Propulsion Systems Design

- Class notes
- Rocket engine basics
- Survey of the technologies
- Propellant feed systems
- Propulsion systems design

Class Notes

- No class on Thursday, March 26
- No midterm exam
- Final exam will be a take-home
- Sorry about the delay - you will be getting problem sets and solutions back ASAP

Thermal Rocket Exhaust Velocity

- Exhaust velocity is

$$V_e = \sqrt{\frac{2\gamma}{\gamma - 1}\frac{\Re T_0}{\overline{M}}\left[1 - \left(\frac{p_e}{p_0}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

where

$$\overline{M} \equiv \textit{average molecular weight of exhaust}$$

$$\Re \equiv \textit{universal gas const.} = 8314.3 \frac{\textit{Joules}}{\textit{mole}\,°K}$$

$$\gamma \equiv \textit{ratio of specific heats} \approx 1.2$$

Ideal Thermal Rocket Exhaust Velocity

- Ideal exhaust velocity is

$$V_e = \sqrt{\frac{2\gamma}{\gamma - 1} \frac{\Re T_0}{\overline{M}}}$$

- This corresponds to an ideally expanded nozzle
- All thermal energy converted to kinetic energy of exhaust
- Only a function of temperature and molecular weight!

Thermal Rocket Performance

- Thrust is

$$T = \dot{m}V_e + \left(p_e - p_{amb}\right)A_e$$

- Effective exhaust velocity

$$T = \dot{m}c \Rightarrow c = V_e + \left(p_e - p_{amb}\right)\frac{A_e}{\dot{m}}$$

$$\left(I_{sp} = \frac{c}{g_0}\right)$$

- Expansion ratio

$$\frac{A_t}{A_e} = \left(\frac{\gamma+1}{2}\right)^{\frac{1}{\gamma-1}}\left(\frac{p_e}{p_0}\right)^{\frac{1}{\gamma}}\sqrt{\frac{\gamma+1}{\gamma-1}\left[1-\left(\frac{p_e}{p_0}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

A Word About Specific Impulse

- Defined as "thrust/propellant used"
  - English units: lbs thrust/(lbs prop/sec)=sec
  - Metric units: N thrust/(kg prop/sec)=m/sec
- Two ways to regard discrepancy -
  - "lbs" is not mass in English units - should be slugs
  - Isp = "thrust/weight flow rate of propellant"
- If the real intent of specific impulse is

$$I_{sp} = \frac{T}{\dot{m}} \text{ and } T = \dot{m}V_e \text{ then } I_{sp} = V_e!!!$$

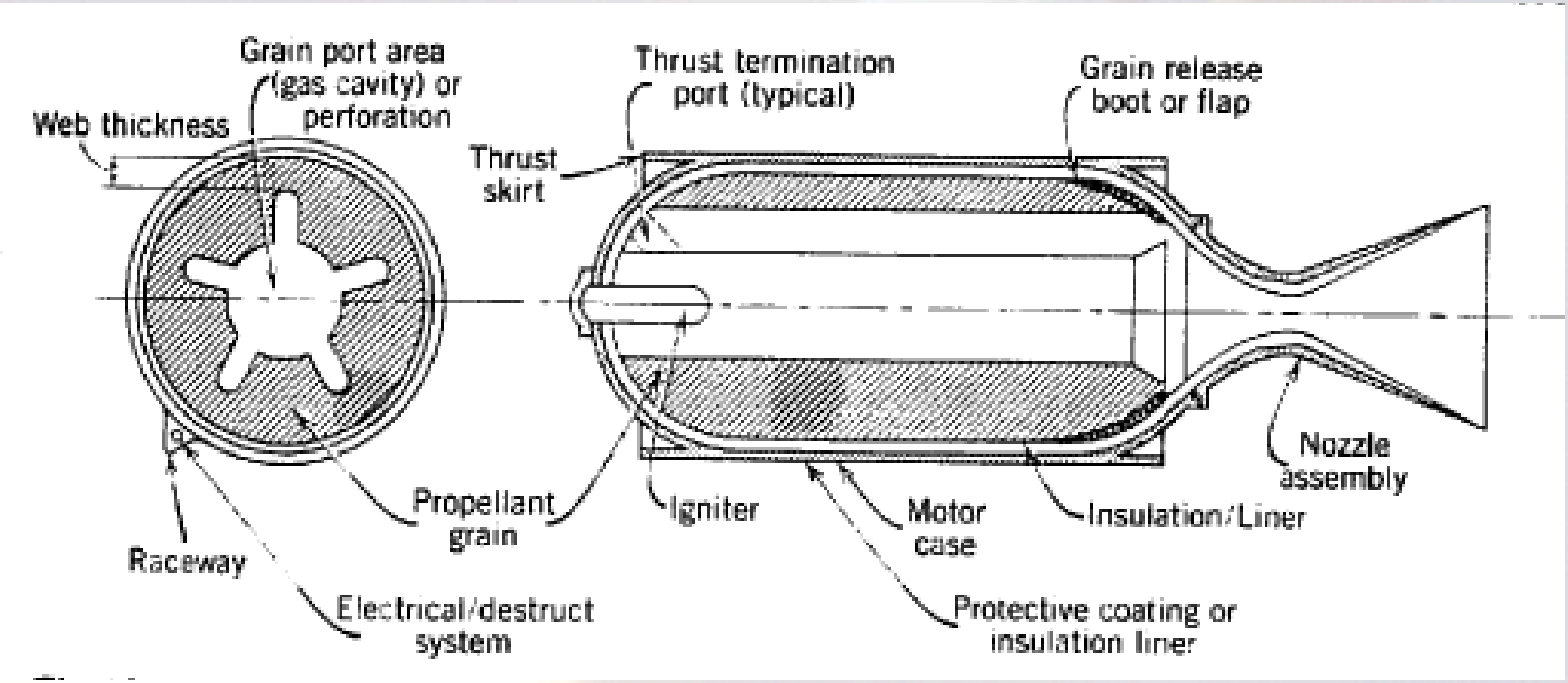
Nozzle Design

- Pressure ratio p_0/p_e=100 (1470 psi-->14.7 psi) A_e/A_t=11.9
- Pressure ratio p_0/p_e=1000 (1470 psi-->1.47 psi) A_e/A_t=71.6
- Difference between sea level and ideal vacuum V_e

$$\frac{V_e}{V_{e,ideal}} = \sqrt{1-\left(\frac{p_e}{p_0}\right)^{\frac{\gamma-1}{\gamma}}}$$

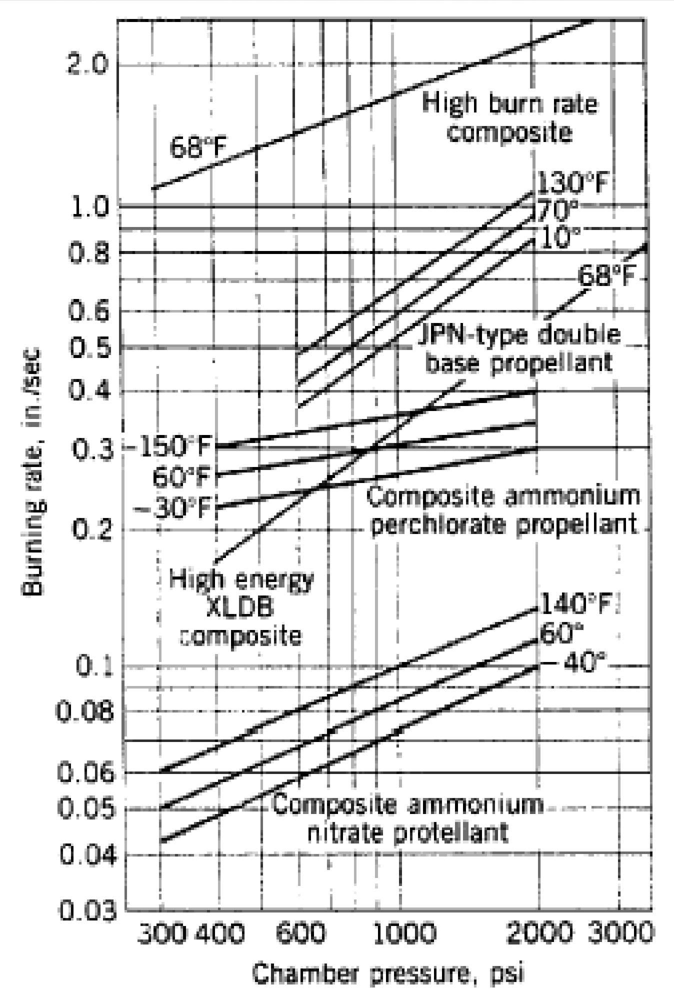
- $I_{sp,vacuum}$=455 sec --> $I_{sp,sl}$=333 sec

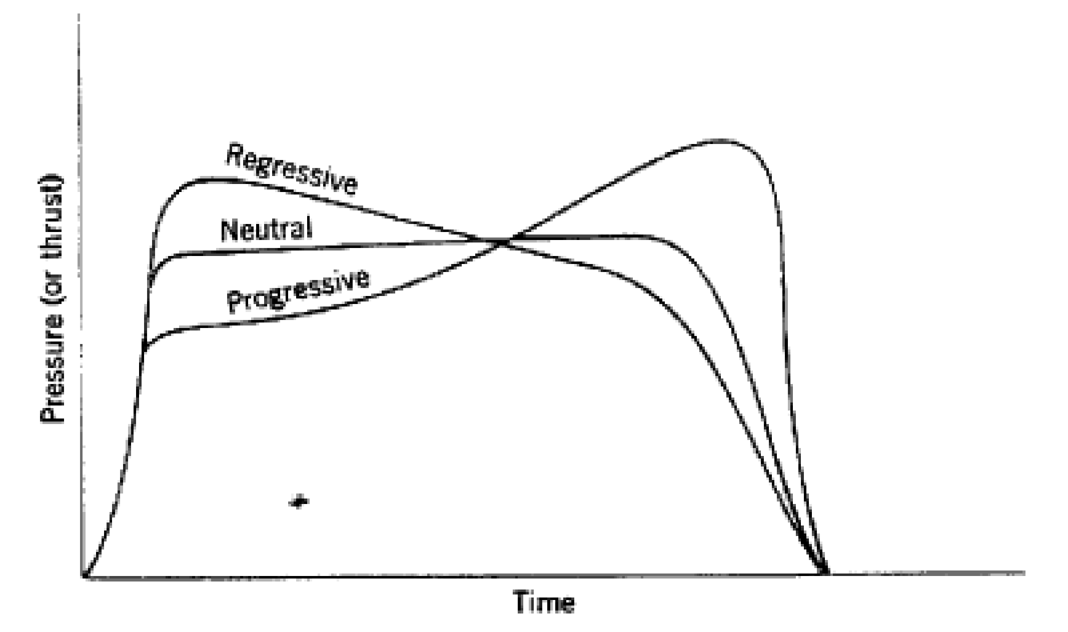
Solid Rocket Motor



From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

Solid Propellant Combustion Characteristics





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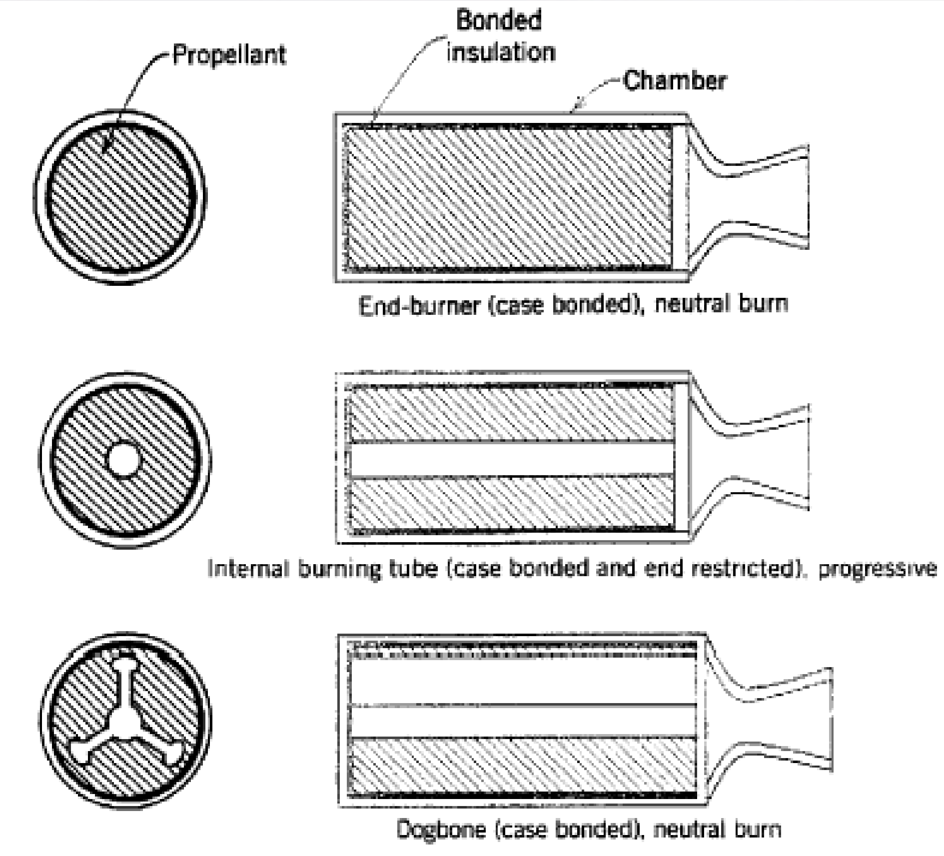
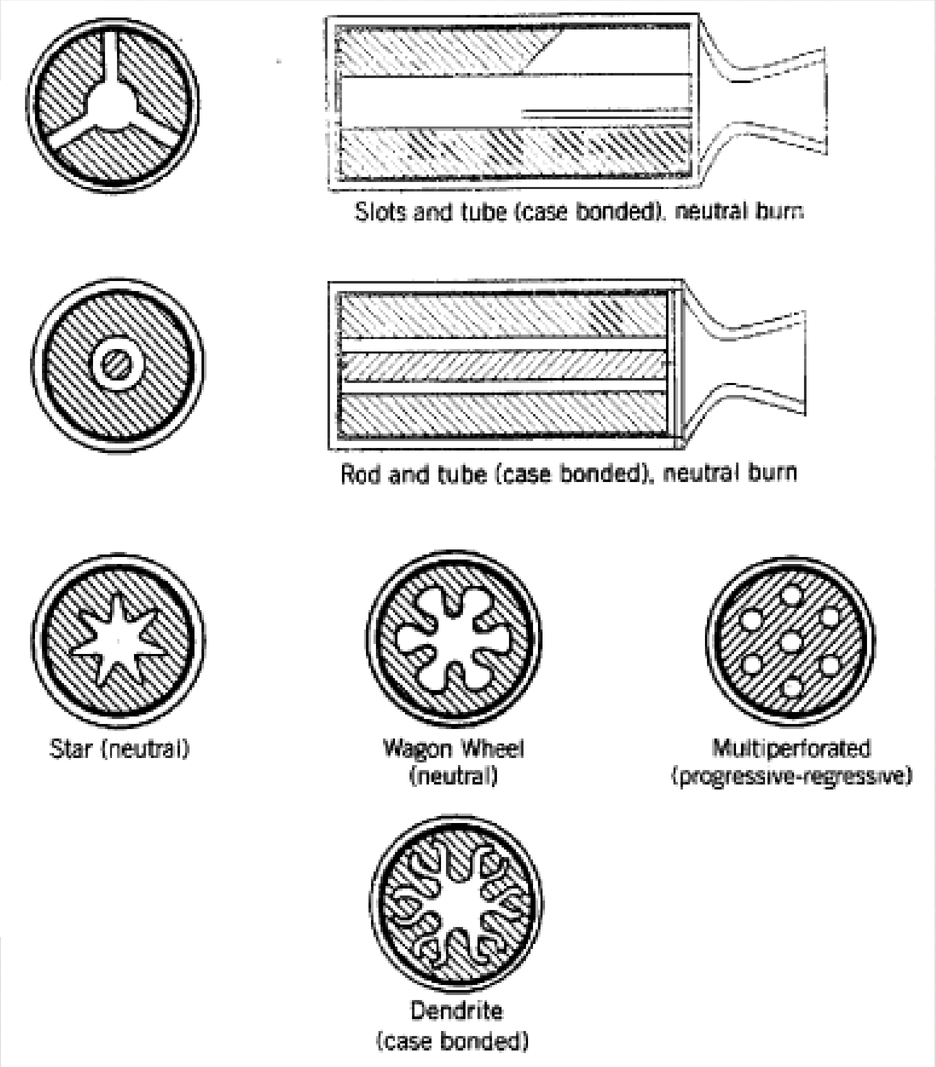
Solid Grain Configurations





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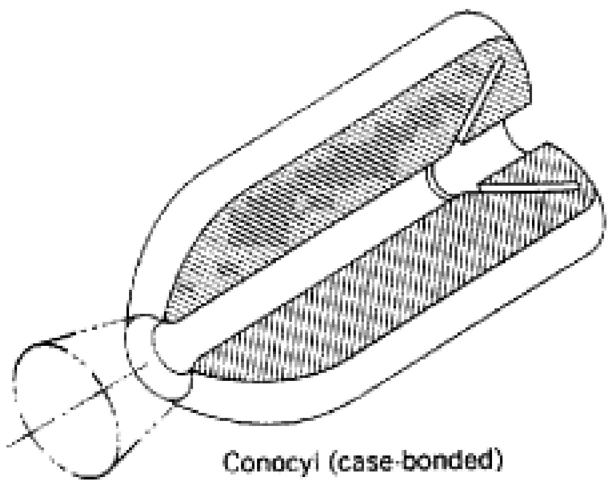
Short-Grain Solid Configurations



Conocyl (case-bonded)

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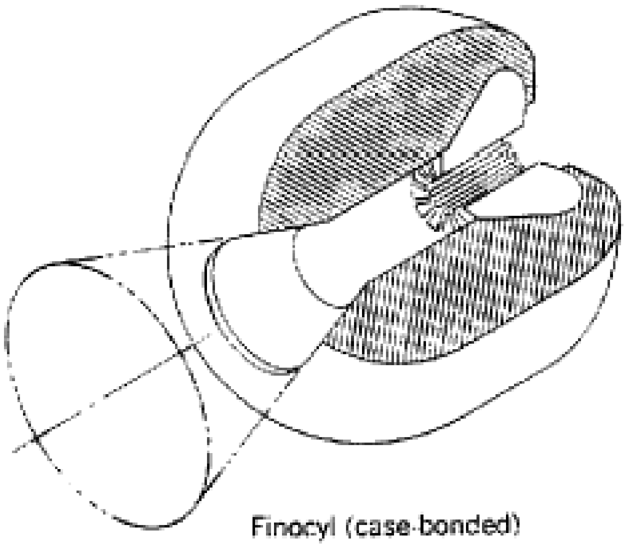
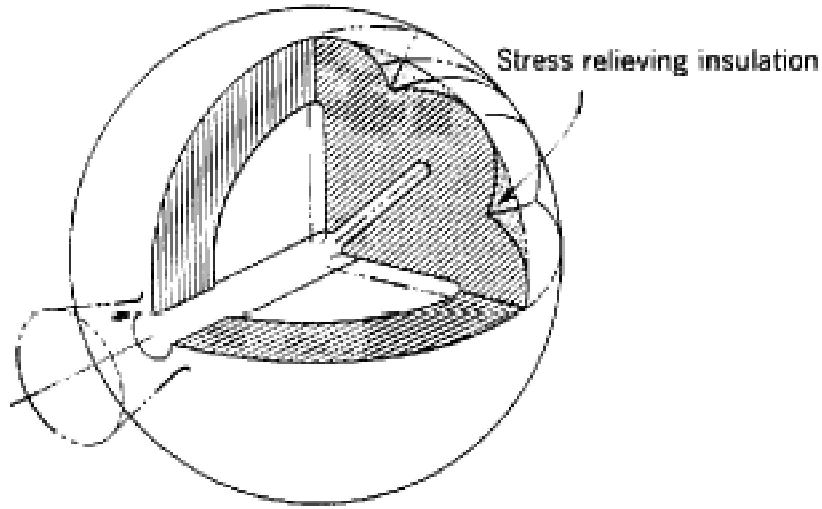
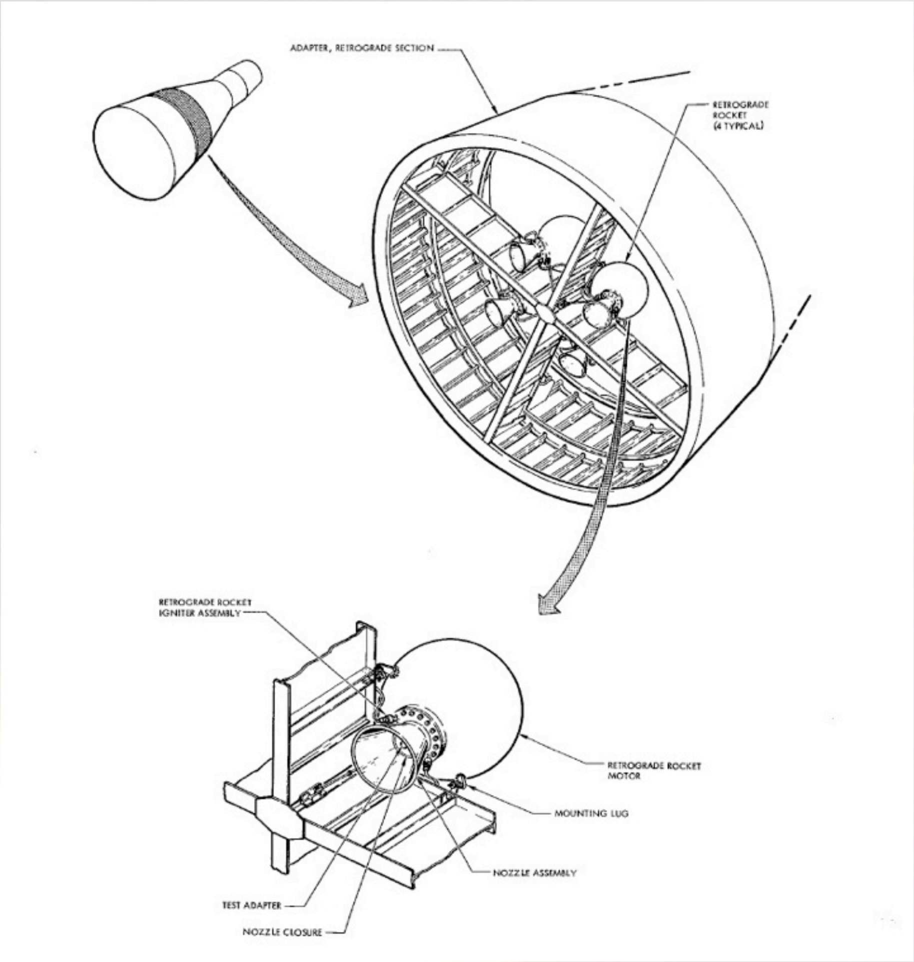


Finocyl (case-bonded)



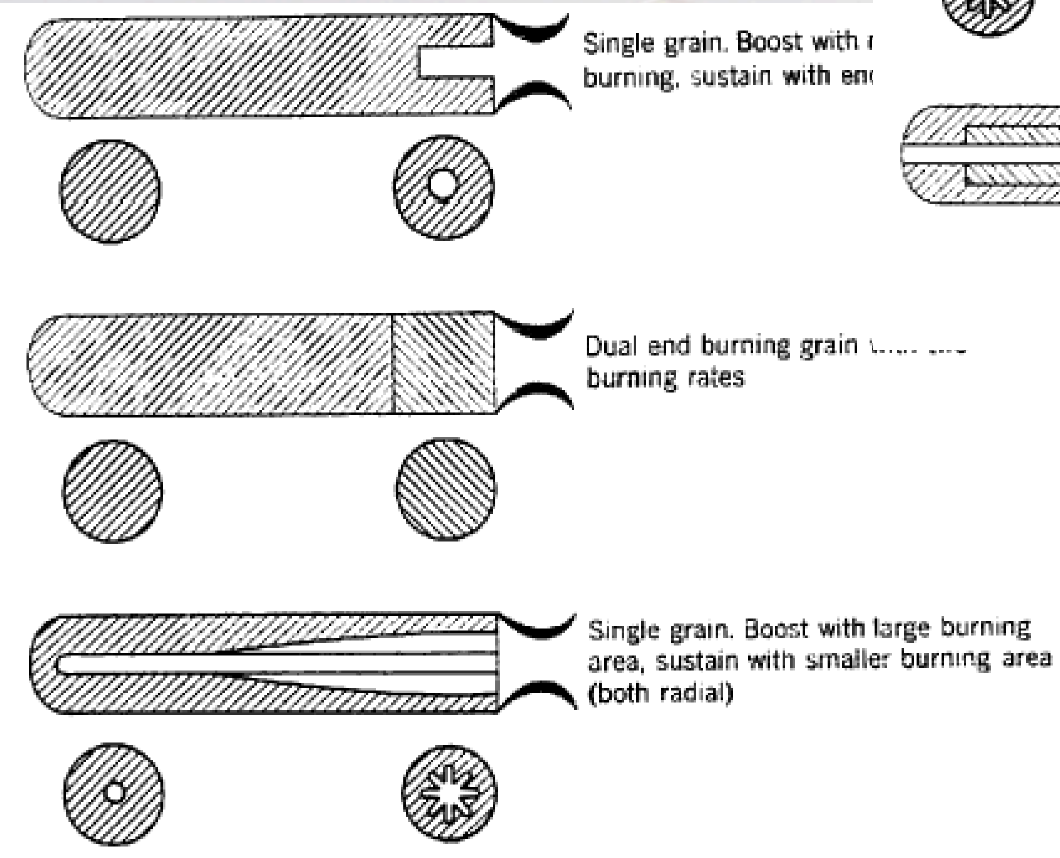
Spherical (case-bonded) with slots and cylinder

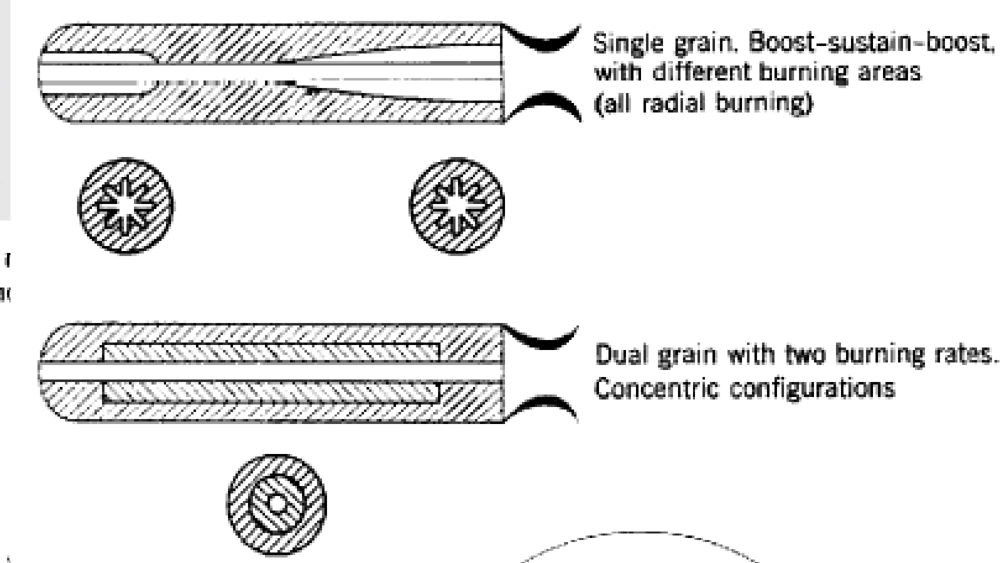
Gemini Retrograde Engine

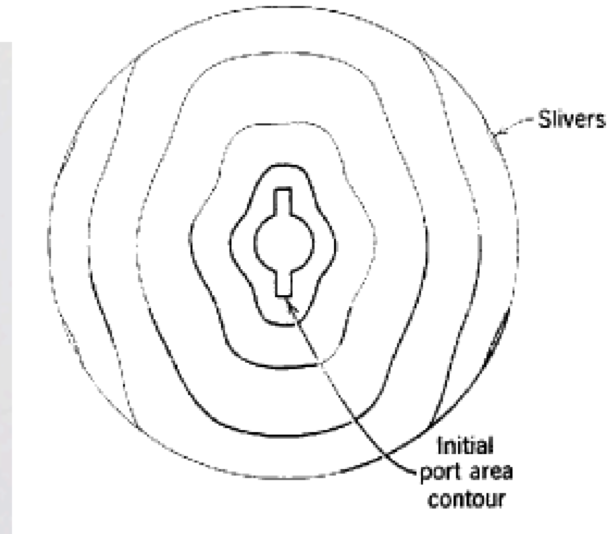


Advanced Grain Configurations

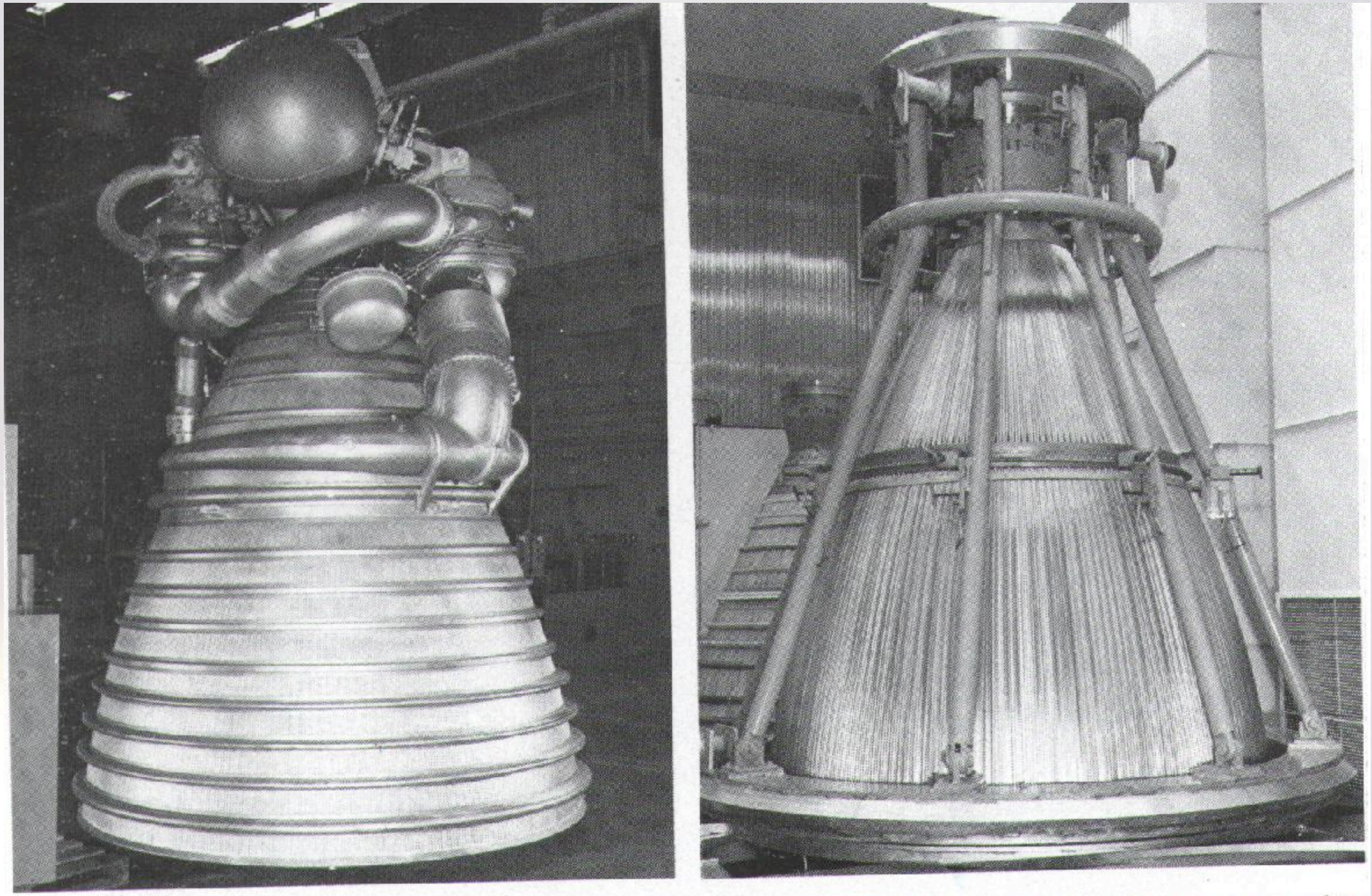
From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986





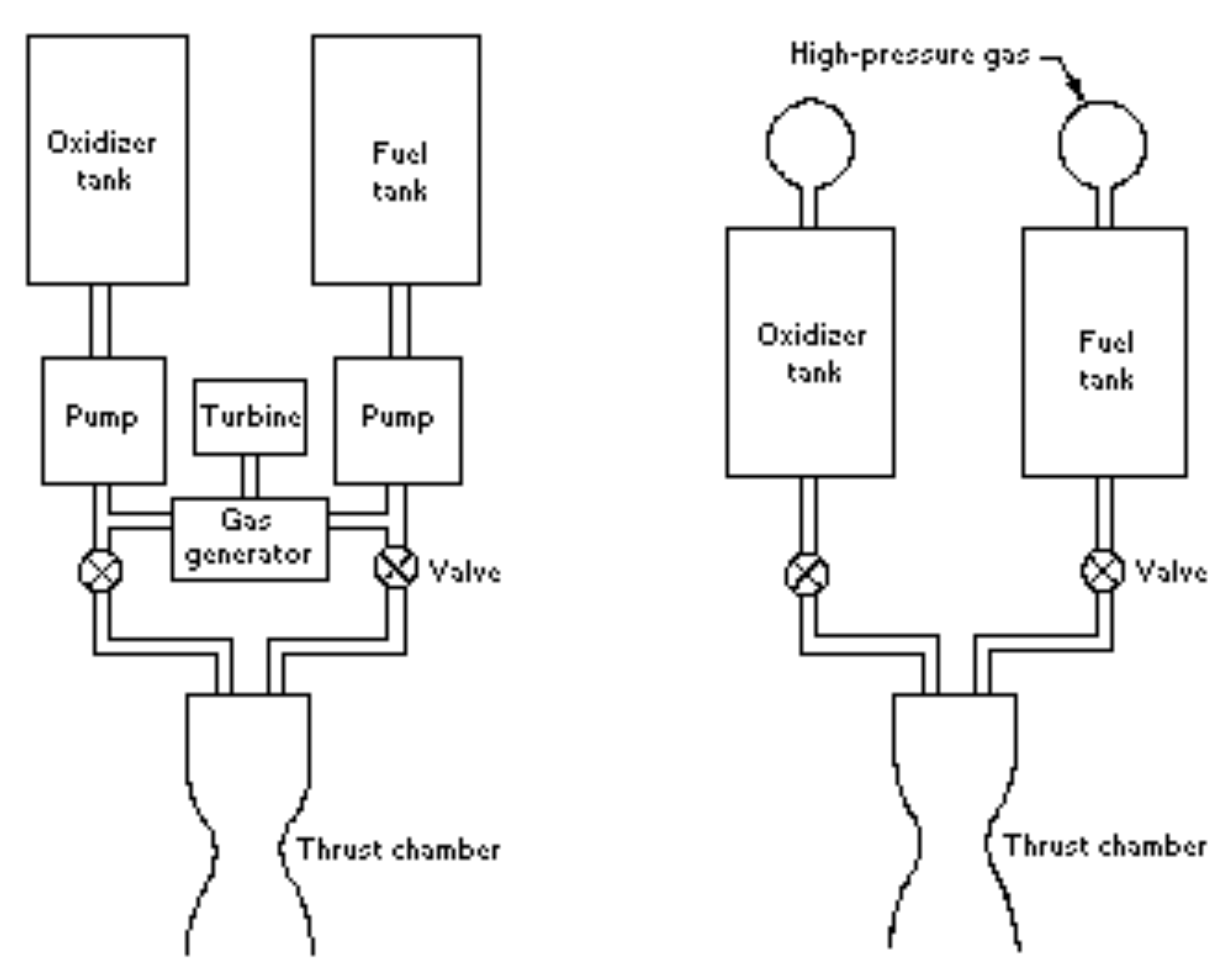


Liquid Rocket Engine



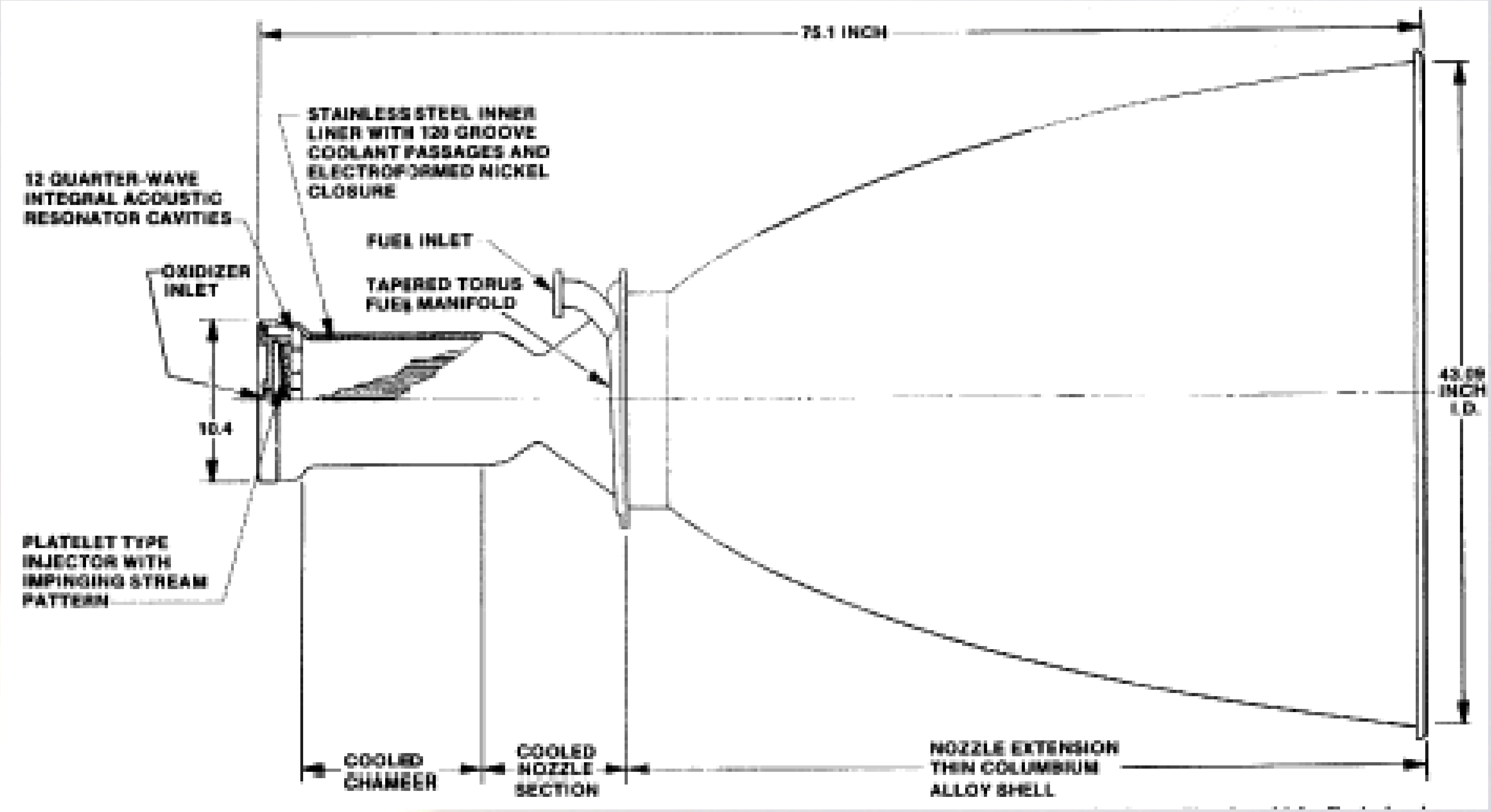
A completed J-2 rocket engine (left), with its pumps and lines installed. The basic engine structure is built up from a series of hollow tubes (right).

Liquid Propellant Feed Systems



(a) Pump-fed rocket (b) Pressure-fed rocket

Space Shuttle OMS Engine



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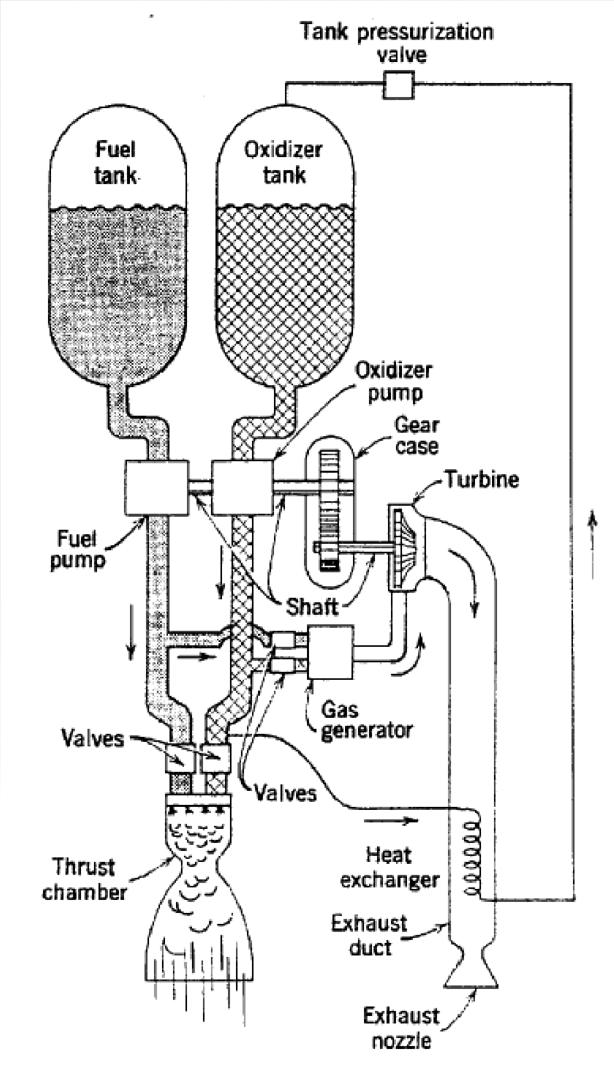
Turbopump Fed Liquid Rocket Engine



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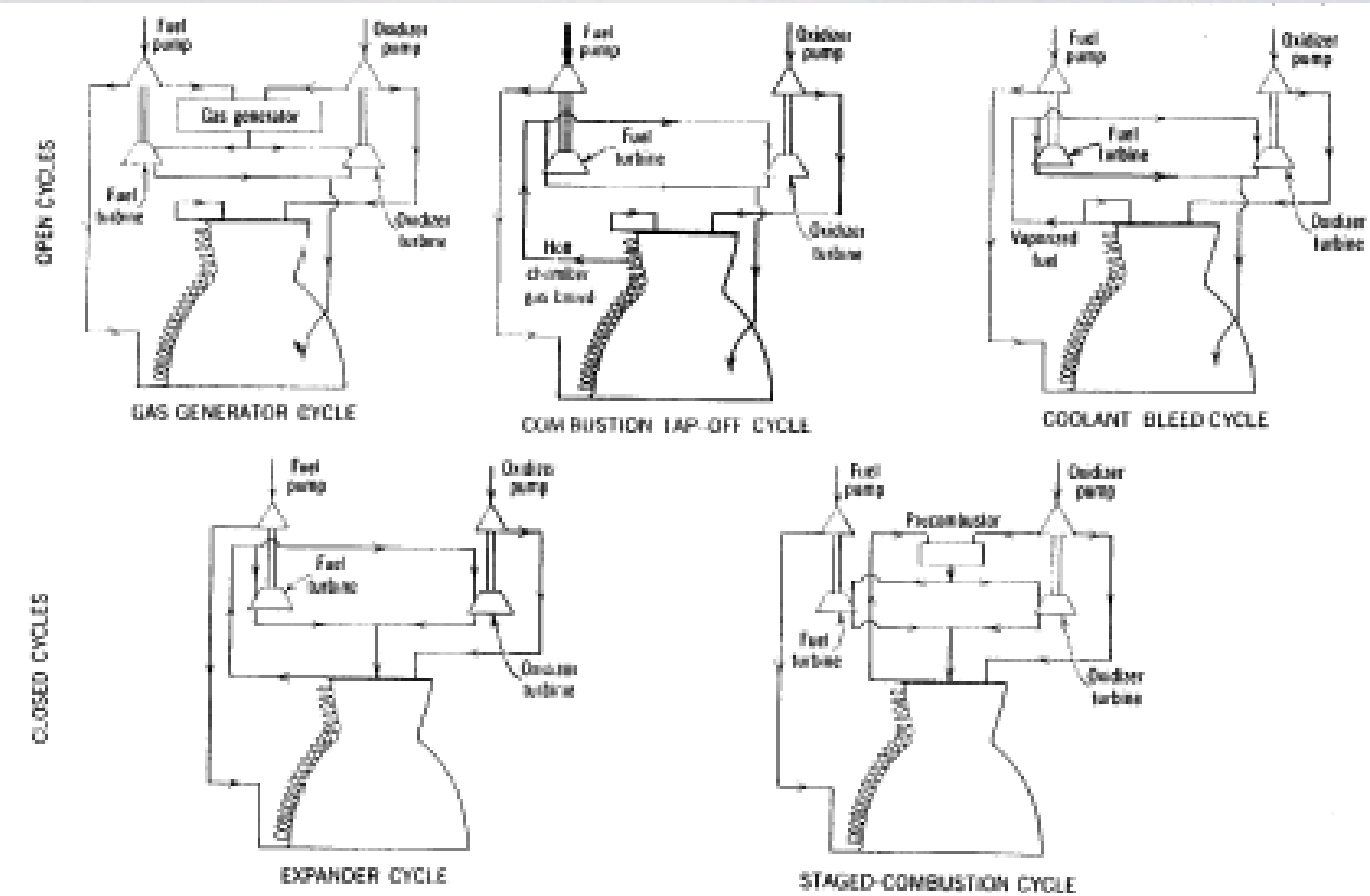
Sample Pump-fed Engine Cycles



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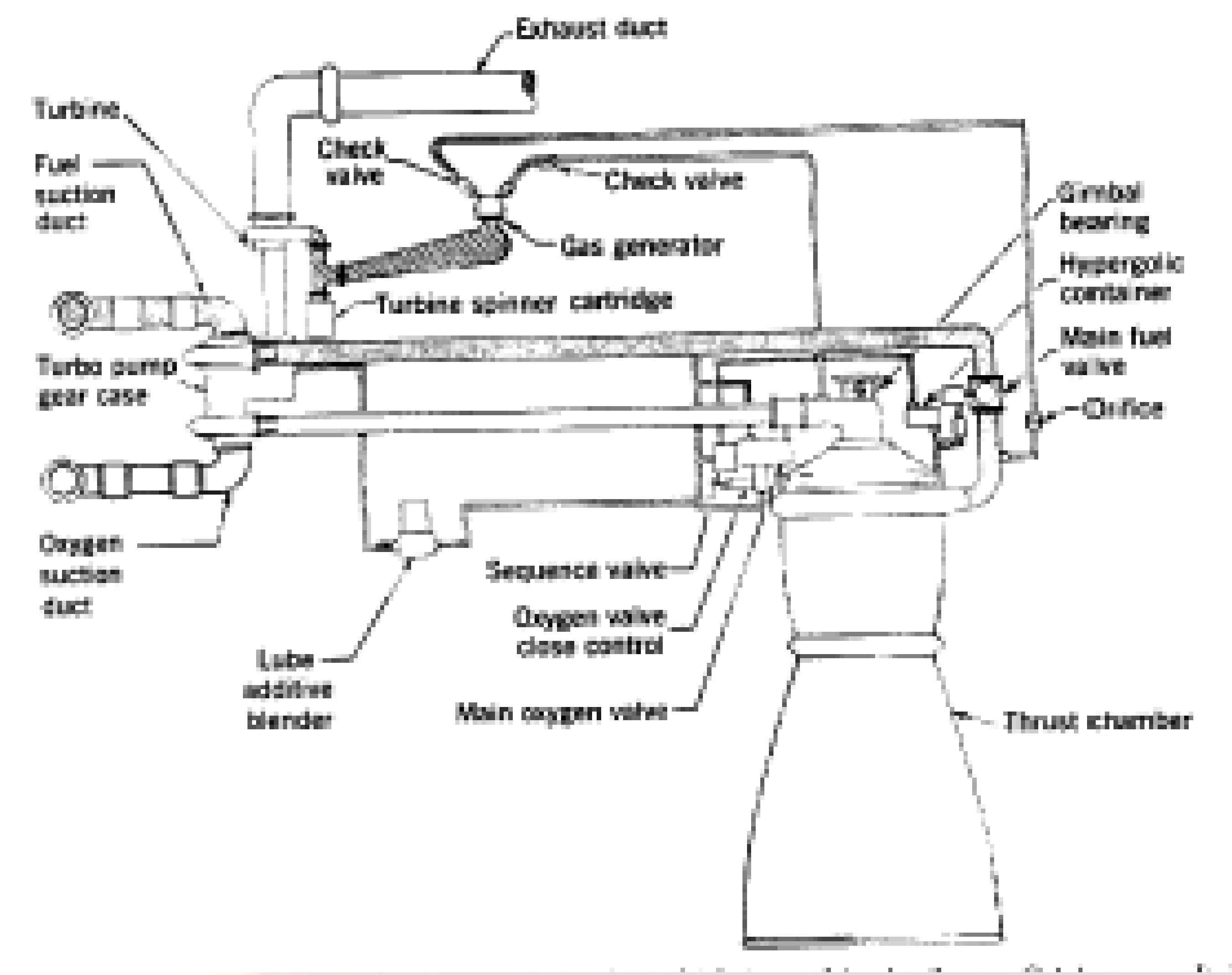
Gas Generator Cycle Engine



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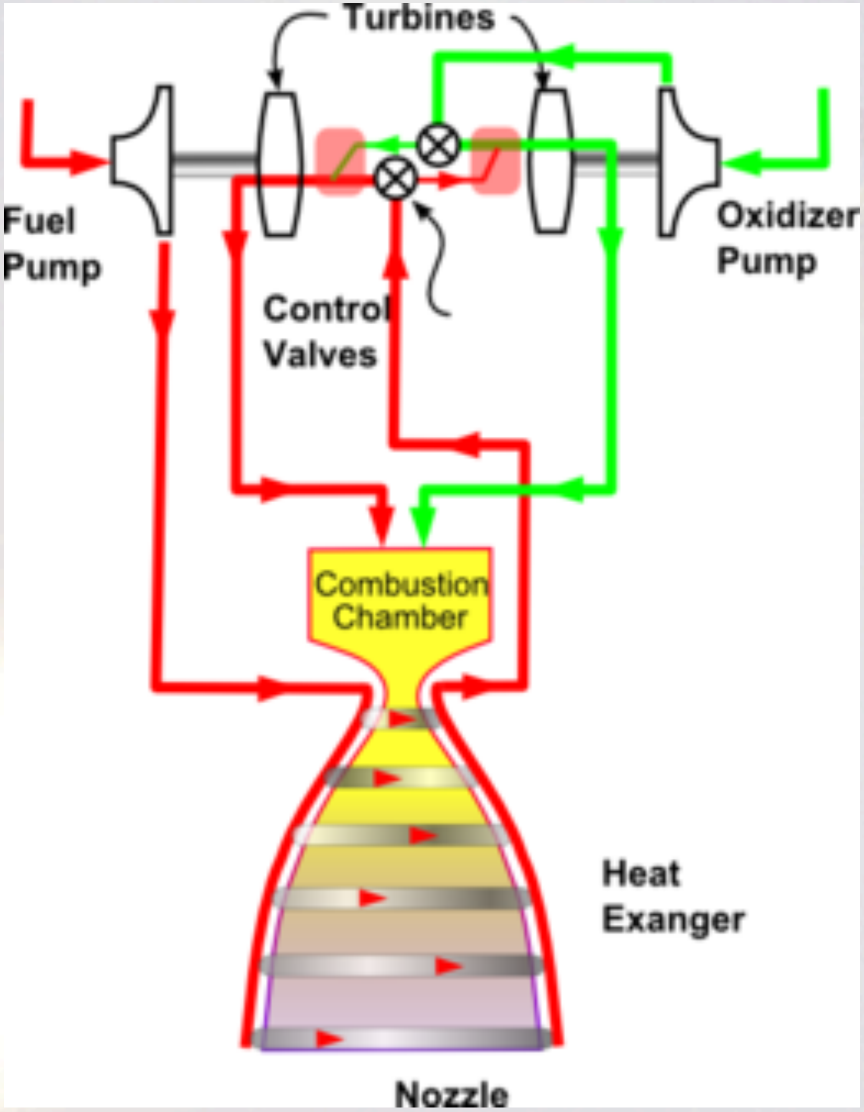
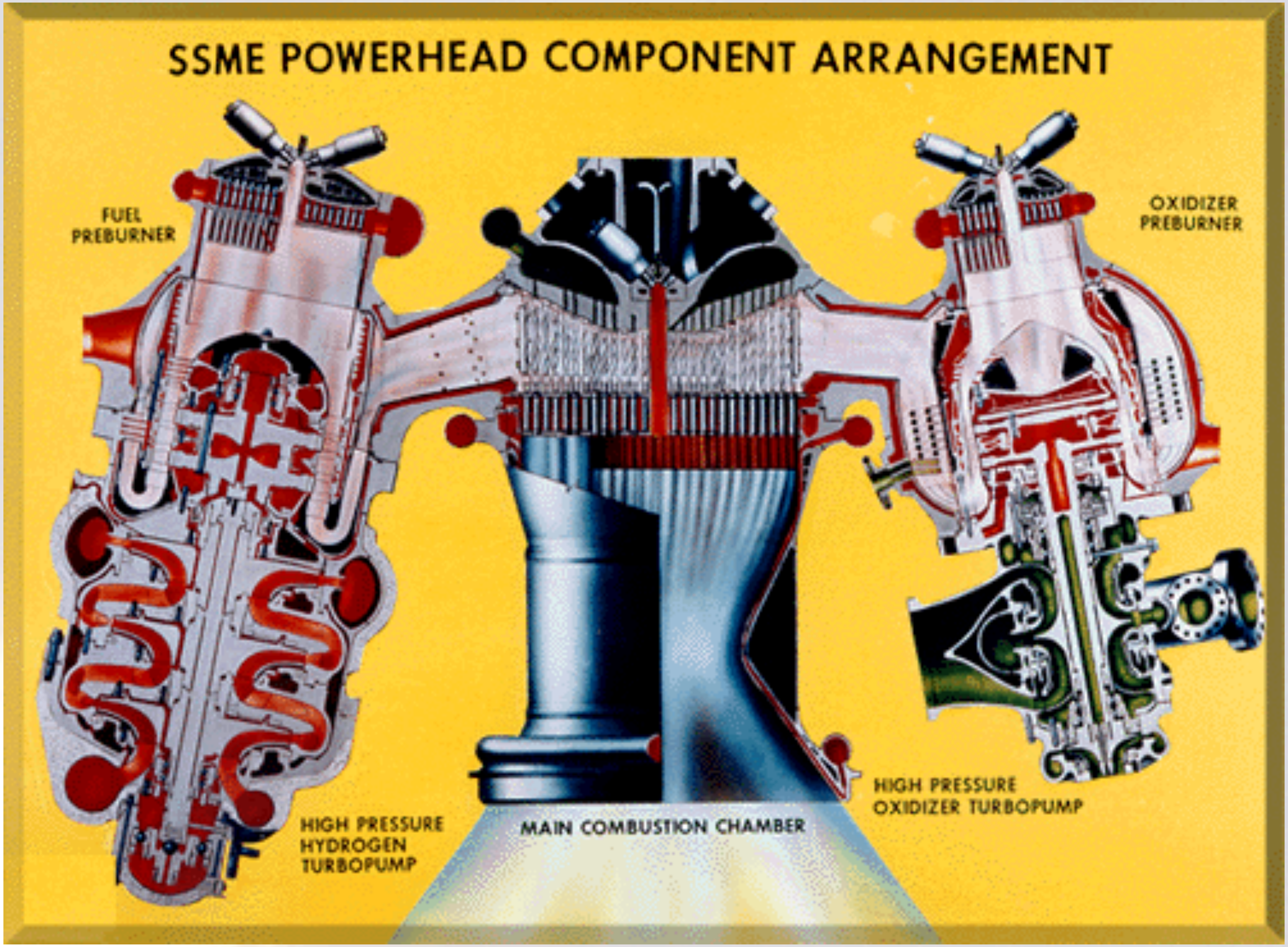
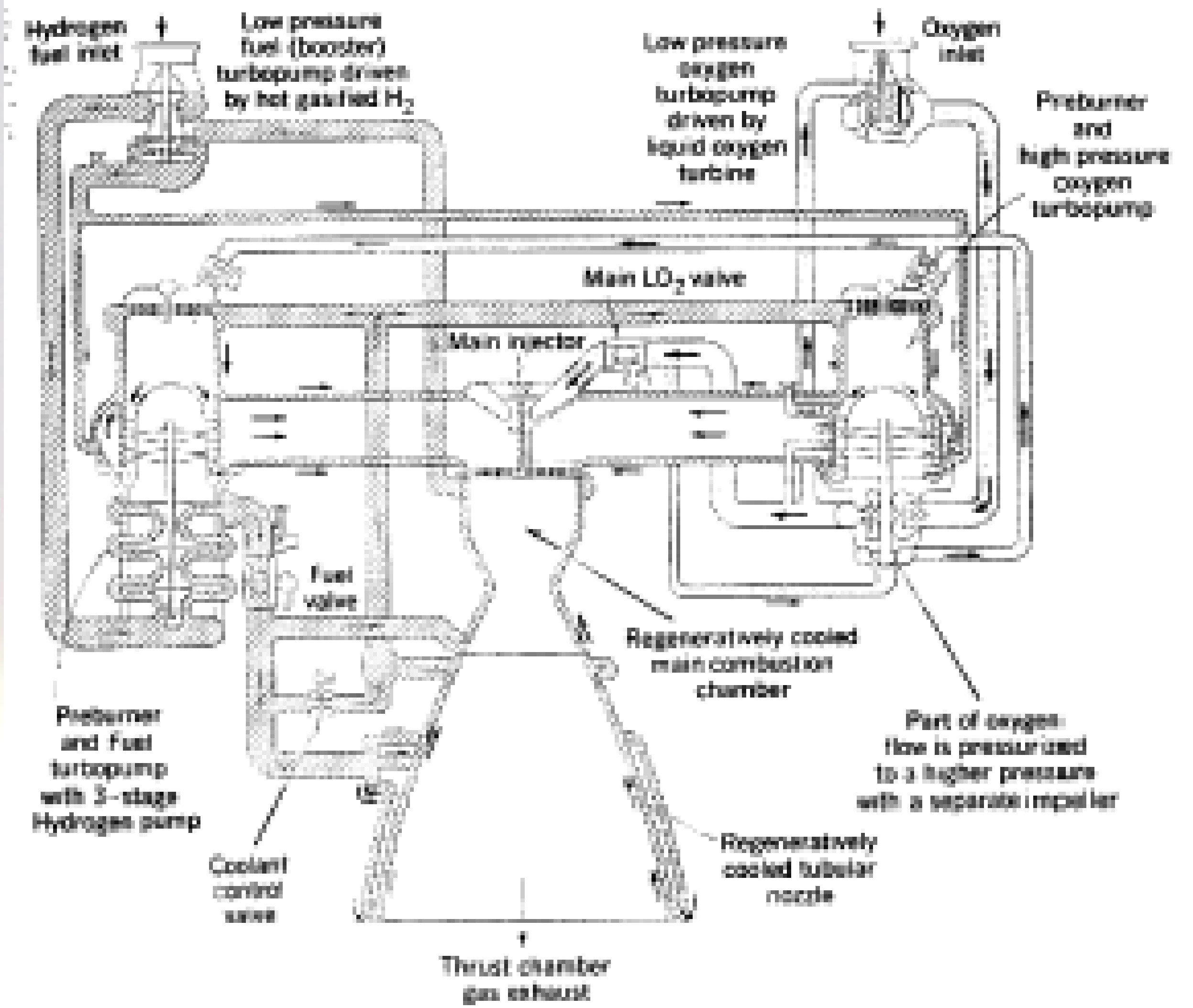
SSME Schematic



SSME Powerhead Configuration



SSME Engine Cycle



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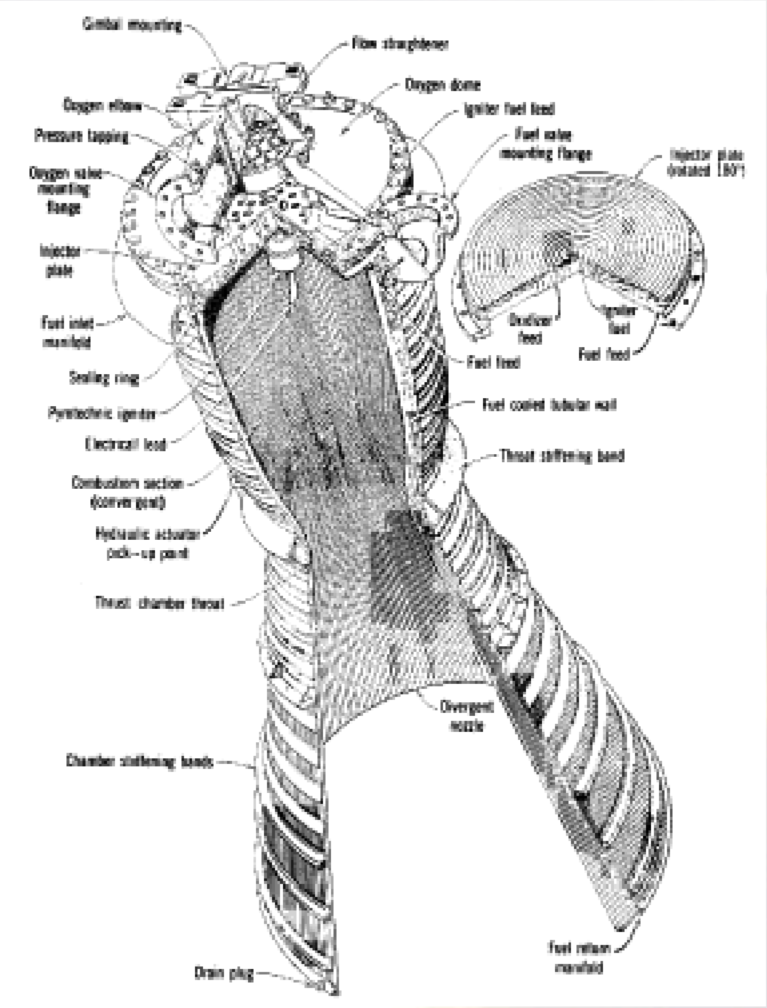
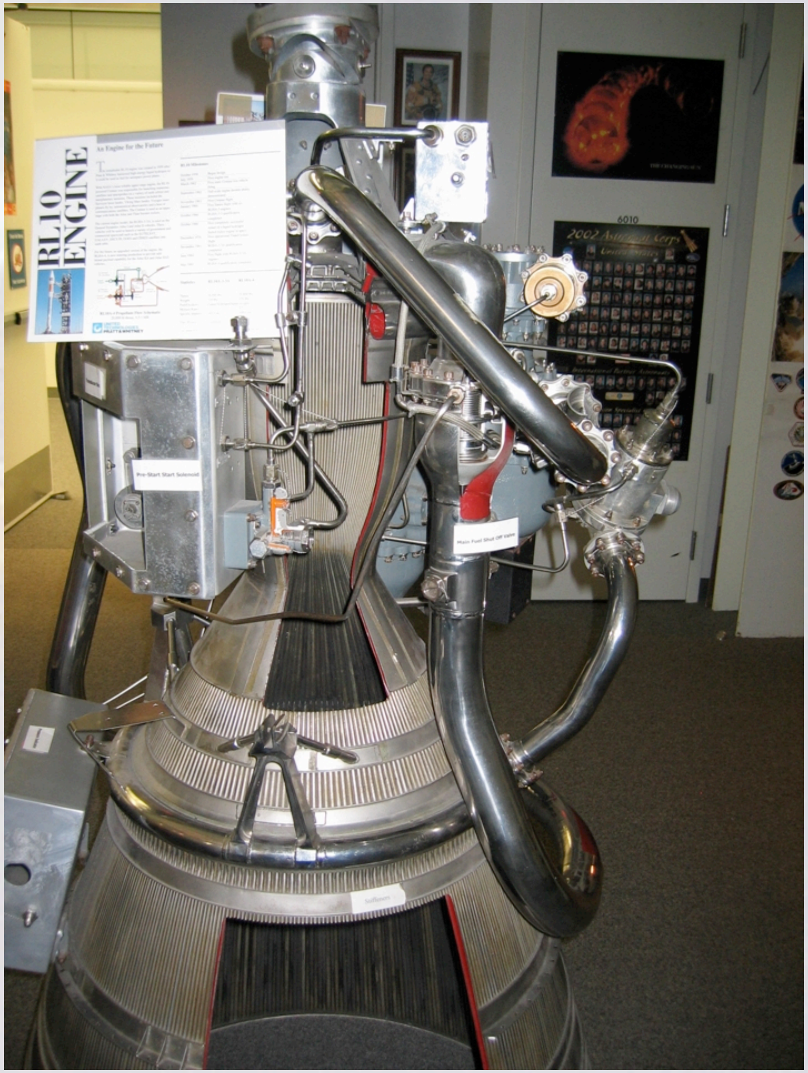
Liquid Rocket Engine Cutaway





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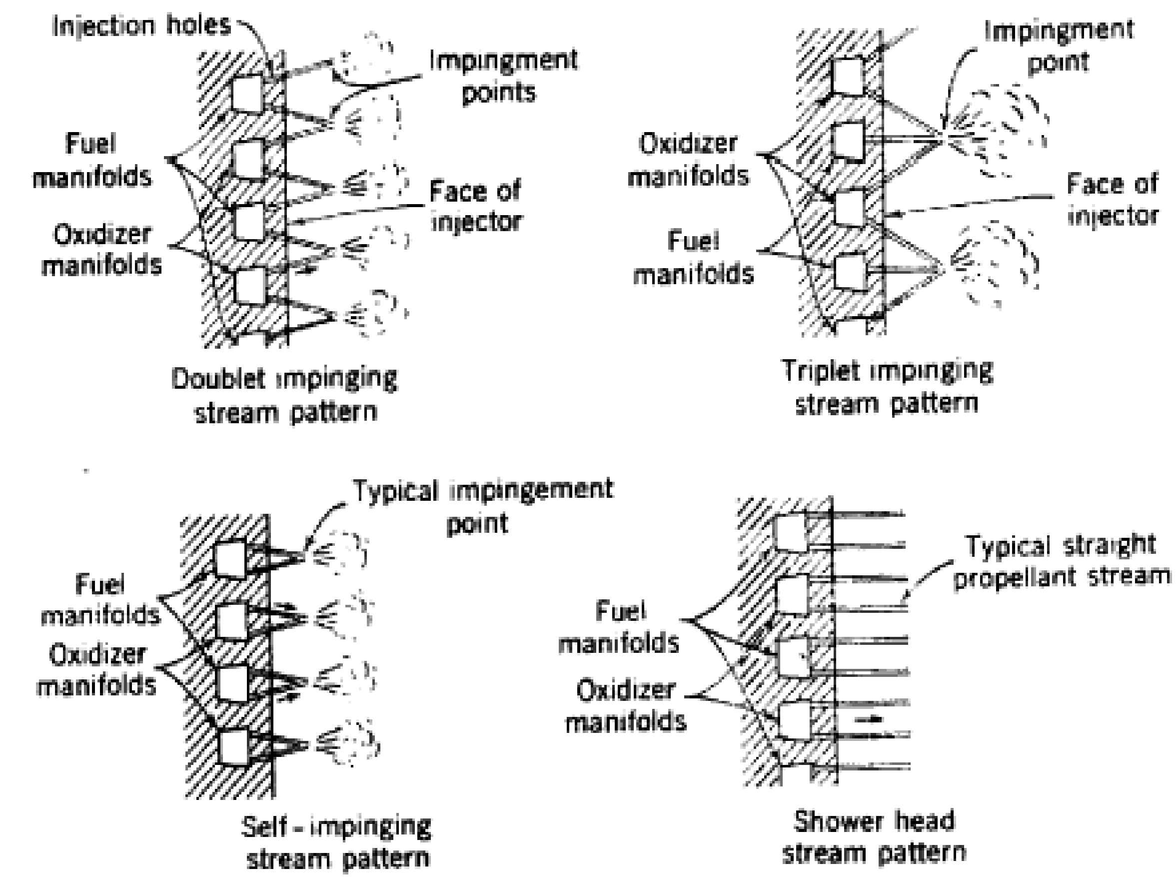
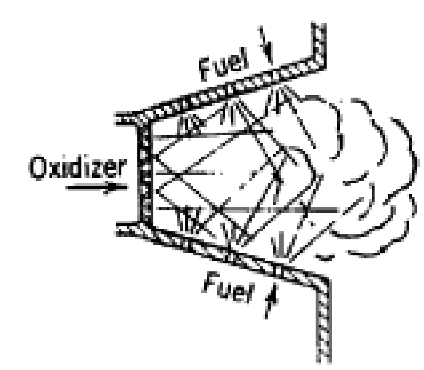
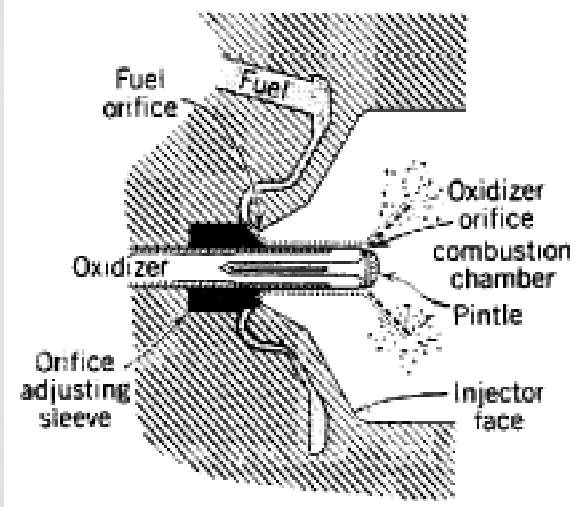
H-1 Engine Injector Plate

Injector Concepts





Non-impinging stream pattern



Variable injection area concentric tube injector

From G. P. Sutton, Rocket Propulsion Elements (5th ed.)
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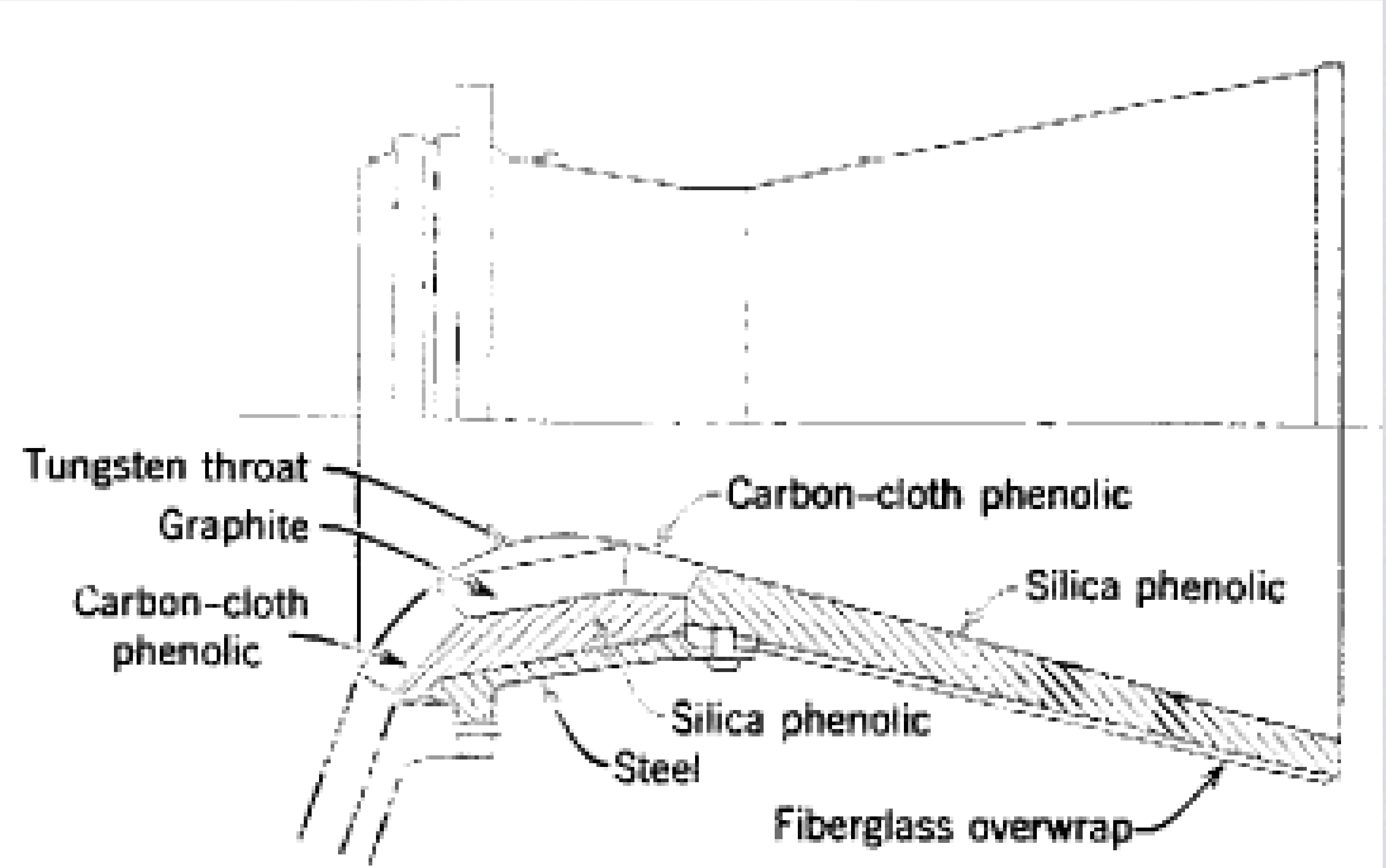
TR-201 Engine (LM Descent/Delta)

Solid Rocket Nozzle (Heat-Sink)



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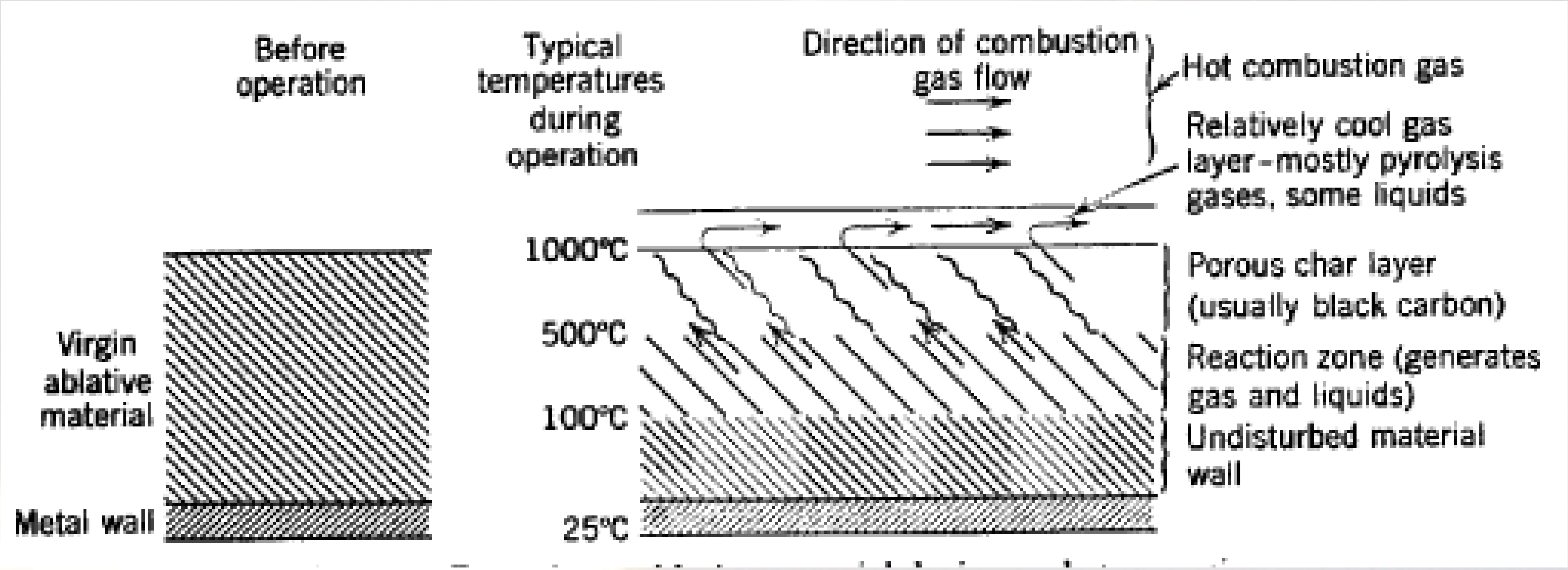
Ablative Nozzle Schematic



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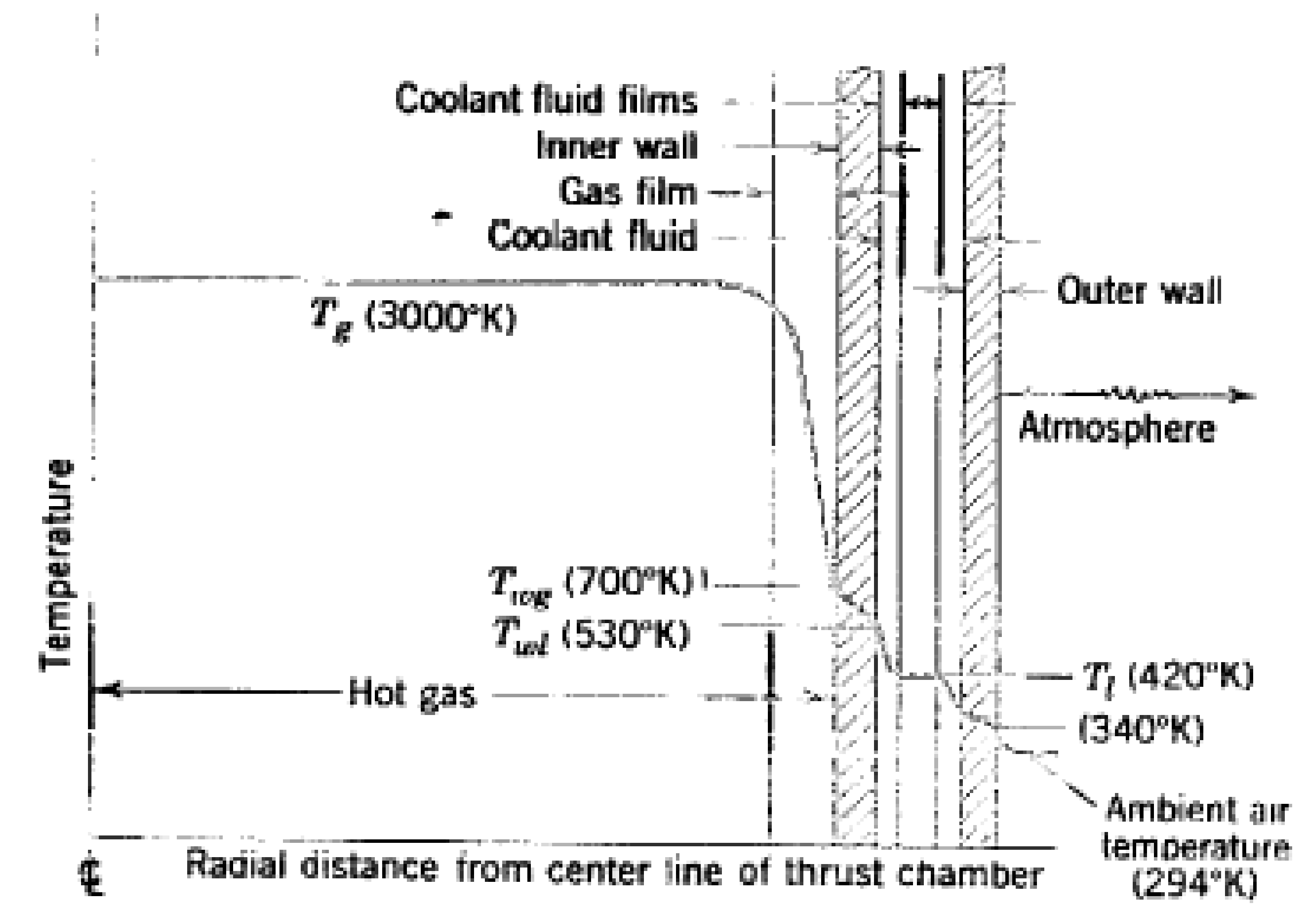
Active Chamber Cooling Schematic



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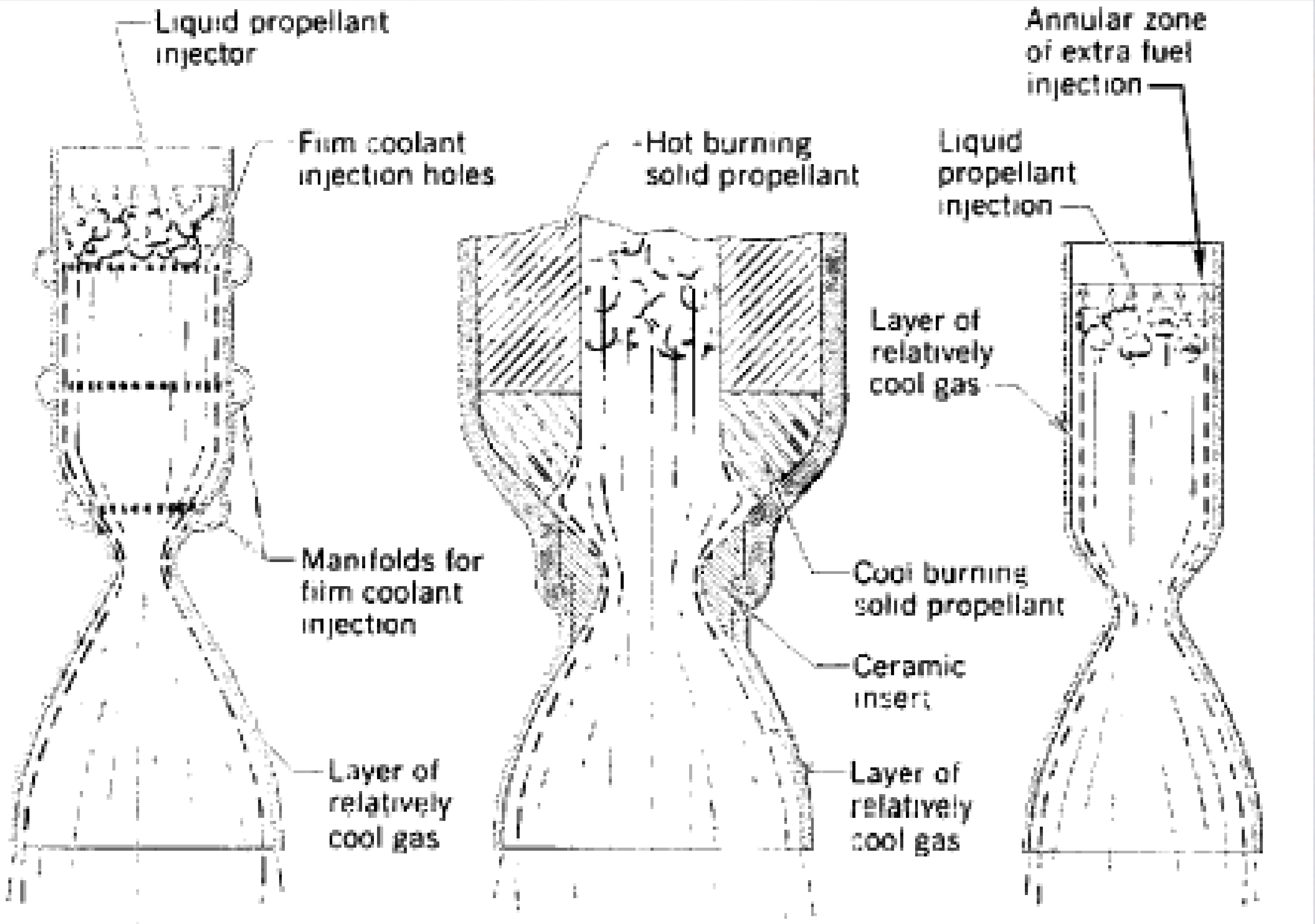
Boundary Layer Cooling Approaches



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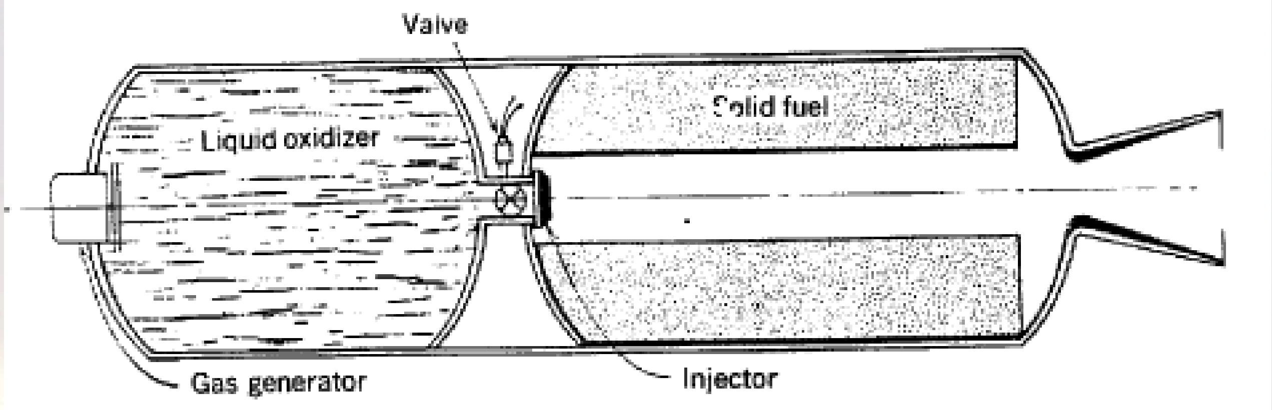
Hybrid Rocket Schematic



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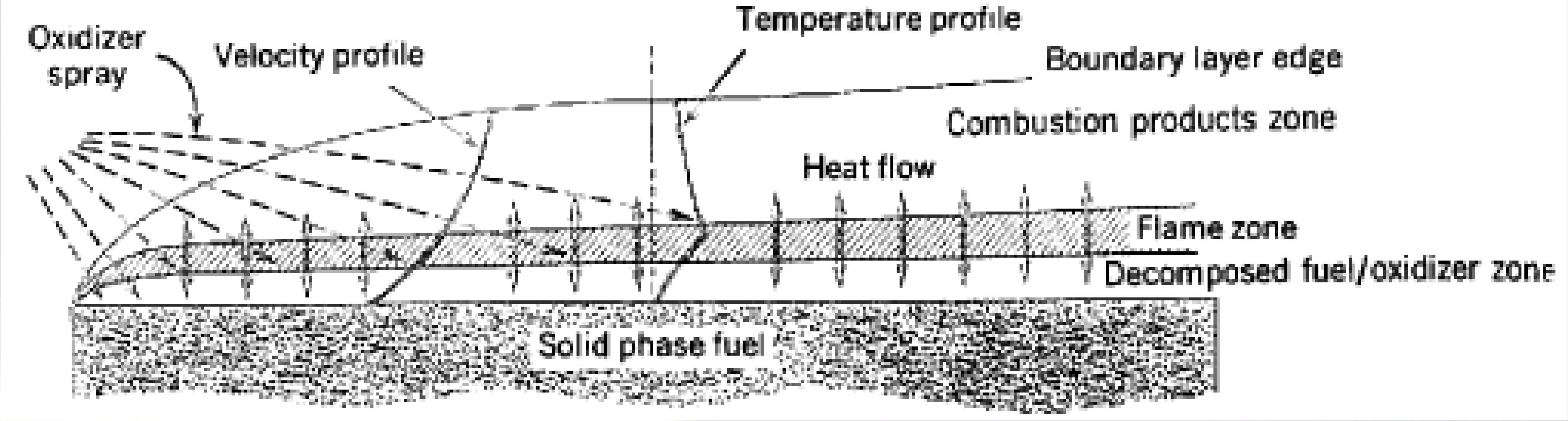
Hybrid Rocket Combustion



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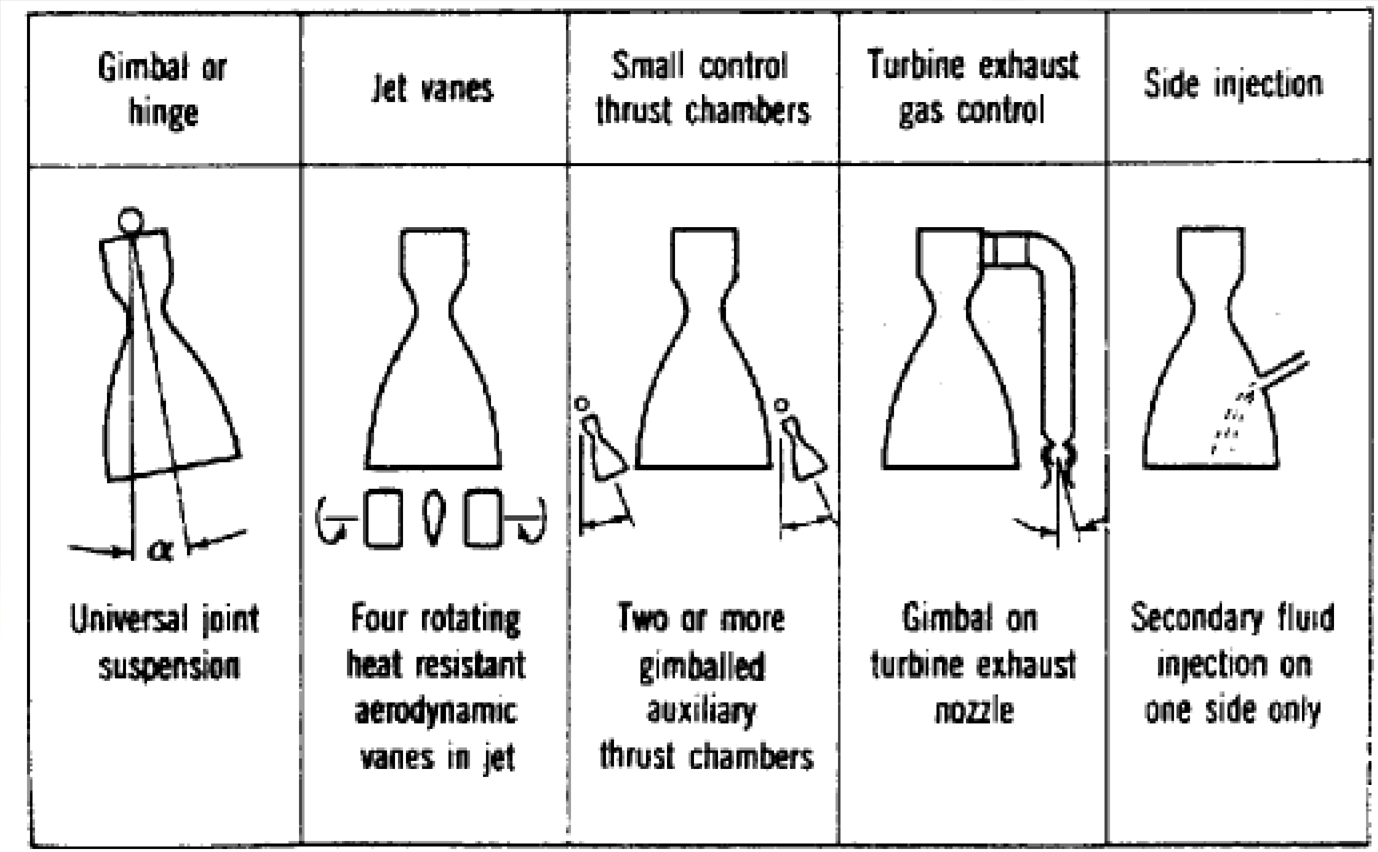
Thrust Vector Control Approaches



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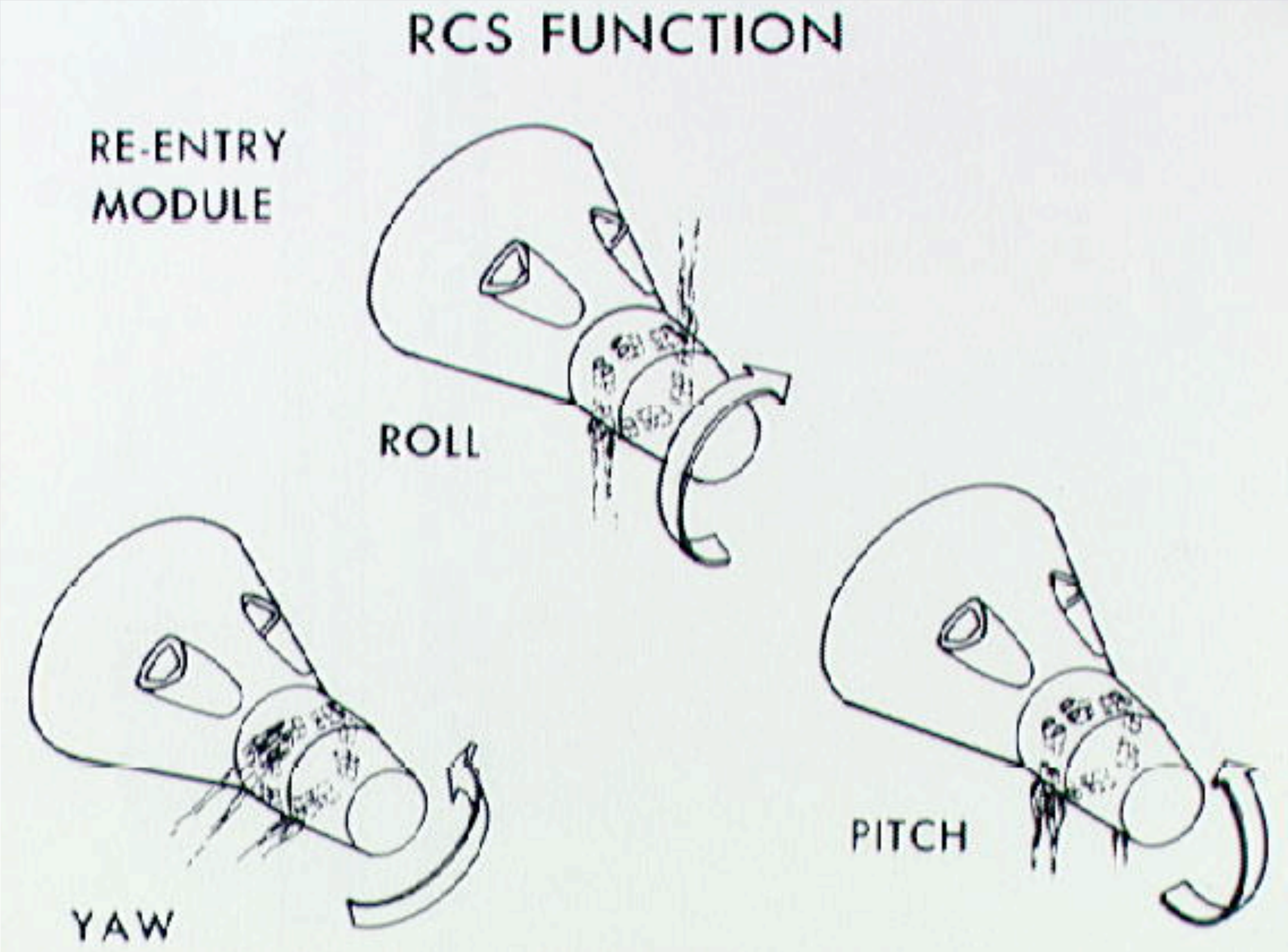
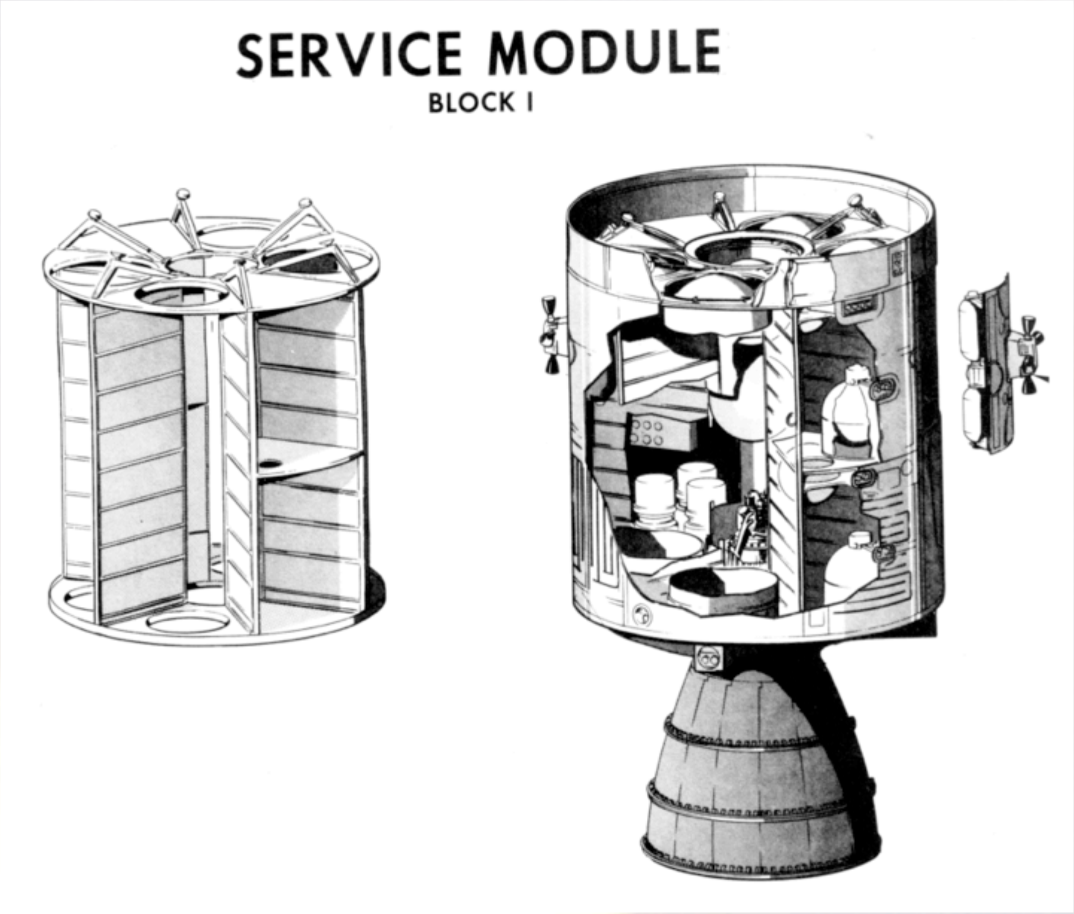
Gemini Entry Reaction Control System

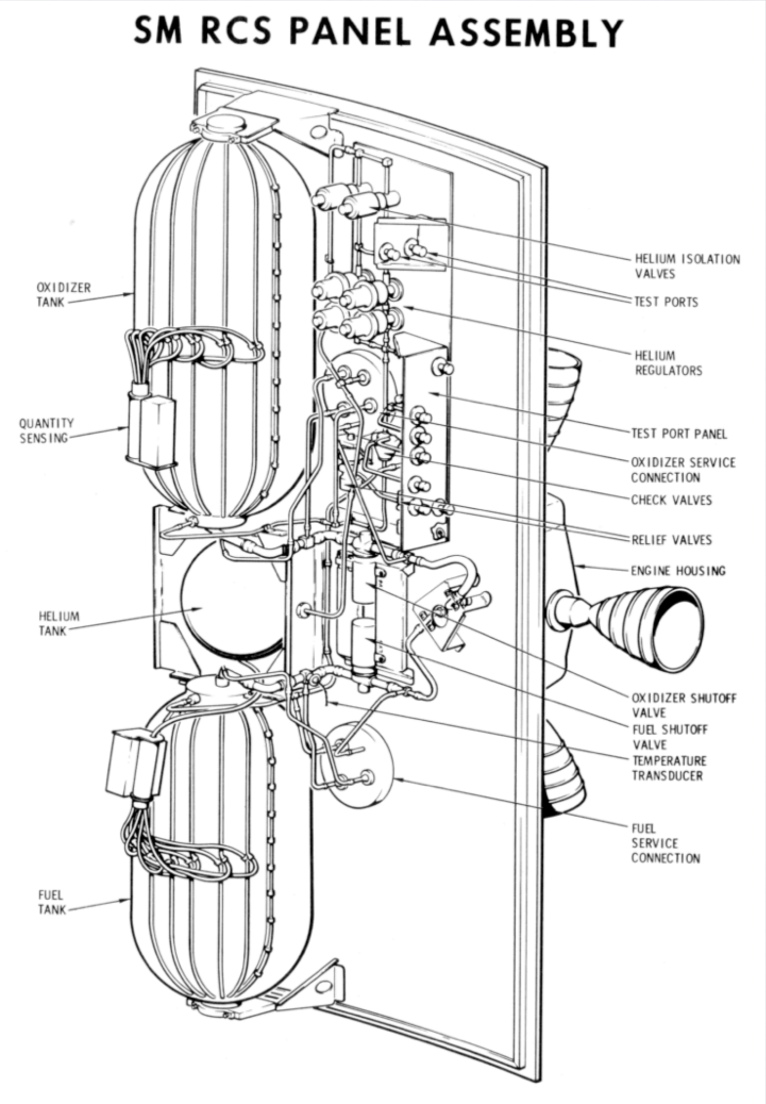


Apollo Reaction Control System Thrusters

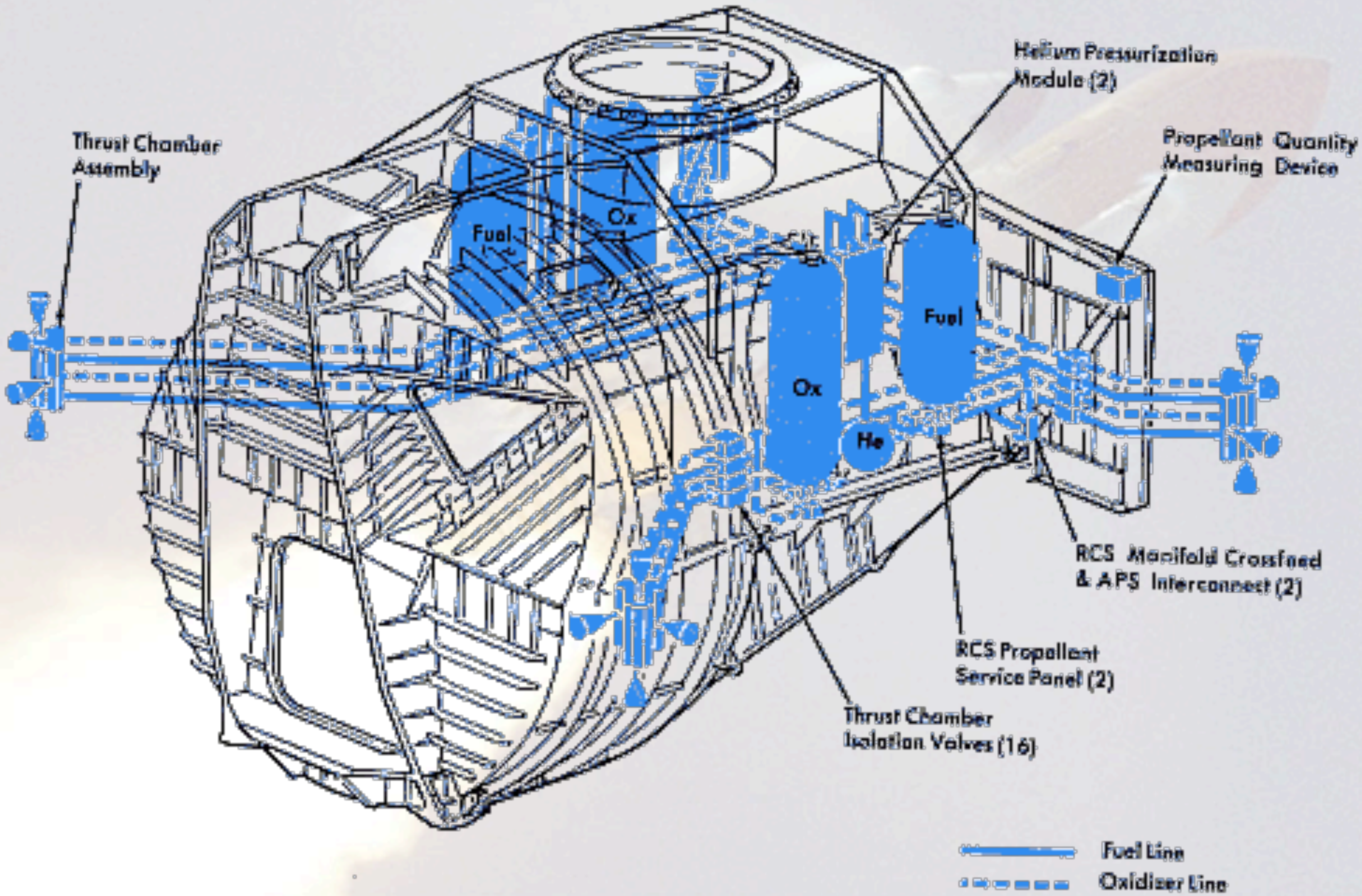
RCS Quad

Apollo CSM RCS Assembly

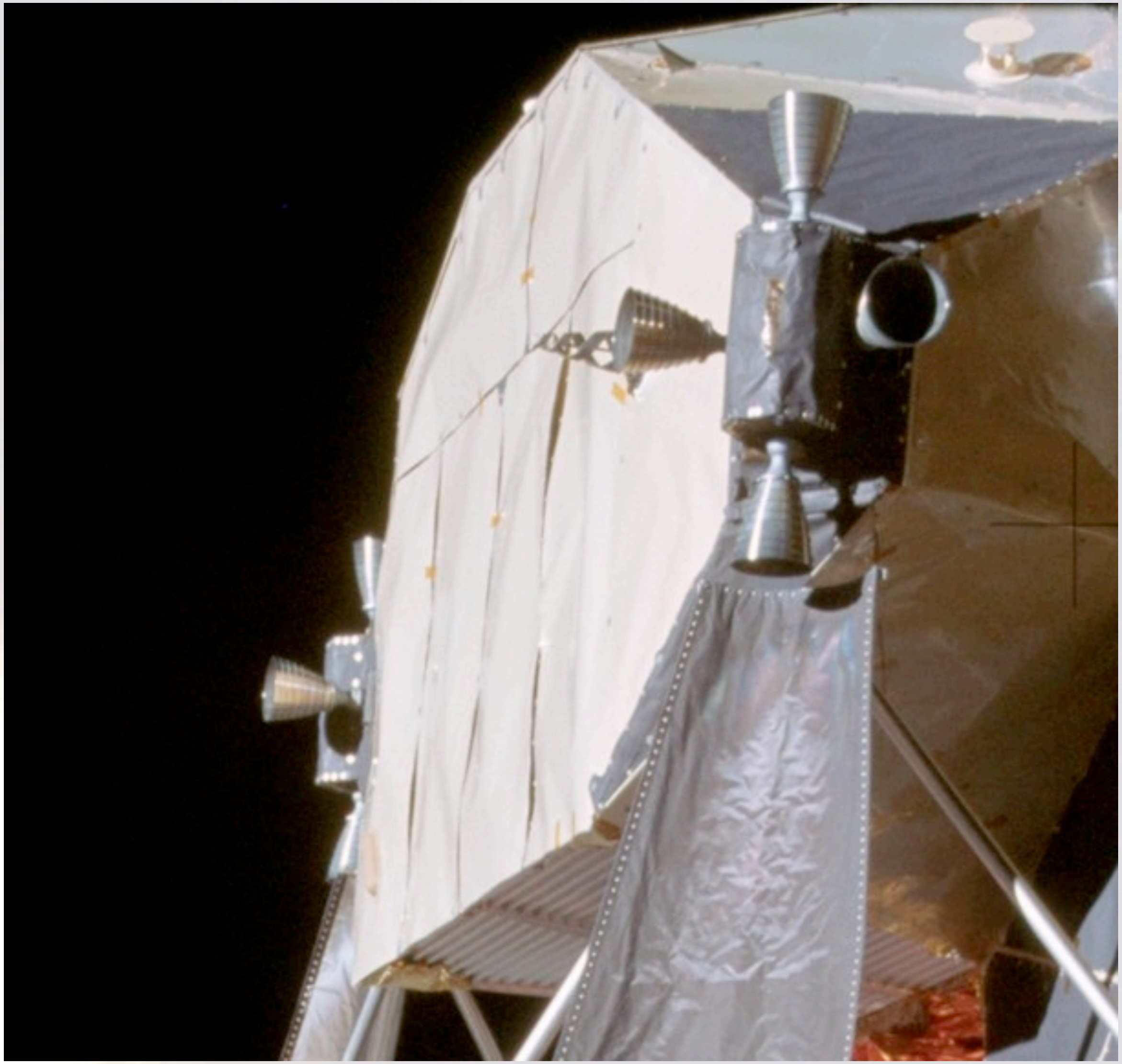




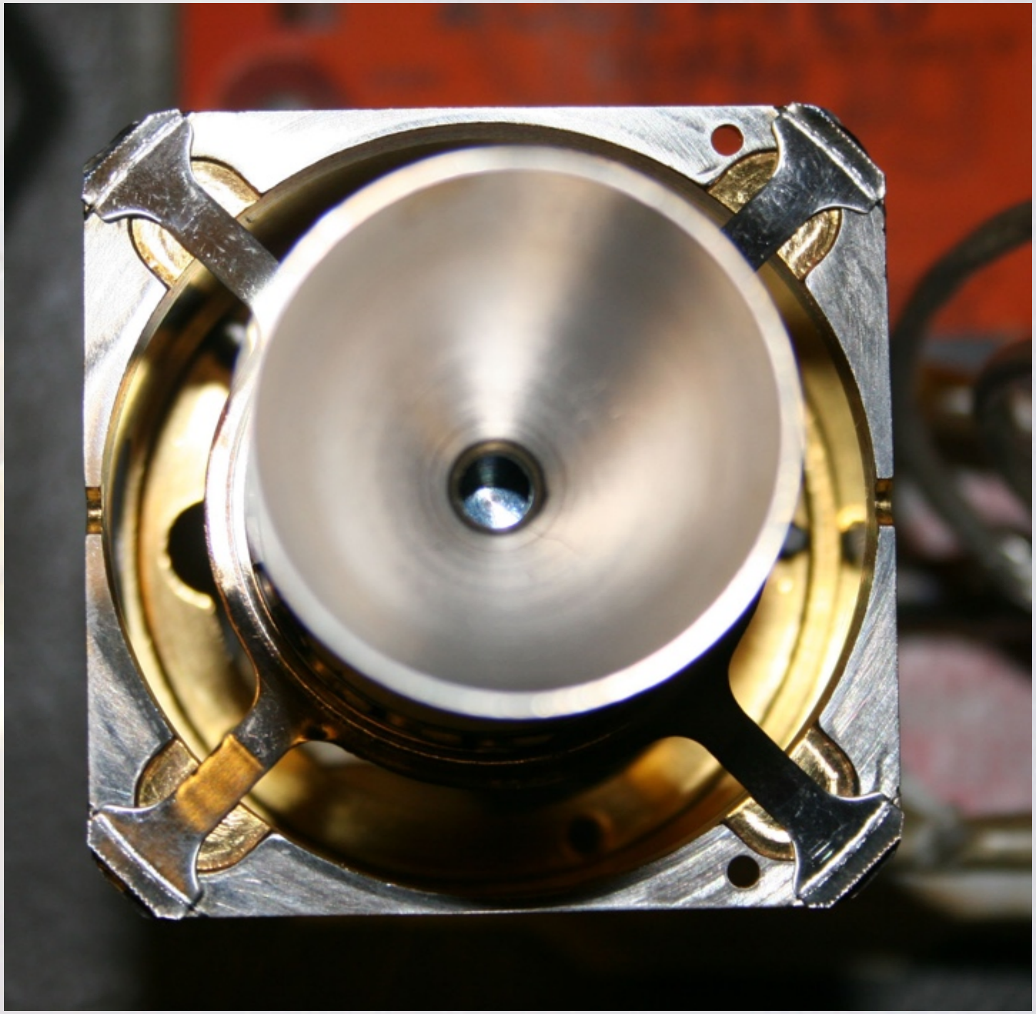
Lunar Module Reaction Control System



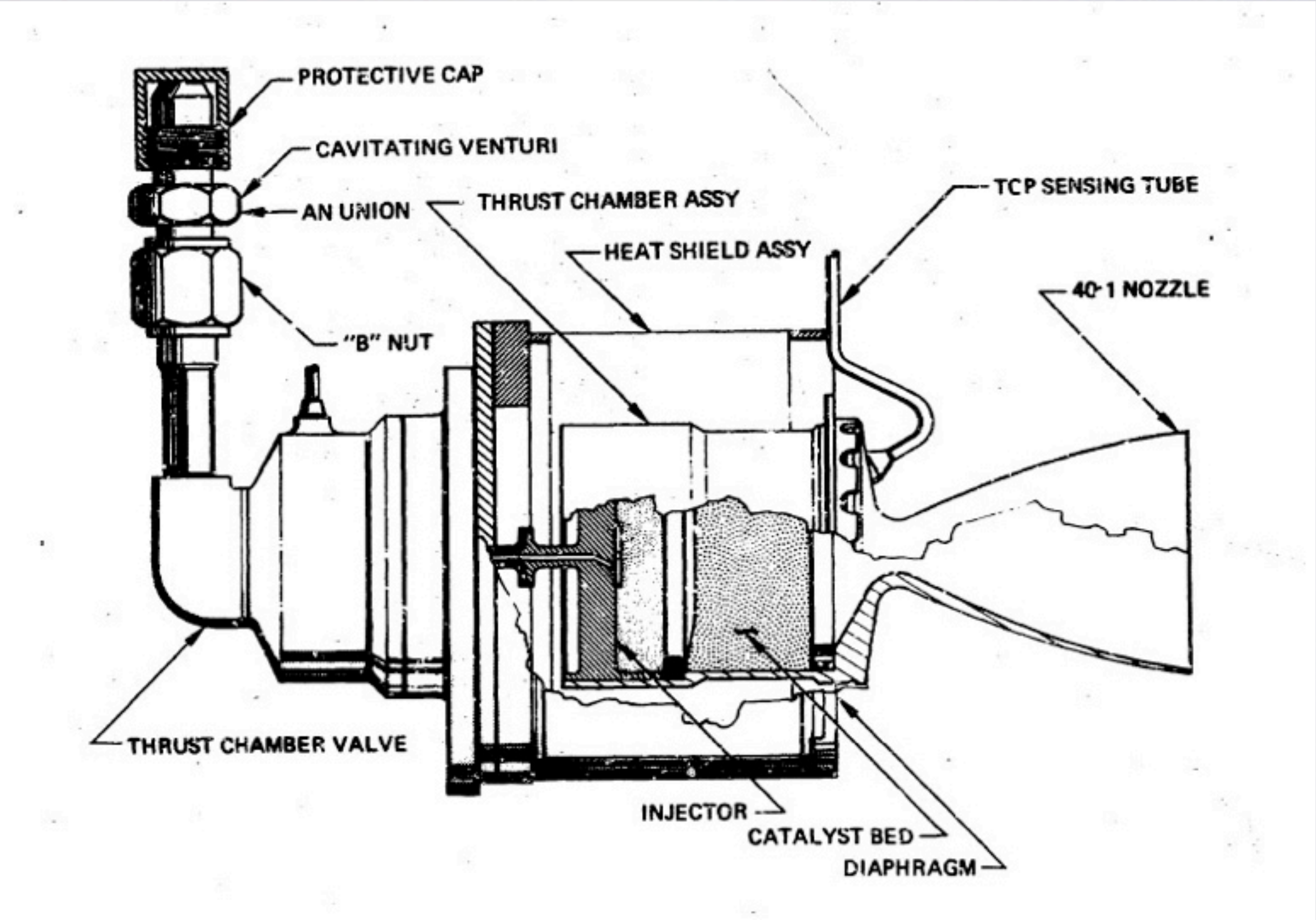
LM RCS Quad



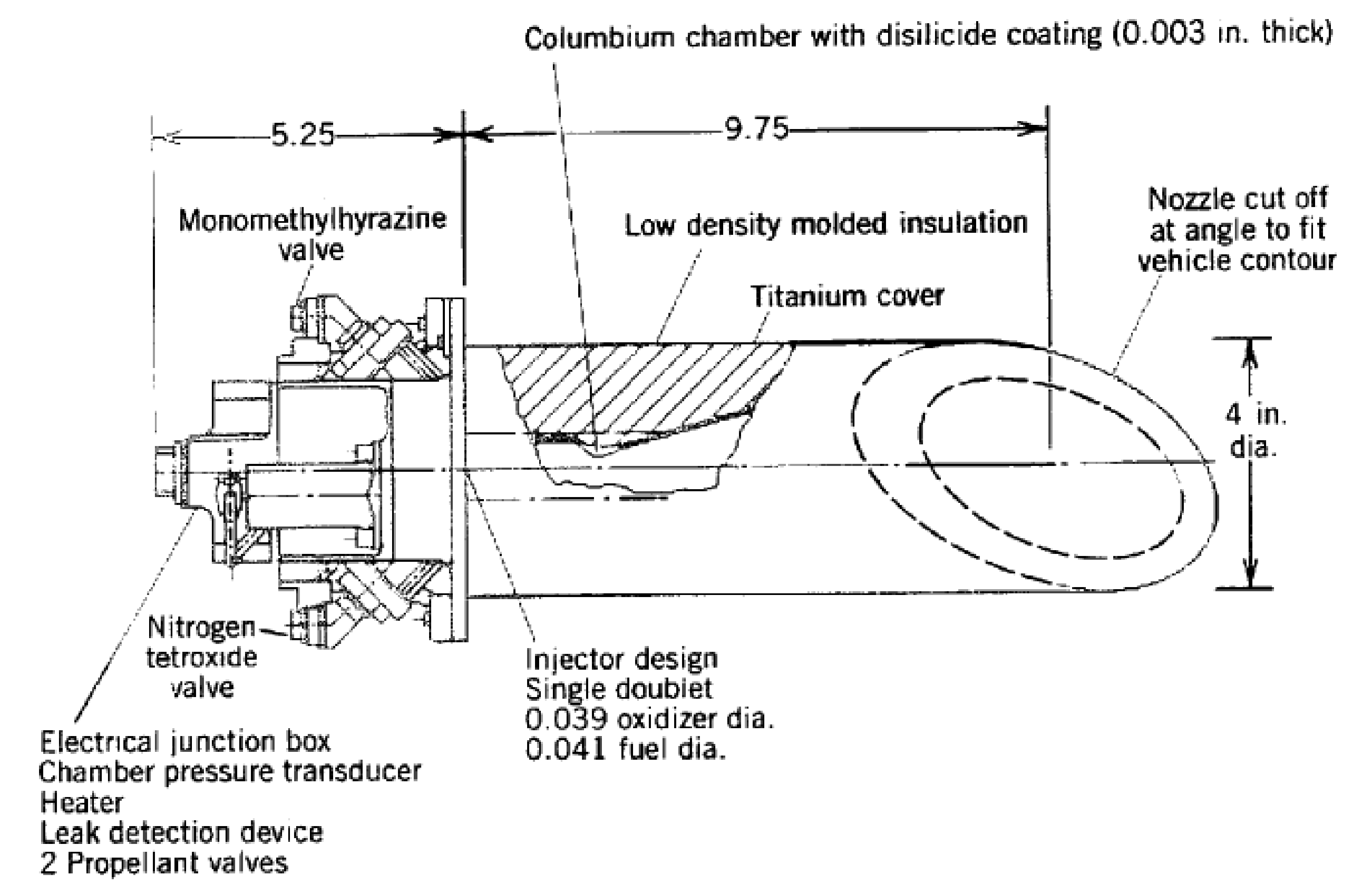
Viking Aeroshell RCS Thruster



Viking RCS Thruster Schematic



Space Shuttle Primary RCS Engine



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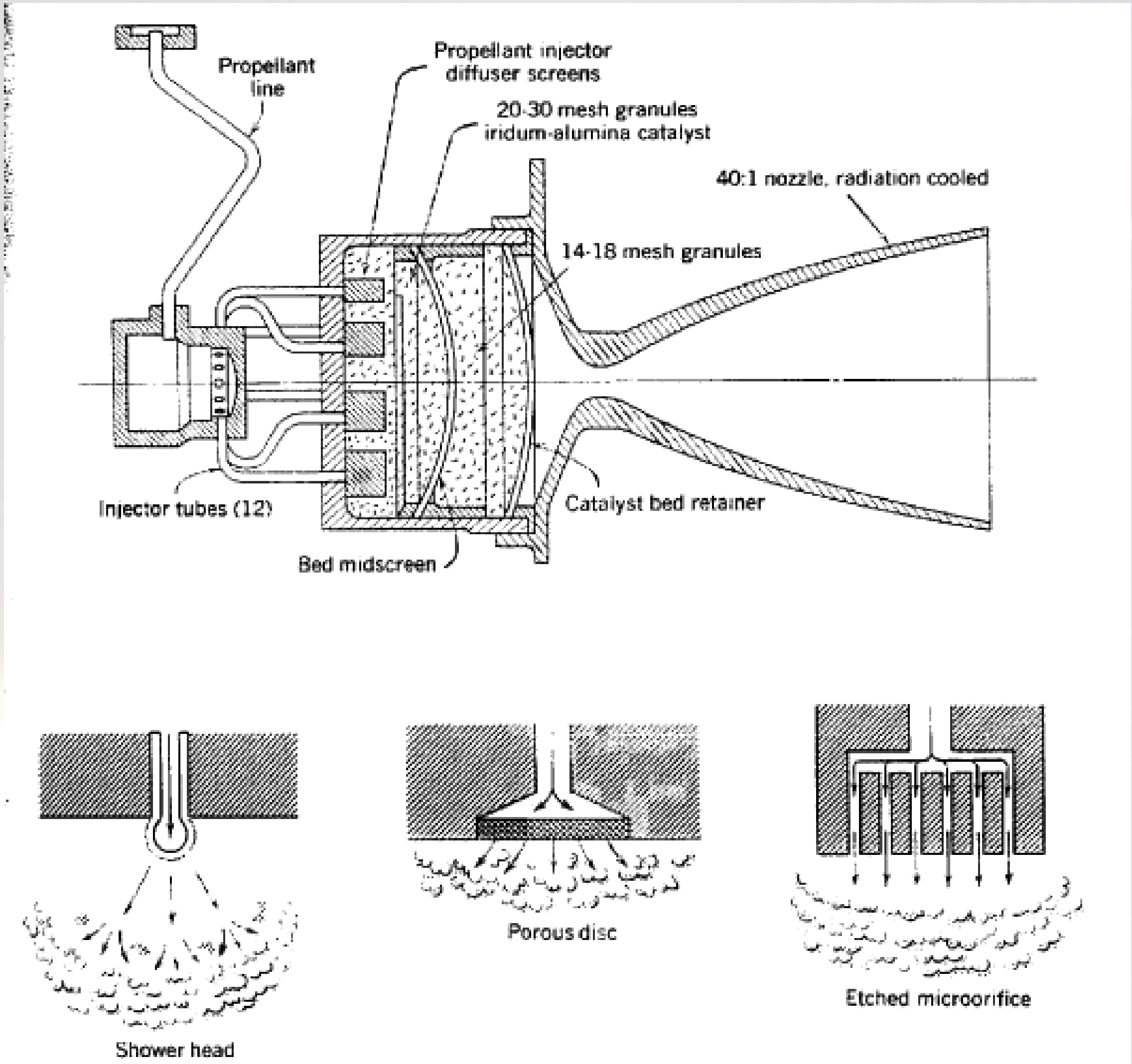
Monopropellant Engine Design



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Cold-gas Propellant Performance

Propellant	Molecular Mass	Density[a] (lb/ft^3)	Theoretical Specific Impulse (sec)
Hydrogen	2.0	1.21	296
Helium	4.0	2.37	179
Methane	16.0	12.10	114
Nitrogen	28.0	17.37	80
Air	28.9	19.3	74
Argon	39.9	27.60	57
Krypton	83.8	67.20	39
Freon 14	88.0	60.01	55
Carbon dioxide	44.0	Liquid	67

[a] At 3500 psia and 0°C.

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

Total Impulse

- Total impulse I_t is the total thrust-time product for the propulsion system, with units <N-sec>

$$I_t = Tt = \dot{m} v_e t$$

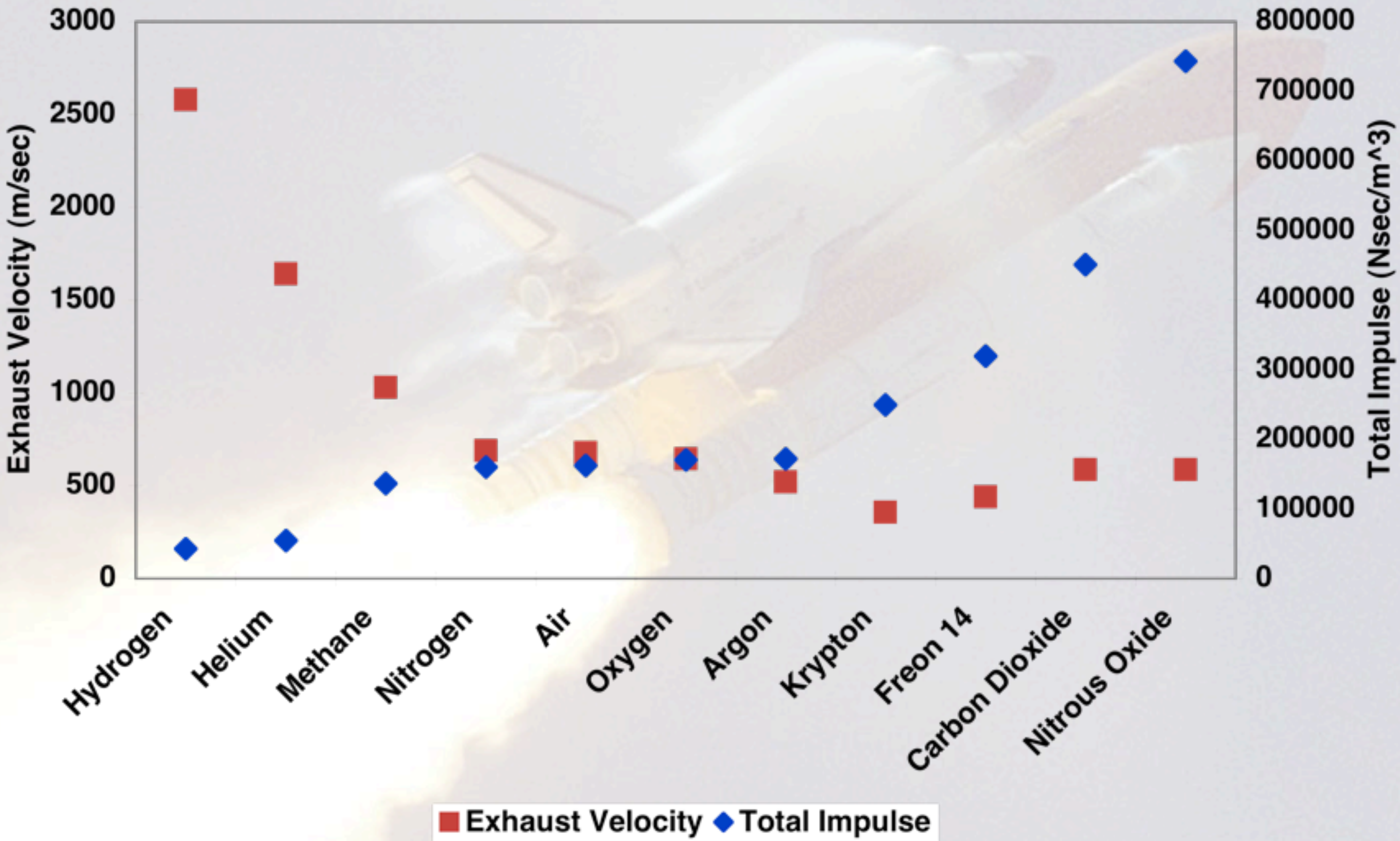
$$t = \frac{\rho V}{\dot{m}}$$

$$I_t = \rho V v_e$$

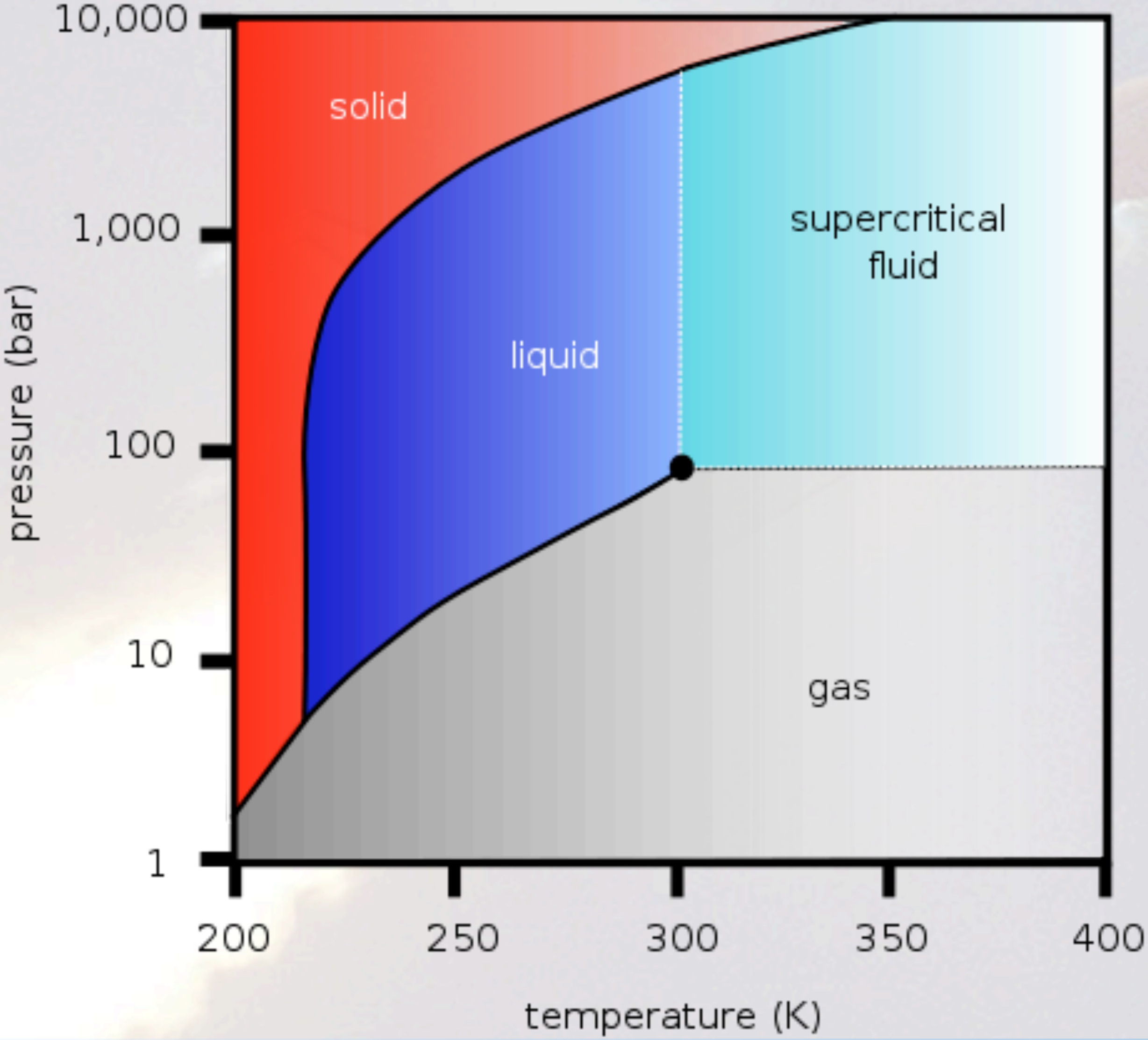
- To assess cold-gas systems, we can examine total impulse per unit volume of propellant storage

$$\frac{I_t}{V} = \rho v_e$$

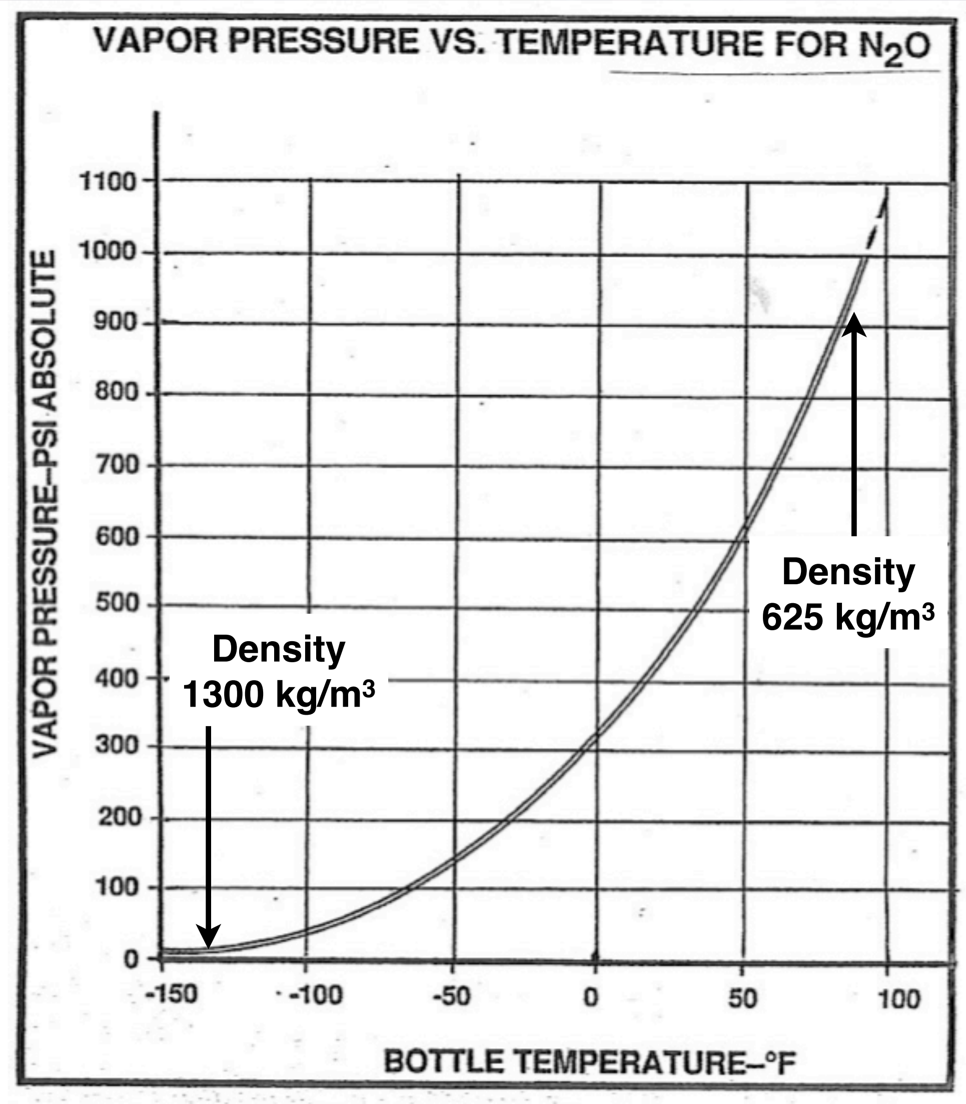
Performance of Cold-Gas Systems



Self-Pressurizing Propellants (CO_2)



Self-Pressurizing Propellants (N_2O)

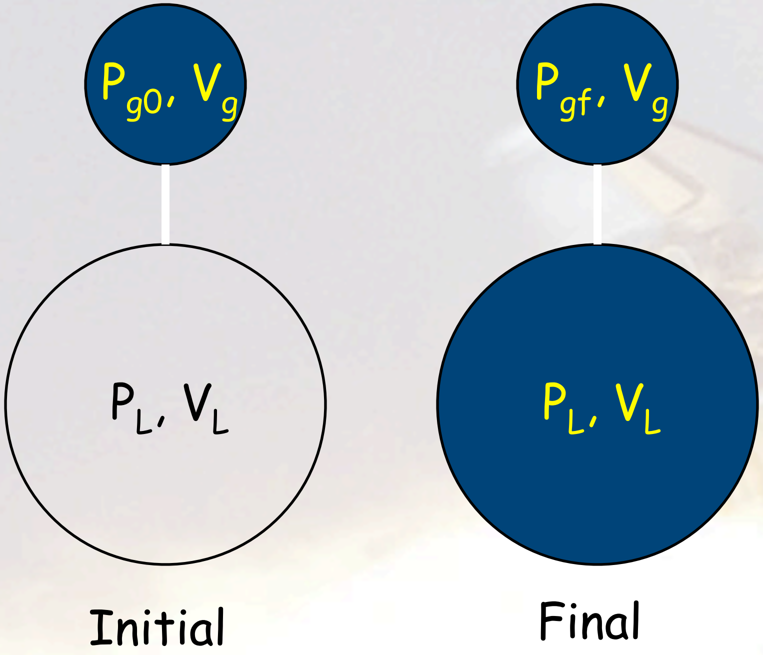


N_2O Performance Augmentation

- Nominal cold-gas exhaust velocity ~600 m/sec
- N_2O dissociates in the presence of a heated catalyst
  engine temperature ~1300°C $2N_2O \longrightarrow 2N_2 + O_2$
  exhaust velocity ~1800 m/sec
- NOFB (Nitrous Oxide Fuel Blend) - store premixed N_2O/hydrocarbon mixture
  exhaust velocity >3000 m/sec

Pressurization System Analysis

Adiabatic Expansion of Pressurizing Gas



$$p_{g,0}V_g^{\gamma} = p_{g,f}V_g^{\gamma} + p_l V_l^{\gamma}$$

Known quantities:

$P_{g,0}$=Initial gas pressure

$P_{g,f}$=Final gas pressure

P_L=Operating pressure of propellant tank(s)

V_L=Volume of propellant tank(s)

Solve for gas volume V_g

Boost Module Propellant Tanks

- Gross mass 23,000 kg
  - Inert mass 2300 kg
  - Propellant mass 20,700 kg
  - Mixture ratio N_2O_4/A50 = 1.8 (by mass)
- N_2O_4 tank
  - Mass = 13,310 kg
  - Density = 1450 kg/m^3
  - Volume = 9.177 m^3 --> r_{sphere}=1.299 m
- Aerozine 50 tank
  - Mass = 7390 kg
  - Density = 900 kg/m^3
  - Volume = 8.214 m^3 --> r_{sphere}=1.252 m

Boost Module Main Propulsion

- Total propellant volume $V_L = 17.39\ m^3$
- Assume engine pressure $p_0 = 250$ psi
- Tank pressure $p_L = 1.25*p_0 = 312$ psi
- Final GHe pressure $p_{g,f} = 75\ psi + p_L = 388$ psi
- Initial GHe pressure $p_{g,0} = 4500$ psi
- Conversion factor 1 psi = 6892 Pa
- Ratio of specific heats for He = 1.67

$$(4500\ psi)V_g^{1.67} = (388\ psi)V_g^{1.67} + (312\ psi)(17.39\ m^3)^{1.67}$$

- $V_g = 3.713\ m^3$
- Ideal gas: T=300°K --> $\rho_{He} = \frac{p_{g,0}\overline{M}}{\Re T_0}$

$\rho = 49.7\ kg/m^3$ (4500 psi = 31.04 MPa) $M_{He} = 185.1$ kg