AE 330/708

AEROSPACE PROPULSION

Instructor

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Liquid Propellant Rockets - Introduction

Use of liquid propellants at ambient or cryogenic temperatures

Wide range of thrust levels 1 milli Newton – Few million Newtons

Controllable flow rates of propellants to the thrust chamber and hence controllable thrust in real time

Restartable with ease, reusable in some cases

Relatively complex system involving moving parts like turbo-pump, hydraulic actuators, valves, etc

Costly as compared to the solid rocket motors

Liquid Propellant Rockets - Introduction

Number of propellants – Biproellant and monopropellant engines

State of the propellants – Earth storable propellant, cryogenic engines, semi-cryogenic engines, gas propellants

Nature of feed system – Pressure feed system, turbopump feed system

Introduction

	Boost propulsion	Auxiliary propulsion
Mission	Impart significant velocity to payload (e.g. launch vehicles)	Attitude control, trajectory control, maneouvers
Application	Boosters, main engines, long range missiles	Satellite thrusters, final stages of anti-ballistic missiles
Impulse	High	Low
No. of thrust chambers	Usually 1; 2-4 in some cases	Between 4-24
Thrust level	High 5 kN – 9 MN	Low 1 mN – 4.5 kN
Feed system	Turbopump	Pressure
Propellants	Storable or cryogenic bipropellant	Storable monopropellant, gaseous propellant

Introduction

	Boost propulsion	Auxiliary propulsion
Chamber pressure	2.4 – 21 MPa	0.14 – 2.1 Mpa
No. of starts in mission	Usually no restart	Several restarts are possible for small thrusters
Cumulative duration of flight	~ Few minutes	~ several hours
Shortest firing duration	5 – 40 sec	~ 0.02 sec
Time for full thrust	Upto several seconds	Very fast 0.004 – 0.08 sec

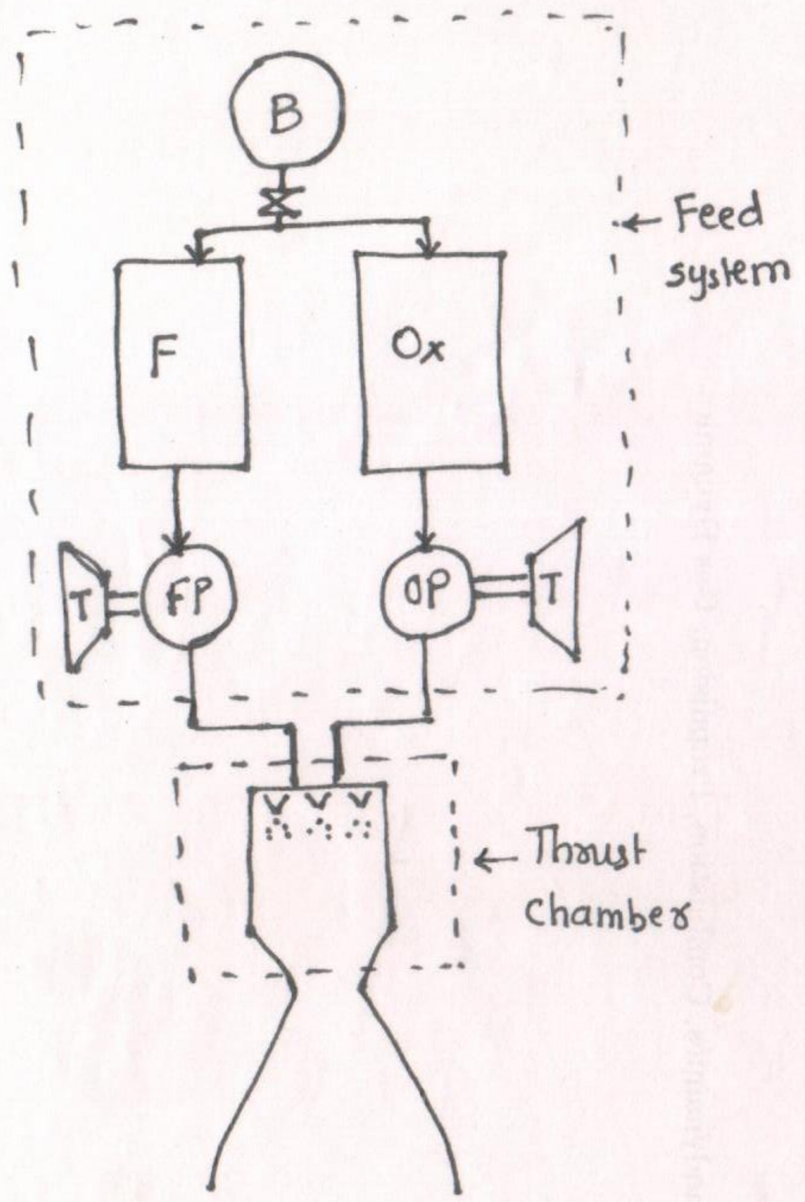
Introduction

Engine systems:

Propellant storage: Tanks, insulations, expulsion devices

Feed systems: Gas bottles, turbopump assemblies, gas generators/precombustors, valves, regulators, flow lines

Thrust chambers: Nozzles, injector/atomizer system, cooling system



Performance

For steady state operation of engine,

Mass flow rate through nozzle = Mass flow rate of fuel propellant + mass flow rate of oxidizer propellant

Overall mass ratio, MR = Ratio of the mass flow rate of oxidizer to the mass flow rate of fuel

Flame temperature and the molecular weight of the gases depend upon the combustion characteristics of the propellants

The performance of the engine depends on the expansion of the gases through the nozzles

Feed systems

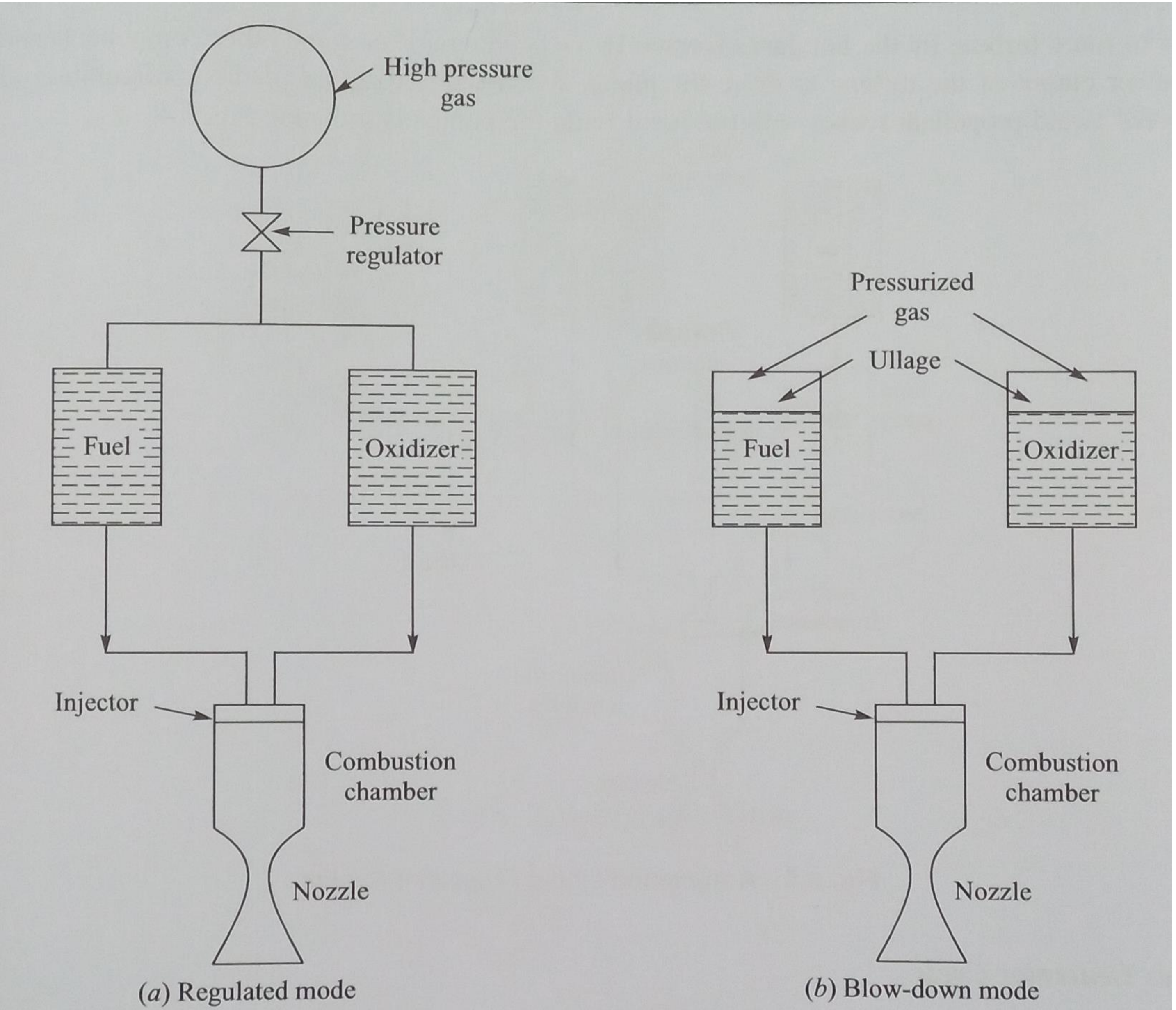
Includes the storage of the propellants, flow circuit for carrying the propellants to the combustion chamber, pressurization system

Two primary functions:

The combustion chamber requires a certain amount of fuel and oxidizer flow rates to maintain constant flow rate through nozzle

The combustion chamber operates at very high pressure and more pressure needs to be used to push the propellants into the combustors

Feed systems



Pressure feed system:

Direct pressurization using a high pressure inert gas

The pressurization gas can be stored at ambient temperature and high pressure in separate gas bottles and a regulated supply is provided to propellant tanks

Alternatively, the gas can be stored directly in the propellant tank empty volume (blowdown mode)

Used for low thrust engines

Feed systems

Approximate estimation of power for pump-fed system:

Engine thrust level == 600 kN with specific impulse = 300 sec and operating with Combustion chamber pressure ~ 10 Mpa

Mass flow requirements = 600 kN / (300 x 10) = 200 kg/s (I have assumed gravitational acceleration of 10 m/s^2 for approximate order of magnitude estimation)

Typical pressure upstream of propellant injectors ~ 11.2 MPa (to ensure good atomization of propellants)

Propellant tanks are usually pressurized (to avoid cavitation) upto 0.2 MPa

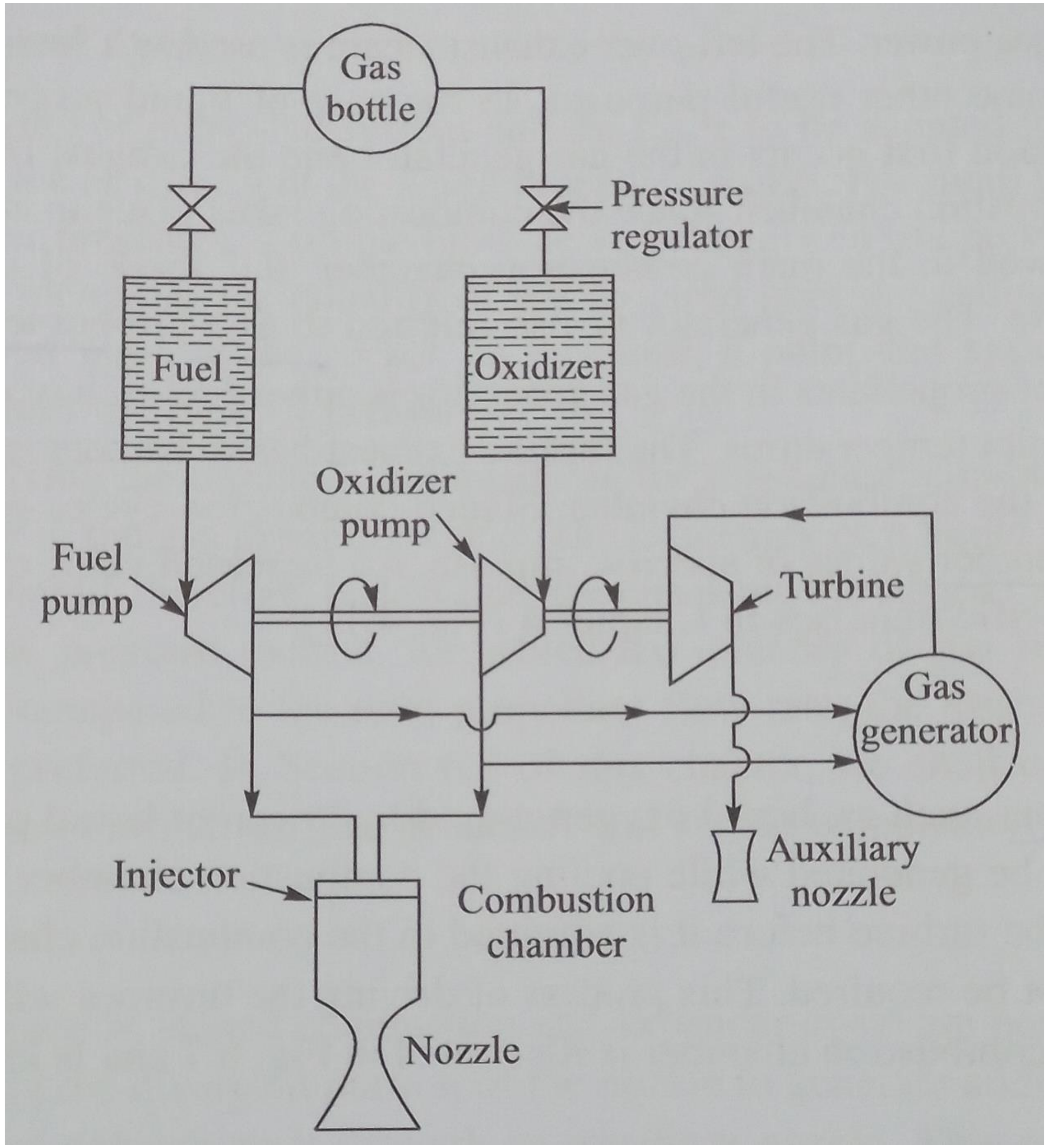
Pumping requirement (pressure rise for propellant) == 11 MPa

Assume average propellant density of 1000 kg/m3

Work done by the pump = pressure rise x flow rate (volumetric) = (11 x 1000000)x(200/1000) = 2.2 MW

Huge power requirement

Propellant Feeding Cycles



Gas generator cycle: (Example: RS-68 engine on Delta vehicle)

High pressure and high temperature gas is created in gas generator by bleeding the main propellants

The gas is expanded in turbine which runs the pumps to deliver the propellants

The turbine exhaust is passed through auxiliary nozzle

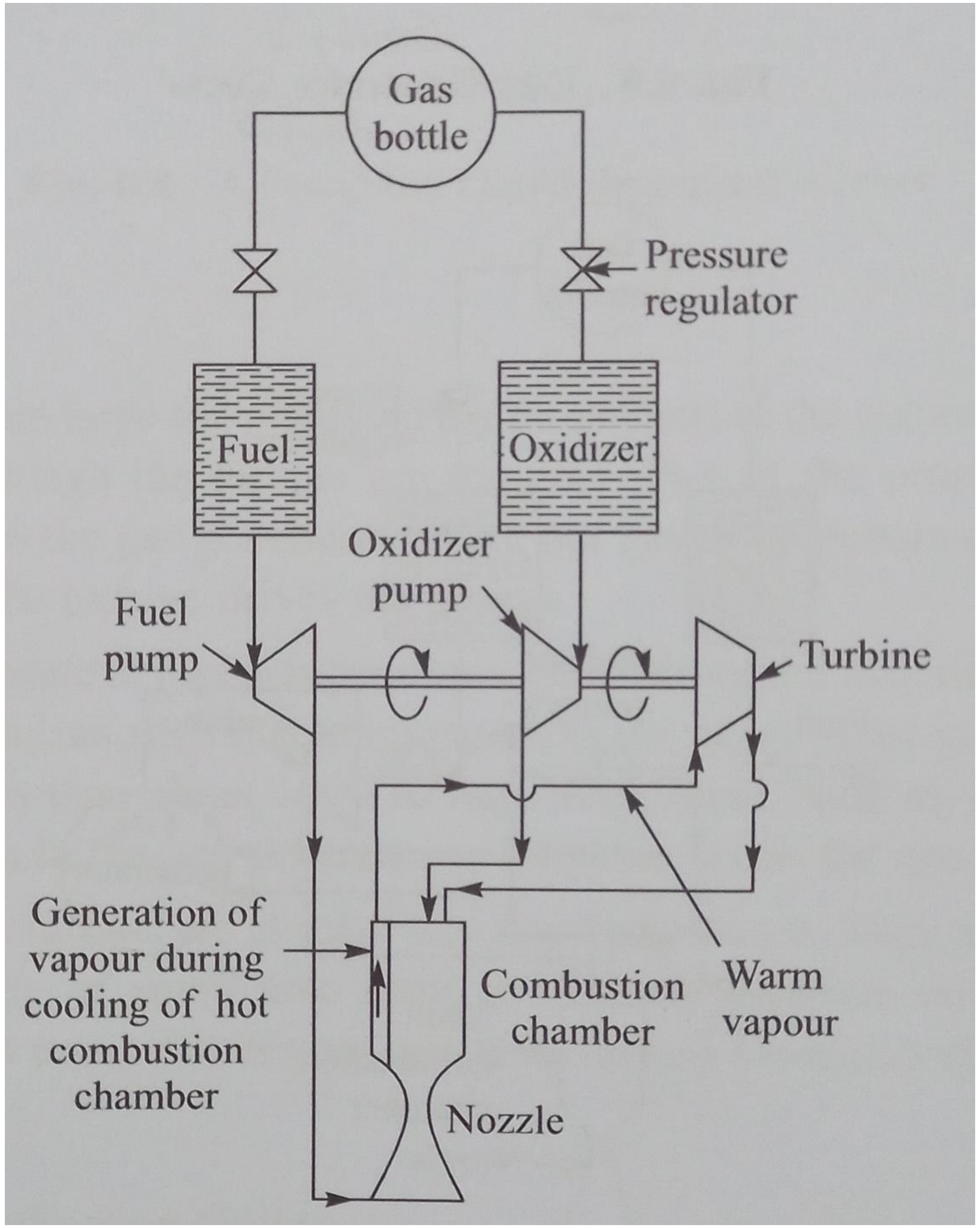
Relatively simple, low pressures in the lines and pumps, low inert mass; Lower specific impulse than other two cycles
Turbine pressure drop is low but good for high mass flow rates

F-rich or Ox-rich mixture is burnt in gas generator for limiting the temperature

Other invariants:

- Use of monopropellant or solid propellants for gas generator (like hydrogen peroxide)
- Turbine exhaust – discharged into a divergent portion of main nozzle

Propellant Feeding Cycles



Expander cycle : (Example: RL-10 engine used in upper stages of launch vehicles

Usually used with Hydrogen fuel (hence in cryogenic systems)

Generation of high pressure and high temperature gas by extracting the heat energy from combustion chamber

Fuel is usually passed through a cooling jacket and temperature and phase of the fuel is changed

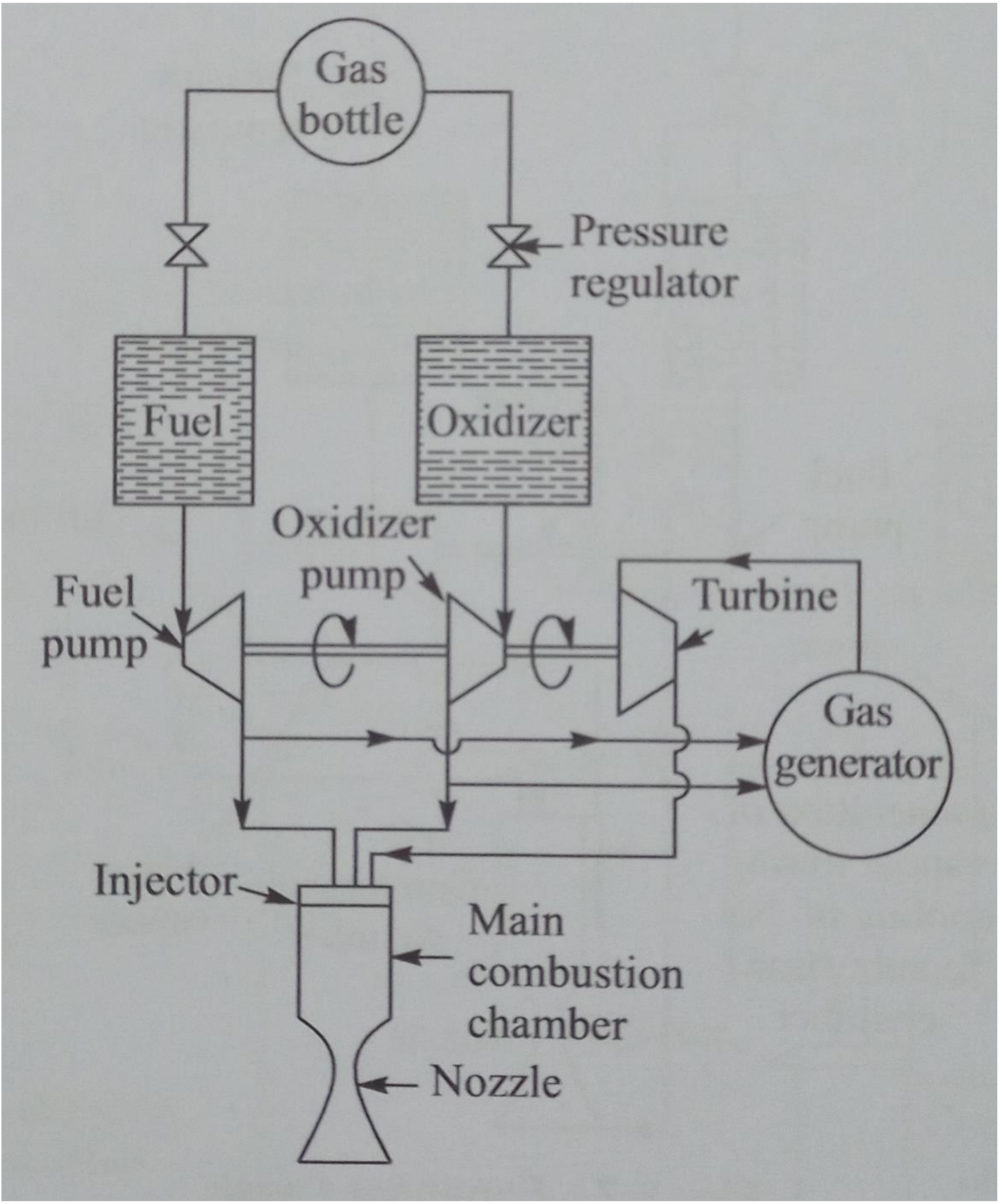
High P and T fuel gas is expanded in turbine before injecting into the main combustion chamber

Good specific impulse, simple in construction, no combustion device, low engine mass

The line pressures and the pump pressures should be higher than the combustion pressure as the fuel gas has to enter the main combustion chamber

Combustion takes place in single stage in main combustor and the gas expands efficiently in the main nozzle

Propellant Feeding Cycles



Staged combustion cycle :

In SSME – F-rich precombustor
RD 120 (semicryogenic system) – Ox-rich precombustor

F-rich or Ox-rich combustion in precombustor from main propellant lines

The precombustor gas is expanded in turbine to generate power for pumps

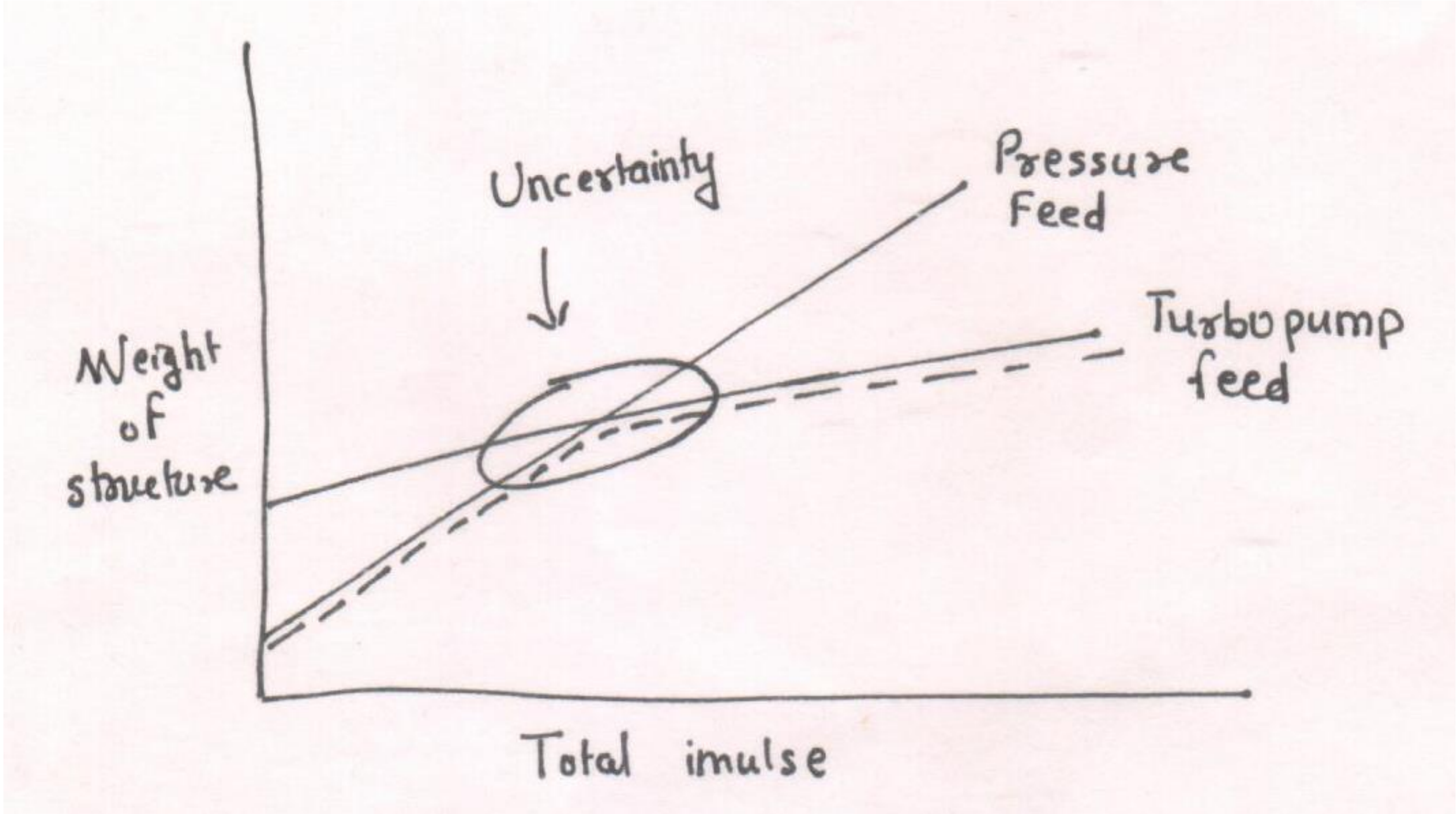
The turbine exhaust is then passed into the main combustor for extracting the remaining energy of the propellants

Low pressure drop and high mass flow rate turbines; High specific impulse, Very efficient cycle among all the cycles

Very high pressures in flow lines, pumps, valves since all the systems are operating at pressures higher than the main combustion chamber pressure

Relatively large mass of the engine, but compensated by the high specific impulse

Feed systems comparison



Pressure feed system:
Superior performance when total impulse or mass flow rates are lower

Typically thrust-to-weight ratio < 0.6

Good for Repeated thrust pulses or several restarts

Turbopump feed system:
Better performance when the thrust levels or mass flow rates are high

Large transient time during the start; continuous operation is suitable

Complex system due to moving parts and many subassemblies

Propellants

Economic factors – Low cost and availability, Cost of transportation and storage must be low

Performance – High flame temperature and low molecular weight, high density/specific gravity

Ignitibility – Spontaneous ignition, low ignition energy and ignition delay

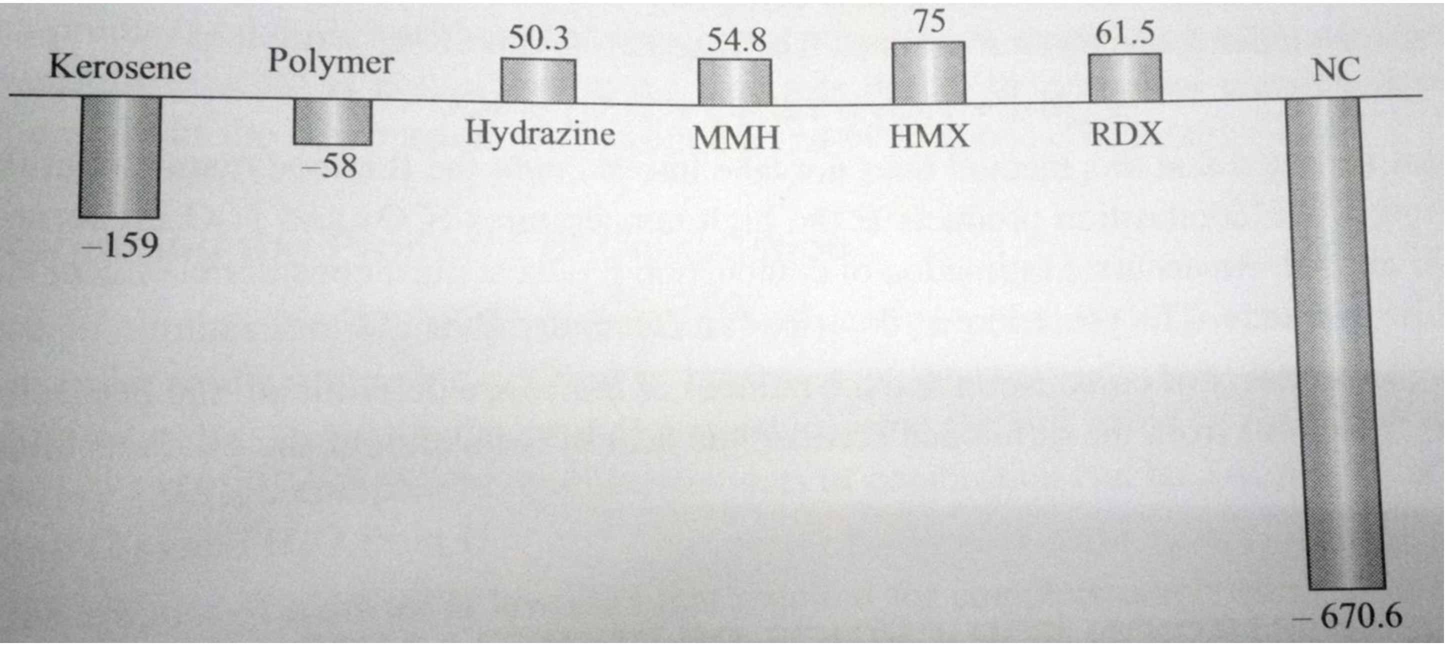
Thermodynamic or heat transfer aspects – Low freezing point, high specific heat, thermal conductivity, high boiling point and decomposition temperature

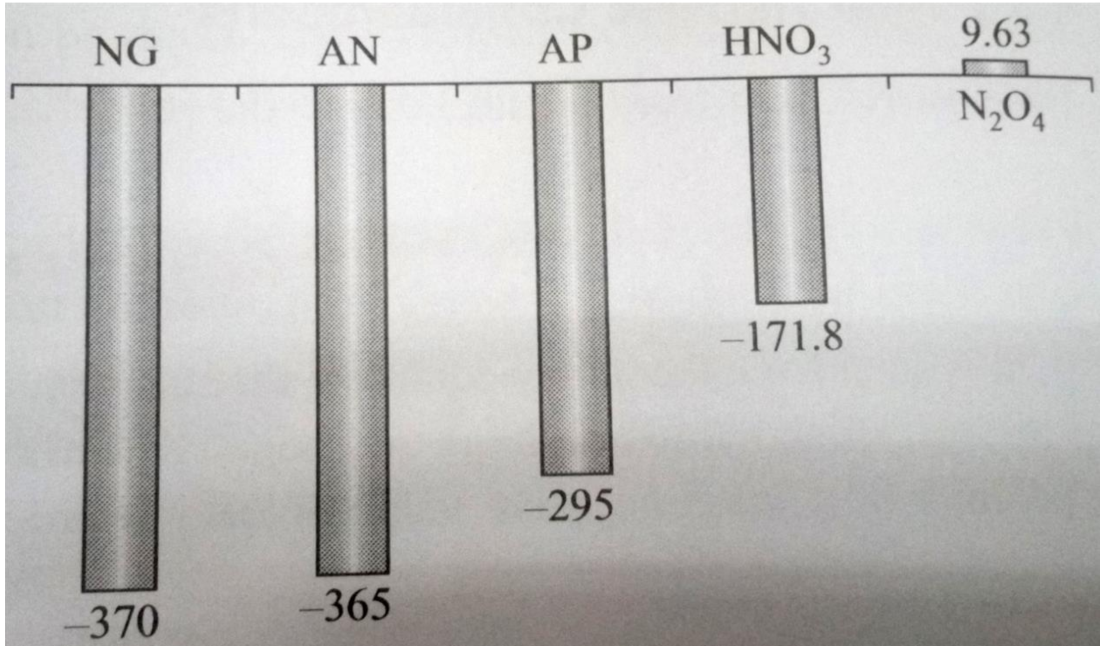
Stability characteristics – No/less decomposition during long term storage, No effect of ambient atmosphere (e.g. low moisture absorption)

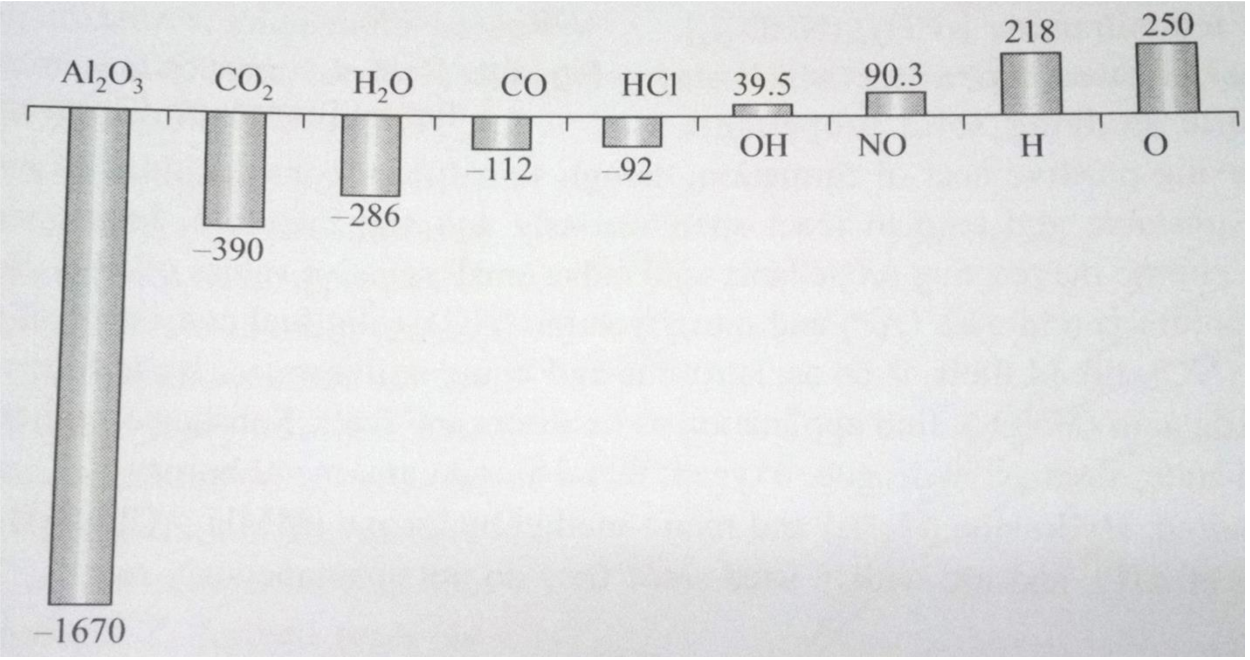
Pumping characteristics – Low viscosity, low vapour pressure

Material compatibility

Physical hazards – (should be minimum) Corrosion (HNO_3, HF), explosion (H_2O_2, cryogenic propellants), Fire (H_2O_2)
Health hazards – Toxicity (all propellants), Carcinogenic (hydrazine), irritation (HNO_3 vapours), burns and cold burns (almost all propellants); Exhaust should be clean and non-toxic







Heat of formation of products should be large negative value and that for reactants should be small negative value or even positive

Large heat of combustion

Propellants - Oxidizers

Liquid Oxygen (LOx):

Cryogenic propellant with boiling point at 90 K
Specific gravity – 1.14
Used extensively with variety of fuels
With Kerosene (RP1) it forms a semicryogenic combination
With liquid H_2, it forms a cryogenic combination
Lox – non-hypergolic propellant
Prepared by boiling nitrogen from liquid air
Explodes in the presence of oils when suddenly pressurized
Causes cold burns due to very low temperatures

Hydrogen peroxide (H_2O_2):
Highly concentrated H_2O_2 is used (70-99%), rest is water

Used as gas generator propellant or monopropellant

Hypergolic with hydrazine and burns well with kerosene
Decomposes at a very slow rate during storage (~ 1% per year)
Green exhaust; but severe burns on skin, fire when comes in contact with wood, cloth, paper

Used extensively in early missions

Hypergolic propellant combination: A propellant combination which does not require ignition system. These propellant start burning as soon as they come in contact with each other (in liquid phase also)

Propellants - Oxidizers

Nitric acid (HNO_3):

Red Fuming Nitric Acid (RFNA)

~ 82% HNO_3 + 15% NO_2 + H_2O
Used extensively in past

Annoying and poisonous fumes
Highly energetic w.r.t. WFNA

Stable in storage; Hypergolic with hydrazine based fuels

Highly corrosive → Problematic with metal storage vessels → 1% HF is added to prevent the corrosion
Called as inhibited RFNA (IRFNA)

Nitric acid (HNO_3):

White Fuming Nitric Acid (WFNA)

~ 98% HNO_3 + 0.5% NO_2 + H_2O
Highly concentrated

Specific gravity : 1.5 – 1.6
Annoying and poisonous fumes

Generally used with kerosene, Hydrazine based propellants

Propellants - Oxidizers

Nitrogen Tetroxide (N_2O_4):

Yellow brown liquid with specific gravity of 1.44

Most widely used oxidizer in recent years. Hypergolic with many fuels

Mildly corrosive; Absorbs moisture

Narrow range of liquid phase: boiling point 21 C and freezing point -11.2 C

Freezing point can be lowered by adding NO (2-30%) ➔ Mixed Oxides of Nitrogen (MON)

Extremely toxic

Vapour pressure is high ➔ need of heavy tanks

Propellants - Fuels

Hydrocarbon Fuels (Kerosene – RP1):
Specially refined petroleum product for rocket

Specific gravity – 0.81
Mixture of saturated and unsaturated hydrocarbons
Used extensively with Lox

Liquid Methane (CH_4):
Heavier than H_2
Cryogenic propellant
Used only in experimental engines; not in commercial use

Liquid Hydrogen (LH2):

Extremely high performance with Fluorine or Lox

Very light fuel with SG – 0.07
Cryogenic fuel with boiling point at 20K

Heavy insulations required for tanks, lines, valves, pumps
Large condensation of air
Vacuum jacketing is provided to insulate

Scavenging of all the system is necessary before loading LH2

Propellants - Fuels

Hydrazine (N_2H_4):

Toxic, colourless liquid
Freezing point 274.3 K

Hypergolic with HNO# and N_2O_4

Also used as monopropellant

Highly stable during storage (~15 years)

Reacts with many materials

Carcinogenic

UDMH [$(CH_3)_2NNH_2$]: Unsymmetrical Dimethyl Hydrazine

More stable at higher temperatures
Freezing point – 215 K and boiling point – 336.5 K

Specific impulse slightly less than N_2H_4
Used in mixtures with Hydrazine (50-50 mixture is widely used – Aerozine used in lunar landing module)

MMH (CH_3NHNH_2): Monomethyl Hydrazine

Mainly used in spacecraft rockets for small thrust operations with N_2O_4
Highly toxic
All hydrazine vapours == explosive mixtures with air

Propellants - Performance

Typical performance values of various propellant combinations
at P1 = 50 atm and assuming optimum expansion at 1 atm

Propellant combination	T1 (K)	MW	c* (m/s)	Is (equilibrium) (sec)	MR
RFNA+UDMH	3048	23.6	1627	254.4	2.92
NTO+UDMH	3400	23.5	1722	272.6	2.61
LOx+RP1	3660	24	1777	284	2.73
LOx+LH2	2951	9.95	2429	374	4

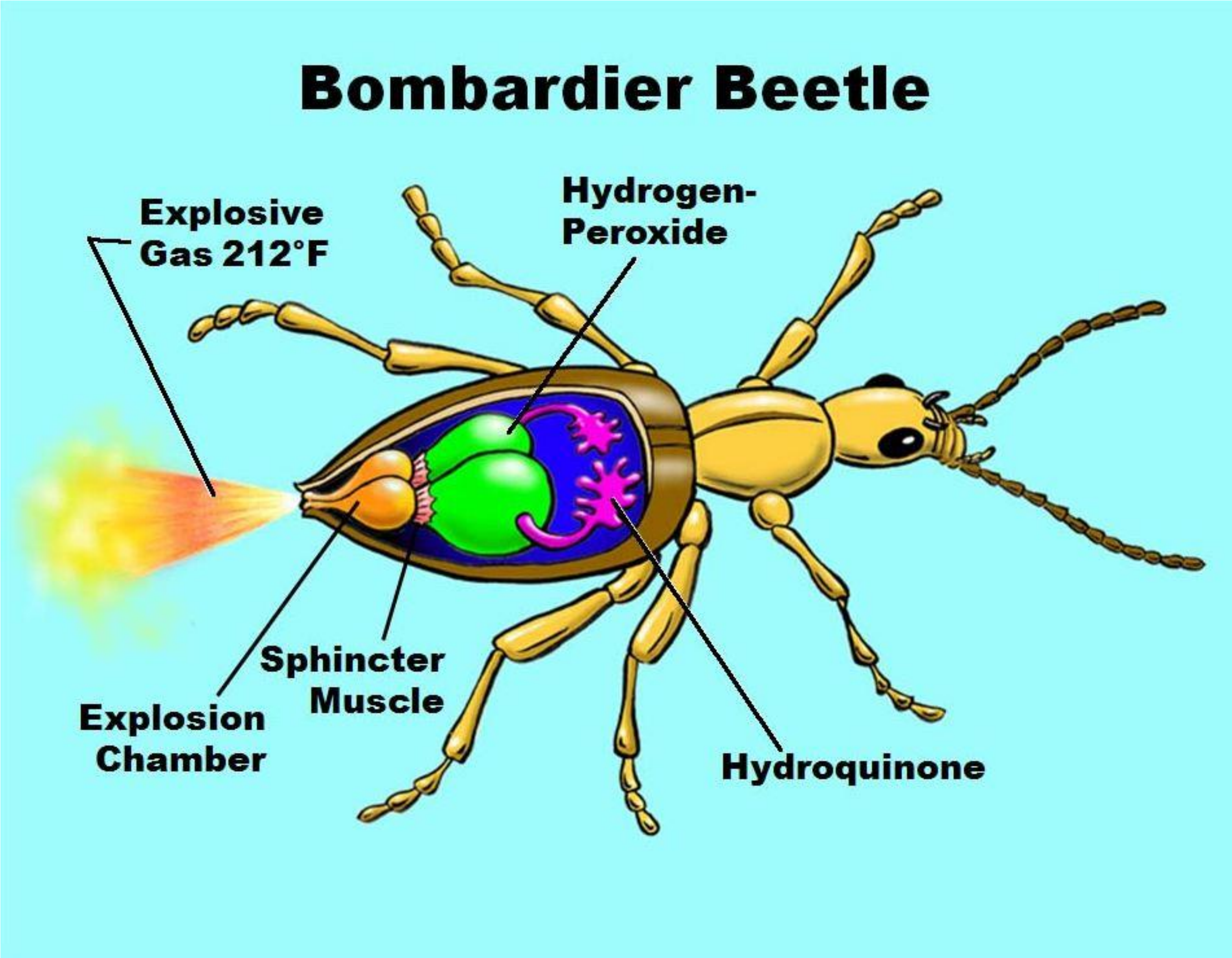
Propellants - Monopropellants

Single propellant decomposes spontaneously and exothermically under certain conditions or in the presence of a catalyst to form high temperature and high pressure gas

Used for small thrusters for auxiliary propulsion requirements or as a gas generator mechanism

Examples: Hydrazine, hydrogen peroxide

Monopropellant – Hydrogen peroxide



Monopropellant – Hydrogen peroxide

Hydrogen peroxide (H2O2) – Used as a monopropellant for auxiliary propulsion applications

$$H_2O_2 \rightarrow H_2O + (1/2)\ O_2 + \text{Heat}$$

Exothermic decomposition – temperature of the gases ~ 600 degree C

Decomposition – Using catalyst like liquid permanganates, solid manganese dioxide, Platinum, iron oxide

90% H2O2 typically generates a specific impulse of 154 sec

Clean exhaust

Monopropellant – Hydrazine

$N_2H_4 \rightarrow \alpha\, NH_3 + [1 - (\alpha/2)]\, N_2 + [2 – (3\,\alpha/2)]\, H_2$ + heat

Catalyst at room temperature – Iridium
Catalyst at elevated temperature (450 K) – Iron, Nickel, Cobalt

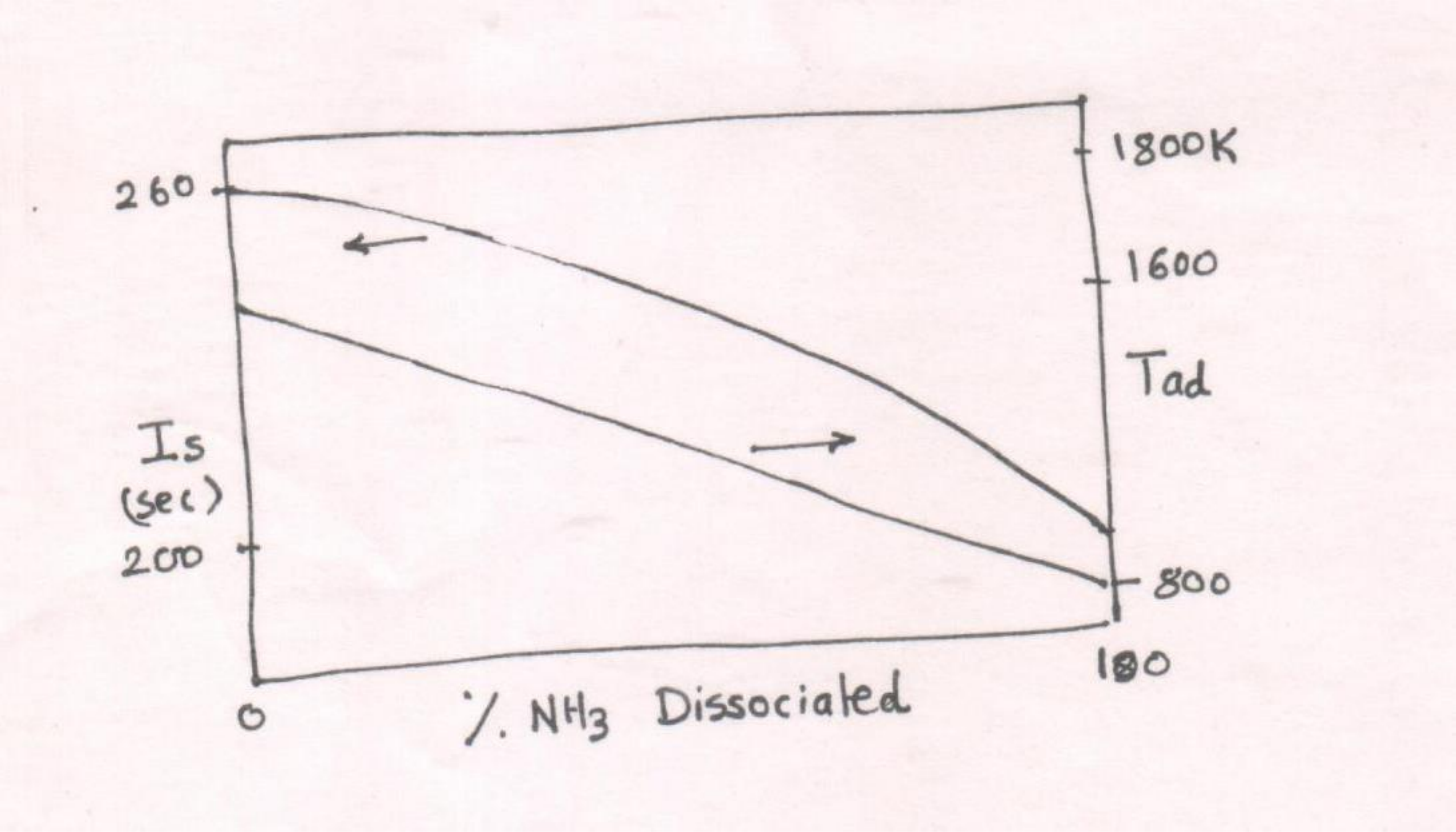
Decomposition of N2H4 leads to the mixture of NH3, N2
When α = 4/3, the products are only NH# and N2 → maximum flame temperature 0f 1649 K

At this high temperature, NH3 becomes unstable and starts to dissociate to form N2 and H2 (endothermic reaction)

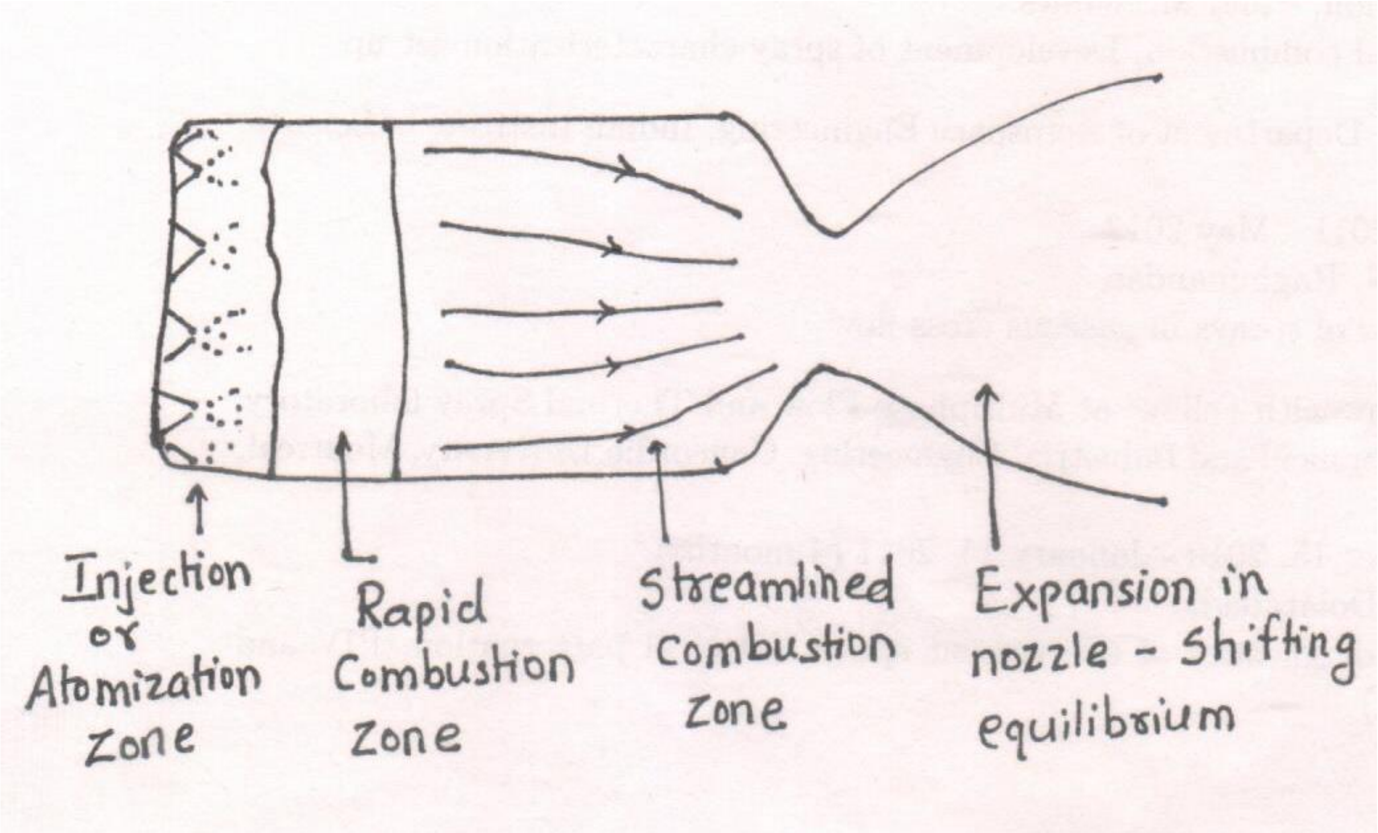
Under equilibrium condition, flame temperature achieved is 867 K

Usually, NH3 dissociates slowly and thus the final flame temperature and composition is governed by the amount of NH3 dissociated

Monopropellant – Hydrazine



Combustion in Liquid Propellant Rockets



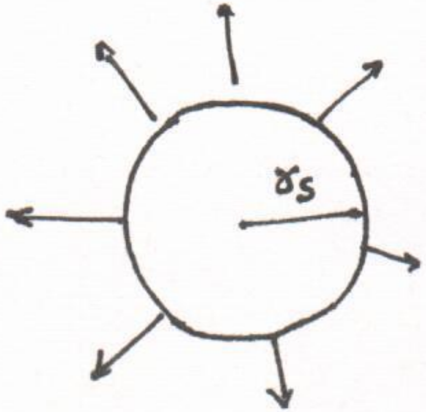
Important processes:

Propellant injection
Atomization
Evaporation
Mixing
Combustion
Expansion

Important zones:
Injection/atomization zone
Rapid combustion zone
Streamline combustion zone

Rate limiting process: Droplet evaporation

Propellant combustion – Droplet evaporation



Mass flow of fuel from surface,

$$\dot{m}_F = 4\pi r_s^2 \underbrace{\dot{m}_F''}_{\text{Mass flux}}$$

Y_F = Mass fraction of fuel

$$\Rightarrow \dot{m}_F'' = \underbrace{Y_F\left(\dot{m}_F'' + \dot{m}_{ox}''\right)}_{\text{Convective flux}} - \underbrace{\rho \mathcal{D} \frac{d}{dr} Y_F}_{\text{Diffusive flux}}$$

$\dot{m}_{ox}''$ = Oxidiser mass flux = 0

$\mathcal{D}$ = Mass diffusivity

Propellant combustion – Droplet evaporation

$\mathcal{D}$ = Mass diffusivity

$$\dot{m}_F = -\frac{4\pi r^2 \rho \mathcal{D}}{1-Y_F}\frac{dY_F}{dr} \Rightarrow \int_{Y_{F_s}}^{Y_{F\infty}} \frac{-dY_F}{1-Y_F} = \int_{r_s}^{\infty} \frac{\dot{m}_F}{4\pi\rho\mathcal{D}}\frac{dr}{r^2}$$

$$\dot{m}_F = 4\pi\rho\mathcal{D}\, r_s \ln\left[\frac{1-Y_{F\infty}}{1-Y_{Fs}}\right]$$

Transport number, $B = \dfrac{Y_{Fs}-Y_{F\infty}}{1-Y_{Fs}} \Rightarrow \dot{m}_F = 4\pi\rho\mathcal{D}\, r_s \ln(1+B)$

Propellant combustion – Droplet evaporation

$$m_F = \frac{\rho_L \pi D^3}{6} \Rightarrow \dot{m}_F = \frac{\rho_L \pi D^2}{2}\frac{dD}{dt} \Rightarrow \frac{dD^2}{dt} = \frac{4\dot{m}_F}{\rho_L \pi D} = \lambda$$

$$\lambda = \text{Evaporation constant} = \frac{8\rho\mathcal{D}}{\rho_L}\ln(1+B) \quad [mm^2/s]$$

$$\text{Evaporation time, } \tau_v = \frac{D_0^2}{\lambda} = \frac{D_0^2\,\rho_L}{8\rho\mathcal{D}\ln(1+B)}$$

$$\tau_v \sim d_0^2$$

$$\tau_v \sim 25\times10^4\, d_0^2$$

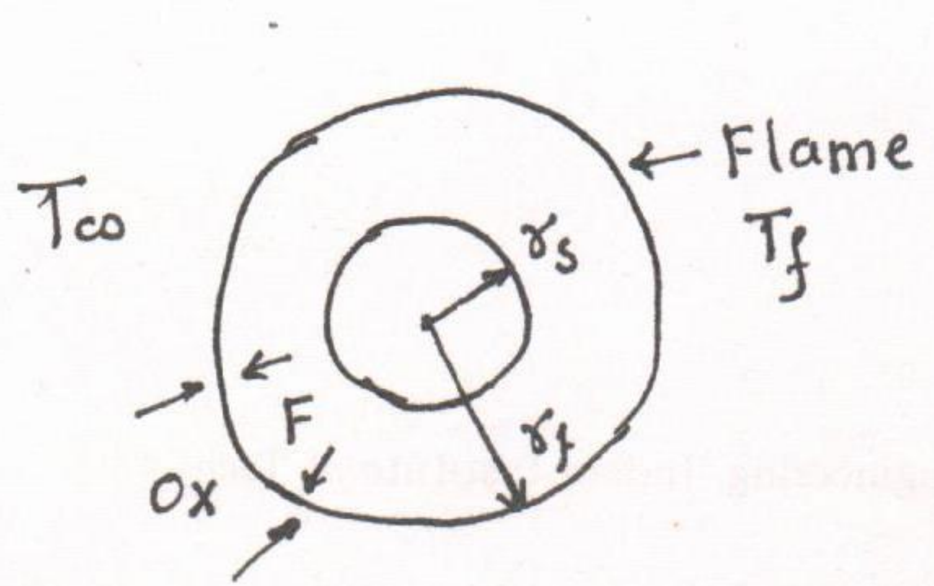
For $d_0 = 100\ \mu m$

$$\tau_v \approx 2.5\ ms$$

D2-law of droplet evaporation: The time required for complete evaporation of a droplet is directly proportional to the square of its initial diameter

Same law holds for the droplet combustion phenomenon

Propellant combustion – Droplet combustion



Fuel	Burn rate constant (mm^2/s) at ambient
Kerosene	1
Hydrazine	21

For Burning droplet, λ = Burn rate constant

Transport number, $B = \dfrac{C_p\,(T_\infty - T_L) + Q\,Y_F}{L}$

Rocket Engine Injectors or Atomizers

Important functions:

To provide the metered flow of the propellants to the combustion chamber as demanded by the thrust requirements

To atomize the liquid propellants to facilitate the efficient combustion of the propellants

Process of Atomization

Conversion of the bulk propellant into the spray consisting of fine droplets

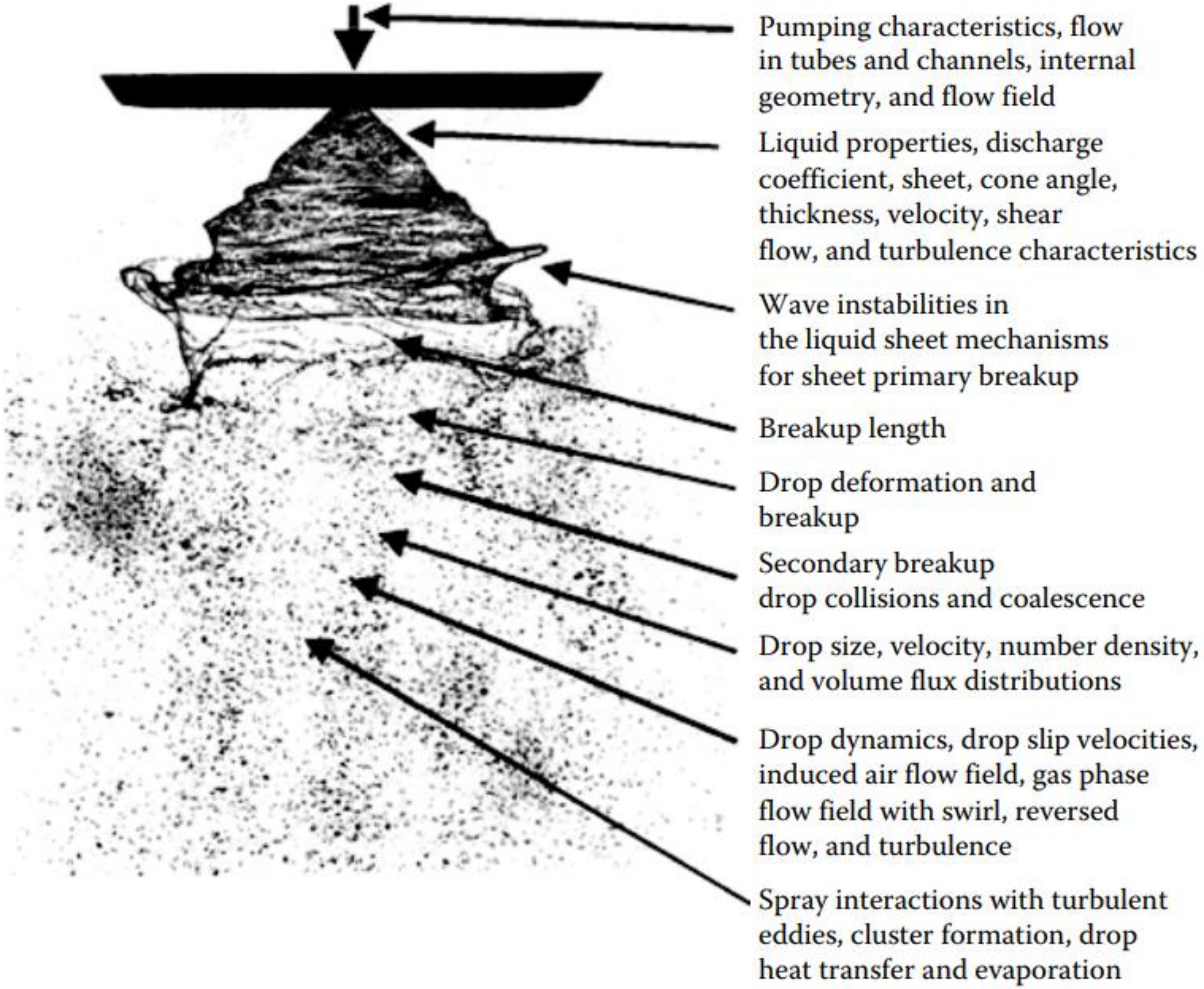
Importance: Droplets increase the surface area of the liquid exposed to high temperature → more heat transfer to the liquid → faster evaporation

Important for the mixing of the fuel and oxidizer propellants

Reduces the time required for the complete combustion of the propellants in the available chamber size

Use of external source of energy such as pressure/kinetic energy, impact, shearing by the high speed gas stream → to overcome the surface tension forces and breakup the liquid into droplets

Process of Atomization



Process of Atomization

Important spray characteristics:

Discharge coefficient: Ratio of actual flow rate to the theoretical flow rate through the orifice

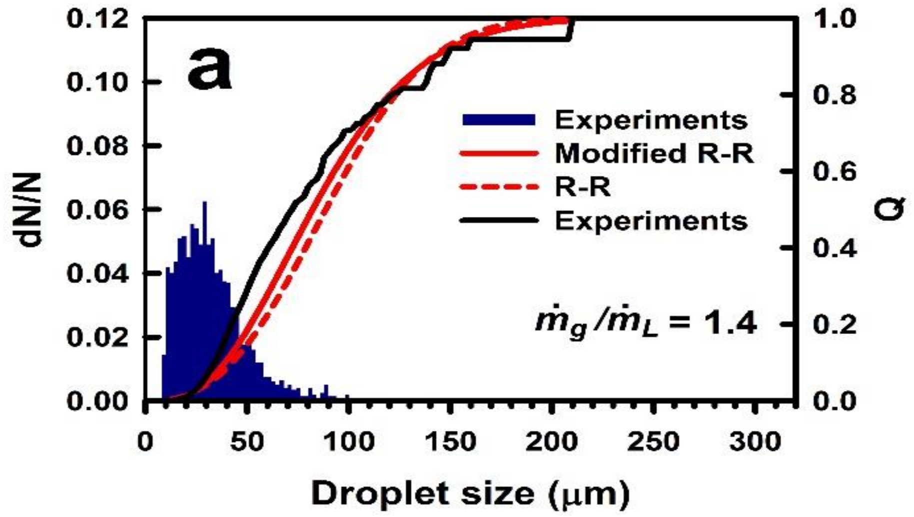
Spray cone angle

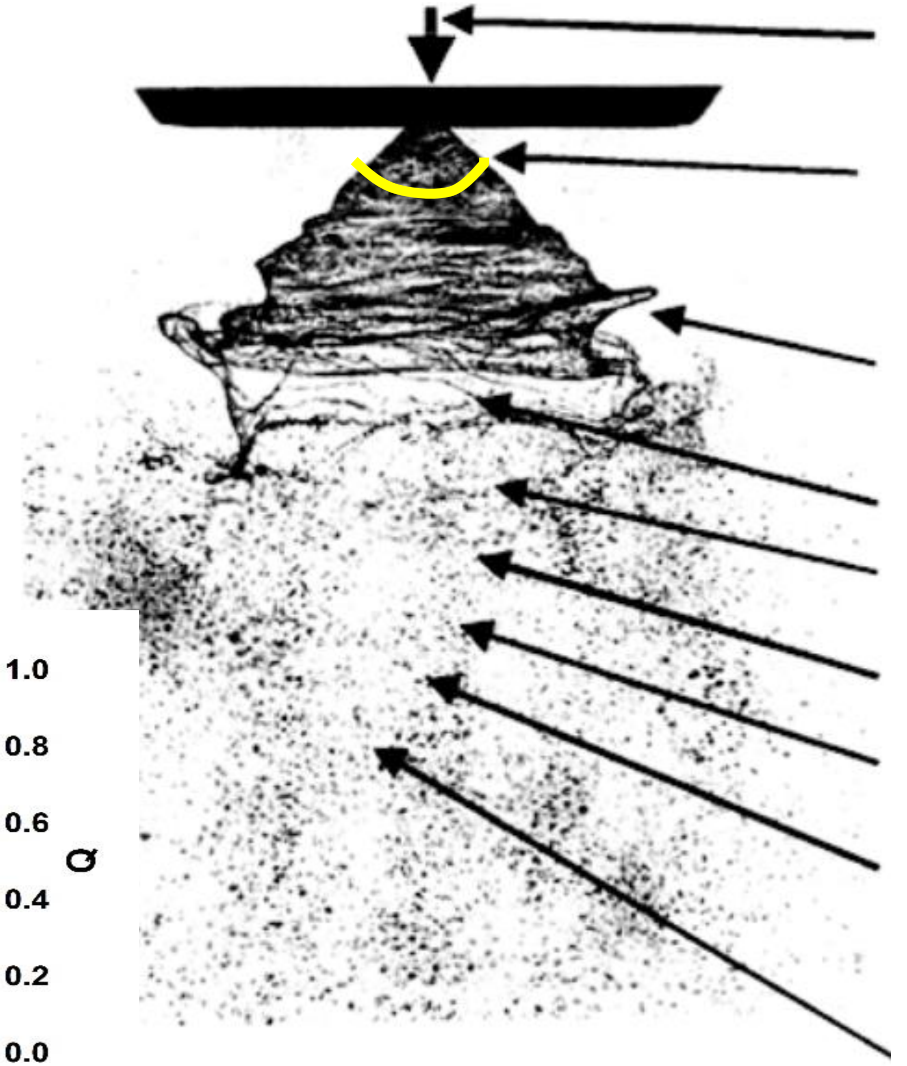
Drop size distribution

Mean drop size : Sauter Mean Diameter

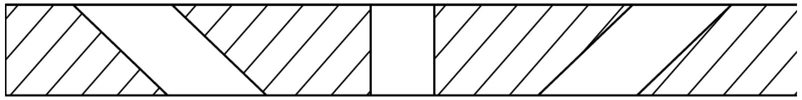
$$SMD = \frac{\sum n_i D_i^3}{\sum n_i D_i^2}$$

Liquid mass distribution

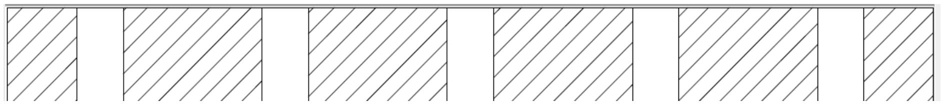




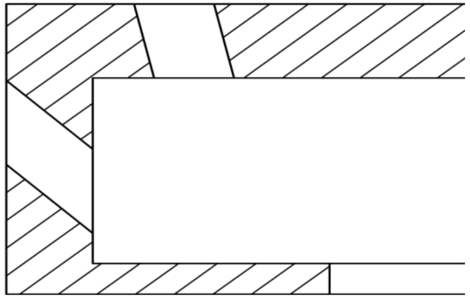
Rocket Engine injectors



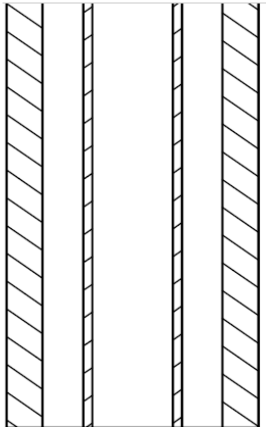
Impinging Jet (types: Like-like, Like-unlike, doublet, triplet)



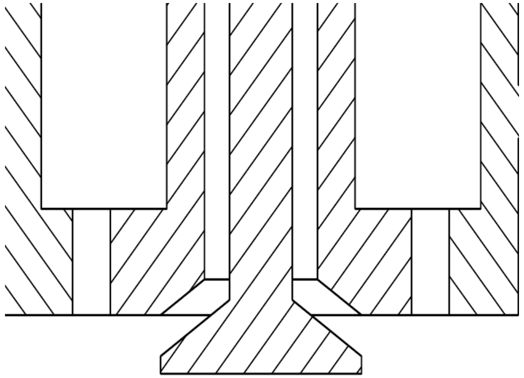
Showerhead



Splash plate



Concentric tube (types: shear coaxial, swirl coaxial(LCSC, GCSC))



Pintle(types: single face valve, dual face valve)

Showerhead injector

Use of high pressure or kinetic energy of the jet to obtain the spray

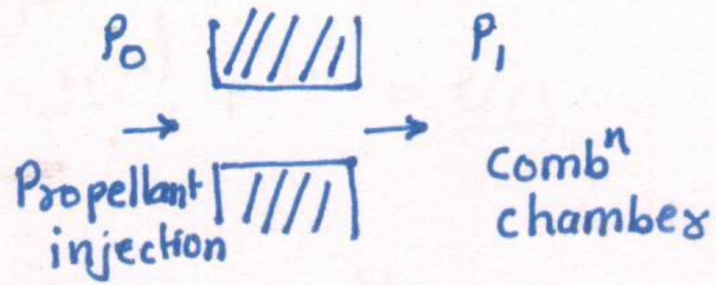
A series of slender parallel jets of liquid fuels and oxidizers (similar to a bath shower)

Important parameters: Coefficient of discharge and the L/D ratio of the orifice

Usually small L/D ratio (between 3-5) is used and the orifices are sharp edged

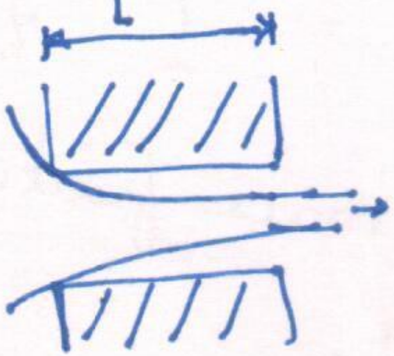
Showerhead injector

$$\dot{m} = C_d A \sqrt{2 \cdot \Delta p \cdot \rho}$$

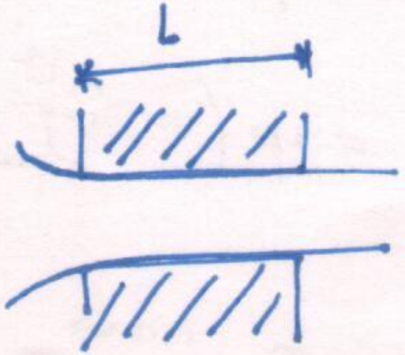


$$\Delta p = P_0 - P_1$$

C_d = Coefficient of discharge



Separated flow $C_d = 0.65$



Attached flow

$C_d = 0.9 - 0.98$

Showerhead injector

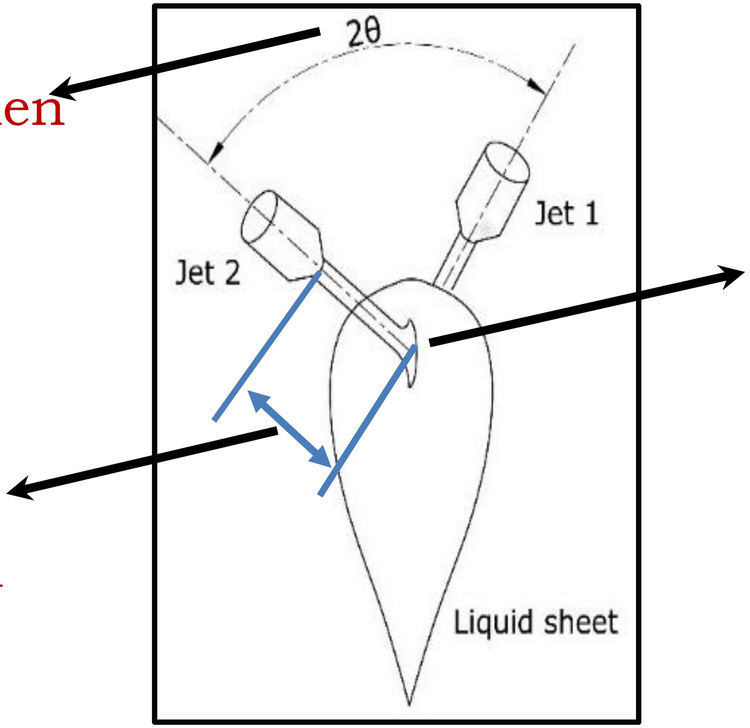
$$\dot{m}_{ox} = n_{ox} \cdot C_d \cdot A_{ox} \sqrt{2(\Delta P)_{ox} \cdot \rho_{ox}}$$

$$\dot{m}_F = n_F \cdot C_d \cdot A_F \sqrt{2(\Delta P)_F \cdot \rho_F}$$

n_{ox} & n_F
No. of orifices of oxidizer & Fuel resp.

$$MR = \frac{n_{ox} \cdot C_d \cdot A_{ox} \sqrt{2(\Delta P)_{ox} \cdot \rho_{ox}}}{n_F \cdot C_d \cdot A_F \sqrt{2(\Delta P)_F \cdot \rho_F}}$$

Impinging Jet Injector



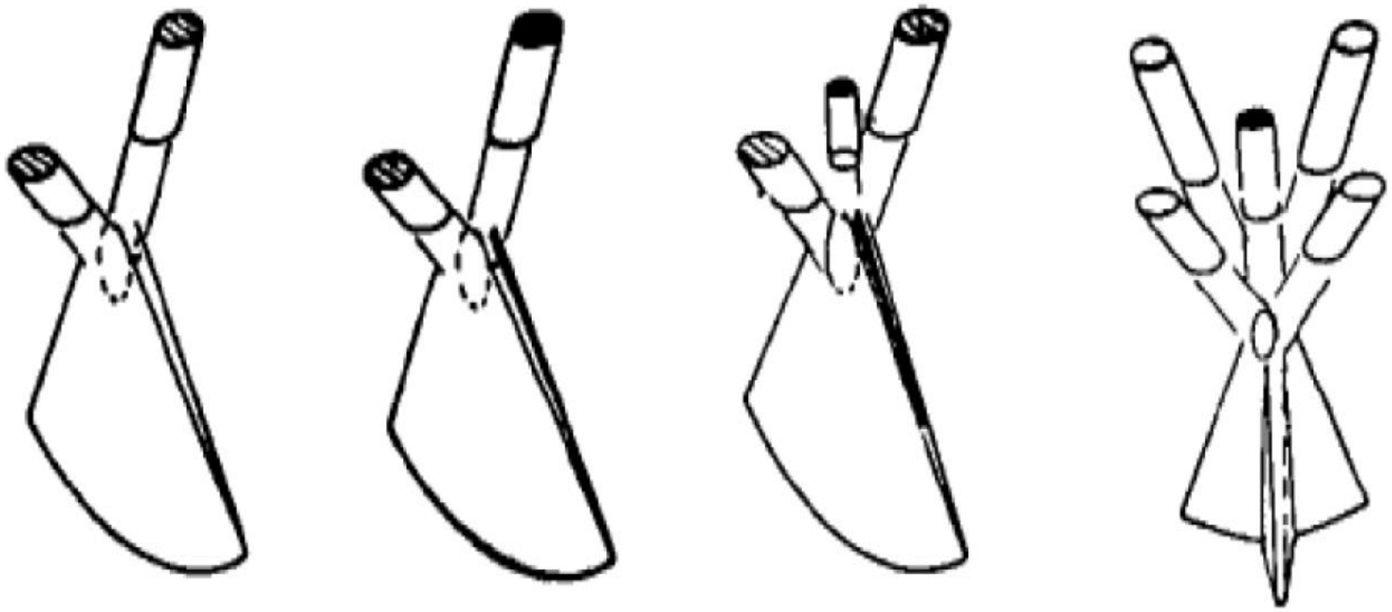
Impact of two or more liquid jets along with kinetic energy of the jets is used of achieving atomization

Impingement of two liquid jets forms a liquid sheet which breaks up to form droplets

Used in case of propellants which enter the combustor in liquid phase

Important operating parameters are impingement angle, momentum of two colliding jets and the diameters of liquid jets

Impinging Jet Injector



Like-on-Like Doublet

Unlike Doublet

Triplet

Pentad

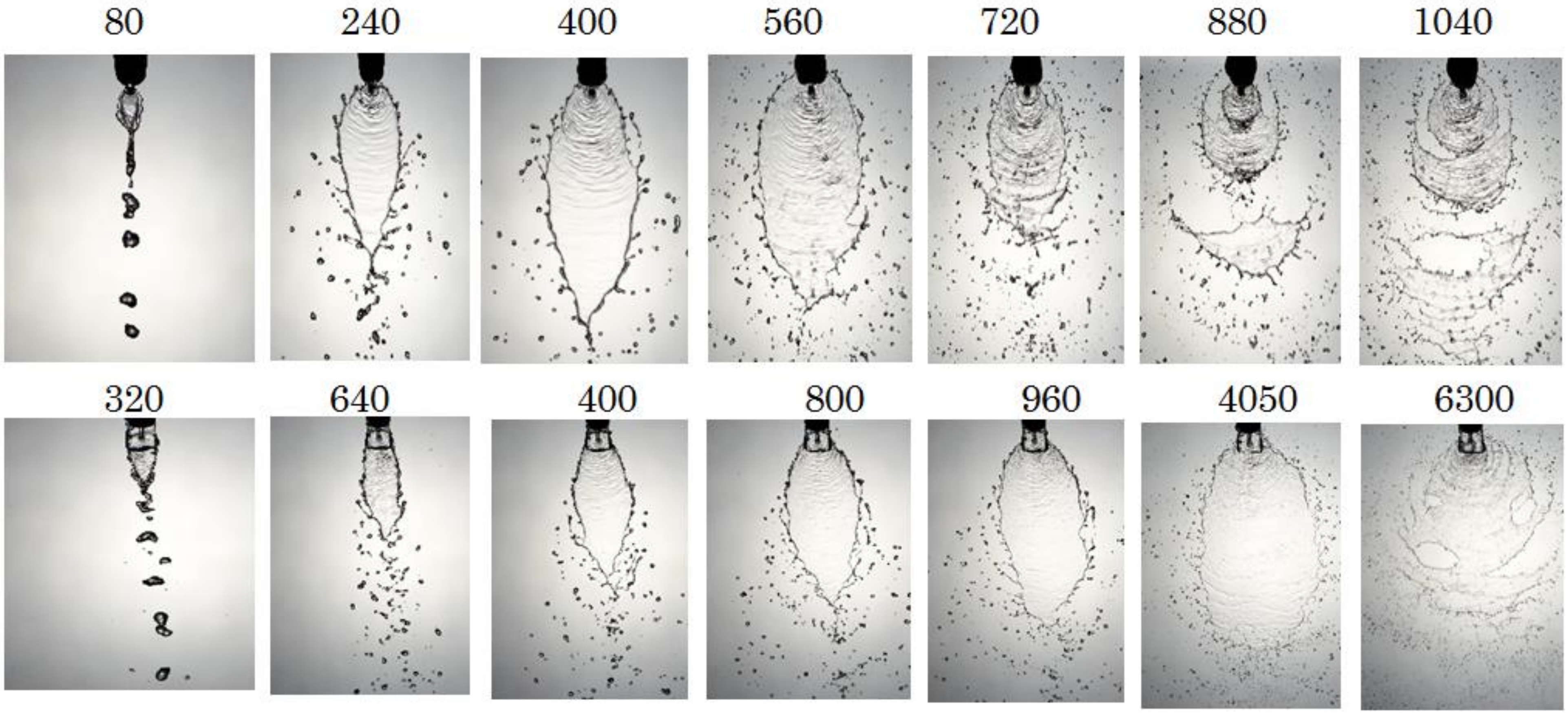
Like impingement – Both the colliding jets are of same liquid

Unlike impingement – the colliding jets are of different liquids

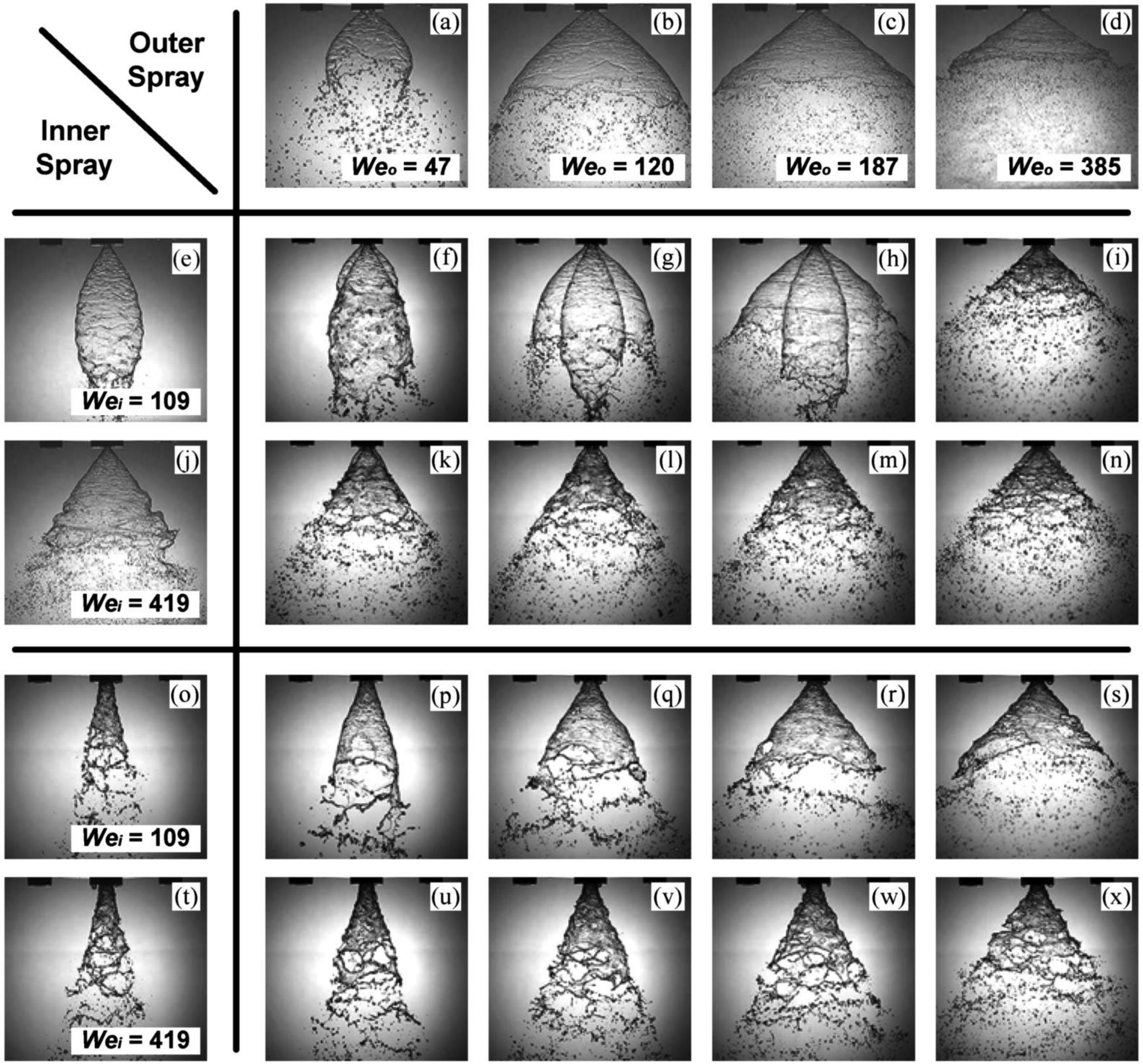
Some times impact of a liquid jet on a flat solid surface is also used – Splash plate atomizer

Easy to fabricate, good atomization and mixing

Impinging Jet Injector



Coaxial injectors



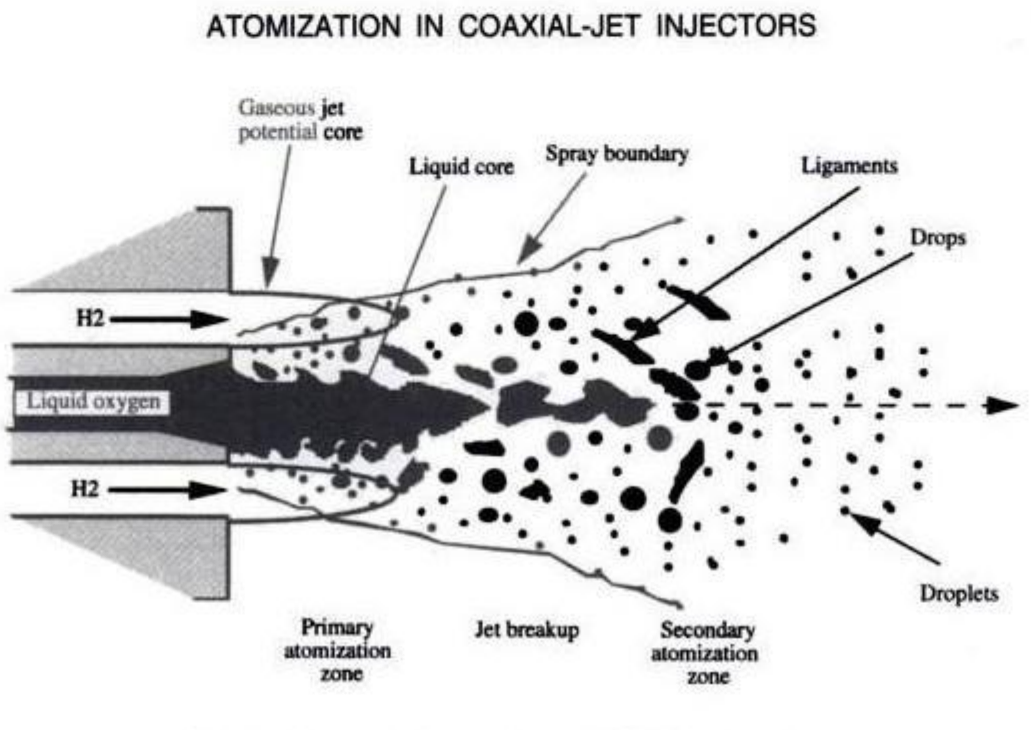
Liquid-Liquid coaxial atomizers

Both the liquid propellants are flowing through the single injector element in an axial direction

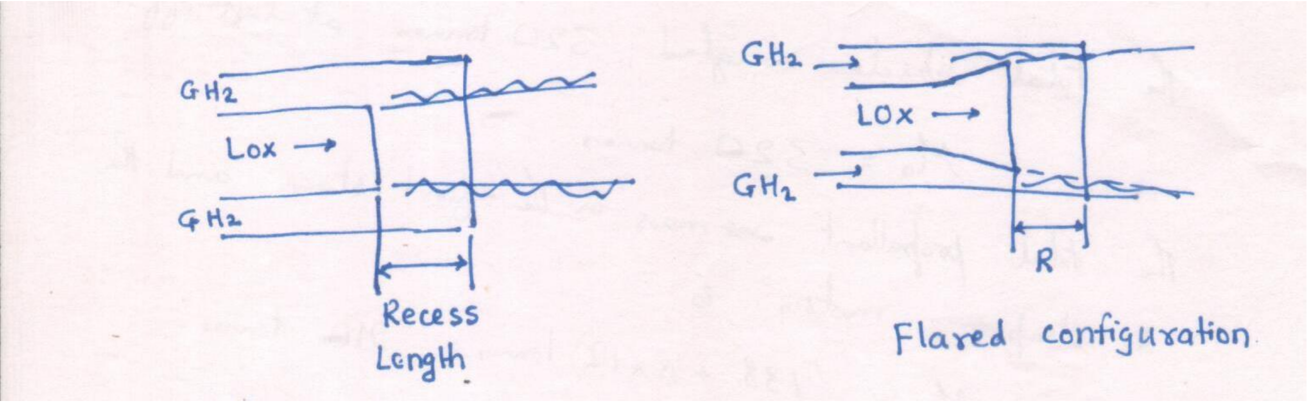
Usually the interaction of liquid sheets (inner and outer or annular) is used to atomize and mix the propellants

The liquid sheets are created by methods such as,
Using a shaped body in the orifice exit
Using swirling motion

Coaxial injectors



General scheme of coaxial jet disintegration.

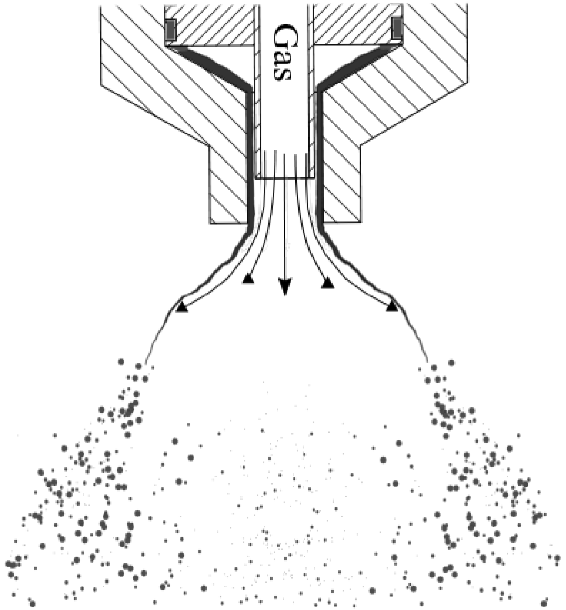


Gas-liquid shear coaxial atomizers

Employed for cryogenic engines where one propellant enters in liquid phase and other in gas phase

Low velocity central LOx jet and high speed GH2 annular jet

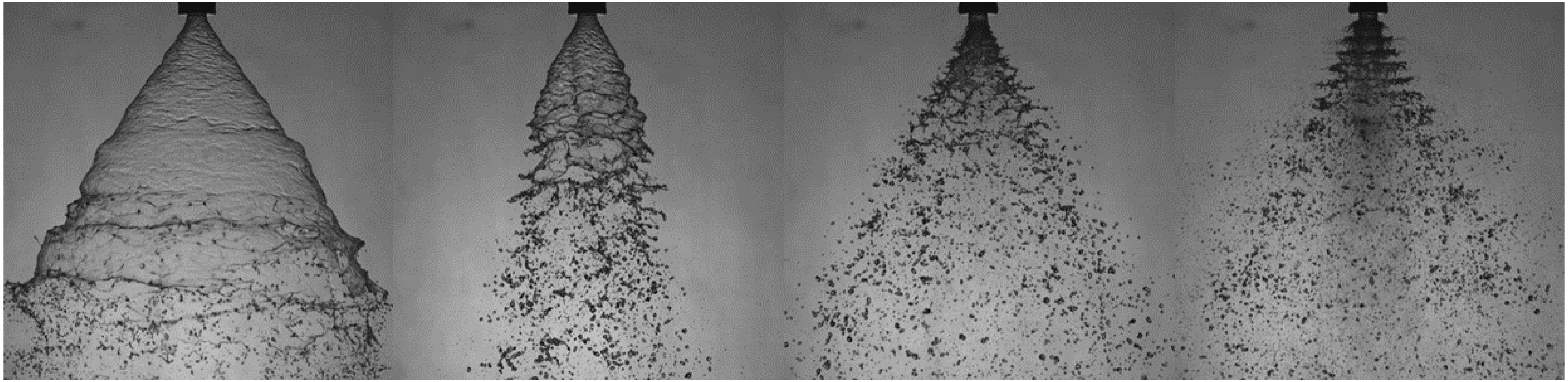
Atomization is realized by the shearing action at the interface



Coaxial injectors

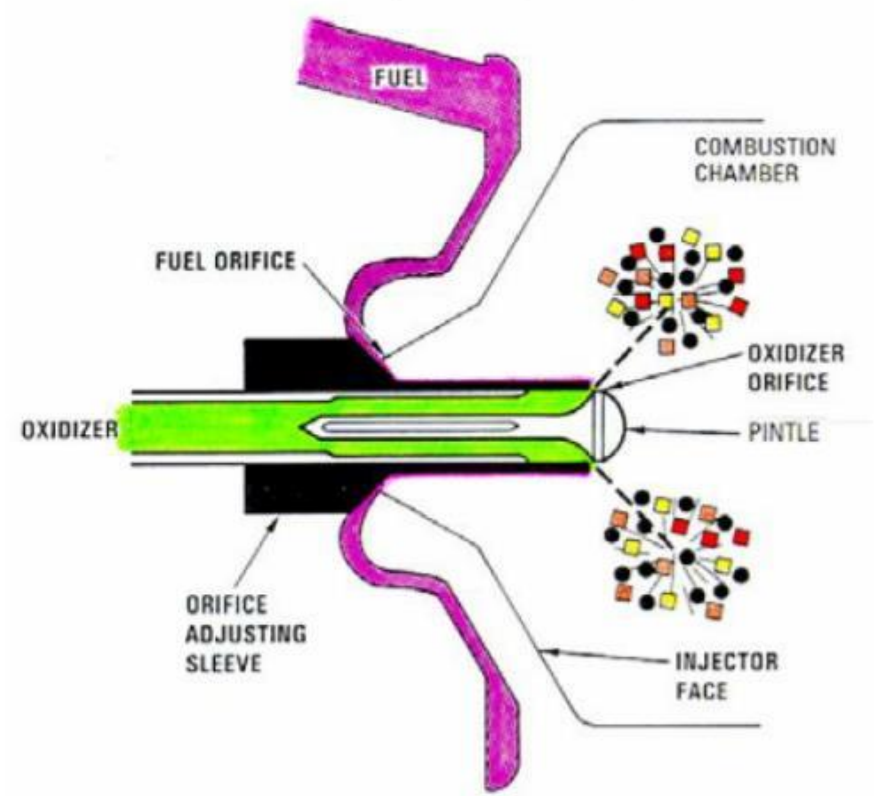
Gas-liquid swirl coaxial atomizer: Central gas jet and swirling annular liquid sheet

Used for semi-cryogenic or cryogenic engines

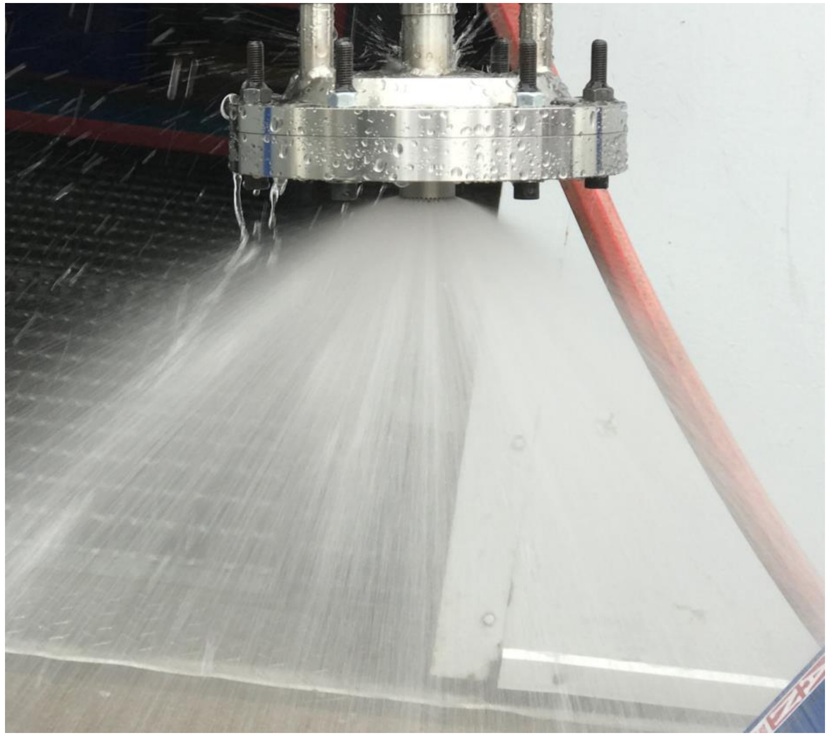


Pintle injector

Pintle Injector (LMDE)



Dressler, G. A., "Summary of Deep Throttling Rocket Engines with Emphasis on Apollo LMDE," AIAA Paper 2006-5220

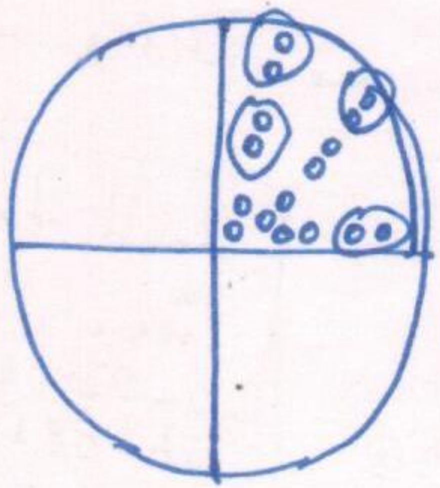


Specifically used for deep throttling rocket engine (for instance, engine used for lunar landing module)

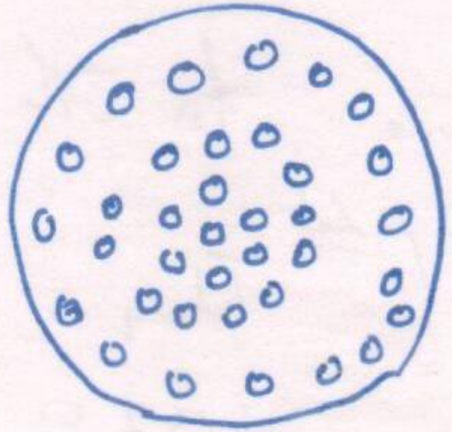
Thrust is varied significantly by changing the flowrate of propellants

Important advantage: Flowrates can be varied by varying flow passage areas without compromising the atomization quality

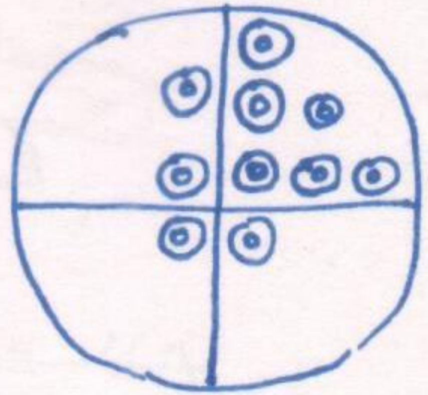
Injector Plate



Injector plate (Impinging)



Injector plate (showerhead)



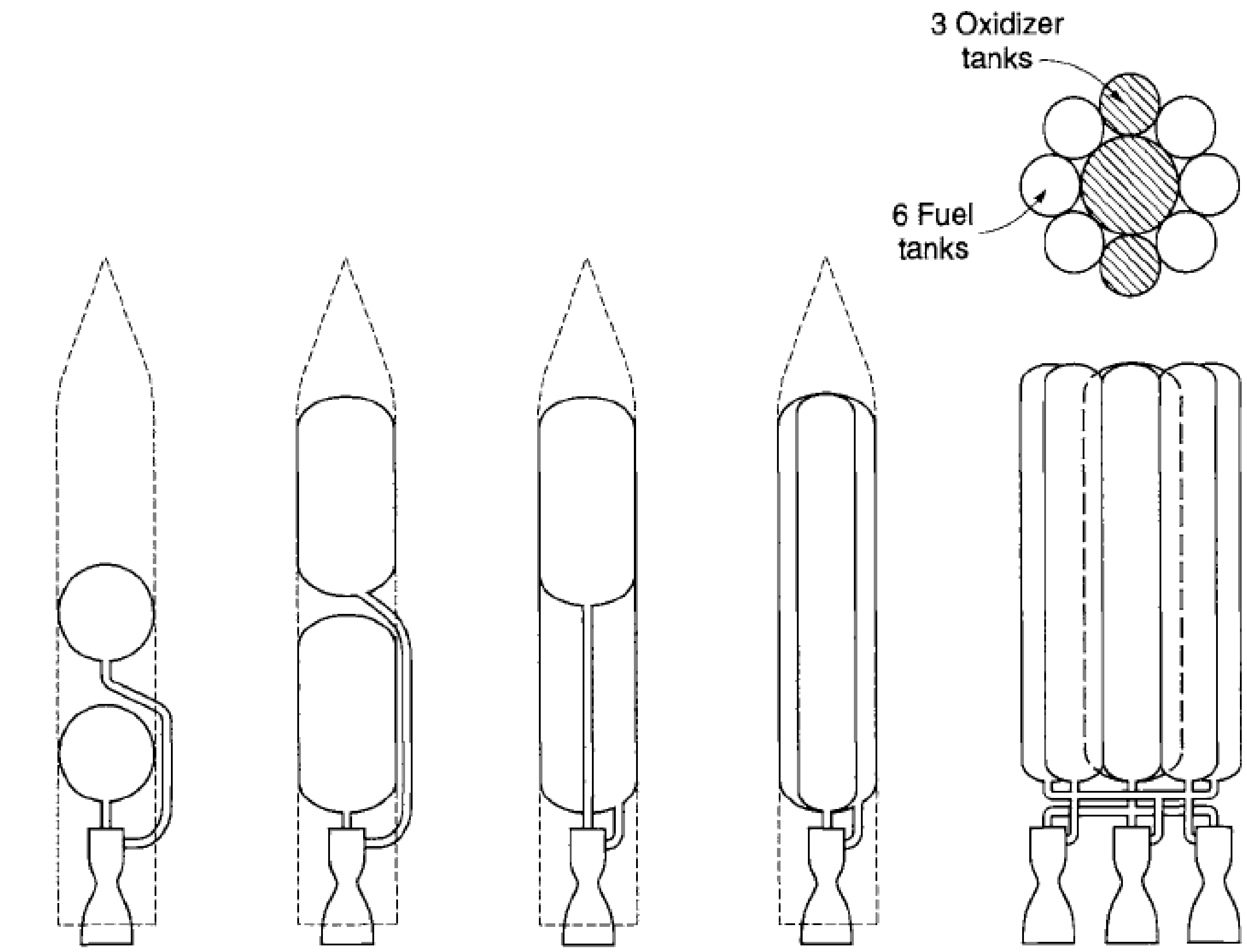
Coaxial

Thrust chamber sizing

Two important parameters:
Ratio of combustion chamber cross-sectional area to throat area (Ac/At) → Practical range 1-3
Reduction in this ratio → reduction in total surface area to be cooled and increase in stagnation pressure loss

Characteristic length → L* = Ratio of combustion chamber volume to throat area (Vc/At) → Practical length 0.8 – 1

Propellant tanks



Spherical tanks | Tandem tanks, external piping | Tandem tanks with common bulkhead, internal piping | Concentric tanks | Multi-tank

Various tank configurations

Common tank materials are aluminum, stainless
steel, titanium, alloy steel, and fiber-reinforced plastics

The extra volume of gas above the propellant in sealed tanks is called ullage.

It is necessary space that allows for thermal expansion of the propellant liquids, pressurization gas, etc

The ullage volume is usually between 3 and 10% of the tank volume.

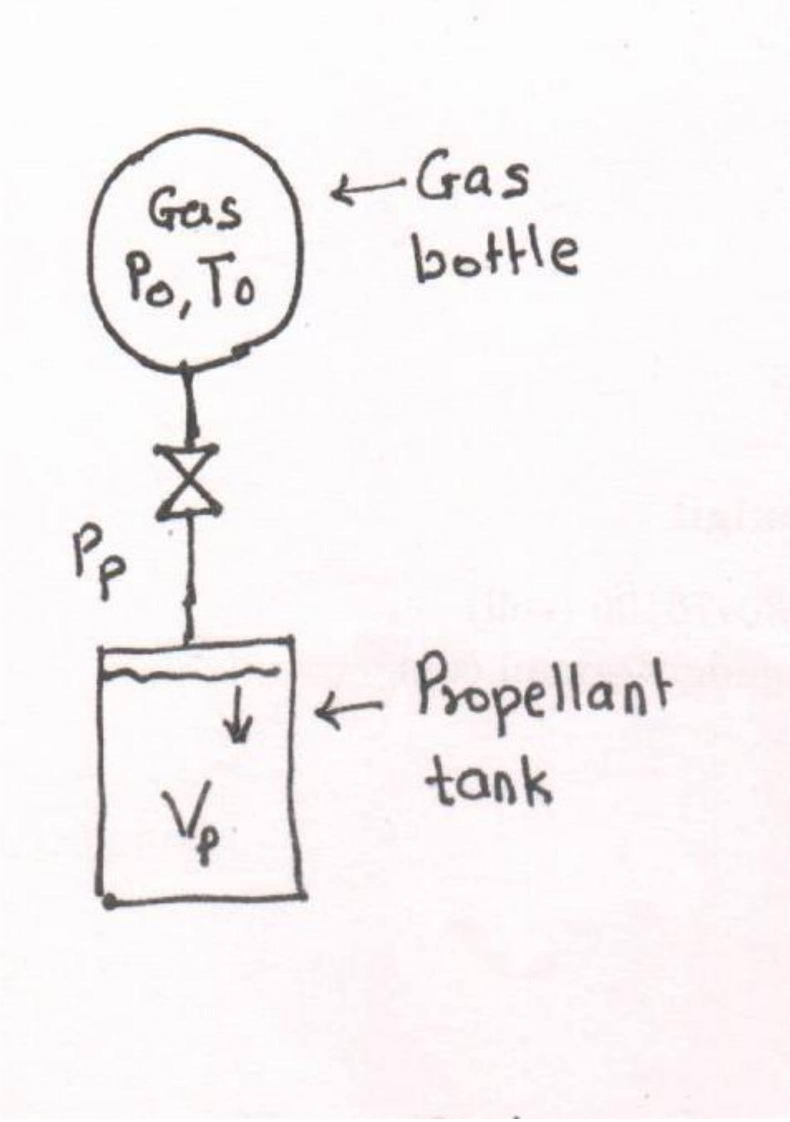
Propellant tank pressurization

For pressurized feed systems the propellant tanks typically operate at an average pressure between 1.3 and 9 MPa. These tanks have thick walls and are heavy.

For high-pressure gas (used to expel the propellants) the tank pressures are much higher, typically between 6.9 and 69 MPa These tanks are usually spherical for minimum inert mass.

For turbopump feed systems it is necessary to pressurize the propellant tanks slightly (to suppress pump cavitation) to average values of between 0.07 and 0.34 MPa. These low pressures allow thin tank walls, and therefore turbopump feed systems have relatively low tank weights.

Estimation of pressurizing gas



High pressure inert gas bottles for tank pressurization

Propellant pressure Pp with the tank volume of Vp

This pressure is maintained with the use of a pressure regulator – P0, T0 (initially)

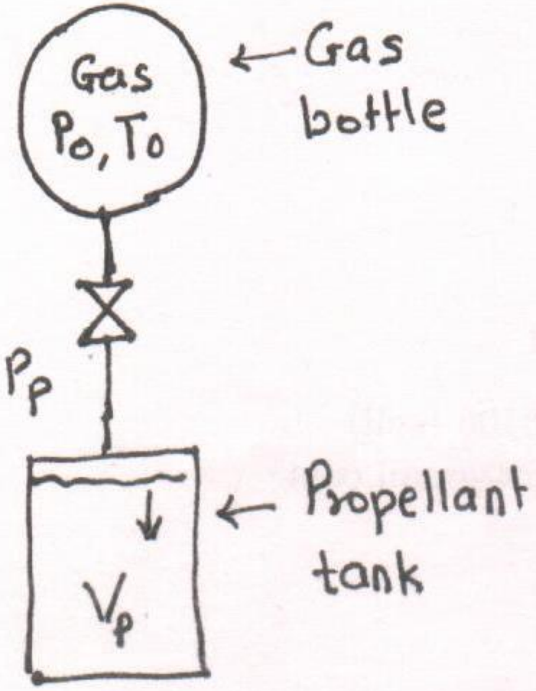
Depletion of propellant → expansion of the gas in gas bottle → Fall in temperature

When all the propellant is expelled → the gas expands in the additional volume of Vp at pressure Pp

The final temperature in the gas bottle – Tr

This is an adiabatic process → change in the internal energy is equivalent to the work done

Estimation of pressurizing gas



~~Change in~~

$$-dU = dW$$

$$-\left[(m_\gamma C_v T_\gamma + m_p C_v T_p) - m_o C_v T_o\right] = P_p \cdot V_p$$

But $m_\gamma = \dfrac{P_\gamma V_o}{R \cdot T_\gamma}$ & $m_p = \dfrac{P_p V_p}{R \cdot T_p}$

Estimation of pressurizing gas

$$m_0 = \frac{P_p V_p}{C_v T_0} + \frac{P_p V_p}{R T_0} + \frac{P_r V_0}{R T_0}$$

$$C_v = \frac{R}{k-1}$$

$$m_0 = \frac{P_p V_p}{R T_0} + (k-1)\frac{P_p V_p}{R T_0} + \frac{P_r V_0}{R T_0}$$

Putting $V_0 = \dfrac{m_0 R T_0}{P_0}$

$$m_0 = \frac{P_p V_p}{R T_0}\left[\frac{k}{1 - \frac{P_r}{P_0}}\right]$$

Mass of pressurizing gas is

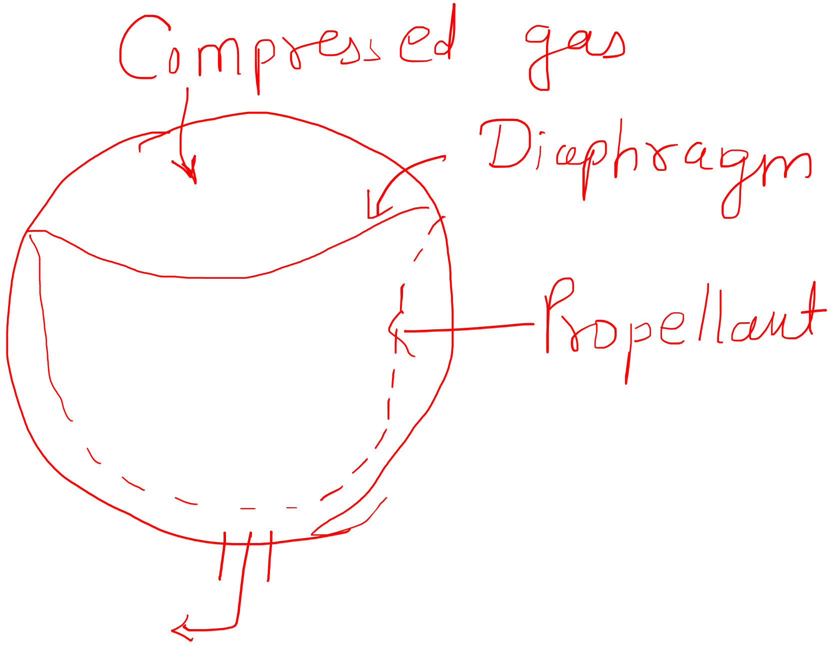
Proportional to the tank pressure and volume

Inversely proportional to the initial gas bottle temperature

Proportional to the molecular weight of the gas

Inversely varies with the ratio (Pr/P0)

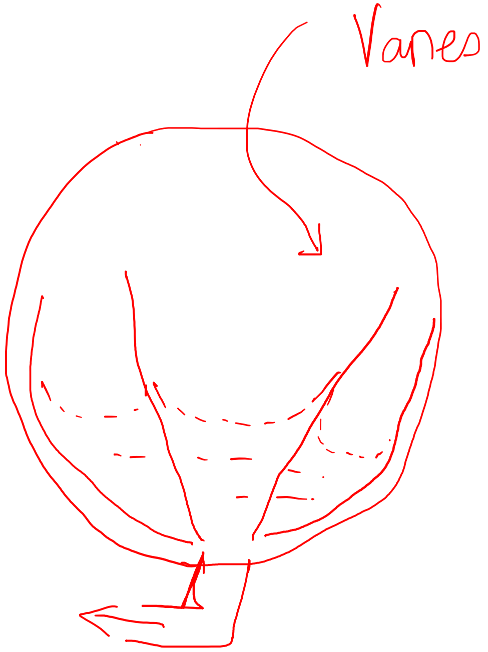
Propellant discharge mechanisms



Positive expulsion device

For effectively utilizing the propellant particularly in microgravity environment

To reduce the unused propellant



Surface tension device

Turbomachinary requirements

Choice of turbomachines → Type of propellant, feed cycle, combustion chamber pressure, thrust levels

For compressors → centrifugal or axial turbomachines are preferred

For turbines → Mostly single stage impulse turbine is used

Typical rotational speeds → 20000 – 50000 rpm or even higher due to smaller dimensions

Cavitation → A major concern on the suction side of the propellant pumps
Suction pressure < vapour pressure of the propellant

Means to avoid → pressurization, subcooling, booster pump

Propellant tank pressures → 2 – 5 atm

Turbomachinary requirements

$$\Delta P = f\left(\omega, \dot{Q}, \rho, d_i\right) \quad \text{where} \quad \omega = \frac{2\pi N}{60} \quad (N = RPM)$$

$$\frac{\Delta P}{\rho \omega^2 d_i^2} = f\left[\underbrace{\frac{\omega \cdot \sqrt{\dot{Q}}}{(\Delta P/\rho)^{3/4}}}_{\text{Specific Speed } (N_s)}, \underbrace{\frac{d_i \, (\Delta P/\rho)^{1/4}}{\dot{Q}^{1/2}}}_{\text{Specific diameter } (D_s)}\right]$$

Net Positive Suction Head (NPSH) $= \dfrac{P_0 - P_v}{\rho g}$

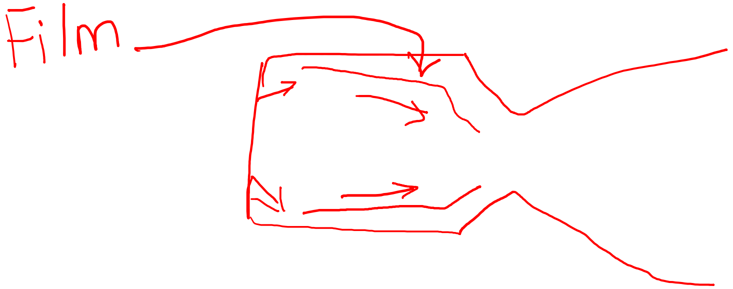
Suction Specific speed $\Rightarrow$ $N_{ss} = \dfrac{\omega \sqrt{\dot{Q}}}{(NPSH \cdot g)^{3/4}}$

Thrust chamber cooling

Heat transfer (cooling) heat flux in a typical high thrust engine like SSME → 35.6 MW/m^2

Film cooling:

Liquid propellant in injected onto the thrust chamber wall through the injectors mounted on the periphery (File generally flows in tangential direction)
Typical injection hole diameters 0.2 – 5 mm and spacing 5 – 10 mm
A relatively cold layer of vapour separates the high temperature region and wall
In cryogenic engine, the cold gas layer is usually provided by the injection of hydrogen gas
Film cooling is always used along with other methods of cooling



Heat sink and radiation cooling:

Heat transferred and stored in the engine hardware
Materials used are capable of storing heat by attaining and sustaining high temperature
Useful for smaller burn times and used for engines < 100 kN thrust
Typical wall temperature 1500 – 1700 K

Radiation cooling → heat transfer by radiation ~ T^4
Materials → Molebdenum, Tungsten, Columbium
Emissivity of outer surface ~ 0.7 – 0.85
Very useful in outer space environment

Ablative cooling:

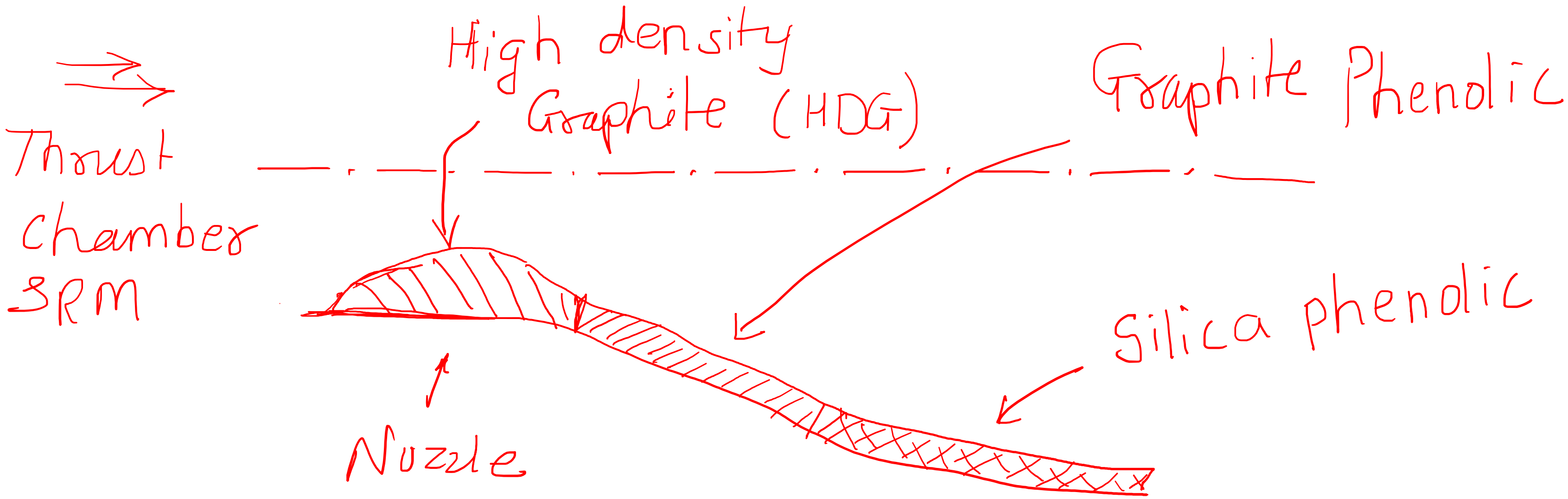
Removal of solid material due to flowing gas stream

Wall material is sacrificed by melting, vaporization and chemical changes to dissipate heat.

As a result, relatively cool gases flow over the wall surface, thus lowering the boundary-layer temperature and assisting the cooling process.

Very effective and used frequently in solid rocket motors

Materials→ High density graphite, Asbestos-phenolic, Silica-phenolic, Carbon-carbon cloth

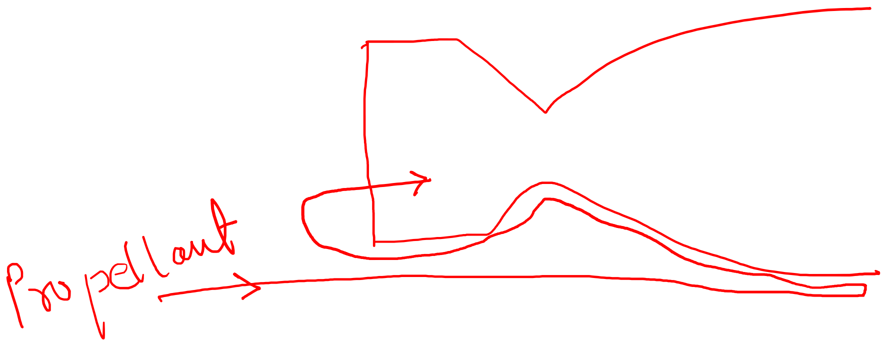


Regenerative cooling:

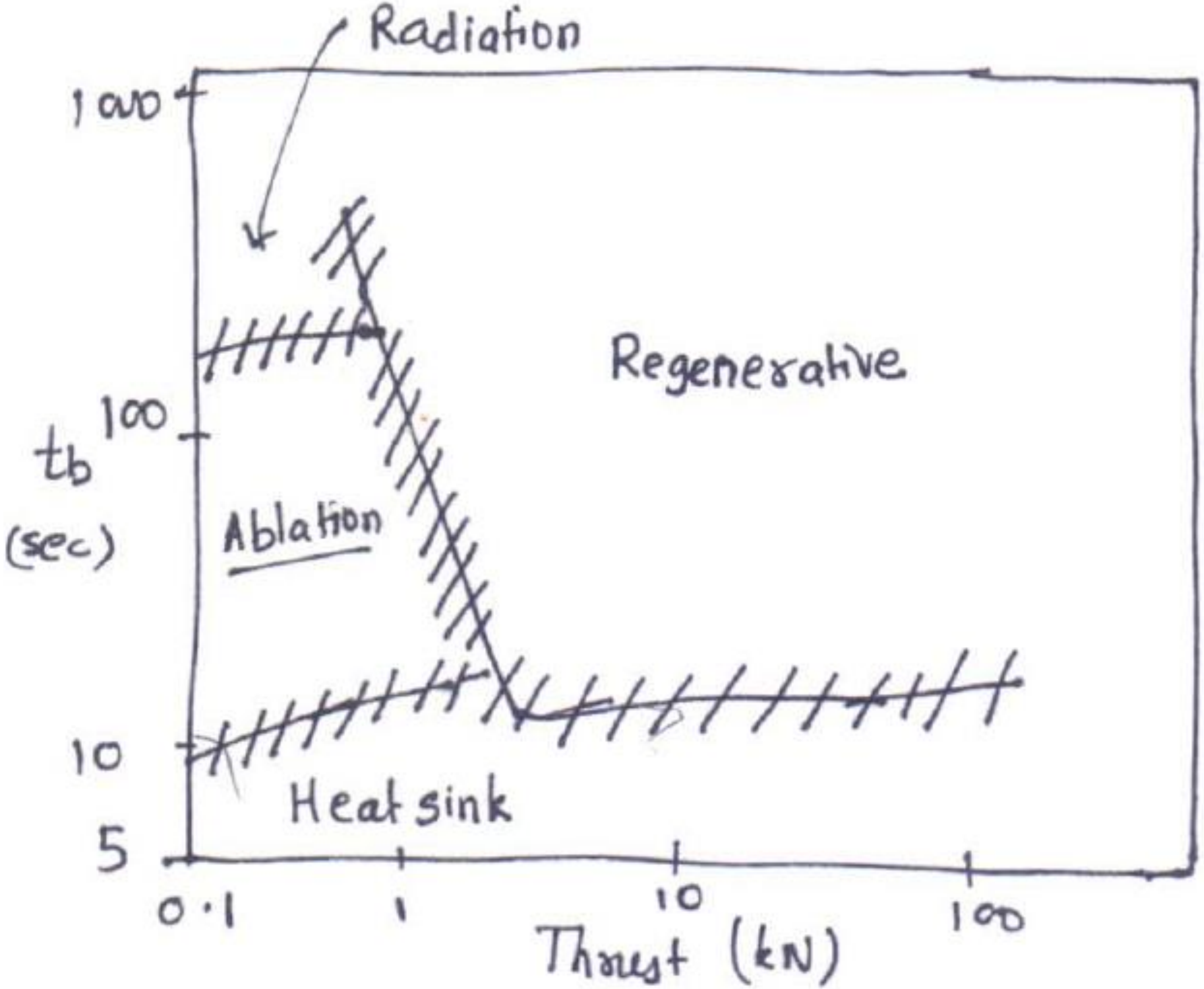
Mainly used in large thrust engines having single long burn time
Liquid propellant is passed through the cooling jacket around the engine before entering into the combustion chamber

Critical issues: Maximum mass flow rates are fixed as determined by the mixture ratios
Cooling strongly depends on the properties of the propellant
Pumping requirements increase

Minimum wall temperatures can be achieved as compared to the other techniques

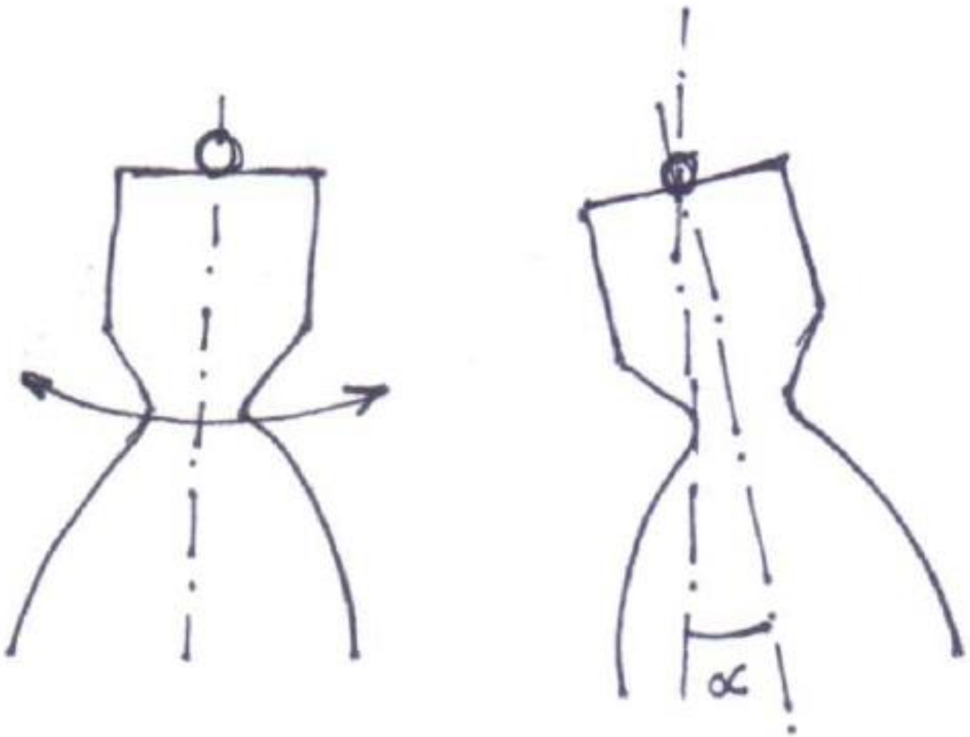


Thrust chamber cooling



Thrust vectoring

To change a rocket path as per the required trajectory
To apply the corrections for the deviations from the intended trajectory



Mechanical deflection of nozzle/thrust chamber

Gimballing (SSME, Russian engines)
Simple and proven technology
Very low power/thrust loss
Max. limits 12 degrees

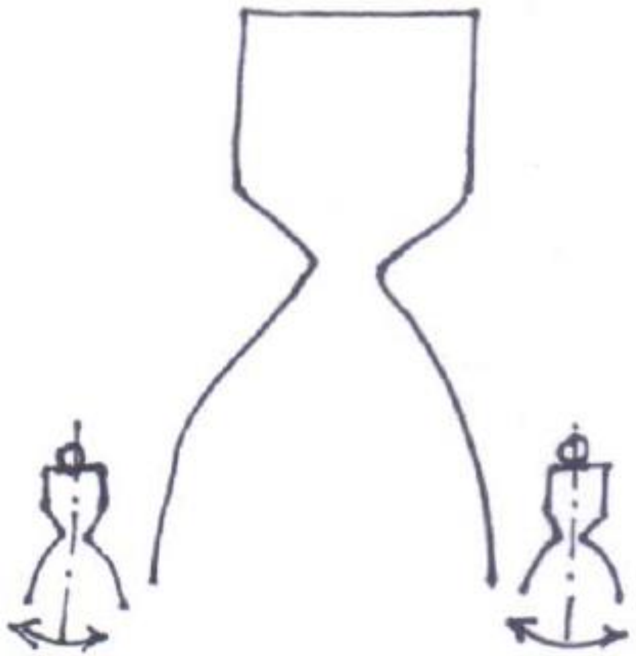
Requires flexible piping, powerful actuators

Thrust vectoring

Jet vanes

Aerodynamic wing shaped surfaces
Low actuation power
Limited to 9 degrees of deflection

Thrust loss ~ 3%
Erosion of vanes
V2 missiles, scud missiles



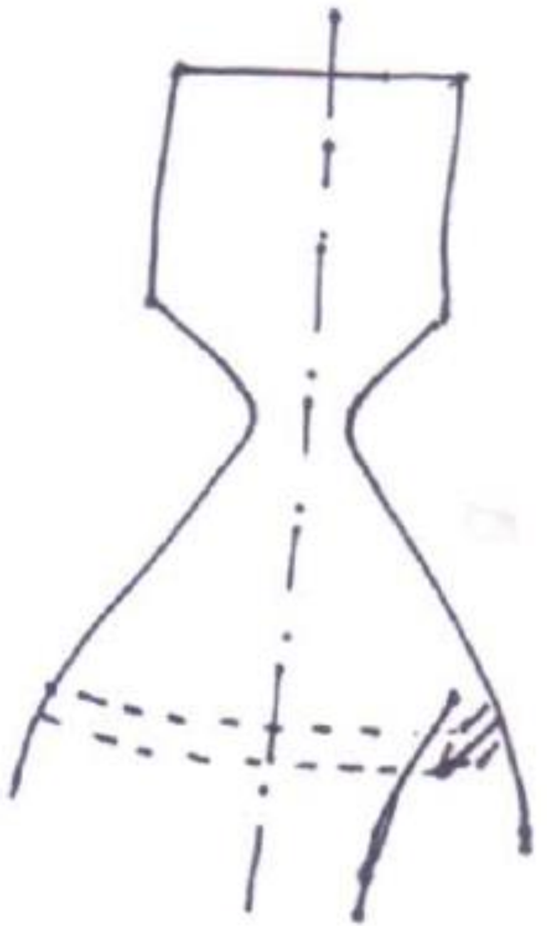
Auxiliary thrust chamber

Good for roll control
Low actuation power, low performance loss in main thrust chamber

Additional components, complex
Small moments
Not used for generic thrust vectoring

Thor, atlas

Thrust vectoring



Secondary Injection Thrust Vector control (SITVC)

Injection of fluid jet into the supersonic gas flow and creating asymmetric distribution of pressure (by shock wave formation and flow separation)

Efficient for low deflections ~ 6 degrees

Loss in power/thrust

Titan, Minuteman

PSLV – Strauntium Perchlorate (high density oxidizer) injection