Liquid propulsion systems

- Applications for small, medium and large systems
- Propellants – small, medium and large systems
- Feed systems – small, medium and large
- Propulsion system options
- Combustion systems – small, medium and large
- Combustion process comparison for Solids, Liquids, and Hybrids

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Applications for small, medium and large systems

- Small systems – satellite propulsion for orbit transfer and on-orbit stabilization for life
  Monopropellant, self-igniting, or decomposing storable, one or more starts or pulse mode, pr-fed
- Medium systems – Upper stages of launch vehicles or propulsion for missiles
  Single start, storable (self-igniting) or cryogenic bi-propellants with ignition, turbo-pump or pr-fed
- Large systems – Lower stages of launch vehicles,
  Usually single start, semi-cryogenic or full cryogenic systems with ignition, turbo-pump fed
- New developments in small systems – green propellants, in large systems, LOX – Methane propellants

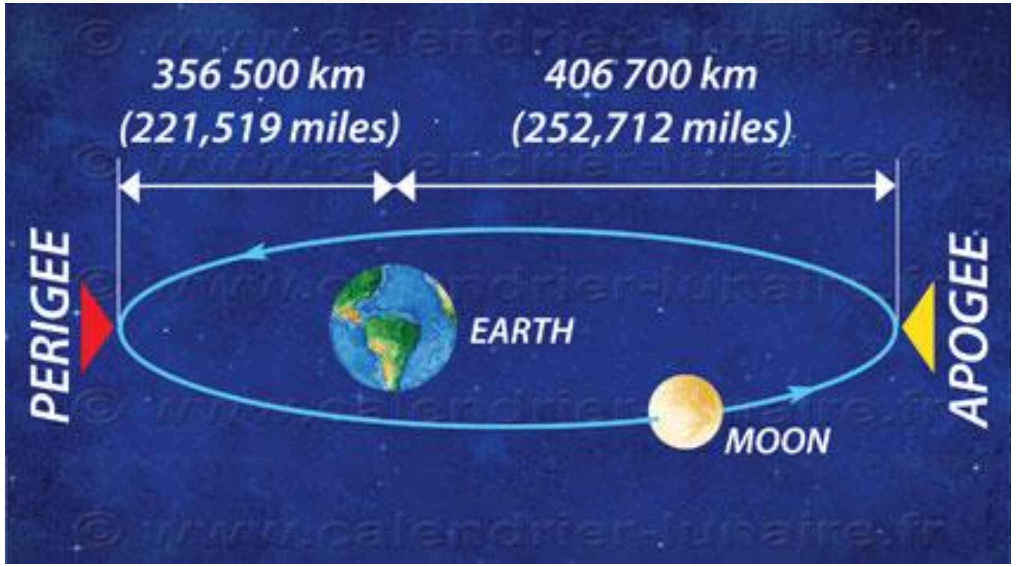
- **Purpose of these propulsion systems: To provide a velocity increment to the payload – stages above its level (stage 1 gives a velocity increment to stages 2+, etc) and satellite finally. Performance of the stages is measured in terms of ΔV**

- **Propulsion control system will also provide correction to the trajectory (orbit raising) or attitude.**
  These can be provided in a single long burn mode, or two burn mode or a number of pulses of fixed duration

Range of Propulsion Requirements

Maneuver	*ΔV, km/s*
Orbit transfer:	3.95 (no plane change required)
LEO to GEO	4.2 (including plane change of 28 deg)
LEO to GEO	1.5 (no plane change required)
GTO to GEO (1)	1.8 (incl. plane change of 28 deg.)
GTO to GEO (2)	3.2
LEO to Earth escape	3.1
LEO to translunar orbit	3.9
LEO to lunar orbit	1.25-1.4
GTO to lunar orbit	5.7
LEO to Mars orbit	8.7
LEO to solar escape	
Orbit control: Station-keeping (GEO)	50-55 m/s per year
Orbit control: Drag compensation	
•alt.: 400-500 km	< 100 m/s per year max. (<25 m/s average)
•alt.: 500-600 km	< 25 m/s per year max. (< 5 m/s average)
•alt.: >600 km	< 7.5 m/s per year max.
Attitude control: 3-axis control	2-6 m/s per year
Auxiliary tasks:	
•Spin-up or despin	5-10 m/s per maneuver
•Stage or booster separation	5-10 m/s per maneuver
•Momentum wheel unloading	2-6 m/s per year

LEO = Low Earth Orbit; GTO = Geo transfer orbit; GEO, Geosynchronous Earth orbit



For a circular orbit, $mV^2/r = mg$ $\quad g = g_0/(1 + h/R_e)^2$ $\quad R_e \quad 6371$ km

$$V = \sqrt{\frac{g_0 R_e}{\left[1 + \frac{h}{R_e}\right]}}$$

$$\boxed{m\frac{dV_z}{dt} = F\sin\theta - mg - D \Longrightarrow \frac{dV_z}{dt} = \frac{F\sin\theta}{m} - g - \frac{D}{m}}$$

$$\frac{dV_z}{dt} = \frac{F}{m} - g$$ Vertical flight, no drag

$$\frac{dV_z}{dt} = I_{sp}\frac{\dot{m}_p}{m} - g = I_{sp}\frac{\dot{m}}{m} - g = \frac{dm}{mdt} - g \qquad dV_z = I_{sp}\frac{dm}{m} - gdt$$

$$\Delta V_z = I_{sp} ln\left[\frac{m_{ini}}{m_f}\right] - gt_b \qquad V = 0 + \Delta V_z$$

ΔV_z is the velocity increment provided by the propulsion system.

Propulsion Function	Typical Requirement
Orbit transfer to GEO (orbit insertion)	
• Perigee burn	2,400 m/s
• Apogee burn	1,500 (low inclination) to 1,800 m/s (high inclination)
Initial spinup	1 to 60 rpm
LEO to higher orbit raising ΔV	60 to 1,500 m/s
• Drag-makeup ΔV	60 to 500 m/s
• Controlled-reentry ΔV	120 to 150 m/s
Acceleration to escape velocity from LEO parking orbit	3,600 to 4,000 m/s into planetary trajectory
On-orbit operations (orbit maintenance)	
• Despin	60 to 0 rpm
• Spin control	±1 to ± 5 rpm
• Orbit correction ΔV	15 to 75 m/s per year
• East-West stationkeeping ΔV	3 to 6 m/s per year
• North-South stationkeeping ΔV	45 to 55 m/s per year
• Survivability or evasive maneuvers (highly variable) ΔV	150 to 4,600 m/s
Attitude control	3–10% of total propellant mass
• Acquisition of Sun, Earth, Star	Low total impulse, typically <5,000 N·s, 1 K to 10 K pulses, 0.01 to 5.0 sec pulse width
• On-orbit normal mode control with 3-axis stabilization, limit cycle	100 K to 200 K pulses, minimum impulse bit of 0.01 N·s, 0.01 to 0.25 sec pulse width
• Precession control (spinners only)	Low total impulse, typically <7,000 N·s, 1 K to 10 K pulses, 0.02 to 0.20 sec pulse width
• Momentum management (wheel unloading)	5 to 10 pulse trains every few days, 0.02 to 0.10 sec pulse width
• 3-axis control during ΔV	On/off pulsing, 10 K to 100 K pulses, 0.05 to 0.20 sec pulse width

GEO = Geo-equatorial orbit, LEO = Low earth orbit

Propellants for small, medium and large systems

Small: Monopropellants – Hydrazine (N_2H_4) with Shell 405 (Iridium) catalyst,
Hydrogen peroxide (H_2O_2) with Silver catalyst
Hydroxyl Ammonium Nitrate (HAN, NH_3-OH-NO_3) with Ammonium nitrate (AN, NH_4NO_3) + methanol (CH_3OH) + water or other combinations – Green propellants for better performance and safer operations….

Medium: Bipropellants - Hypergolic (No ignition system needed) - MMH - N_2O_4 [Mono-methyl-hydrazine CH_3-NH-NH_2 – Nitrogen tetroxide] or MON (Mixed oxides of N_2 - N_2O_4 + some NO)

Large: Bipropellants – Storable, hypergolic – UDMH - N_2O_4 (Unsymmetrical dimethyl hydrazine. $H_2NN(CH_3)_2$)
Non-hypergolic, semicryo – Kerosene – Liquid oxygen (LOX, LO_2)
Non-hypergolic, cryo – Liquid hydrogen (LH_2) - LOX

Notes: Hypergolicity – a property by which when fuel and oxidizer are brought together, they ignite (within a few millisecs)

Hydrazine gives the best performance as a rocket fuel, but it has a high freezing point (2˚C) and is too unstable for use as a coolant. MMH is more stable with a freezing point, - 52 ˚C, good for spacecraft propulsion applications. UDMH has the lowest freezing point and has enough thermal stability to be used in large regeneratively cooled engines. The freezing point of N_2O_4 is ~ -9 °C, MON-3 is -15 °C, MON-25 is -55 °C. Boiling point of LH_2 – 20 K, LO2 – 90 K

The basic performance of propellant combinations

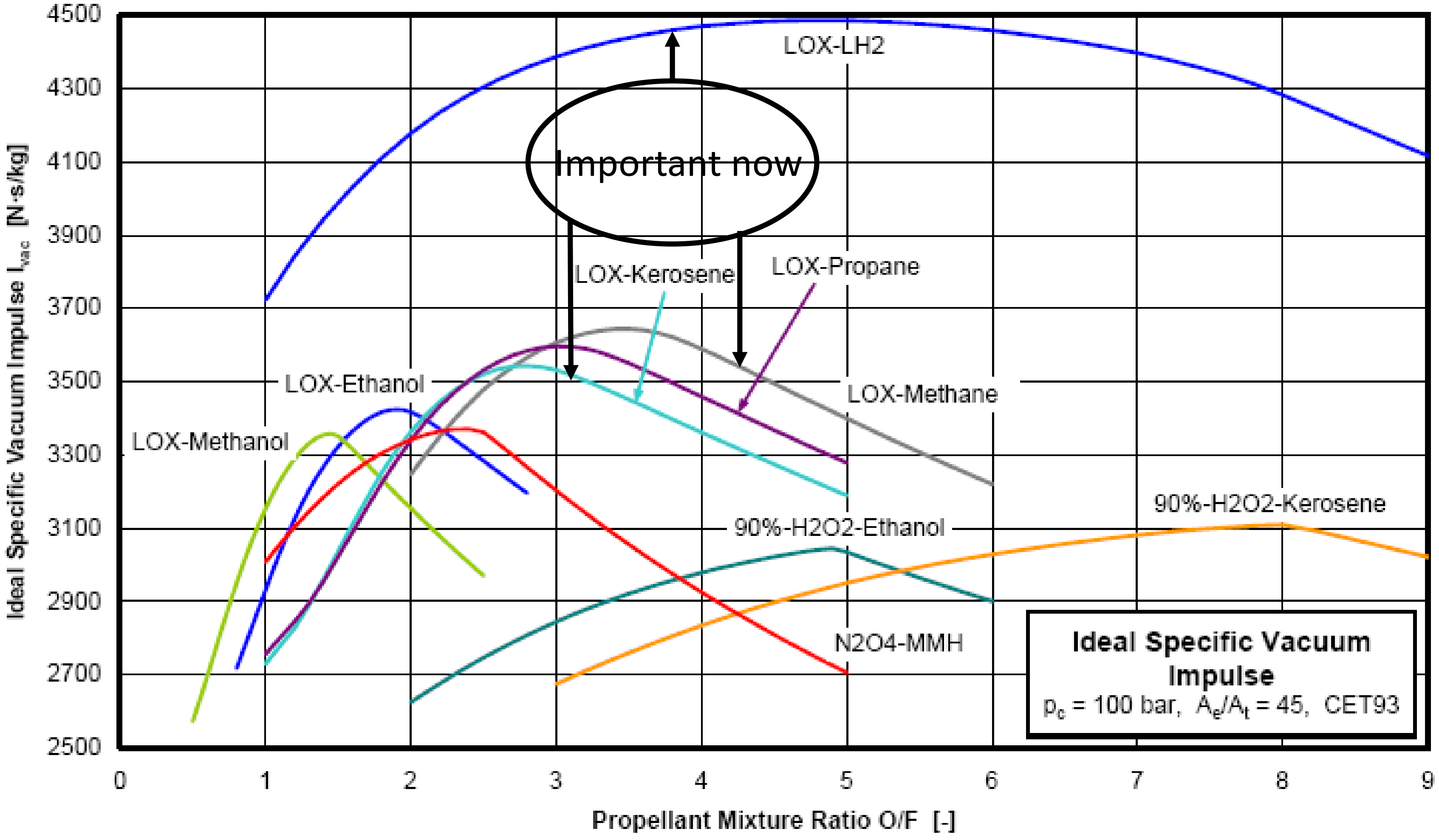
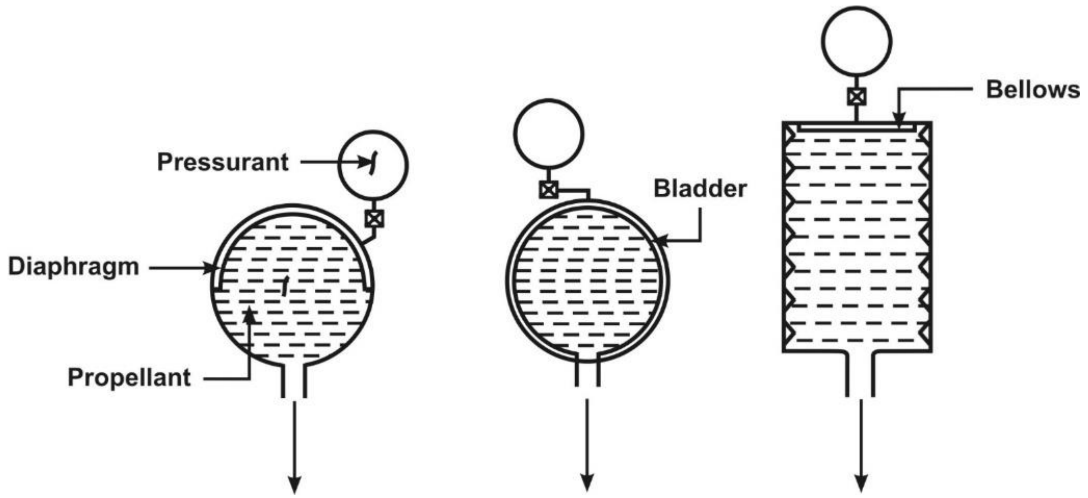
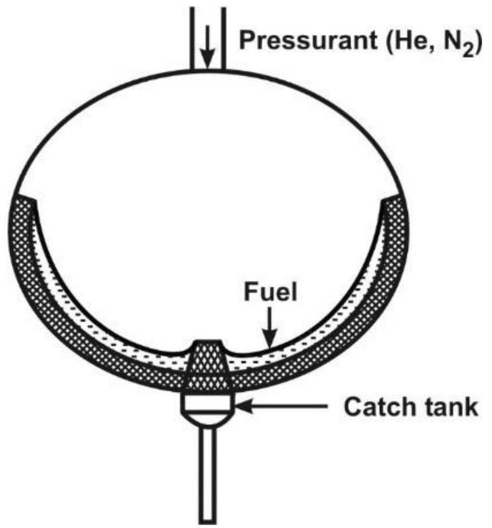
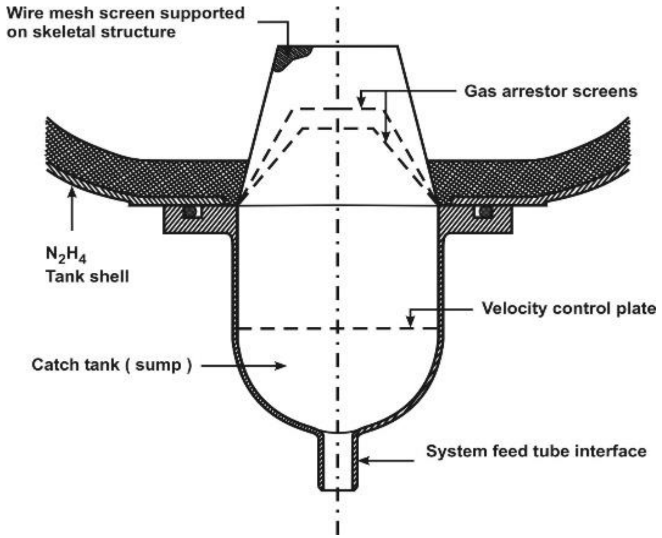


Figure 3: Ideal specific impulse of various propellant combinations

Feed systems – for small engines





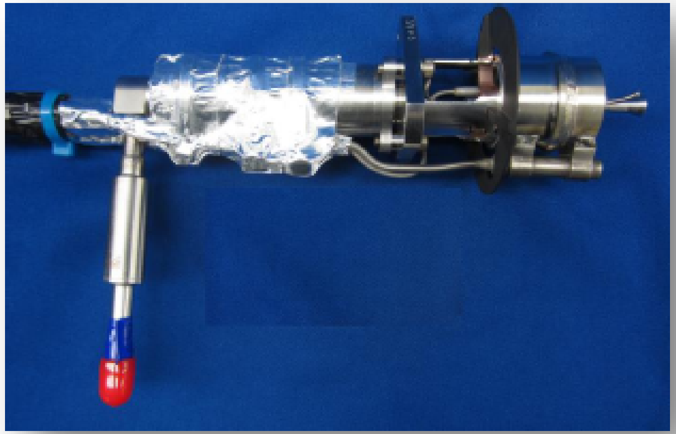


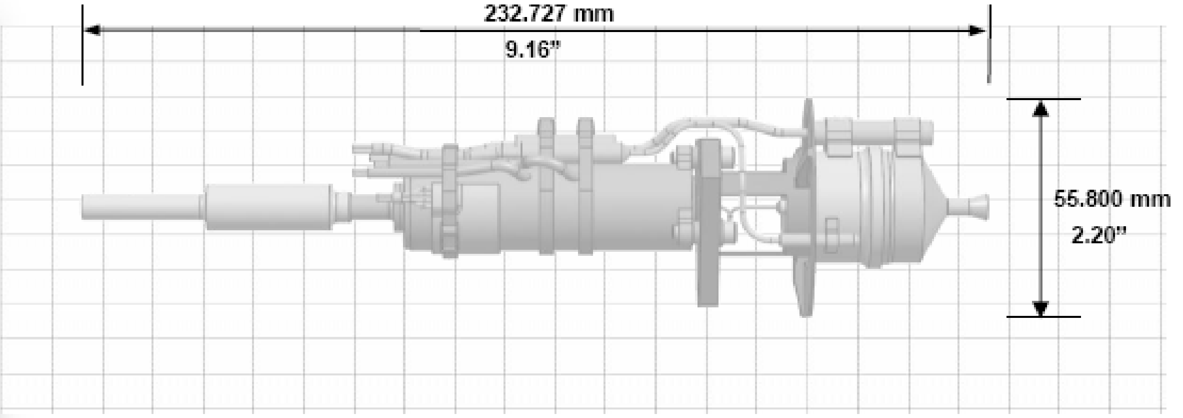
When satellites orbit under near zero-g conditions, acquiring propellant cannot get the aid of gravity.

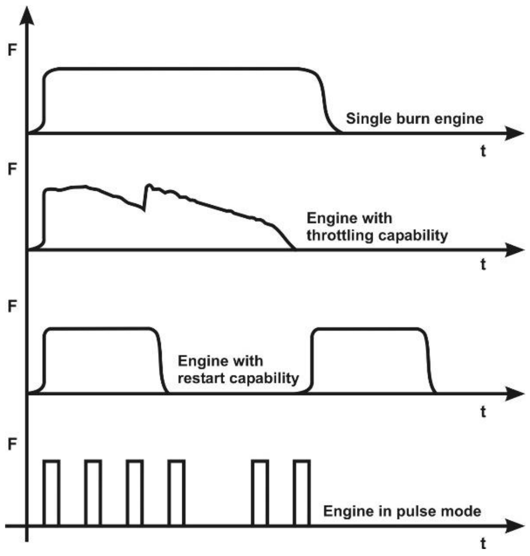
Propellant acquisition requires other approaches. Arranging compatible bellows, bladder or a diaphragm between the liquid mono-propellant (most usual), and pressuring the surface will help maintain pressure on the liquid for it to pass through the plumbing system downstream to get delivered Into the combustion chamber.

An alternate system is surface tension device in which fine mesh is designed to hold the liquid and under pressure release it to a "catch tank" for deliver it downstream. The performance is measured by "expulsion efficiency". It is the highest - 99.5 % + for surface tension device. This matters a whole lot because on this depends the life of the satellite – 10 to 12 years

A typical small hydrazine propellant based thruster







Design Characteristics

- Propellant.. Hydrazine
- Catalyst.. S-405
- Thrust/Steady State...........0.07 – 0.09 N (0.016 - 0.020 lbf)
- Feed Pressure..................14.8 – 18.6 bar (215 - 270 psia)
- Flow Rate......... 154.2 – 181.4 g/hr (0.34 – 0.40 lbm/hr)
- Valve.. Dual Seat
- Valve Power................. 8.25 Watts Max @ 28 Vdc & 21°C
- Valve Heater Power....... 1.9 Watts Max @ 28 Vdc & 21°C
- Cat. Bed Heater Pwr...... 1.8 Watts Max @ 28 Vdc & 21°C
- Mass.. 0.60 kg (1.32 lbm)
  - Engine.................................... 0.33 kg (0.74 lbm)
  - Valve...................................... 0.20 kg (0.44 lbm)
  - Heaters................................. 0.065 kg (0.14 lbm)

Performance

- Specific Impulse, steady state....... 180 - 184 sec (lbf-sec/lbm)
- Specific Impulse, cumulative.........1 50 - 177 sec (lbf-sec/lbm)
- Total Impulse...................... 199,693 N-sec (44,893 lbf-sec)
- Total Starts/Pulses..5,960
- Min Impulse Bit............ 4.0 N-sec @ 14.8 bar & 60 sec ON
  .... (0.9 lbf-sec @ 215 psia & 60 sec ON)
- Steady State Firing.................. 0 - 900 sec Single Firing
  .... 720 hrs Cumulative

Status

- Flight Proven
- Currently in Production

Reference

- JANNAF, 2011, paper 2225

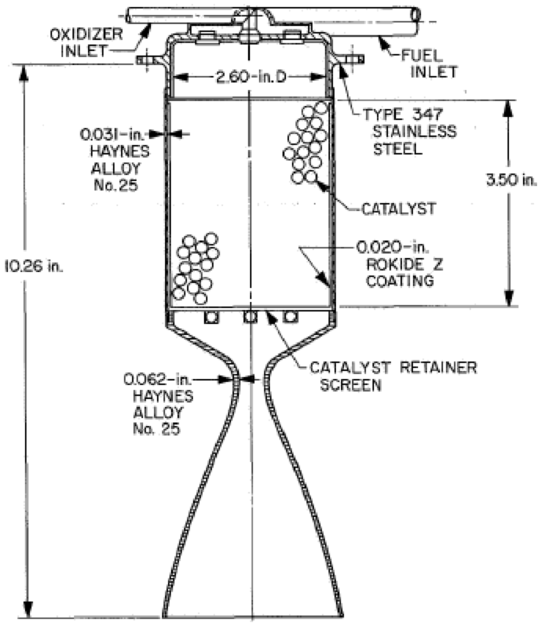


Fig. 12. ***Ranger-Mariner*** **monopropellant hydrazine 50-lbf thrust rocket engine (nonspontaneous catalyst)**

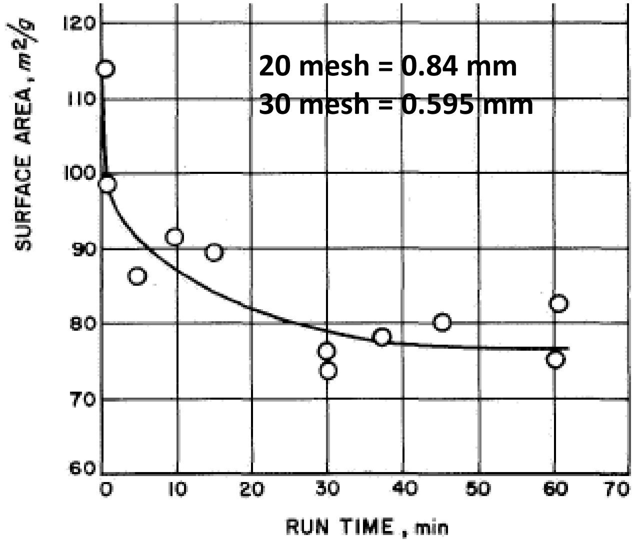


Fig. 1. Micropore surface area vs run time for Shell 405 catalyst (20-30 mesh) (reprinted from Ref. 21)

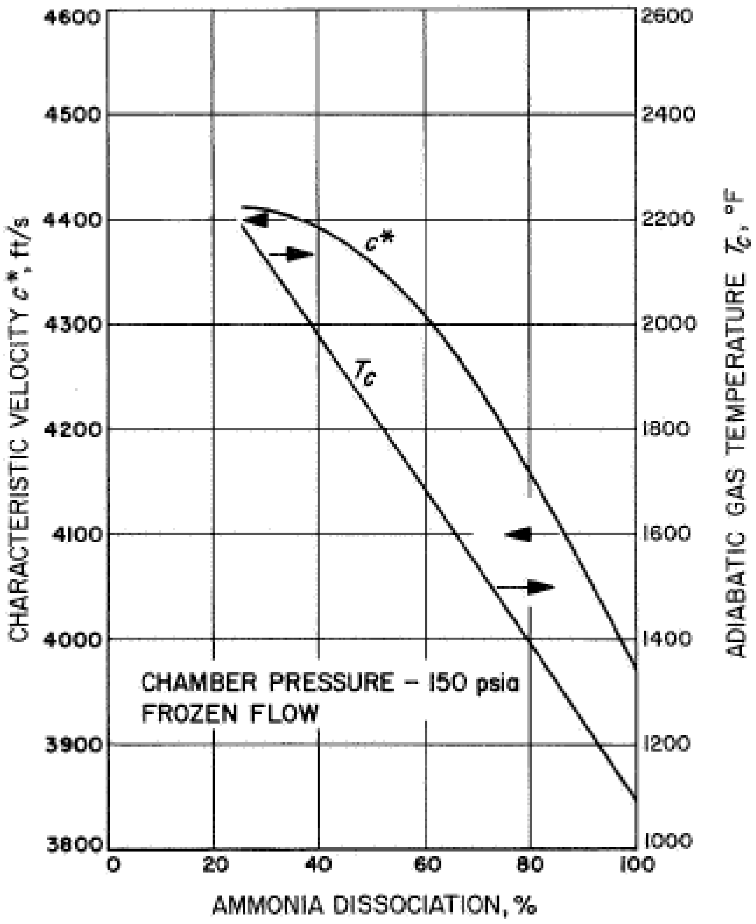


Fig. 2. Theoretical characteristic velocity and adiabatic gas temperature of monopropellant hydrazine

$$N_2H_4 \longrightarrow \frac{4}{3}(1-X)\,NH_3 + \frac{1}{3}(1+2X)\,N_2 + 2X\,H_2$$

X = Fraction of Ammonia dissociated

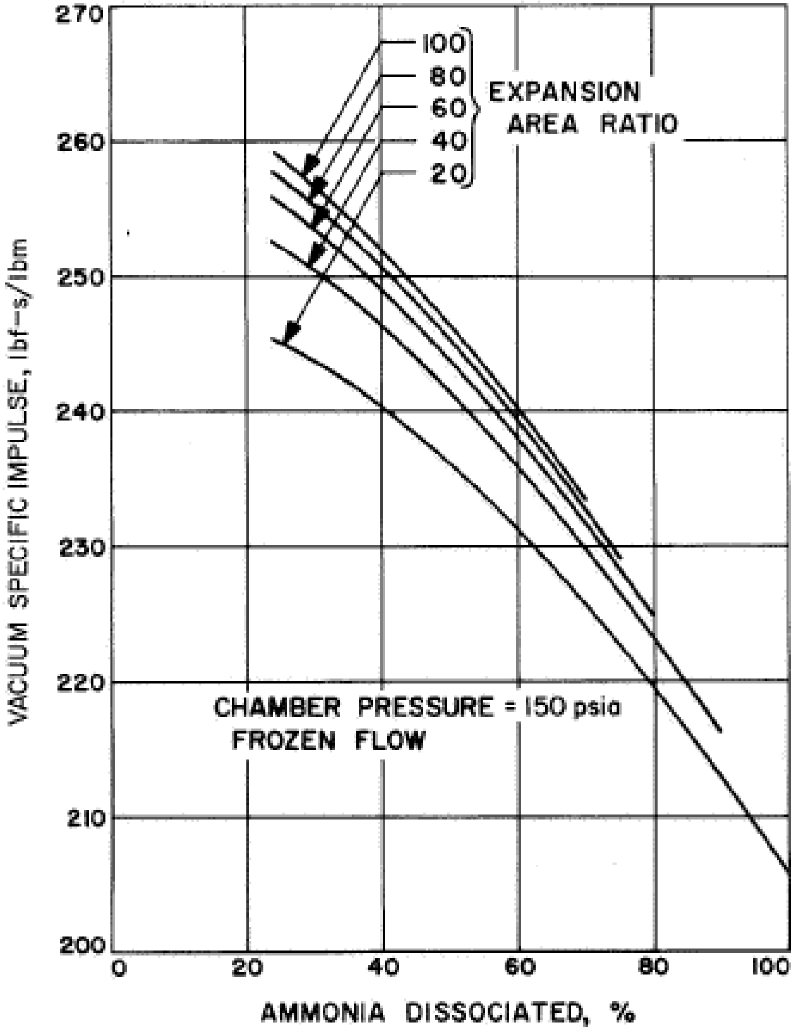
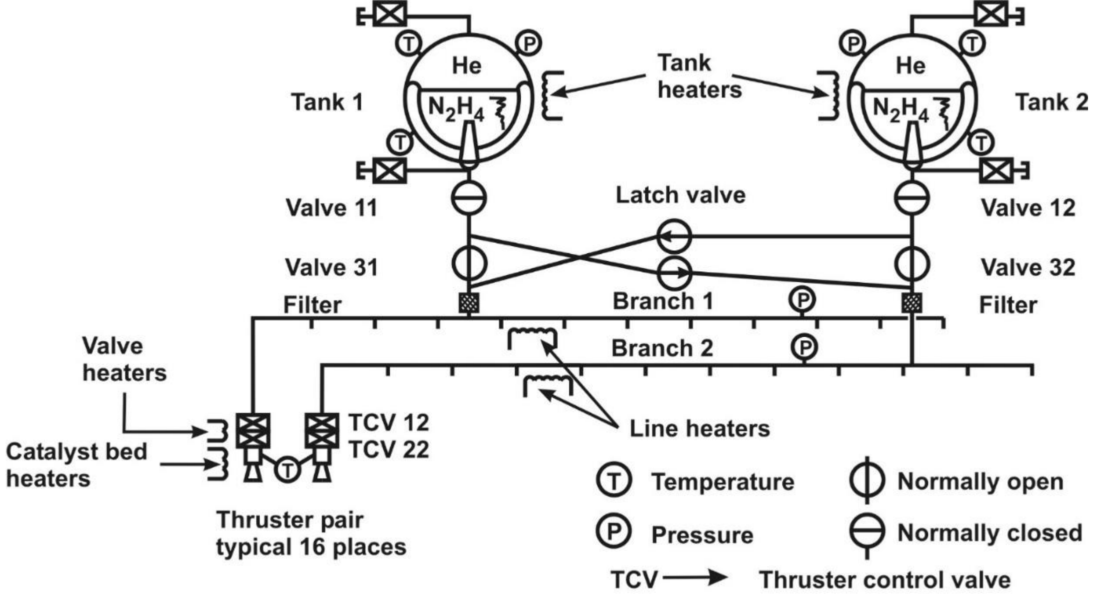
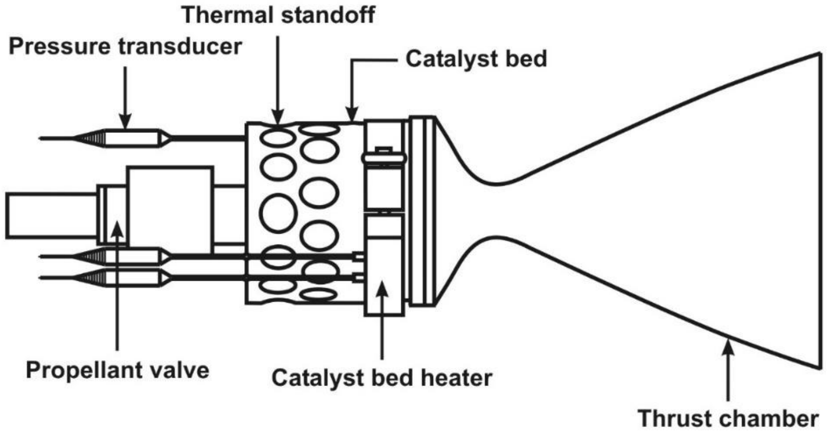


Fig. 3. Theoretical vacuum specific impulse of monopropellant hydrazine



(a)



(b)

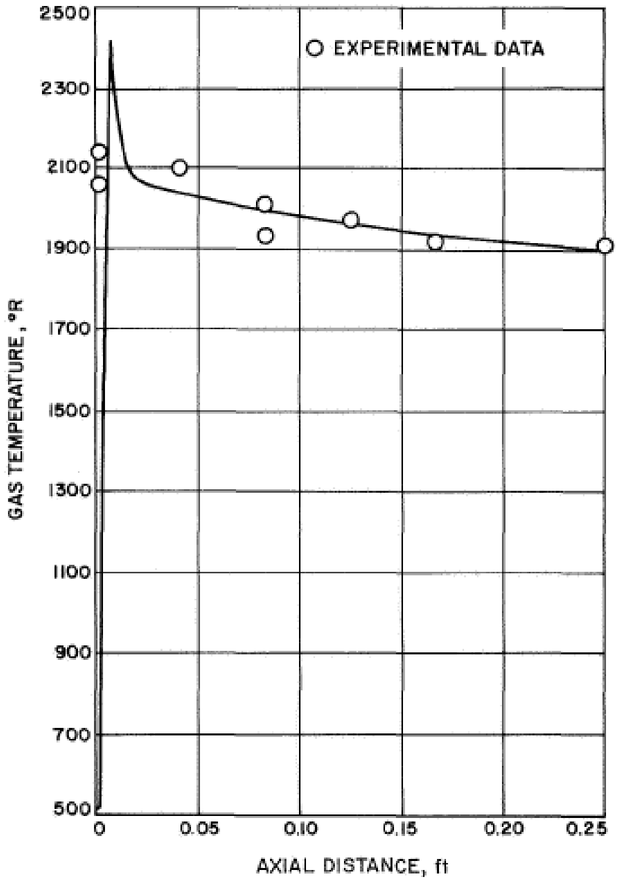


Fig. 4. Steady-state axial profiles in a catalyst bed (reprinted from Ref. 26)

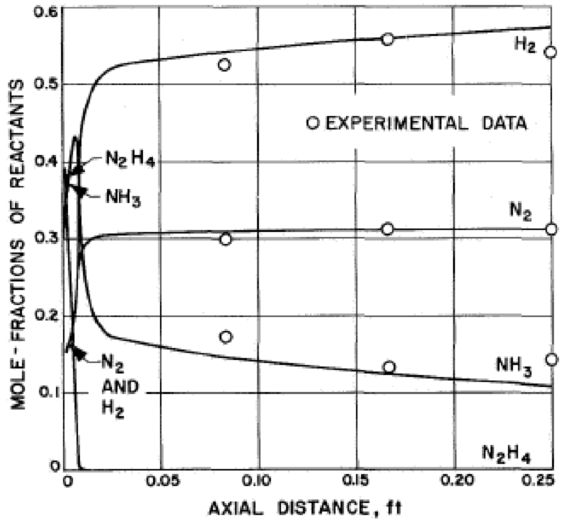
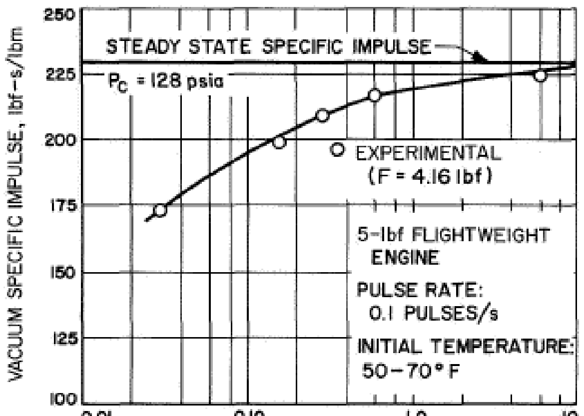


Fig. 5. Steady-state axial profiles of reactant concentrations in a catalyst bed (reprinted from Ref. 26)



New propulsion system plans – green propellant

The following figure shows an overview over the application areas for space propulsion, current toxic (in red) and non-toxic (in green) technologies that are used and non-toxic technologies that are currently investigated or under introduction into the market.

The figure also shows that for key applications where today still toxic propellants are in use green technology is already available. Further maturation is necessary to fulfil all the various demands of the different applications and use cases.

ECAPS LMP-103S Technology: LMP-103S is a liquid propellant where the solid oxidizer (ADN) is solved in water and Ethanol is used as fuel. The ADN technology is closely linked to the Swedish company ECAPS that developed the propellant LMP-10S and subsequent thrusters which are for the first time used on a commercial mission [4].

Advantage of LMP-103S	Disadvantage of LMP-103S
▪ Non Toxic, non-carcinogenic ▪ Most COTS components can be used ▪ Lower cost of handling ▪ High Density (1250 kg/m³) ▪ Higher performance (235 s) ▪ Allowed to be transported by airplane	▪ Higher cost of propellant and thruster ▪ Preheating of thruster to > 250°C required and has to be controlled ▪ High combustion temperature requires high temperature combustion chamber material ▪ ADN salt itself is a friction sensitive explosive with subsequent handling effort

Table 1 Advantages / Disadvantages of LMP-103S

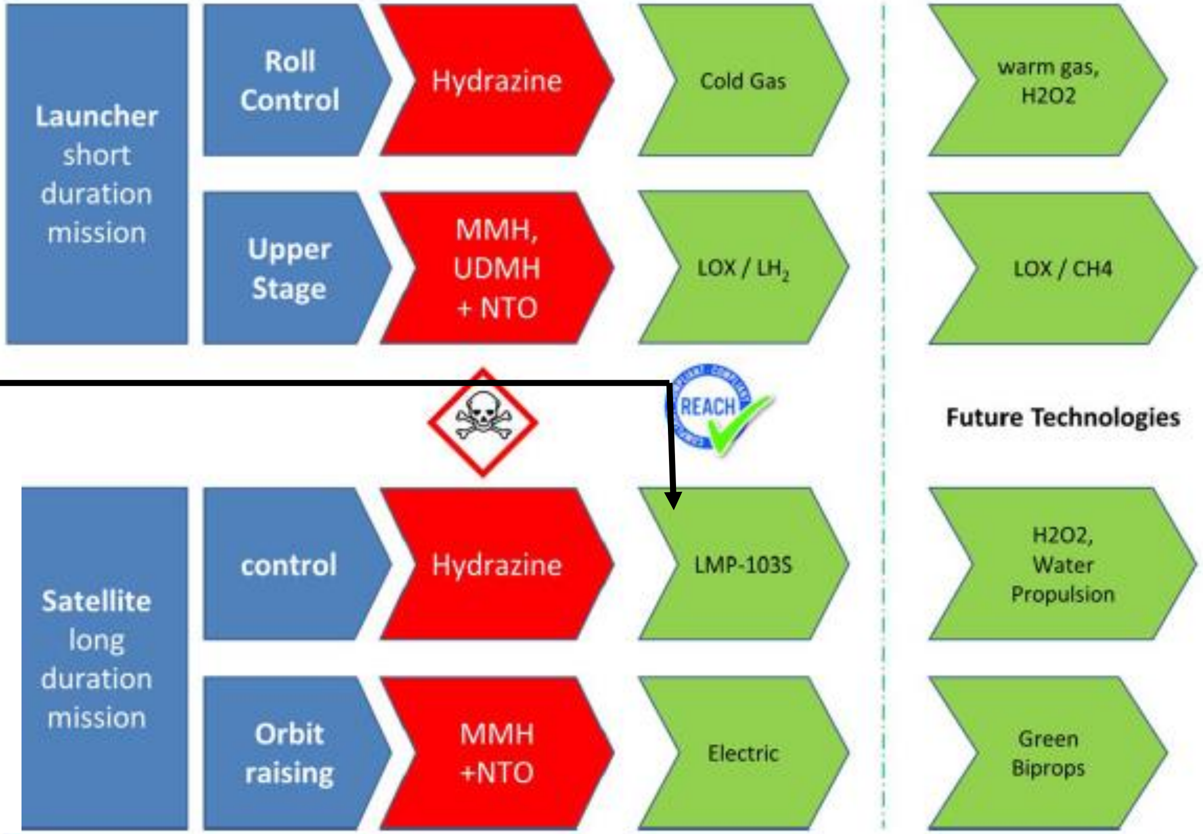


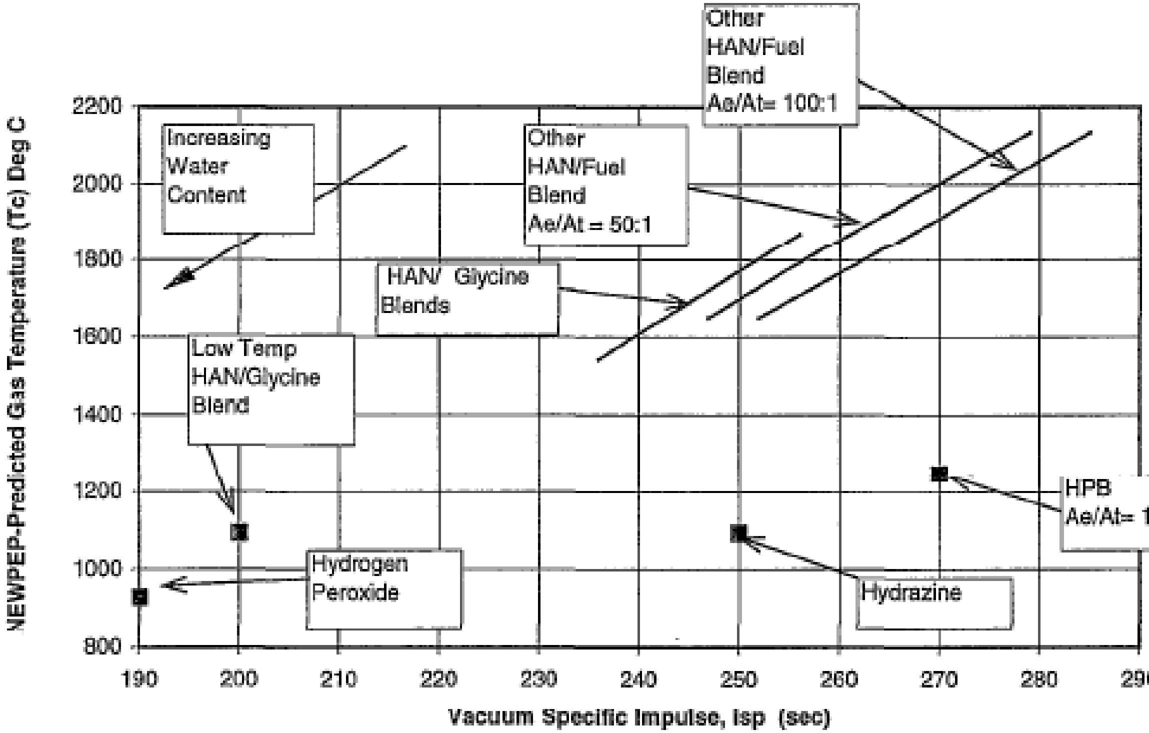
Figure 3: Application areas for space propulsion and technologies that are used

Performance of new propellants

2.2 ADN Technology

The term "ADN Technology" is used for a liquid monopropellant where solid oxidizer (ADN - Ammonium dinitramide salt) is solved in water and then fuel and stabilizer are added. In the combustion chamber the oxidizer and fuel are burned with subsequent high combustion temperatures. It is considered as non-toxic and air transportable. Typically these propellants have a higher ISP and a higher density compared to Hydrazine. A comparable technology is the HAN technology where HAN (Hydroxylammonium Nitrate) is used as a solid oxidizer, e.g. in the propellant AF-315ME [3].

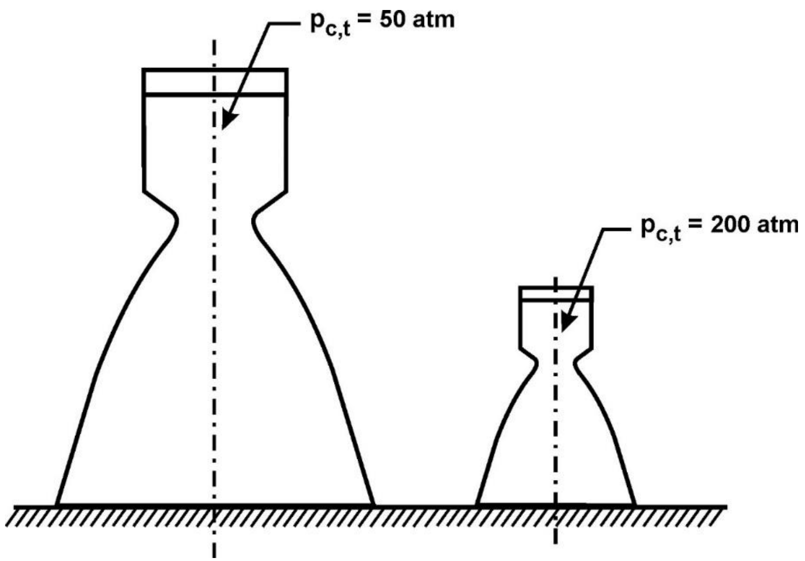
ISRO research:
…..The in-house formulation consists of HAN, ammonium nitrate, methanol and water. While methanol was added to reduce combustion instability, the choice of AN was dictated by its capacity to control the burn rate and lower the freezing point of the propellant.

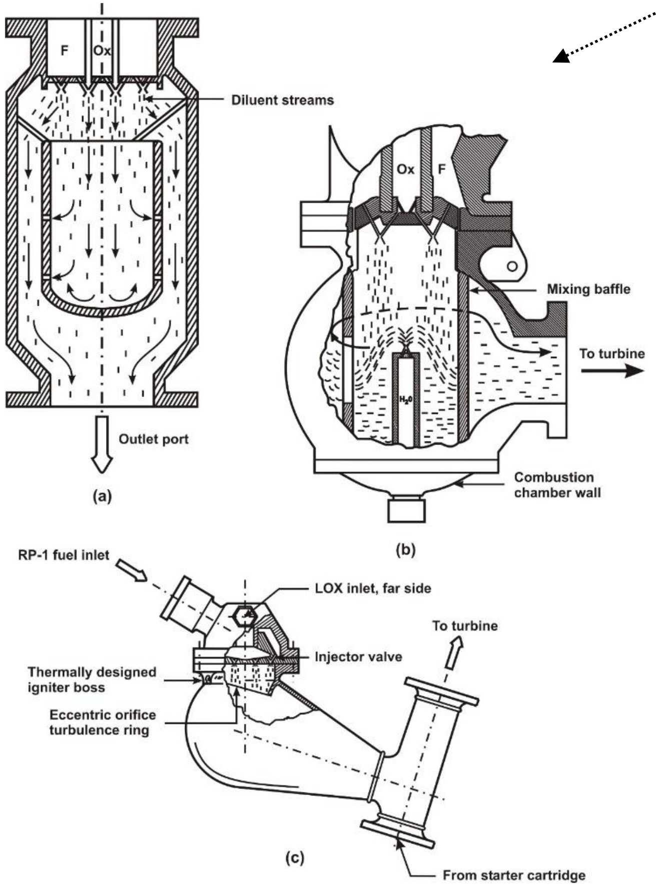


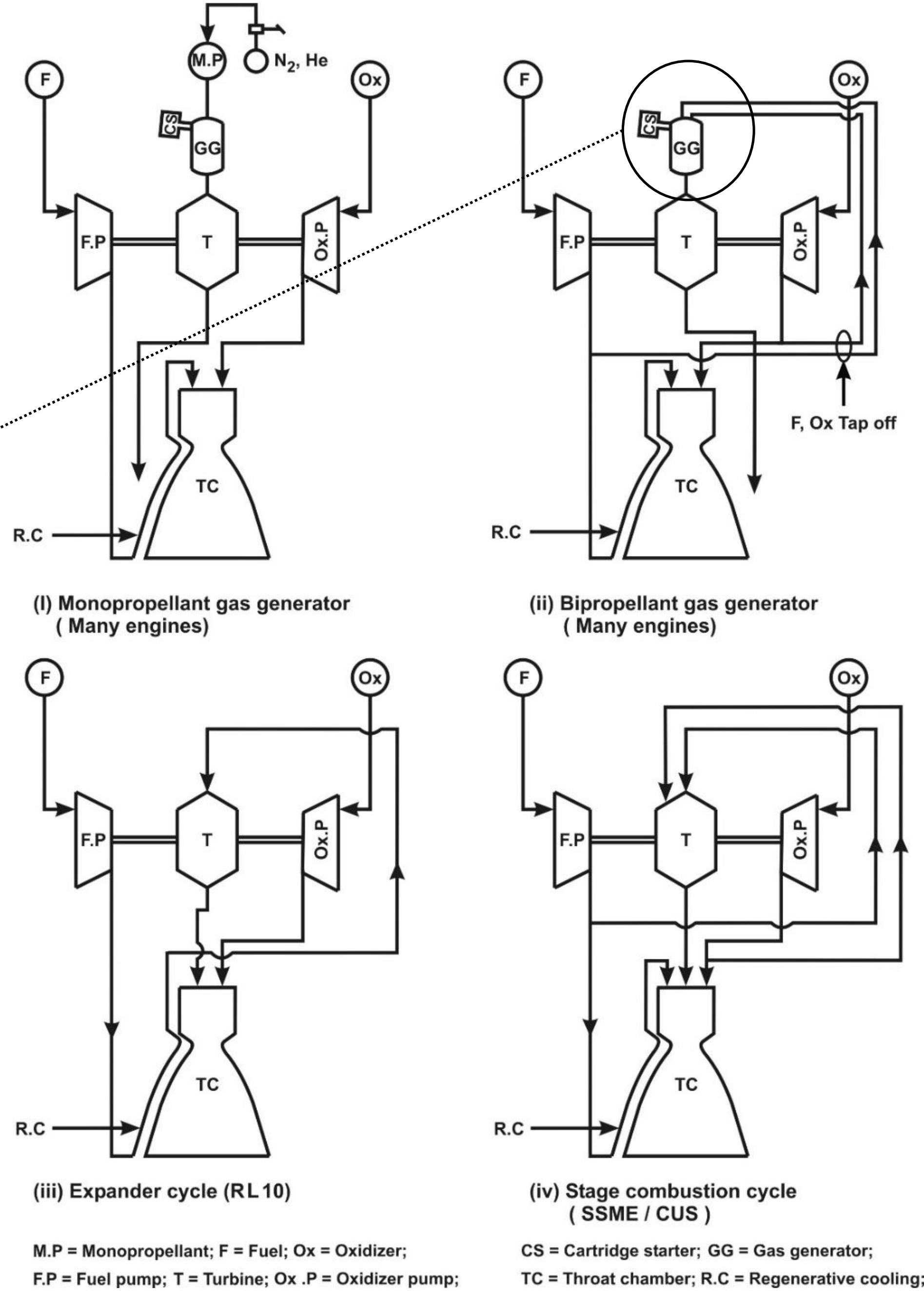
Japanese technology demonstration satellite Innovative Satellite Technology Demonstration-1, launched in **January 2019**, contains a demonstration thruster using HAN and operated successfully in orbit

More developments will take place in coming times in ISRO

Various cycles, feed systems and thrust chambers for large propulsion systems

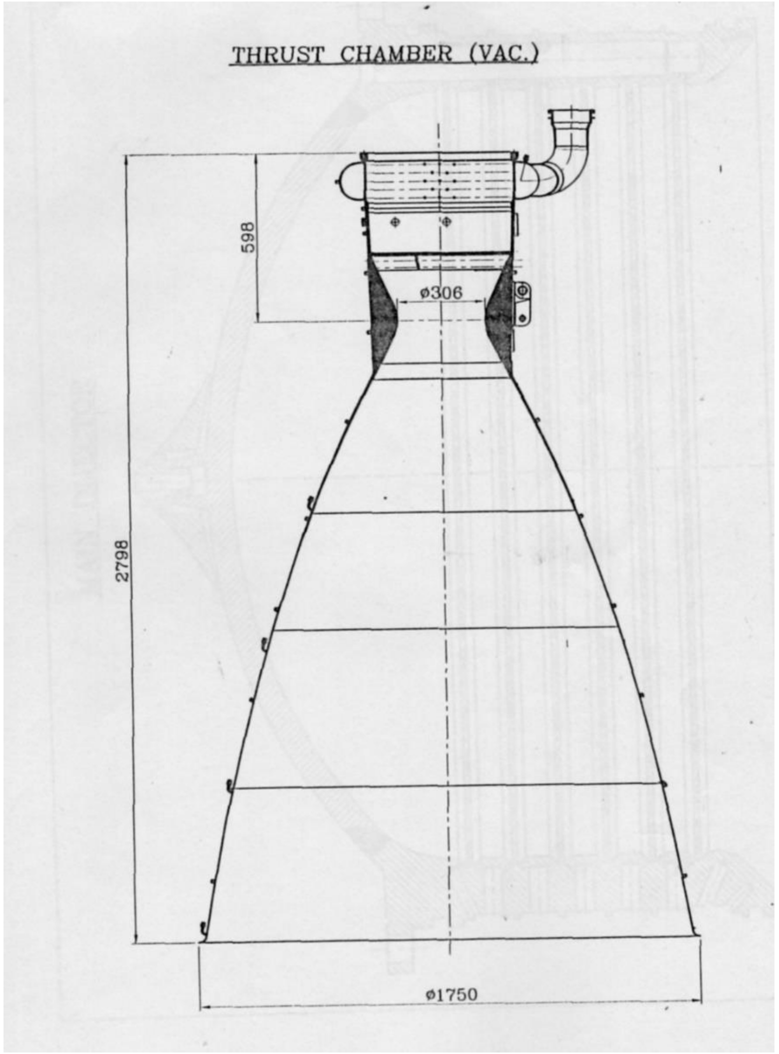


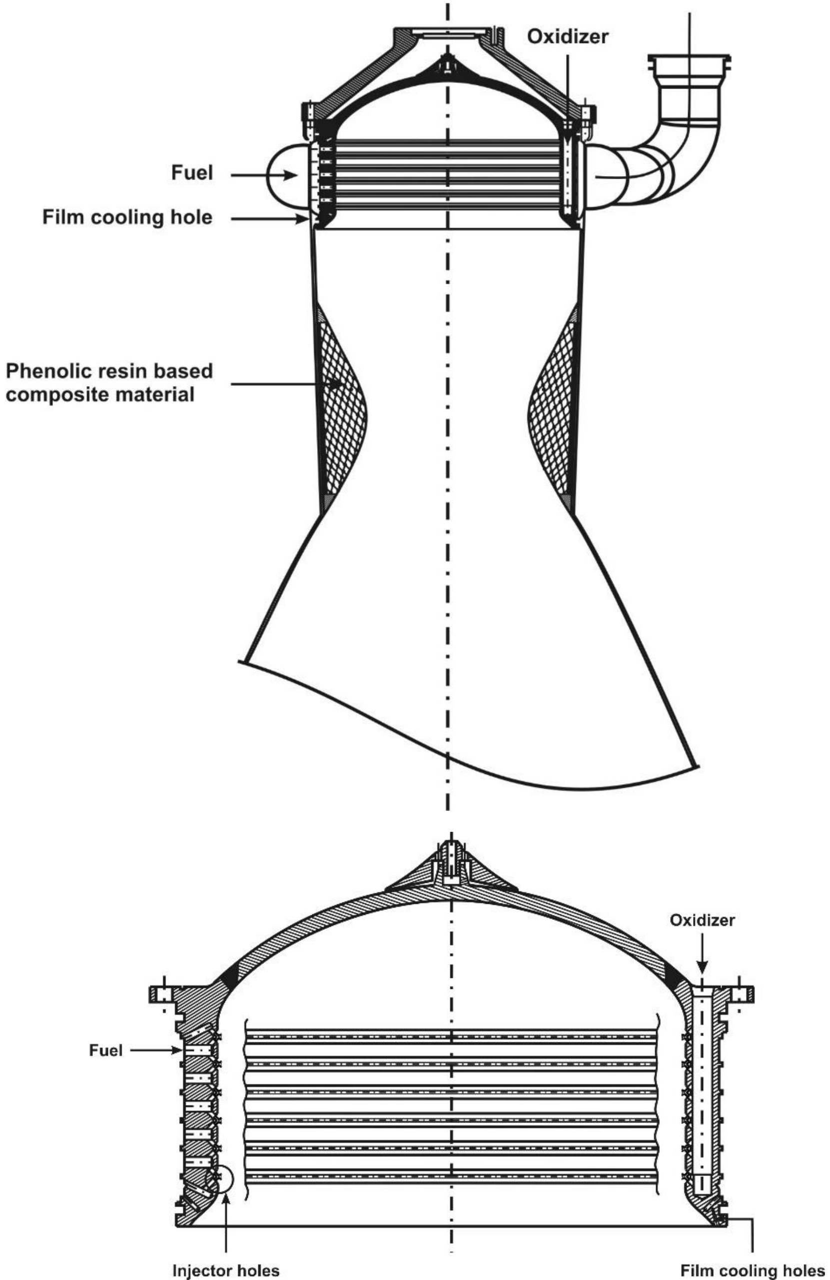




Choice of higher p_c reduces the envelope smaller, engine lighter

Large propulsion system – Vikas engine





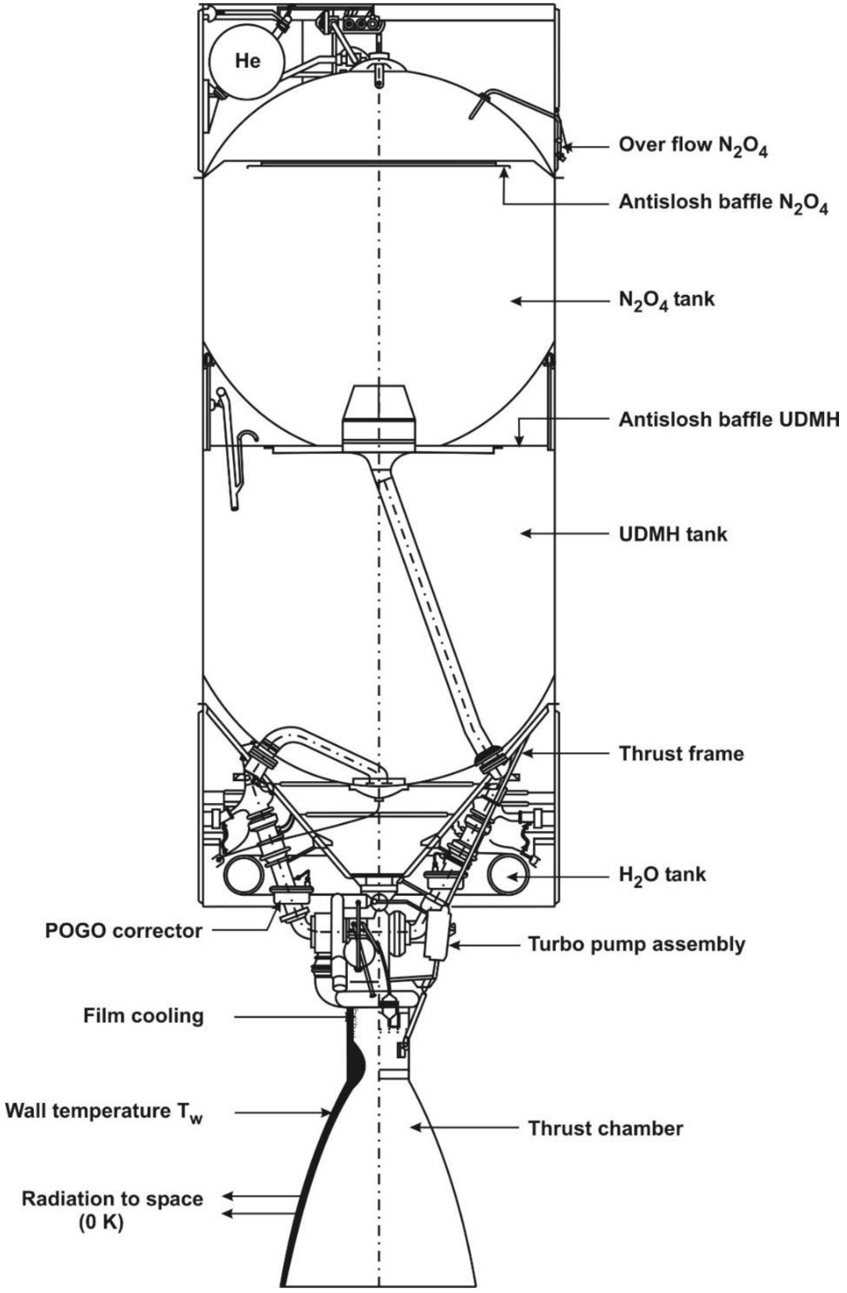


Table 3: Characteristic data of liquid booster engines [2, 8]

Rocket Engine	Engine Cycle	Propellant combination	Thrust [MN]	spec. impulse (sl) [s]	Chamber pressure [MPa]	Burn time [s]
RD-170	SC	LO_2/Kerosene	7,65	310	25,1	150
RD-180	SC	LO_2/Kerosene	3,82	311	25,5	150
RD-107	SC	LO_2/Kerosene	0,81	257	5,9	119
F-1	GG	LO_2/RP1	6,91	264	6,6	161
MA-5A	GG	LO_2/RP1	1,84	263	4,4	263
RS-27	GG	LO_2/RP1	0,91	263	4,8	265
RD-253	SC	N_2O_4/UDMH	1,47	285	15,2	130
YF-20	GG	N_2O_4/UDMH	0,76	259	7,4	170
Viking 6	GG	N_2O_4/UH25	0,68	249	5,9	142
RS-68	GG	LO_2/LH_2	2,89	360	9,7	249

Table 5: Characteristic data of upper stage engines [2, 8]

Rocket Engine	Engine Cycle	Propellant combination	Thrust (vac.) [kN]	Spec. impulse (vac.) [s]	Chamber pressure [MPa]	Burn Time [s]
11D58M	SC	LO_2/Kerosene	79.5	353	7,6	680
RD-0210	SC	N_2O_4/UDMH	582	327	14.8	230
AESTUS	PF	N_2O_4/MMH	30	325	1,0	1100
J-2	GG	LO_2/ LH_2	890	426	4.4	
YF-75	GG	LO_2/ LH_2	79	440	3.7	470
LE-5B	EC	LO_2/ LH_2	137	447	3.6	534
HM7-B	GG	LO_2/ LH_2	70	447	3.5	731
VINCI	EC	LO_2/ LH_2	180	465	6.1	
RL-10B	EC	LO_2/ LH_2	110	462	4.3	700

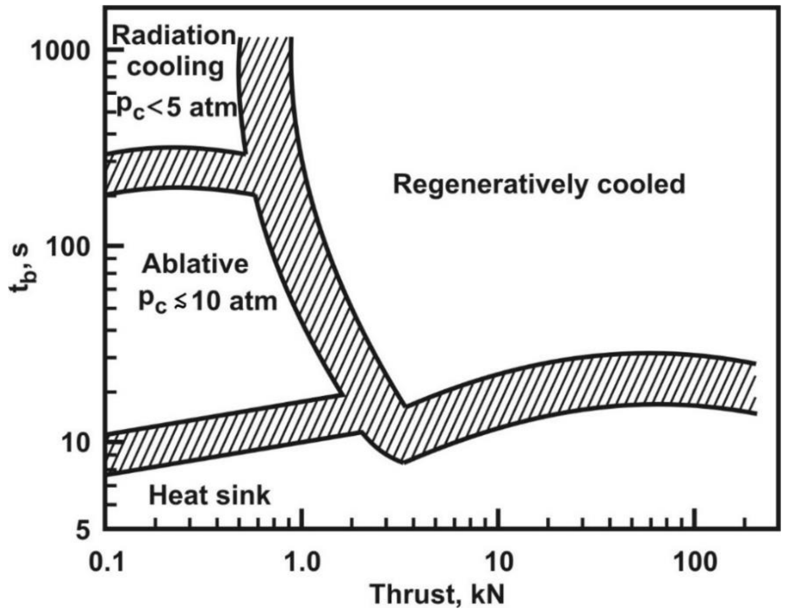
Table 6: Characteristic data of sample gas generators and pre-burners [10]

	Vulcain 2	SSME	LE-7	RD-0120	RF-1	RD-180
Propellant Combination	LOX/H_2	LOX/H_2	LOX/H_2	LOX/H_2	LOX/RP1	LOX/RP1
T [K]	875	940 / 870	810	846	816	820
p_{GG} [MPa]	10.1	35 / 36	21.0	42.4		55.6
r_{of} [-]	0.9	0.89 / 0.8	0.55	0.81	~55	54
m [kg/s]	9.7	80 / 30	53	78.6	?	887
P [MW]	5 / 14	56 / 21	4.5 / 19	62	44	93.5
p_c [MPa]	11.6	20.6	12.7	21.8	6.6	25.7

As can be seen a variety of choices have been made for engines built in Russia, USA and France. Russian engines using staged combustion cycle have always used oxygen rich mode for combustion and all others have used fuel rich mode.

Oxidizer rich operation allows for sufficient reduction of turbine inlet temperature since larger mass flow rates are available. Russians alone seem to have mastered this even though Americans have been trying. The choice of materials with high temperature oxidizing environment is the key.

Cooling aspects of thrust chambers

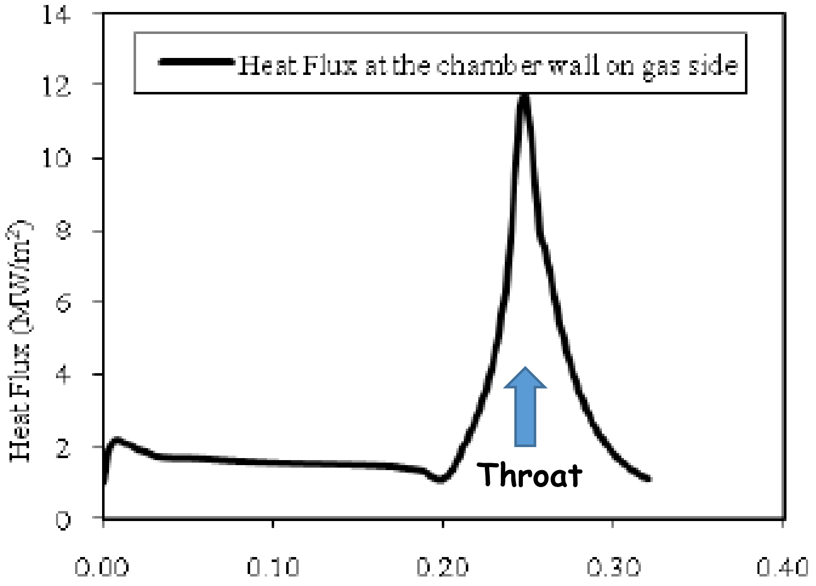


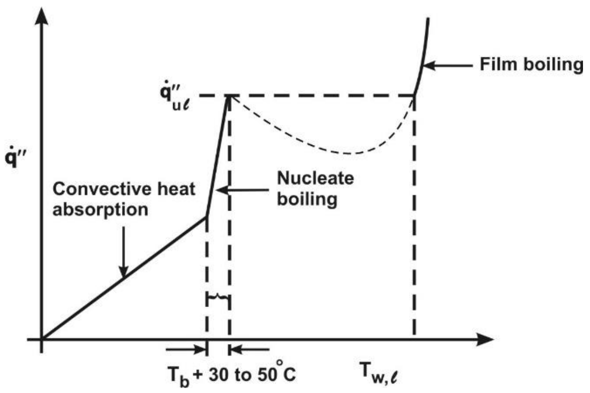
$$Nu = K\,\mathrm{Re}^a\,\mathrm{Pr}^b\left(\frac{\rho}{\rho_w}\right)^c\left(\frac{\mu}{\mu_w}\right)^d\left(\frac{k}{k_w}\right)^e\left(\frac{\bar{c}_p}{c_p}\right)^f\left(\frac{p}{p_{cr}}\right)^g\left(1+\frac{2D}{L}\right)^m$$

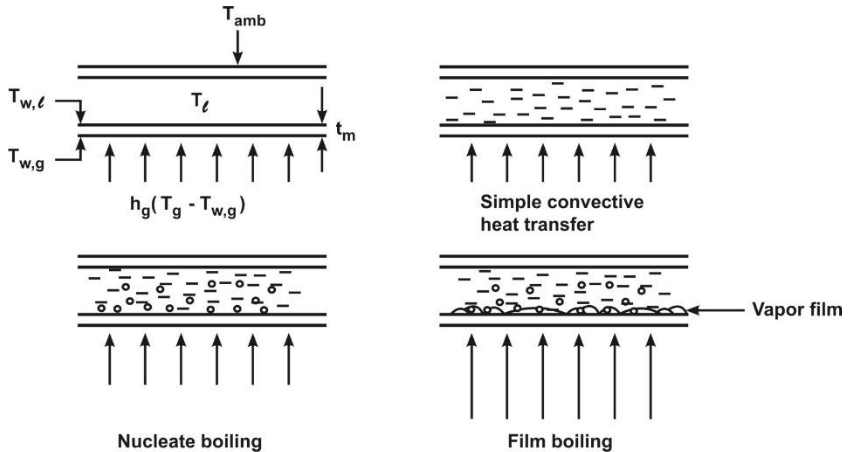
Re = ṁ/ (4 π d μ)

Table 8: Coefficients for Nusselt –correlations coolant side heat transfer

Fuel	Pc MPa	Uch m/s	q MW/m²	Tb K	Tw K	K	a	b	c	d	e	f	g	m	n
C_2H_6	13.7	6-30		420-810		0.00538	0.80	0.4	-0.125	0.242	0.193	0.395	-0.024	1	0
	3-12.4	15-45	0.3-16.4	120-395		0.00545	0.90	0.4	-1.1	0.23	0.27	0.53	0	1	0
	7-13.8	30-60	3.1-18.2	230-300	230-810	0.0280	0.80	0.4	0	0	0	0	0	0	0
						0.00696	0.88	0.5	0	0	0	0	0	0	-1.0
CH_4	27-34	55-238	2.6-139	146-275		0.0220	0.80	0.4	0	0	0	0	0	0	-0.45
	27-34	55-238	2.6-139	146-275		0.0230	0.80	0.4	0	0	0	0	0	0	-0.57
	7-13.8	30-60	3.1-18.2	230-300	230-810	0.0230	0.80	0.4	0	0	0	0	0	0	0
						0.0230	0.80	0.4	0	0	0	0	0	0	-0.80
						0.0230	0.80	0.4	0	0	0	0	0	0	0
RP-1	7-13.8	30-60	3.1-18.2	230-300	230-810	0.0440	0.76	0.4	0	0	0	0	0	1	0
	7-13.8	30-60	3.1-18.2	230-300	230-810	0.0068	0.94	0.4	0	0	0	0	0	0	0
						0.0056	0.90	0.4	0	0	0	0	0	0	0







Combustion in Liquid rocket combustion chamber - 1

- The combustion process is quite complex in the case of self-igniting (hypergolic) propellants.
- The liquid – to - product conversion process involves significant liquid-liquid reaction. This is unlike non-hypergolic propellants where atomization process has more direct role.
- The extent of liquid phase mixing depends on the injection diameter and the velocity.
- Injection process is designed to reduce the coupling between the between the combustion chamber processes and the feed system dynamics. This reduces incidence of low frequency instability.
- Typical pressure drop across the injectors (Δp) is about 8 to 12 atm. This leads to velocities of 30 to 50 m/s (allowing for frictional resistance accounted by a coefficient of discharge, c_d as in $V_l = c_d\, A_{inj} \sqrt{2\, \rho_l \Delta p}$)
- The drop size due to impingement and other processes is proportional to the injection hole diameter and reduces with increasing velocity {$d_l/d_{inj} \sim 1/We^n$, We = Weber number = $\rho_l U_l^2/(\sigma/d_l)$, $n \sim 0.5$}
- When jets impinge, the liquids mix and also break up into droplets. Liquid phase reaction leads to heat release and break-up of the liquid into finer droplets. These droplets interact with each other at varying mixture ratios and release heat.
- There will also be fast gas phase reactions that lead to near-chemical equilibrium composition and $T_{f,\, adiab}$
- The time it takes for this to occur and the distance travelled in this period settles the combustor size.
- In this case, it is only decided by experiments on actual systems. Combustor size is decided by $L^* = V_c/A_t$ where L^* = Characteristic length, V_c = Combustion chamber volume (up to throat), A_t = throat area
- Typical value of L^* for hypergolic propellants is about 0.7 to 0.9 m, with higher p_c having lower L^*
- For given engine thrust and a choice of p_c, one can get A_t. With this A_t, and a choice of L^*, we get V_c. With a choice of chamber to throat cross sectional area, we get chamber diameter.

Combustion in Liquid rocket combustion chamber - 2

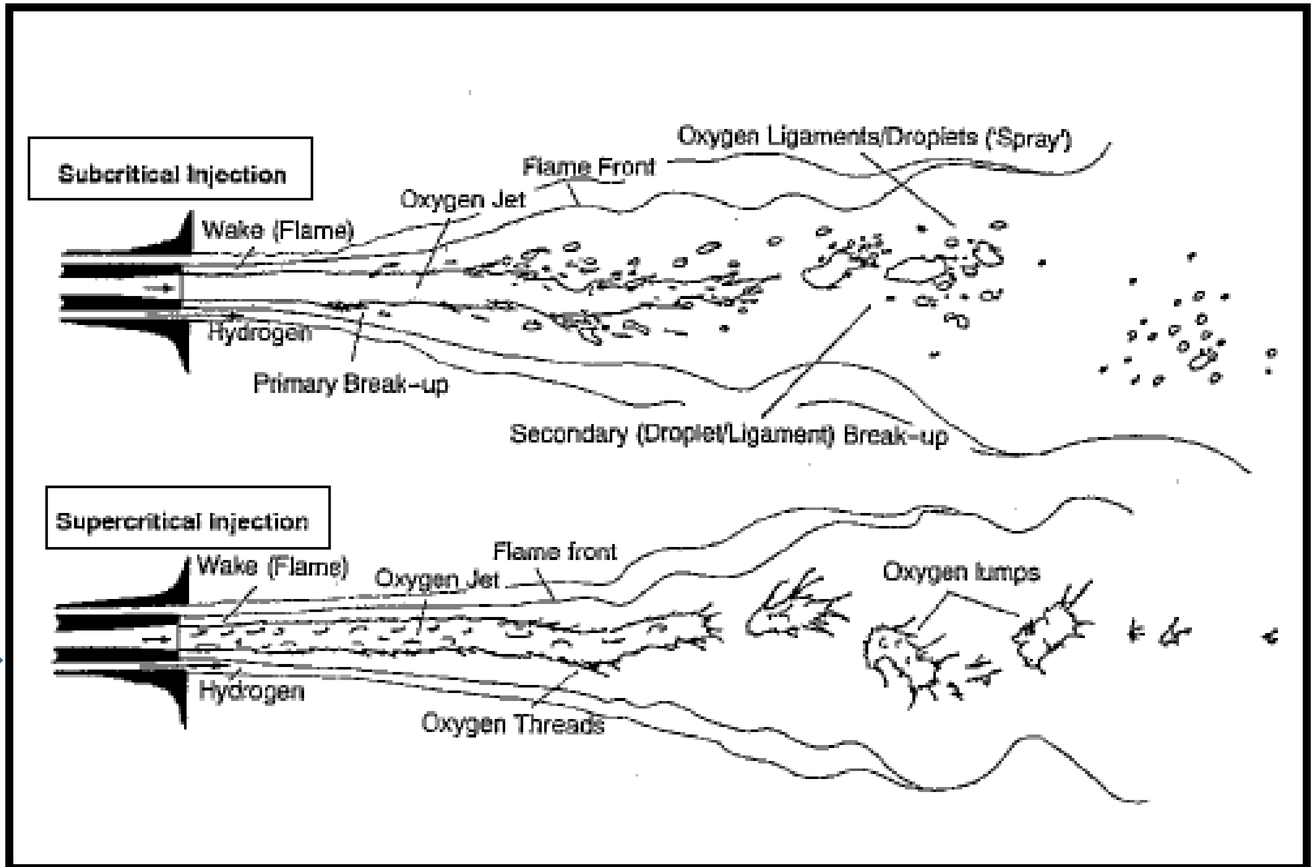
- The non-hypergolic propellants used are LOX-Kerosene and LOX – LH_2.
- Both kerosene and hydrogen are used as a regenerative coolant. Kerosene is close to boiling and hydrogen will always be a gas
- V_{inj} for liquids ~ 30 m/s, for gas ~ 150 m/s
- For impinging jets or swirling jets,drop size due to impingement and/or primary and secondary atomization processes is proportional to the injection hole diameter and reduces with increasing velocity
- Coaxial injection systems show atomization, vaporization and reaction processes depending on whether the combustion process occurs under supercritical conditions. Experiments have shown the difference between the two.

LOX ~ 100 K GH_2 injection temp ~ 150 K,

LOX vel ~ 30 m/s, GH_2 vel ~ 150 m/s,

Dox = 1 mm, Fuel – 3 mm (OD) x 1.6 mm (ID)

O/F ~ 4, p_c ~ 10 to 100 atm, η_{c*} ~ 90 %+



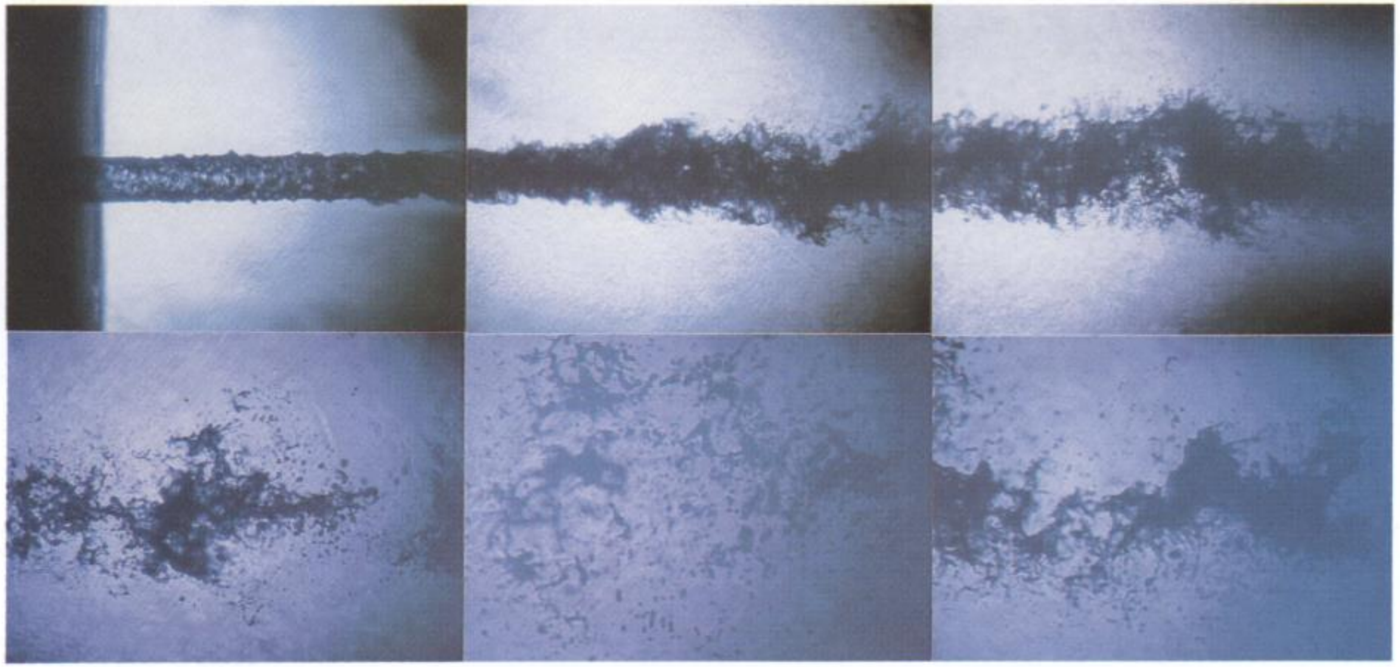
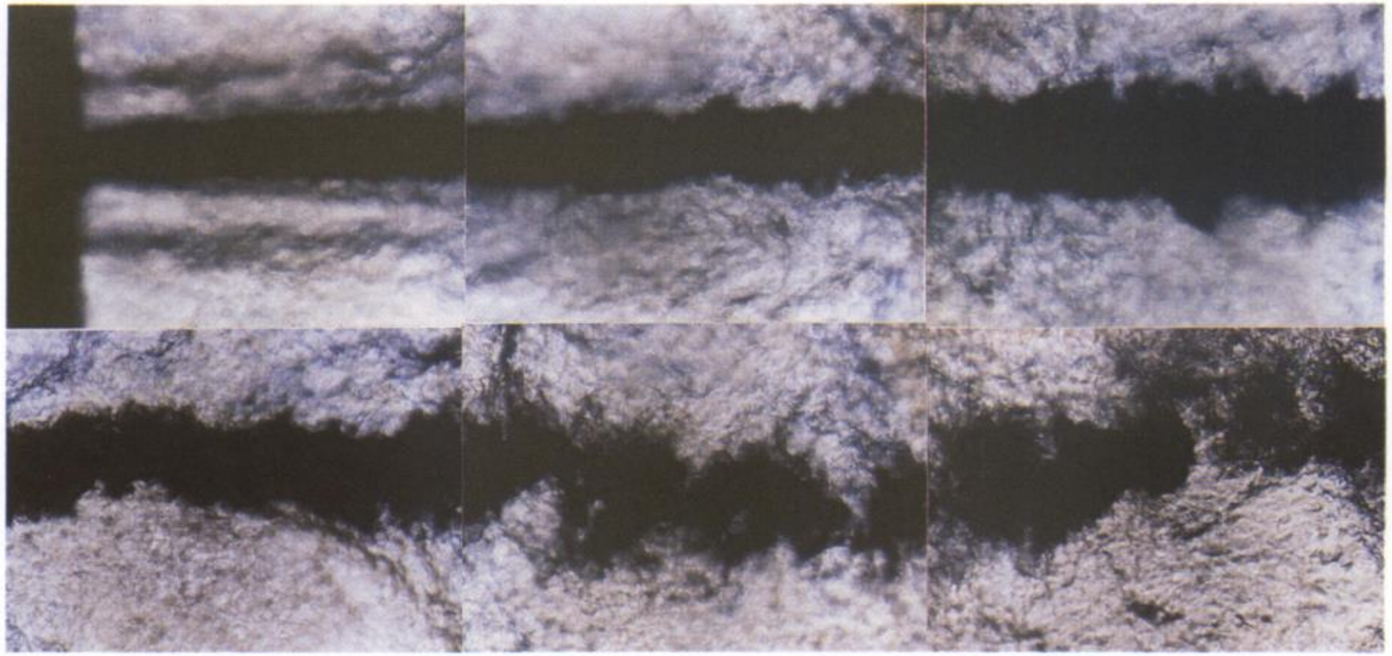
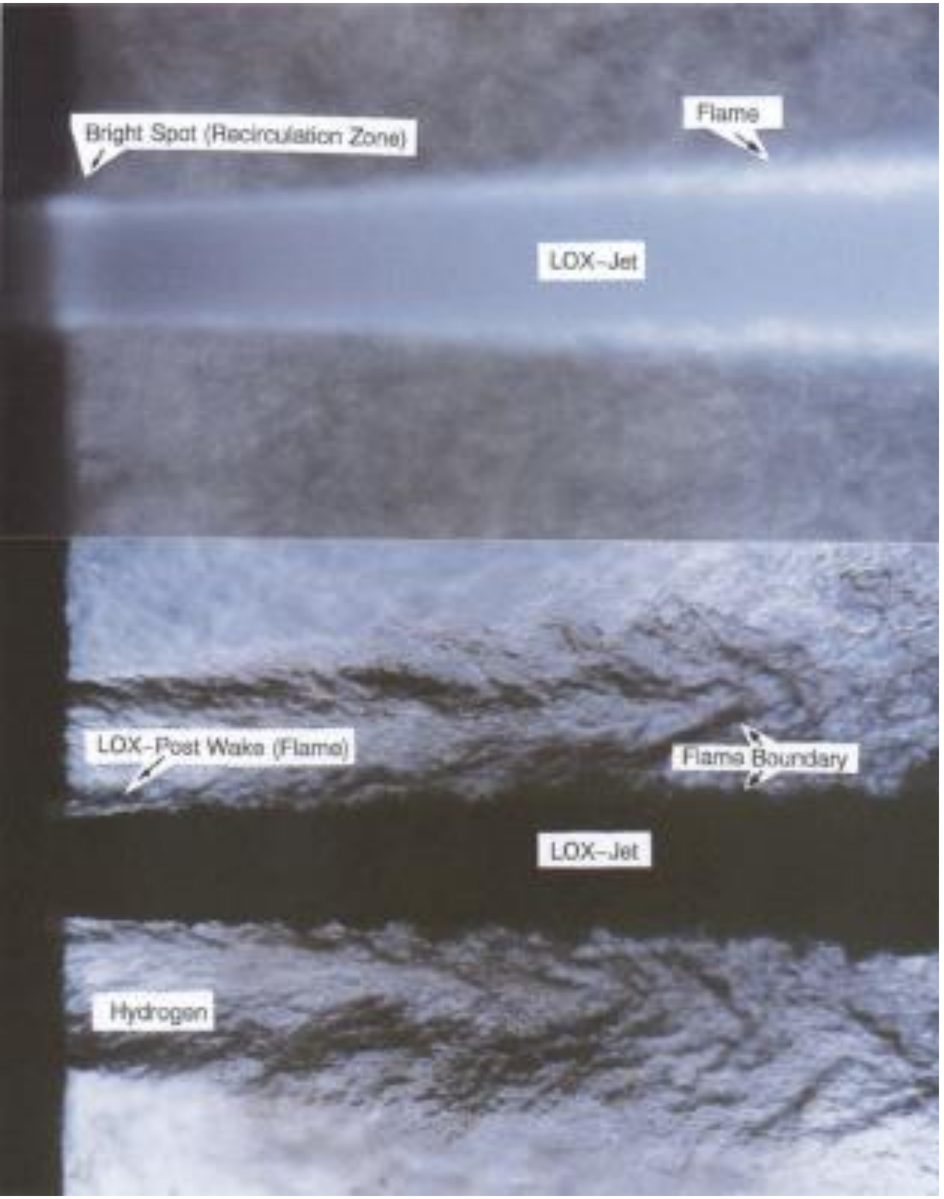


Fig. 4 Oxygen jet, subcritical injection, combusting condition: oxygen velocity is 10 m/s, hydrogen velocity is 300 m/s, $d = 1$ mm, chamber pressure 1.5 MPa (test case 1), from left to right and top to bottom: axial position $x = 0$ (faceplate), 12, 24, 36, 48, and 60 mm.

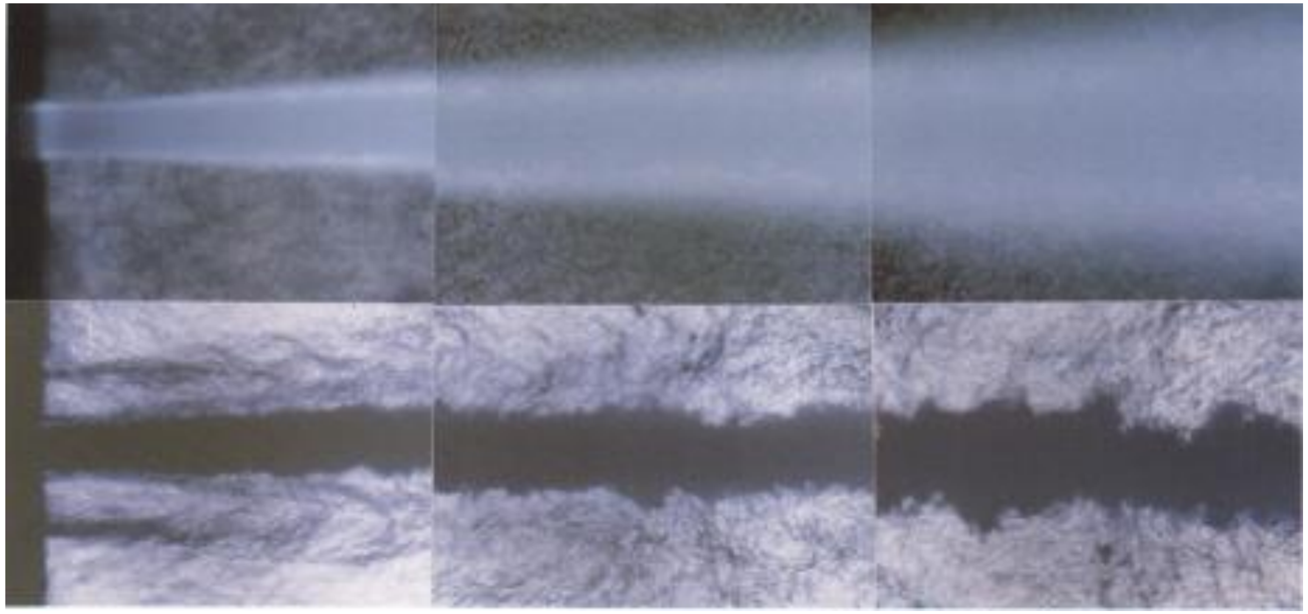


From the LOX jet core, thread like structures develop and grow. They do not detach, but dissolve and fade away. Several tens of diameters downstream, LOX core breaks up into large LOX lumps dissolving the same way.

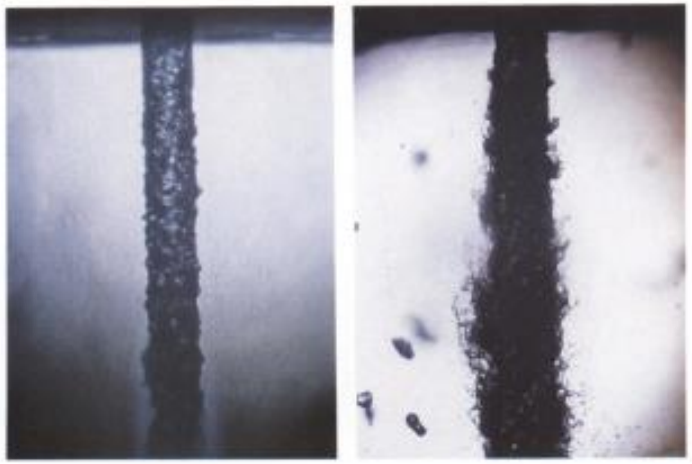
Jet break up length decreases with pc. for the 10 MPa case, oxygen lumps have completely depleted by 70 diameters



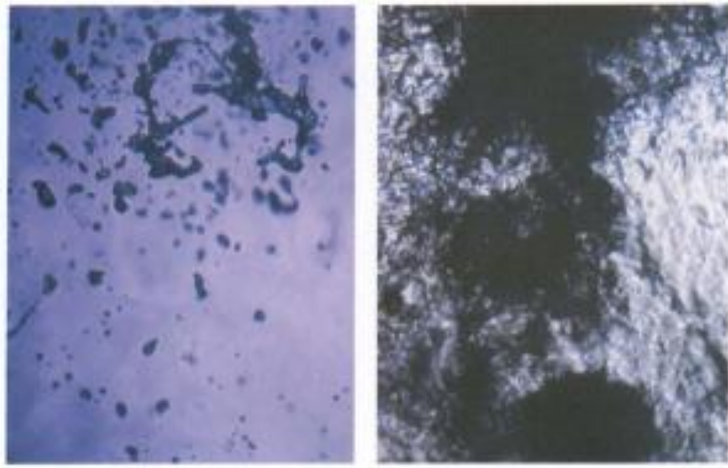
Near injector region, 30 m/s (LOX), 300 m/s (GH2)
pc = 4.5 MPa. Top – Flame, bottom – flow field



Composite picture – top – burning, bottom – flow field,
Same case as in the left



Near Injector region,
Left – burning, right – before ignition



Burning, 60 mm downstream
6 mm x 4.8 mm zone
left: 1.5 MPa, right – 6 MPa

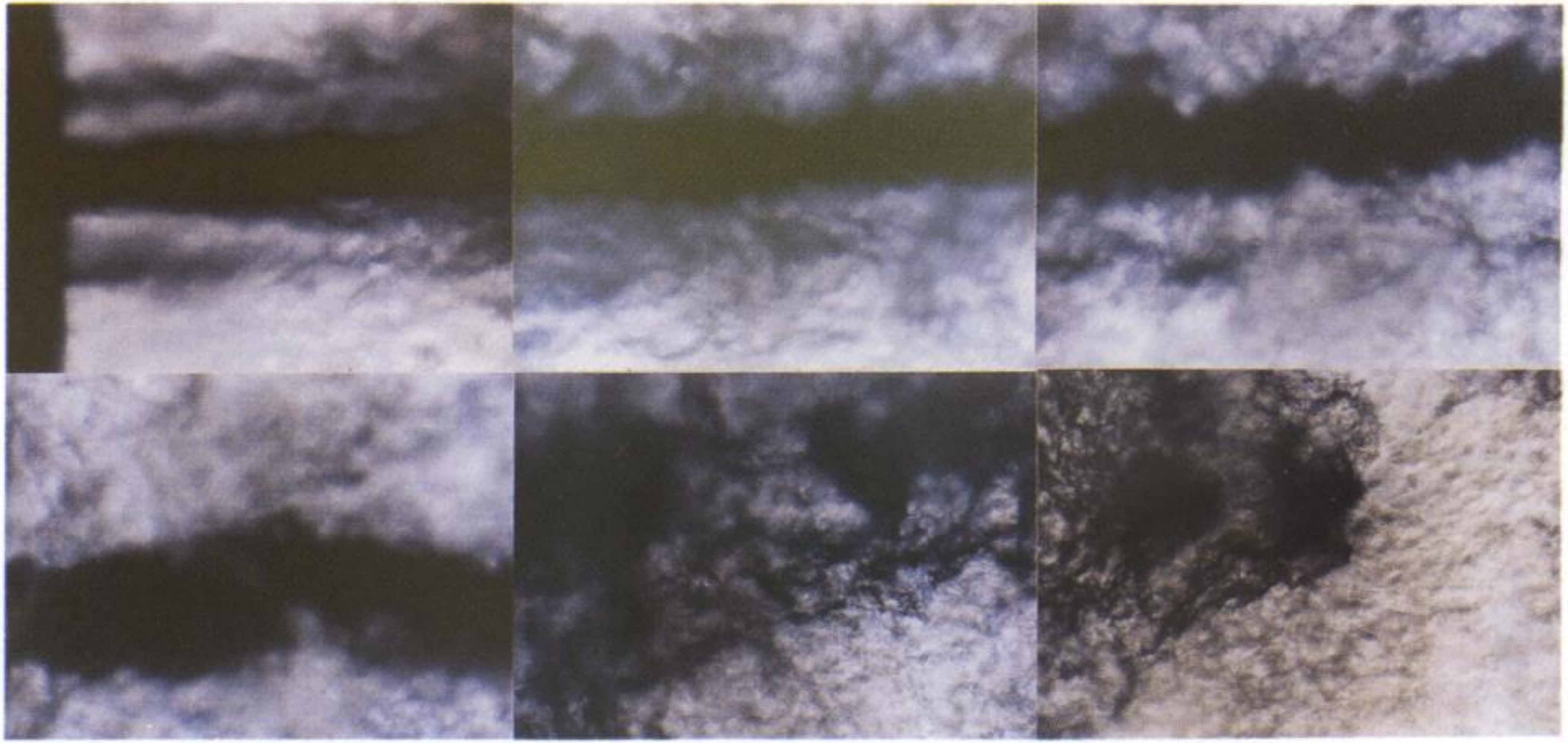


Fig. 6 Burning oxygen jet, supercritical injection, combusting condition: oxygen velocity is 30 m/s, hydrogen velocity is 300 m/s, d = 1 mm, chamber pressure 10.0 MPa (test case 4), from left to right and top to bottom: axial position x = 0 (faceplate), 12, 24, 36, 48, and 60 mm.

Combustion and Atomization process

Classically, one would think as follows:

- When you want to burn them efficiently as in a diesel engine or rocket engine, you arrange such that the liquid becomes a fine droplet. Why so?
- The burn rate of a droplet with diameter *d* is given by

$$\dot{m} = 2\,\pi\,\rho_l\,d\,\ln(1 + B) \quad \rightarrow \quad t_b = \text{Const.}\ d^2$$

where d = drop diameter, ρ_l = density, B = thermo-chemical parameter.

- The aim of the design of a diesel engine or a rocket engine system is to burn as much of fuel per unit time in a given volume. This is translated to saying: the combustion process must be completed within a certain time. This is about a 2 to 4 millisecond.
- It takes about 3 s to burn a 1 mm dia diesel droplet (Const = 3 s/mm^2).
- Therefore, the drop dia should be $\sqrt{3 \times 10^{-3} \times 10^{-6}/3}$ ~ 31 micrometers. This is in fact the typical diameter aimed to be obtained by atomization of the liquids. To appreciate what all this means….

Liquid atomization…..

- Consider d_0 = 1 mm and d_1 = 30 micrometer (μm)droplets
- For equal mass or volume, $(\pi/6)\, d_0^3 = N\,(\pi/6)\, d_1^3$; so, $N = (d_0/d_1)^3$,

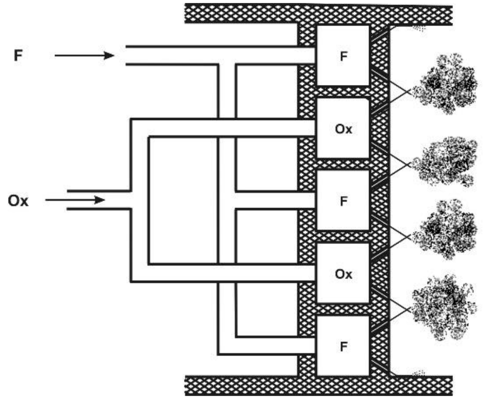
  N = 33000. Hence, a 1 mm drop produces 33000, 30 μm drops.
- The surface area ratio is $N\, d_1^2 / d_0^2 = d_0/d_1 = 32$.

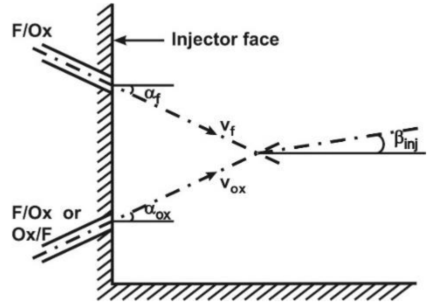
- Thus by atomizing the liquid, one increases the number of drops enormously and increase the surface area as well. Since this reduces their diameter, their burn time is reduced enormously – by a factor of 1000!
- To achieve this level of atomization, the diesel engine injects the liquid through an orifice of ~ 150 to 200 μm dia (0.15 to 0.2 mm) at pressures of 300 to 500 atm for a brief while ~ 0.5 ms! every cycle which for a 1500 rpm engine is about 25 ms and this injection occurs once in 50 ms for a four stroke engine

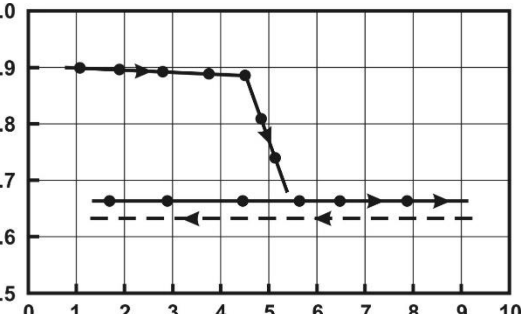
- Following the combustion dynamics of droplets makes sense at high density of fuel injection without any impingement.
- Otherwise, impingement dynamics needs to accounted true for for rocket engines…….

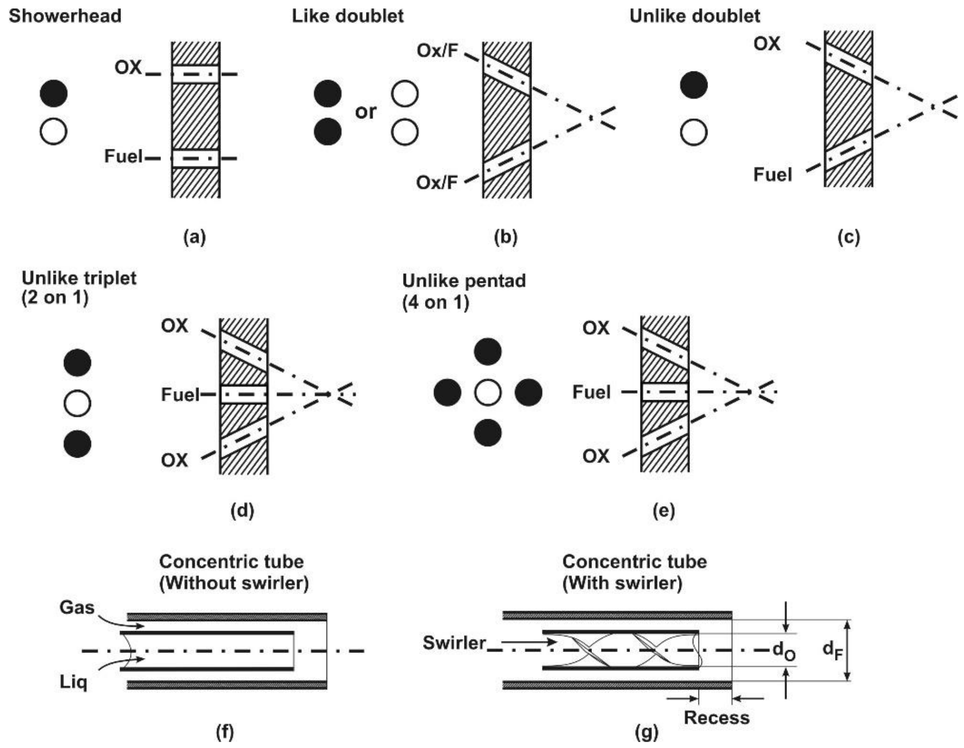
Aspects of atomizers in practical systems

Impinging arrangement

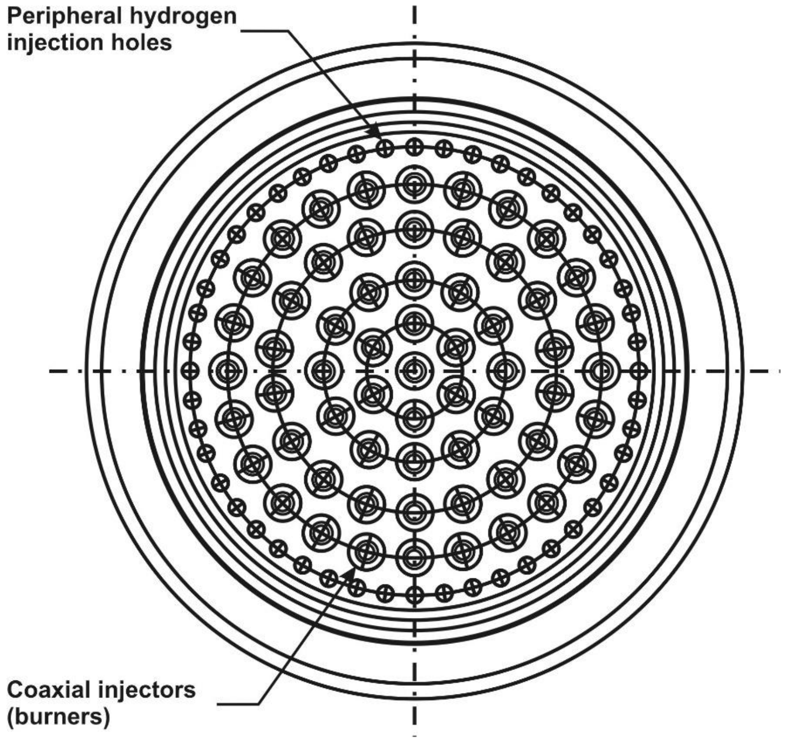




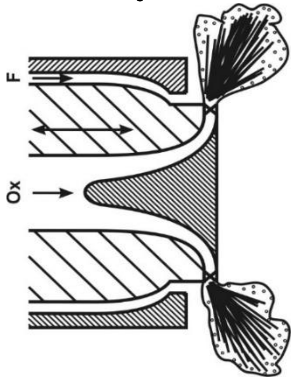


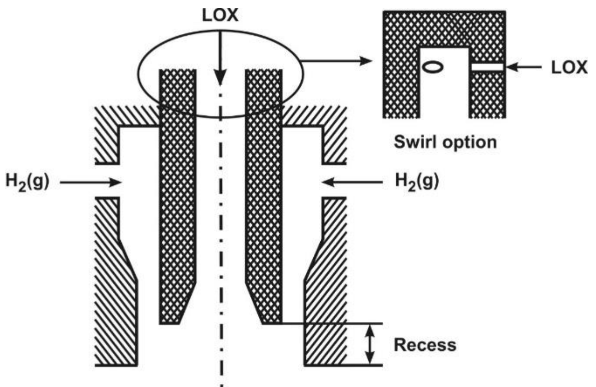


Coaxial injector



Pintle injector



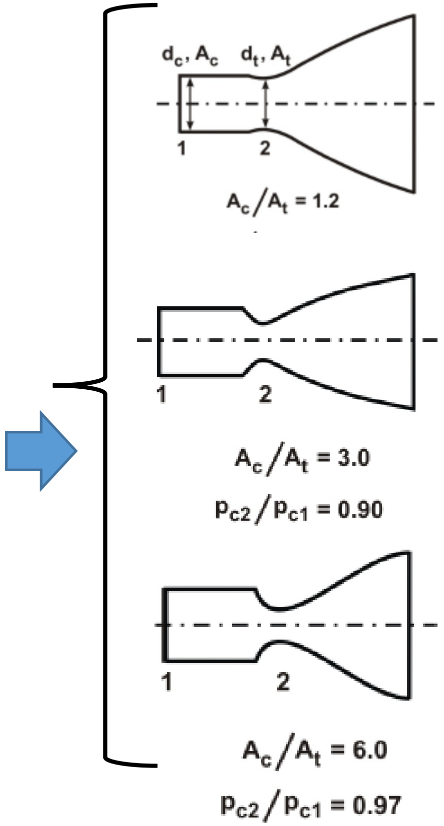


Further on combustion in liquid rockets - 1

- The combustion processes in advanced liquid rocket engines at very high pressures borders on critical to supercritical combustion processes with the need to evolve a proper equation of state.
- Impinging injectors are used in upper stage engines and also catalytic monopropellant thrusters
- Most large engines use coaxial injectors like one discussed earlier (for LOX-LH_2 and LOX-Kerosene)
- Many researchers have experimentally investigated the subject in the last two decades.
- An examination of the combustion behavior shows that the propellants coming out of the injectors have very limited interaction with neighbors – lateral mixing is small. This is particularly true of coaxial injectors.
- Thus creating a near uniform O/F distribution across the cross-section (excepting the near wall region that has film cooling of the fuel) seems appropriate to get high performance.
- Impinging jet injectors have a mass flux distribution involving the droplets over a distance of five to ten injector hole diameters and this is susceptible in response to acoustic oscillations
- The flow behavior for coaxial injectors seems more complex, particularly when the process is close to critical conditions
- It appears efforts are needed to create a well justified computable model of the steady combustion process from the physical processes in coaxial injectors

Further on combustion in liquid rockets - 2

- Two key parameters of combustor design are chamber diameter and length. There are others that need to be definition are no. of injection holes/coaxial injectors that control the density of propellant injection.
- Experiments have shown that L^* for LOX-LH_2 system is about 0.7 m and for LOX-Kerosene is about 1.2 m.
- From the demand of thrust and a choice of chamber pressure, one can calculate A_t and then V_c. With a choice of contraction ratio, A_c/A_t (1.5 to 3), one can get the chamber diameter.
- Smaller contraction ratio means smaller combustion chamber diameter – smaller cooling surface area, desirable for the optimization of regenerative cooling process.
- On the other hand, the mass flux through the combustion chamber will be higher and causes stagnation pressure drop due to friction and hence loss of specific impulse.
- The compromise is dependent on the designer.
- Generally, Russians have used higher contraction ratio compared to Americans.



Further on combustion in liquid rocket engines - 3

Table 7: Characteristic data of sample injectors [10]

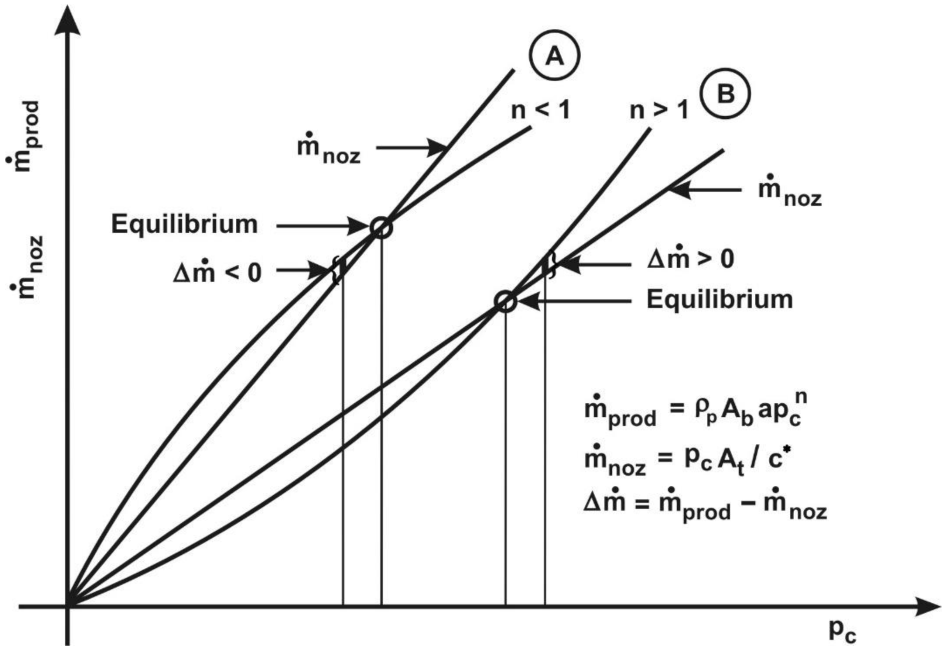
Engine	F-1	RD-170	SSME	RD-0120	LE-7A	RS-68	Vulcain 2
Thrust (sl) [MN]	6.9	7.6	1.8	1.52	0.86	2.9	0.94
Propellant combination	LOX / RP-1	LOX / kerosene	LOX / LH2	LOX / LH2	LOX / LH2	LOX / LH2	LOX / LH2
Injector type	Imping-ing	Swirl coax	Shear coax	Shear coax	Shear coax	Shear coax	Shear coax
Flow rate / element [kg/s]	1.7	1.8	0.9	~1	0.85	~0.9	0.55
Thrust / element [kN]	4.6	5.5	3.8	3.4	3.1	~ 3.2	1.6

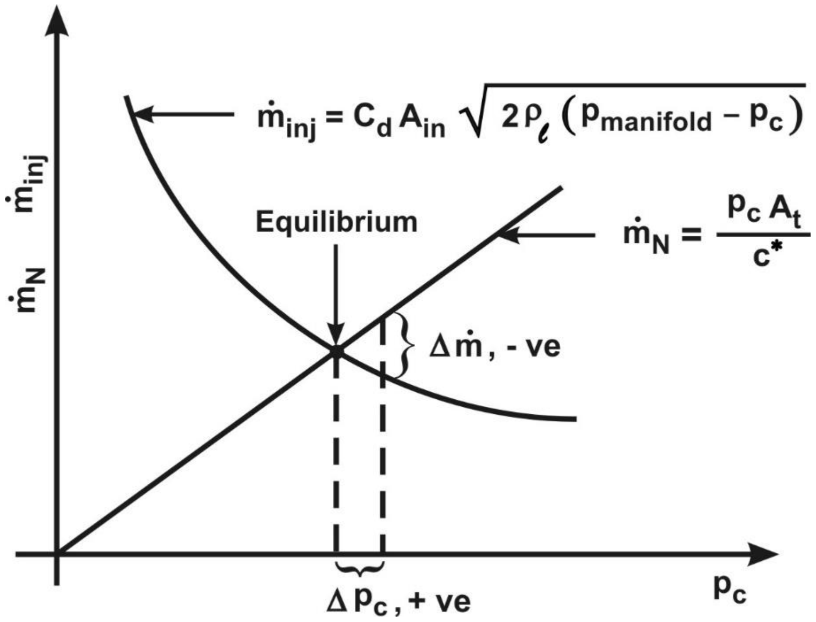
Flow rate per element is an indication of intensity of combustion process. As can be noted, Russian engines make a choice of much higher density of propellant injection with coaxial injection system That troubled them much less in terms of instability than for F-1 engine

In Summary,

- Many aspects of liquid propellant rockets have been explored – very briefly
- Changes in space propulsion systems will occur through the introduction of green propellants. Research is underway at this time in ISRO laboratories.
- Upper stage propulsion systems will use self-igniting MMH-N_2O_4 pressure fed systems
- Large engines will
  - **get sustained with current UDMH-N_2O_4 (VIKAS) systems. Minor developments may also take place.**
  - **involve absorption of technologies on staged combustion cycle based semi-cryo LOX-kerosene engines**
  - **involve including minor improvements into LOX-LH_2 engines**
  - **involve development of LOX-CH_4 engines around the current LOX-LH_2 engines**
- Injection systems on the semi-cryo and full cryo engines will be coaxial injectors. Issues of steady combustion are already dealt with or will be dealt with.
- Dealing with possible problems of combustion instability on the semi-cryo engines will require better physics based computational approach in coming times
- Combustion processes inside high pressure combustion chambers use larger injector diameter to be with in stable operating range. This has ideas to create stable liquid rocket engines.

Static stability of rocket engines





In a solid rocket engine, the burn rate index, n should be less than 1 so that the operation is statically stable: (A) in the above figure.

In a liquid rocket engine, the operation is always statically stable due to propellant Injection rate decreasing when chamber pressure increases and vice versa.

This is also true of hybrid rockets because with an O/F ~ 2 to 3, the behavior of total flow rate from the combustion chamber is controlled by the oxidizer just as above.

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