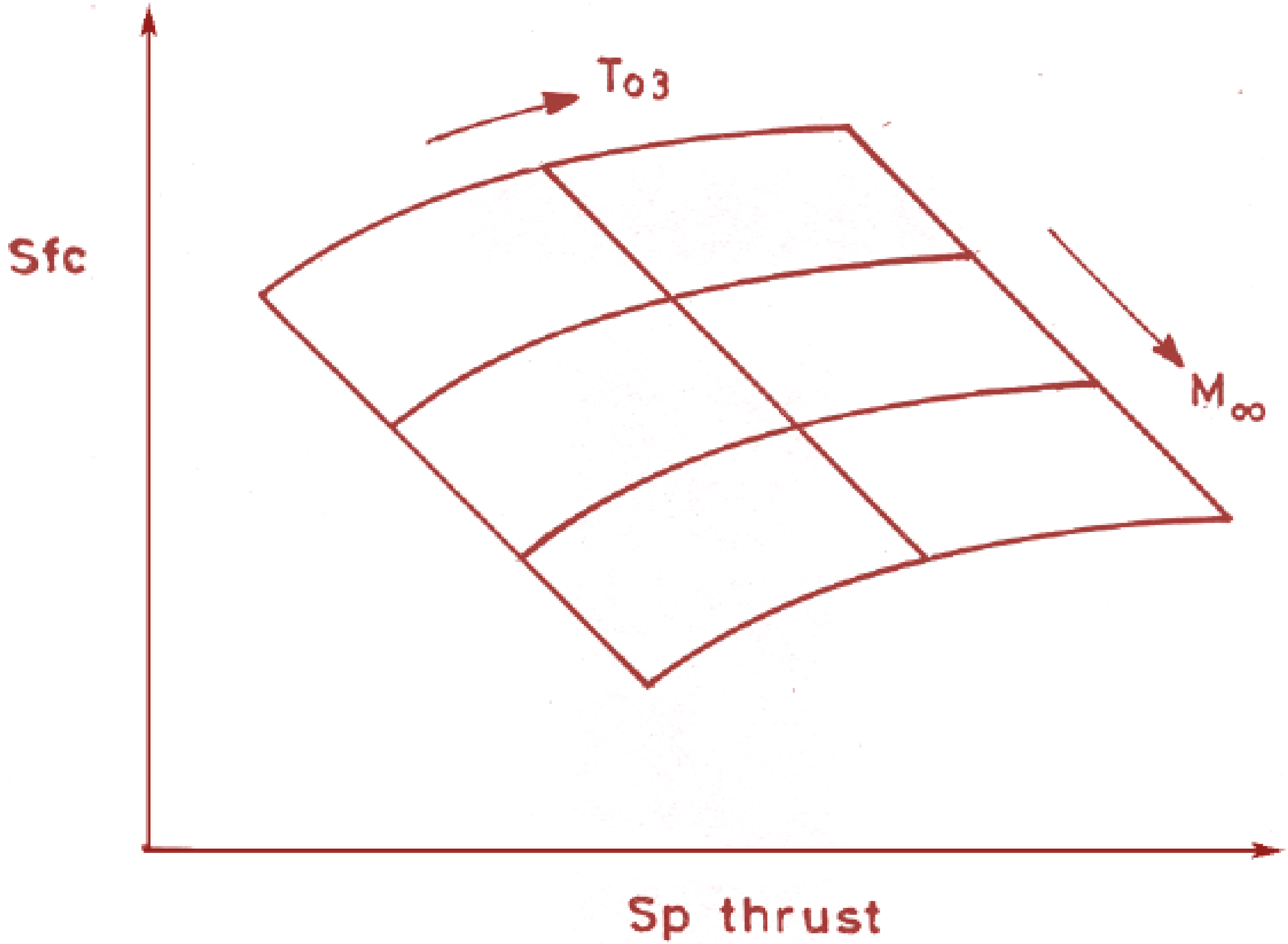
- Comparison of these methods has shown that in a *subsonic* flow arbitrary specific heat method is useful for engineering approximations.
- But at *supersonic* flow conditions only the first two methods should be used for results within acceptable limits.
- Thus in a ramjet (or scramjet), where major portion of the flow is supersonic, last two methods can provide only approximate estimates.
- More accurate estimates shall require use of the first method in a scramjet.

- Under ‘design point’ flow condition accurate analysis is desirable to arrive at or optimize the engine internal flow path geometry.
- On the other hand “off-design point” flow analysis is carried out with engine geometry already available.
- But a number of flow parameters may be unknown variables, requiring simpler approach at the initial stages of analysis, to be followed later on with more rigorous estimations (e.g. CFD).

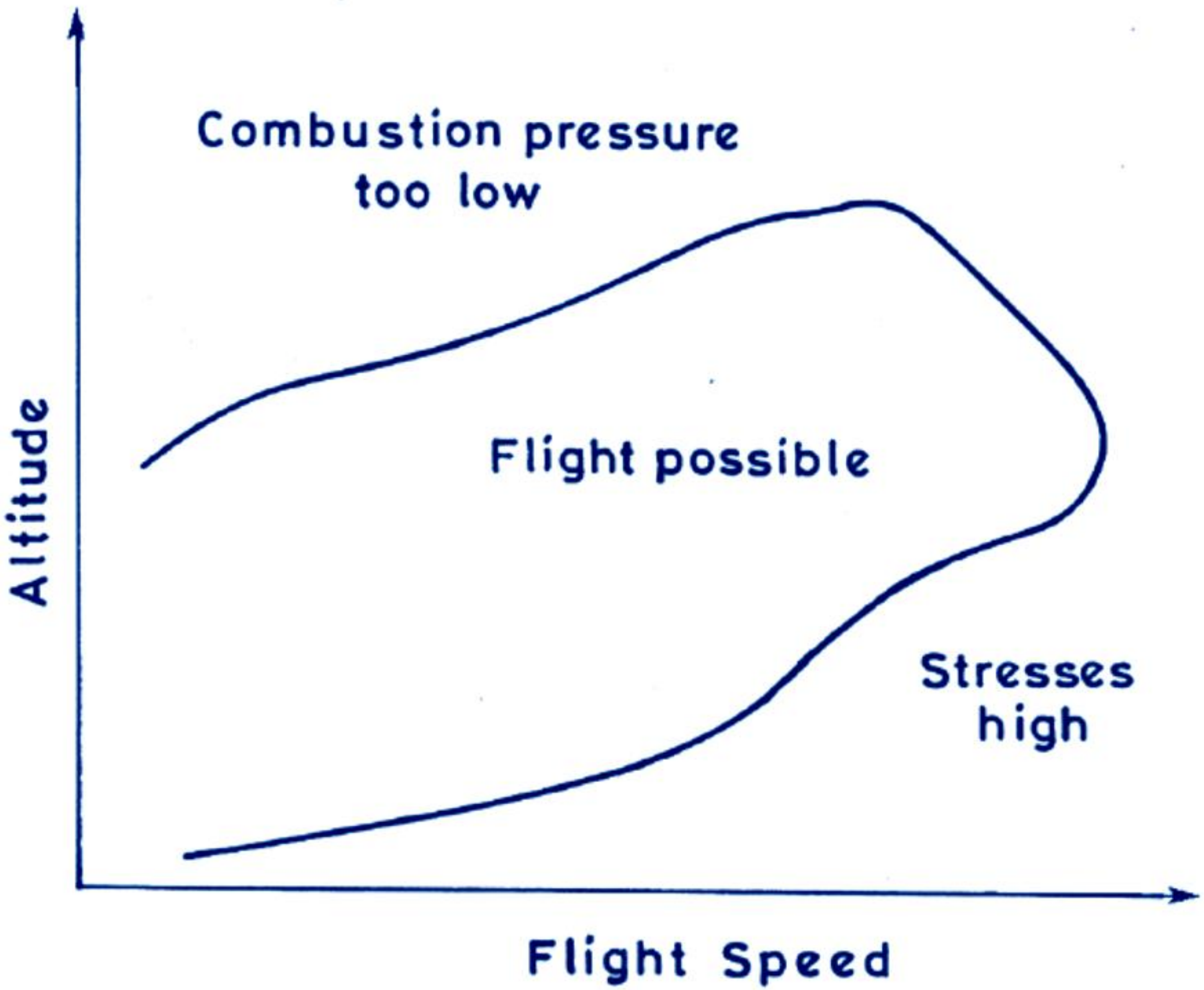
Performance variation of ramjet engine



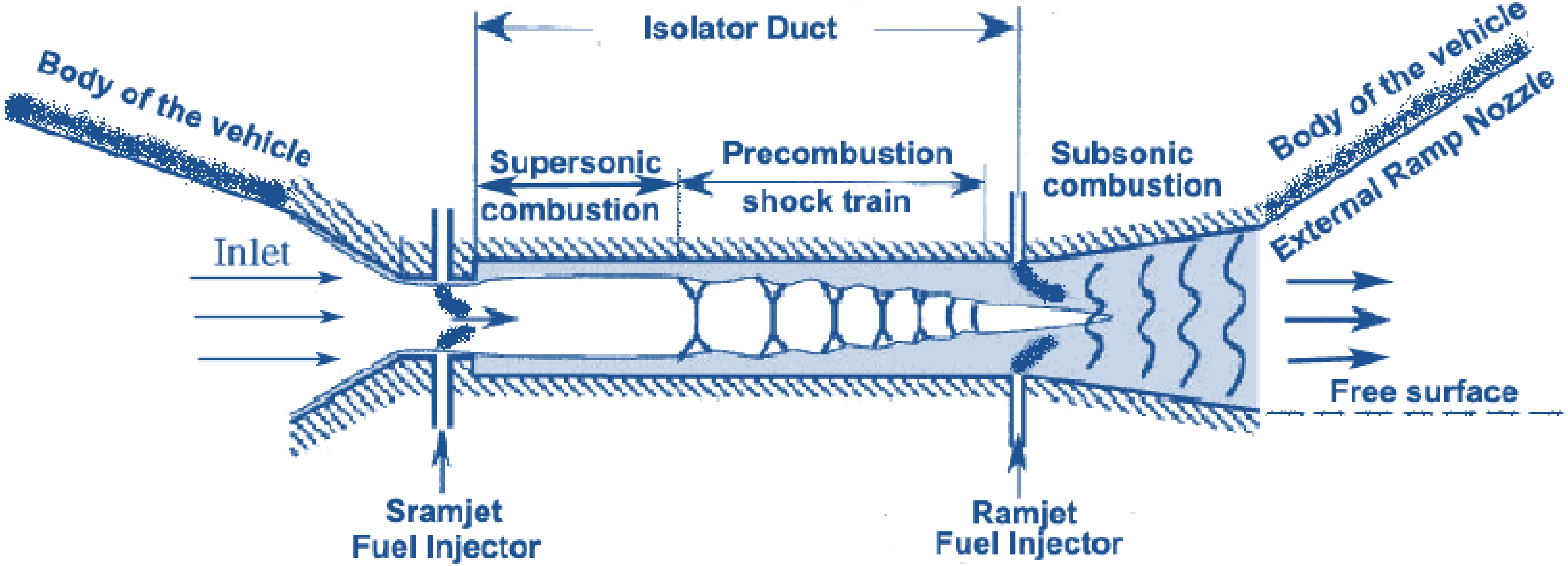
Performance variation of ramjet engine



Performance variation of ramjet engine



Typical Ramjet Engine (with Dual combustion – Sub and Supersonic Combustion)



Need for supersonic combustion-

- Supersonic combustion air-breathing engines have long been recognized as the most well-suited for hypersonic propulsion in the Mach 5-10 range.
- Designs for hypersonic engines have been around since the early 1900's.
- Ramjet technology has been developing over the past eight decades and, except for marginal improvements, has been shown to be suited for atmospheric flight speeds up to Mach 5.
- The desire for faster, more efficient engines gave birth to the idea of a scramjet, utilizing supersonic combustion and potentially expanding the speed envelope to the Mach 15 range.
- The promise of covering the entire planet at high speed from horizontal takeoff for both civil and military aircraft is an attractive prospect.

Implications criticality of efficient diffusion and acceleration, problems of combustion in high speed flow

- Although the concept of scramjet engines appears simple, supersonic combustion remains a complex field of study.
- Chemical kinetics, temperature, pressure, equivalence ratio, mixing rate and stream velocity all affect the combustion process.
- As it stands, supersonic combustion is very difficult to maintain and continues to be a formidable task.
- The ignition delay time of a fuel-air mixture continues to be the limiting factor for all scramjet engine designs. Decreasing the delay time allows for shorter combustors and/or higher flight velocities.
- Initially, the ignition delay time of a fuel is fixed for a given set of conditions and the type of fuel. Increasing the temperature of the fuel and/or air stream reduces this time.

- Pressure plays a somewhat more complex role. Increasing the pressure, usually, but not always, improves the combustion conditions. Increasing pressure usually reduces the ignition delay time, but there exists a critical value of pressure, above which, the delay time increases dramatically, followed by a slow decrease.
- Therefore, it is not always advantageous to increase the pressure. The equivalence ratio does not strongly affect the ignition delay time, except for equivalence ratios below 0.3, where the delay time increases sharply. Therefore, these effects need to be considered in designs.

Proposed Solutions to Supersonic Combustion Difficulties

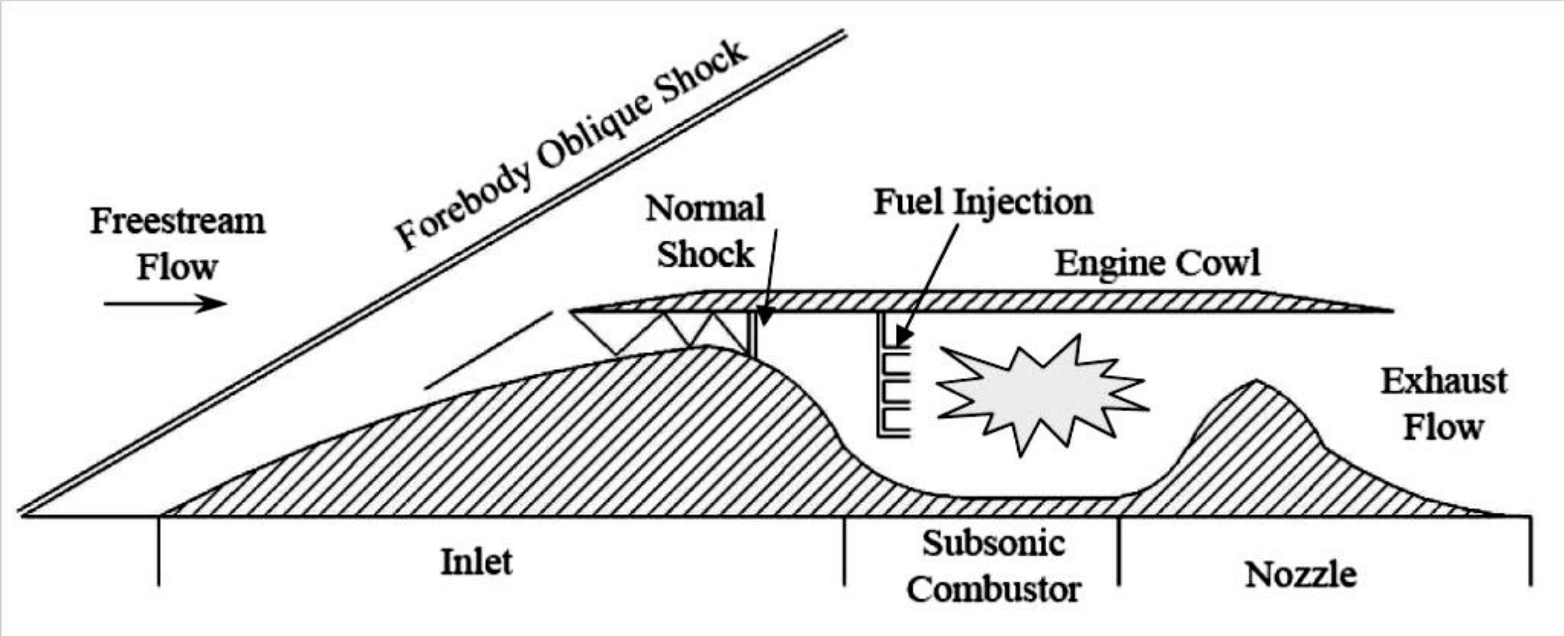
- Improved schemes for injection patterns have been designed and studied to overcome the obstacles of inadequate fuel penetration and mixing.
- In addition, the problem of ignition and flame holding can be handled in one of two ways; injecting combustion enhancing radicals by use of a plasma torch can reduce the induction time of the mixture or recirculation zones can be created using aerodynamic bodies such as wedges, ramps or cavities to slow down the flow and provide an environment where combustion can occur.
- A combination of these methods can also be used.

- Throughout the recent decades, scramjet developers have overcome problems with material capabilities, ignition, flame holding and internal drag, to name a few.

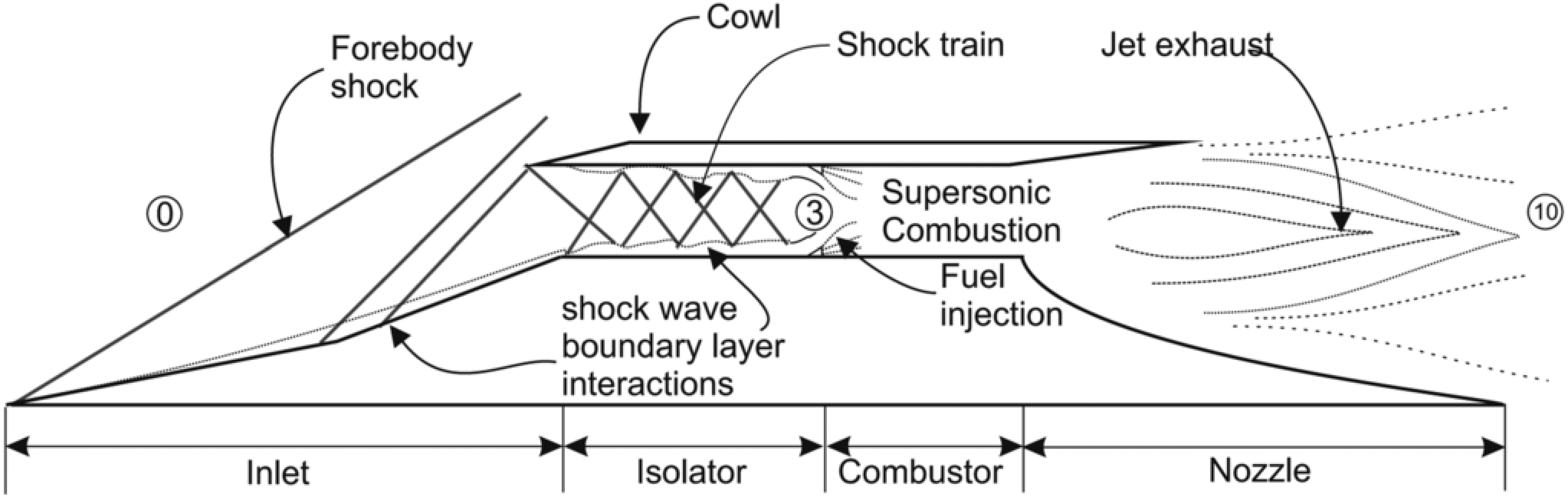
- **Fuel Injection**
- **Recessed Cavity Flameholders**
- **Ramps and Wedges**
- **Plasma Torches**

The Scramjet Engine

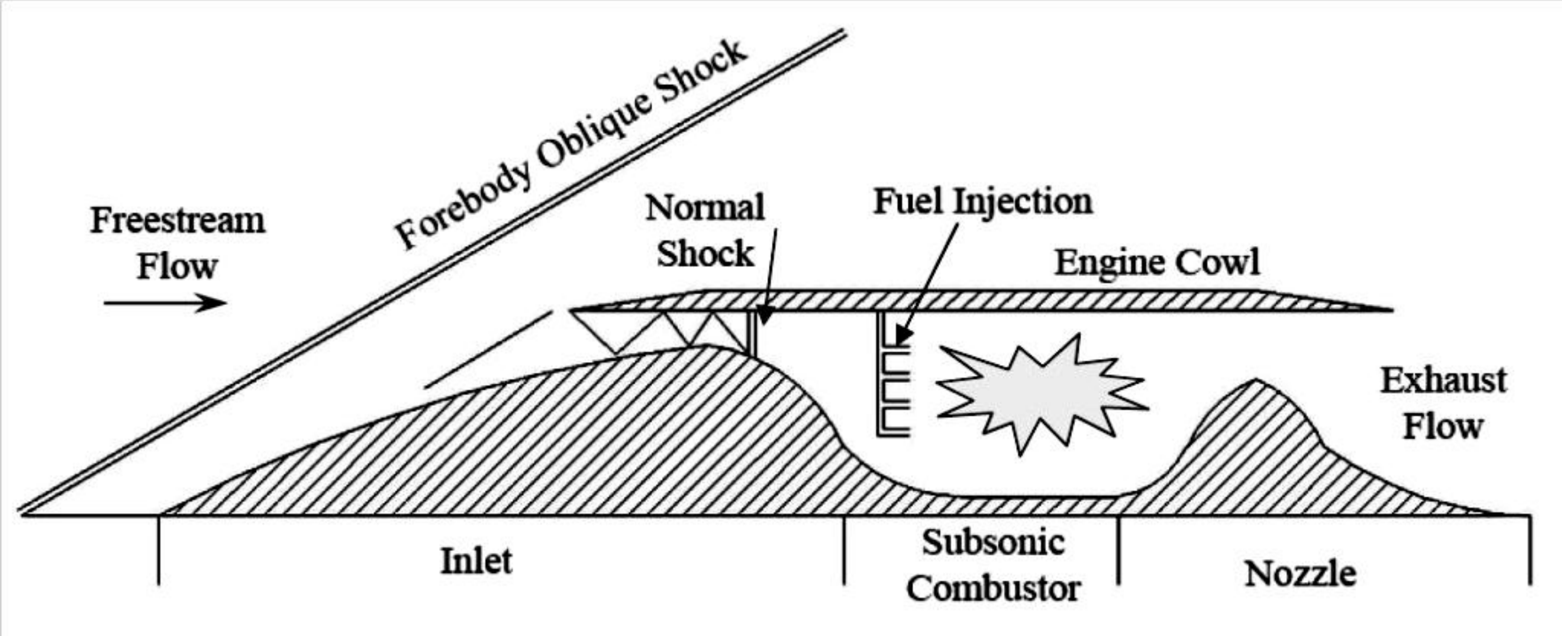
- The definition of a ramjet engine is first necessary, as a scramjet engine is a direct descendant of a ramjet engine.

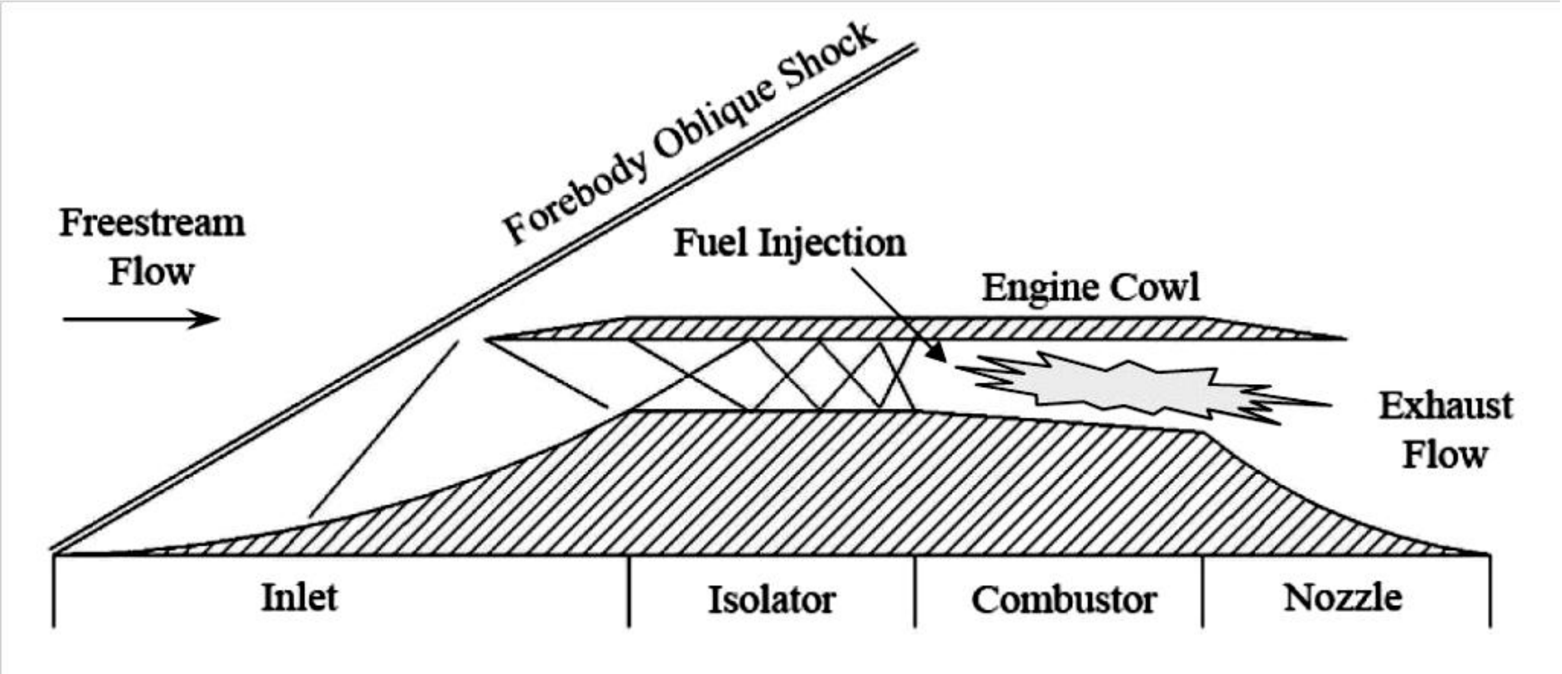


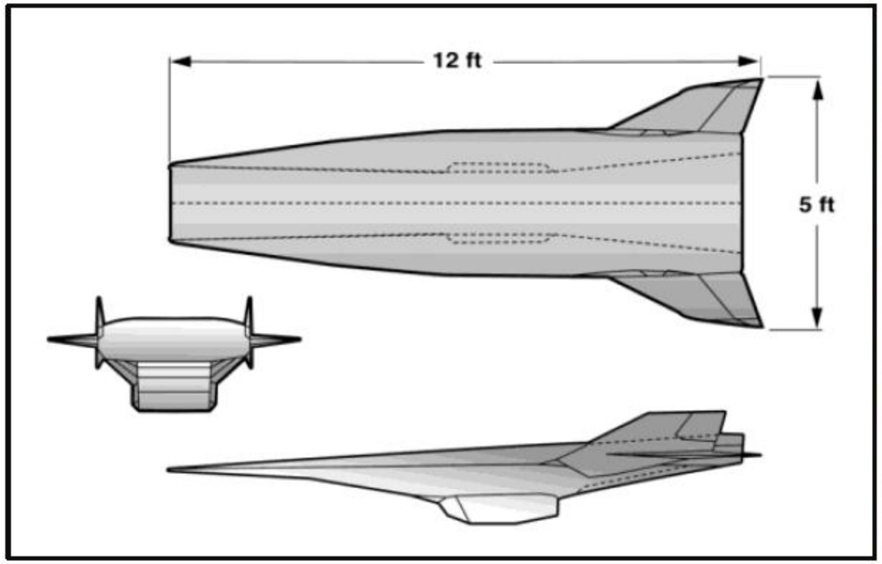
Schematic diagram of a ramjet engine

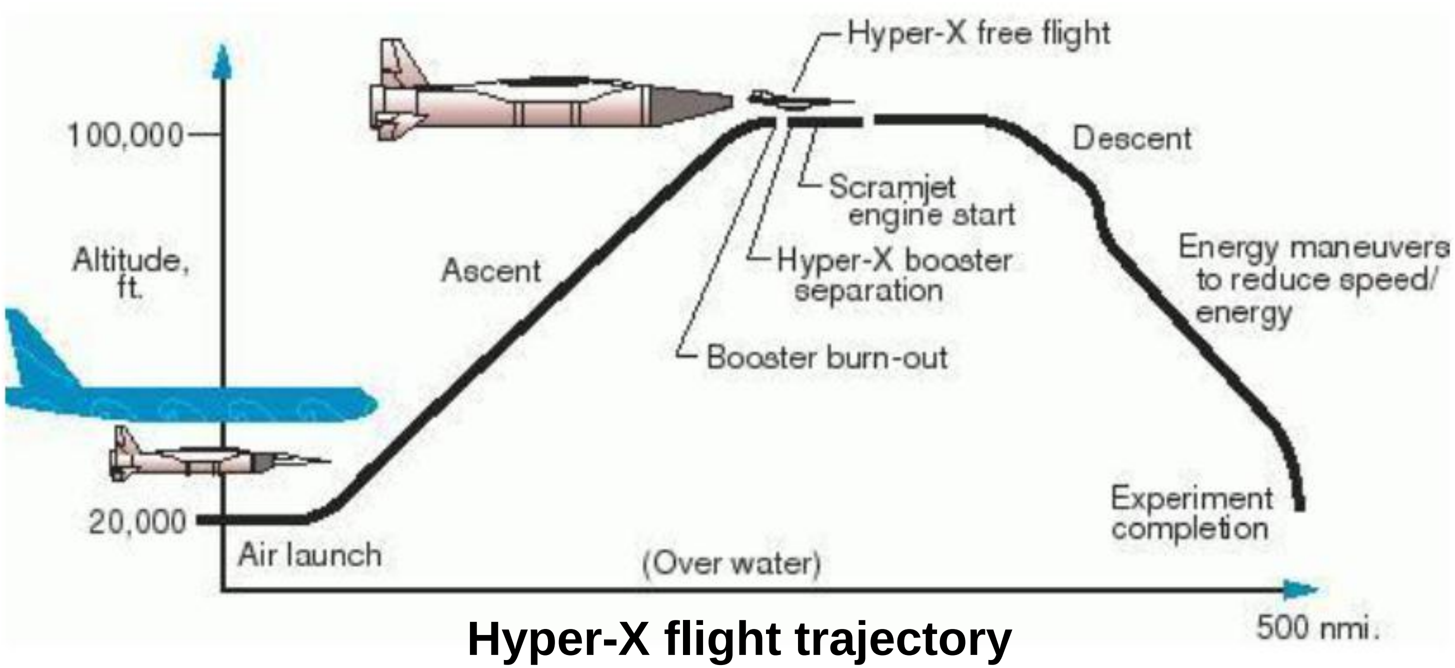


Schematic diagram of a scramjet engine


Forebody Oblique Shock
Freestream Flow
Normal Shock
Fuel Injection
Engine Cowl
Exhaust Flow
Inlet
Subsonic Combustor
Nozzle


Forebody Oblique Shock
Freestream Flow
Fuel Injection
Engine Cowl
Exhaust Flow
Inlet
Isolator
Combustor
Nozzle

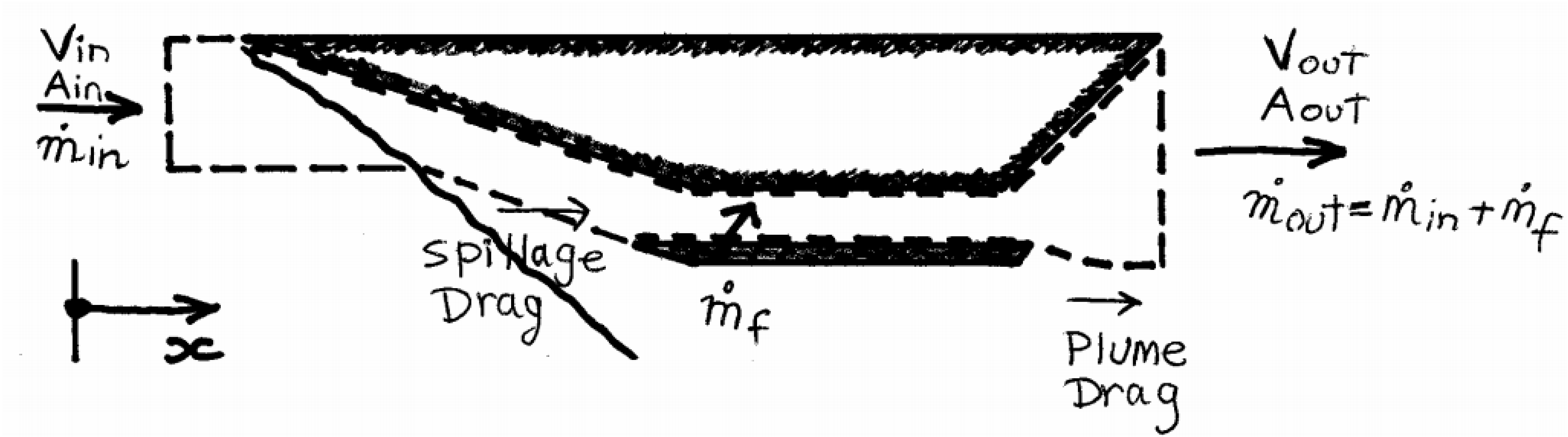




Hyper-X flight trajectory

SCRAMJET PERFORMANCE ANALYSIS

- The performance of a scramjet engine, either uninstalled or when integrated on a hypersonic vehicle, is most easily determined by what is called stream thrust analysis.
- This technique conserves the fluxes of mass, momentum and energy on strategically placed control volumes to determine the propulsive forces on the vehicle.
- Figure shows a schematic of a scramjet powered vehicle with a control volume surrounding all the airflow that passes through the engine.
- 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 usually represented by a flux-conserved onedimensional average of the non-uniform exhaust plume. In the current analysis only the axial forces will be considered, however, similar relations can be developed for the transverse direction to determine the lift forces generated by the propulsion system.



Schematic of control volume used for scramjet performance analysis

- Assuming for simplicity that fuel is added with no component of velocity in the streamwise direction, application of Newtons law to the control volume in Fig. in the streamwise direction yields the following relation:

$$\dot{m}_{in}V_{in} + p_{in}A_{in} - (\dot{m}_f + \dot{m}_{in})V_{out} + p_{out}A_{out} + \sum F_{xs} = 0$$

where $\sum F_{xs}$ = sum of the pressure and viscous forces on the top and bottom boundaries of the control volume.

It is customary to separate the addative drag due to inlet spillage and the nozzle plume from $\sum F_{xs}$ as follows:

$$\sum F_{xs} = F_{add} + F_{surface}$$

Re-arranging eqn. 1 yields:

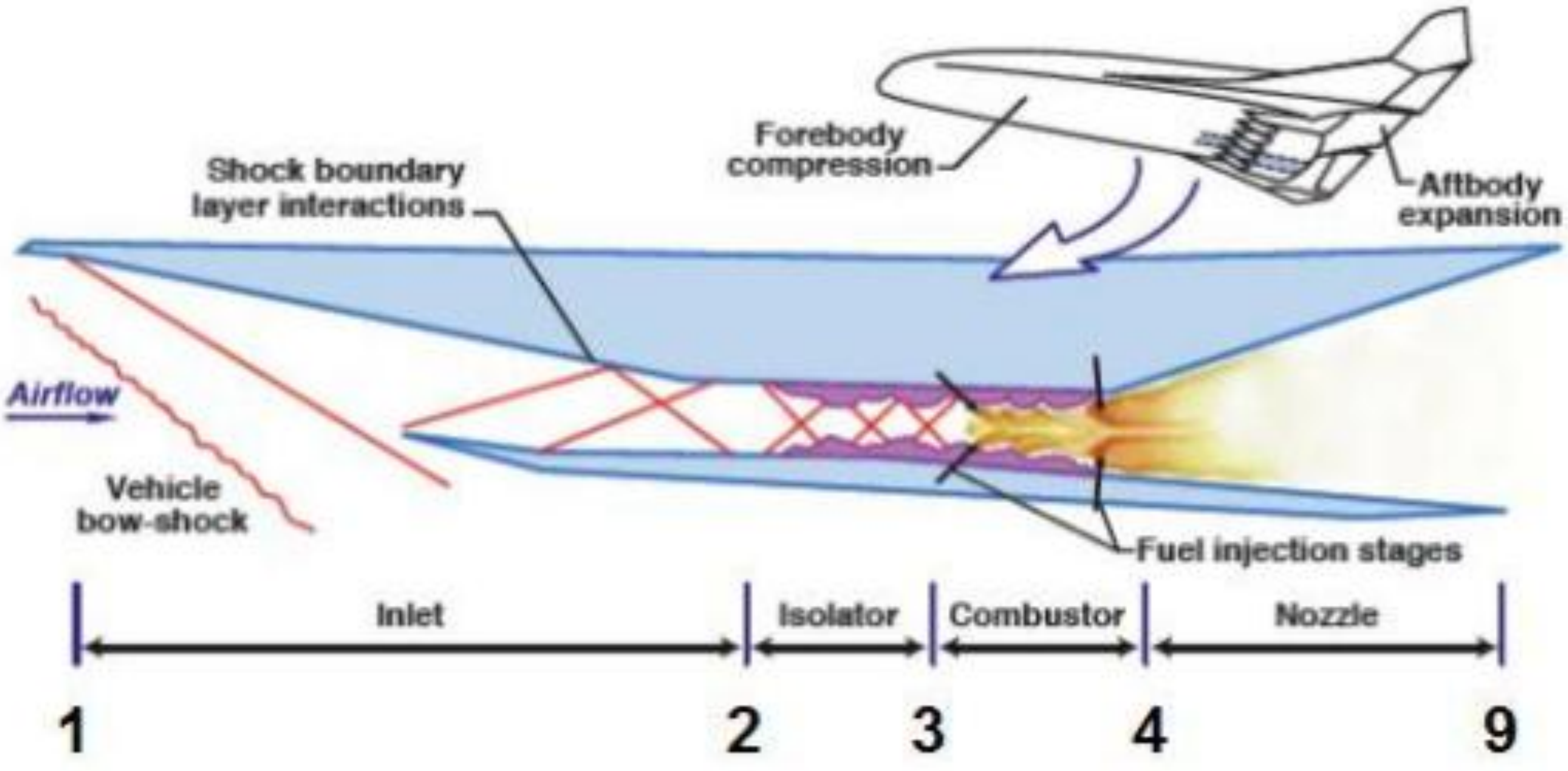
$$F_{surface} = (\dot{m}_f + \dot{m}_{in})V_{out} + p_{out}A_{out} - \dot{m}_{in}V_{in} - p_{in}A_{in} - F_{add} \qquad (2)$$

The left hand side of eqn. 2 is the thrust of the uninstalled engine, F_{un}. Using the definition of stream thrust, $F = pA + \dot{m}V$, we can express eqn. 2 as:

$$F_{un} = F_{out} - F_{in} - F_{add} \qquad (3)$$

- Equation 3 indicates that the uninstalled thrust of an engine can be determined with knowledge of the stream thrust of the air entering the engine, the addative drag, and the stream thrust exiting the engine nozzle.
- The flow enters the engine at ambient conditions and at the flight velocity, so determination of F_{in} reduces to a determination of the freestream capture area.
- Air spillage (and therefore spillage drag) decreases as the vehicle speed approaches the design point, and the plume drag varies depending on the amount of under-expansion in the nozzle.
- Both these are usually estimated through CFD analysis, or through rules-of-thumb based on empirical or experimental databases. Determination of F_{ex} requires an involved analysis that follows the air through the complete scramjet flowpath.

Component Analysis



- The different parts of a scramjet engine: air inlet, isolator, combustor and nozzle.
- The scramjet engine must be integrated with the fuselage of the aircraft, specially the air inlet and the nozzle.
- Part of the forebody aircraft fuselage makes the function of air inlet compressing the freestream air, and similarly, the aftbody acts as a nozzle expanding the gases from the combustion.
- The flow path through a scramjet engine follows a Brayton thermodynamic cycle. The air is compressed, after that combustion takes place to increase the flow temperature and pressure, and finally, the products from the combustion are expanded.

Air inlet

- The air inlet can be considered as a diffuser in which takes place the compression of the freestream air gathered. This compression is achieved by successive shock waves.
- For an oblique shock wave, the flow is always deflected towards the shock wave, and the flow properties vary as

$$
\begin{array}{ccc}
M_{2n} < 1 & M_2 < M_1 & \\
T_{t2} = T_{t1} & p_{t2} < p_{t1} & \rho_{t2} < \rho_{t1} \\
T_2 > T_1 & p_2 > p_1 & \rho_2 > \rho_1
\end{array}
$$

- Here the subscript 1 and 2 corresponds respectively just before and just after the shock wave.

- Therefore, at the exit of the air inlet, the supersonic freestream air has reduced its velocity and has raised its static temperature and pressure. Furthermore, the total temperature through the air inlet is constant and the total pressure changes

$$\frac{T_{t2}}{T_{t0}} = 1 \qquad \frac{p_{t2}}{p_{t0}} < 1$$

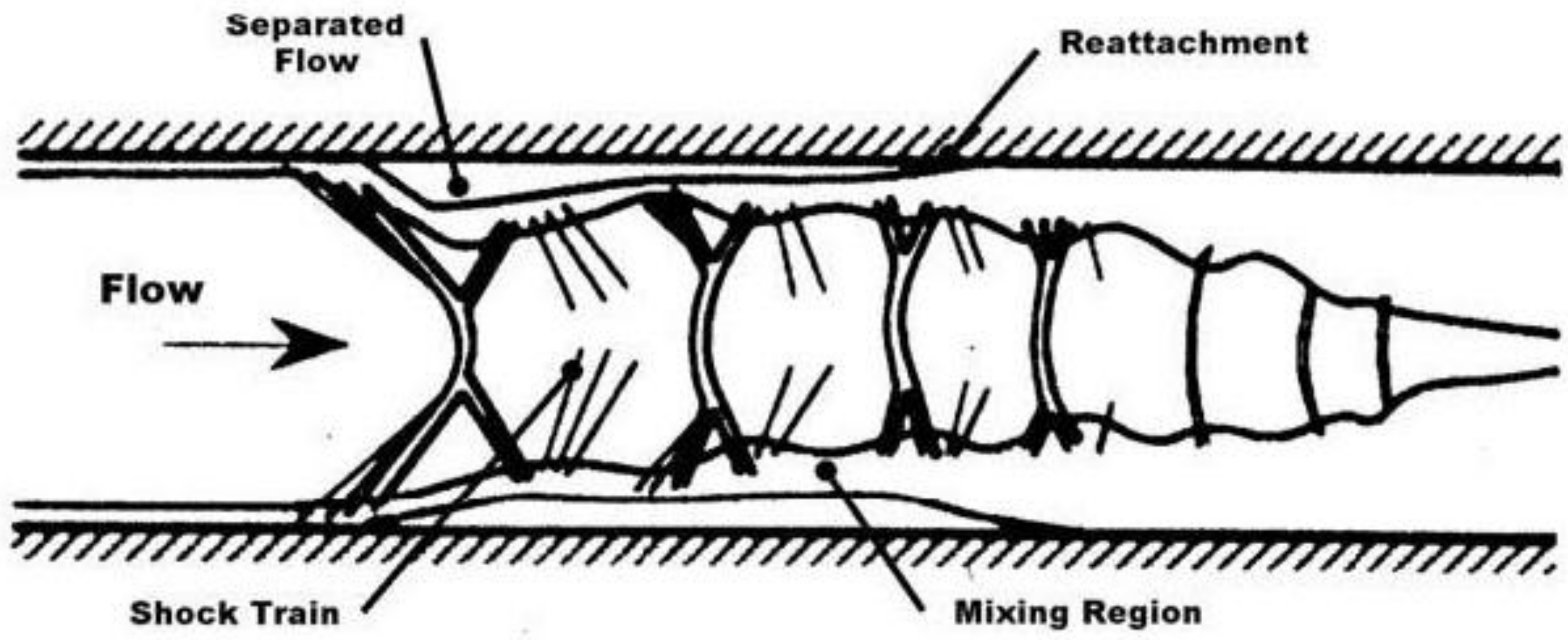
- The subscript 0 represents freestream conditions before forebody compression.
- The performance of the air inlet compression can be separated into two key parameters: capability, or how much compression is performed, and efficiency, or what level of flow losses does the inlet generate during the compression process.
- A common parameter used to quantify the efficiency of the forebody/inlet compression is the kinetic energy efficiency.

- The definition is the ratio of the kinetic energy the compressed flow would achieve if it were expanded isentropically to freestream pressure, relative to the kinetic energy of the freestream.

$$\eta_d = \frac{\frac{1}{2}u'^2_2}{\frac{1}{2}u_0^2} = \frac{u'^2_2}{u_0^2}$$

Isolator

- At flight speeds below Mach 8, combustion in a scramjet engine can generate a large local pressure rise and separation of the boundary layer on the surfaces of the combustion duct. This separation, which can feed upstream of fuel injection, acts to further diffuse the core flow in the duct, and will affect the operation of the inlet, possibly causing an unstart of the engine.
- The method use to alleviate this problem is the installation of a short duct between the inlet and the combustor known as an isolator.
- In some engines (those which operate in the lower hypersonic regime between Mach 4 and 8) the combination of the diffusion in the isolator and heat release in the combustion decelerate the core flow to subsonic conditions, in what is called dual-mode combustion.
- At speeds above Mach 8 the increased kinetic energy of the airflow through the engine means that the combustion generated pressure rise is not strong enough to cause boundary layer separation. Flow remains attached and supersonic throughout, and this is termed pure scramjet. In this case an isolator is not necessary.

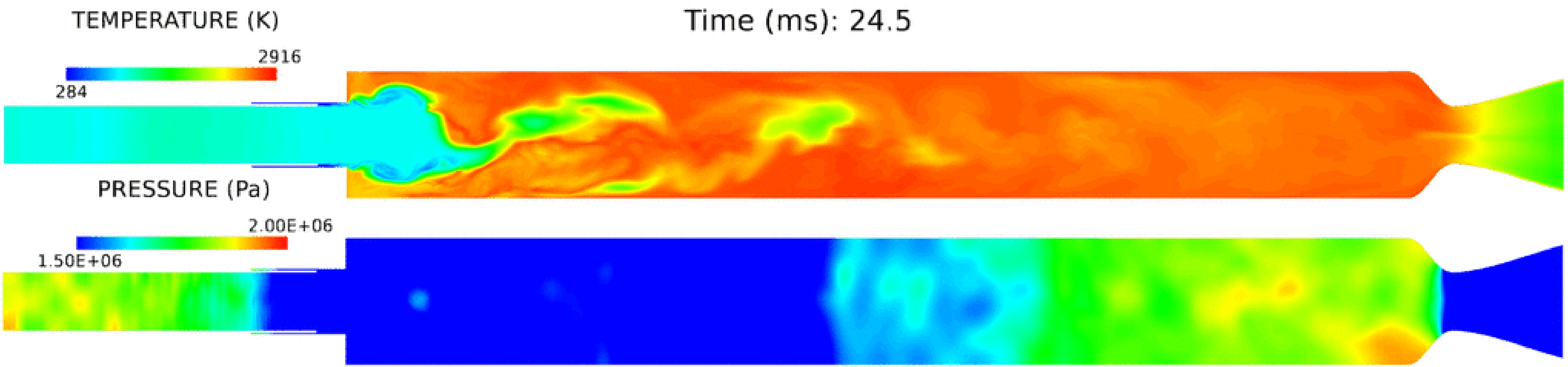


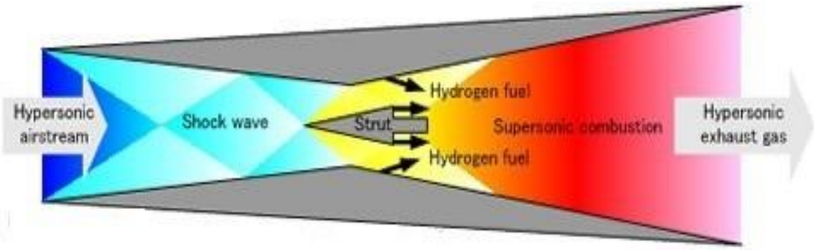
Schematic of flow structure in an isolator

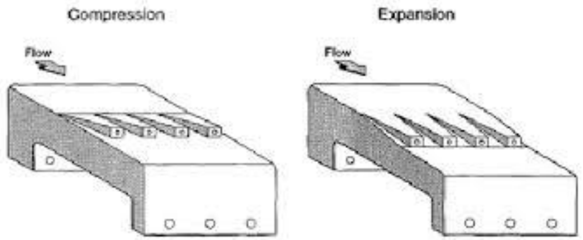
- The structure of the supersonic flow in confined ducts under the influence of a strong adverse pressure gradient is of interest in the design of scramjet isolators.
- As shown in Figure, a pressure gradient is imposed on the incoming supersonic flow, and with the presence of a boundary layer, a series of crossing oblique shocks are generated. This phenomenon, known as pseudo shock or shock-train, is characterized by a region of separated flow next to the wall, together with a supersonic core that experiences a pressure gradient due to the area restriction of the separation and forms the series of oblique shocks mentioned before.
- Finally, the flow reattaches at some point and mixes out to conditions that match the imposed back-pressure. Being able to predict the length scale of this flow structure is the key component of isolator design for dual-mode scramjets.

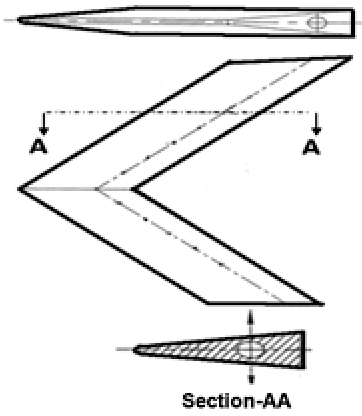
Combustor

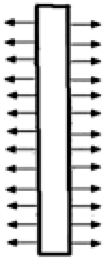
- The combustor chamber is a duct where the combustion between freestream air and fuel takes place. This combustion is supersonic, so there are some aspects that require more attention on the contrary of the conventional combustion.
- At very high velocities, a properly fuel injection and mixing could be a problem, as well as flame holding. That is why over the past decades a lot of different configurations have been studied and developed.
- Some techniques used today for fuel injection in scramjet engines are: wall, ramp, strut, pylon and pulsed injectors. And for keeping the combustion, there is a technique quite used called cavity flame holders.

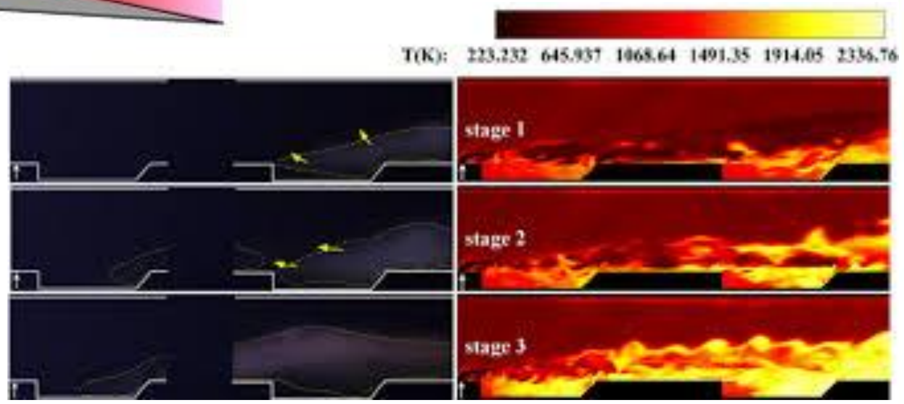

TEMPERATURE (K)
Time (ms): 24.5
2916
284
PRESSURE (Pa)
2.00E+06
1.50E+06

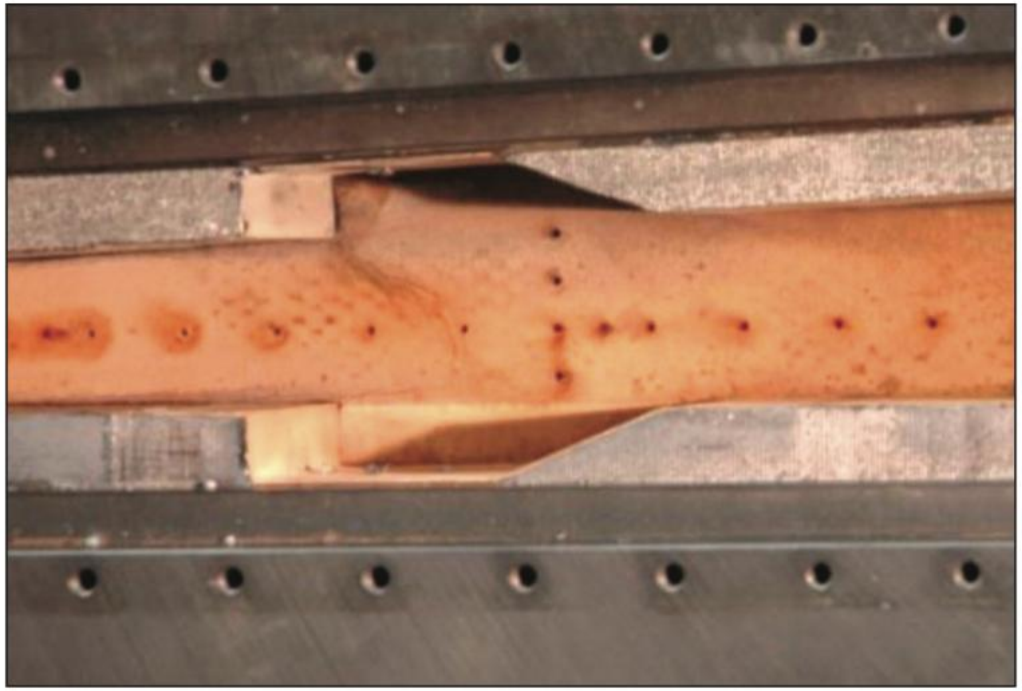

Hypersonic airstream
Shock wave
Strut
Hydrogen fuel
Supersonic combustion
Hydrogen fuel
Hypersonic exhaust gas


Compression
Expansion
Flow
Flow


A
A
Section-AA




T(K): 223.232 645.937 1068.64 1491.35 1914.05 2336.76
stage 1
stage 2
stage 3



- Another significant aspect to take into account is the dissociation. At the entrance of the combustor the flow static temperature and pressure are very high, and with the heat release due to chemical reactions, the temperature and pressure could reach extremely high values which involve dissociation of combustion products.
- Because of the heat addition, the velocity or Mach number decreases while the static temperature and pressure increases.
- The total temperature is raised and the total pressure is reduced. The total pressure loss is proportional to the square of Mach number; hence, it is better to have a small combustor inlet Mach number, on the contrary for the dissociation phenomenon.

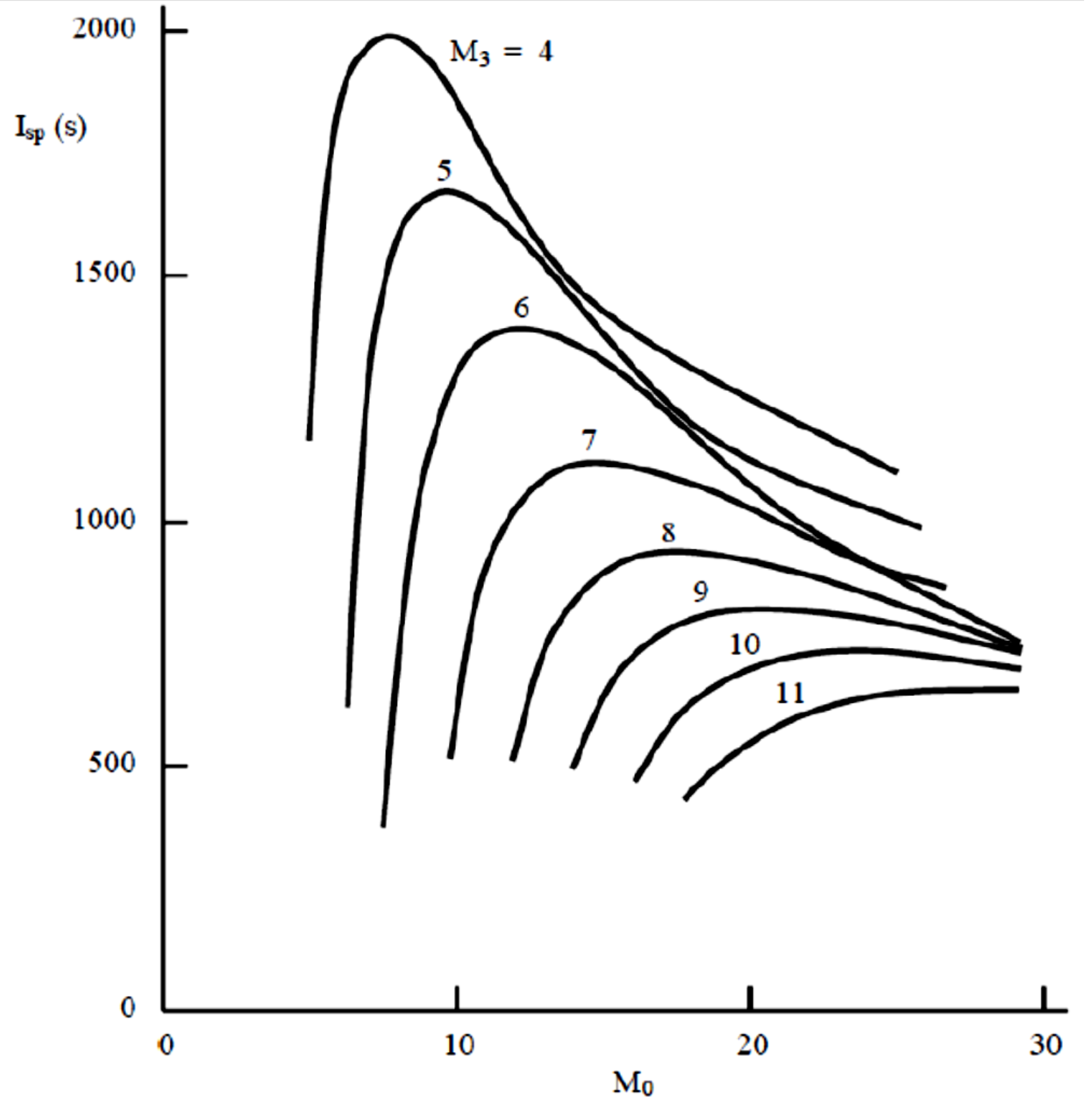
Nozzle

- The nozzle is a divergent duct that accelerates the supersonic flow and at the same time expands it reducing its static temperature and pressure.
- The expansion process converts the potential energy of the combusting flow to kinetic energy and then it results in thrust. An ideal expansion nozzle would expand the engine plume isentropically to the freestream pressure assuming chemical equilibrium.
- Nevertheless, loss mechanisms are present in real expansion processes and are due to under-expansion, failure to recombine dissociated species, flow angularity and viscous losses.
- The weight of a fully-expanded nozzle would be prohibitive at most hypersonic flight conditions; hence under-expansion losses are usually traded against vehicle structural weight.

- Dissociation losses result from chemical freezing in the rapid expansion process in the nozzle, essentially locking up energy that cannot be converted to thrust.
- Flow angularity losses are product of varying flow conditions in the nozzle, and viscous losses are associated with friction on the nozzle surfaces.
- The flow enters the nozzle in a highly reactive state. As it expands to lower pressure and temperature, chemical reactions will occur toward the completion of combustion, with consequent additional heat release.
- If the expansion is slow enough chemical equilibrium is approached, but in most cases, due to the high velocities reached in a scramjet, the flow composition freezes and becomes fixed.
- Two limiting cases can be treated fairly easily: equilibrium flow, where equilibrium is maintained through all nozzle length, and frozen flow, where the flow doesn’t change its composition from the combustor exit. The true situation lies between these two cases.

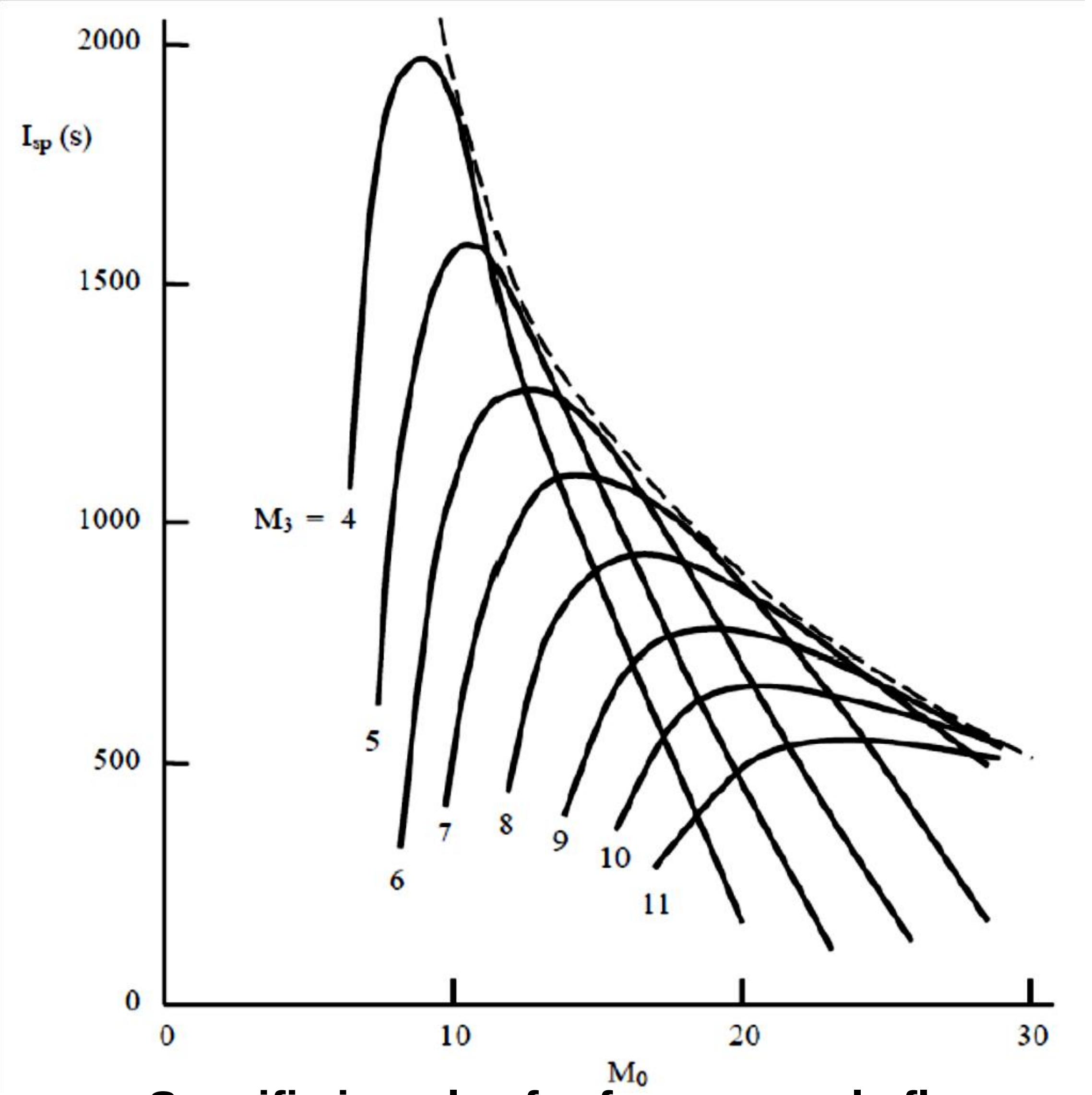
- The choice of combustor inlet Mach number is a key aspect for the performance of the scramjet and it is related to the nozzle expansion.
- If the static temperature at the combustor entrance is too high, dissociation will be present and then chemical energy is not available as thermal energy for conversion to kinetic energy in the nozzle.
- So, the question to be dealt with quantitatively is then what static temperature, or what combustor inlet Mach number, is best for any given flight Mach number.
- The existence of such an optimum M_3, which depends on finite chemical reaction rates, can be seen by comparison of the specific impulse for two limiting cases: one which chemical equilibrium is assumed throughout the flow, and another in which the flow is assumed to be in equilibrium up to the combustor exit but frozen at that composition during the nozzle expansion.

- As it can be seen in Figure, for equilibrium nozzle flow there is no optimum M3, the specific impulse increases continuously as M_3 decreases.
- This is not surprising, because for equilibrium flow the chemical energy invested in dissociation is recovered as thermal energy and then kinetic energy as recombination occurs in the nozzle, and the lower the combustor Mach number the lower the entropy increase in the combustor. Therefore, independently on the flight Mach number interests a low combustor inlet Mach number.

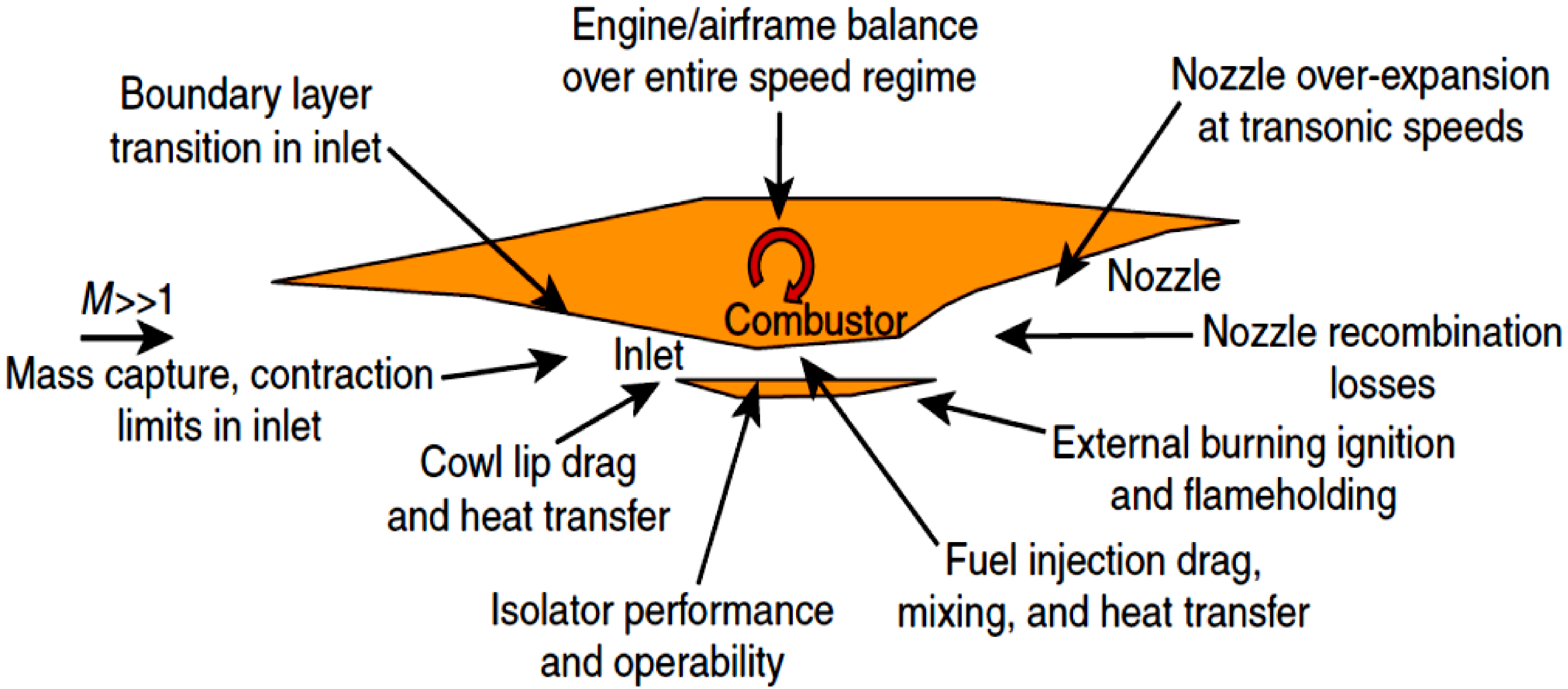


Specific impulse for equilibrium nozzle flow (combustor pressure fixed at 1 atm)

- In contrast, for frozen nozzle flow, there is a clear optimum M_3 for flight Mach numbers above about 10. It is defined by the envelope, drawn as a dashed line. The optimum value of M3 depends on the extent to which recombination occurs in the nozzle, as well as on the degree of dissociation at the combustor exit. For a very high flight Mach number, if the flow is decelerated to a very low M3, the total pressure loss will be small but then losses of dissociation will be too high, and in the opposite case, if the flow is slowed to a relative high M3, dissociation losses will be small but the total pressure loss will be very elevated. So, balancing each term it can be found an optimum combustor inlet Mach number.



Specific impulse for frozen nozzle flow (combustor pressure fixed at 1 atm)



Engine issues for hypersonic airbreathing propulsion systems

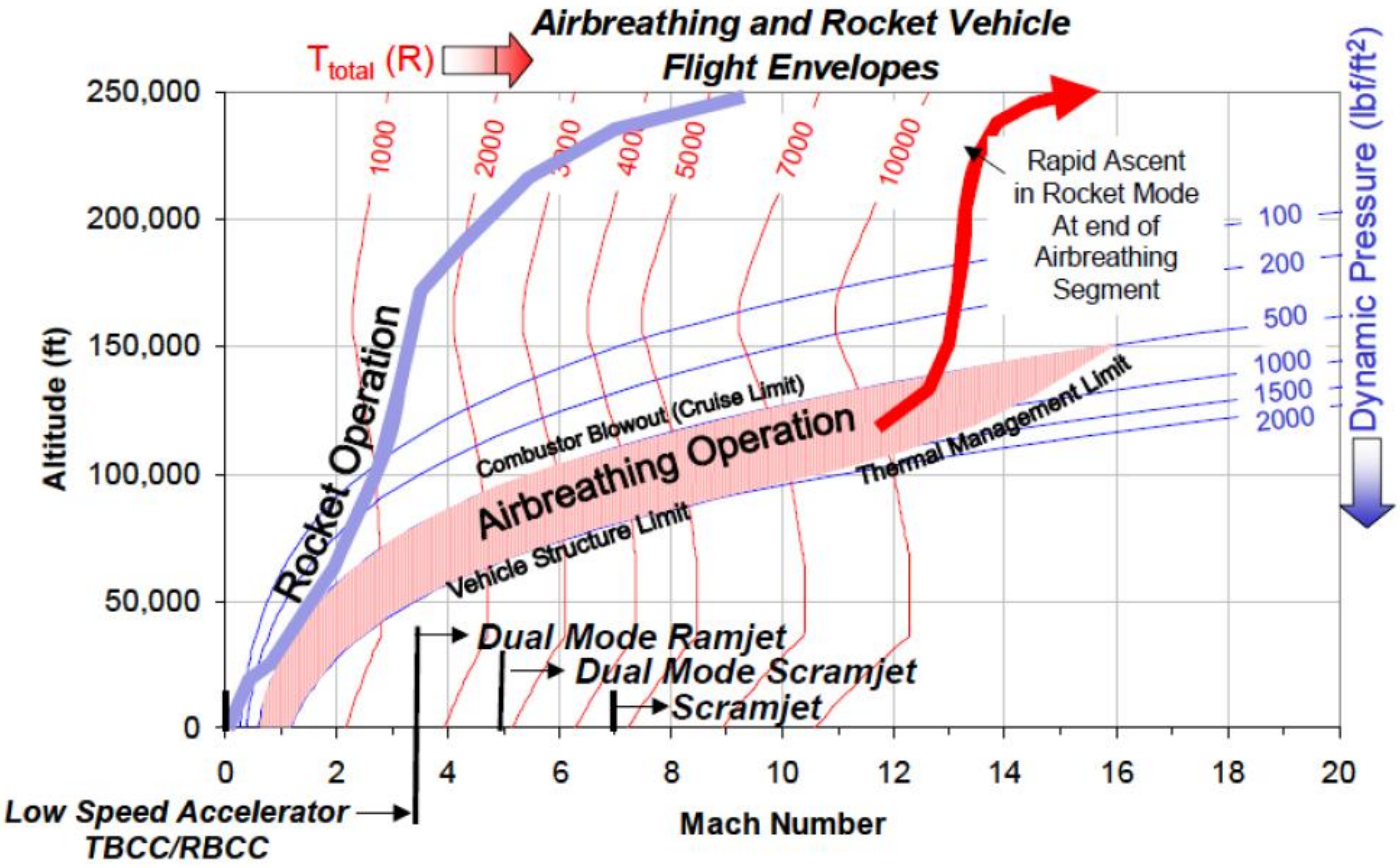
Scramjet applications

- The “holy grail” of hypersonic airbreathing propulsion is its use as part of a system for reaching low earth orbit, either for satellites insertion or manned operations.
- At the current stage of scramjet technology development, single-stage-to-orbit systems are not viable, however many multi-stage options have been studied.
- Turbojets are a propulsion candidate for the initial phase of a flight to LEO, but are currently limited to Mach 3+. Scramjets are a desirable candidate for the middle phase, particularly if the upper limit of their operation can be stretched to Mach 10+. However, scramjet use in conjunction with turbojets is problematic, as the take-over Mach number of a scramjet designed to operate at Mach 10 and above is likely to be Mach 5-6, in the absence of significant variable geometry.

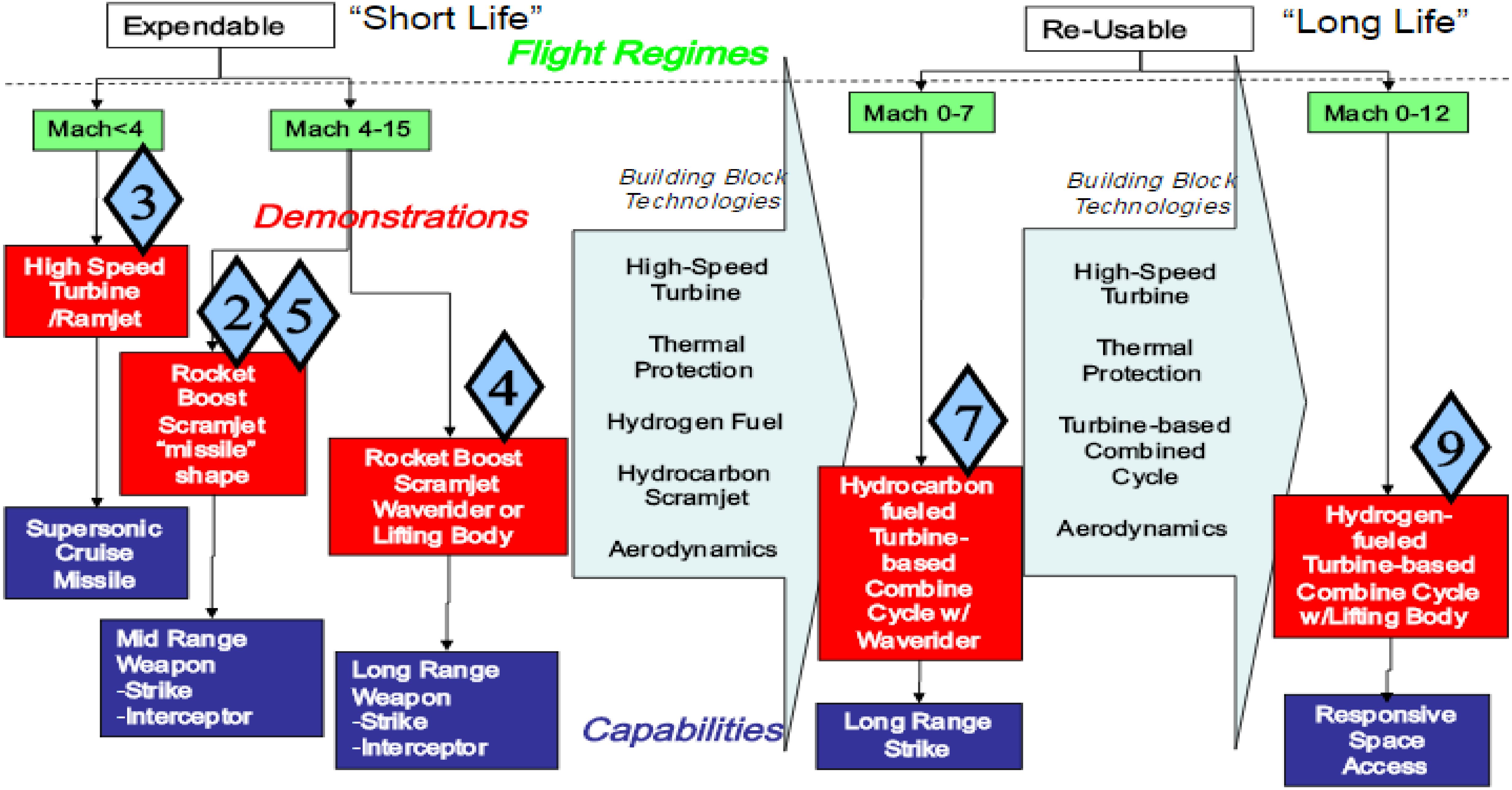
- An efficient liquid fuelled rocket is a desirable candidate for the last phase to LEO. An example of a possible system for acceleration to low earth orbit is described here, based on a rocket scramjet- rocket three-stage vehicle design to lift approximately 100 kg to LEO.
- The first stage is a solid rocket, chosen for its simplicity of operation, despite its low efficiency.
- The second stage is a scramjet powered hypersonic vehicle with an initial mass of 3000 kg that can operate between Mach 6 and 12.
- This is followed by a liquid fuelled rocket third stage to boost the payload to LEO.

Scramjet applications

- ISRO currently uses rocket launch vehicles like the PSLV to deliver satellites into orbit.
- PSLVs are expendable, meaning that can only be used once, and are designed to carry both fuel and oxidizer with for launch.
- Scramjets use ambient air to burn fuel, thus saving the need to carry an oxidizer – thus increasing the payload of a craft.
- ISRO claims that using Avatar for satellite launches will cut down launch costs by half. Since there are no rotating parts in a scramjet, the chances of failure are also measurably reduced.



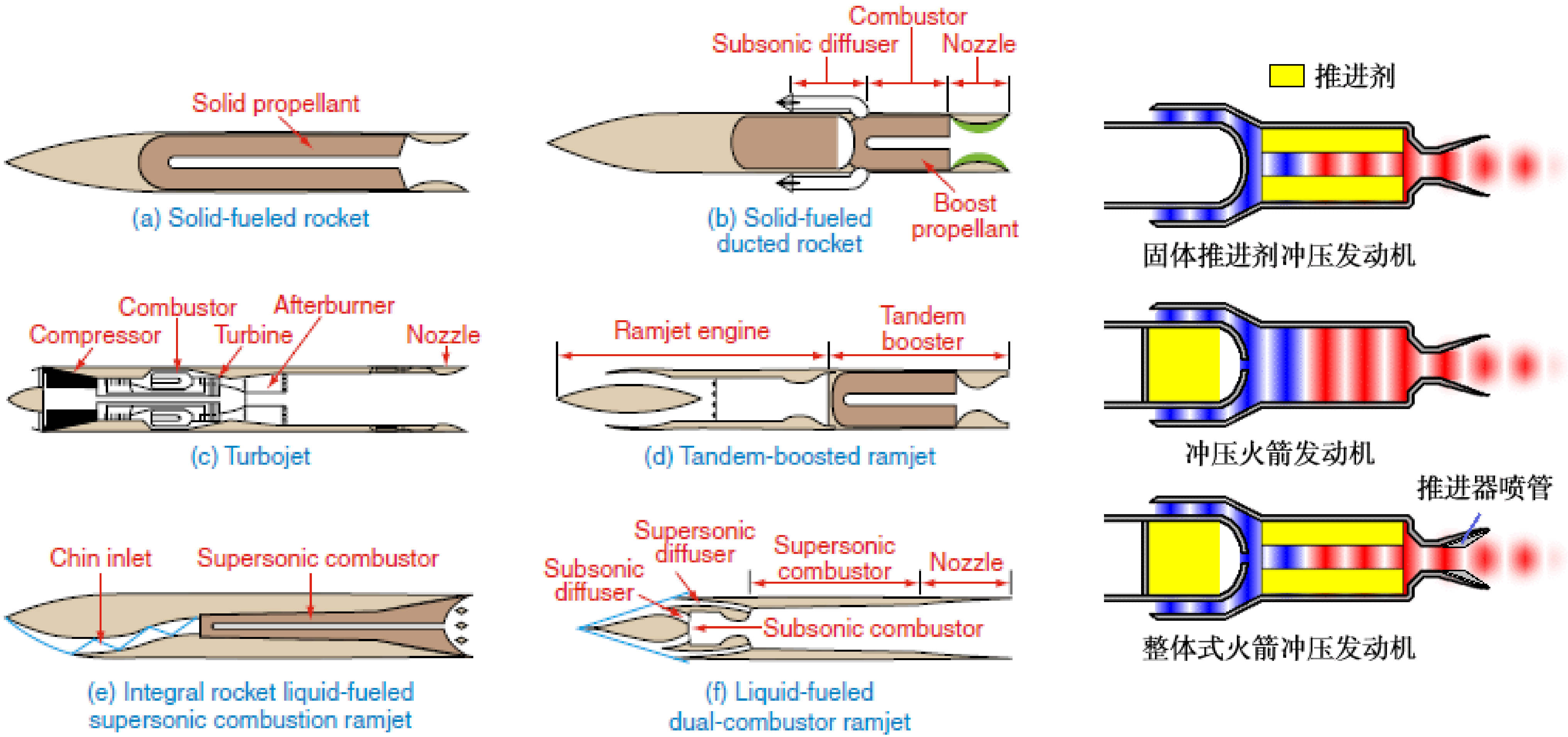
Total Temperature Rise with Increasing Mach Number in Trans-atmospheric Flight


Expendable
"Short Life"
Flight Regimes
Re-Usable
"Long Life"
Mach<4
Mach 4-15
Mach 0-7
Mach 0-12
3
Demonstrations
Building Block Technologies
High Speed Turbine /Ramjet
2
5
Rocket Boost Scramjet "missile" shape
4
Rocket Boost Scramjet Waverider or Lifting Body
High-Speed Turbine
Thermal Protection
Hydrogen Fuel
Hydrocarbon Scramjet
Aerodynamics
7
Hydrocarbon fueled Turbine-based Combine Cycle w/ Waverider
Building Block Technologies
High-Speed Turbine
Thermal Protection
Turbine-based Combined Cycle
Aerodynamics
9
Hydrogen-fueled Turbine-based Combine Cycle w/Lifting Body
Supersonic Cruise Missile
Mid Range Weapon -Strike -Interceptor
Long Range Weapon -Strike -Interceptor
Capabilities
Long Range Strike
Responsive Space Access

Combined cycle engines

- Hypersonic propulsion systems can be categorized as liquid- and solid-fueled rockets, turbojets, ramjets, ducted rockets, scramjets, and the dual-combustion ramjet (DCR). All existing hypersonic systems use either liquid or solid rockets as their propulsion system.
- Ramjets and scramjets can operate efficiently at supersonic and hypersonic speeds, but there tend to be limitations to the range of Mach numbers over which they can operate.
- For instance, the need to have sufficient compression in the inlet ordinarily requires that the ramjet engine operate supersonically. The inefficiencies of slowing the flow down to subsonic speeds makes the ramjet difficult to use for speeds exceeding Mach 5.

- Scramjets can be used above approximately Mach 5 but below that there is in general insufficient energy in the captured airstream to enable efficient combustion in the supersonic combustor.
- Both the ramjet and scramjet must be coupled with some additional form of propulsion (for missiles, this is chiefly a rocket) to accelerate the vehicle to its “take-over” Mach number.
- To overcome these limitations, combined cycle engines have been developed.



Engine cycles for hypersonic vehicles

- A subsequent version of the X-43B has had to be fully multi-use, and its main objective was to test the hybrid power plant type RBCC (rocket-based-combined-cycle), TBCC (turbine-based-combined cycle) and AAR (air-augmented rocket).
- The engine type is TBCC turbines placed separately over a high-speed ram and part has its own entrance channel and nozzle. Most comprehensive and most complex the engine AAR.
- He has all kinds of drive concentrated in a single flow channel. During take-off of the rocket engine flows into the combustion chamber, additional fuel in excess of oxygen, thereby increasing the tension of almost 50%. After reaching a speed of Mach 2, the rocket motor shuts down and re-activation occurs only in the absence of atmospheric oxygen in orbit. Wiring engine AAR and the principle of its operation is shown next slide. Rbcc engine is its principle similar to AAR, but can operate in a wider speed range by changing the geometry of the engine. The advantages of such a drive train are clear. After minimal adjustments feasible in flight, can work in rocketry, Ramjet, maximum or jet mode.

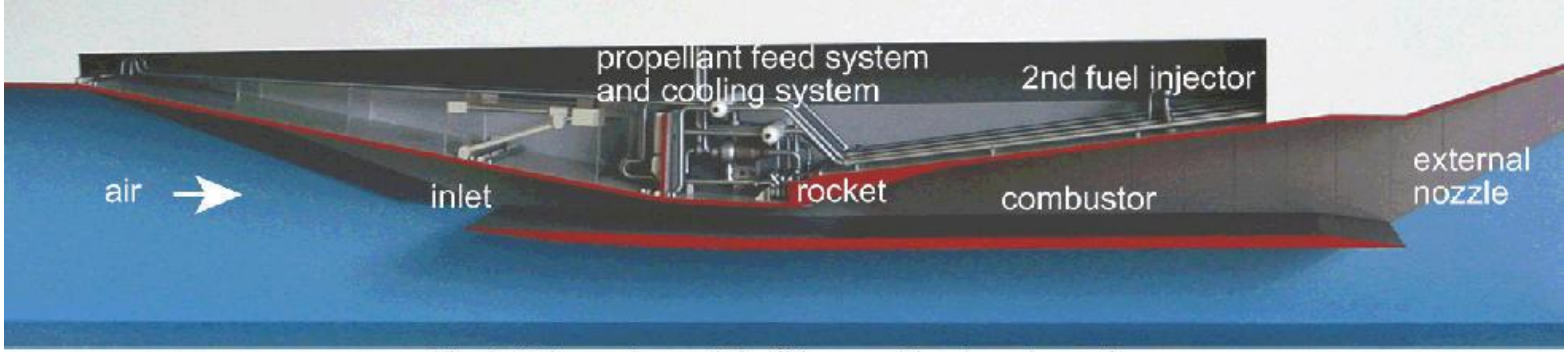


Fig. Schematic model of the combined cycle engine.

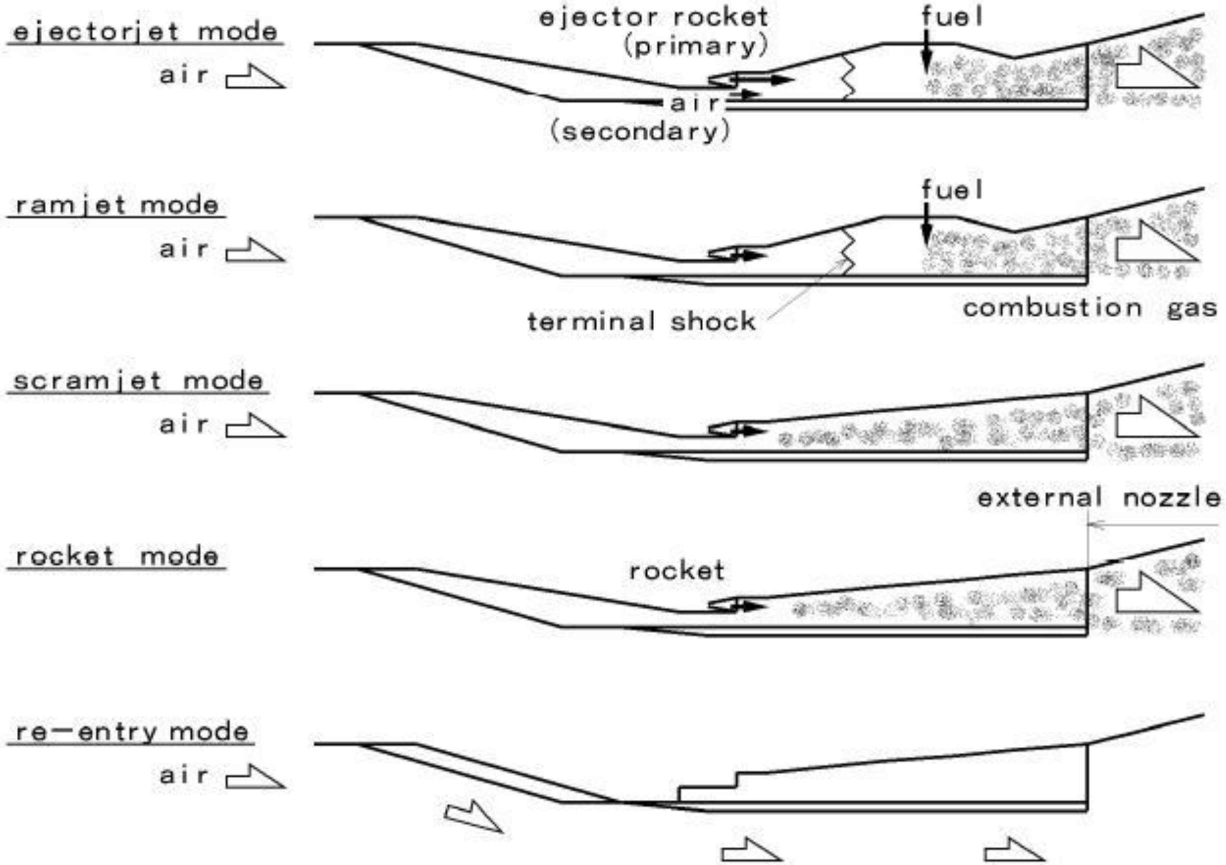
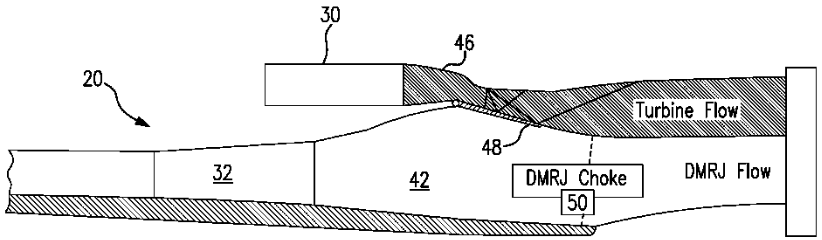
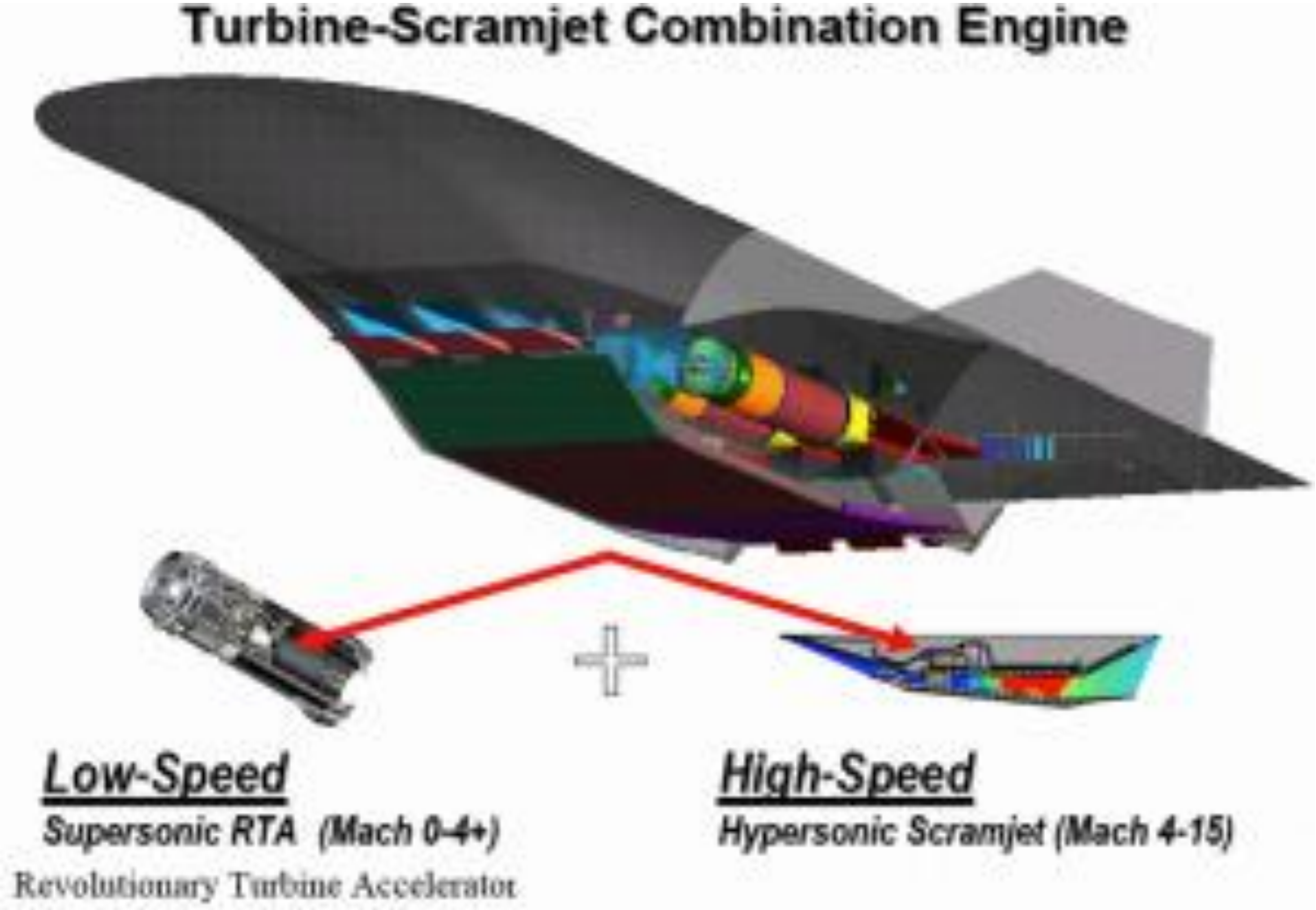


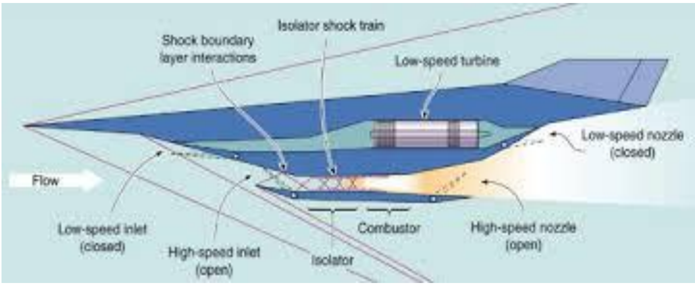
Fig. Schematic diagram of operating conditions of the combined cycle engine

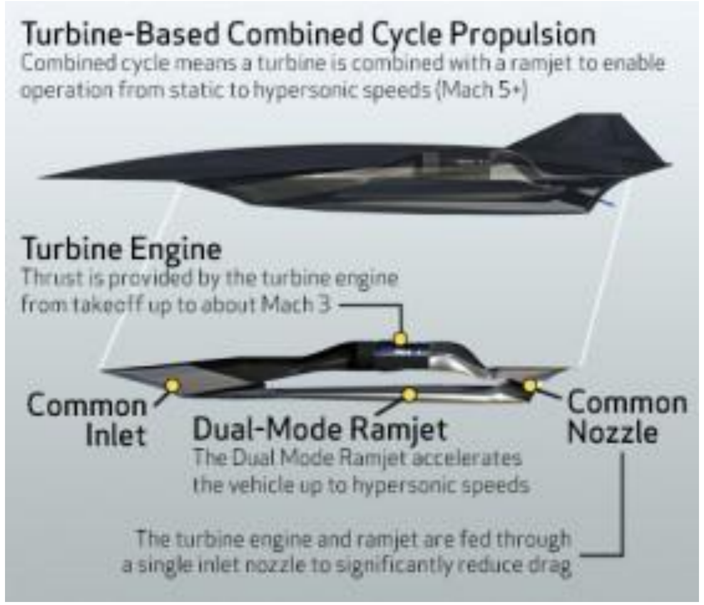
Combined cycle engines- Turbo-Ra



- A simple combined cycle is a turbojet (or turbofan)/ramjet in which a secondary flow bypasses the core turbojet and participates to produce thrust in an afterburner.
- As the Mach number increases, typically beyond M = 3, the afterburner transitions to operation as a ramjet while the turbojet maximum cycle temperature is reduced to maintain an acceptable load on the rotating machinery while maintaining the airflow path open to contribute to thrust generation in the afterburner.
- The main issue in this configuration is, evidently, the matching of the core flow with the bypass flow to avoid reversed flow on any of the sides. Additional operational difficulties derive from the broad bypass ratio range during acceleration and deceleration and the thermal management of the moving parts during high enthalpy flight.

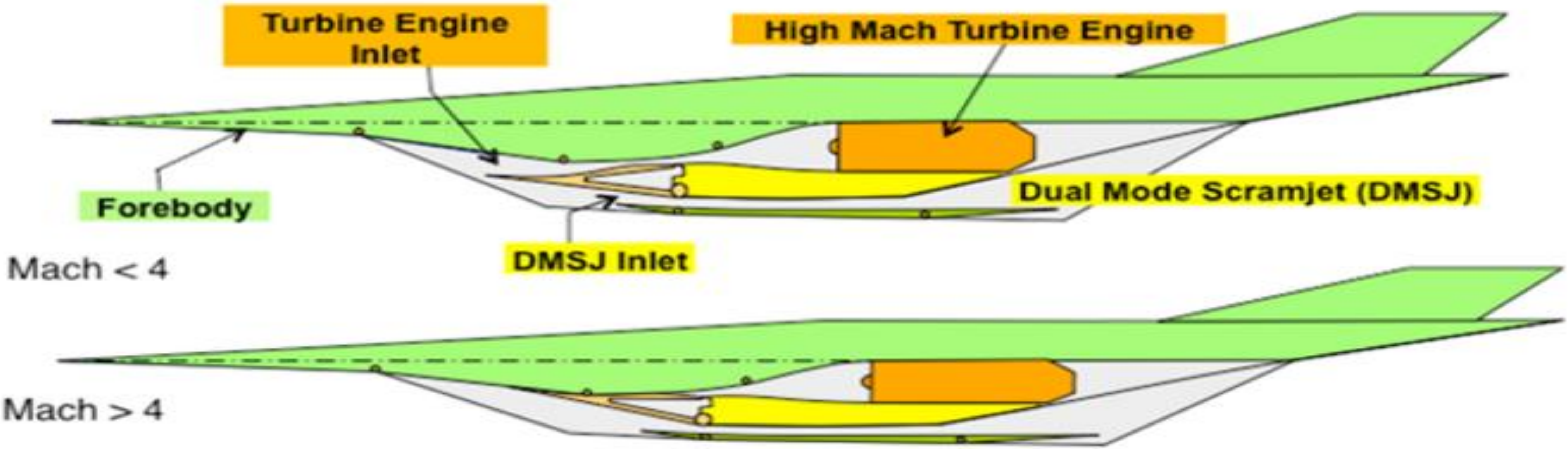

Turbine-Scramjet Combination Engine
Low-Speed
Supersonic RTA (Mach 0-4+)
Revolutionary Turbine Accelerator
High-Speed
Hypersonic Scramjet (Mach 4-15)


Isolator shock train
Shock boundary layer interactions
Low-speed turbine
Low-speed nozzle (closed)
Flow
Low-speed inlet (closed)
High-speed inlet (open)
Isolator
Combustor
High-speed nozzle (open)


Turbine-Based Combined Cycle Propulsion
Combined cycle means a turbine is combined with a ramjet to enable operation from static to hypersonic speeds (Mach 5+)
Turbine Engine
Thrust is provided by the turbine engine from takeoff up to about Mach 3
Common Inlet
Dual-Mode Ramjet
The Dual Mode Ramjet accelerates the vehicle up to hypersonic speeds
Common Nozzle
The turbine engine and ramjet are fed through a single inlet nozzle to significantly reduce drag

Turbine-Based Combined Cycle Concept

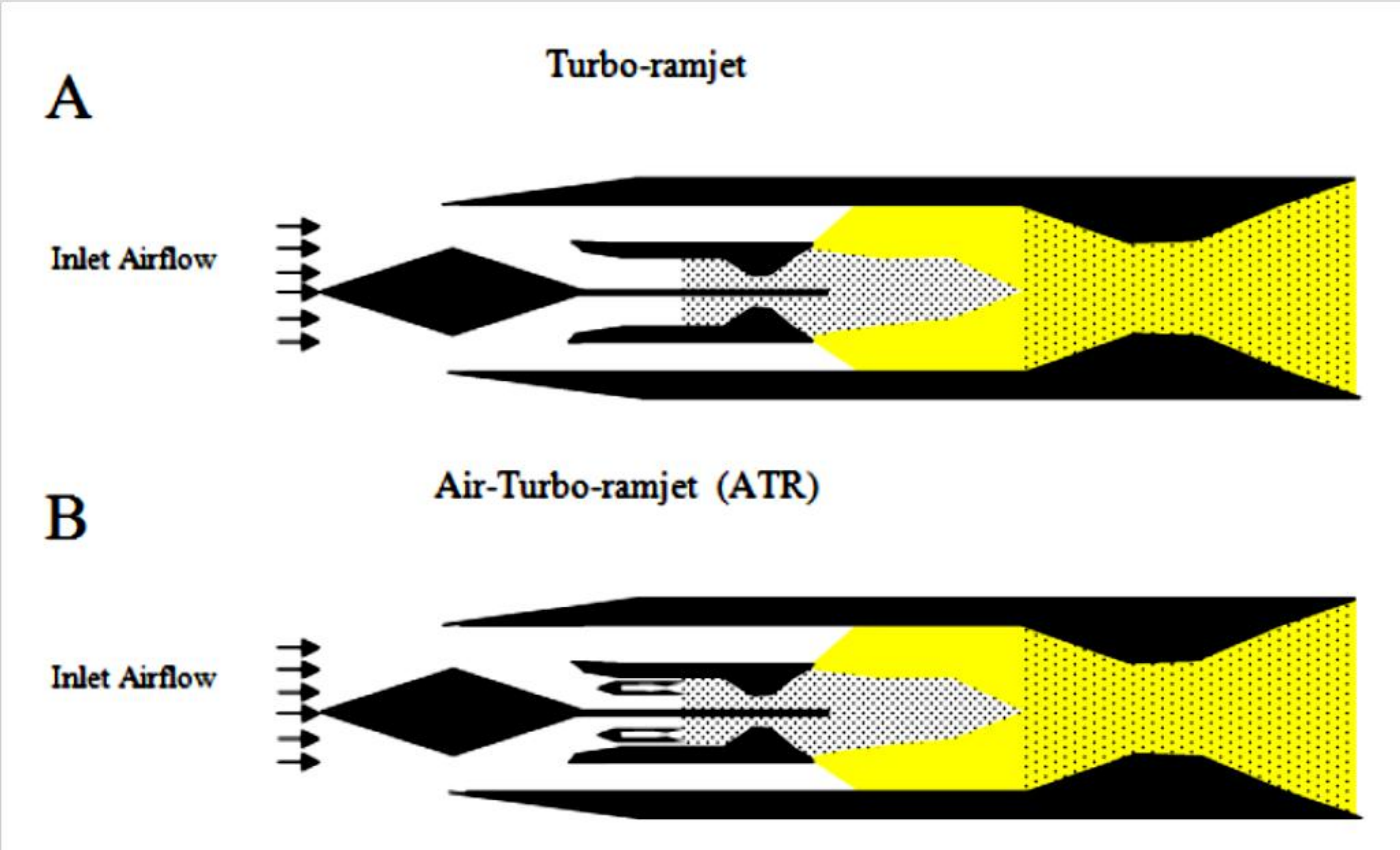
- The Turbine-Based Combined Cycle (TBCC) concept involves an over/ under configuration with a split in the propulsion flowpath.
  - From take-off to Mach 4, a turbine engine provides propulsion.
  - Above Mach 4, a dual-mode supersonic combustion ramjet /scramjet (DMSJ) engine provides propulsion.



Air Turbo-Rocket (ATR)

- The Air-Turbo-Ramjet is an upgrade on the simplest form of the Turbine-Based-Combined-Cycle Engine, the turbo-ramjet. The turbo-ramjet uses the best performance features of the two systems in a hybrid configuration.
- In Fig. a it can be seen that the variable geometry (translating center body in this case) inlet of a turbo-ramjet bi-furcates the inlet mass flow allowing some of the flow to go into the turbojet, and the rest into the ramjet.
- The turbojet is located upstream of the ramjet (or more precisely, the turbojet is contained inside of the ramjet). At low speeds the center body is extends further upstream of the turbine, allowing more airflow to go into the turbojet than into the ramjet.
- At higher supersonic speeds the center body is retracted, eventually closing of the turbojet inlet and allowing all of the flow to go into the ramjet, bypassing the entire turbojet.

- The Air-Turbo-Ramjet (Figure. b) provides enhanced performance over the turbo-ramjet by avoiding the high temperatures and pressures associated with conventional turbo-compression.
- The ATR powers the turbine in the system by the use of high temperature, fuel rich gas generator.
- As a result the temperature remains unchanged since it is independent of inlet Mach number and the resulting aerodynamic heating associated with it.
- The Air-Turbo-Rocket/Ramjet is a modification which has been made to integrate rockets in the ATR congfiuration. The ATR-ramjet uses rockets as the main source of gas generation.



Turbo-ramjet (a) and Air Turbo Ramjet (b) Combined Cycle Engines

Ejector Ramjet

- Ejector ramjet engines hold promise for several high-speed air vehicle applications, from sounding rockets to "Aurora" type aircraft to space launch booster aircraft.
- Another method of enhancing the performance of a rocket in the low speed regime is thrust augmentation of the ejector rocket (see Fig. a).
- The ejector operates by producing static thrust in a duct open to the inlet air, and as a result, creates flow into and over the ejector.
- This results in a pressure rise inside of the duct which can slow down the flow for stable combustion.
- Downstream of the ejector, the rocket exhaust and the air stream is mixed. Further down from the mixing location, fuel is added to the mixed flow, and the flow is heated, creating more thrust and thus enhancing the overall performance.

- The flow must be choked, and thermal choking can be employed to avoid the necessity of variable geometry in the nozzle and the complex problems associated with it.
- The ejector rocket is a promising candidate for integration in an SSTO combined-cycle engine because it can be easily integrated say in a scramjet in such configurations as the ejector-scramjet (SERJ).

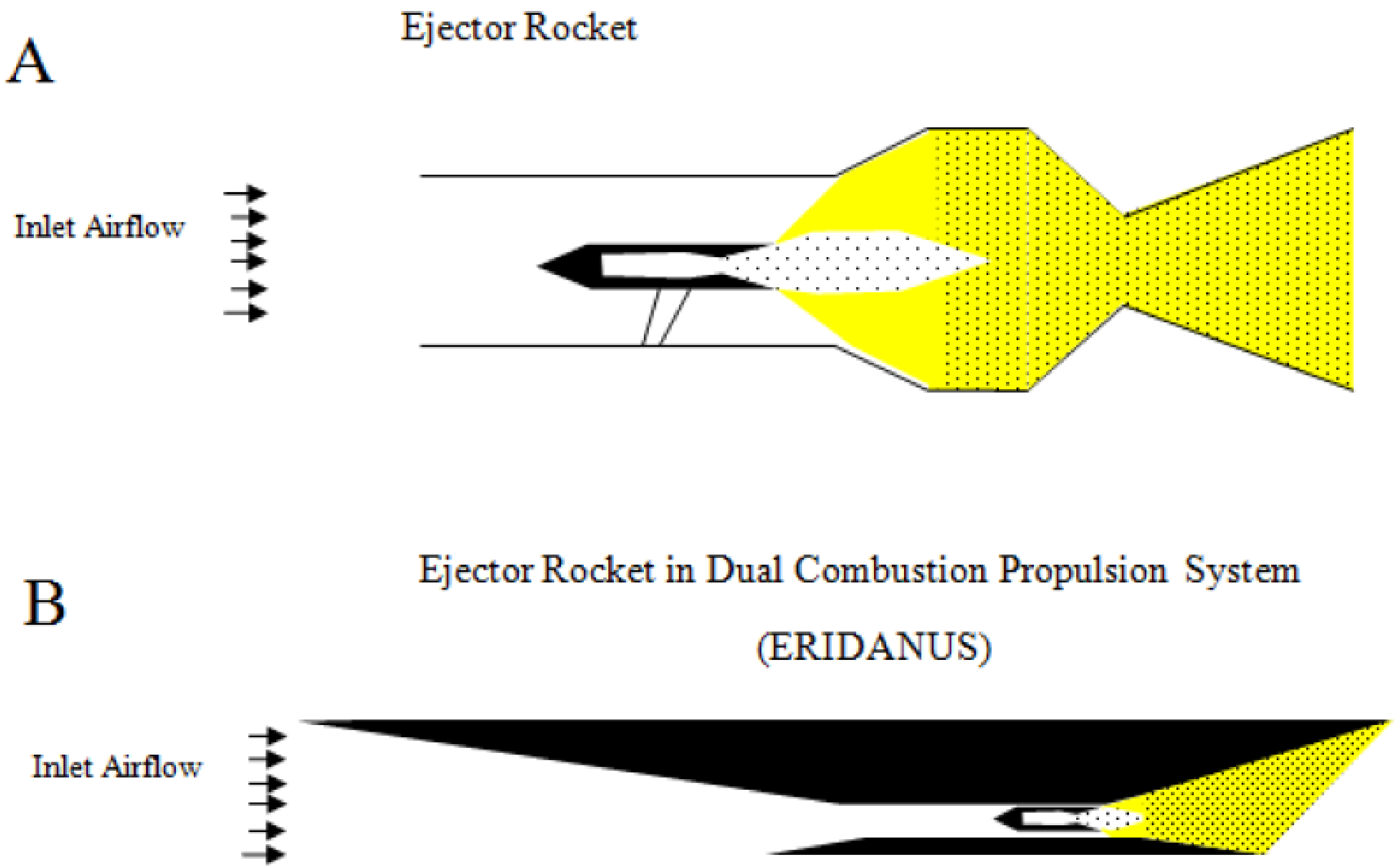


Figure : Ejector Rocket (a) and ERIDANUS (b) Combined Cycle Engines

Liquid-air collection engine(LACE)

- Another candidate for low-speed combined-cycle propulsion are engines generally classified as cryo-cooled compression cycle engines. Technically these engines are the so called `air-breather' rockets.
- The LACE engine has an advantage over the turbo-accelerator based combined cycle engines in that it does not require the use of the entire turbo-accelerator system, although they may employ a turbo compressor.
- The heat exchanger reduces the temperature (and pressure) of the incoming airflow to burner entry conditions and resulting stable combustion.
- Thus it avoids the heavy machinery which is required to cool and compress flows in turbo-accelerators. The most attractive feature of these devices is that they attempt to enhance the performance (increasing specific impulse) by making use of oxygen from the surrounding air during flight through the atmosphere, and thus reducing the required weight of the entire vehicle.

- From Fig. a it can be seen that in the LACE cycle first liquefies the free-stream air in a precooler, and then pumps the liquefied air in a turbopump, preparing it for the high required pressures for heat addition in the combustion chamber.
- LACE engines also have physical limitations which make them not the best candidates for SSTO propulsion in their purest form. LACE engines require extremely high fuel consumption relative to other airbreathing engines, with specific impulse limitations in the 800 to 1000 second range.
- Modifications of the LACE to enhance its performance include combining the LACE cycle with the scramjet cycle (ScramLACE engine) Fig. b, or more modern forms of the LACE engine includeing HOTOL's R545, or SKYLON's Synergetic Air-breathing Rocket Engine. Each of these requires complex cooling systems, and although are promising ideas, are extremely dependant on complicated heat exchanger research, design, and development.

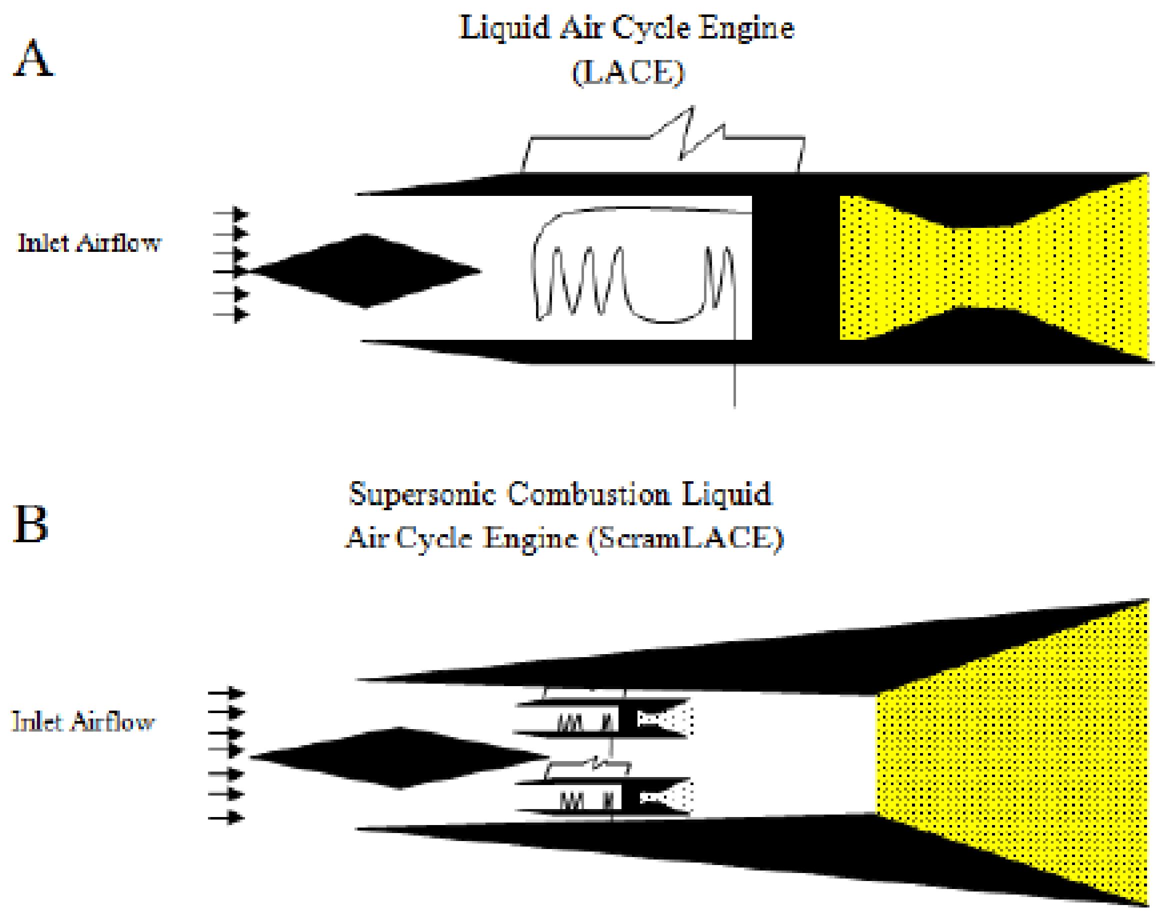
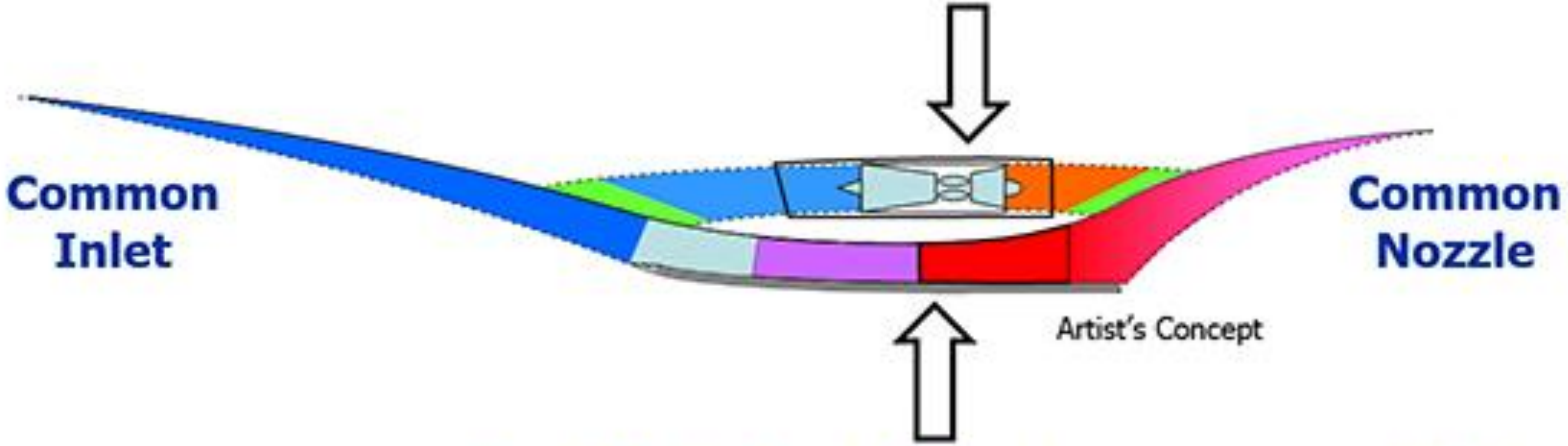


Figure : LACE (a) and ScramLACE (b) Combined Cycle Engines

Advanced Full Range Engine (AFRE)

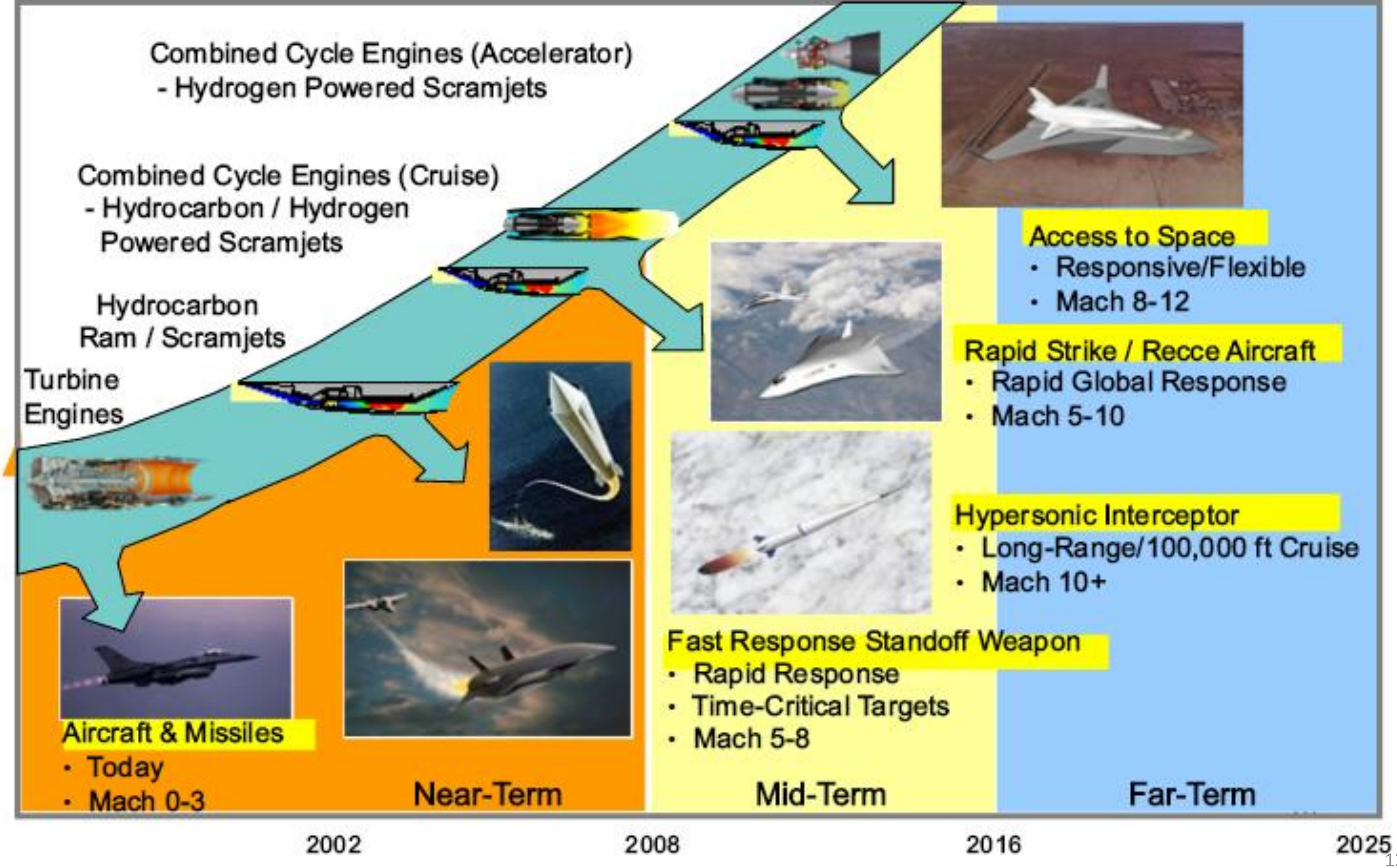
Off-the-Shelf (OTS) Turbine Engine

for Low-Speed Operations



Dual-Mode Ramjet Engine

for High-Speed Operations


Combined Cycle Engines (Accelerator)
- Hydrogen Powered Scramjets
Combined Cycle Engines (Cruise)
- Hydrocarbon / Hydrogen Powered Scramjets
Hydrocarbon Ram / Scramjets
Turbine Engines
Access to Space
• Responsive/Flexible
• Mach 8-12
Rapid Strike / Recce Aircraft
• Rapid Global Response
• Mach 5-10
Hypersonic Interceptor
• Long-Range/100,000 ft Cruise
• Mach 10+
Fast Response Standoff Weapon
• Rapid Response
• Time-Critical Targets
• Mach 5-8
Aircraft & Missiles
• Today
• Mach 0-3
Near-Term
Mid-Term
Far-Term
2002
2008
2016
2025

Hypersonic missiles

Speed is the new stealth

10

RGM-84 Harpoon Block II

(United States)

Chapter 2

AIR BREATHING ENINES FOR HYPERSONIC TRANSPORT PLANES AND

MILITARYMISSILES- SUPERSONIC COMBUSTION- THE SCRAM-JET ENGINE

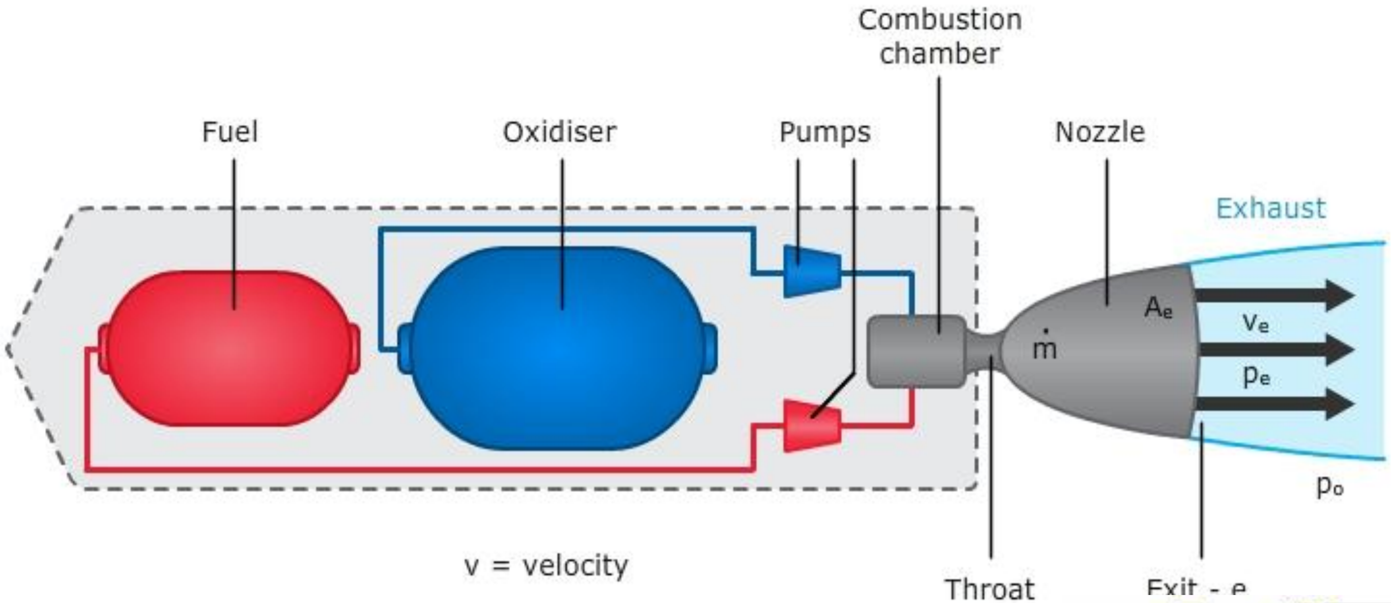
Combined cycle engines- turbo-ramjet, air turbo-rocket (ATR), ejector ramjet- Liquid-air collection engine(LACE)- need, principle, construction, operation, performance, applications to hypersonic transport plane and missile propulsion

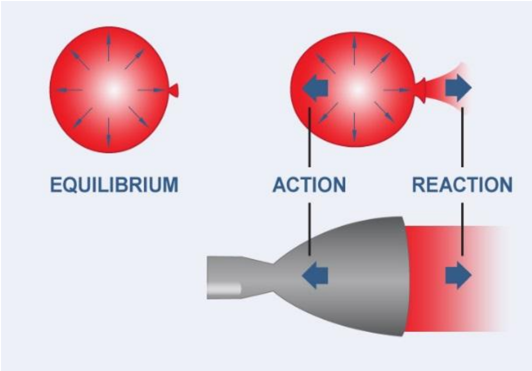
Chapter 3
CHEMICAL ROCKET ENGINES

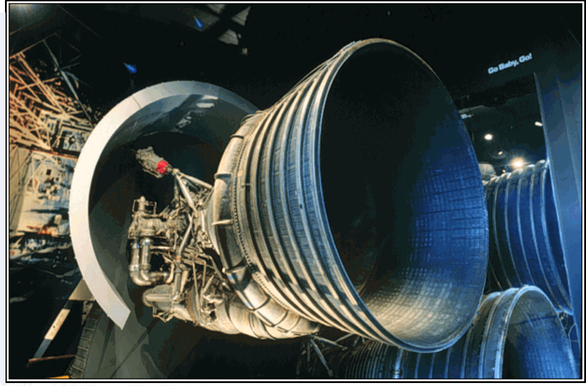
Rocket propulsion- history, principles, types, applications. The rocket equation. Vehicle velocity, jet exit velocity, mass ratio. Effect of atmosphere. Engine parameters, propellants. Chemical rockets- the thrust chamber- processes- combustion, expansion- propellants. Thermo-chemical analysis of combustion, equilibrium energy balance, mass balance, combustion efficiency. Equilibrium composition, recombination.

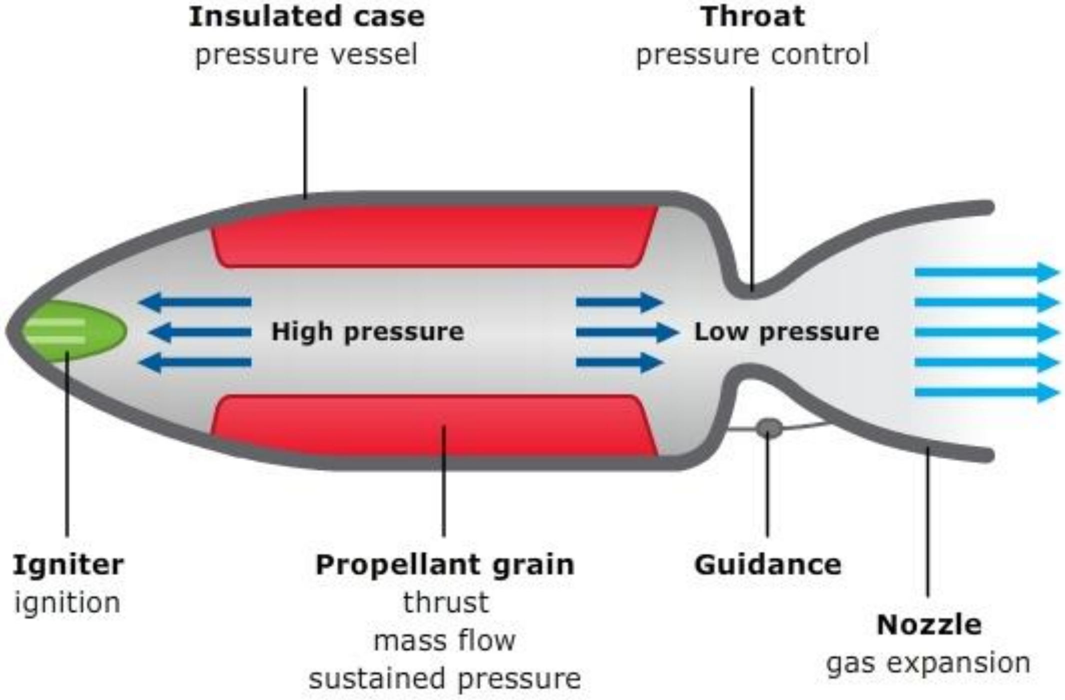
Nozzle expansion, performance, design parameters, analysis- non-equilibrium expansion- frozen equilibrium, shifting equilibrium. One dimensional, two dimensional flows, presence of liquid drops and solid particles- two phase flow, losses, efficiency.

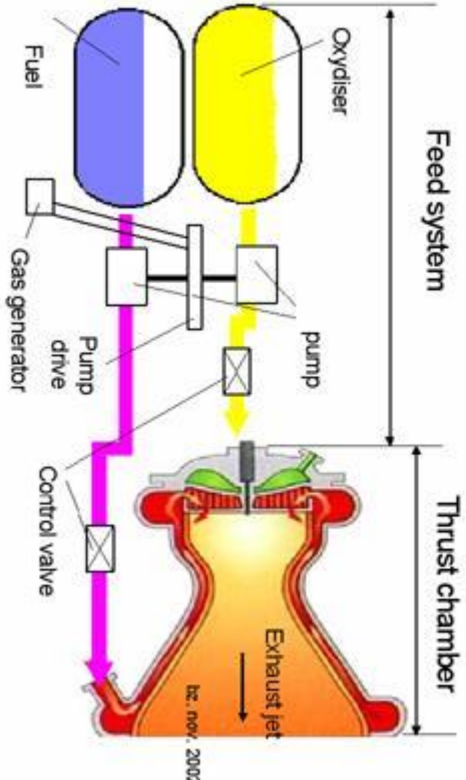
Performance measures of chemical rocket engines- thrust coefficient, specific impulse; engine parameters thrust chamber pressure, temperature, characteristic velocity, exhaust velocity, effective velocity. Computing rocket engine performance- theoretical, delivered performance, performance at standard operating conditions, guaranteed minimum performance.

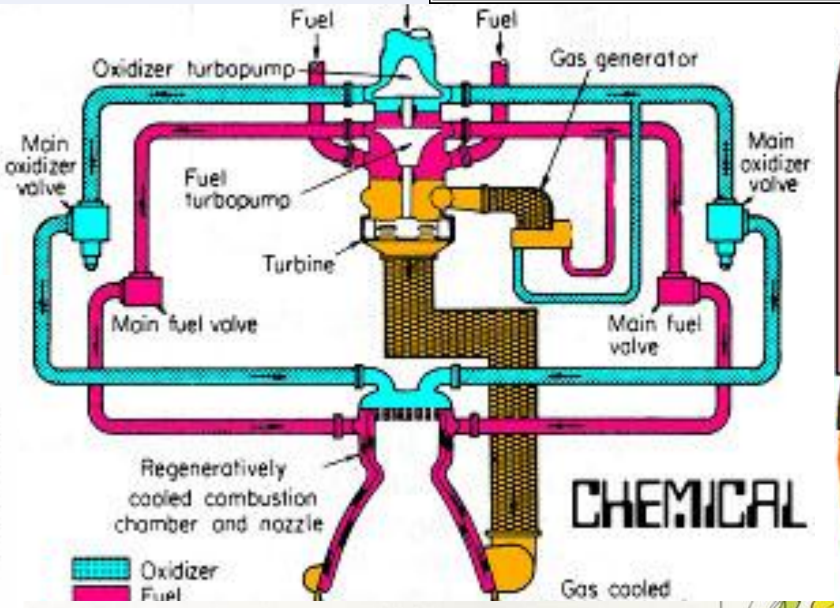

Combustion chamber
Fuel
Oxidiser
Pumps
Nozzle
Exhaust
A_e
$\dot{m}$
v_e
p_e
p_o
v = velocity
Throat
Exit - e


EQUILIBRIUM
ACTION
REACTION

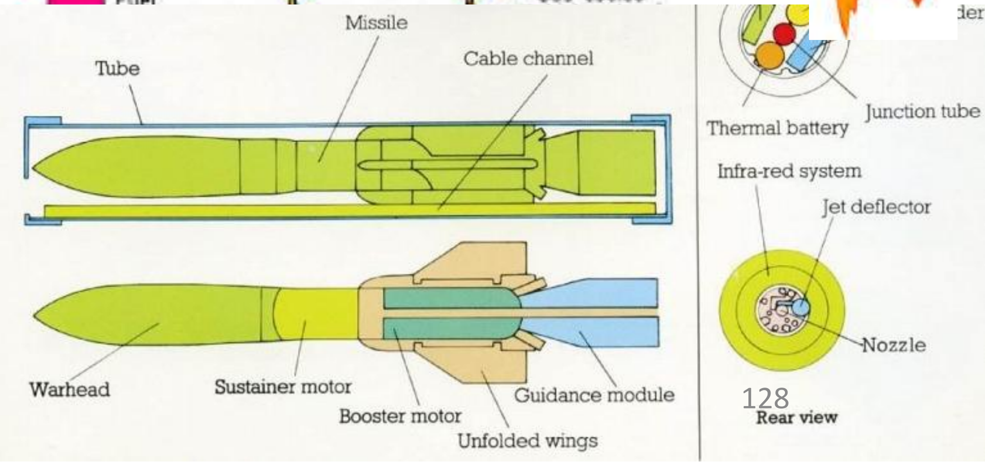



Insulated case
pressure vessel
Throat
pressure control
High pressure
Low pressure
Igniter
ignition
Propellant grain
thrust
mass flow
sustained pressure
Guidance
Nozzle
gas expansion


Fuel
Oxydiser
Feed system
Gas generator
Pump drive
pump
Control valve
Thrust chamber
Exhaust jet


Fuel
Fuel
Oxidizer turbopump
Gas generator
Main oxidizer valve
Main oxidizer valve
Fuel turbopump
Turbine
Main fuel valve
Main fuel valve
Regeneratively cooled combustion chamber and nozzle
CHEMICAL
Oxidizer
Fuel
Gas cooled


REACTION
ACTION
©2003 HowStuffWorks


Missile
Tube
Cable channel
Thermal battery
Junction tube
Infra-red system
Jet deflector
Nozzle
Rear view
Warhead
Sustainer motor
Booster motor
Guidance module
Unfolded wings

History

- The mighty space rockets of today are the result of more than 2,000 years of invention, experimentation, and discovery. First by observation and inspiration and then by methodical research, the foundations for modern rocketry were laid.
- Building upon the experience of two millennia, new rockets will expand human presence in space back to the Moon, to Mars and the asteroids, and beyond. These new rockets will be versatile.
- They will support Earth orbital missions, such as the International Space Station, and off-world missions millions of kilometers from home.
- Already, travel to the stars is possible. Robot spacecraft are on their way into interstellar space as you read this. Someday, they will be followed by human explorers.

What Is Propulsion?

- Initiating or changing the motion of a body
  - *Translational* (linear, moving faster or slower)
  - *Rotational* (turning about an axis)
- Space propulsion
  - Rocket launches
  - Controlling satellite motion
  - Maneuvering spacecraft
- Jet propulsion
  - Using the momentum of ejected mass (propellant) to create a *reaction force*, inducing motion

At one time it was believed that rockets could not work in a vacuum -- they needed air to push against!!

Jet Propulsion Classifications

- Air-Breathing Systems
  - Also called *duct propulsion*.
  - Vehicle carries own fuel; surrounding air (an *oxidizer*) is used for combustion and thrust generation
  - Gas turbine engines on aircraft…

- Rocket Propulsion
  - Vehicle carries own fuel and oxidizer, or other expelled propellant to generate thrust:
  - Can operate outside of the Earth’s atmosphere
  - Launch vehicles, upper stages, Earth orbiting satellites and interplanetary spacecraft … or

Space Propulsion Application

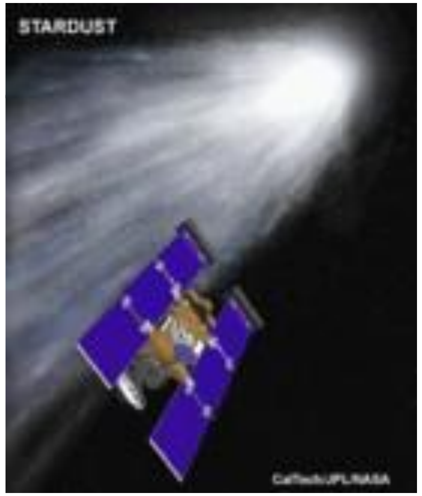
- Launch Vehicles
- Ballistic Missiles
- Earth Orbiting Satellites
- Upper Stages
- Interplanetary Spacecraft
- Manne

en.wikipedia.org

www.army-technology.

www.britannica .com

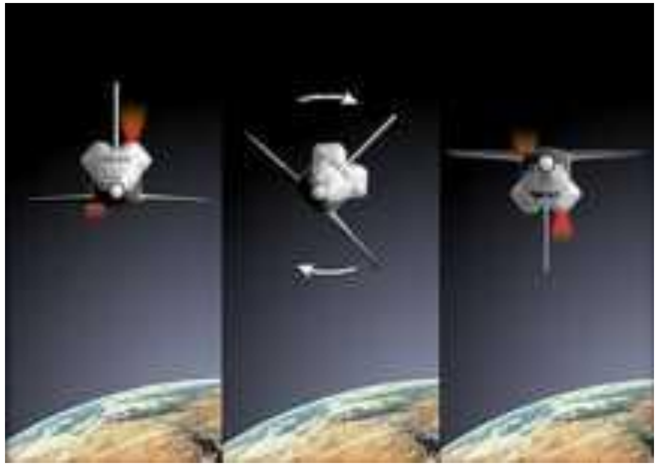
blog.wired.com



www.psrd.hawaii.edu

Space Propulsion Functio

- Primary propulsion
  - Launch and ascent
  - Maneuvering
    - Orbit transfer, station keeping, trajectory correction
- Auxiliary propulsion
  - Attitude control
  - Reaction control
  - Momentum management



www.nasm.si.edu

A Brief History of Rocketry

- China (300 B.C.)
    - Earliest recorded use of rockets
    - Black powder
- Russia (early 1900's)
    - Konstantin Tsiolkovsky
    - Orbital mechanics, rocket equation
- United States (1920's)
    - Robert Goddard
    - First liquid fueled rocket (1926)
- Germany (1940's)
    - Wernher von Braun
    - V-2
    - Hermann Oberth

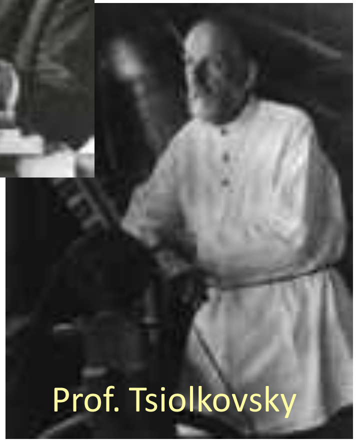
onenew.wordpress.com

Wan-Hu who tried to launch himself to the moon by attaching 47 black powder rockets to a large wicker chair!

Dr. Goddard

Dr. von Braun

www.britannica.com



Prof. Tsiolkovsky

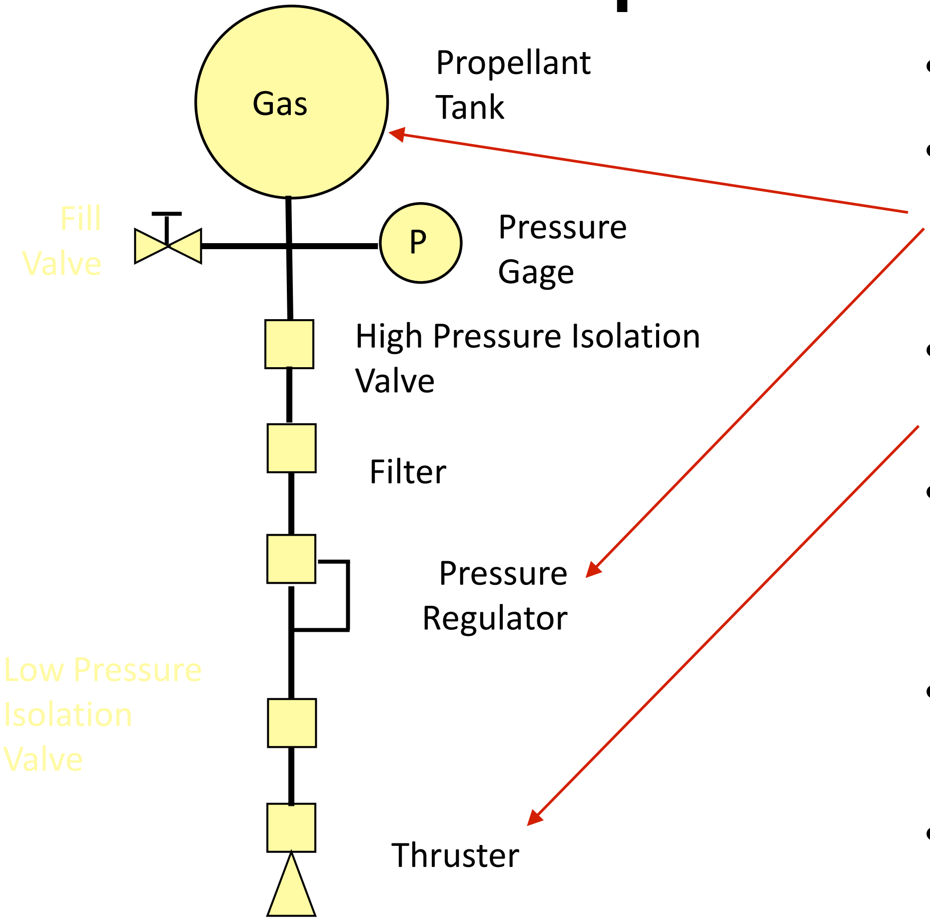
www.geocities.com

Space Propulsion System Classifications

- Stored Gas
- Chemical
  - Solid
  - Liquid
    - Pressure Fed
    - Pump Fed
    - Bipropellant
    - Monopropellant
  - Hybrid
- Electric
  - Electrothermal
  - Electrostatic
  - Electrodynamic
- Advanced
  - Nuclear
  - Solar thermal
  - Laser
  - Antimatter

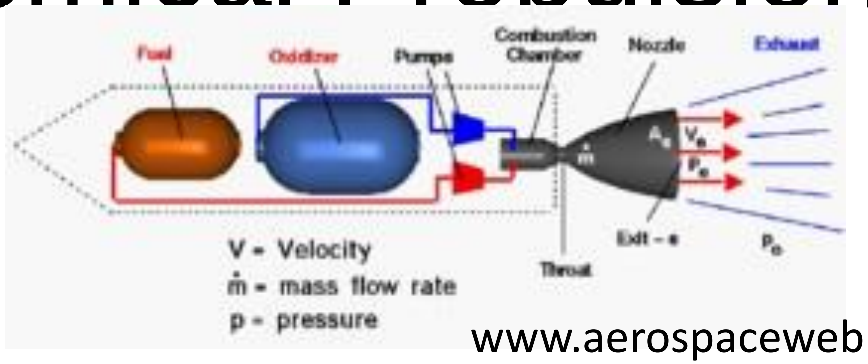
Space propulsion systems are classified by the type of energy source used.

Stored Gas Propulsion

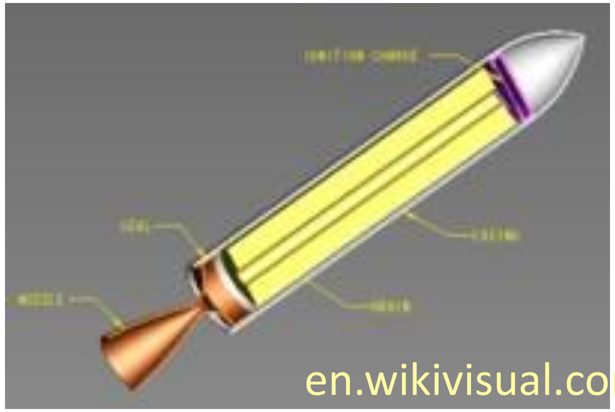


- Primary or auxiliary propulsion.
- High pressure gas (*propellant*) is fed to low pressure nozzles through pressure regulator.
- Release of gas through nozzles (*thrusters*) generates thrust.
- Currently used for momentum management of the Spitzer Space telescope.
- Propellants include nitrogen, helium, nitrous oxide, butane.
- Very simple in concept.

Chemical Propulsion Classifications



www.aerospaceweb.org

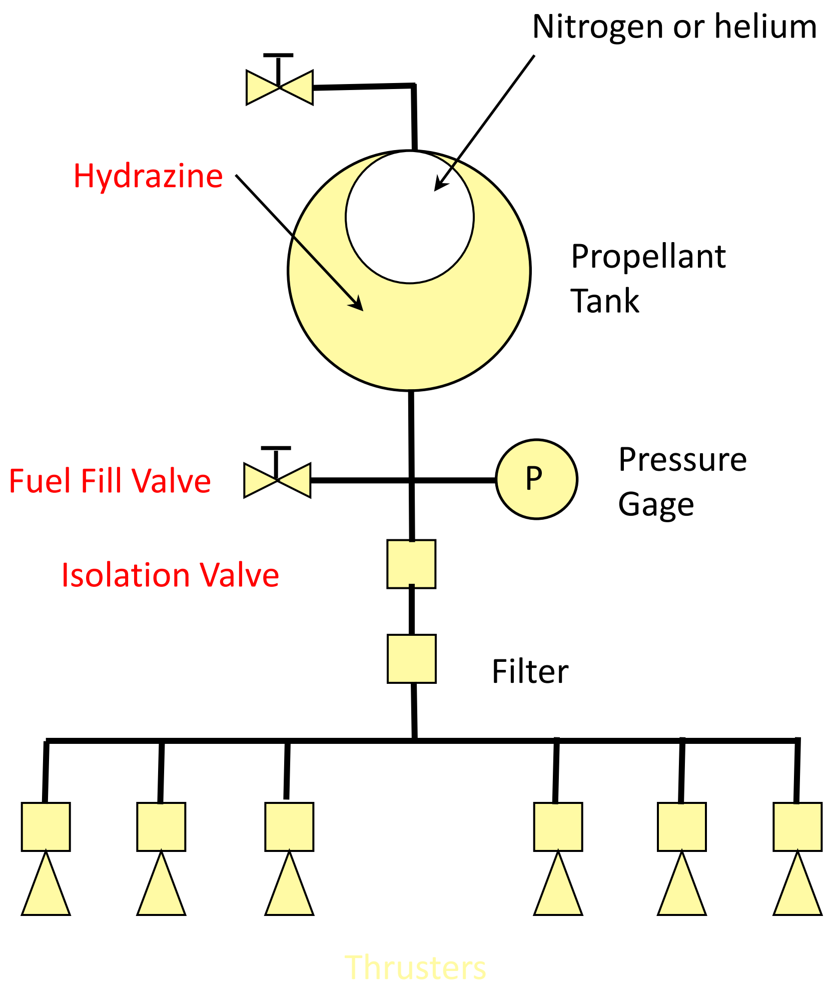


en.wikivisual.com

news.bbc.co.uk

- Liquid Propellant
  - Pump Fed
    - Launch vehicles, large upper stages
  - Pressure Fed
    - Smaller upper stages, spacecraft
  - Monopropellant
    - Fuel only
  - Bipropellant
    - Fuel & oxidizer
- Solid Propellant
  - Launch vehicles, Space Shuttle, spacecraft
  - Fuel/ox in solid binder
- Hybrid
  - Solid fuel/liquid ox
  - Sounding rockets

Monopropellant Systems



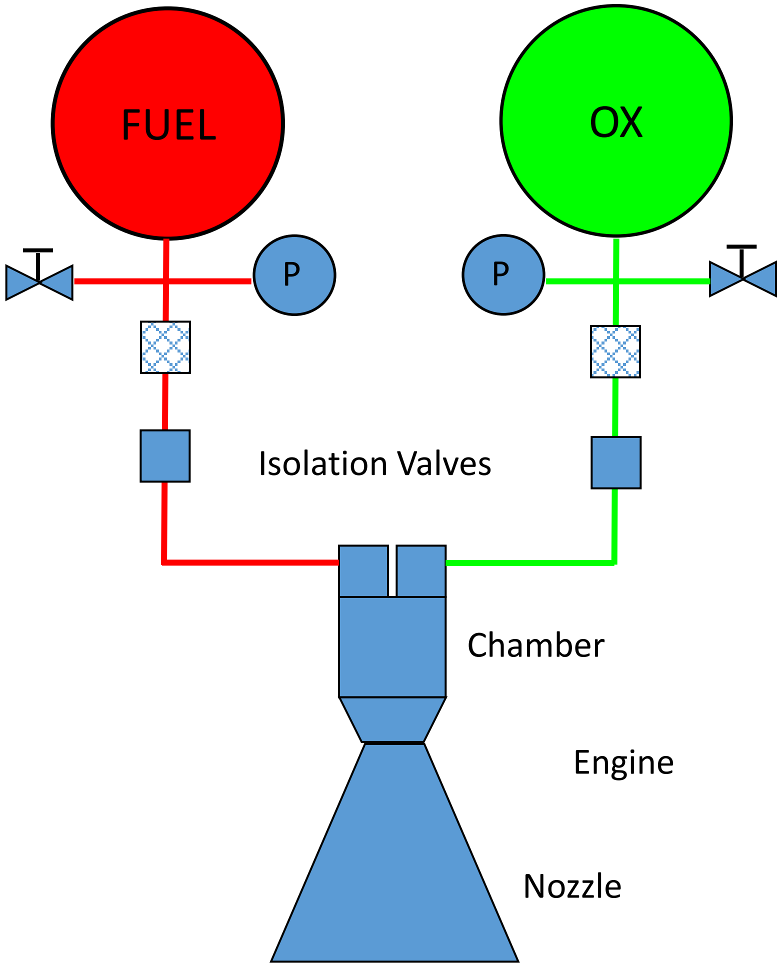
- Hydrazine fuel is most common monopropellant.
  - N_2H_4
- Decomposed in thruster using catalyst to produce hot gas for thrust.
- Older systems used hydrogen peroxide before the development of hydrazine catalysts.
- Typically operate in *blowdown mode* (pressurant and fuel in common tank).

Monopropellant Systems

MONOPROPELLANT THRUSTERS		
ENGINE MONARC-1	Nominal Thrust: 0.2 lbf Specific Impulse: 230 sec Propellants: Hydrazine Status: Flight proven	
ENGINE MONARC-5	Nominal Thrust: 1 lbf Specific Impulse: 232 sec Propellants: Hydrazine Status: Flight proven	
ENGINE MONARC-22	Nominal Thrust: 5 lbf Specific Impulse: 232 sec Propellants: Hydrazine Status: Flight proven	
ENGINE MONARC-90	Nominal Thrust: 20 lbf Specific Impulse: 235 sec Propellants: Hydrazine Status: Flight proven	
ENGINE MONARC-445	Nominal Thrust: 100 lbf Specific Impulse: 235 sec Propellants: Hydrazine Status: Flight proven	

www.aerojet.com

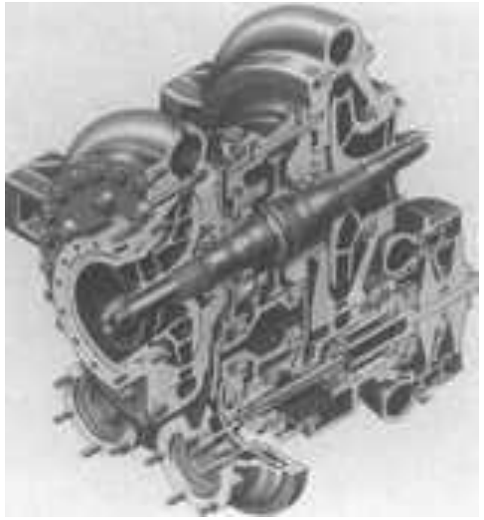
Bipropellant Systems



- A *fuel* and an *oxidizer* are fed to the engine through an *injector* and combust in the *thrust chamber*.
- *Hypergolic*: no igniter needed -- propellants react on contact in engine.
- *Cryogenic* propellants include LOX (-423 ºF) and LH2 (-297 ºF).
  - Igniter required
- *Storable* propellants include kerosene (RP-1), hydrazine, nitrogen tetroxide (N2O4), monomethylhydrazine (MMH)

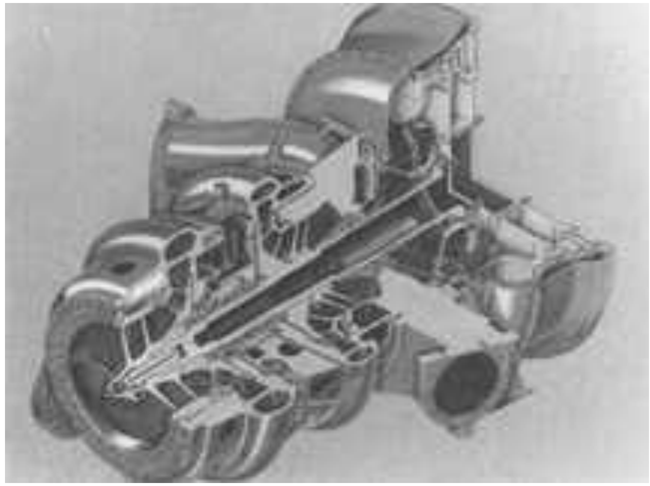
Liquid Propellant Systems

- Pump fed systems
  - Propellant delivered to engine using *turbopump*
    - Gas turbine drives centrifugal or axial flow pumps
  - Large, high thrust, long burn systems: launch vehicles, space shuttle
  - Different *cycles* developed.



H-1 Engine Turbopump

A 35’x15’x4.5’ (ave. depth) backyard pool holds about 18,000 gallons of water. How quickly could the F-1 pump empty it?



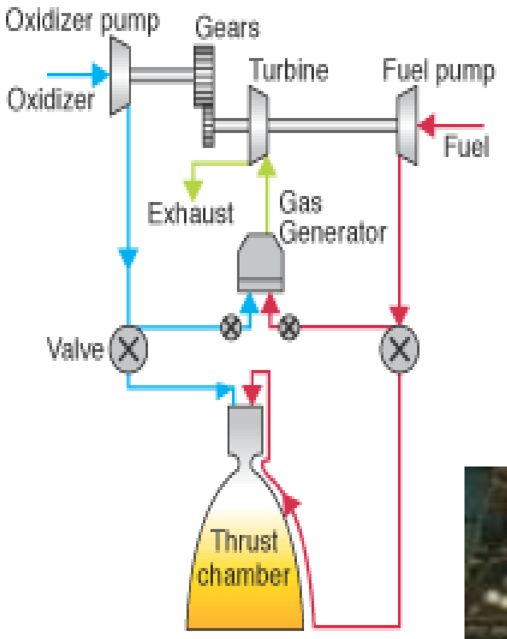
F-1 Engine Turbopump

F-1 engine turbopump:
- 55,000 bhp turbine drive
- 15,471 gpm (RP-1)
- 24,811 gpm (LOX)

Ans: In 27 seconds!

Photos history.nasa.gov

Rocket Engine Power Cycles

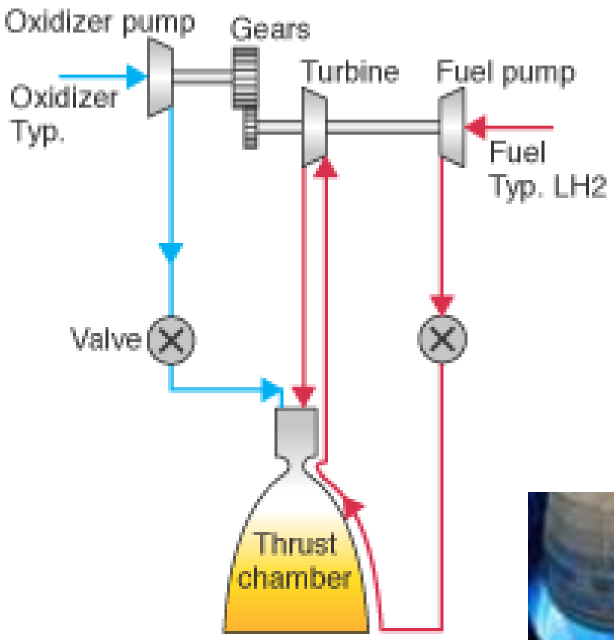


www.aero.org/publications/crosslink/winter2004/03_sidebar3.html

www.answers.com

- Gas Generator Cycle
  - Simplest
  - Most common
  - Small amount of fuel and oxidizer fed to gas generator
  - Gas generator combustion products drive turbine
  - Turbine powers fuel and oxidizer pumps
  - Turbine exhaust can be vented through pipe/nozzle, or dumped into nozzle
  - Saturn V F-1 engine used gas generator cycle

Rocket Engine Power Cycles - cont

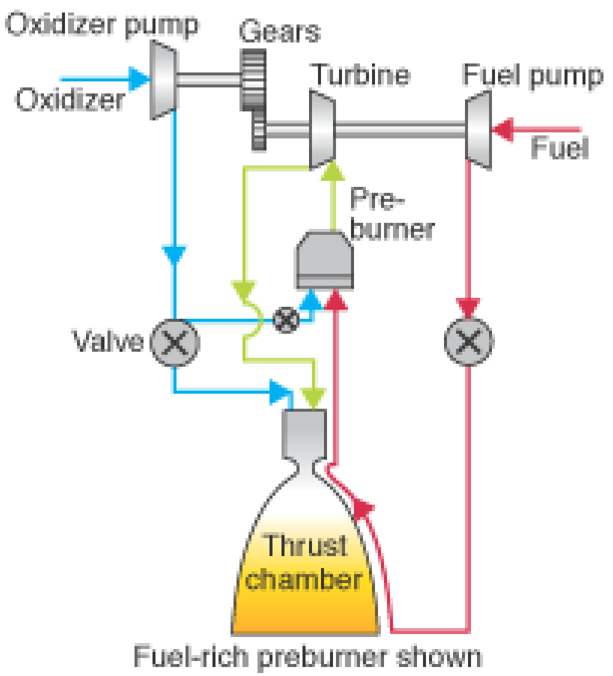


www.aero.org/publications/crosslink/winter2004/03_sidebar3.html

science.nasa.gov

- Expander
  - Fuel is heated by nozzle and thrust chamber to increase energy content
  - Sufficient energy provided to drive turbine
  - Turbine exhaust is fed to injector and burned in thrust chamber
  - Higher performance than gas generator cycle
  - Pratt-Whitney RL-10

Rocket Engine Power Cycles - cont



www.aero.org/publications/crosslink/winter2003_sidebar3.html

www.rocketrelics.com

shuttle.msfc.nasa.gov

- Staged Combustion
  - Fuel and oxidizer burned in *preburners* (fuel/ox rich)
  - Combustion products drive turbine
  - Turbine exhaust fed to injector at high pressure
  - Used for high pressure engines
  - Most complex, requires sophisticated turbomachinery
  - Not very common
    - SSME (2700 psia)

The Big Engines…

F-1 Engine
Saturn V
1.5 million lbs thrust (SL)
LOX/Kerosene

www.flickr.com

Main Engine
Space Shuttle
374,000 lbs thrust (SL)
LOX/H_2

spaceflight.nasa.gov

RD-170
1.78 million lbs thrust (SL)
LOX/Kerosene

www.aerospaceguide.net

Solid Propellant Motors

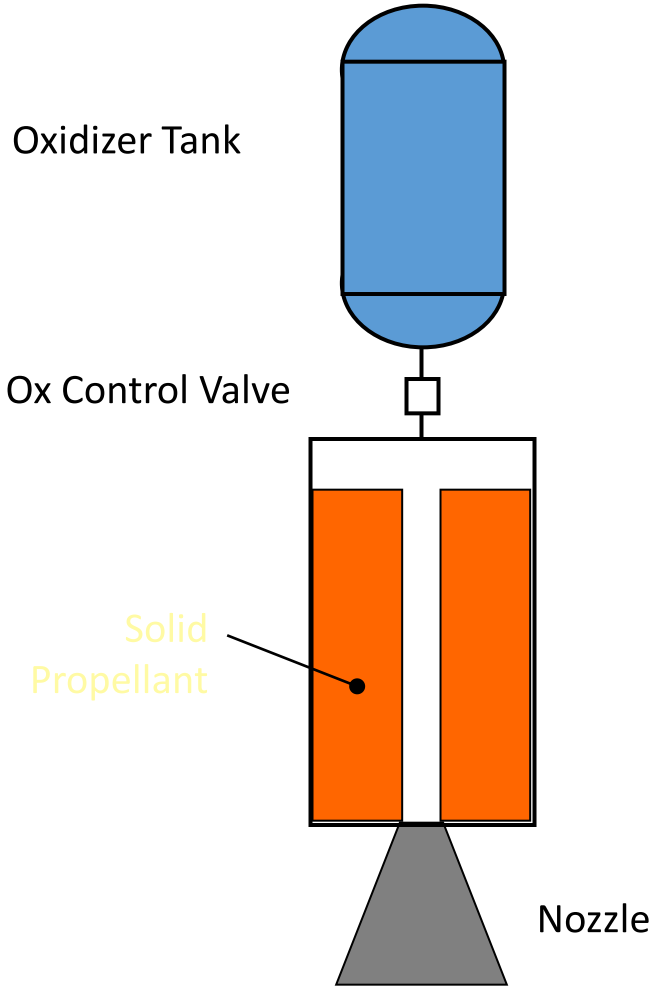
www.aerospaceweb.org

- Fuel and oxidizer are in solid binder.
- Single use -- no restart capability.
- Lower performance than liquid systems, but much simpler.
- Applications include launch vehicles, upper stages, and space vehicles.


www.propaneperformance.com


www.nationalmuseum.af.mil

Hybrid Motors



- Combination liquid-solid propellant
  - Solid fuel
  - Liquid oxidizer
- Multi-start capability
  - Terminate flow of oxidizer
- Fuels consist of rubber or plastic base, and are inert.
  - Just about anything that burns…
- Oxidizers include LO_2, hydrogen peroxide (N_2O_2) and nitrous oxide (NO_2)
- Shut-down/restart capability.

Rocket Performance Calculations

- Thrust & Specific Impulse
  - *Thrust* is the amount of force generated by the rocket.
  - *Specific impulse* is a measure or engine performance (analogous to miles per gallon)
    - Units are *seconds*

$$\boxed{I_{sp} = \frac{F}{w}}$$

$F =$ rocket thrust

$w =$ weight flowrate of propellant

- Rocket Equation

$$\boxed{\Delta V = g I_{sp} \ln \frac{m_i}{m_f}}$$

$g = 9.8 \text{ m/s}^2$

$m_i =$ mass of vehicle before burn

$m_f =$ mass of vehicle after burn

$m_p =$ mass of propellant for ΔV

$= m_i - m_f$

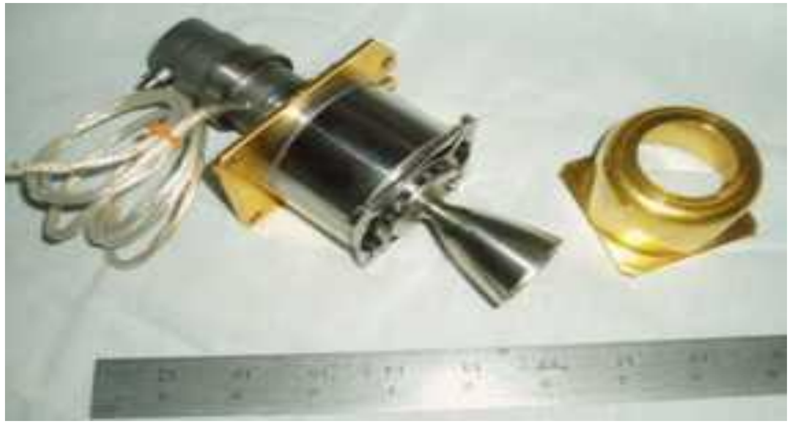
$$\boxed{m_p = m_i \left(1 - e^{\frac{-\Delta V}{g I_{sp}}} \right)}$$

Rocket equation assumes no losses (gravity effects, aerodynamic drag). Actually very accurate for short burns in Earth orbit or in deep space!

Specific Impulse Comparison

- Stored gas
- Monopropellant hydrazine
- Solid rocket motors
- Hybrid rockets
- Storable bipropellants
- LOX/LH2

- 60-179 sec
- 185-235 sec
- 280-300 sec
- 290-340 sec
- 300-330 sec
- 450 sec



Specific impulse depends on many factors: altitude, nozzle expansion ratio, mixture ratio (bipropellants), combustion temperature.

This thruster was used on the Viking Lander. It has a specific impulse of about 225 seconds.

Mission Delta-V Requirements

Mission (duration)	Delta-V (km/sec)
Earth surface to LEO	7.6
LEO to Earth Escape	3.2
LEO to Mars (0.7 yrs)	5.7
LEO to Neptune (29.9 yrs)	13.4
LEO to Neptune (5.0 yrs)	70
LEO to alpha-Centauri (50 yrs)	30,000

LEO = Low Earth orbit (approx. 274 km)

Propellant Calculation Exercise

- Determine the mass of propellant to send a 2500 kg spacecraft from LEO to Mars (0.7 yr mission).
  - Assume the 2500 kg includes the propellant on-board at the start of the burn.
  - Assume our engine has a specific impulse of 310 sec (typical of a small bipropellant engine).
  - Use the *rocket equation:*

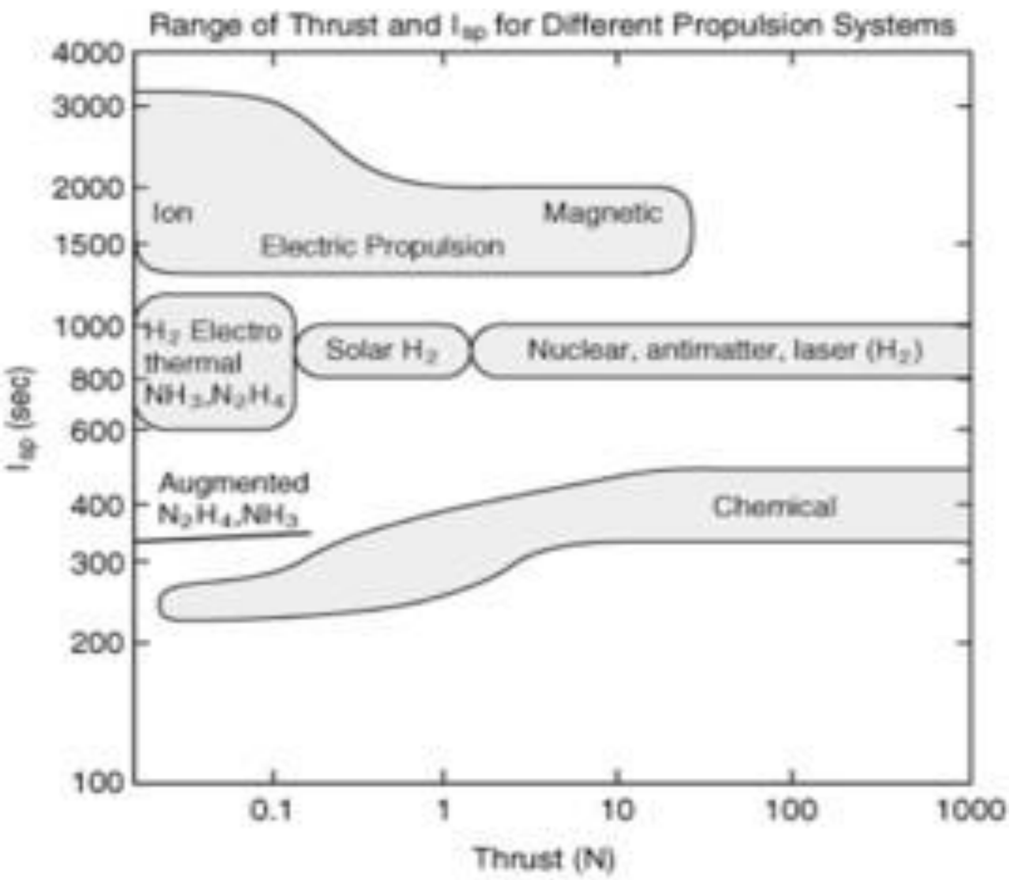
$$m_p = (2500)\left[1 - e^{\frac{-5700}{(9.8)(310)}}\right] = 2117 \text{ kg}$$

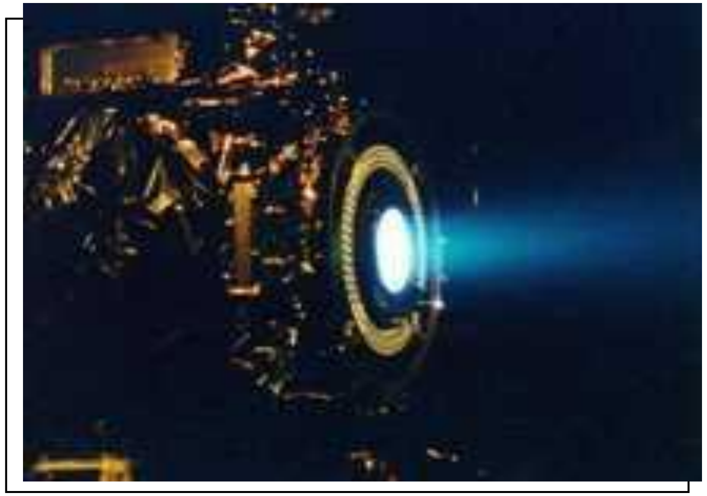
Most of our spacecraft is propellant! Only 383 kg is left for structure, etc! How could we improve this?

Electric Propulsion

- Classifications
  - Electrothermal
  - Electrostatic
  - Electromagnetic
- Characteristics
  - Very low thrust
  - Very high Isp
    - > 1000 sec
  - Requires large amounts of power (kilowatts)

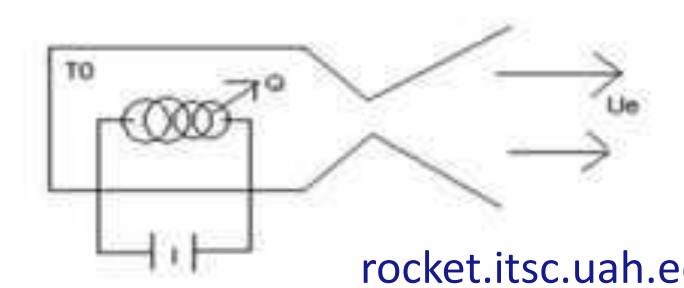
www-ssc.igpp.ucla.edu



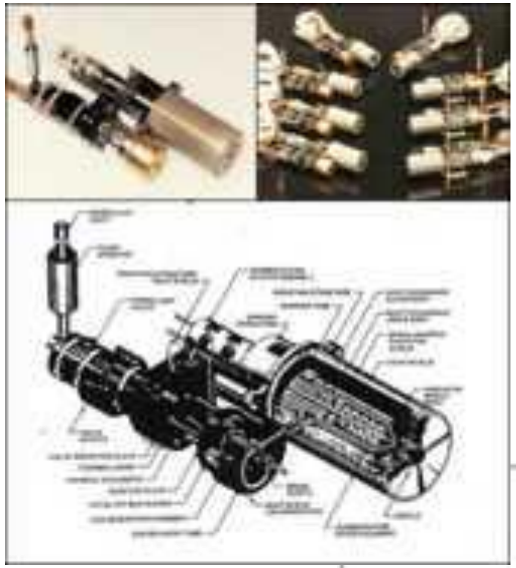


This image of a xenon ion engine, photographed through a port of the vacuum chamber where it was being tested at NASA's Jet Propulsion Laboratory, shows the faint blue glow of charged atoms being emitted from the engine. The ion propulsion engine is the first non-chemical propulsion to be used as the primary means of propelling a spacecraft.

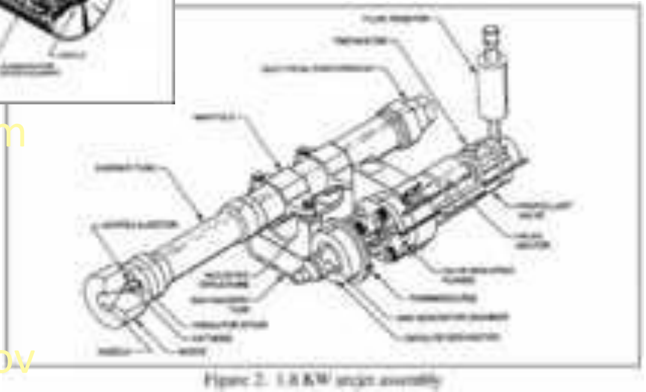
Electrothermal Propulsion



rocket.itsc.uah.edu

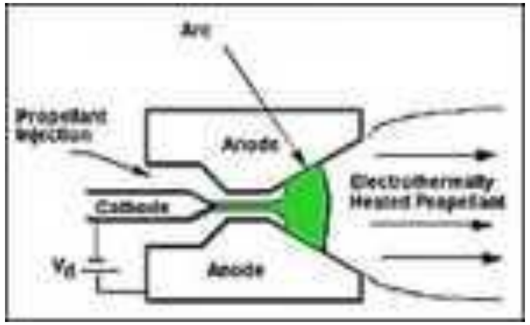


www.fathom.com



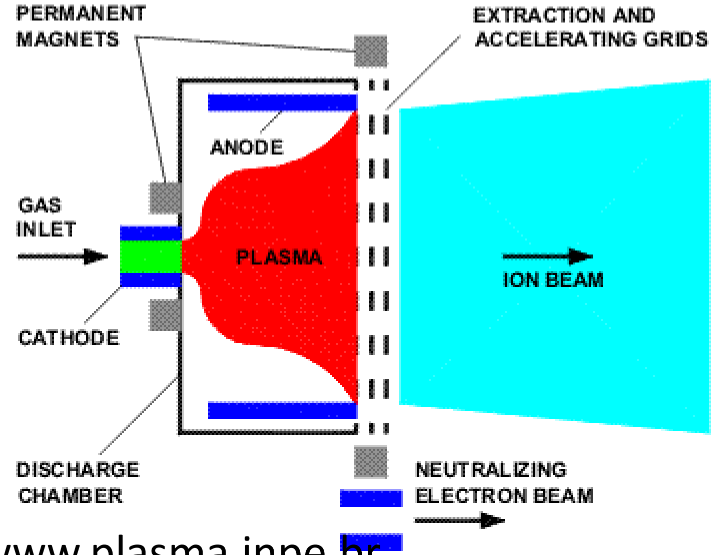
www.nasa.gov

- Electrical power is used to add energy to exhaust products
- Resistojet
  - Catalytic decomposition of hydrazine is augmented with high power electric heater
  - 800 – 5,000 W
- Arcjet
  - High voltage arc at nozzle throat adds thermal energy to exhaust
  - Various gaseous or vaporized propellants can be used.



www.waynesthisandthat.com

Electrostatic Propulsion



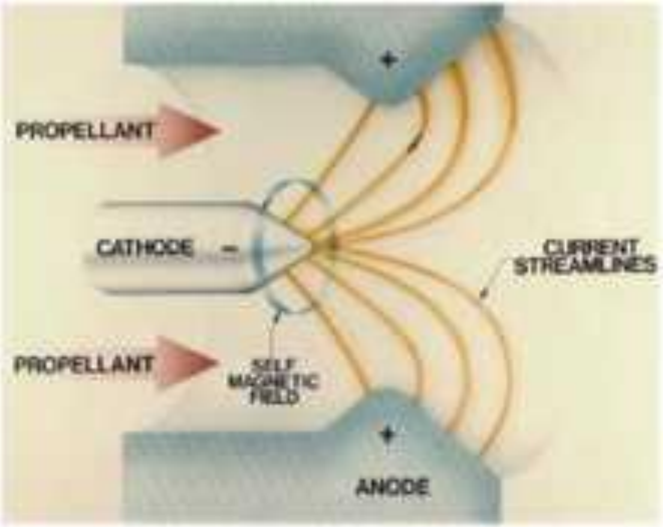
www.plasma.inpe.br

aerospace.engin.umich.edu

ESA's SMART-1 uses a xenon ion propulsion system (XIPS)

- Xenon Ion Thruster
  - Xenon propellant
  - Electrostatic forces are used to accelerate charged particles to very high velocities
  - Xenon is ionized by electron bombardment
    - *Thermionic cathode*
  - Positively charged particles accelerated by grid
  - Electrons routed to second anode and injected into beam to neutralize

Electromagnetic Propulsion

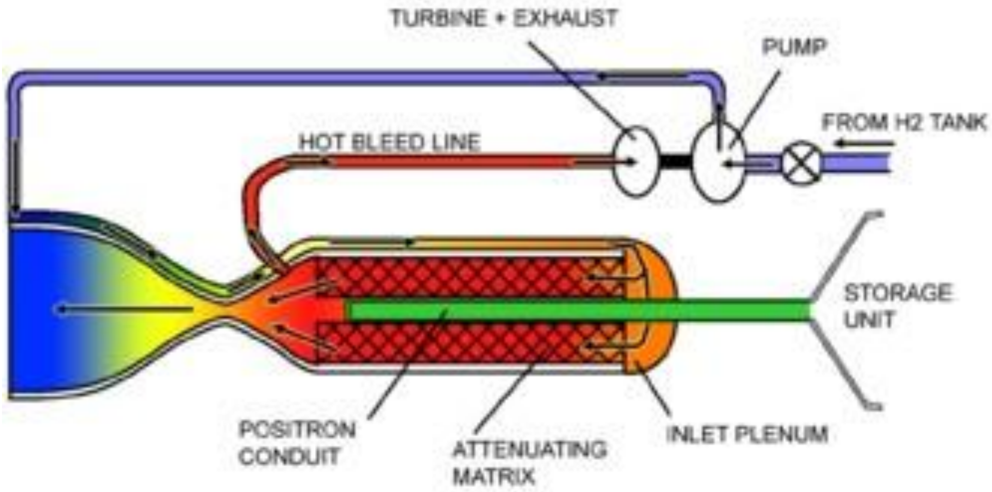


www.nasa.gov

- Electromagnetic forces are used to accelerate a *plasma*
  - A gas consisting of positive ions, electrons
  - 5000 – 9000 ºR
- Neutral beam is produced
- Higher thrust per unit area than electrostatic thruster
- Classifications
  - Magnetoplasmadynamic
  - Pulsed plasma
    - Electric discharge creates plasam from solid Telfon
  - Hall effect
    - Developed in Russia
    - Flew on U.S. STEx mission (1998)

The Future

- Interplanetary travel will require advanced forms of propulsion technology:
  - Antimatter
  - Nuclear fusion
  - Non-rocket methods



Effect of atmosphere

- The development of thrust, and the effect of atmosphere, can be examined through the derivation of the thrust equation, which relates the thrust of the rocket to the actual exhaust velocity, the pressure in the combustion chamber and the atmospheric pressure.

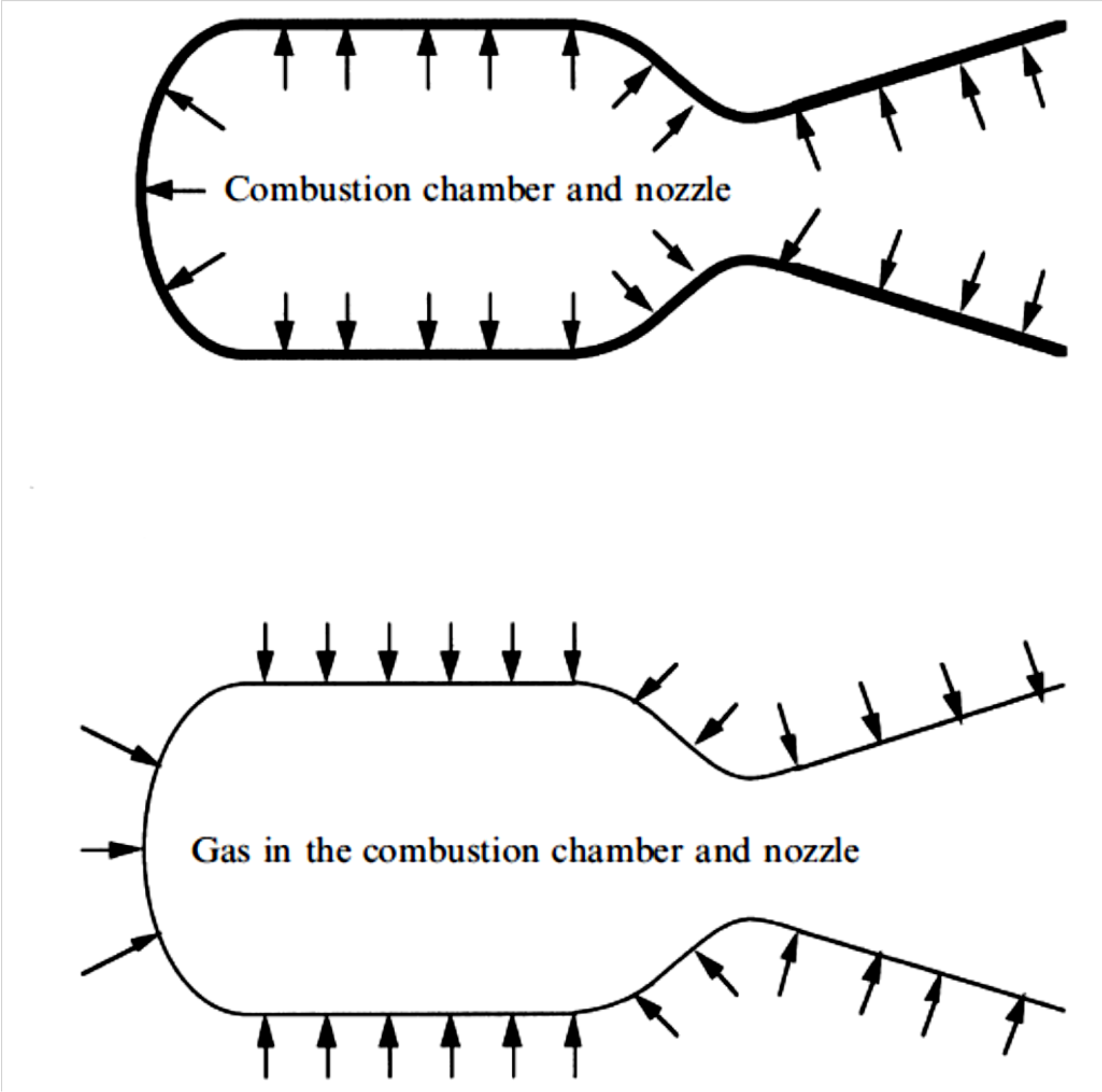
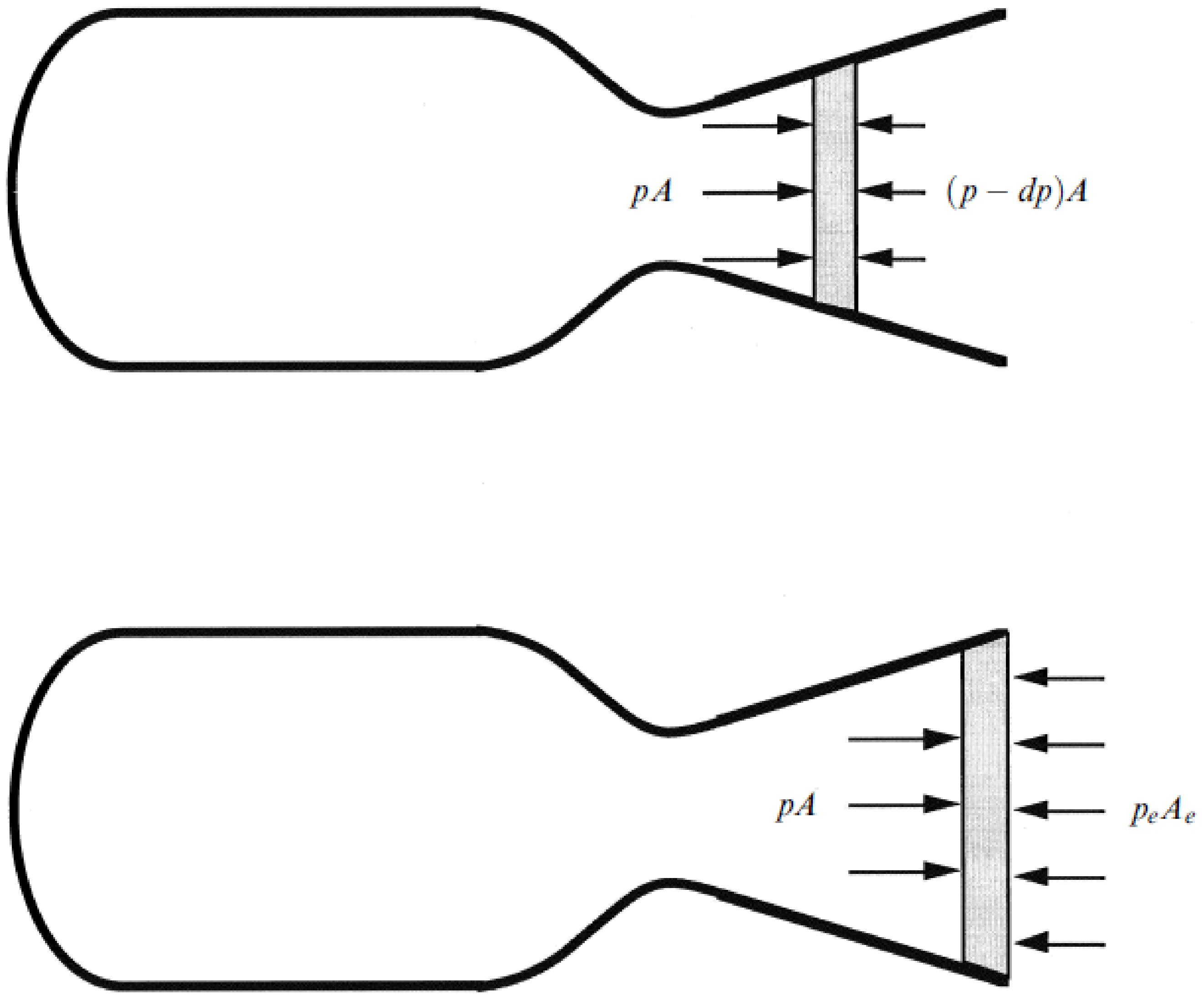
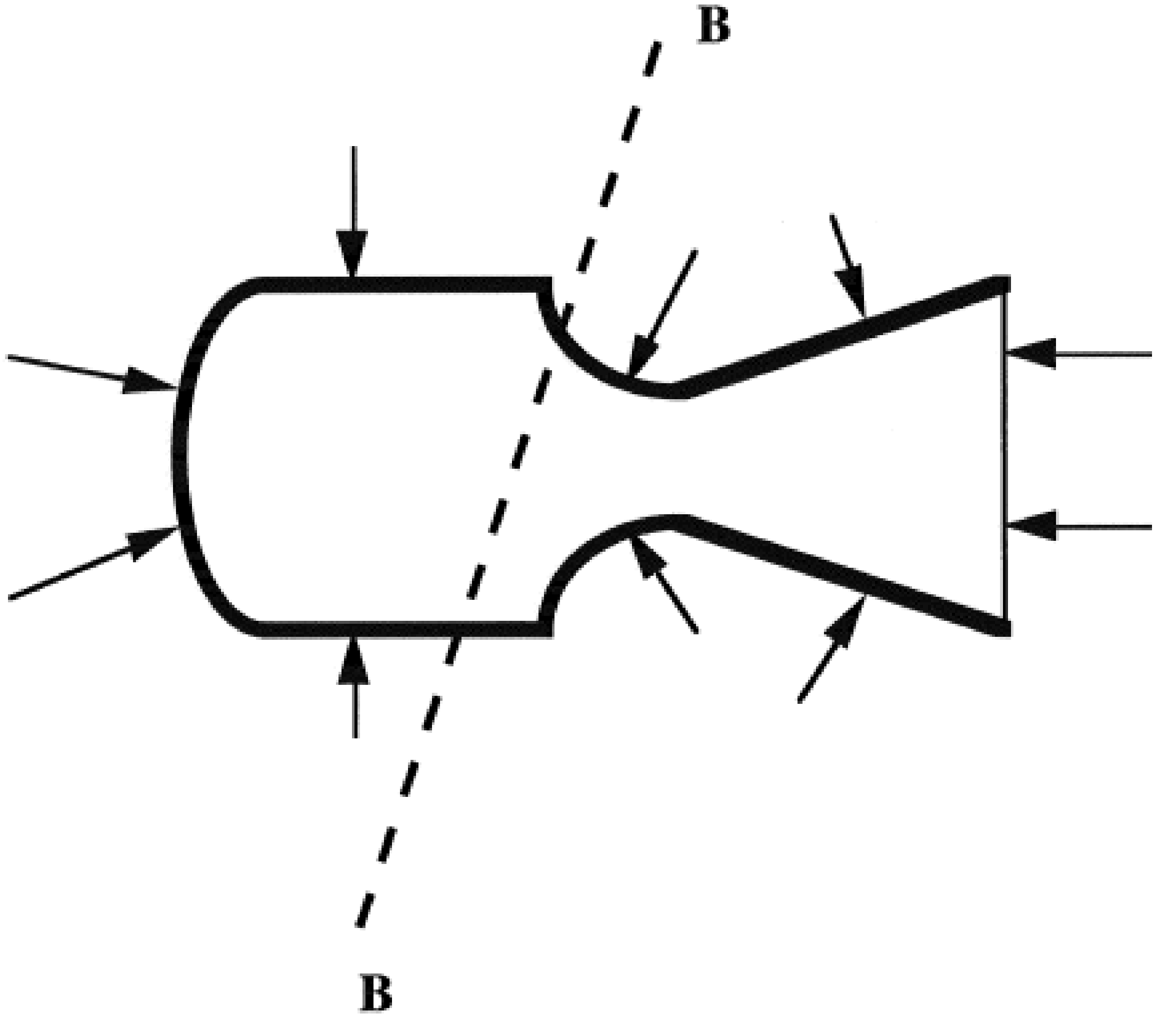


Fig. Forces in the combustion chamber and exit nozzle



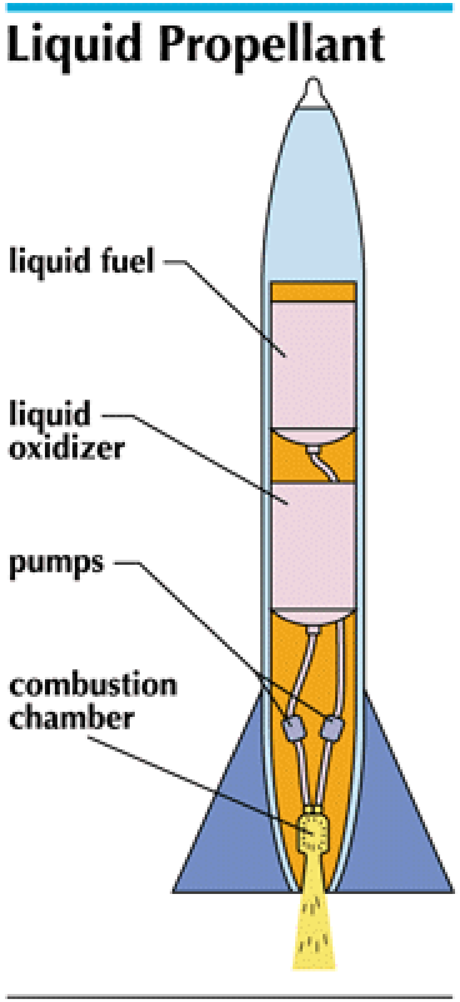
Gas Flow through Nozzle



Static force due to atmospheric Pressure

Chemical rockets

- Chemical propulsion is propulsion in which the thrust is provided by the product of a chemical reaction, usually burning (or oxidizing) a fuel. A chemical reaction combines two or more kinds of chemicals and makes a different chemical as a product. A commonly used reaction is combining hydrogen with oxygen to make water.
- The function of a chemical rocket engine system is to generate thrust through combustion by release of thermal energy derived from the chemical energy of the propellants.
- These combustion gases are ejected through a nozzle at high velocity.



Chemical rockets- the thrust chamber- processes- combustion, expansion- propellants.

- The analysis is usually divided into two somewhat separate sets of calculations:

1. The *combustion process*
2. The *nozzle gas expansion process*

- The *combustion process* is the first part. It usually occurs in the combustion chamber at essentially constant chamber pressure (isobaric) and the resulting gases follow Dalton's law. The chemical reactions or the combustions occur very rapidly. The chamber volume is assumed to be large enough and the residence time in the chamber long enough for attaining chemical equilibrium in the chamber.
- The *nozzle gas expansion process* constitutes the second set of calculations. The fully reacted, equilibrated gas combustion products enter the nozzle and undergo an adiabatic expansion in the nozzle. The entropy remains constant during a reversible (isentropic) nozzle expansion, but in real nozzle flows it increases slightly.

EXIT VELOCITY EQUATIONS

$$U_e = \sqrt{2C_p T_{o2}\left[1-\left(\frac{p_e}{p_{02}}\right)^{(\gamma-1)/\gamma}\right]} = \sqrt{\frac{2\gamma\overline{R}}{(\gamma-1)\overline{M}}T_{o2}\left[1-\left(\frac{p_e}{p_{02}}\right)^{(\gamma-1)/\gamma}\right]}$$

$$U_e \approx \sqrt{2\frac{\overline{Q}_R}{\overline{M}}\left[1-\left(\frac{p_e}{p_{02}}\right)^{(\gamma-1)/\gamma}\right]}$$

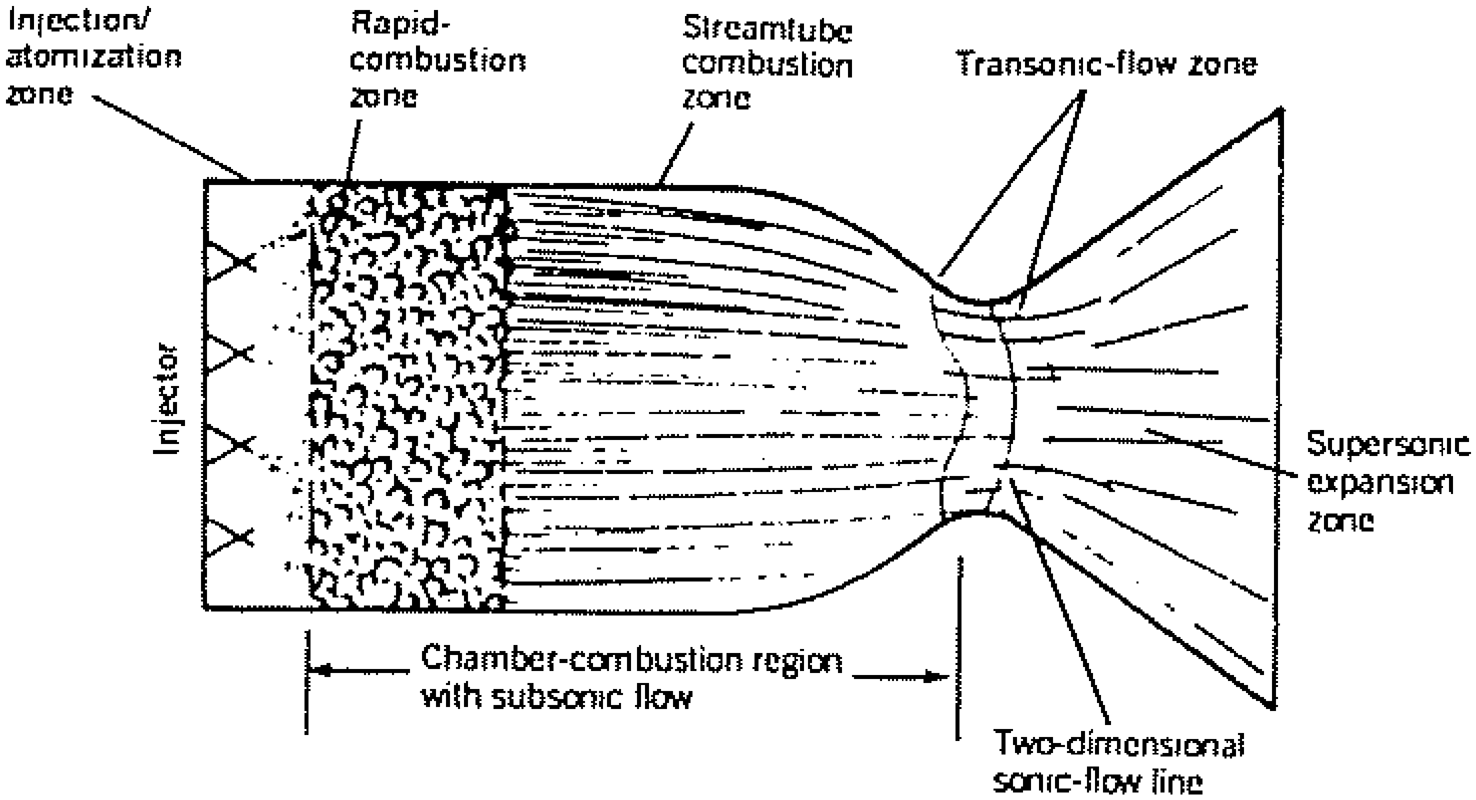
- Key Assumptions
  - Quantity of heat added at constant pressure
  - Constant specific heats
- For high Ue (for all thermal rockets), desire:
  - Propellants with low molecular weight, M
  - Propellant mixtures with large Q_R/M
  - High combustion chamber pressure, P_{02}

TYPES OF COMBUSTION CHAMBERS

- Two Types of rockets
  1. Liquid Rockets
     - Mono-Propellant (single compound)
     - Bi-Propellant (fuel and oxidizer)
  2. Solid Rockets
- Nozzles may look similar for these two types, but combustion chamber design quite different
- Goals same for each type:
  1. Understand factors that govern combustion process
  2. Choice of fuels for each of these options
  3. Design of combustion chamber
     - Sizing: Length, Area, Volume, residence time
  4. Performance of the solid vs. liquid rocket thrust chambers

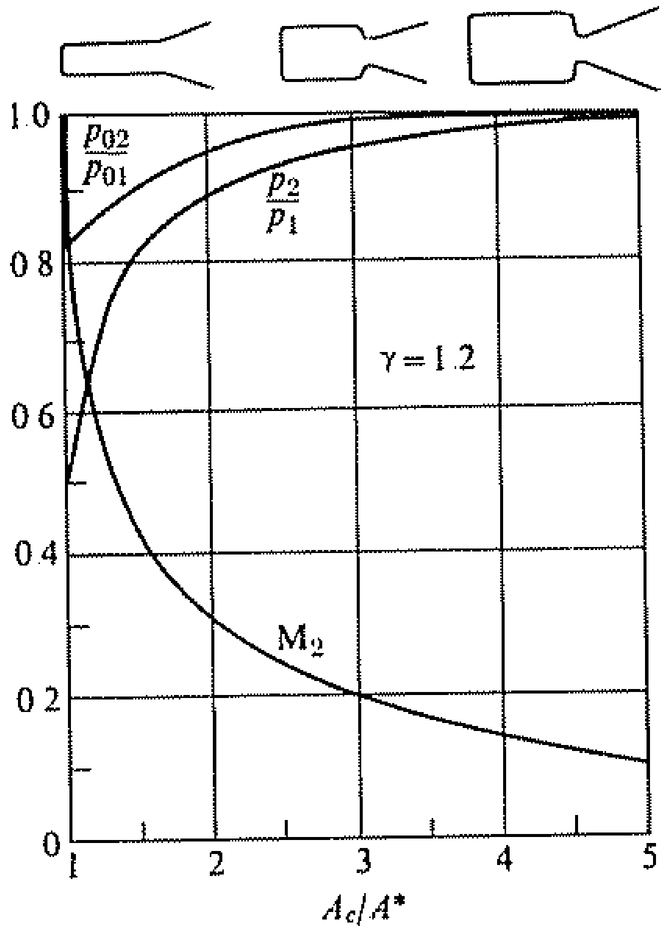
LIQUID COMBUSTION CHAMBER OVERVIEW

Atomization/vaporization



Combustor Chamber zones

Sizing of contraction ratio



Variation of Static and Stagnation Pressure ratios and Mach number with contraction ratio of γ = 1.2 Ac = Chamber Cross-sectional Area

PROPELLANTS: BASIC DEFINITIONS

- Propellant: general term to describe combustion materials (both oxidizer and fuel)

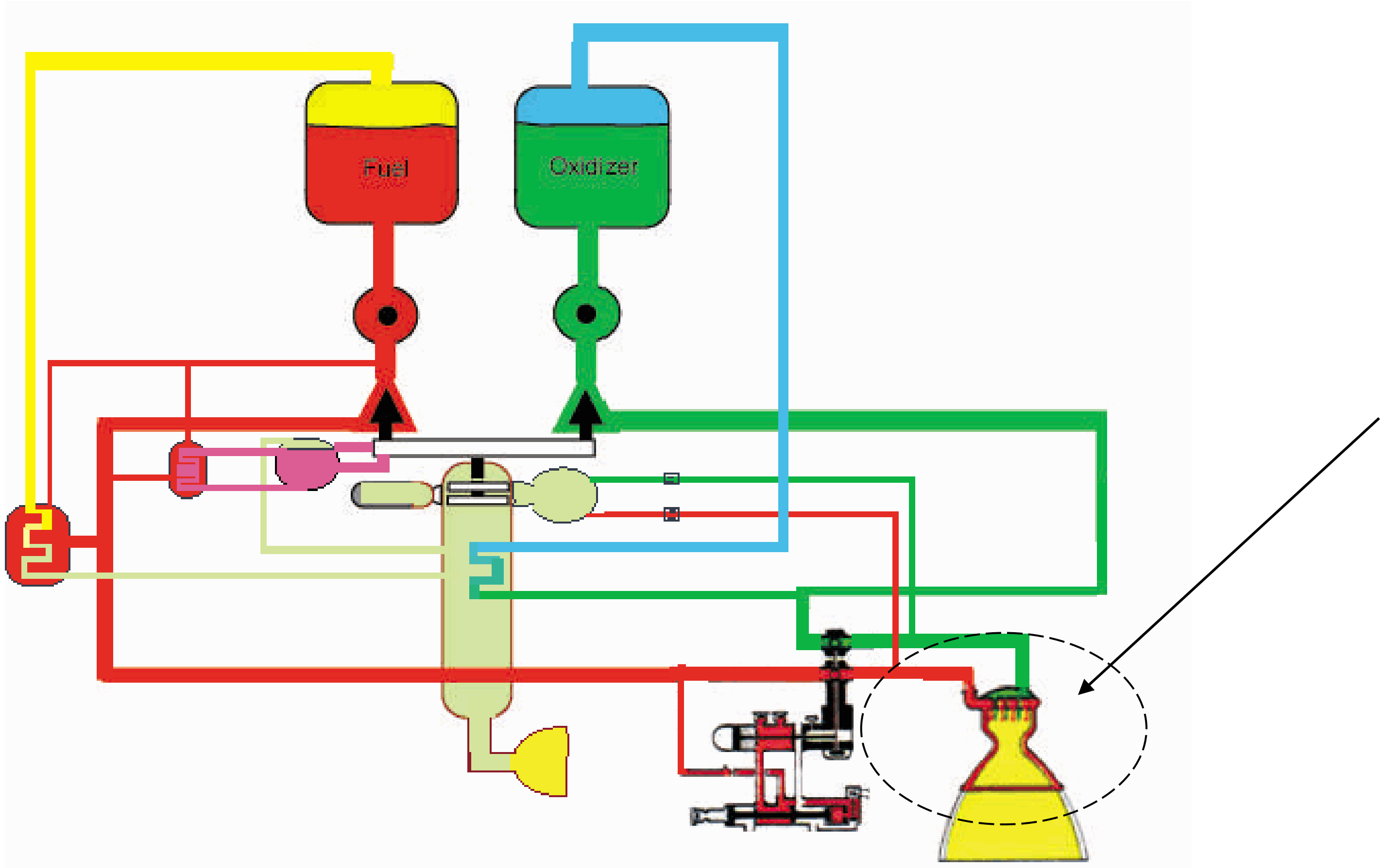
- Some liquid propellants are gases at atmospheric conditions
    - Only when temperature is dropped sufficiently and material pressurized does propellant become a liquid
        - Why do we have to do this?
    - Called **cryogenic** - maintained in low temperature condition
    - Common cryogenic propellants are liquid hydrogen and liquid oxygen

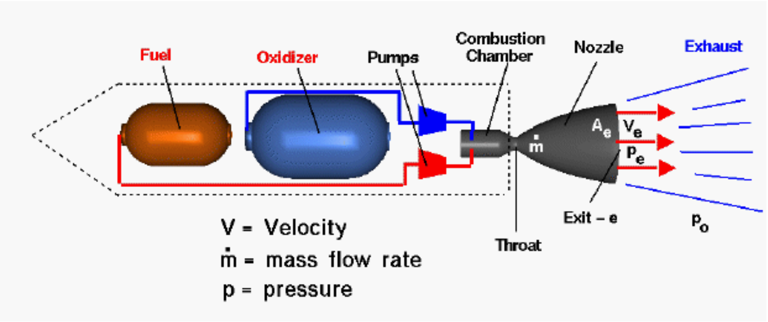
- In contrast to many liquid propellants **storable** at room temperature conditions
    - Storable liquid propellants easier to use, but don't provide as high Isp
    - High reliability, Shuttle orbital maneuvering and re-entry

- Cryogenics add significant complexity to a rocket system, but extra Isp is worth it
    - Venting system, special tanks, shrinkage, flexible joints, leakage, etc.

Propellant also serves as coolant

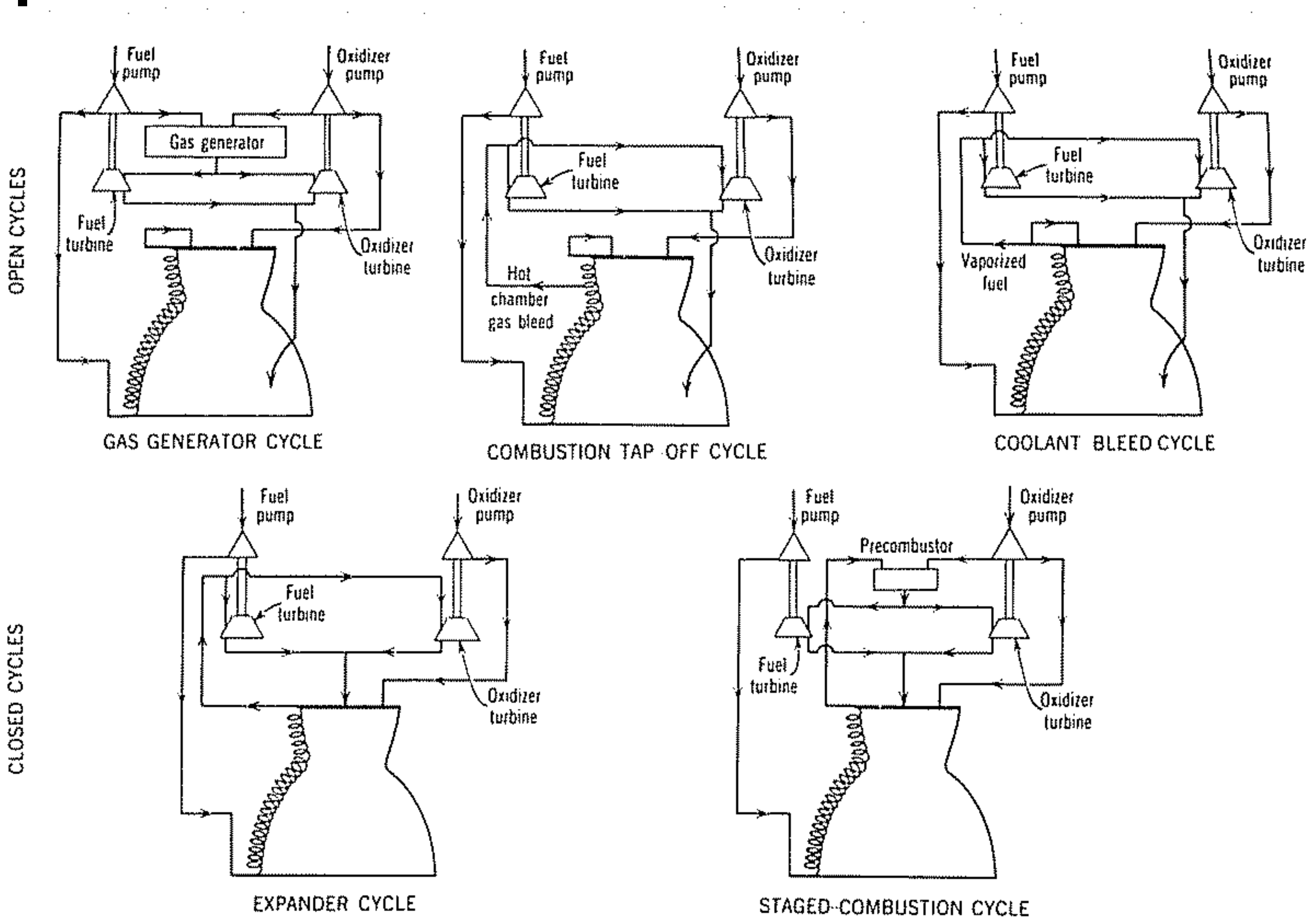
- Propellants are often used to cool portions of the rocket (combustion chamber and nozzle) prior to entering combustion chamber to be burned





Combustion chamber and portion of nozzle are cooled with propellants

PROPELLANT WILL VARY WITH CYCLE



COMMON LIQUID ROCKET FUELS

Oxidizer	**Fuel**	**Specific Impulse (maximum)**
Liquid Oxygen	Liquid Hydrogen	391
"	RP-1	300
"	Ammonia	296
"	95% Ethyl Alcohol	287
"	Hydrazine	313
"	50% UDMH 50% Hydrazine	312
"	UDMH	310
Liquid Fluorine	Liquid Hydrogen	410
"	Hydrazine	363
"	Ammonia	357
Nitrogen Tetroxide	Hydrazine	292
"	50% UDMH 50% Hydrazine	288
"	UDMH	285
"	RP-1	276
"	92.5% Ethyl Alcohol	267
95% Hydrogen Peroxide	Hydrazine	285
"	50% UDMH 50% Hydrazine	279
"	UDMH	278
"	RP-1	273
NONE (Monopropellant)	Hydrogen Peroxide	140
NONE (Monopropellant)	Hydrazine	205
NONE (Monopropellant)	Nitromethane	180
NONE (Monopropellant)	Methylacetylene	160

- Most Common Liquid Oxidizers
  - LOX
  - Hydrogen Peroxide
  - Nitric Acid
  - Nitrogen Tetroxide
  - Liquid Florine

- Most Common Liquid Fuels
  - Hydrocarbon fuels (RP1, kerosense, methane)
  - Liquid hydrogen
  - Hydrazine (also mono)
  - Unsymmetrical Dimethylhydrazine

Propellants: general comments

- Combustion temperature directly reflects chemical energy of reaction
- Using O_2 and RP-1 produces higher temperature than O_2 and H_2
  - This is because RP-1 contains more chemical energy than H_2
  - However, H_2 has much lower molecular weight
  - Result is that O_2 and H_2 produces highest exit velocity (except F_2-H_2)
- Additional H_2 may be added, which actually raises exit velocity
- Maximum exhaust velocity is shifted away from stoichiometric value, in direction of lower molecular weight

- Why use a propellant with high molecular weight?
  - Exhaust velocity is not only criterion
  - Also interested in high thrust
  - Heavy propellants giver higher mass flow rate
    - Velocity might be lower, but overall thrust might be increased
  - This is an application dependent design trade-off

PROPELLANTS: GENERAL COMMENTS

- RP-1 is an inexpensive fuel, but provides lower performance (similar to kerosene)

- Fluorine and LH_2 produces highest temperature of any bi-propellant system
  - Highly toxic and corrosive

- Hydrogen peroxide (H_2O_2) deteriorates at about 1% a year, so cannot be used for long term missions.

- Hydrazine (N_2H_4), Florine (F_2) and UDMH (unsymmetrical dimethylhydrazine) are good propellants but highly toxic and flammable. Very dangerous to handle

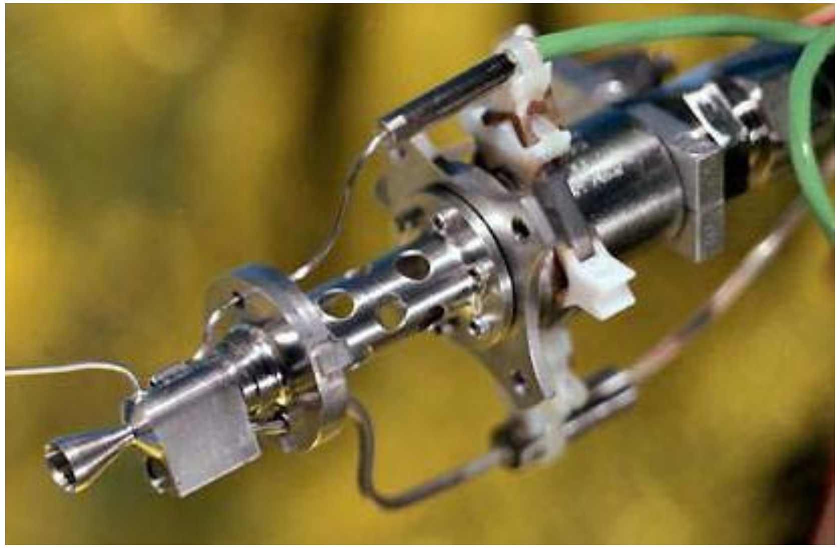
- Nitrogen tetroxide (N_2O_4) is a very good oxidizer
  - Easy to store and transport, but also very toxic, hazardous to skin contact and creates toxic cloud (BFRC) in when exposed to atmosphere

PROPELLANTS: ADDITIVES

- Sometimes additives are mixed into propellants for:
  - Lower molecular weight
  - Improve cooling characteristics
    - Rocket designs may be limited by temperature on turbine, combustion chamber, or nozzle
  - Lower freezing point (prevent propellant from freezing in space)
  - Reduce corrosive effects
  - Facilitate easier ignition
  - Stabilize combustion

Mono-propellant system

- Monopropellant hydrazine thrusters for satellites, space probes and spacecraft
- Used for station-keeping (orbital correction) and small thrust level applications
- Launch vehicle roll control and upper stage orientation and precision maneuvers (used on Ariane 5 launcher)

Examples of 0.1N Hydrazine mono-propellant thrusters, Isp~210 s

Why Mono-Propellants?

- Less complicated
  - Only need 1 propellant
  - Only need 1 set of drive turbines and pumps
  - Less plumbing
- Easy to turn on and off, reliable
- Less dangerous, no chance of interaction of high pressure fuel and oxidizer
- Less costly to build and low maintenance

- Drawback is limited performance, Isp and thrust

- Mono-propellants
  - Most common propellant in use today are Hydrazine (N_2H_4) and Hydrogen Peroxide (H_2O_2)
  - Also useful with catalyst to decompose to produce heat and gas
  - Storable for long periods of time
  - Useful for short bursts of thrust

BI-PROPELLANT LIQUID ROCKET SYSTEM

- Most liquid fuel rockets are bipropellant
- Most also require a spark or some other method of activation to ignite
- Mixtures of liquid oxygen (LO_2) with hydrocarbons such as kerosene and RP1 or with liquid hydrogen (LH_2) are most typical
- These types of rockets produce enormous thrust due to massive amount of propellant used
  - Example: External tank on the space shuttle contains over 1 million pounds of LO_2 and over 200,000 pounds of LH_2

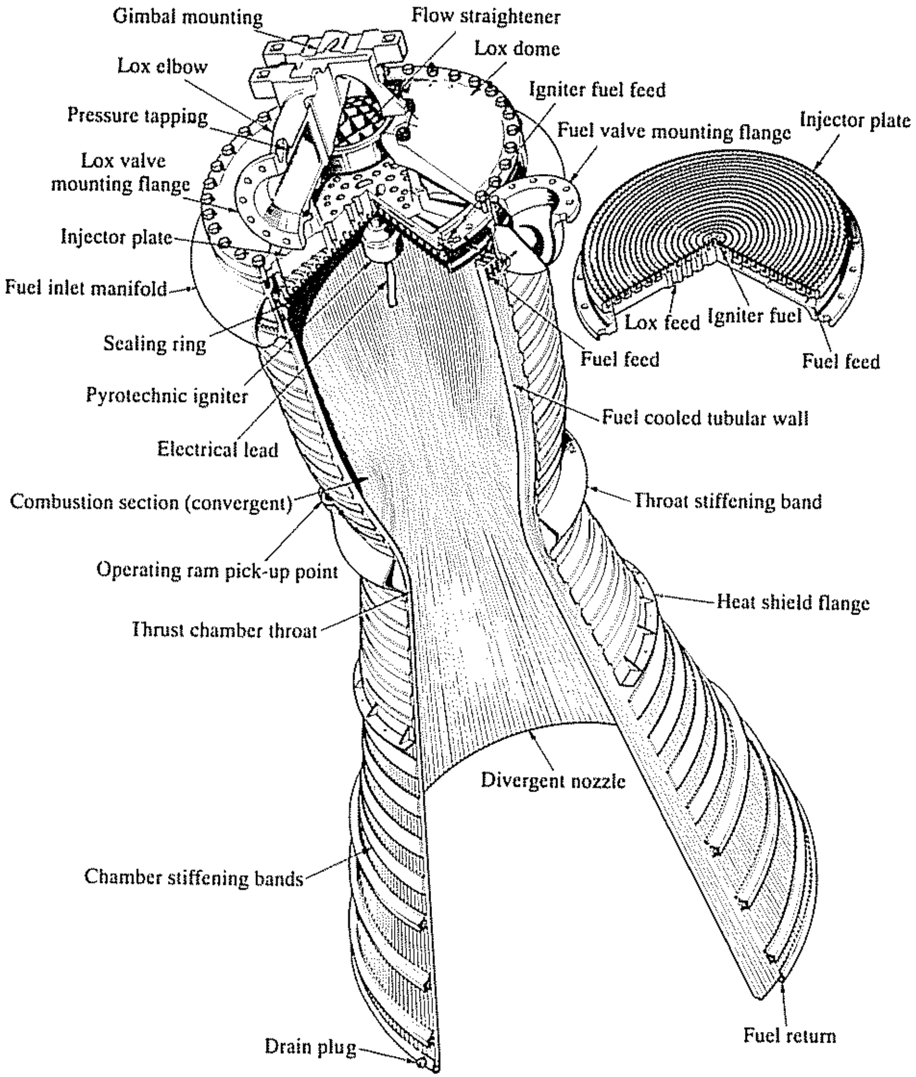
- Some work being done on Tri-Propellant Rockets
  - These work in basically same way as bipropellant rockets
  - Combine two fuels and one oxidizer to give a mixture of efficiency and thrust

ROCKET FUEL SELECTION GUIDE

- Desirable Physical Properties
  - Low freezing point
  - High specific gravity (dense propellant)
  - Stability (with time)
  - Heat transfer properties
  - Pumping properties (low vapor pressure, low viscosity)
  - Small variation in physical characteristics with temperature
  - Ignition, combustion, and flame properties

- Performance of Propellants
- Economic Factors
- Physical Hazards (Explosion, Fire, Spills)
- Health Hazards
- Corrosion

BI-PROPELLANT LIQUID ROCKET



WHY DO LH_2-LOX ROCKETS RUN FUEL RICH?

- Desire high heating value, Q_R, and low molecular weight, M
  - Running rocket fuel rich (or lean) will reduce Q_R
  - However, if fuel is lighter in molecular weight than oxidizer, fuel rich will reduce average molecular weight
  - The overall ratio Q_R/M is increased
  - "In many cases fuel-rich mixtures are burned, as it is found that the resultant reduction in M more than offsets the accompanying reduction in Q_R"
  - Also note that when dissociation is present, O/F ratio, much less than stoichiometric for maximum specific impulse

Combustion Reactions and Analysis

- **Mechanism of Combustion**
- The chemical reaction equations presented here do not portray the actual mechanism of combustion; they merely indicate the initial and final chemical compositions of a reaction.
- In most cases the reactions involve a sequence of steps, leading from the reactants to the products, the nature of which depends on the temperature, pressure, and other conditions of combustion.
- Fuel molecules, for instance, may undergo *thermal cracking*, producing more numerous and smaller fuel molecules and perhaps breaking the molecules down completely into carbon and hydrogen atoms before oxidation is completed.
- In the case of solid fuels, combustion may be governed by the rate at which oxidizer diffuses from the surrounding gases to the surface and by the release of combustible gases near the surface.
- Combustion of solids may be enhanced by increasing the fuel surface area exposed to the oxidizer by reducing fuel particle size.

- We know that, for combustion to occur, molecules of oxidizer must affiliate with fuel molecules, an action enhanced by the three T's of combustion: turbulence, time, and temperature.
- Chemical reactions take place more rapidly at high temperatures but nevertheless require finite time for completion.
- It is therefore important that burners be long enough to retain the fuel-air mixture for a sufficiently long time so that combustion is completed before the mixture leaves.
- *Turbulence*, or *mixing*, enhances the opportunities for contact of oxidizer and fuel molecules and removal of products of combustion.

- A flame propagates at a given speed through a flammable mixture. It will propagate upstream in a flow of a combustible mixture if its *flame speed* exceeds the flow velocity.
- If a fixed flame front is to exist at a fixed location in a duct flow in which the velocity of the combustion gas stream exceeds the propagation speed, some form of *flame stabilization* is required. Otherwise the flame front is swept downstream and flameout occurs.
- Stabilization may be achieved by using fixed *flameholders* (partial flow obstructions that create local regions of separated flow in their bases where the flame speed is greater than the local flow velocity) or by directing a portion of the flow upstream to provide a low-speed region where stable combustion may occur.

- Each combination of oxidizer and fuel has been seen to have a particular stoichiometric oxidizer-fuel ratio for which the fuel is completely burned with a minimum of oxidizer.
- It has also been pointed out that it is usually desirable to operate burners at greater than the theoretical air-fuel ratio to assure complete combustion of the fuel and that this is sometimes referred to as a *lean mixture*.
- Occasionally it may be desirable to have *incomplete combustion*, perhaps to produce a stream of products in which carbon monoxide exists or to assure that all the oxidizer in the mixture is consumed. In that case a burner is operated at less than the stoichiometric air-fuel ratio with what is called a *rich mixture*.
- There are limits to the range of air-fuel ratios for which combustion will occur called *limits of flammability*. Here the density of the mixture is important. The limits of flammability around the stoichiometric A/F are reduced at low densities. If combustion is to occur reliably in mixtures at low densities, it is necessary to closely control the air-fuel ratio.

- **Combustion Analysis of Solid Fuels**
- In the determination of the air-fuel ratio and flue gas composition for the combustion of solid fuels, it is important to account for the ash and moisture in the fuel in the as-fired condition.
- In the following analyses, all of the elements of the reactants in the fuel and oxidizer are assumed to be present in the flue gas products except for the ash, which is assumed to fall as a solid or flow as molten slag to the furnace bottom.
- Nitrogen and oxygen are present in many solid fuels and should be accounted for in predicting the flue gas composition.
- While both carbon monoxide and oxygen may be present in combustion products at the same time because of imperfect mixing of combustibles and oxygen in some instances, we will assume for prediction of the flue gas composition that perfect mixing occurs such that no carbon monoxide is present when excess air is supplied.

- Stoichiometric Reaction: A stoichiometric reaction is defined such that the only products are carbon dioxide and water. The components on the left side of a chemical reaction equation which are present before the reaction are called reactants, while the components on the right side of the equation which are present after the reaction are called products or exhaust.
- One kmole of a substance has a mass in kilograms equal in number to the molecular mass (molar mass) of that substance. Mathematically, $m = NM$ [kmole] [kg/kmole], where: m = mass[kg], N = number of moles [kmole], M = molecular mass[kg/kmole], 1 kmole = 6.02×10^{26} molecules.

Examples

$$CH_4 + 2O_2 = CO_2 + 2H_2O$$

$$C + O_2 = CO_2$$

$$CH_4 + 2O_2 \Rightarrow CO_2 + 2H_2O$$
$$[12 + 4(1)] + 4(16) \Rightarrow [12 + 2(16)] + 2[2(1) + 16] = 80$$

There are 2 moles of water in the 3 moles of combustion products, and therefore a *mole fraction* of water in the combustion products of $x_{water} = 2/3 = 0.667$.
Similarly, $x_{Carbon\ dioxide} = 1/3 = 0.333$ moles of CO_2 in the products.

There are 44 mass units of CO_2 in the 80 mass units of products for a *mass fraction of* CO_2 *in the products,*

$$\mathrm{mf}_{\mathrm{carbon\ dioxide}} = 44/80 = 0.55$$

Likewise, the mass fraction of water in the products is 2(18)/80 = 0.45.

Combustion Efficiency

- Combustion efficiency is defied as the fraction of the fuel energy supplied which is released in the combustion process.
- Combustion efficiency of liquid rocket engines can be divided into two parts

1. Vaporization Efficiency ($\eta_{c,}$*, $_{vap}$)
2. Mixture Efficiency ($\eta_{c,}$*, $_{mix}$)

Chemically Reacting Gas Mixtures

- The working fluids in engines are mixtures of gases. Depending on the problem under consideration and the portion of the engine cycle in which it occurs chemical reactions may:

(1) be so slow that they have a negligible effect on mixture composition (the mixture composition is essentially "frozen");

(2) be so rapid that the mixture state changes and the composition remains in chemical equilibrium;

(3) be one of the rate-controlling processes that determine how the composition of the mixture changes with time.

Chemical Equilibrium

- It is a good approximation for performance estimates in engines to regard the burned gases produced by the combustion of fuel and air as in chemical equilibrium.
- It means that the chemical reactions, by which individual species in the burned gases react together, produce and remove each species at equal rates. No net change in species composition results.

DISSOCIATION: GENERAL COMMENTS

Q: What does combustion products are dissociated mean?

A: Consider burning (combustion) of a hydrogen and oxygen mixture

- In complete combustion we have: $H_2 + \frac{1}{2}O_2 \rightarrow H_2O$

- All reactants go into forming products
  - In this case hydrogen and oxygen form water
- At high enough temperature products may further breakdown
- Reaction may proceed in reverse, H_2O breaking down into H_2 and O_2

$$H_2 + \frac{1}{2}O_2 \rightarrow \alpha H_2O + \beta H_2 + \delta OH + \nu H + \varepsilon O_2 + \eta O$$

DISSOCIATION: GENERAL COMMENTS

$$H_2 + \frac{1}{2}O_2 \rightarrow \alpha H_2O + \beta H_2 + \delta OH + \nu H + \varepsilon O_2 + \eta O$$

- Think about terms β, δ, ν, ε, η as a 'wasted energy opportunity'

- Instead of liberating all chemical energy of reaction into just water, some of it is subsequently 'wasted' by using high T energy to form other products

- Highest U_e and I_{sp} will always be achieved with **<u>no</u>** dissociation
  - Never occurs at operating T of actual rockets
  - Must deal with dissociation of products

- Also note, left side of equation still has stoichiometric ratio of O/F
  - Ratio is not best one to use when dissociation occurs (lower side)

NOZZLE FLOW OF REACTING GASES

Suppose that we have determined composition of gases in combustion chamber

Now compute flow through nozzle taking into account chemical reactions

- 3 major deviations from simple model based on ideal gas behavior
  1. Composition of gas is not necessarily constant in flow
     - Properties that are composition dependent must be treated as variables along flow direction
     - Specific heats, gas constant and ratio of specific heats
  2. Sum of thermal energy and kinetic energy is no longer constant
     - Exchange of chemical energy and thermal energy
  3. Non-isentropic
     - We can neglect this entropy change for 2 special cases

NOZZLE FLOW OF REACTING GASES

- To deal with the flow, we first note that energy conservation for the gas flow is:

$$H_c = H(T,P) + \frac{u^2}{2}$$

- This replaced the ideal gas energy equation:

$$C_p T_c = C_p T + \frac{u^2}{2}$$

- Need statement about variation of entropy
  - Transfer from chemical to thermal energy takes place at finite rate, therefore entropy increase
- There are 2 limiting cases for which entropy change is very small
  1. **Chemical Equilibrium**
  2. **Frozen Flow**

NOZZLE FLOW OF REACTING GASES

- **Chemical Equilibrium**
  - Approached if the reactions occur fast enough to keep up with T & P change caused by expansion
  - Flow time >> Reaction time
  - Chemical energy is transferred through an infinitesimal ΔT

$$S = \sum x_i(P,T)S_i(P,T) = S_c$$

  - x_i determined by equilibrium at local T & P

$$S = \sum x_i(P_c,T_c)S_i(P,T) = S_c$$

- **Frozen Flow**
  - Reactions occur slowly, x_i are fixed at their chamber values
  - Flow time << reaction time
  - No chemical energy release

NOZZLE FLOW OF REACTING GASES

- Chemical Equilibrium and Frozen Flow are 2 limiting cases
  - Provide an upper and lower limit for the velocity at a given pressure

- **Chemical Equilibrium** gives **maximum** thermal energy availability for conversion for kinetic energy
- **Frozen Flow** gives **minimum**

Phenomenological explanation of why this is so:

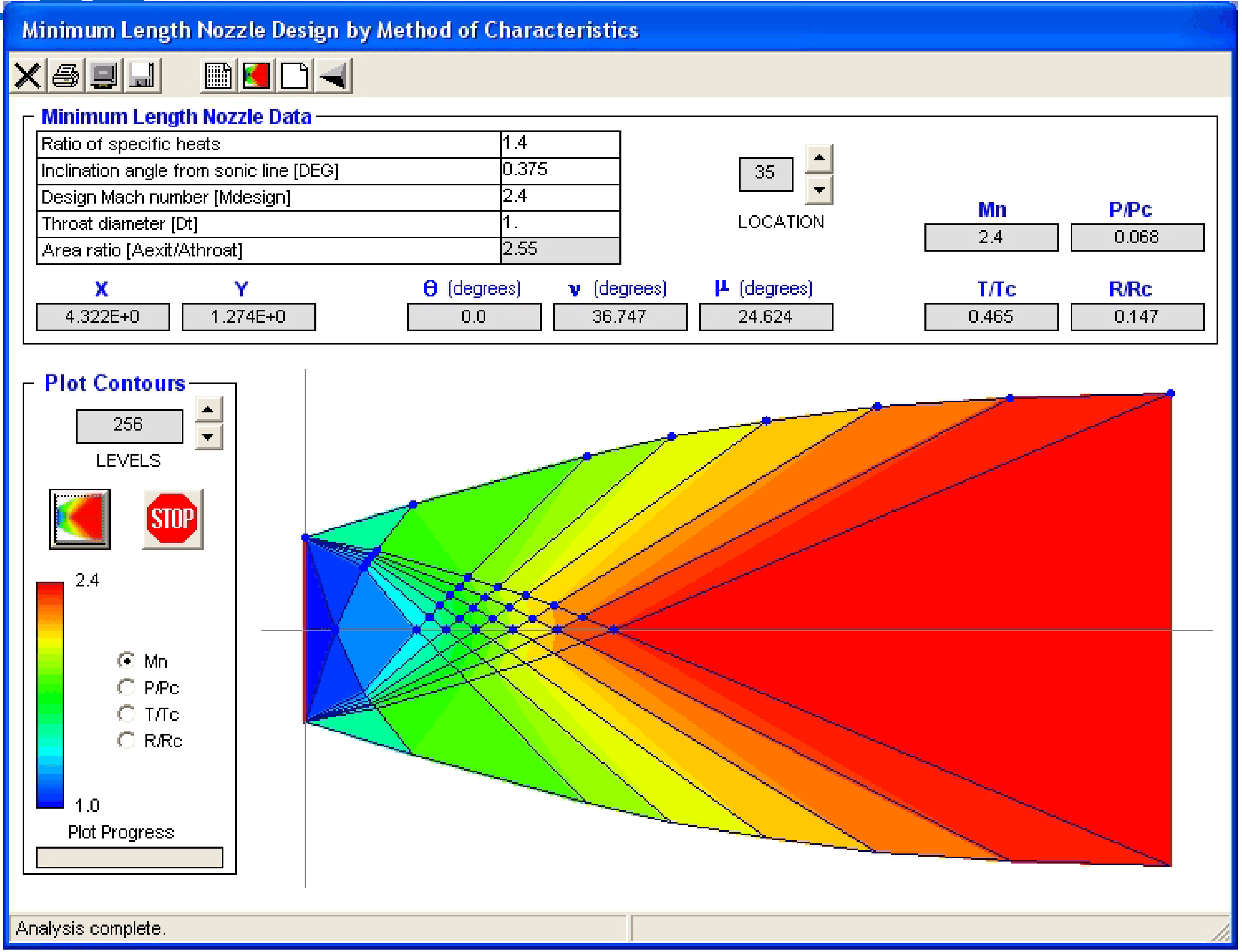
- Gases leaving combustion chamber are so hot products of combustion are highly dissociated
- Dissociated compounds will tend to recombine because of large drop in T during nozzle expansion process
- Recombination is exothermic and acts as heat source in flow (changes in C_p and γ)

- **However, Isp always lower with dissociation than with no dissociation at all**

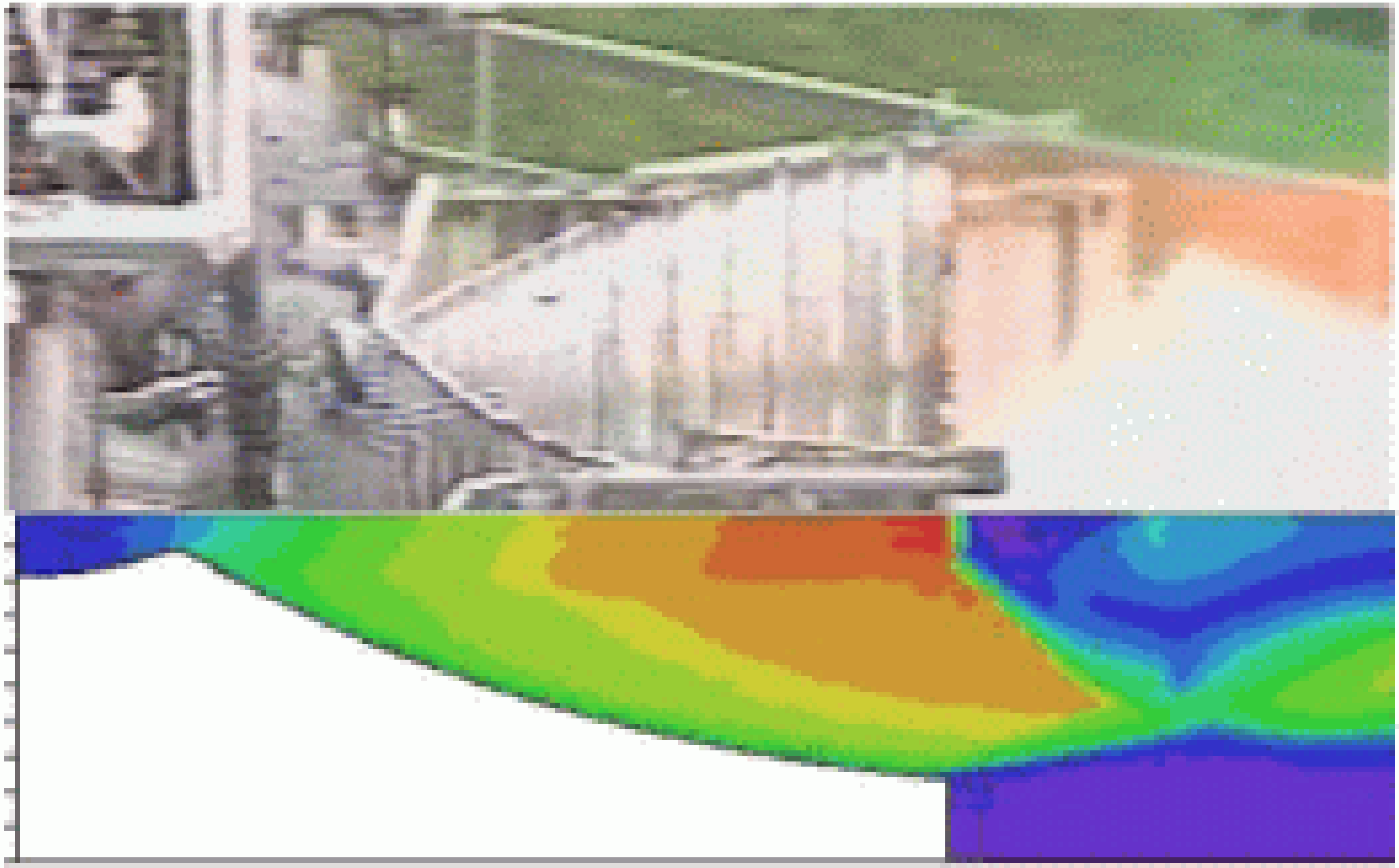
HOW DO WE DESIGN NOZZLES?

- Start with 1-D approximation
  - Always first step
  - Tools in place to do this
- At design stage
  - 2D analysis
    - Method of characteristics (2D or axisymmetric)
    - Curve-Fitting to specify shape $L < L_{opt(M.O.C.)}$ and iteration with M.O.C.
    - Boundary layer approximations
    - Divergence approximation
  - Complete 3D analysis of flow field
    - CFD
    - Consider non-uniformities
    - Critical operating regimes (near separation)
  - These analyses lead to a few percentage points in Isp, but well worth it

METHOD OF CHARACTERISTICS EXAMPLE



CFD EXAMPLE



Rocket Performance:

- -Equation of state (Ideal gas law)
- -Continuity equation (Conservation of mass)
- -Conservation of momentum
- -Conservation of energy (first law of thermodynamics)
- -Isentropic expansion (second law of thermodynamics)
- Considerable numerical computations are thus necessary. Sophisticated computer modelling & simulation are to be done exclusively for accurate prediction of performance.

Design Parameters

1.Thrust levels

2.Performance

3.Burn time

4.Propellant mixture ratio

5.Burnout mass.

6.Envelope/Size

7.Reliability

8.Cost

9.Schedule.

1.Thrust Level.

-Lift off weight

-Minimum & maximum acceleration allowed.

-Single engine or multiple engine.

-Variable thrust.

2. Performance

-Specific Impulse

-Theoretical

-Tested

-Flight

3.Burn duration

-Tank capacity

-Pressurant storage

-Nozzle cooling

-Thrust build up

-Shut down

4.Mixture Ratio

-Stoichiometric ratio

-(max. temp & heat)

-Exhaust velocity

-(gas properties)

-Optimum ratio

-(Residence time in combustion chamber; Chamber wall cooling)

5.Burnout Mass

-Dry mass + residual propellants

-Burn out mass

-Wet mass –CG & MI

-Wet gimbal mass.

6.Envelope/Size

-Vehicle Structure

-Handling

-Servicing

-Realisation

-Expansion Ratio

7.Reliability

-Unmanned/manned mission

-Review of design, calculation & drawing

-Paintaking execution of the above

-Familiarizing with correct application of accepted, prior design standards & procedures

-Written statements & instruction

-Simplicity, Redundancy & Safety

-Test condition vs. Flight condition

8.Cost

-Design Phase

-Engineering Phase

-Programme phase.

9. Schedule.

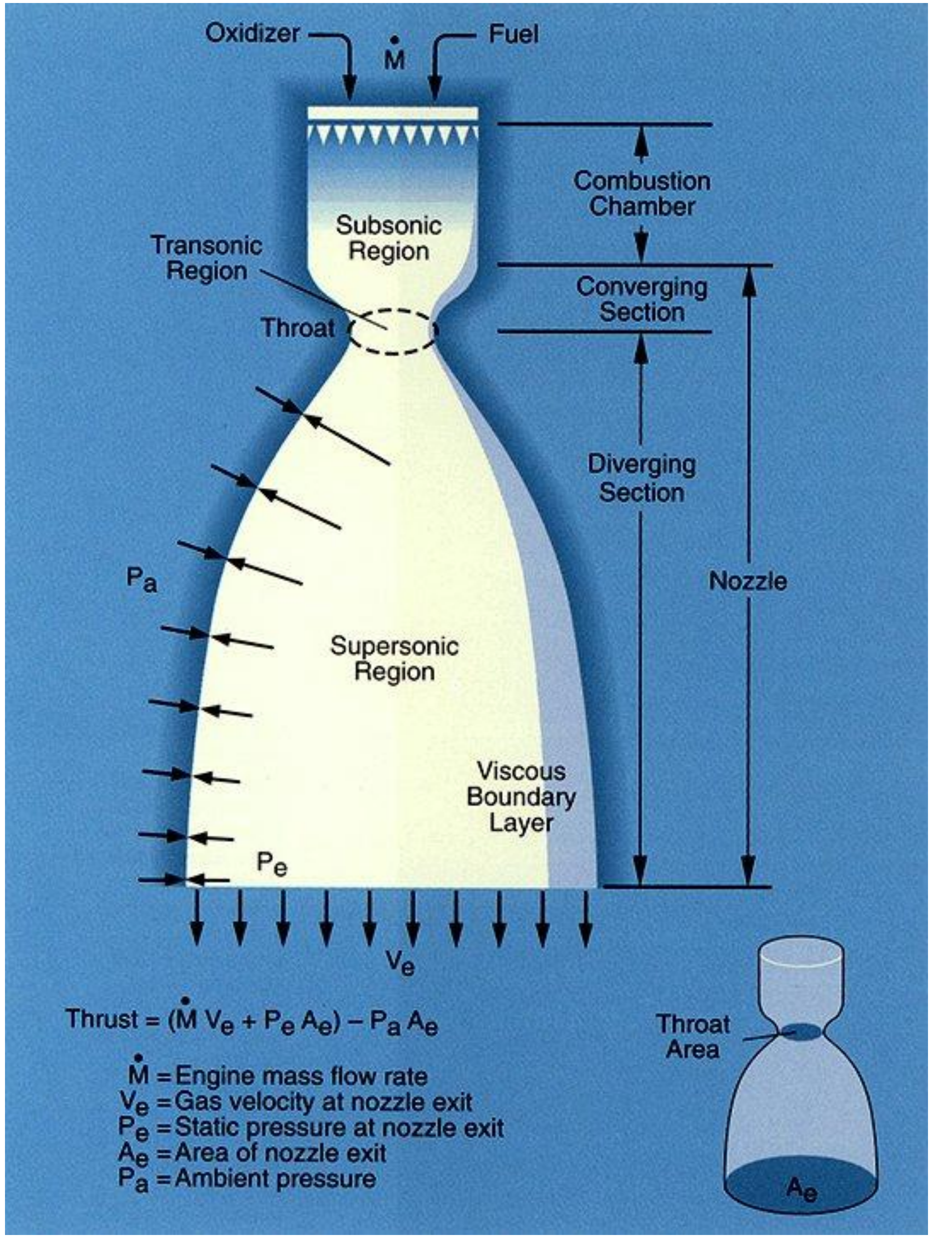
-Availability of subsystems

-Design Quality

-System Analysis

-Materials, Fabrication, Handling.

NOZZLE BASICS REVIEW



- Nozzle produces thrust
- Convert thermal energy of hot chamber gases into kinetic energy and direct that energy along nozzle axis
- Exhaust gases from combustion are pushed into throat region of nozzle
- Throat is smaller cross-sectional area than rest of engine → gases
- Nozzle gradually increases in cross-sectional area allowing gases to expand and push against walls creating thrust
- Mathematically, ultimate purpose of nozzle is to expand gases as efficiently as possible so as to maximize exit velocity

$$F = \dot{m}_e V_e + \left(P_e - P_a\right) A_e$$

$$F = \dot{m}_e V_e$$

NOZZLE BASICS REVIEW

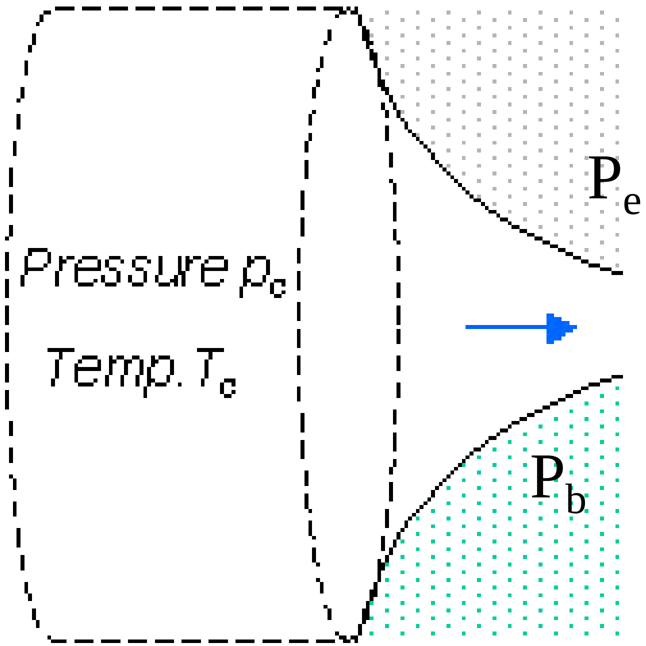
- Expansion Area Ratio:
  - Most important parameter in nozzle design is expansion area ratio, ε

$$\varepsilon = \frac{A_{exit}}{A_{throat}} = \frac{A_e}{A^*}$$

  - Fixing other variables (primarily chamber pressure) → only one ratio that optimizes performance for a given altitude (or ambient pressure)
  - However, rocket does not travel at only one altitude
    - Should know trajectory to select expansion ratio that maximizes performance over a range of ambient pressures

- Other factors must also be considered
  - Nozzle weight, length, manufacturability, cooling (heat transfer), and aerodynamic characteristics.

OPERATION OF CONVERGING NOZZLES



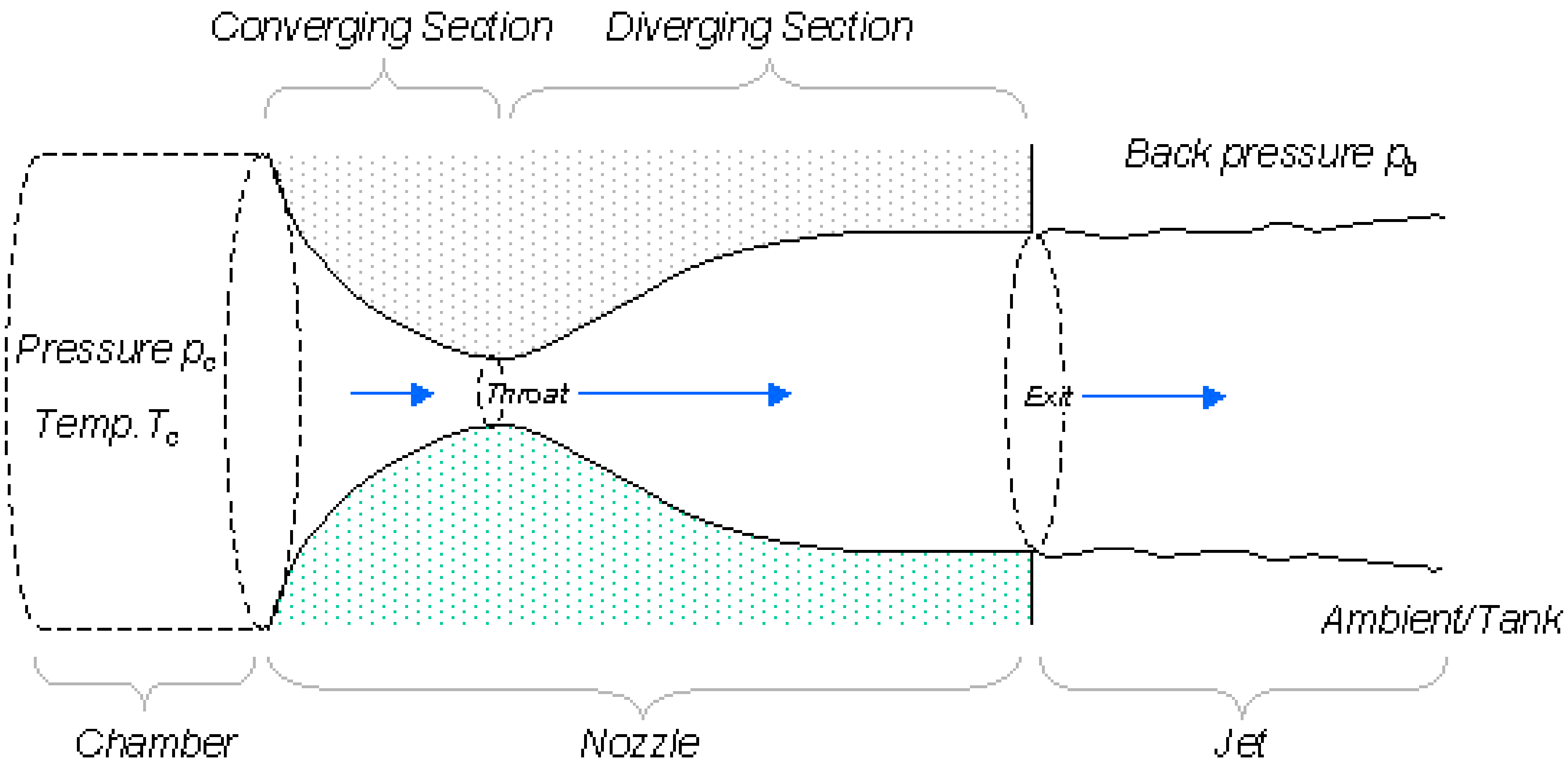
- A converging nozzle will choke when P_b is reduced to a critical value P* (found from isentropic relations)
- P_e = P* is called design pressure ratio
- For P_b > P*
    - Flow will be subsonic
    - $P_e = P_b$
- For any P_b < P*
    - Flow will be sonic at nozzle exit
    - $P_e > P_b$
    - Flow is choked and P_e = P*

Questions?

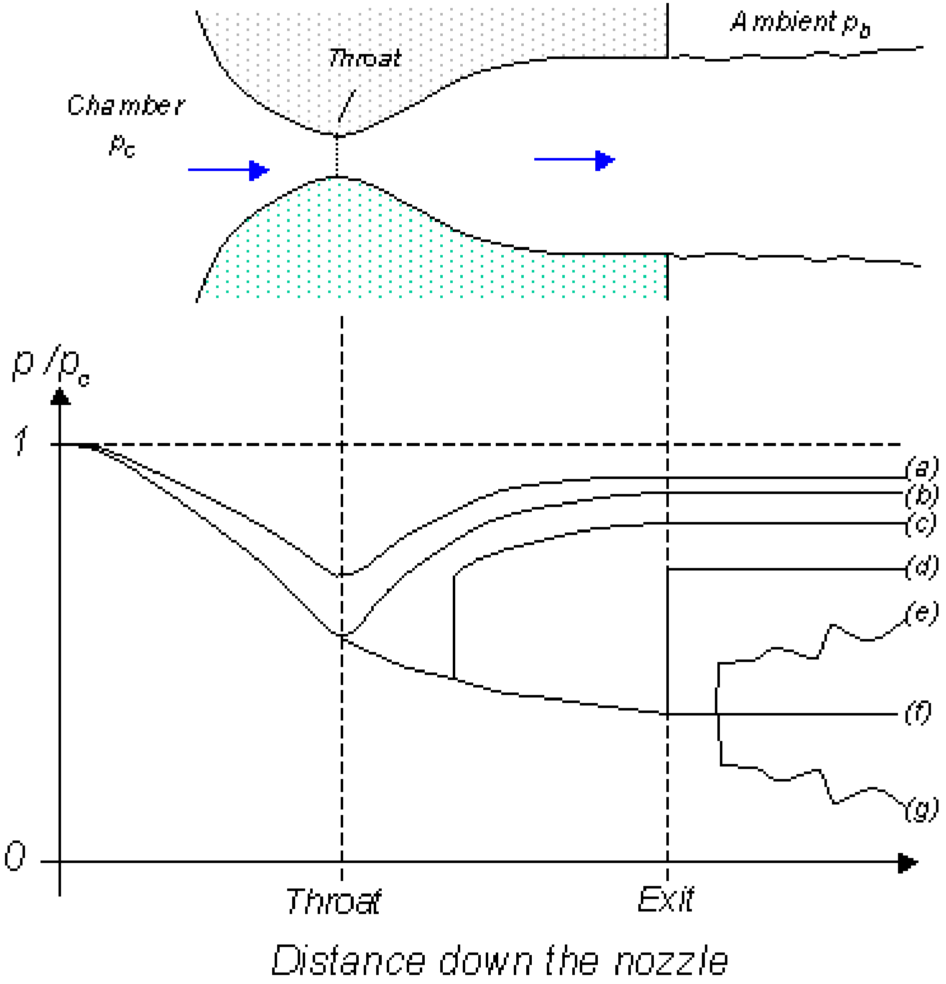
- Can this nozzle ever produce a supersonic flow in the converging portion?
- Can this nozzle ever produce a supersonic region in the exhaust? Is this region expanding or being compressed?
- Can this nozzle ever be over-expanded, such that $P_e < P_b$?

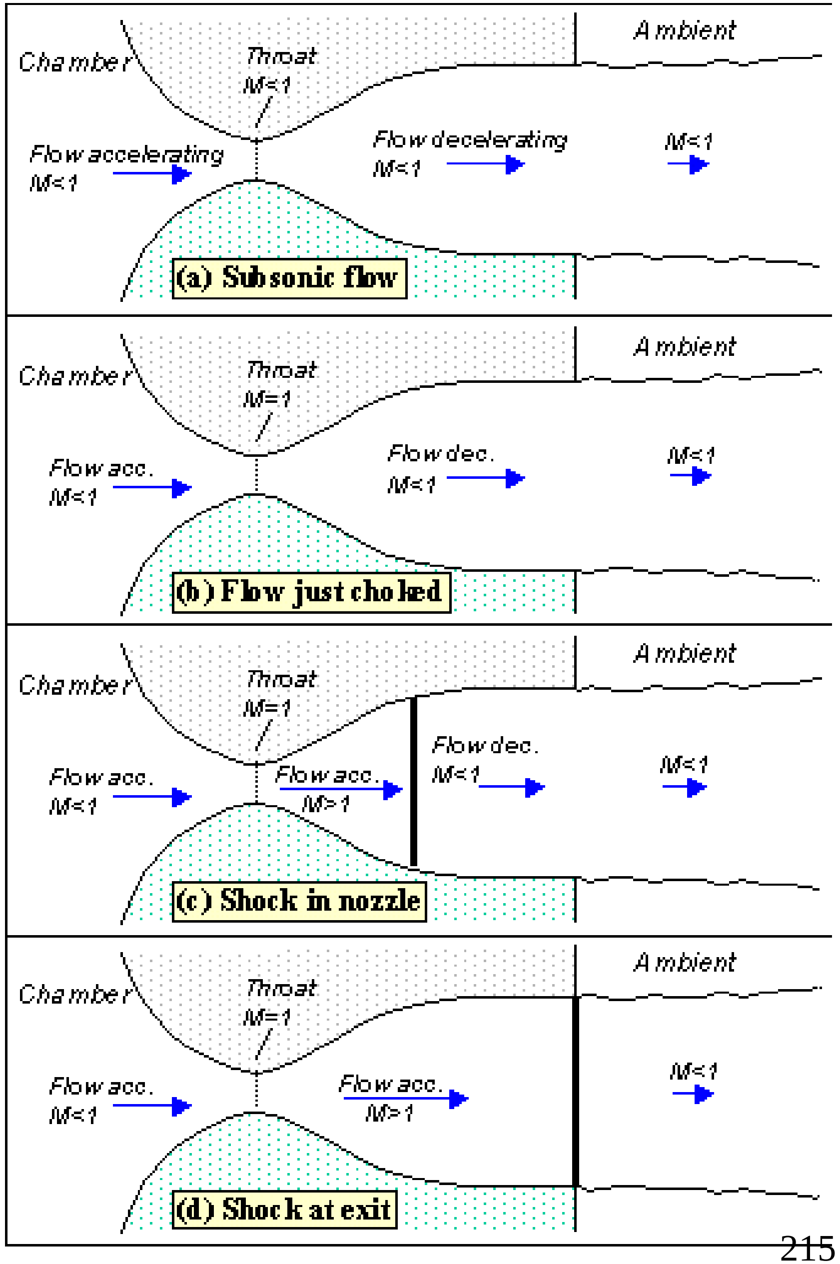
OPERATION OF CD NOZZLES

- Configuration for converging-diverging (CD) nozzle is shown below
- Gas flows through nozzle from region of high pressure (chamber) to low pressure (ambient)
- The chamber is taken as big enough so that any flow velocities are negligible
- Gas flows from chamber into converging portion of nozzle, past the throat, through the diverging portion and then exhausts into the ambient as a jet
- Pressure of ambient is referred to as back pressure

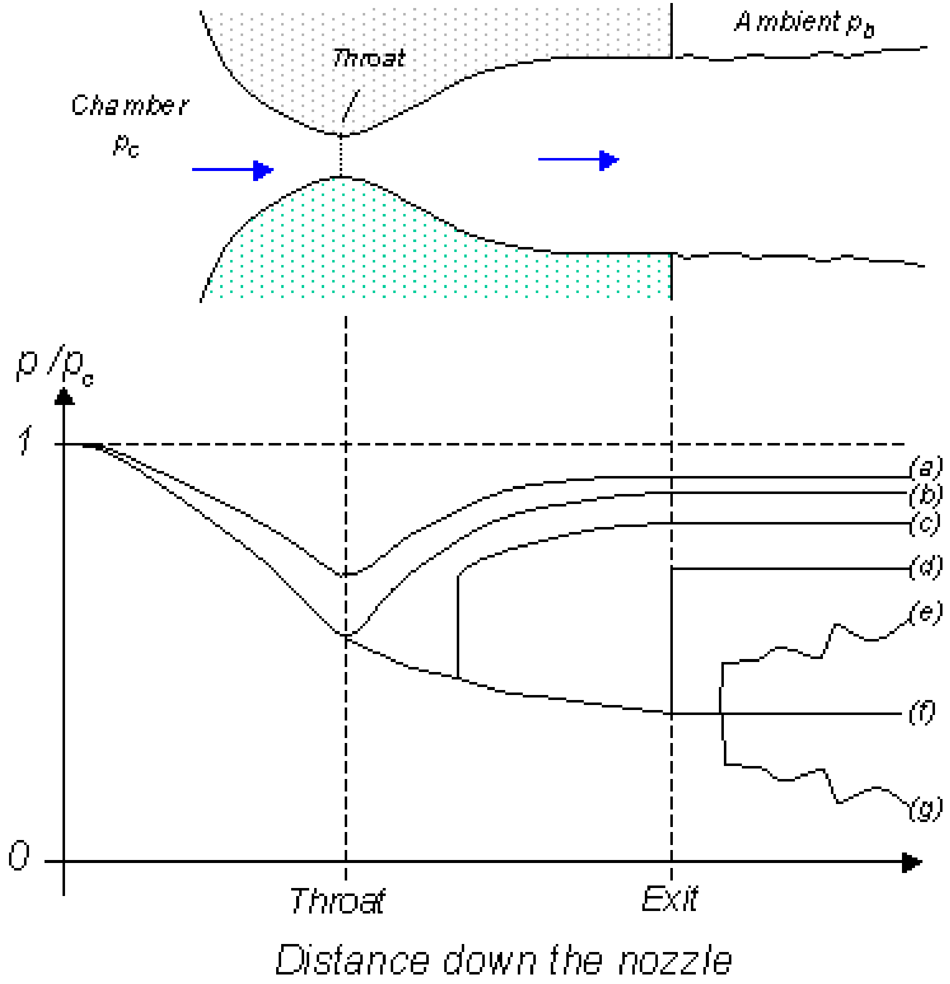


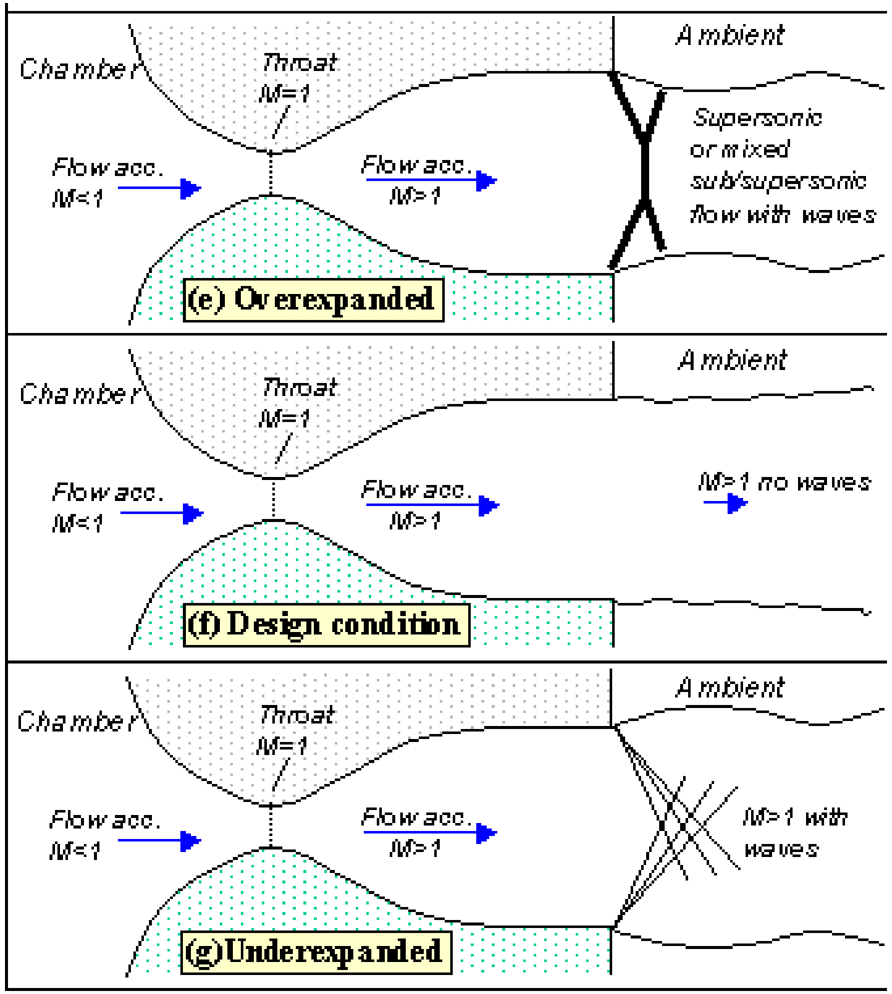
OPERATION OF CD NOZ





OPERATION OF CD NOZZLES



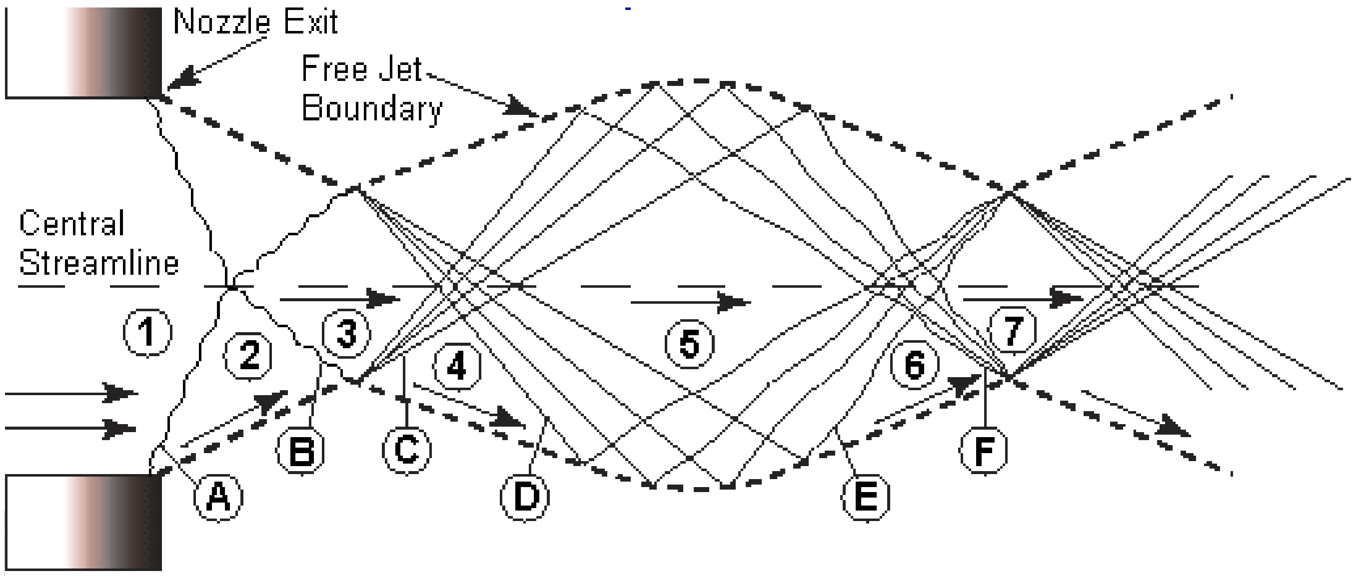


All practical rockets operate in regimes (e)-(g)

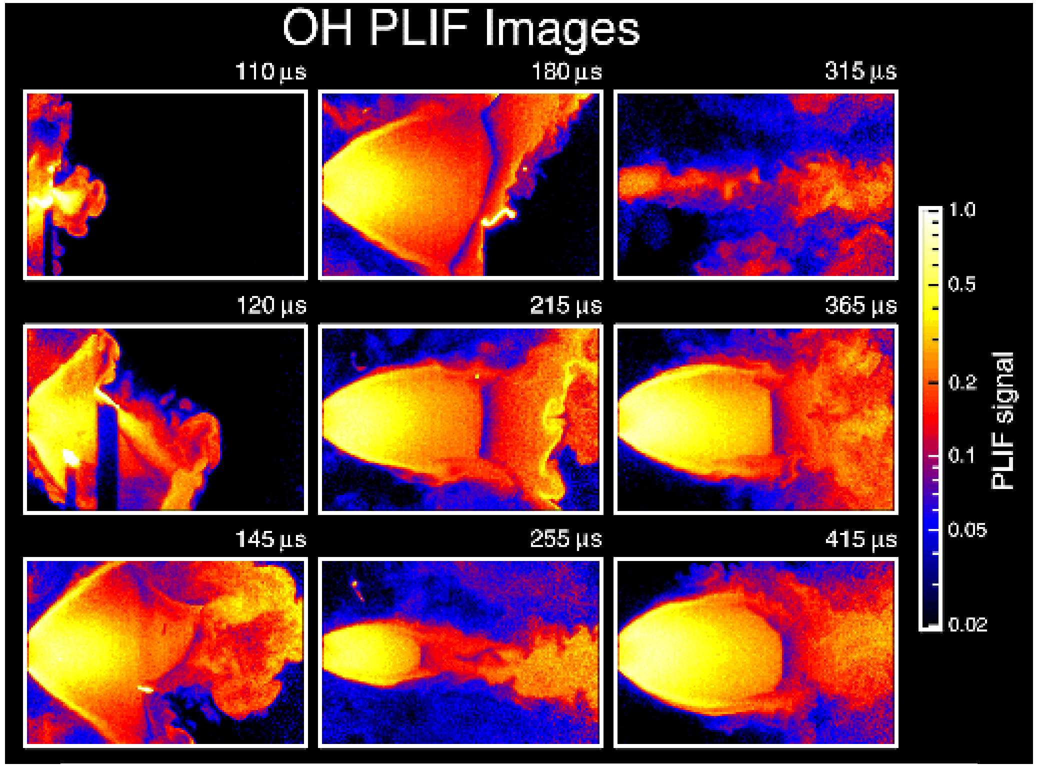
EXAMPLE: SPACE SHUTTLE MAIN ENGINE, ε ~ 77

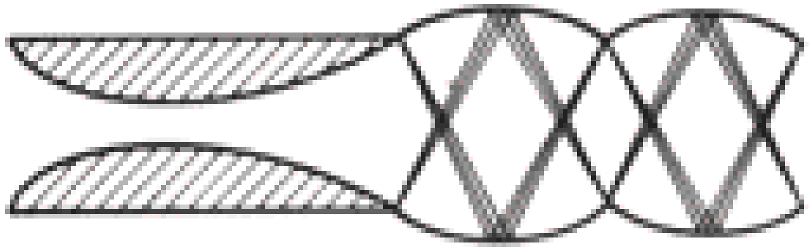
- Static pressure at exit of Space Shuttle Main Engine nozzle is considerably less than ambient pressure at sea level
- Mismatch in pressure gives rise to Mach "disc" in nozzle exhaust
- Extremely strong shock wave that creates a region of subsonic flow and produces a characteristic white luminescent glow
- Flow in picture is over-expanded (lift-off)

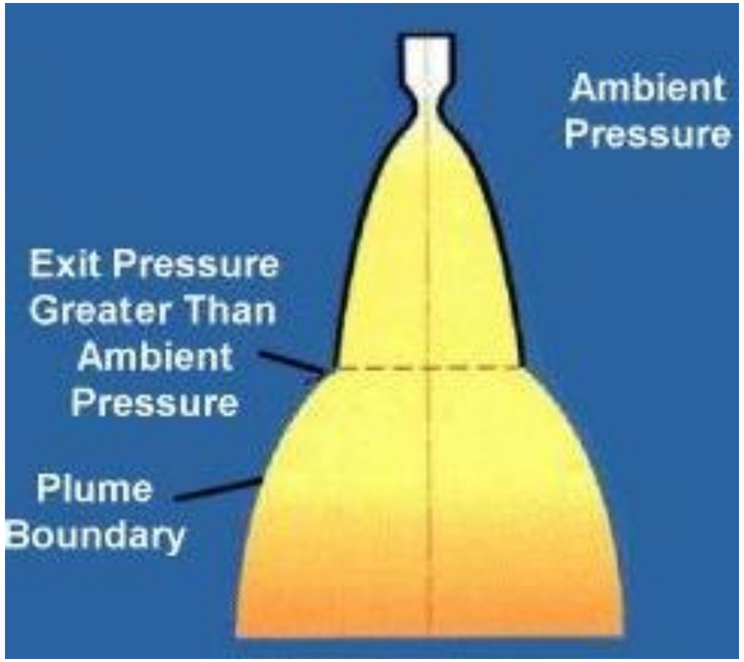
OVER-EXPANDED FLOW



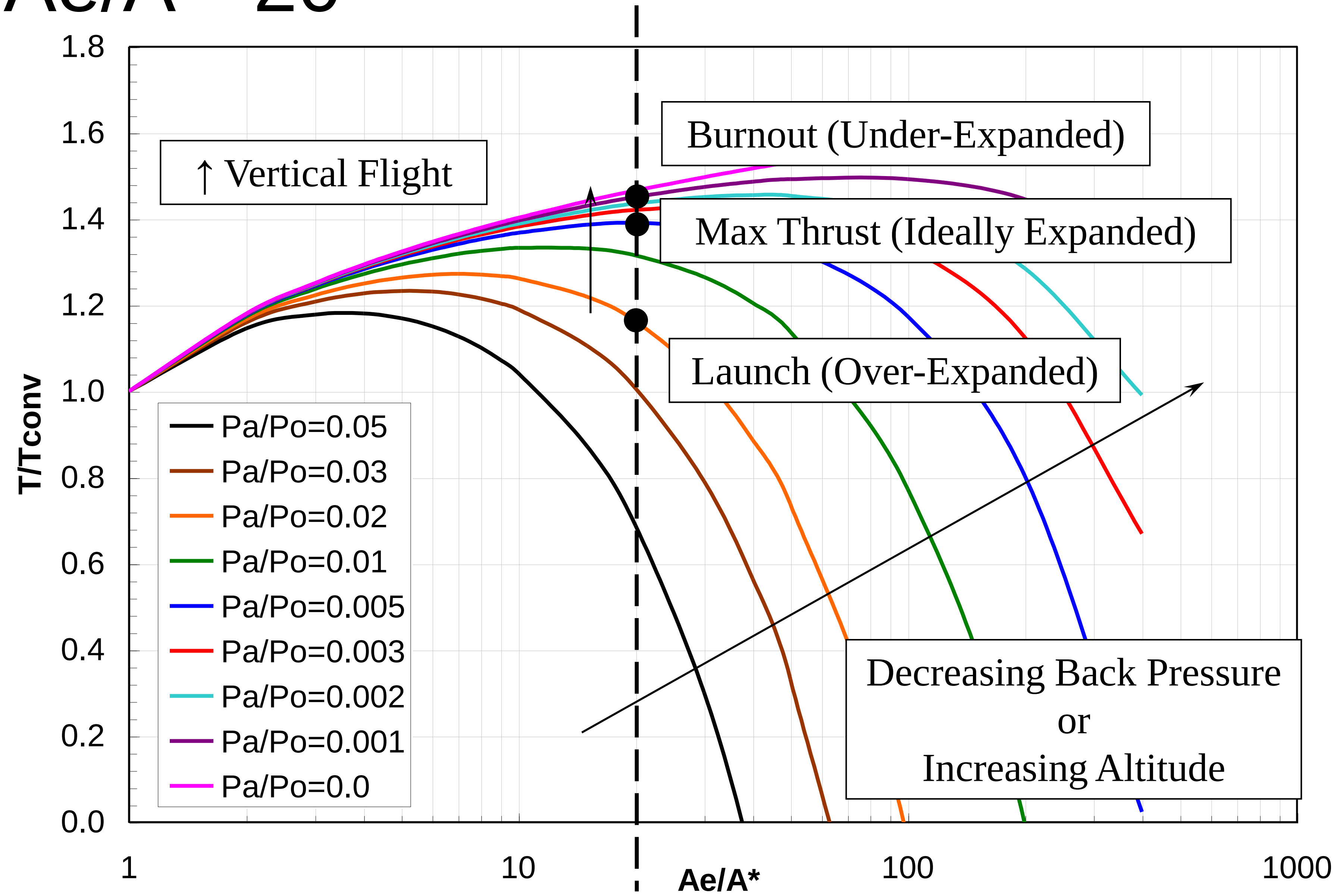
UNDER-EXPANDED FLOW



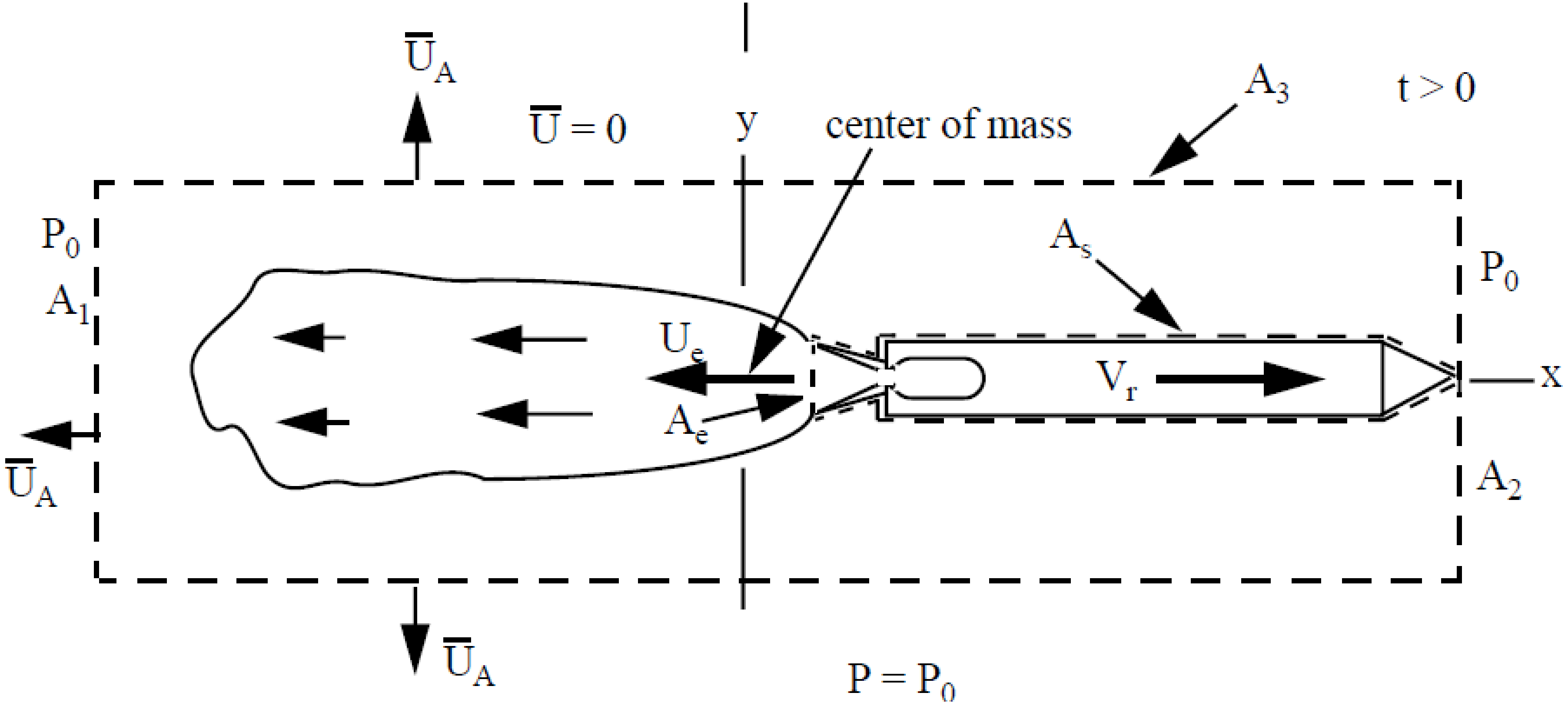




EXAMPLE: ROCKET LAUNCH
Ae/A*=20



MOMENTUM BALANCE FOR A ROCKET



Rocket mass X Acceleration = Thrust – Drag -gravity effect

$$M_r \frac{dV_r}{dt} = F_T - g\sin\theta - F_{drag}$$

Rocket Principles

- High pressure/temperature/velocity exhaust gases provided through combustion and expansion through nozzle of suitable fuel and oxidiser mixture.

- A rocket carries both the *fuel* and *oxidiser* onboard the vehicle whereas an *air-breather* engine takes in its oxygen supply from the atmosphere.

Criteria of Performance

- Specific to rockets only.
  - thrust
  - specific impulse
  - total impulse
  - effective exhaust velocity
  - thrust coefficient
  - characteristic velocity

Thrust (F)

For a rocket engine:

$$F_T = \dot{m}_{ejects} U_{ejects} + A_e \left(p_e - p_{amb} \right)$$

Where:

$\dot{m}$ = propellant mass flow rate

p_e = exit pressure, p_{aamb} = ambient pressure

U_{ejects} = exit plane velocity, A_e = exit area

Specific Impulse (I or I_{sp})

- The ratio of thrust / ejects mass flow rate is used to define a rocket's *specific impulse*-best measure of overall performance of rocket motor.

$$I_{sp} = \frac{F_T}{\dot{m}_{ejects}}$$

- In SI terms, the units of I are m/s or Ns/kg.
  - In the US:
  - with units of seconds - multiply by g (i.e. 9.80665 m/s^2) in order to obtain SI units of m/s or Ns/kg.
- Losses mean typical values are 92% to 98% of ideal values.

Total Impulse (I_{tot})

- Defined as: $$I_{total} = \int_0^{t_b} F_T \, dt$$

  where t_b = time of burning

- If F_T is constant during burn:

$$I_{total} = F_T \times t_b$$

- Thus the same total impulse may be obtained by either :
    - high F_T, short t_b (usually preferable), or
    - low F_T, long t_b
- Also, for constant propellant consumption (ejects) rate:

$$I_{total} = \frac{F_T}{\dot{m}_{ejects}} \times \dot{m}_{ejects} t_b$$

Effective Exhaust Velocity (c)

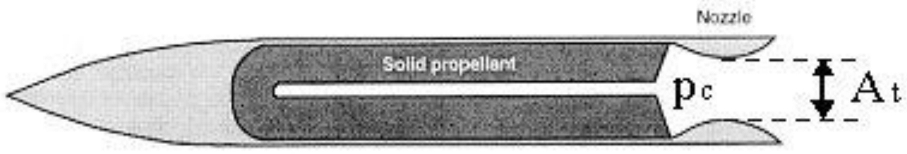
- Convenient to define an effective exhaust velocity (c), where:

$$F_T = \dot{m}_{ejects} c \qquad c = \frac{F_T}{\dot{m}_{ejects}} = I$$

$$c = U_e + \frac{(p_e - p_{amb}) A_e}{\dot{m}_{ejects}}$$

Thrust Coefficient (C_F)

- Defined as:



$$C_F = \frac{F_T}{P_c A_t}$$

where p_c = combustion chamber pressure,

A_t = nozzle throat area

- Depends primarily on (p_c/p_a) so a good indicator of *nozzle* performance – dominated by pressure ratio.

Characteristic Velocity (c*)

- Defined as: $$C^* = \frac{P_c A_t}{\dot{m}_{ejects}}$$

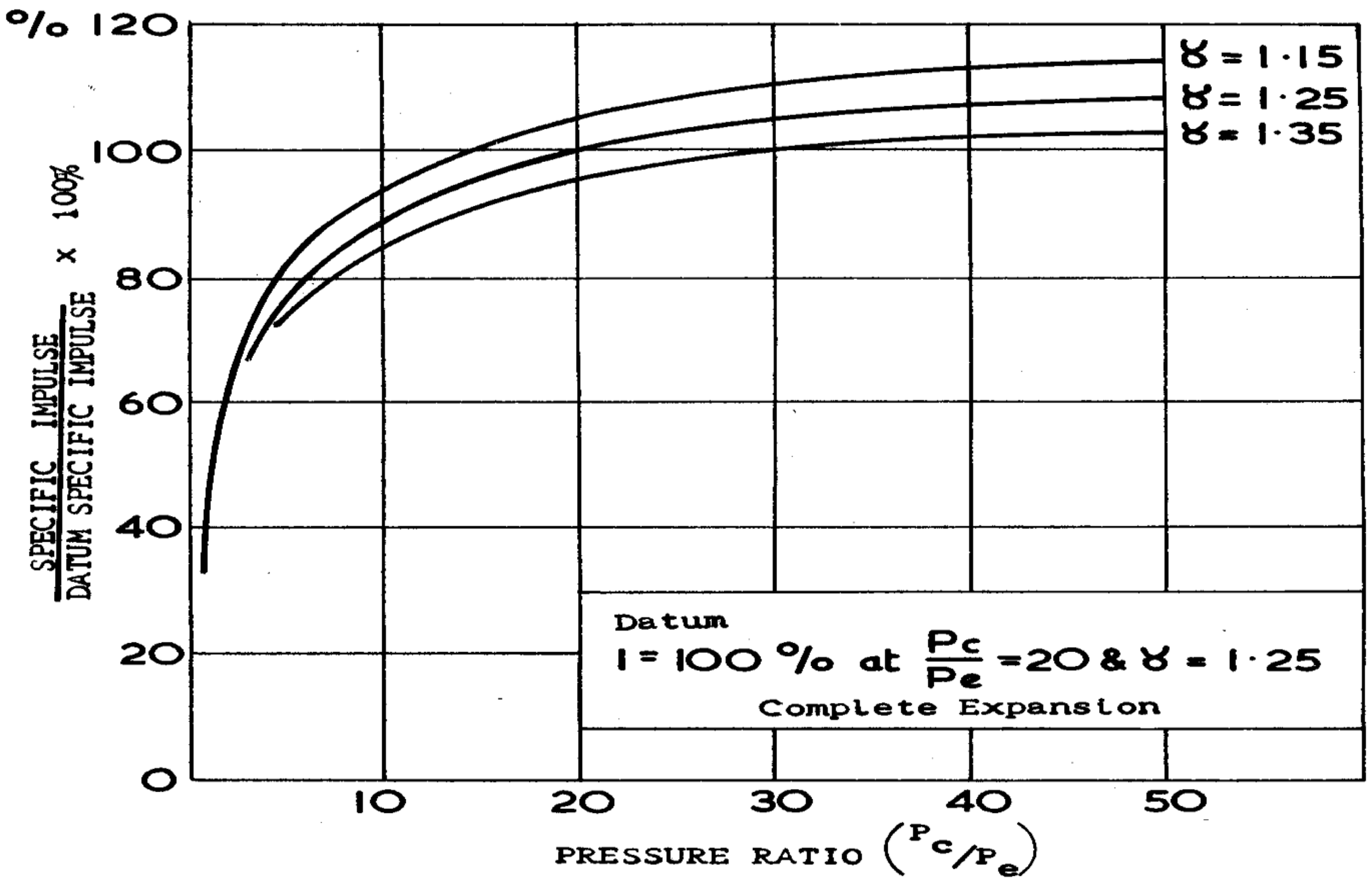
•Calculated from standard test data.

• It is independent of nozzle performance and is therefore used as a measure of *combustion* efficiency – dominated by T_c (combustion chamber temperature).

Thermodynamic Performance - Thrust

- Parameters affecting thrust are primarily:
  - mass flow rate
  - exhaust velocity
  - exhaust pressure
  - nozzle exit area

Thermodynamic Performance - Specific Impulse

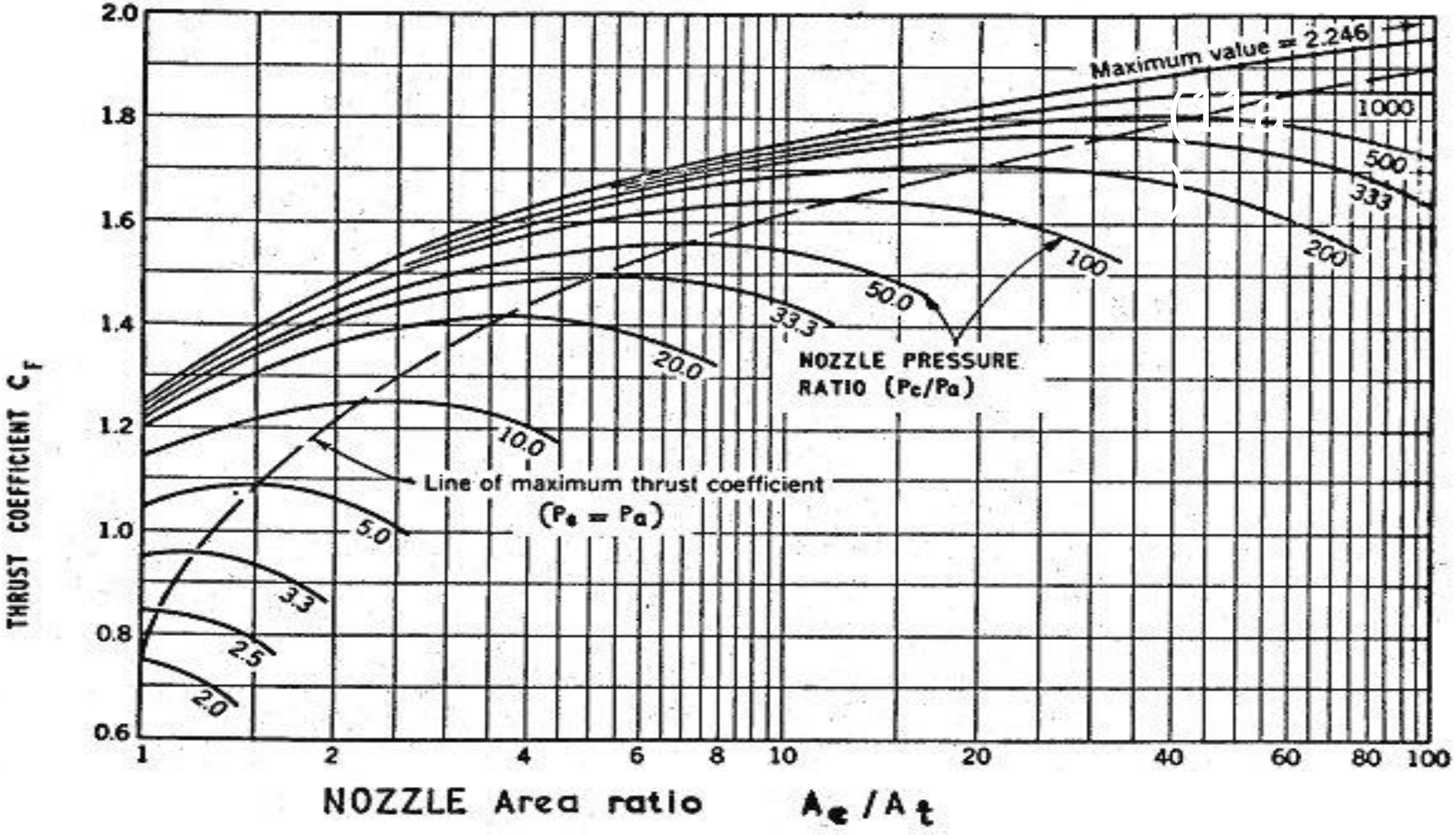


Thermodynamic Performance - Specific Impulse

<u>Variable Parameters - Observations</u>

- Strong pressure ratio effect - but rapidly diminishing returns after about 30:1.
- High T_c value desirable for high I - but gives problems with heat transfer into case walls and dissociation of combustion products – practical limit between about 2750 and 3500 K, depending on propellant.
- Low value of molecular weight desirable – favouring use of hydrogen-based fuels.
- Low values of γ desirable.

Thrust Coefficient (C_F)



Thrust Coefficient (C_F) - Observations

- More desirable to run a rocket under-expanded (to left of optimum line) rather than over-expanded.
- Uses shorter nozzle with reduced weight and size.
- Increasing pressure ratio improves performance but improvements diminish above about 30/1.
- Large nozzle exit area required at high pressure ratios – implications for space applications.

Actual Rocket Performance

- **Performance may be affected by any of the following deviations to simplifying assumptions:**
  - Properties of products of combustion vary with static temperature and thus position in nozzle.
  - Specific heats of combustion products vary with temperature.
  - Non-isentropic flow in nozzle.
  - Heat loss to case and nozzle walls.
  - Pressure drop in combustion chamber due to heat release.
  - Power required for pumping liquid propellants.
  - Suspended particles present in exhaust gas.

Internal Ballistics

- Liquid propellant engines store fuel and oxidizer separately - then introduced into combustion chamber.
- Solid propellant motors use propellant mixture containing all material required for combustion.
- Majority of modern use solid propellant rocket motors, mainly due to simplicity and storage advantages.
- *Internal ballistics* is study of combustion process of solid propellant.

Solid Propellant Combustion

- Combustion chamber is high pressure tank containing propellant charge at whose surface burning occurs.

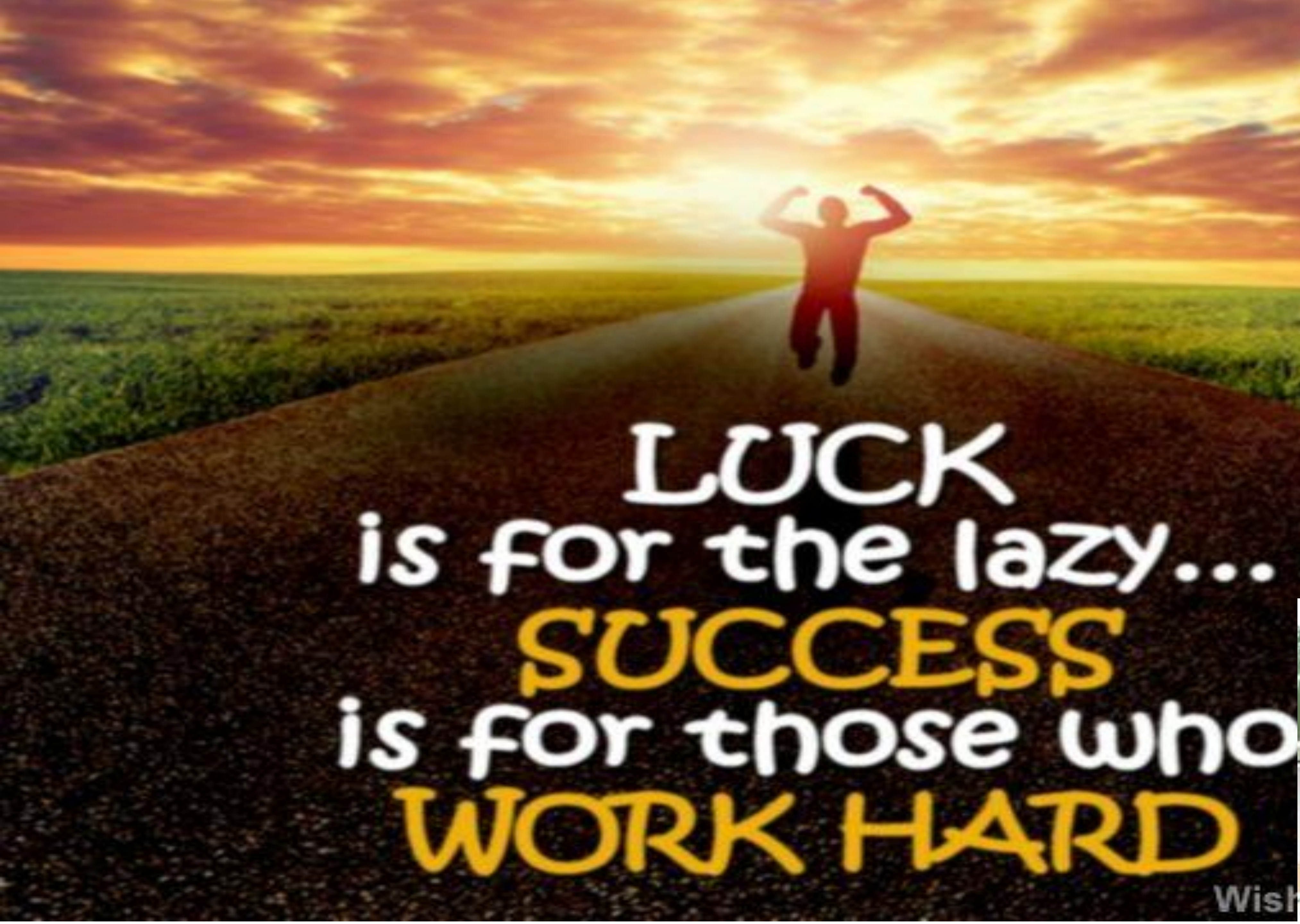
- No arrangement made for its control – charge ignited and left to itself so must *self-regulate* to avoid explosion.

- Certain measure of control provided by charge and combustion chamber design and with inhibitor coatings.

CHEMICAL ROCKET ENGINES

presence of liquid drops and solid particles- two phase flow, losses, efficiency.

Computing rocket engine performance- theoretical, delivered performance, performance at standard operating conditions, guaranteed minimum performance.


LUCK
is for the lazy...
SUCCESS
is for those who
WORK HARD


90 minutes exam
The first 85 minutes
The last 5 minutes

Rocket engine performance

RPA is a program which calculates chemical equilibrium product concentrations from specified set of propellant components, determines thermodynamic properties for the product mixture, and calculates theoretical as well as estimated test (actual) nozzle performance.

The method used for obtaining equilibrium compositions is the minimization of Gibbs free energy. Applied to the combustion chamber, the method allows:

•to determine the equilibrium product concentrations from adiabatic, isenthalpic combustion of two or more reactants

•to determine the equilibrium product concentrations from monopropellant decomposition

•to calculate the isentropic quasi-one-dimensional nozzle flow for both shifting and frozen equilibrium flow models

To obtain the rocket performance, the tool calculates conditions at several sections of the chamber. It always includes the calculation of combustion (injector section) and nozzle throat parameters, as well as nozzle exit parameters, defined by either nozzle exit pressure p_e, expansion pressure ratio p_t/p_e, or expansion area ratio A_e/A_t.

The user can force the program to calculate the performance with respect of pressure drop between injector and nozzle inlet, defining such parameters as a chamber mass flux or a nozzle inlet contraction area ratio A_c/A_t.

- Default nozzle flow model is a shifting equilibrium: combustion products continue to react and reach chemical equilibrium at each temperature and pressure conditions along the nozzle.
- The user can trigger the "freezing" of nozzle flow composition downstream of the throat.
- In this case, it is assumed that composition is "frozen" (infinitely slow reaction rates) during expansion along the nozzle. The location of "freezing" nozzle section is defined by either pressure ratio p_t/p_{fr}, or area ratio A_{fr}/A_t.
- In case of overxpanded nozzle flow, the tool calculates the performance with respect of flow separation.

Electric Propulsion

Unit 5

Electric Propulsion (EP)

- Electric propulsion (EP) encompass a broad variety of strategies for achieving very high exhaust velocities in order to reduce the total propellant burden and corresponding launch mass of present and future space transportation systems.

Limitations of Chemical Rockets

- Chemical rocket: exhaust ejection velocity intrinsically limited by the propellant-oxidizer reaction
- Larger velocity increment of the spacecraft could be obtained only with a larger ejected mass flow.
- Mission practical limitation: exceedingly large amount of propellant that needs to be stored aboard

ELECTRIC PROPULSION OVERVIEW

- **Chemical rockets: Energy limited**
  - Limited to energy contained within propellants they carry
  - High power (W=J/s) due to rapid conversion of energy
- **Electric systems: Power limited**
  - No limit to energy added to propellant (in theory)
  - However, rate of conversion of energy to power limited by mass of conversion equipment which must be carried, $M_{electrical}$
  - Possible to achieve very high exhaust velocities at cost of high power consumption
- **Electric propulsion broadly defined as acceleration of propellants by:**
  - Electrical heating (thermal rocket)
  - Electric body forces
  - Magnetic body forces

These techniques group broadly into three categories:

- ***Electrothermal propulsion***, wherein the propellant is electrically heated, then expanded thermodynamically through a nozzle.

- ***Electrostatic propulsion***, wherein ionized propellant particles are accelerated through an electric field.

- ***Electromagnetic propulsion***, wherein current driven through a propellant plasma interacts with an internal or external magnetic field to provide a stream-wise body force.

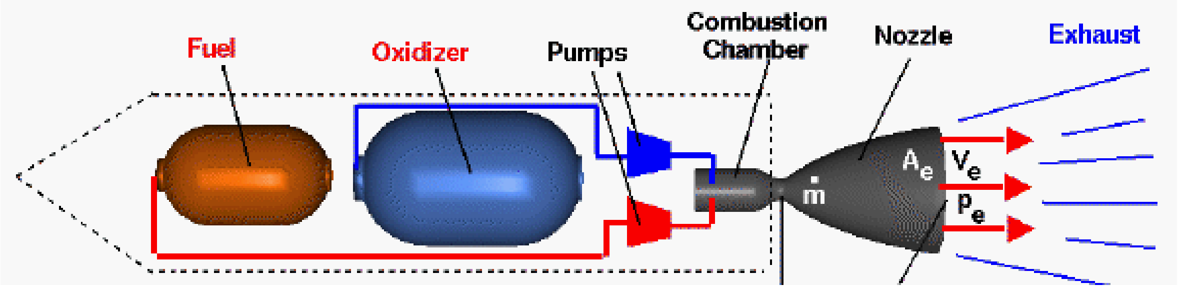
COMMENTS / CHALLENGES

- **Electrothermal: Use electricity to heat propellant (resistojet, arcjet)**
  - Limited by wall temperature heating
  - Specific impulse not be much greater than H_2-O_2 chemical rocket
    - Unless pure H_2 as propellant

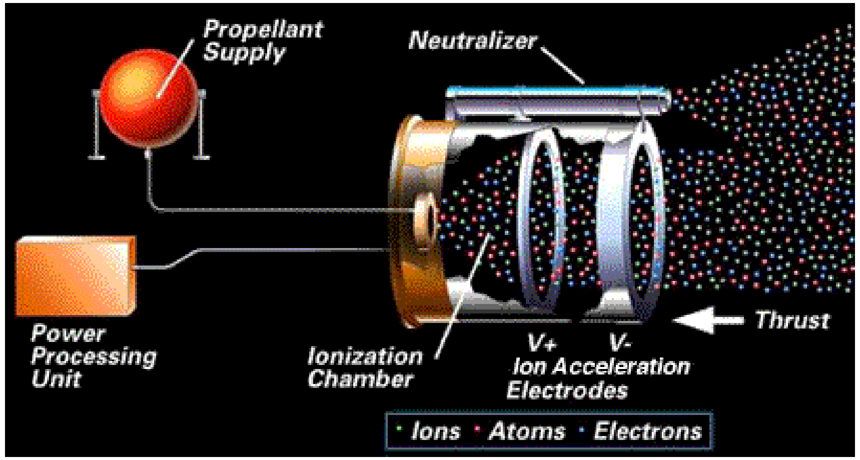
- **Electrostatic and Electromagnetic not influenced in this way**
  - Propellant is not heated
  - **Ue (or Isp) not set by temperature**
  - Gas is ionized and accelerated by electric and magnetic fields
  - Higher exit velocities, but very limited by power
  - Low thrust
  - Long mission times

COMPARISON EXAMPLE: ORDER OF MAGNITUDE

- **Liquid Rocket: Energy Limited**
  - Ue ~ 4,500 m/s
  - Isp ~ 450 s
  - Energy ~ 100 GJ
  - Power ~ 300 MW
  - Thrust ~ 2,000,000 N



- **Ion Rocket (Electrostatic): Power Limited**
  - Ue ~ 30,000 m/s
  - Isp ~ 3,000 s
  - Energy ~ 1,000 GJ
  - Power ~ 1 kW
  - Thrust ~ 0.1 N



DEFINITIONS

$$V = V_e \ln(R) = V_e \ln\left(\frac{M_o}{M_f}\right)$$

General form of rocket equation
Chemical (thermal) and electric rockets

Remember: Propulsive force developed by an electric thruster same physical origin as chemical thruster: It is momentum transferred to propellant

$$\frac{M_o}{M_f} = \frac{M_{payload} + M_{structure} + M_{propellant} + M_{electrical}}{M_{payload} + M_{structure} + M_{electrical}}$$

$M_{electrical}$ is mass of electrical conversion equipment that rocket must carry

Mass is a significant part of rockets total weight (depends strongly on power)

SOME TYPICAL POWER (W) VALUES

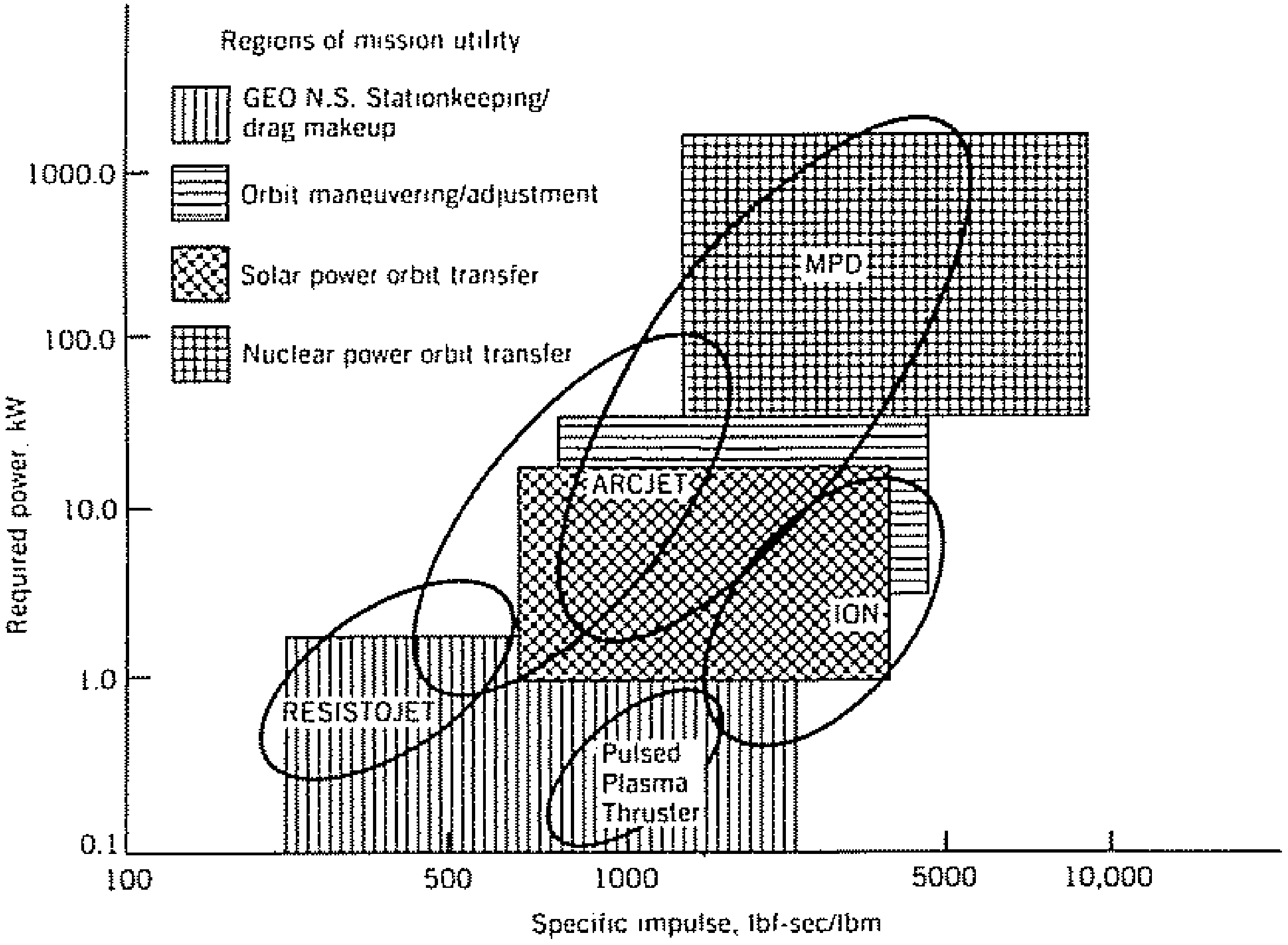
- **Light Bulb**
  - 40 – 100 W

- **Resistojets**
  - ~ 500 W
  - ~ 200 mN of thrust

- **1970’s: SERT II (Space Electrical Rocket Test II) Ion Propulsion (electrostatic)**
  - Photovoltaic energy source
  - ~ 850 W
  - 30 mN of thrust, Isp ~ 4,200 s

- **Nuclear-Electric generator (see next slides)**
  - ~ 100 kW range (up to MW range, but strong dependence on mass)
  - 600,000 pounds of chemical fuel = 1 pound of nuclear fuel

POWER REQUIREMENTS



Overview of the approximate regions of applications of different electrical propulsion systems in terms of power and specific impulse.

TYPICAL VALUES: SPECIFIC MASS (α = kg/kW)

- **Lightweight (Advanced) Solar Arrays**
  - Small: 10 kg/kW
  - Large: 300 kg/kW
  - Becomes prohibitively large as power requirement increases

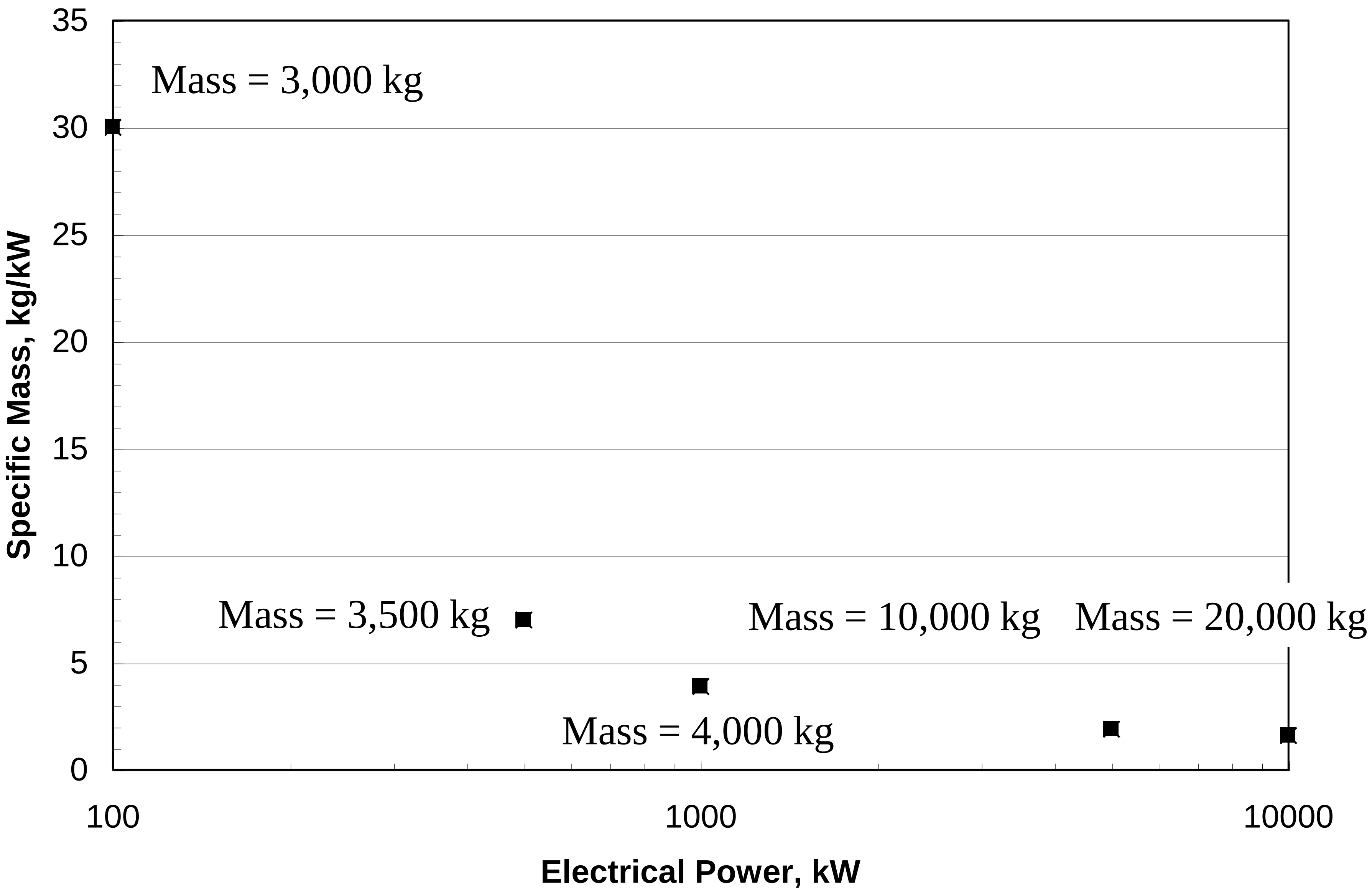
- **Photovoltaic power supply**
  - ~ 400 kg/kW

- **RTG (Radioisotope Thermoelectric Generator)**
  - 125 kg/kW – 500 kg/kW
  - New Horizons (Pluto) Mission (and over 25 others)

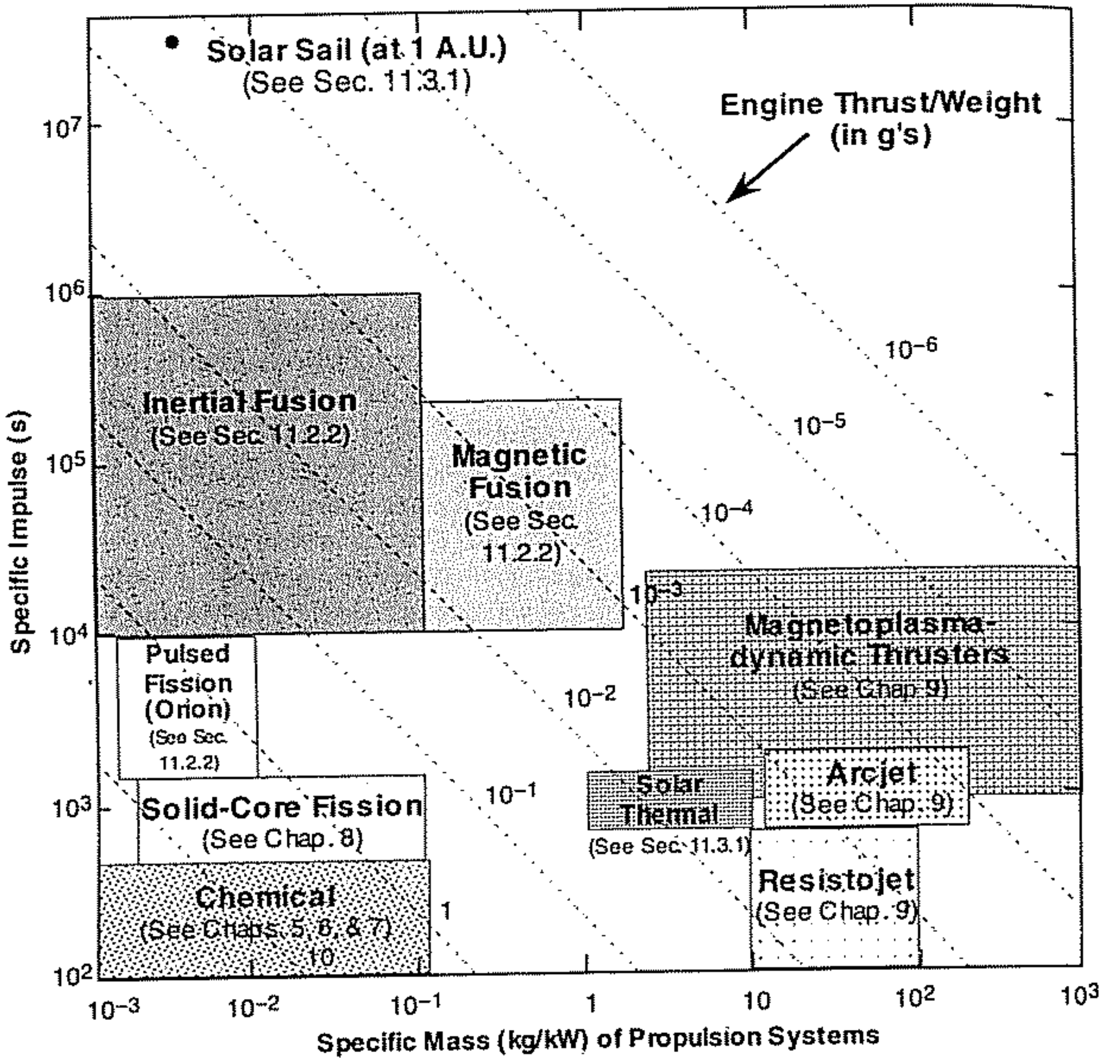
- **100 kW lunar nuclear reactor**
  - 30 kg/kW

- **Fuel cells (wide range of values)**
  - 50 – 500 kg/kW

SPECIFIC MASS (kg/kW) OF NUCLEAR POWER PLANTS FOR SPACE APPLICATIONS



ROCKET PERFORMANCE COMPARISON



DEFINITIONS

P = Power of Conversion Equipment

$$\alpha \equiv \frac{M_{electrical}}{P}\left[\frac{kg}{W}\right]$$

α = **<u>Specific Mass</u>** of Electric Conversion Equipment
α = **<u>Mass-to-Power Ratio</u>**

$$\beta \equiv \frac{P}{M_{electrical}}\left[\frac{W}{kg}\right]$$

β = **<u>Specific Power</u>** of Electric Conversion Equipment
β = **<u>Power-to-Mass Ratio</u>**

$$\eta = \frac{\frac{1}{2}\dot{m}V_e^2}{P}$$

η = efficiency of power conversion
η = kinetic energy flux in exhaust/electric power supplied

$$\dot{m} = \frac{M_{propellant}}{t_b}$$

t_b = ‘burn’ time (remember that propellant is not actually burning, but is being used)

ELECTRIC VEHICLES: GENERAL TRENDS

- Power does not diminish during flight
  - Propellant diminishes during flight
- Power proportional to mass of power supply
  - Mass of rocket depends on mass of power supply and hence on power
- Mass flow rate usually constant

- Exhaust velocity no longer a free parameter
  - Fixed by power and mass flow rate
  - Mass flow rate related to burn time and mass of propellant

- If we want to increase exhaust velocity or mass flow rate
  - Requires an increase in power supplied to thruster (larger $M_{electrical}$)
- If we want to increase mass flow rate
  - Leads to shorter burn time

ELECTRIC VEHICLES: GENERAL TRENDS

- Importance of burn time
  - Defines rate at which propellant is used and hence power that has to be applied
  - For same mass of propellant, short burn time requires higher power and heavier power supply

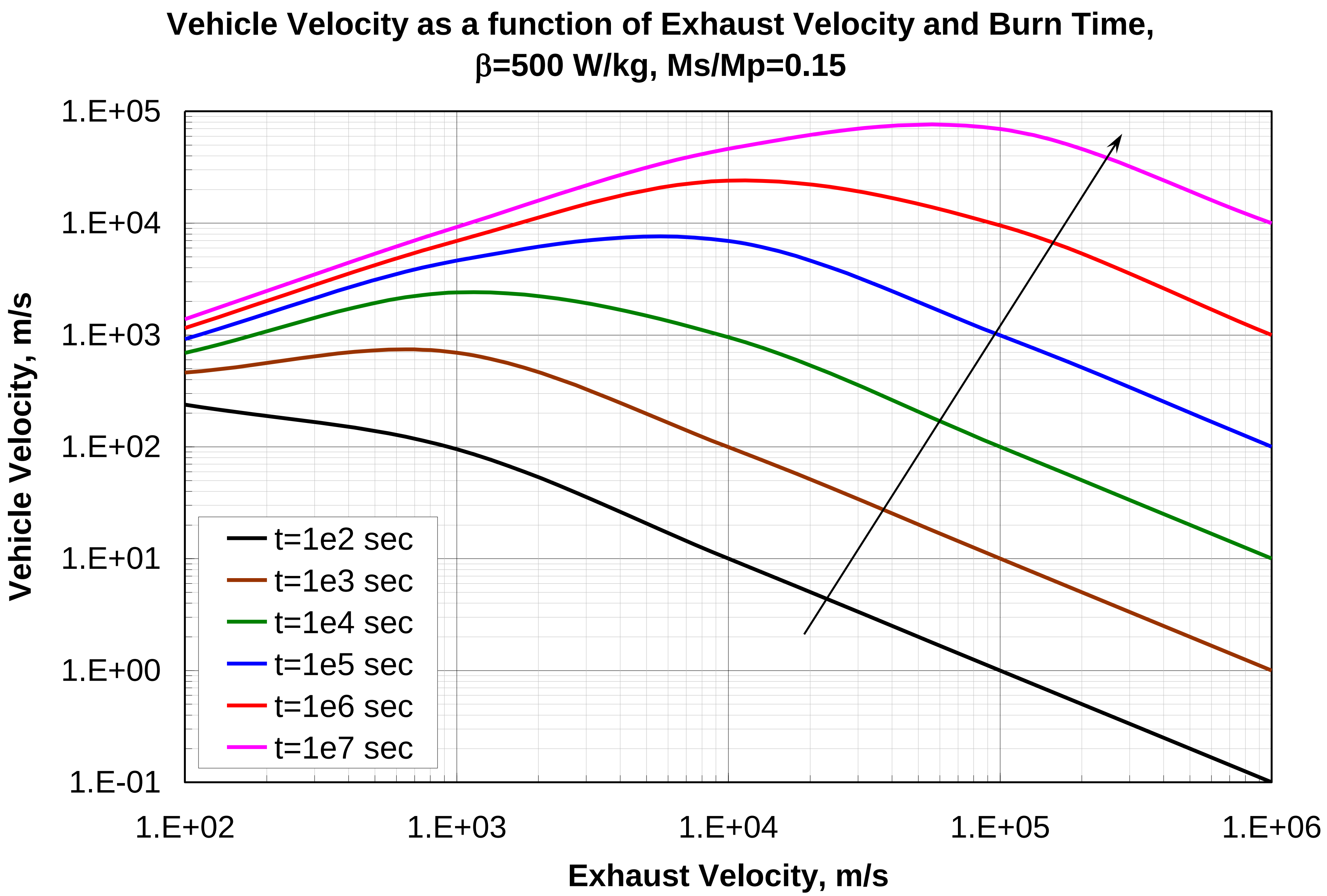
- Mass ratio ($R = M_0/M_{final}$) for a given propellant mass decreases as exhaust velocity increases, due to increased power supply mass
  - Not true for chemical rocket, in which exhaust velocity and mass ratio, are in principle, independent

- For electric vehicle, increase in exhaust velocity requiring increase in power could result in NO improvement in vehicle velocity, due to increased mass
  - Not true for chemical rocket

➢ **There Is An Optimum Exhaust Velocity For Electric Rockets**

PARAMETRIC INVESTIGATION

PARAMETRIC INVESTIGATION (t_b)



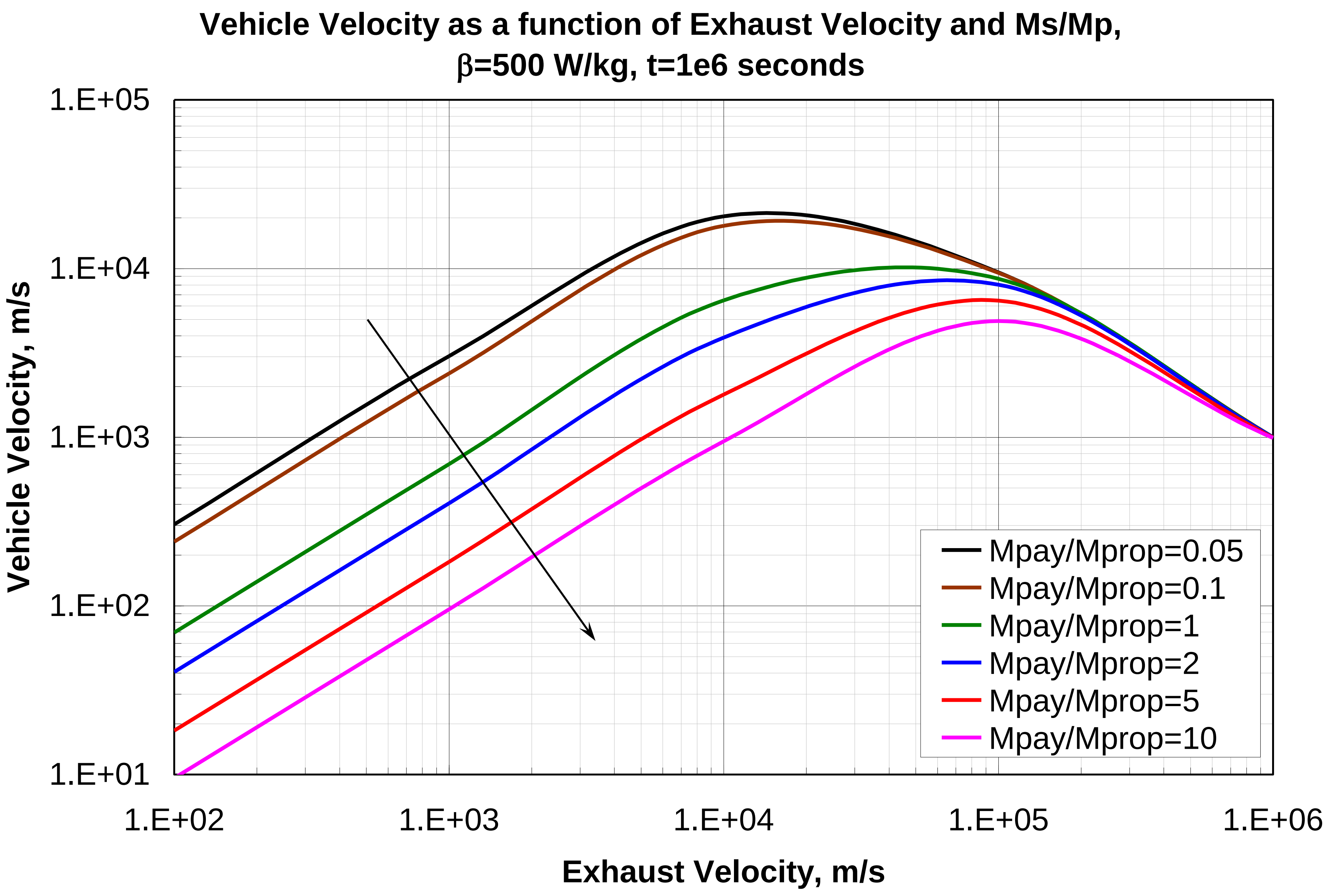
KEY POINTS (PARAMETER: t_b)

- Plot shows velocity of vehicle as a function of exhaust velocity, assuming a fixed relationship between exhaust velocity and power supply mass
- Power-to-Mass ratio, β, is fixed at 500 W/kg (corresponds to α = 2 kg/kW)
- Ratio of structural mass to propellant mass, M_s/M_p, is fixed at 0.15
- Burn time, t_b, is the parameter being varied
- Vehicle velocity peaks for a certain value
  - Does not increase monotonically
- Increasing burn time increases peak value of vehicle velocity and optimal exhaust velocity

- Decrease in vehicle velocity above a certain point is due to increasing relative mass of power supply (and hence a reduction in mass ratio)

- Exhaust velocity for a given power depends inversely on mass flow rate
    - Low mass flow rates or long burn times are beneficial
- Thrust is inversely proportional to burn time
    - Long burn times and high exhaust velocities imply low thrust

PARAMETRIC INVESTIGATION (M_{struct}/M_{prop} or M_{pay}/M_{prop})

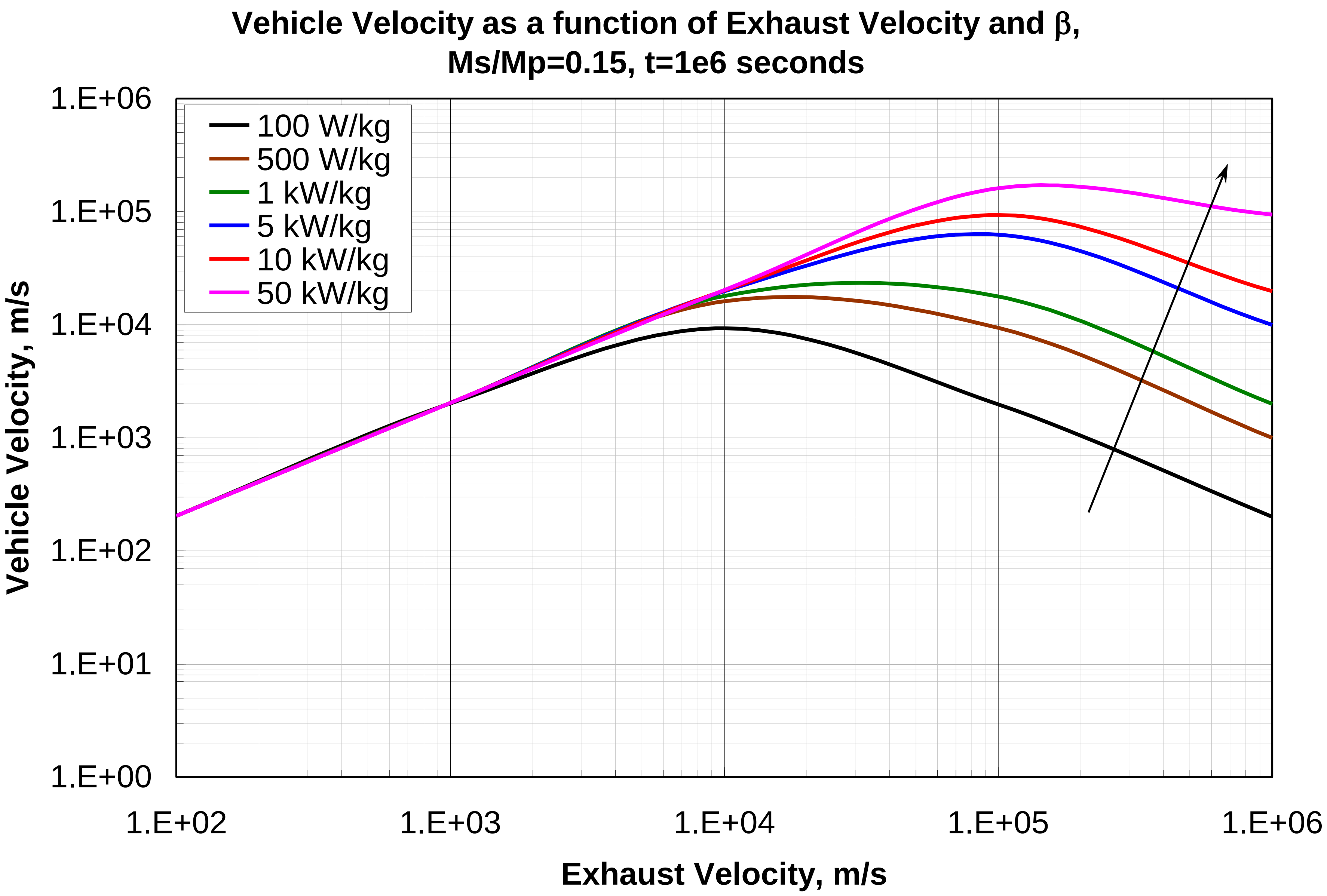


KEY POINTS (PARAMETER: M_{struct}/M_{prop})

- Examines vehicle velocity as a function of ratio of structural mass (or payload) to propellant mass, M_s/M_p as a parameter
- Burn time is fixed a 1 million seconds (~12 days)
- Power-to-Mass ratio fixed at 500 W/kg

- Vehicle velocity increases as the fraction of propellant increases
- Peak vehicle velocity moves to higher exhaust velocities as the payload mass increases
  - Confirms that high exhaust velocity is advantageous for long missions

PARAMETRIC INVESTIGATION (β)



KEY POINTS (PARAMETER: β)

- Examine role of Power-to-Mass ratio of power supply
- Burn time fixed at 1 million seconds
- Payload-to-Propellant mass ratio set to 0.15

- Increasing Power-to-Mass ratio increases the vehicle velocity
- Peak velocity moves towards higher exhaust velocities as power-to-mass ratio increases
- For very large velocity increments, a high power-to-mass ratio must be matched by high exhaust velocity
- Importance of exhaust velocity
  - High exhaust velocity allows much higher payload-to-propellant mass ratios
  - Power-to-mass ratio of power supply is crucial in obtaining best performance

EXAMPLE DEVICES

- Electrothermal
  - Resistojet, Arcjet
- Electrostatic
  - Ion, Hall Effect
- Electromagnetic
  - MPD (magnetoplasmadynamic), PPT (pulsed plasma thrusters)

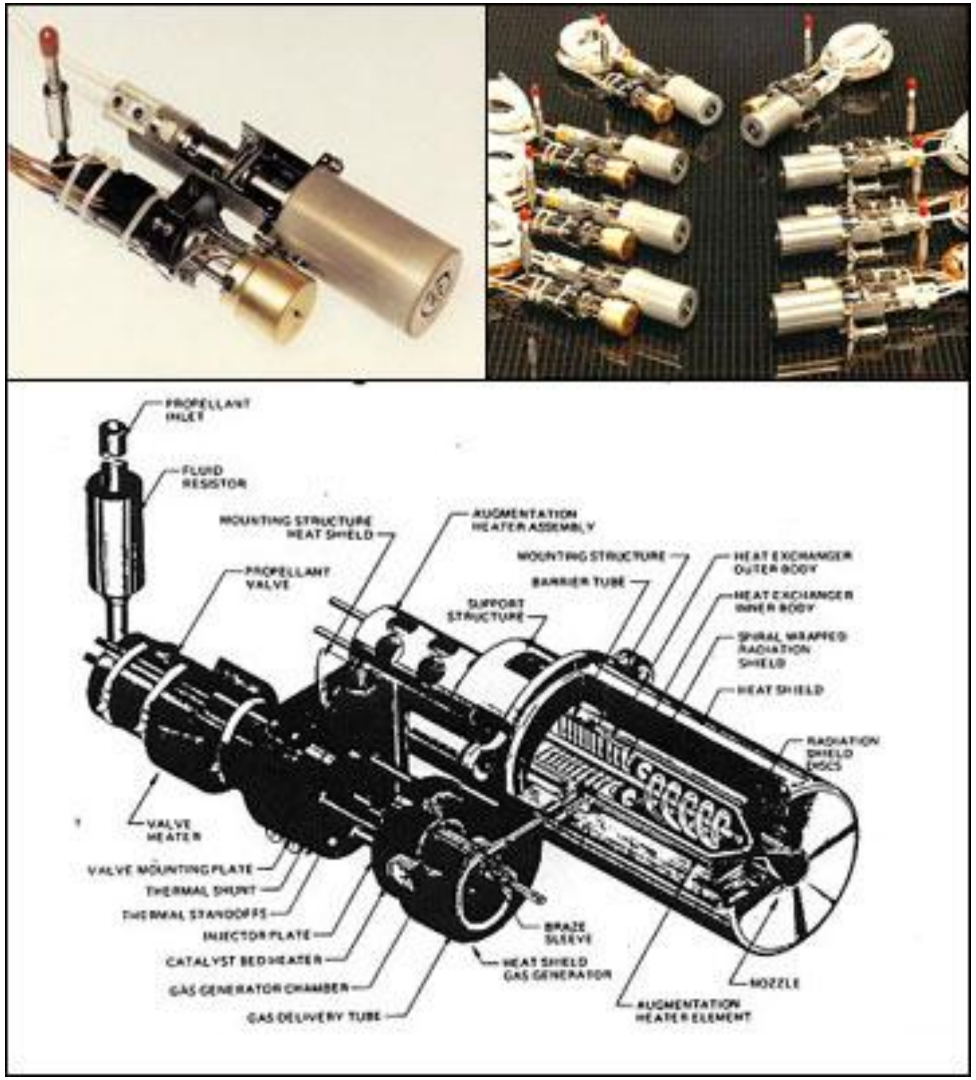
TABLE 19–1. Typical Performance Parameters of Various Types of Electrical Propulsion Systems[a]

Type	Thrust Range (milliNewton)	Specific Impulse (sec)	Thruster Efficiency[b] (%)	Thrust Duration	Typical Propellants	Specific Power (W/milliNewton)
Resistojet (thermal)	2–100	200–300	65–90	Months	NH_3, N_2H_4, H_2	0.5–6
Arcjet (thermal)	2–700	400–1500	40–50	Months	H_2, N_2, N_2H_4, NH_3	2–3
Ion contact	0.001–100	1500–5000	60–80	Months	Cs, Hg	10–70
Ion bombardment	0.01–200	1500–5000	60–80	Months	Xe, Hg, Ar, Kr	10–70
Solid pulsed plasma	0.05–10	1000–2000	20–30	Years	Teflon	10–50
MPD arcjet	0.001–2000	1000–8000	30–50	Weeks	Ar, Xe, H_2	100
Hall effect MPD	0.01–2000	3000–5000	30–50	Months	Cs, Bi, Ar, N_2, Xe, H_2	100
Monopropellant rocket	30–100,000	200–230	87–97	Hours or minutes	N_2H_4	—

[a] Listed for comparison only

[b] $\text{Efficiency} = \dfrac{\text{thrust-power output}}{\text{electrical-power input}}$

ELECTROTHERMAL: RESISTOJET



Aerojet MR-501B resistojet

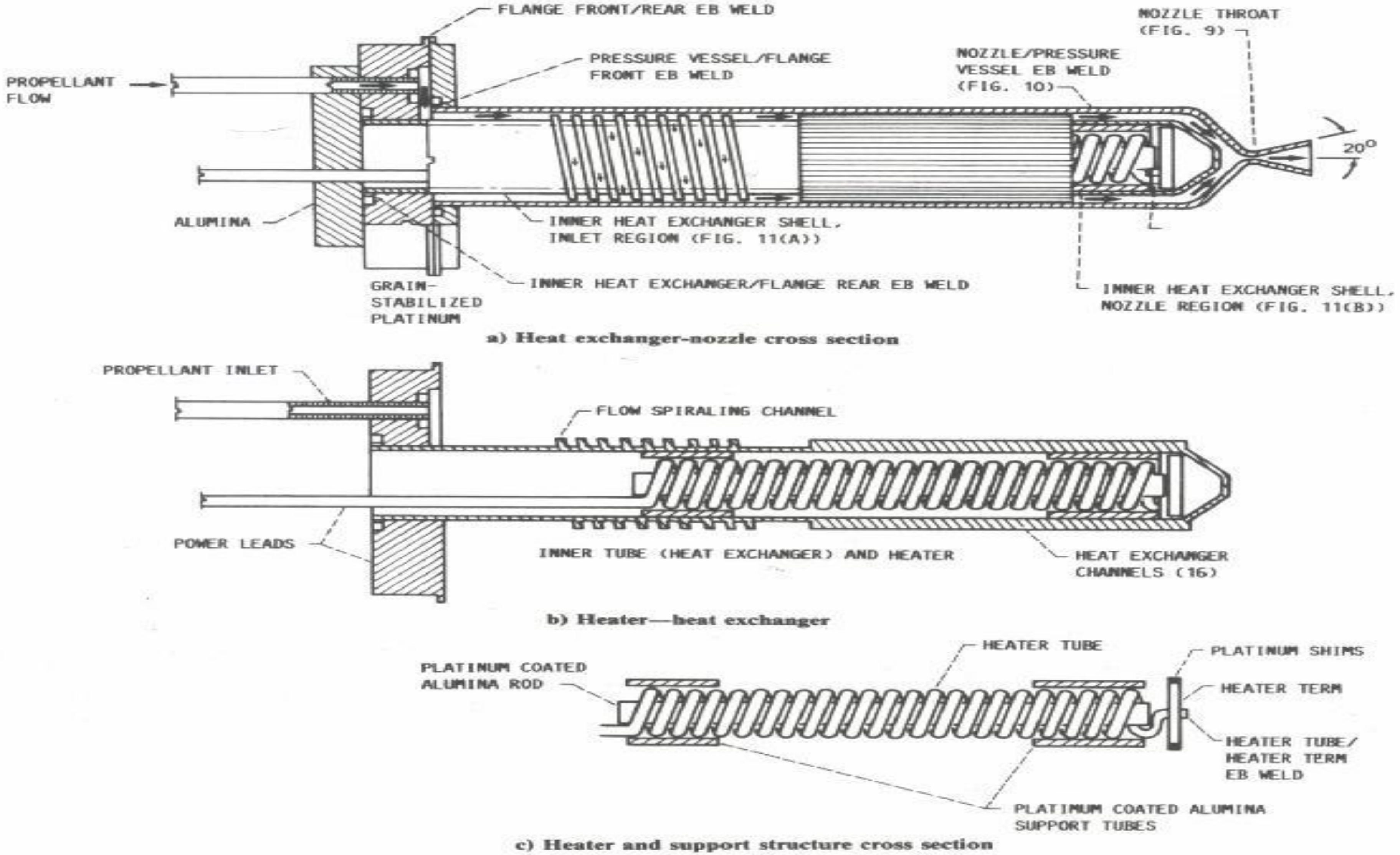
Uses hydrazine propellant at a flow rate of 0.045-0.1225 g/s. It consists of two main assemblies:

1) a small catalyst bed with its electro-magnetically operated propellant valve and with heaters to prevent freezing of the propellant, and
2) an electrical resistance spiral-shaped heater surrounded by thin radiation shields made from tungsten and high-temperature electric insulators for supporting the power leads.

Power input level may be up to 500 W @ 25 V. Thruster mass is 0.9 kg.

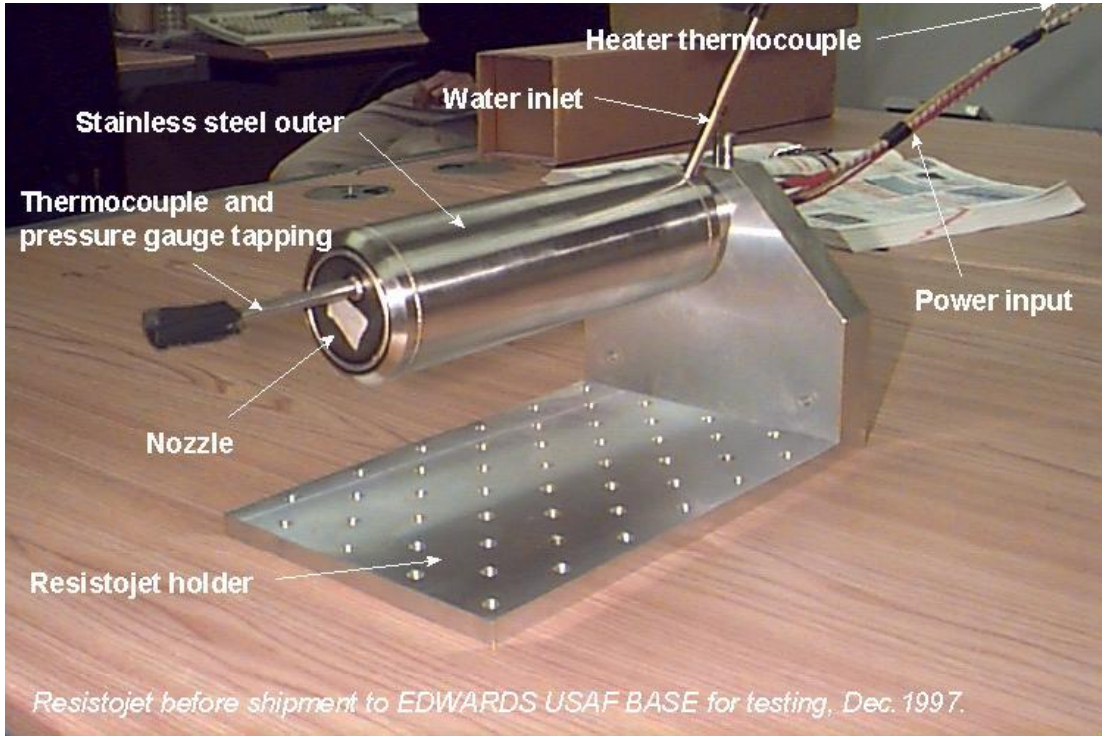
- MR-501B resistojet in detail
- Hydrazine thruster used on INMARSAT III satellites
- Thrust (N): 0.8-0.36
- Operation Pressure (bar): 26.5-6.2
- Spec. Impulse (s): 299
- Min. Impulse Bit (mNs): 88.96
- Total Impulse (kNs): 524.9
- Mass (kg): 0.871
- Valve Power (W): 8.25
- Valve Heater Power (W): 1.54
- Cat. Bed Heater Power (W): 3.93
- Augmentation Heater Power (W): 885-610
- Augmentation Heater Voltage (VDC): 29.5-24.5
- Steady State Firing:
    - 2.0 hrs - single firing
    - 370 hrs - cumulative

NASA MULTI-PROPELLANT RESISTOJET

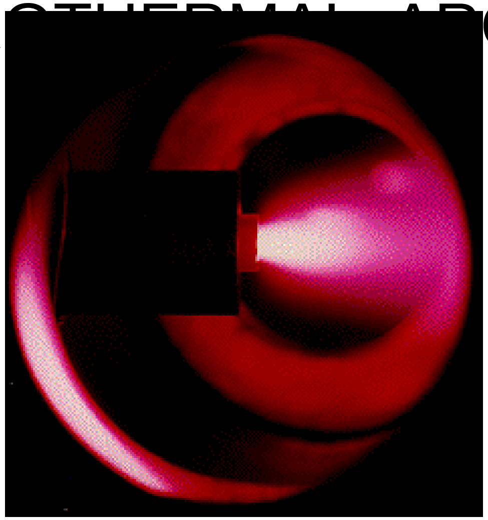


- **A schematic of a NASA developed model multi-propellant resistojet.**
- **It consists of a radiation heating element located in an evacuated cavity within an annular heat exchanger body.**
- **The heat exchanger consists of two concentric tubes sealed together to permit contained gas flow within the annular region between them.**
- **A spiral channel near the rear (inlet end) of the heat exchanger directs the flow circumferentially to reduce heat loss from the rear of the thruster.**
- **The flow is then directed axially by 16 small channels in the forward (hottest) section of the heat exchanger after which the gases are expanded in the nozzle.**
- **The heating element is made from a coiled tube comprised of 22 turns over a length of 5.8 cm The platinum thruster components are joined by electron beam (EB) welds.**
- **To minimise radiative heat losses from the outer surface of the heat exchanger, the thruster is wrapped with radiation shielding consisting of two layers of 0.03 mm platinum foil followed by 13 layers of 0.13 mm stainless steel foil.**
- **The layers of the shielding are separated by small-diameter wires.**

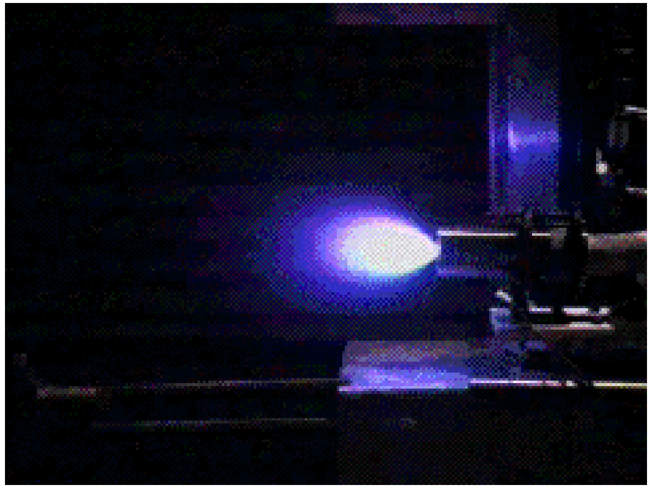
- Other propellants: water, N_2, N_2O, He, CH_3OH, CO_2, NH_3, N_2H_4
  - Higher molecular weight, higher thrust
  - Higher molecular weight, lower Isp
- Typical power ranges: 10-600 W
  - Up to 60 kW (pulsed)
  - Thrust levels: mN – N range
- Efficiencies typically around 80%

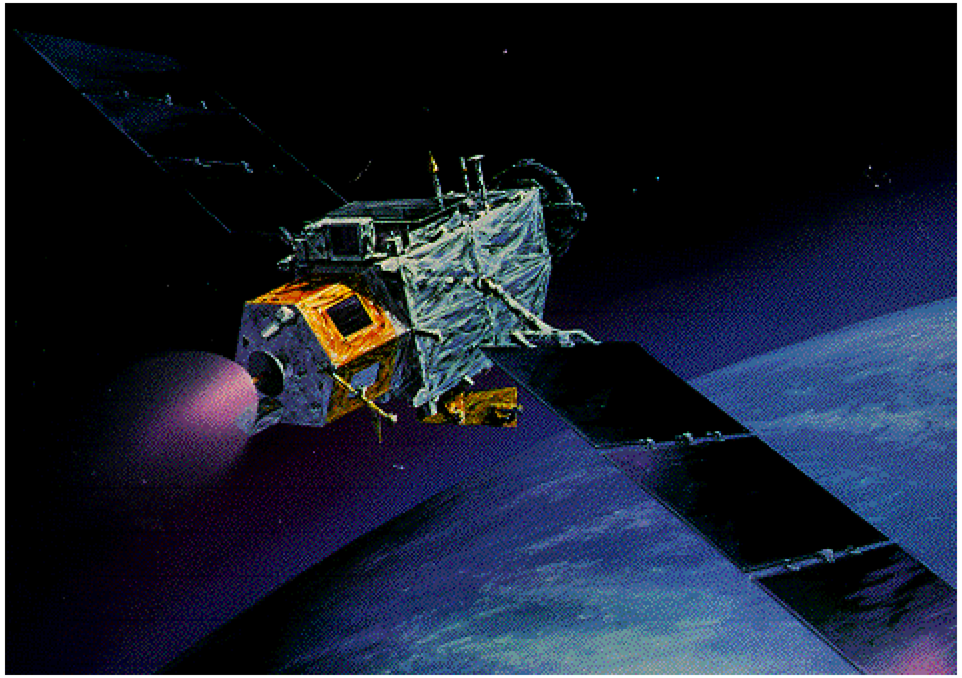


ELECTROTHERMAL ARCJET



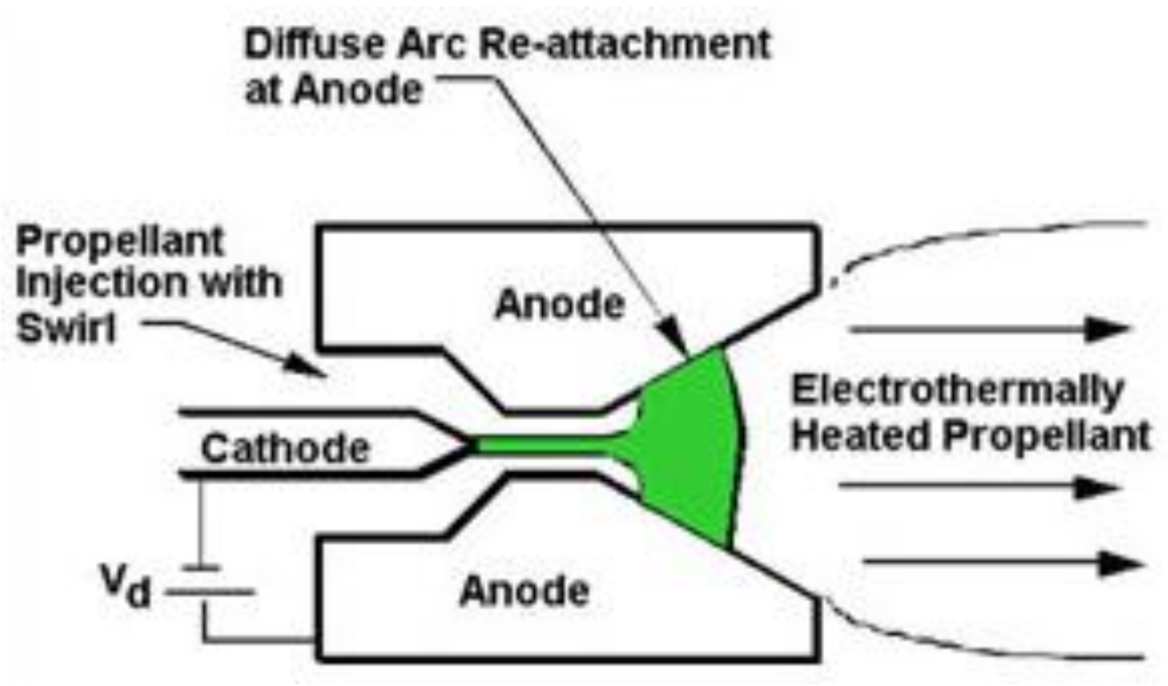
- Low power hydrazine arcjets in use on TelStar IV communication satellites
- First space test of a high power arcjet for use in orbit raising operations launched in 1997 under Air Force ESEX program





Hydrogen arcjet plume firing
Runs on H_2, N_2, or a mixture, at about 12A, 1 kW
peak around 600 Isp, (20-30% efficiency)

ELECTROTHERMAL: ARCJET IN DETAIL



- Arc is a beam of electrons emitted from tip of cathode and collected at positively biased anode
- Between cathode and anode is a narrow passageway called constrictor
- As electrons leave cathode, electric fields that exist between cathode and anode accelerate them
- Gas is injected near base of cathode with an azimuthal swirl
- Swirling flow surrounds cathode and electric arc in constrictor
    - Swirl prevents arc from kinking and touching walls, thereby keeping constrictor from melting
    - Swirl helps circulate gas through arc, which can have an effective temperature over 15,000 °C.
- Arcjets developed for a multitude of applications ranging from station keeping of moderate-sized spacecraft (500 W, hydrazine) to a piloted mission to Mars (100 kW, hydrogen)
- Specific impulses range from approximately 500-600 s on hydrazine to ~2,000 s on hydrogen

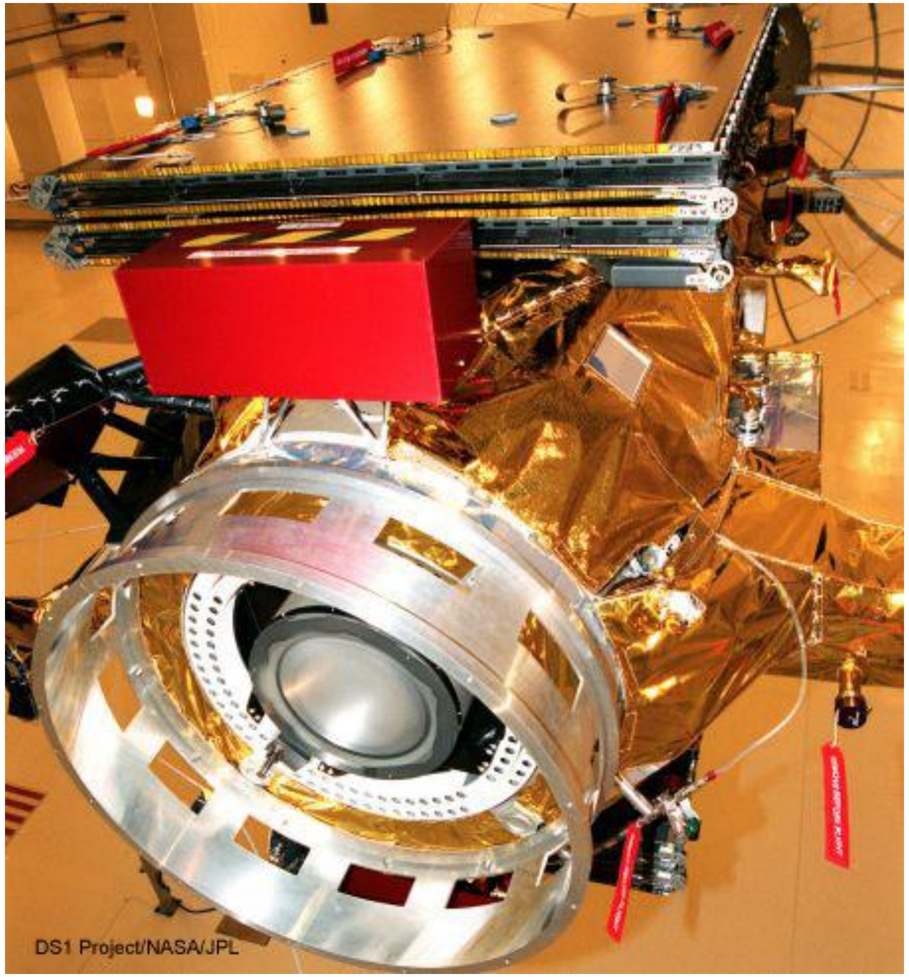
ELECTROTHERMAL: ARCJET

- Primary limitations on performance
  - Energy losses through dissociation and ionization
  - Heat transfer losses
    - Walls
    - Radiation losses
  - Loss of material at electrode surfaces
  - Since gas conductivity increases rapidly with temperature, attempts to produce high exhaust velocity (high T_0) are hampered by associated decrease of arc-column resistance relative to rest of circuit
    - This results in a decreasing fraction of total electrical power liberated in arc column
- Electrothermal systems have limited utility for a number of deep-space missions with large ΔV because of performance constraints of excessive frozen flow and electrode losses
  - 20% of input power is deposited into electrodes as heat
  - Specific impulse and thrust efficiency of arcjets operating on standard space-storable propellants (e.g., hydrazine) are limited to less than 700 s and 41%, respectively
  - Recent US Air Force arcjet tests have demonstrated specific impulses of over 800 s on ammonia, which is also space storable, at 30% thrust efficiency
  - Researchers in Germany have shown that arcjets can produce specific impulses of 2,000 s with hydrogen as propellant, but also at relatively low efficiency.

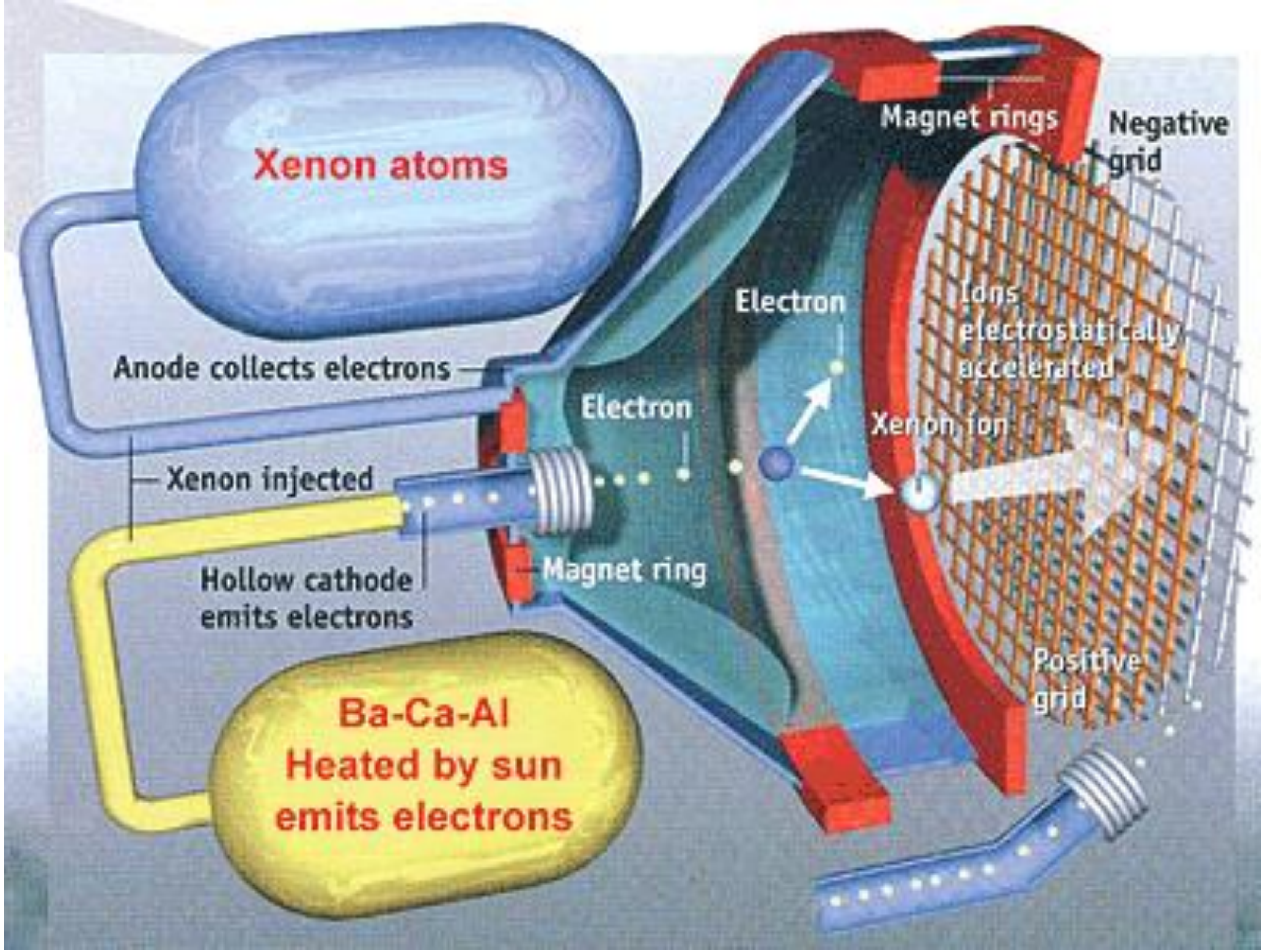
Electrostatic Thrusters

Ion thruster
Hall thruster

ION ENGINES: NASA NSTAR

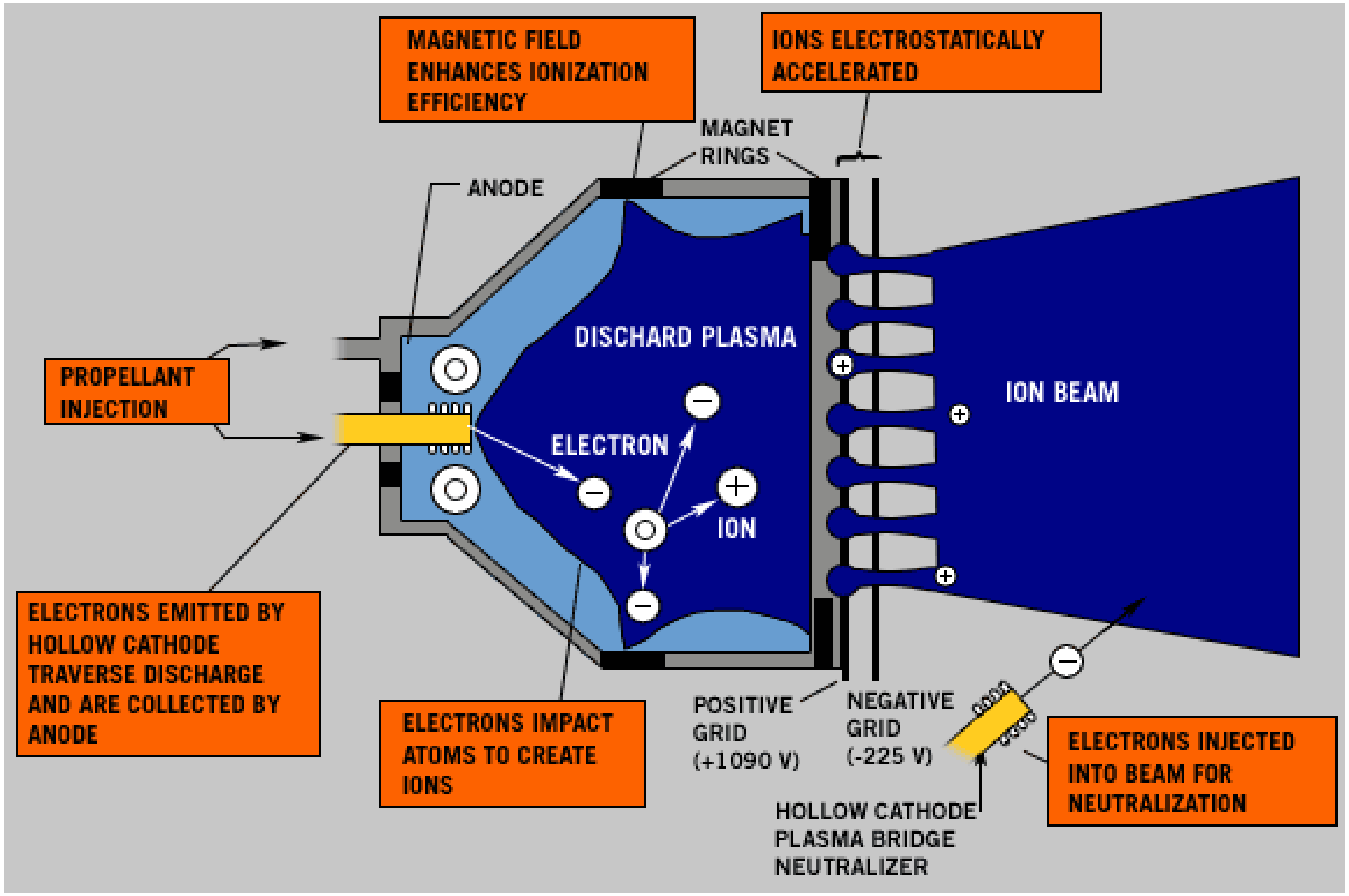


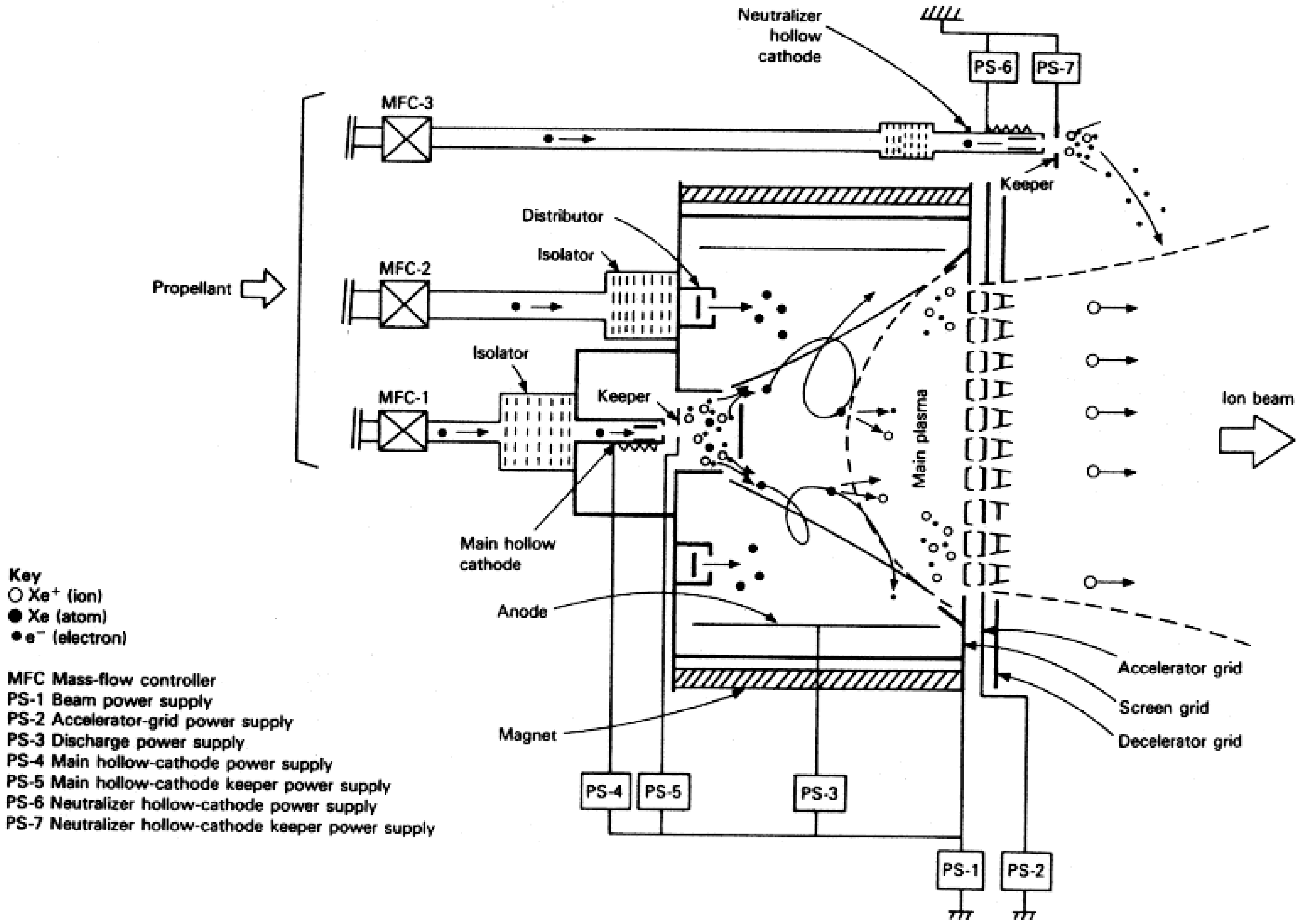
- Ion thrusters are electrostatic propulsion engines
- Ions (typically Xenon or Krypton) are produced in a discharge chamber via collisions between neutral atom and energetic electrons generated by a hollow cathode in discharge chamber
- Ions are accelerated through two fine grids with roughly 1300 V difference between them for 2.3 kW operation
- Ion beam is "neutralized" by electrons emitted from a second hollow cathode external to discharge assembly.
- The **N**ASA **S**olar Electric Propulsion **T**echnology **A**pplication **R**eadiness (NSTAR) program developed 2.3 kW ion engine for use as a primary propulsion engine for orbit transfer and intra-solar system trajectories
  - NSTAR engine is primary propulsion for Deep Space 1 (DS-1) probe currently in route for comet and asteroid rendezvous



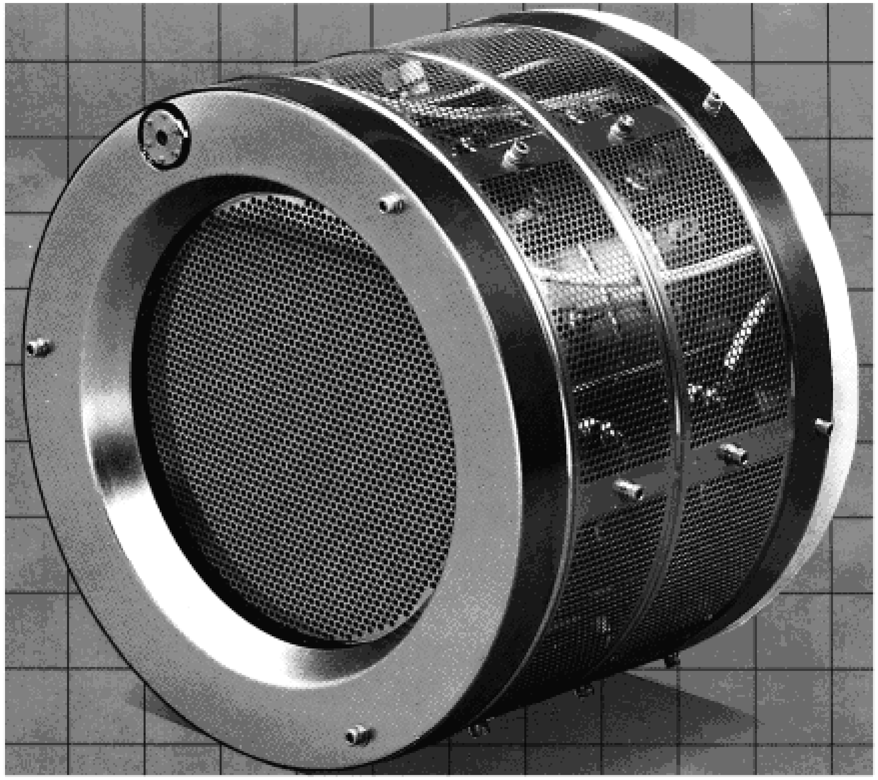
Xenon atoms
Magnet rings
Negative grid
Electron
Ions electrostatically accelerated
Anode collects electrons
Electron
Xenon ion
Xenon injected
Hollow cathode emits electrons
Magnet ring
Positive grid
Ba-Ca-Al
Heated by sun
emits electrons

OPERATING PRINCIPLES OF DS1 ION THRUSTER





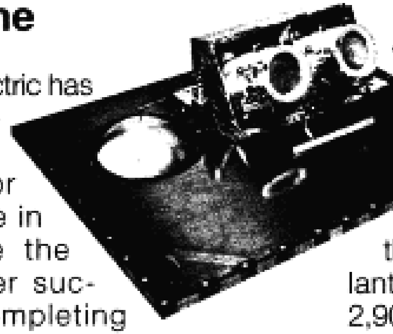
Neutralizer hollow cathode
PS-6
PS-7
MFC-3
Keeper
Distributor
Isolator
MFC-2
Propellant
Isolator
Keeper
MFC-1
Main hollow cathode
Main plasma
Ion beam
Anode
Magnet
Accelerator grid
Screen grid
Decelerator grid
PS-4
PS-5
PS-3
PS-1
PS-2
Key
Xe⁺ (ion)
Xe (atom)
e⁻ (electron)
MFC Mass-flow controller
PS-1 Beam power supply
PS-2 Accelerator-grid power supply
PS-3 Discharge power supply
PS-4 Main hollow-cathode power supply
PS-5 Main hollow-cathode keeper power supply
PS-6 Neutralizer hollow-cathode power supply
PS-7 Neutralizer hollow-cathode keeper power supply



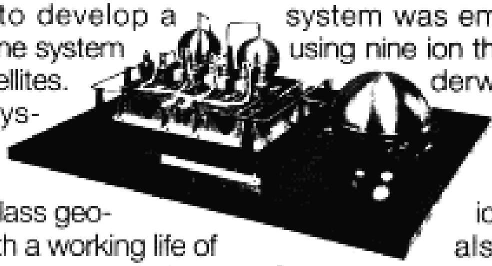
Thrust Method	Electric propulsion with electron-bombardment xenon ion engine
Operating Configuration	Two thrusters are operated simultaneously to perform north-south stationkeeping
Operating Frequency	209 times/year (4 times every 1 week) at nodal point
Thrust/Specific Impulse **Individual Thruster** **Combined Thruster**	 23 mN/2906 sec 40.3 mN/2516 sec for two thrusters canted at 30°
Power Consumption	1570 W
Mass	95 kg
Propellant Mass	up to 41 kg
Total Operation Time	up to 6,500 hours for 10-year mission
Number of Firings	up to 2,920 firings for 10-year mission

TECHNOLOGY

Ion Engine



Mitsubishi Electric has been developing ion engine systems for practical use in space since the 1970s. After successfully completing flight tests on Engineering Test Satellite III (ETS-III), which was launched in 1982, we started to develop a 20mN class ion engine system for geostationary satellites.

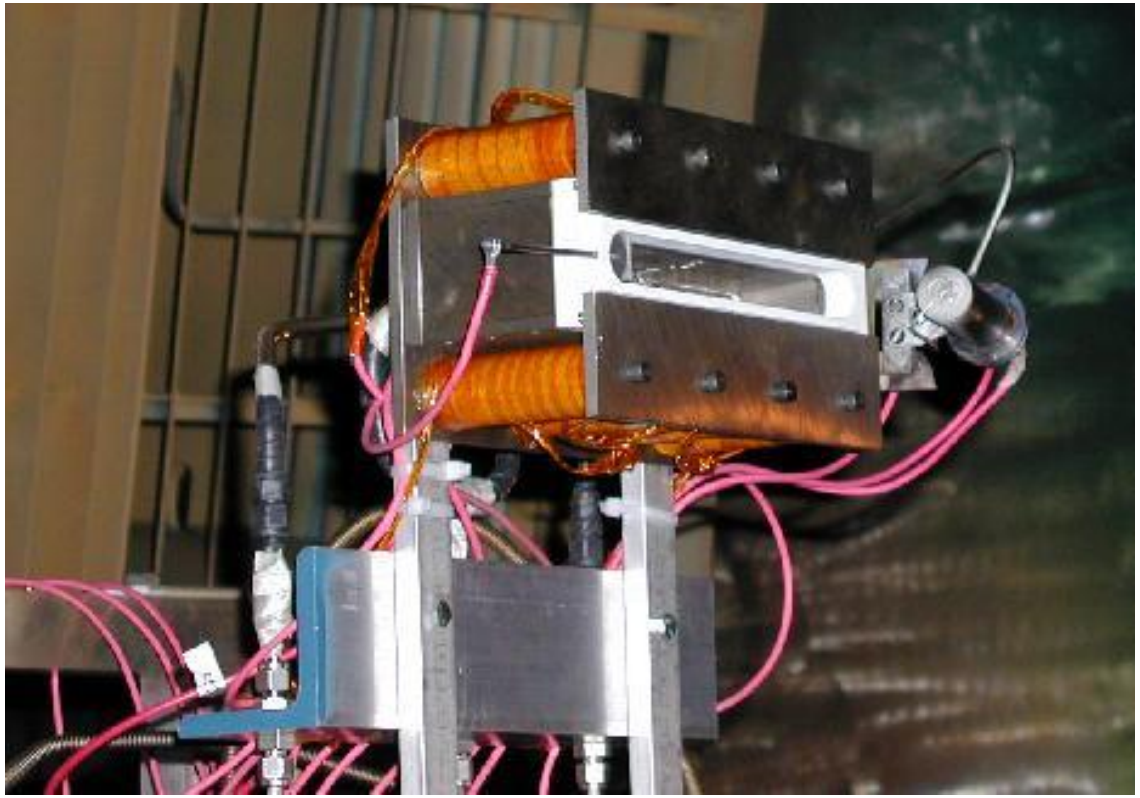


This ion engine system will be used for north-south station keeping on a 2-ton class geostationary satellite with a working life of ten years. The system was optimized for this mission and is to be used for Engineering Test Satellite VI (ETS-VI), which will be launched in 1994. Flight hardware is currently being manufactured.

This ion engine system has electron bombardment, Kaufman type thrusters which use Xe as a propellant and have a specific impulse of 2,900 sec and thrust of 20-30 mN. As ETS-VI requires a working life of over 6,000 hours, a three-grid ion extraction system was employed. Life tests using nine ion thrusters are now underway and three have already achieved over 6,500 hours of operation. This ion engine system will also be used for the Communication Engineering Test Satellite COMETS, which will be launched in 1997.

ION: 2 kW LINEAR GRIDLESS ION THRUSTER (LGIT)

- Linear Gridless Ion Thruster (LGIT) is two-stage device designed to incorporate efficient ionization process found in gridded ion thrusters with high thrust density and crossed-field acceleration mechanism of Hall thrusters
- Can think of this thruster as ionization stage of an ion thruster combined with acceleration stage of a Hall thruster



ELECTROSTATIC: PROPELLANTS

- Alkali Metals: H, Li, Na, K, Rb, Cs
  - Low ionization potential (easy to create ions), 1 electron in outer shell
- Inert: He, Ne, Ar, Kr, Xe, outer shell full
- Hg: two electrons in outer shell

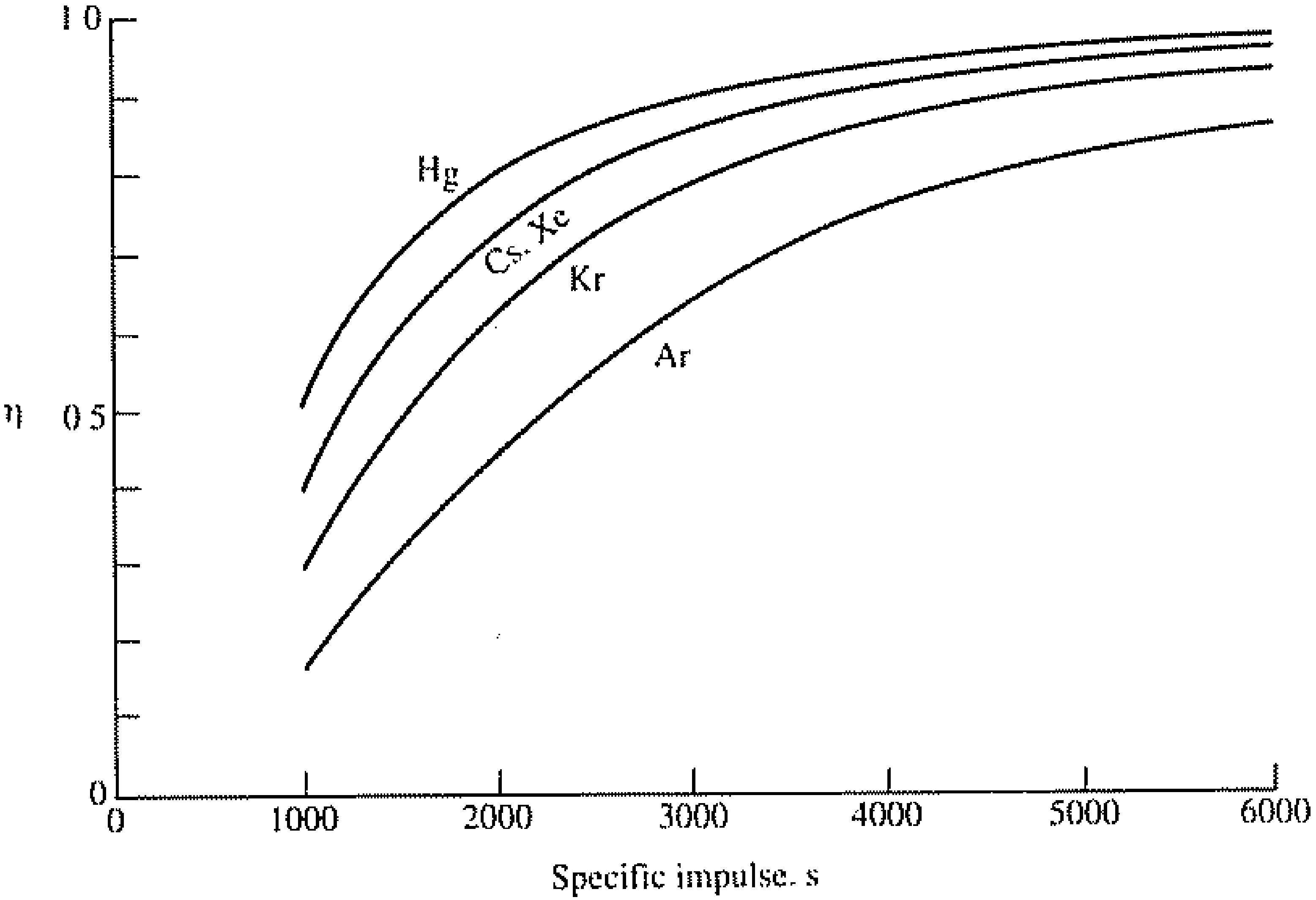
KEY PERFORMANCE PARAMETERS

Atomic masses and ionization potentials

Element	Atomic weight	m (10^{-25} kg)	q/m (singly charged) (10^5 C/kg)	Atomic number	1st ionization potential (eV)[a]	2nd ionization potential (eV)[a]
Cs	132.9	2.21	7.25	55	3.89	25.1
Hg	200.6	3.33	4.80	80	10.44	18.75
Xe	131.30	2.18	7.34	54	12.13	21.21
Kr	83.80	1.39	11.50	36	13.999	24.359
Ar[b]	39.948	0.66	24.13	18	15.799	22.629

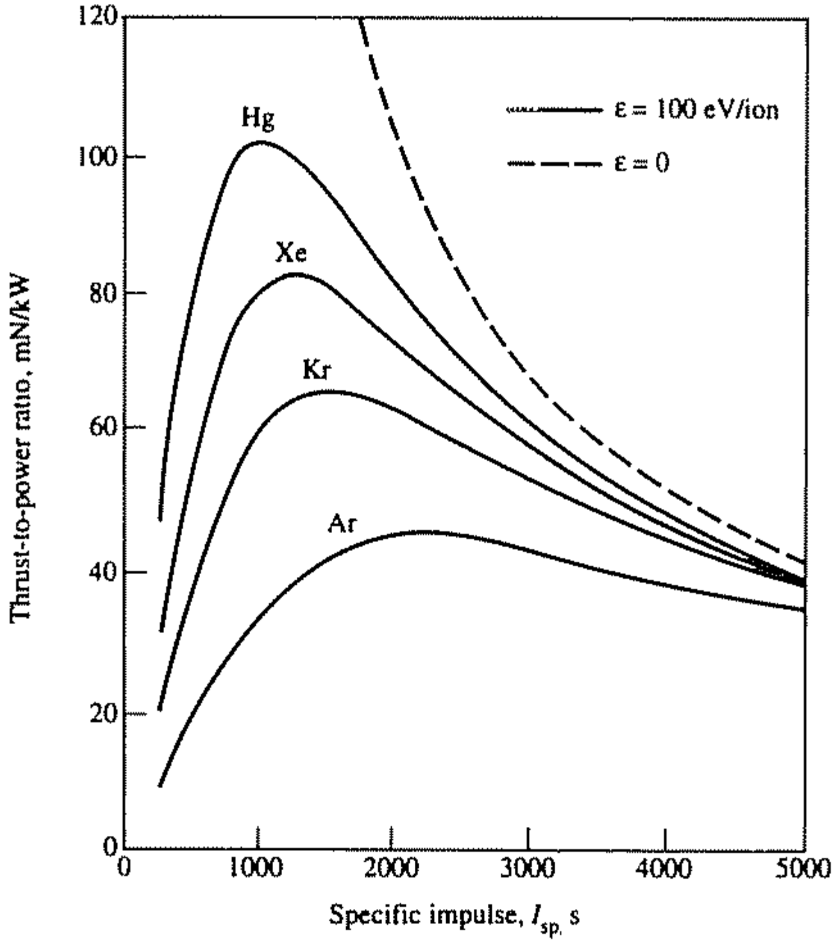
[a] $1\ eV = 1.60 \times 10^{-19}\ J$
[b] $A = 6.0225 \times 10^{26}$ *molecules/kmol*

ELECTROSTATIC ROCKET PERFORMANCE TRENDS



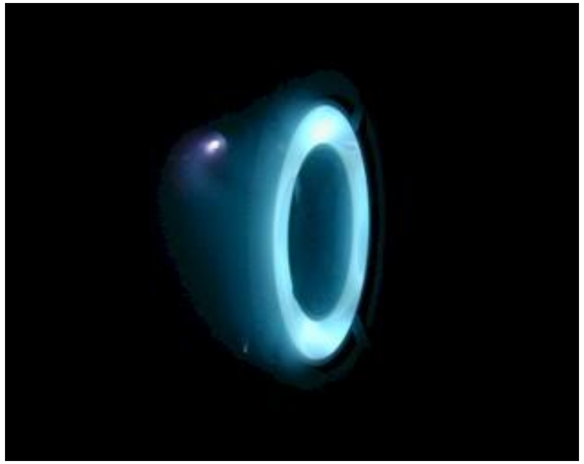
Efficiency of ion thrusters with 100 eV loss per ion

ELECTROSTATIC ROCKET PERFORMANCE TRENDS

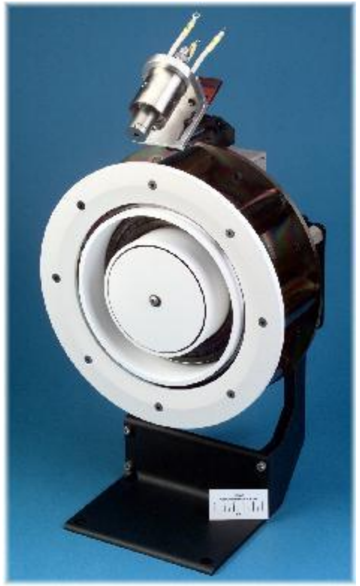


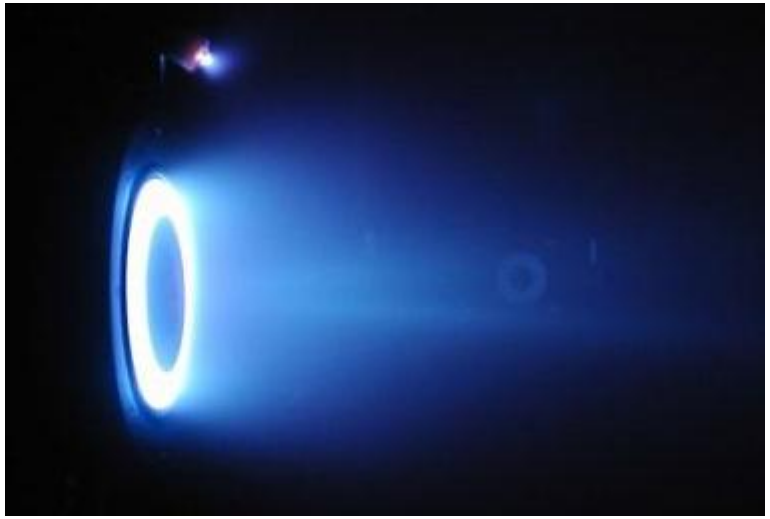
Thrust-to-power ratios for high-performance ion propulsion subsystem operated on various propellants (Courtesy Beattie, Matossian, and Poeschl [6] © AIAA Reprinted with permission)

NASA-173GT 2-Stage Hybrid Hall/Ion Thruster



HALL: UM-NASA, NASA-173M Hall Thruster





HALL THRUSTER

3 kW T-140 HALL THRUSTER (P&W)

ELECTROMAGNETIC: MPD

- Lack of high performance in electrothermal systems that may have first led to development of one kind of electromagnetic engine; the magnetoplasmadynamics (MPD) thruster

- MPD thruster "invented" by accident

- Arcjet researchers were investigating effect of mass flow rate on thrust
  - They noted that while thrust of arcjet initially dropped with decreasing mass flow rate as expected, thrust began to increase with decreasing flow rate once a sufficiently low flow rate was reached
  - This seemingly impossible result marked transition from electrothermal heating to electromagnetic acceleration as flow rate decreased

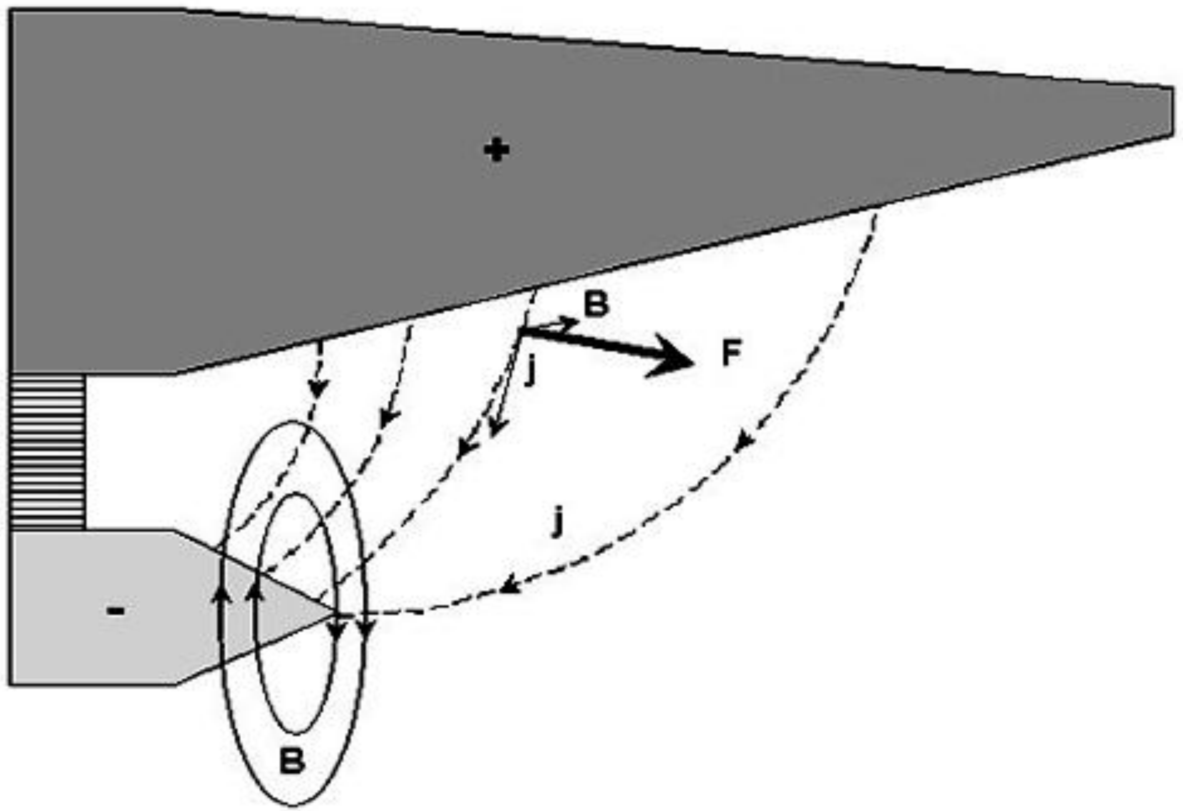
ELECTROMAGNETIC: MPD

- Electromagnetic devices pass a large current through a small amount of gas to ionize propellant
- Once ionized, plasma is accelerated by electromagnetic body force called Lorentz force which is created by interaction of a current (j) with magnetic field (B):

F*=*j x ***B***

- Current provided between energized positive and negative electrodes, while magnetic field is either induced by (created from) current itself, applied externally via an electromagnet or both
- Strength of Lorentz force for an MPD thruster with a self-induced magnetic field is roughly proportional to ratio J^2 / mdot, where J is total thruster current
- While gas-phase propellants like hydrogen and lithium (after vaporization) can be used, solid propellants can also be used in pulsed electromagnetic accelerators called pulsed plasma thrusters (PPTs).

ELECTROMAGNETIC: MPD



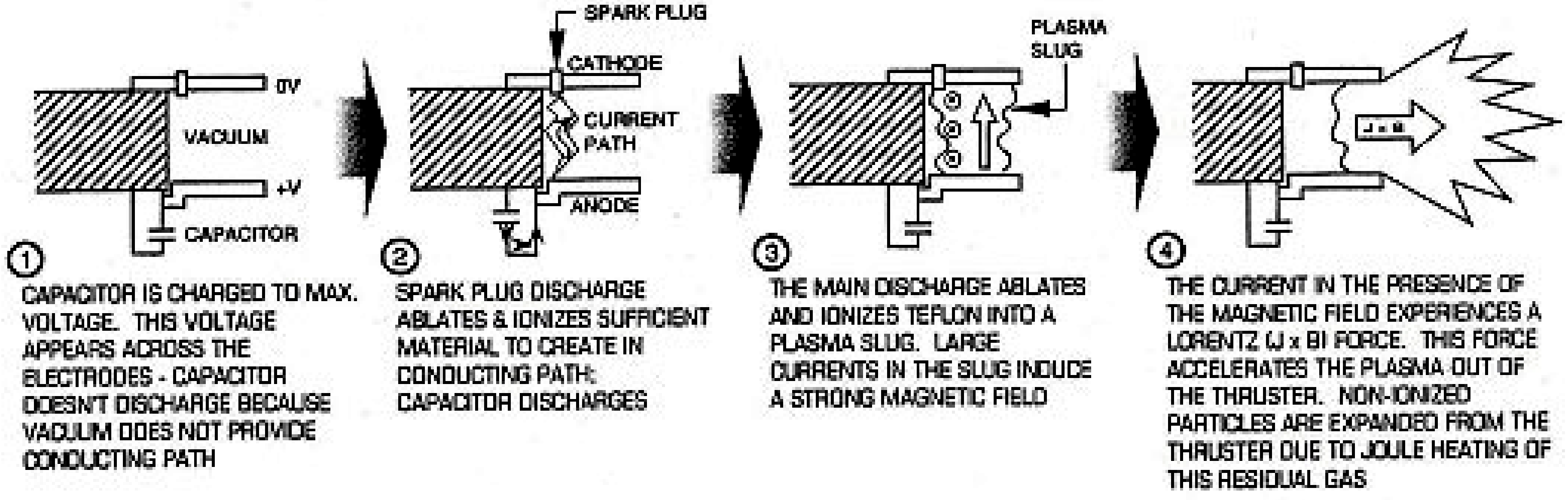
$$\vec{F} = \vec{j} \times \vec{B}$$

ELECTROMAGNETIC: PPT

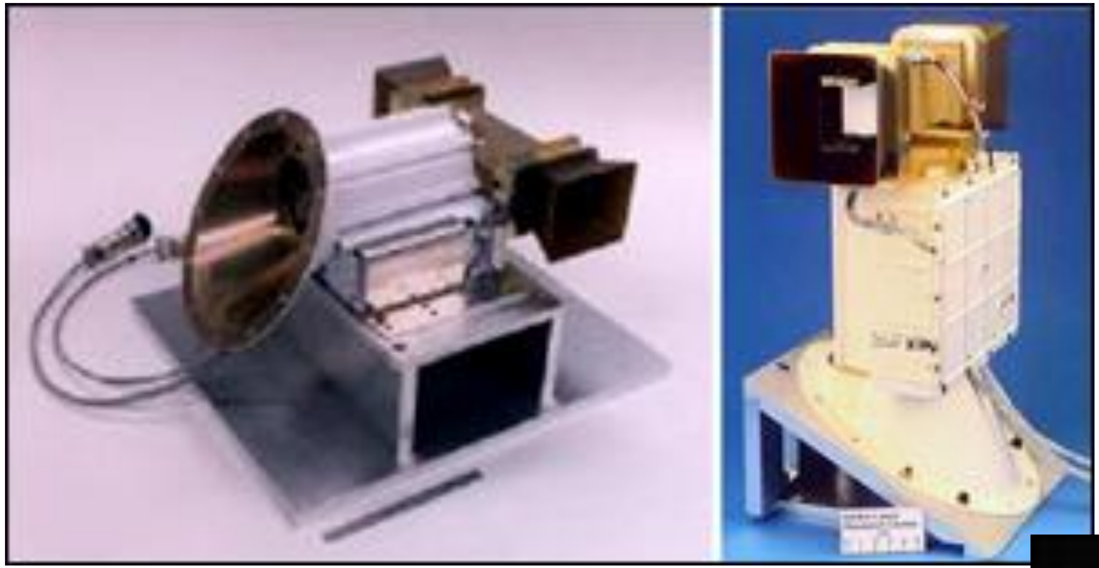
- PPTs use solid Teflon propellant to deliver specific impulses in the 900 - 1,200 s range and very low, precise impulse "bits" (10-1,000 µNs) at low average power (< 1 to 100 W)
- PPTs inherently inefficient (η ~5%)
  - Simplicity and low impulse bits provide highly useful
  - Precision-flying of a spacecraft constellation
- PPT consists of a coiled spring that feeds Teflon propellant bar, an igniter plug to initiate a small-trigger electrical discharge, a capacitor, and electrodes through which current flows
- Plasma is created by ablating Teflon from discharge of capacitor across electrodes
- Plasma is then accelerated to generate thrust by Lorenz force that is established by current and its induced magnetic field
- PPT flown on both American and Soviet/Russian spacecraft since the 1960s
- PPT was used to maintain fine pitch attitude control for NASA New Millennium Program's Earth Observing-1 mission launched in 2000

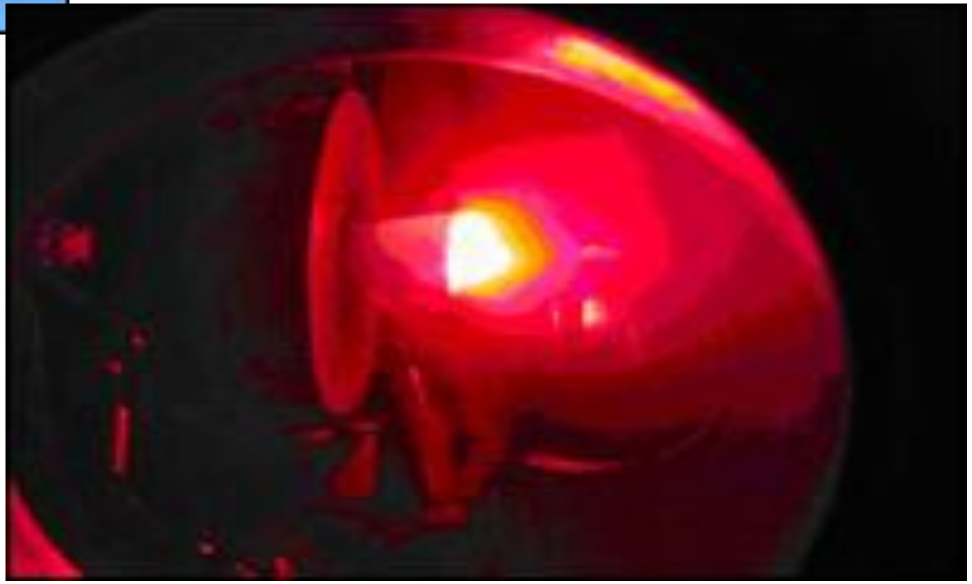
ELECTROMAGNETIC: PPT

- MPD thrusters can be operated in pulsed mode like PPTs or continuously
- MPD thruster's high specific impulse (> 4,000 s) and high power density make it an excellent candidate for high-power, high-ΔV missions
- MPD thrusters suffer from relatively low efficiency (< 50%) due to frozen flow losses and electrode deposition
- Figure shows a 30 kW MPD thruster operating on lithium propellant at Princeton University
    - Lithium-propellant MPD thrusters at power levels above 200 kW



PULSED PLASMA THRUSTER





THANK YOU