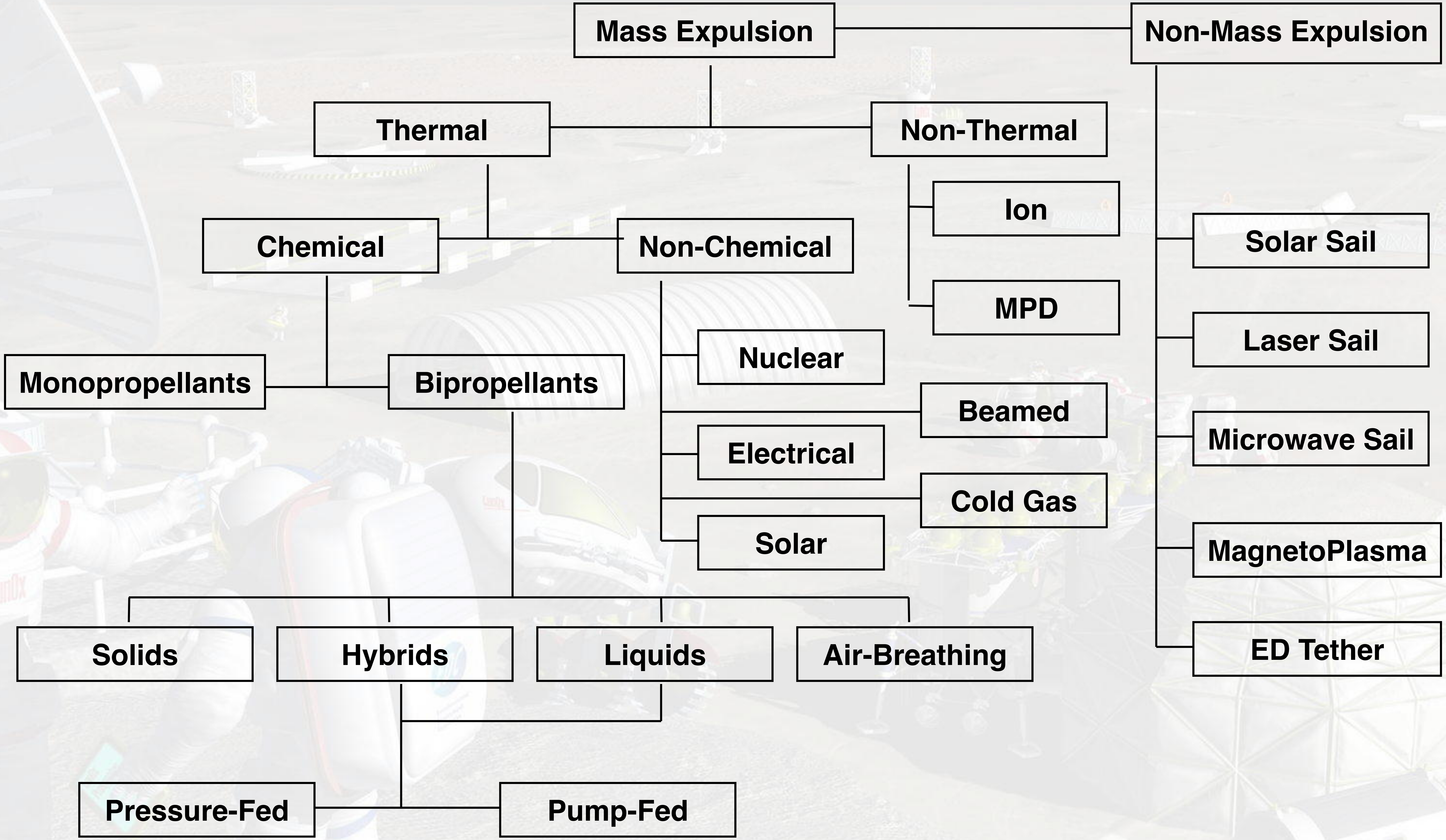
Propulsion Systems Design

- Lecture #20 - November 2, 2023
- Rocket engine basics
- Survey of the technologies
- Propellant feed systems
- Propulsion systems design

Propulsion Taxonomy



Thermal Rocket Exhaust Velocity

- Exhaust velocity is

$$v_e = \sqrt{\frac{2\gamma}{\gamma - 1}\frac{\Re T_o}{\bar{M}}\left[1 - \left(\frac{p_e}{p_o}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

where

$$\bar{M} \equiv \text{average molecular weight of exhaust}$$

$$\Re \equiv \text{universal gas constant} = 8.3143\ \frac{Joules}{mole^o K}$$

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

Ideal Thermal Rocket Exhaust Velocity

- Ideal exhaust velocity is

$$v_{e,ideal} = \sqrt{\frac{2\gamma}{\gamma - 1}\frac{\Re T_o}{\bar{M}}}$$

- 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

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

- Effective exhaust velocity

$$T = \dot{m} c \Longrightarrow c = v_e + (p_e - p_{amb}) \frac{A_e}{\dot{m}} \qquad \left(I_{sp} = \frac{c}{g_o} \right)$$

- Expansion ratio

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

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 vacuum V_e

$$\frac{v_{e1}}{v_{e2}} = \sqrt{\frac{p_o^{\frac{\gamma-1}{\gamma}} - p_{e1}^{\frac{\gamma-1}{\gamma}}}{p_o^{\frac{\gamma-1}{\gamma}} - p_{e2}^{\frac{\gamma-1}{\gamma}}}}$$

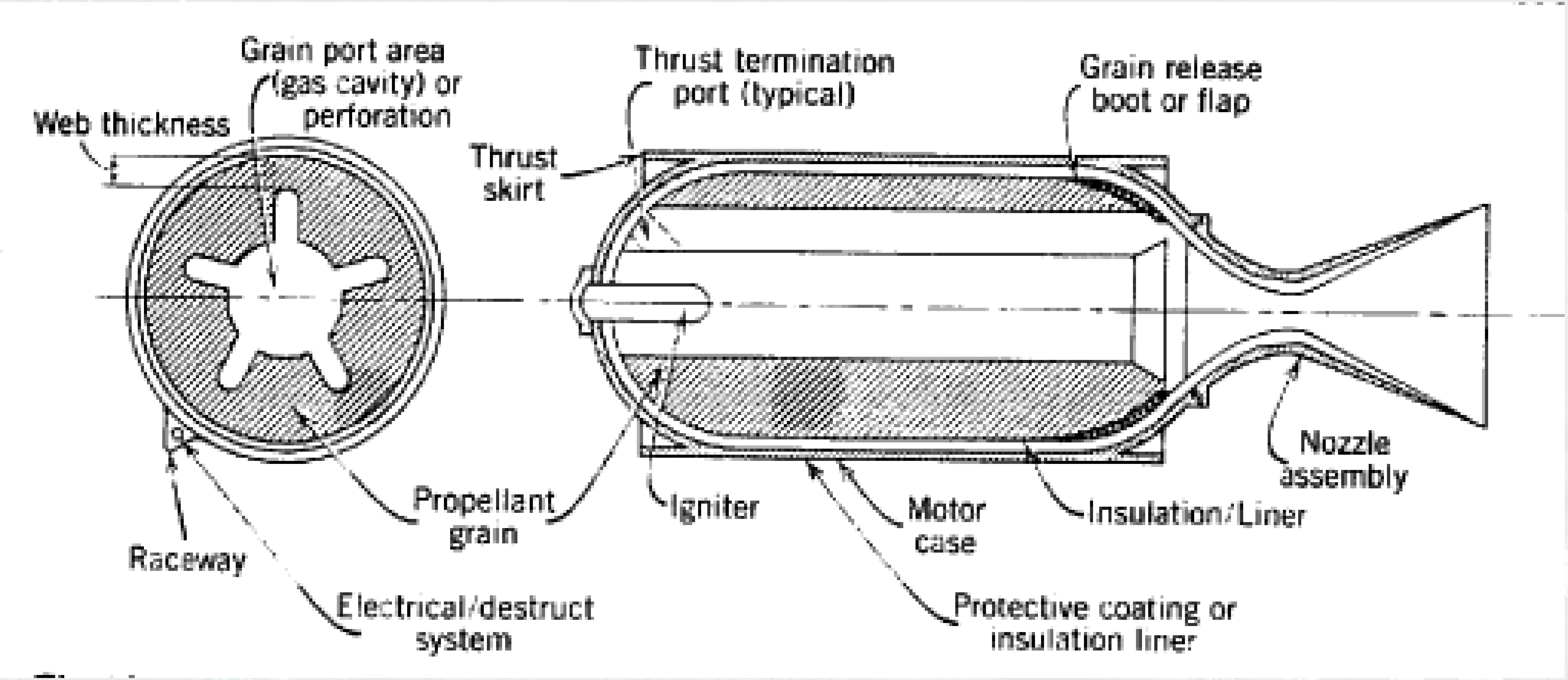
- $V_{e,vacuum}$=4460 sec $\Longrightarrow$ $V_{e,sea\ level}$=3890 sec

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

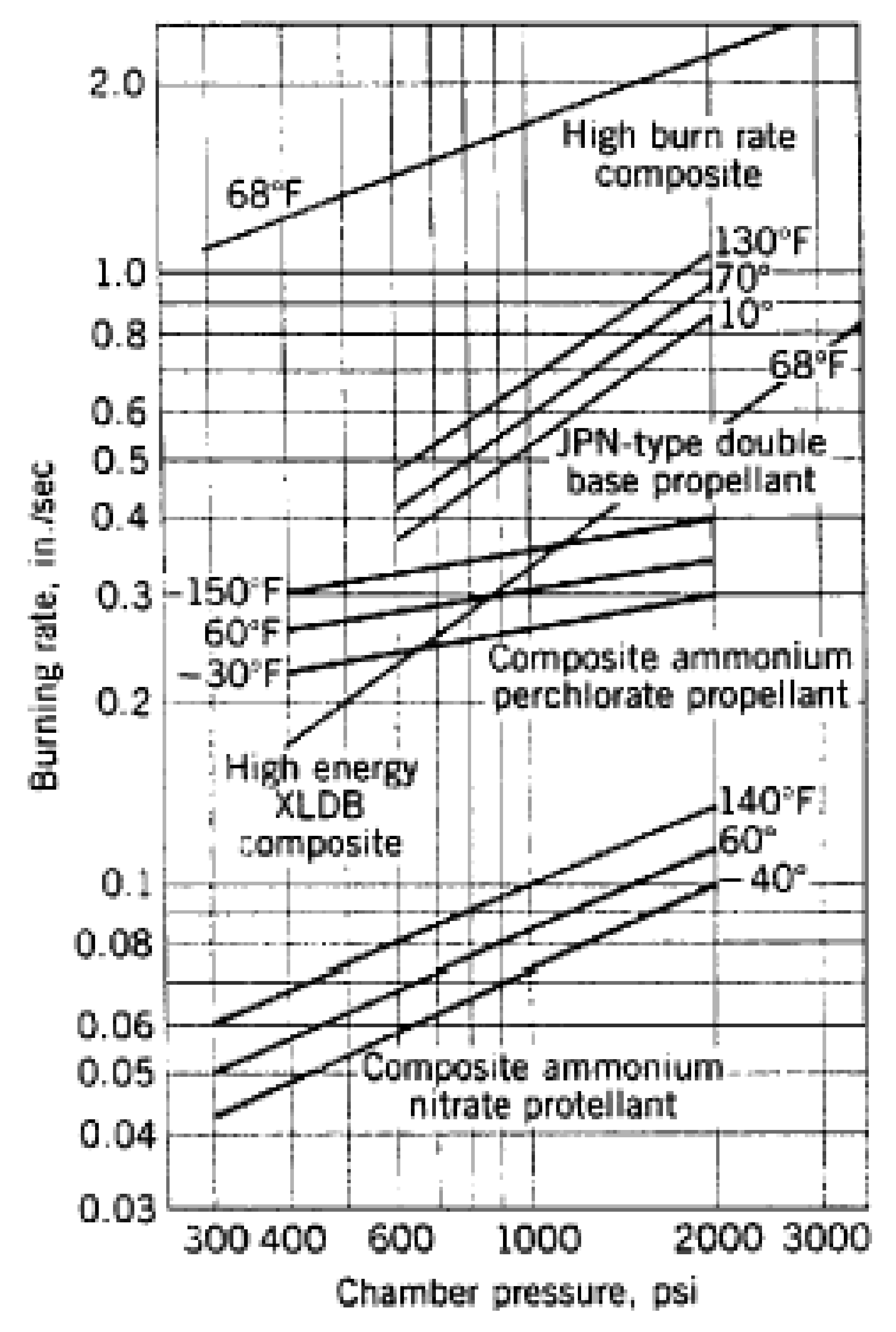
$$I_{sp} = \frac{T}{\dot{m}} \text{ and } T = \dot{m}V_e \text{ then } I_{sp} = V_e!!!$$

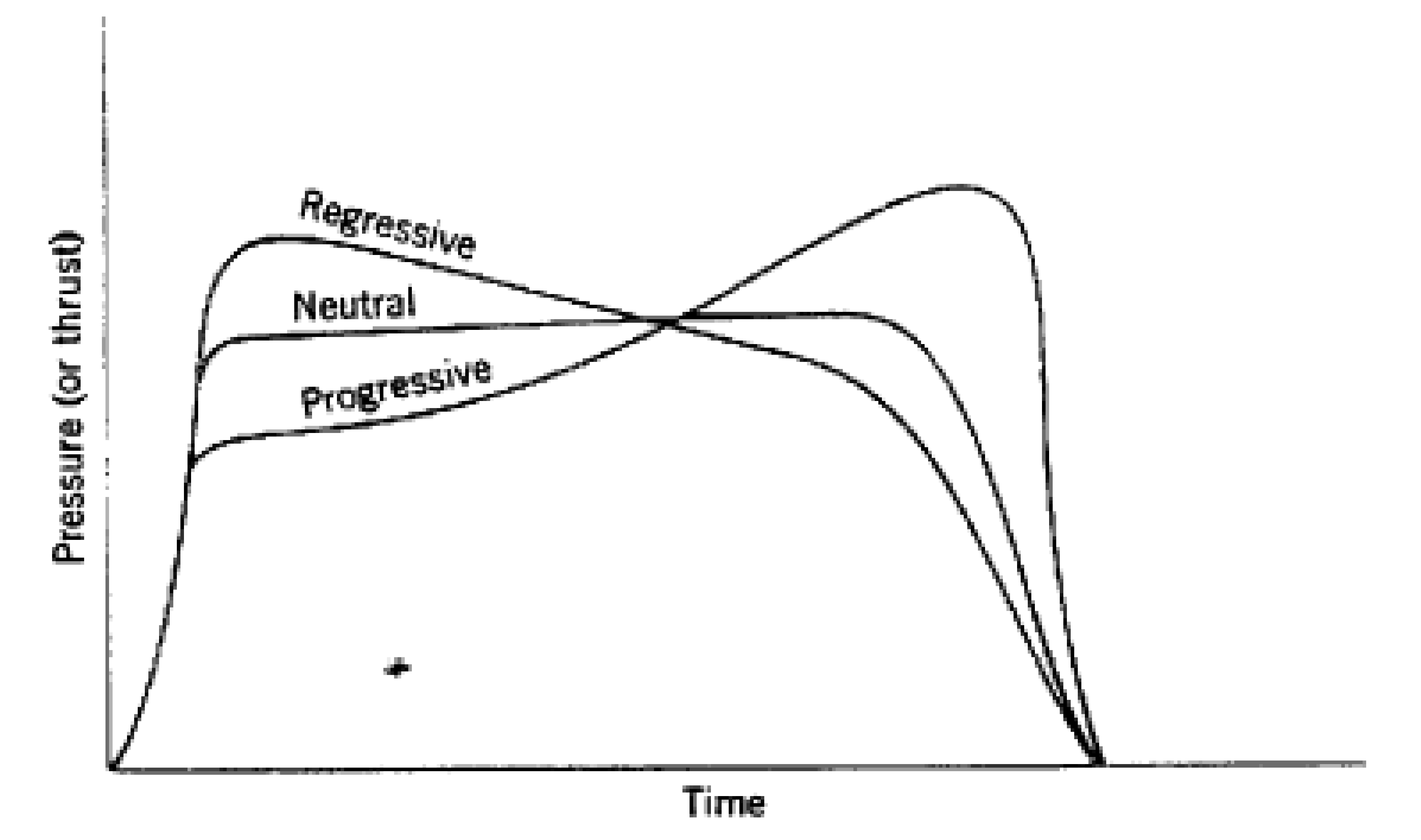
Solid Rocket Motor



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

Solid Propellant Combustion Characteristics





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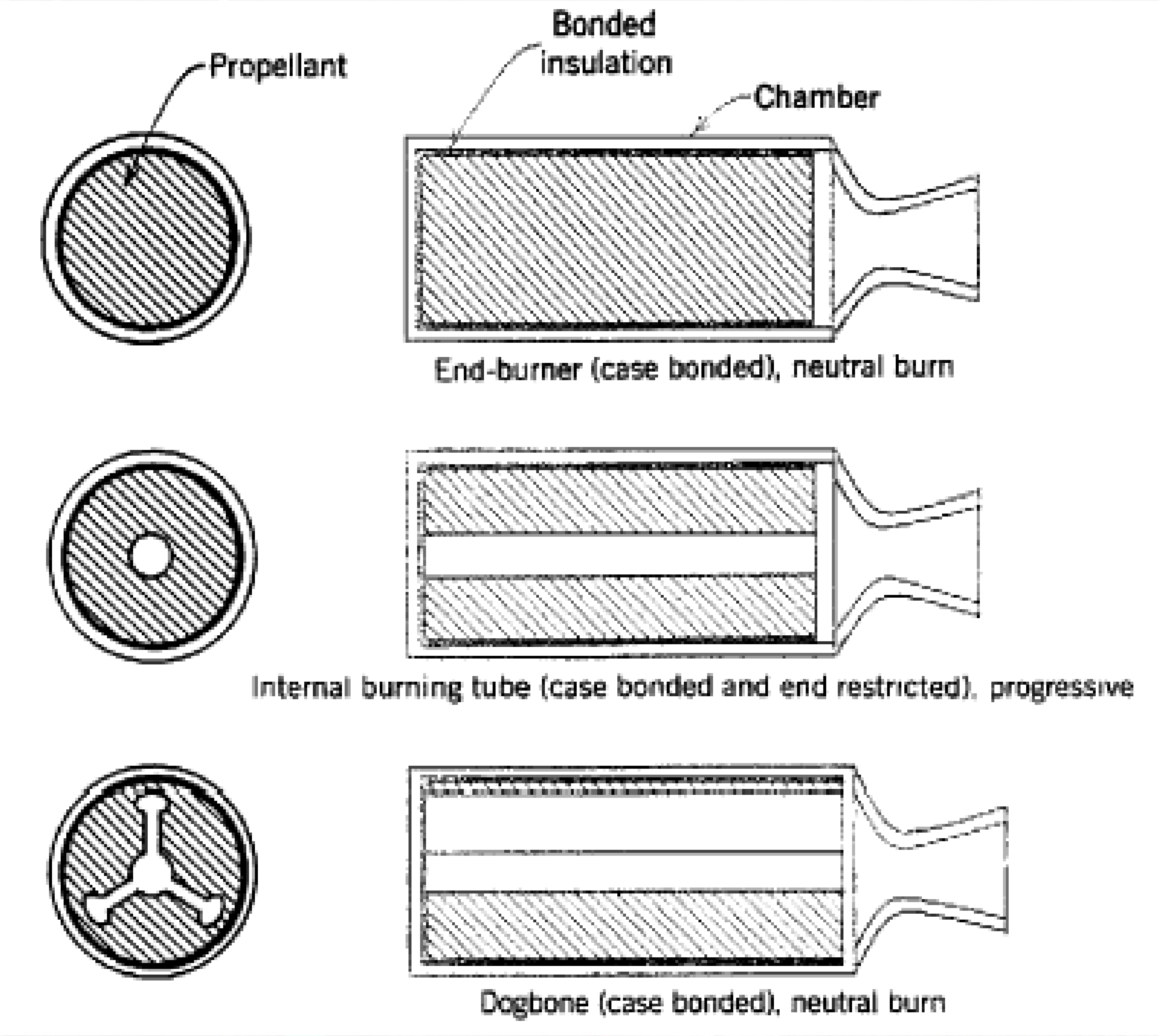
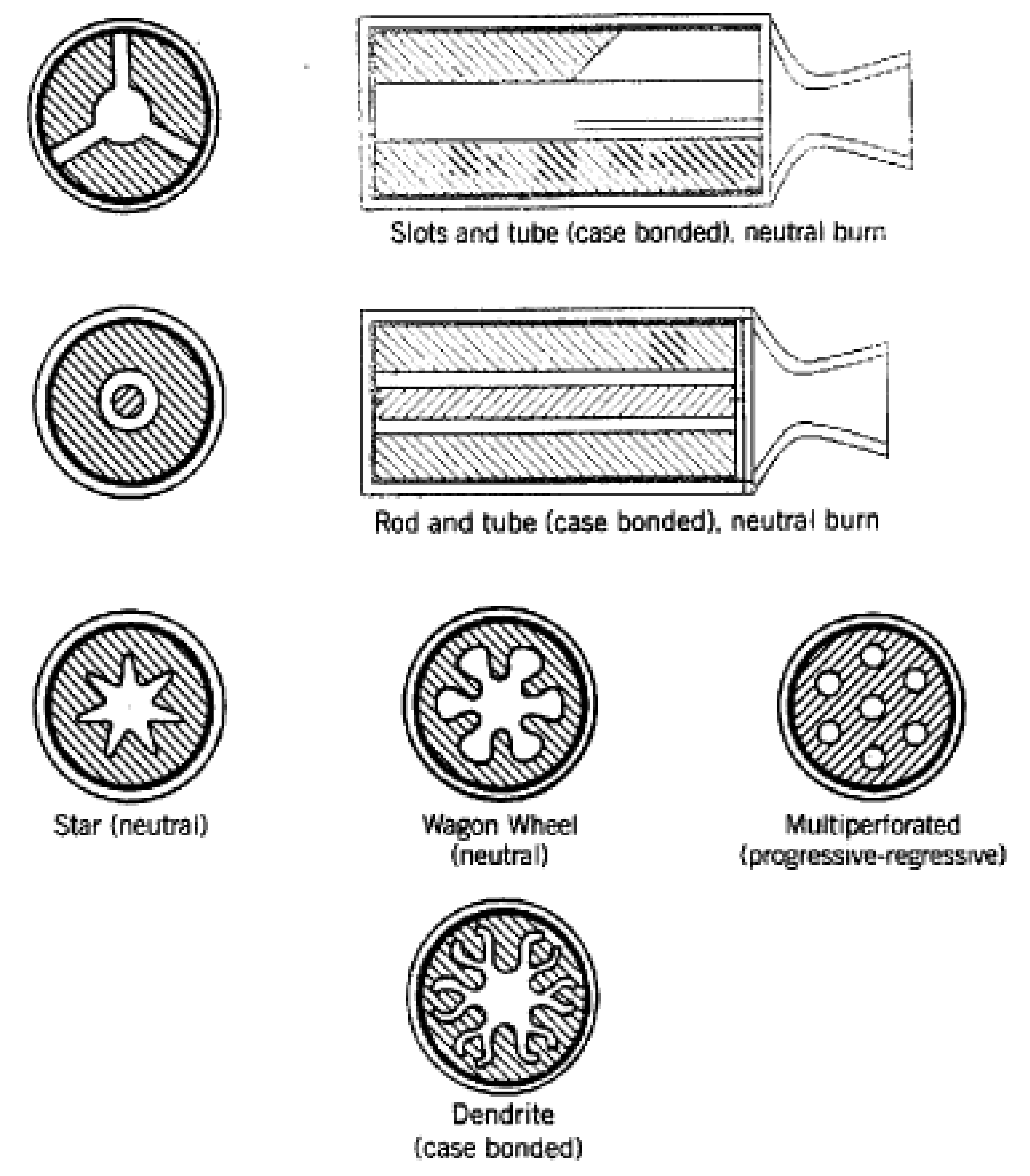
Solid Grain Configurations





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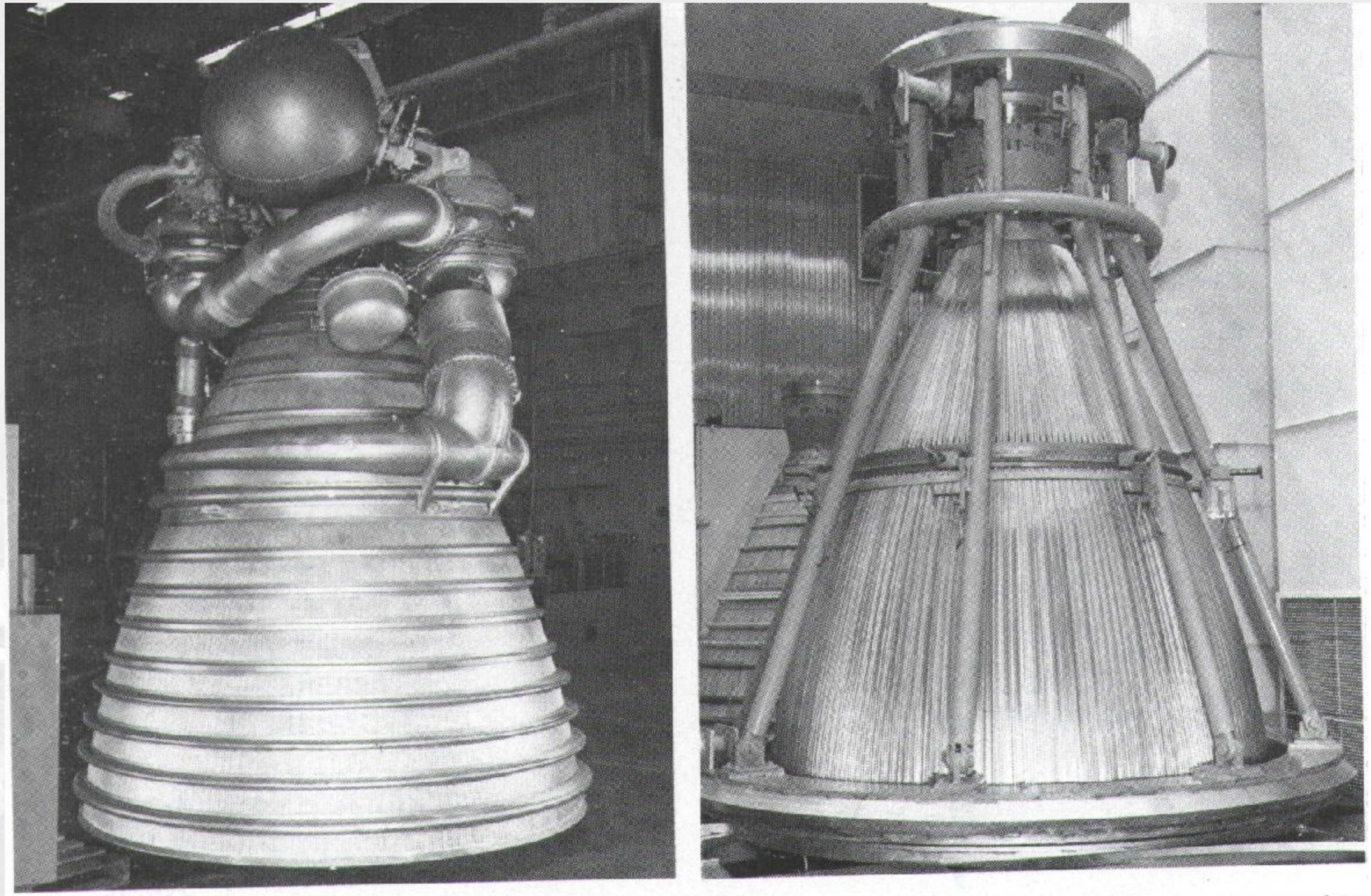
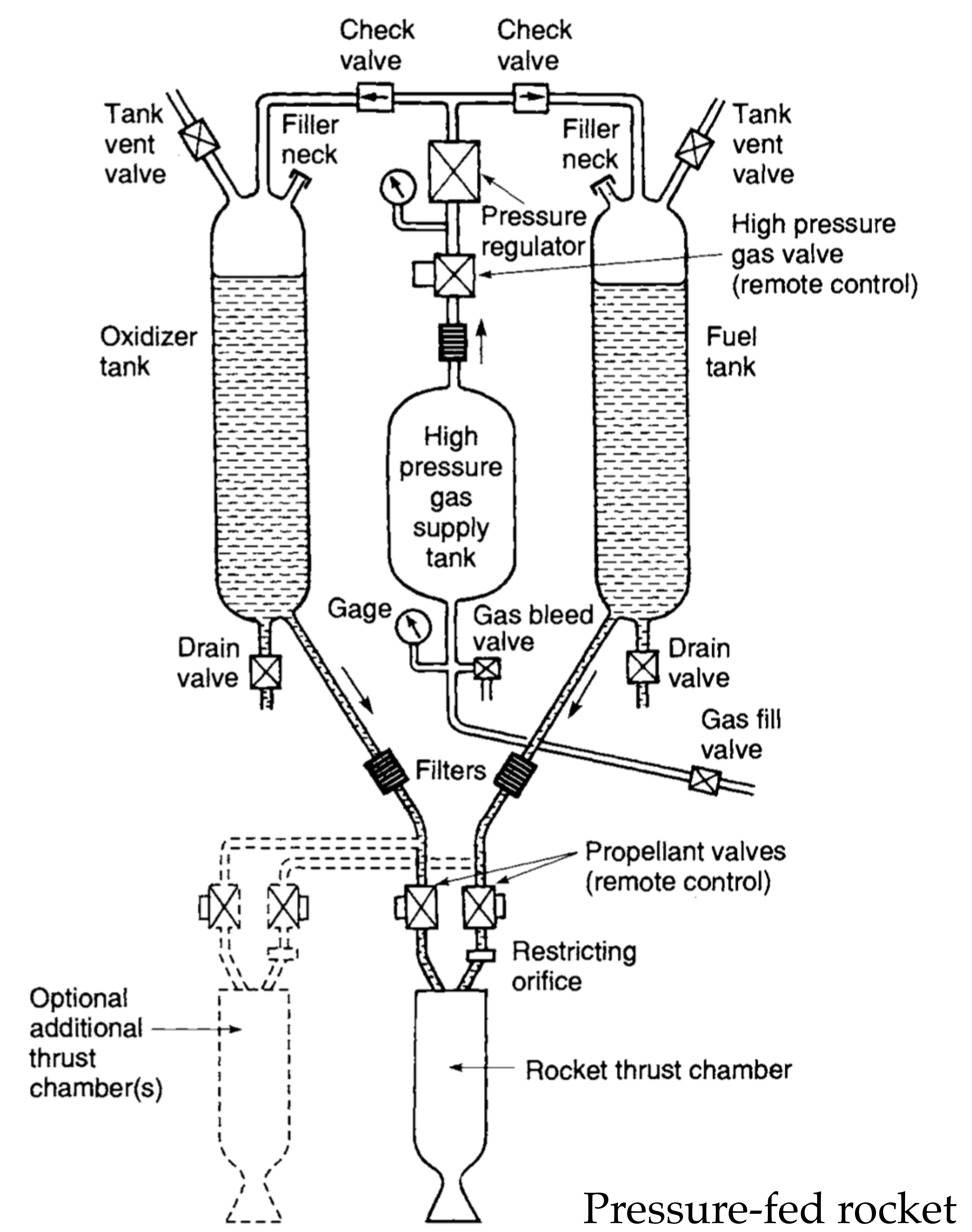
Liquid Rocket Engine

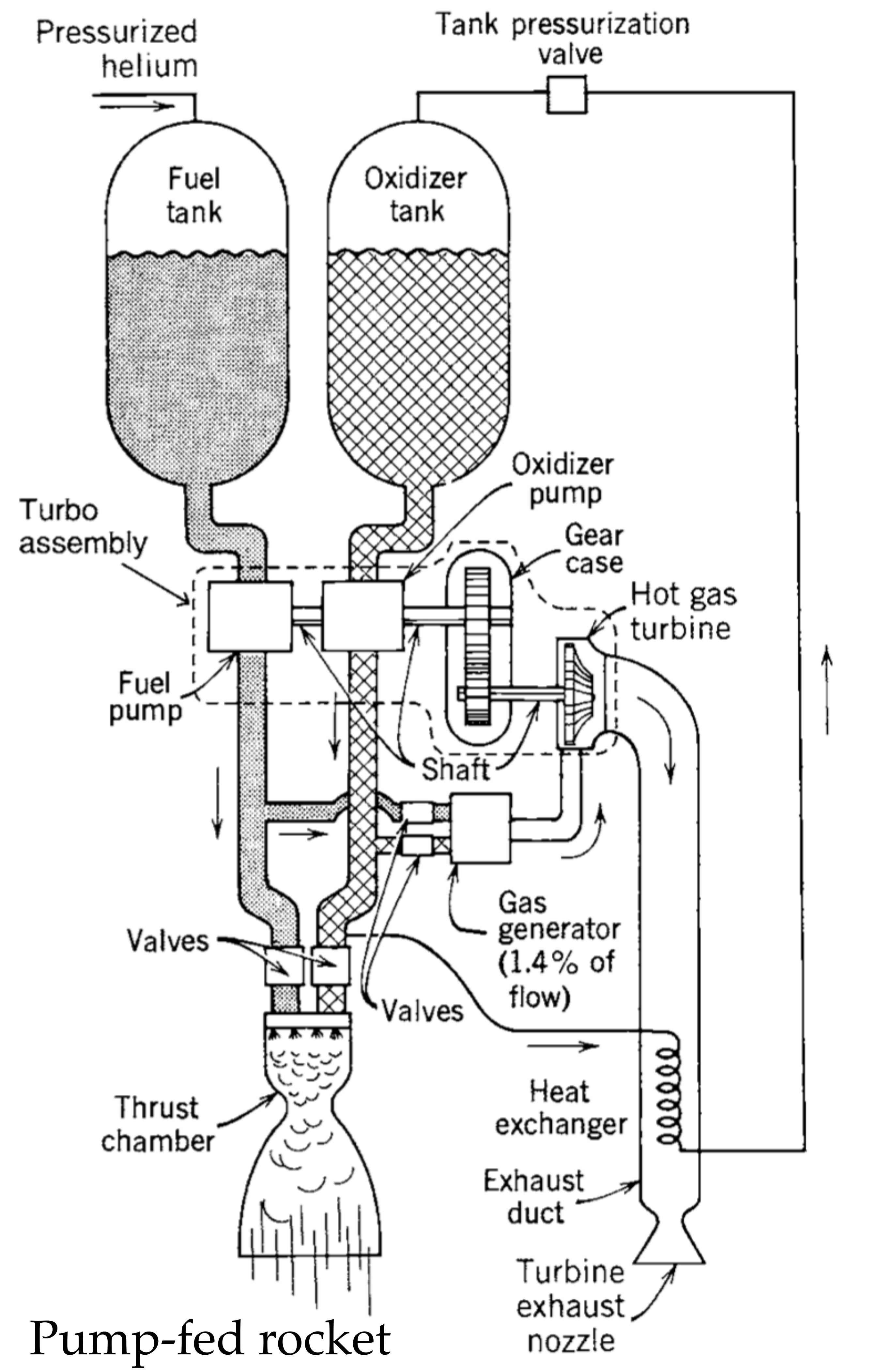


A completed J-2 rocket engine (left), with its pumps and lines installed. The basic engine structure is built up from a series of hollow tubes (right).

Liquid Propellant Feed Systems



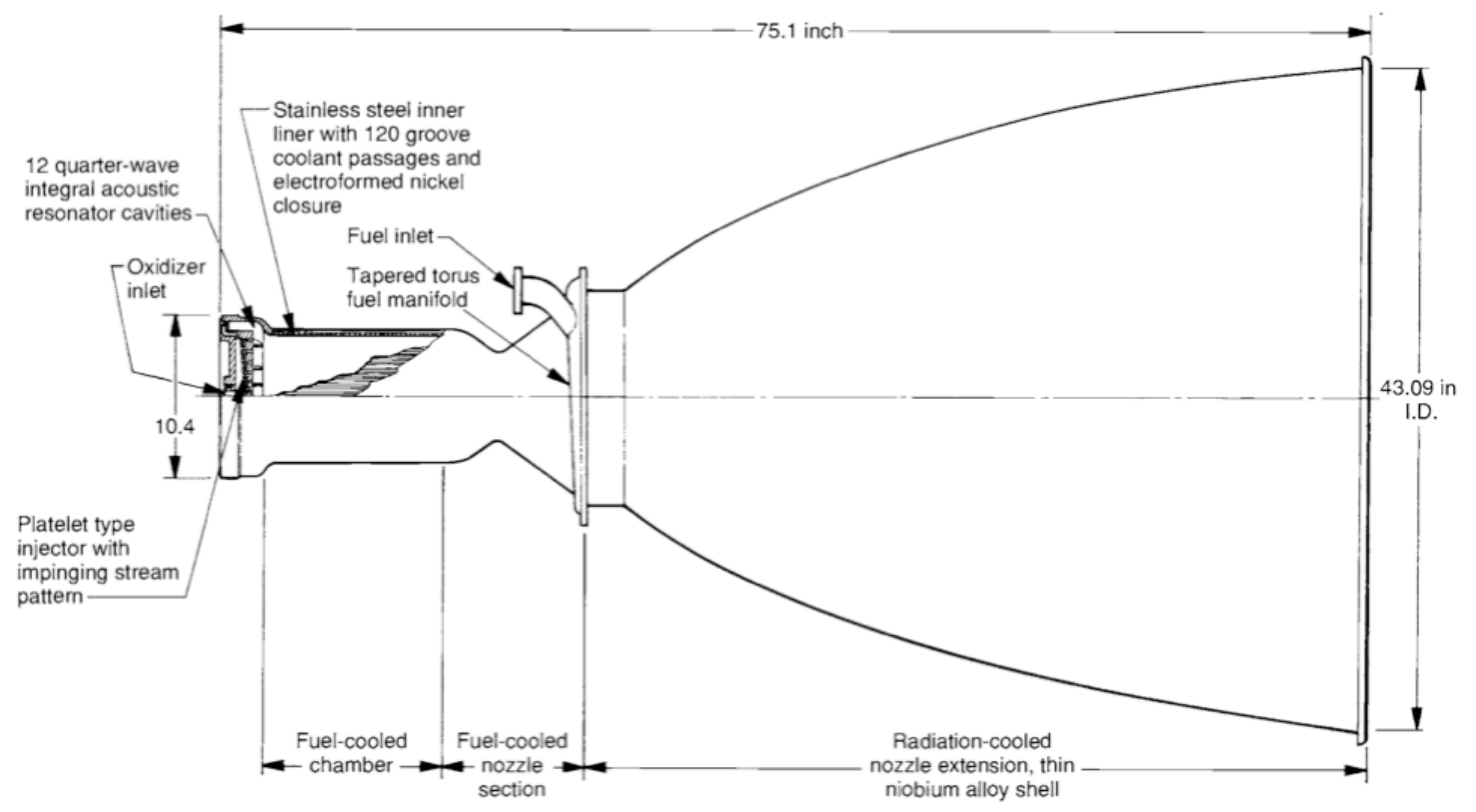
Pressure-fed rocket



Pump-fed rocket

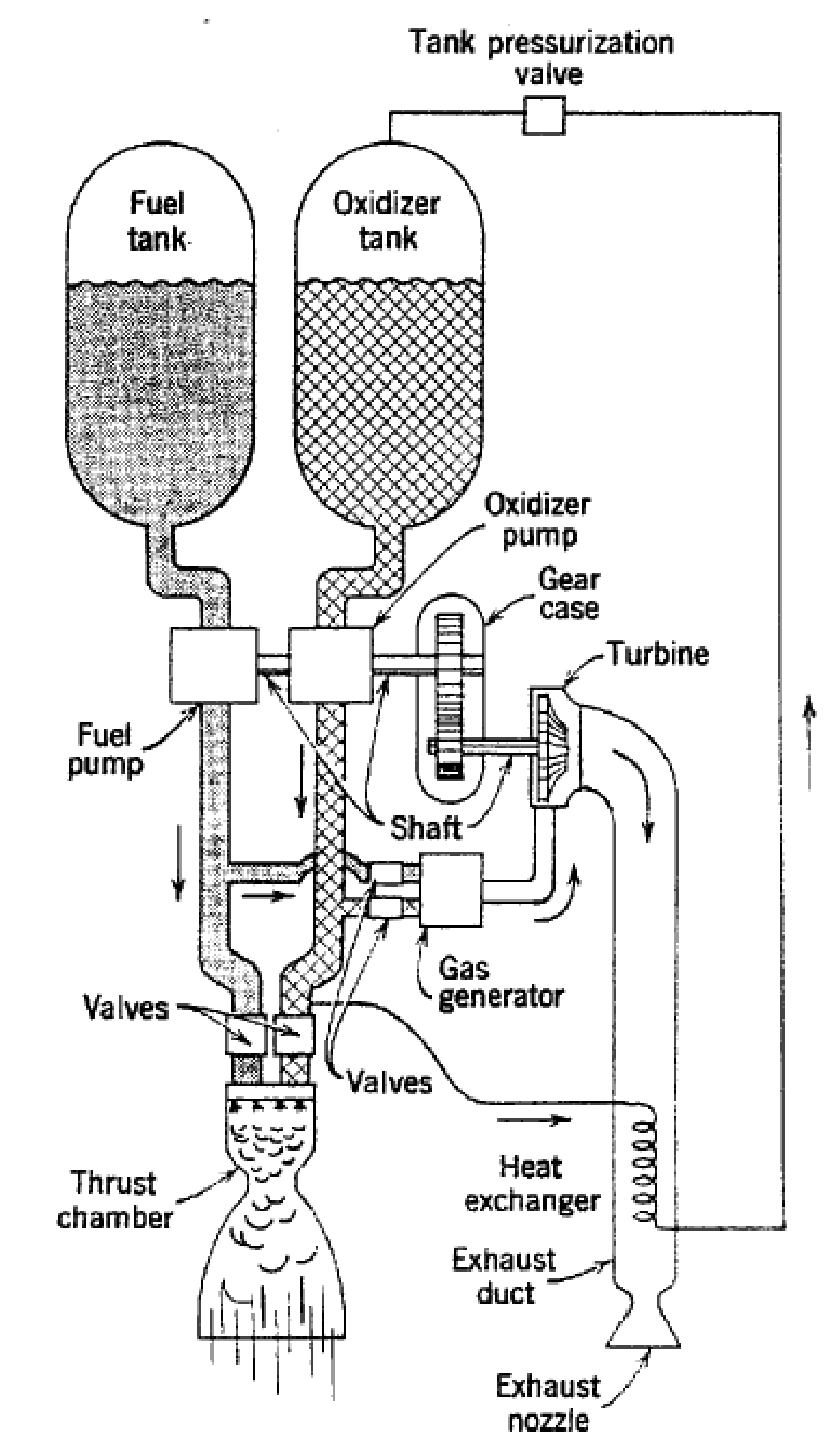
From Sutton and Biblarz, *Rocket Propulsion Elements (7th Edition)*, Wiley-Interscience, 2001

Space Shuttle OMS Engine



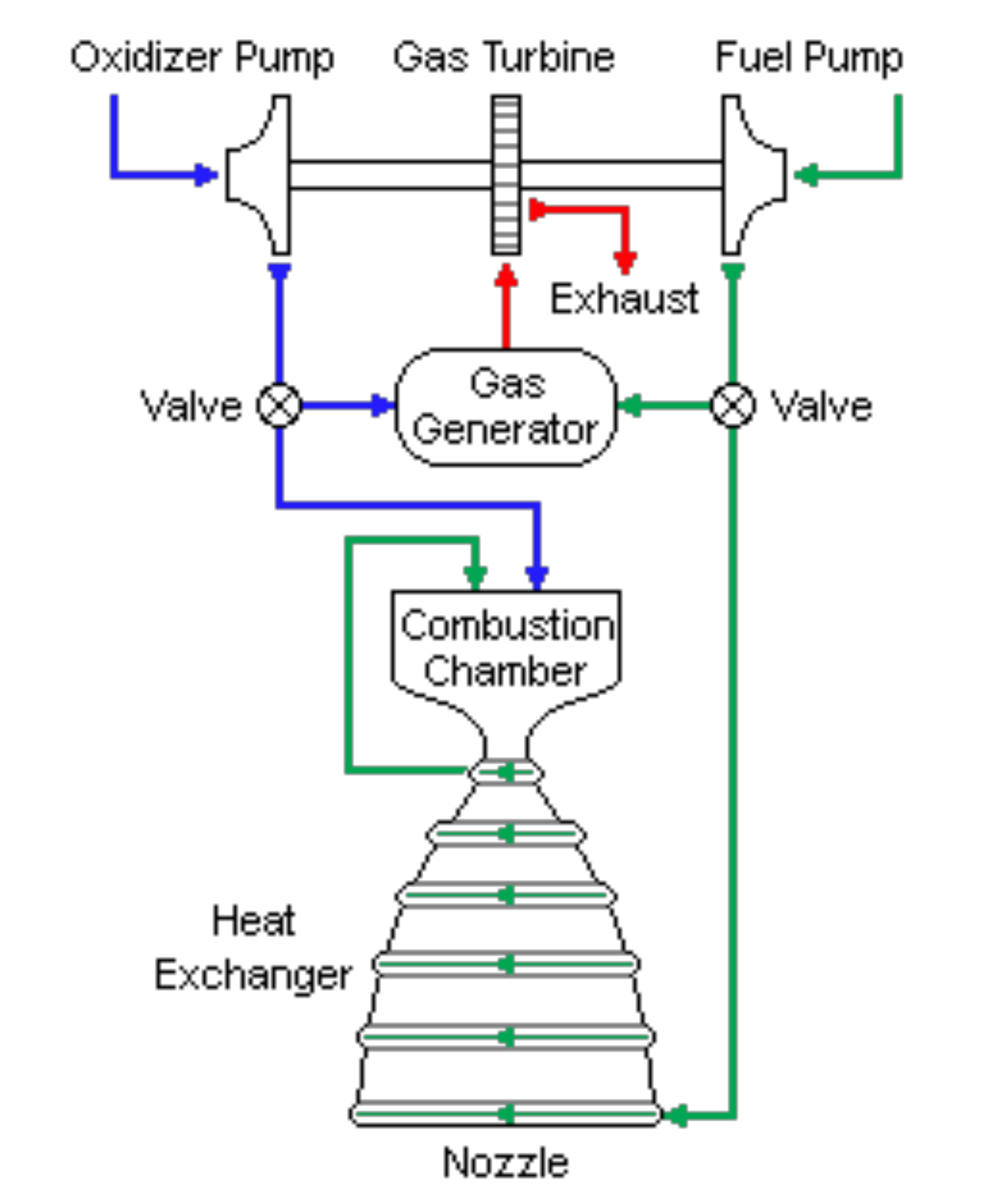
From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 2001

Turbopump Fed Liquid Rocket Engine



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

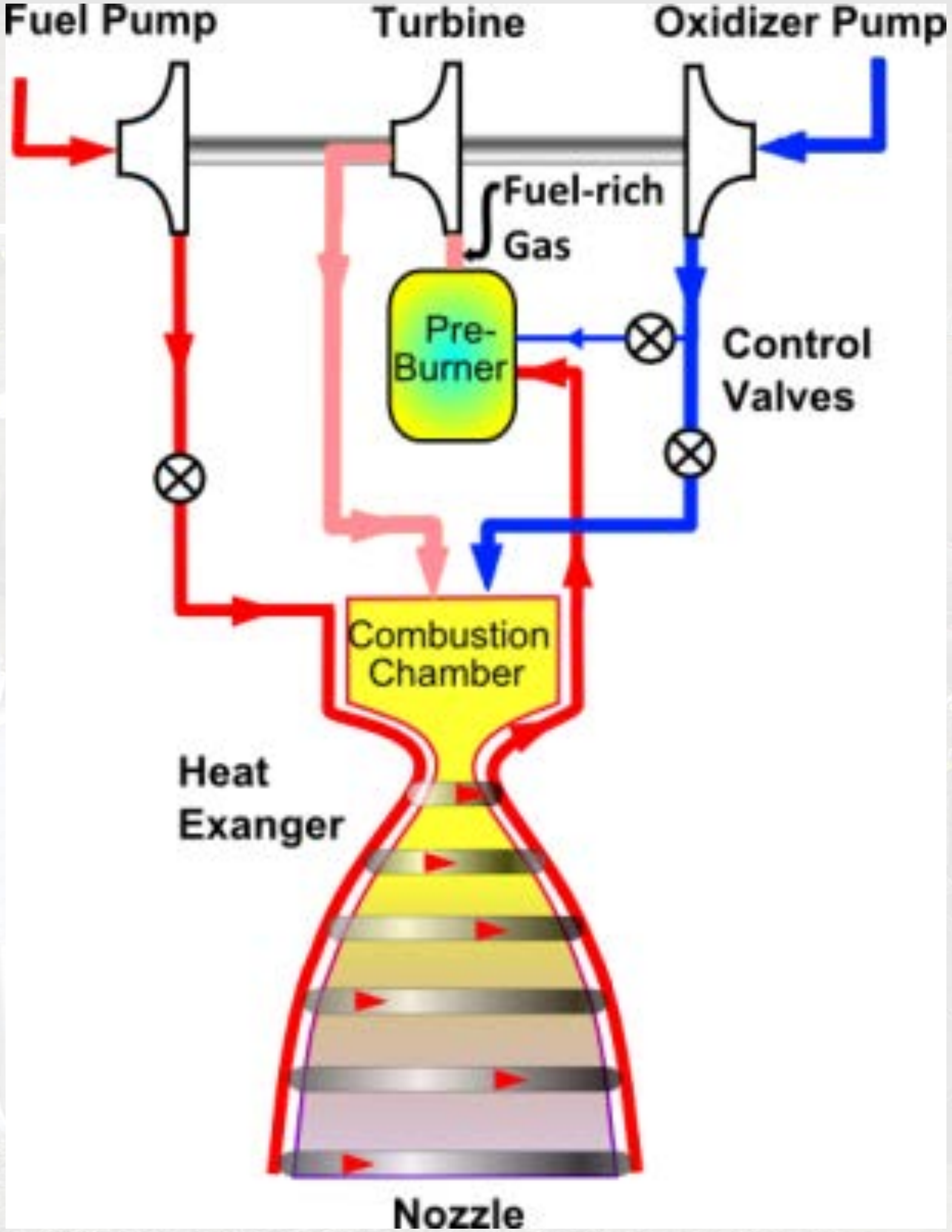
Gas Generator Engine Schematic



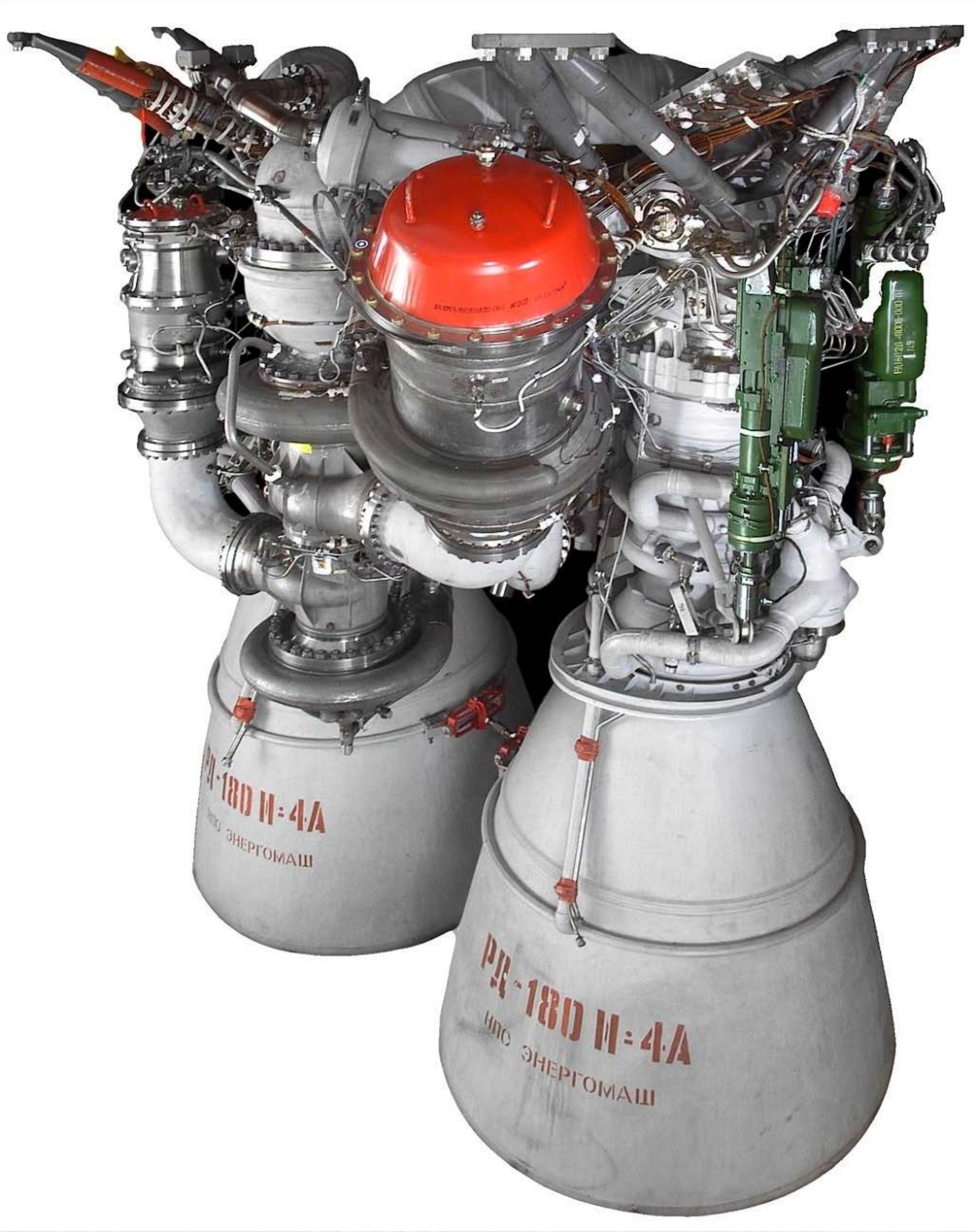
SpaceX Merlin 1D Engines

Falcon 9 Octoweb Engine Mount

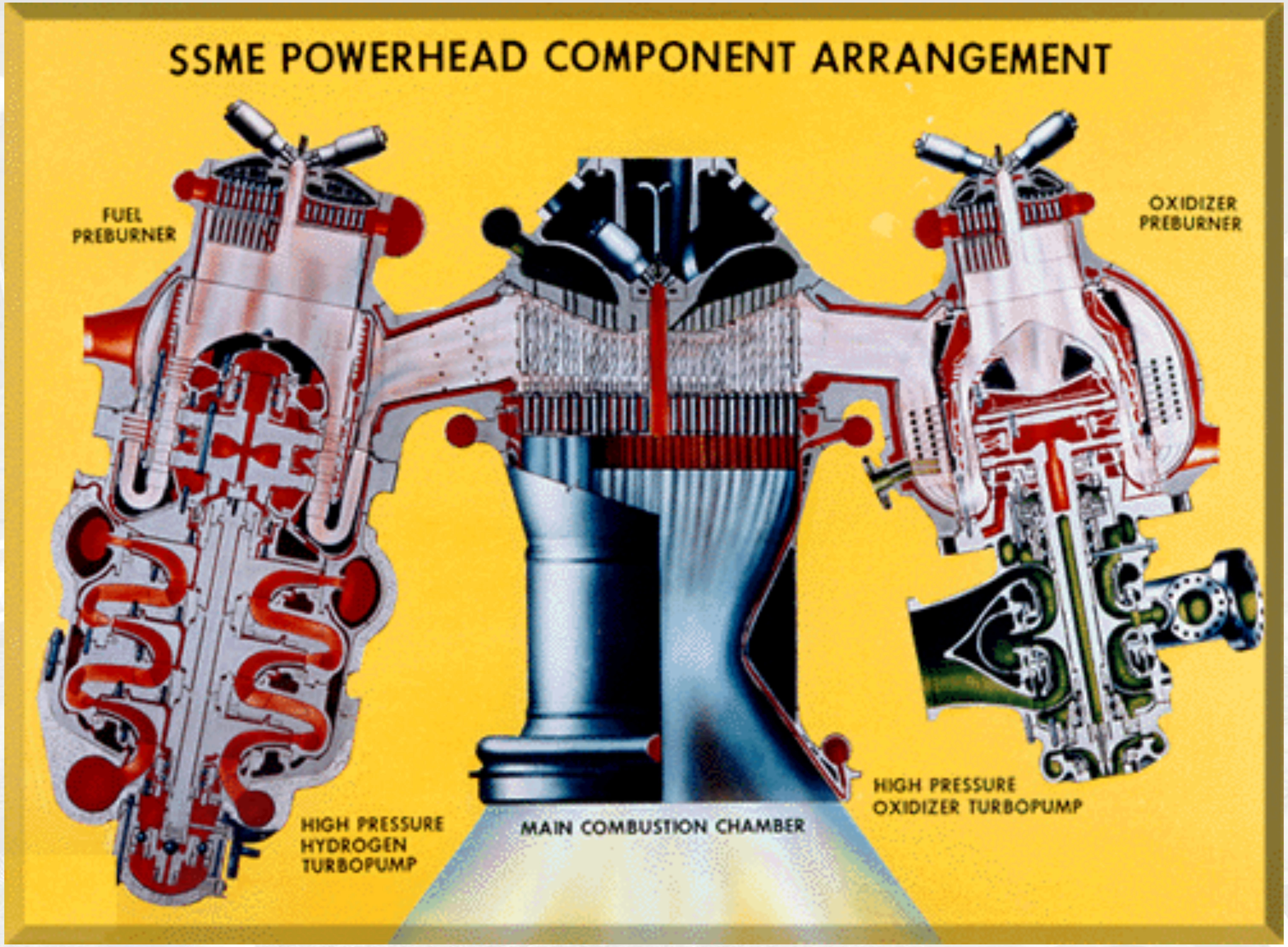
Staged-Combustion Engine Schematic



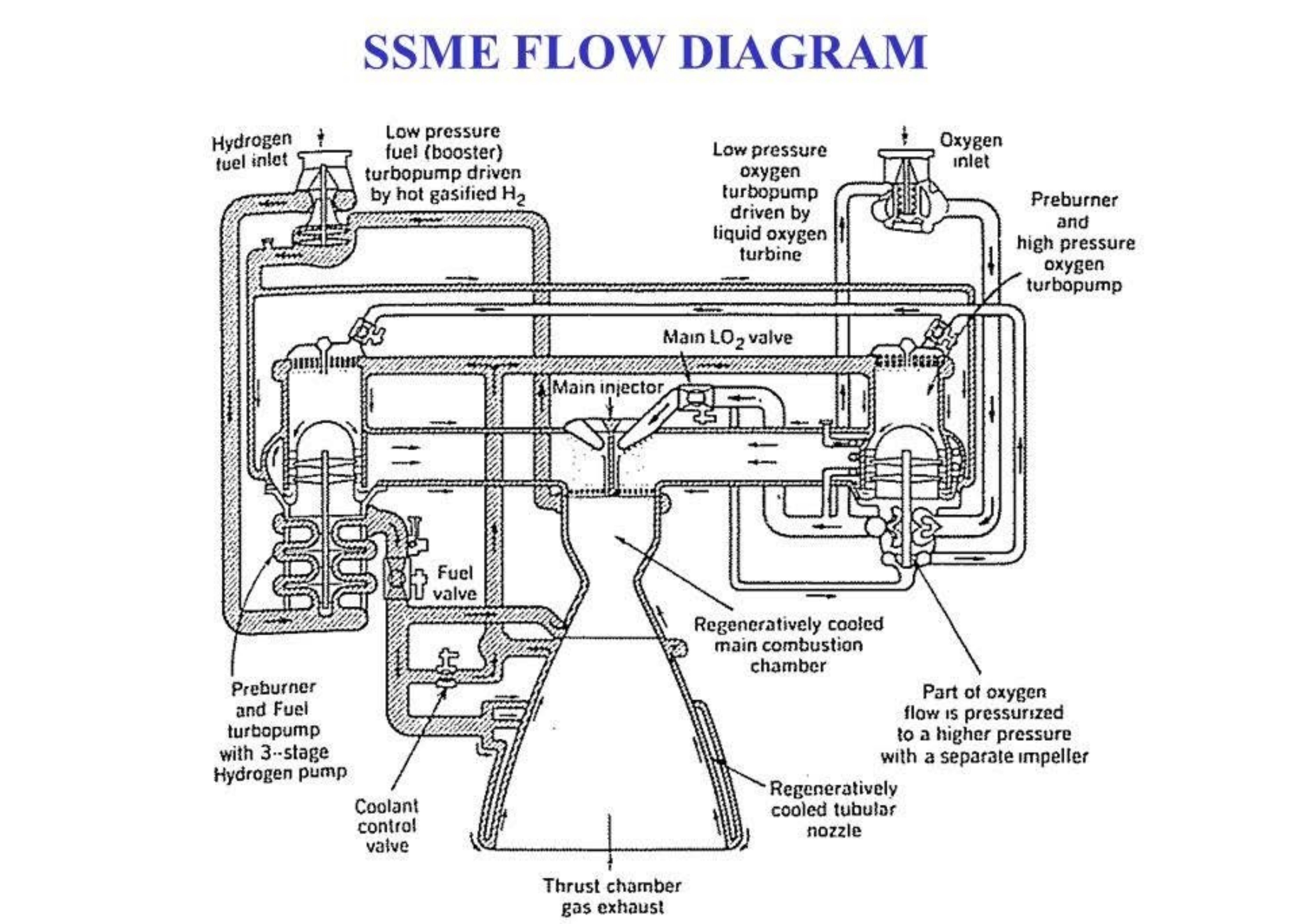
RD-180 Engine(s) (Atlas V)



SSME Powerhead Configuration



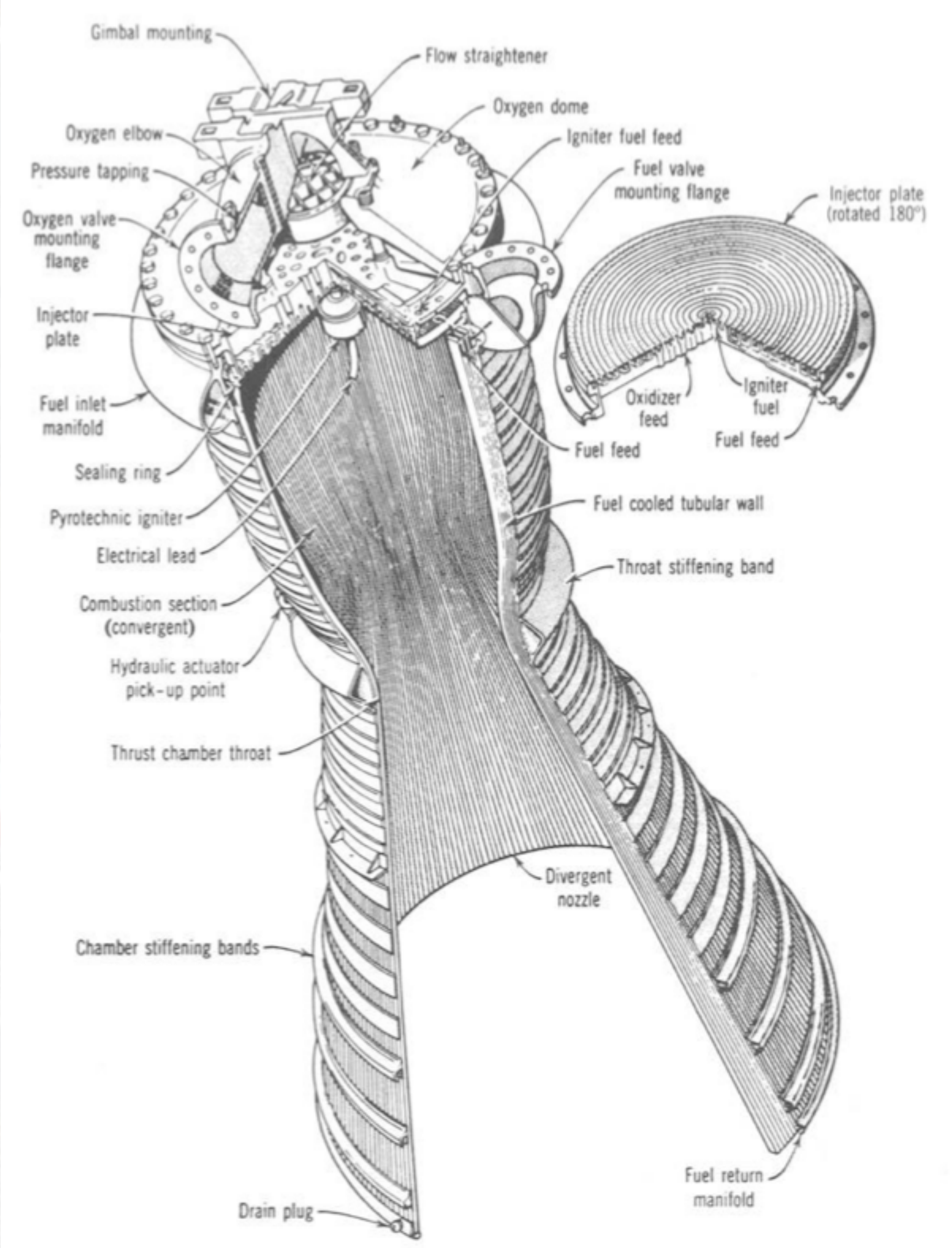
SSME Engine Cycle

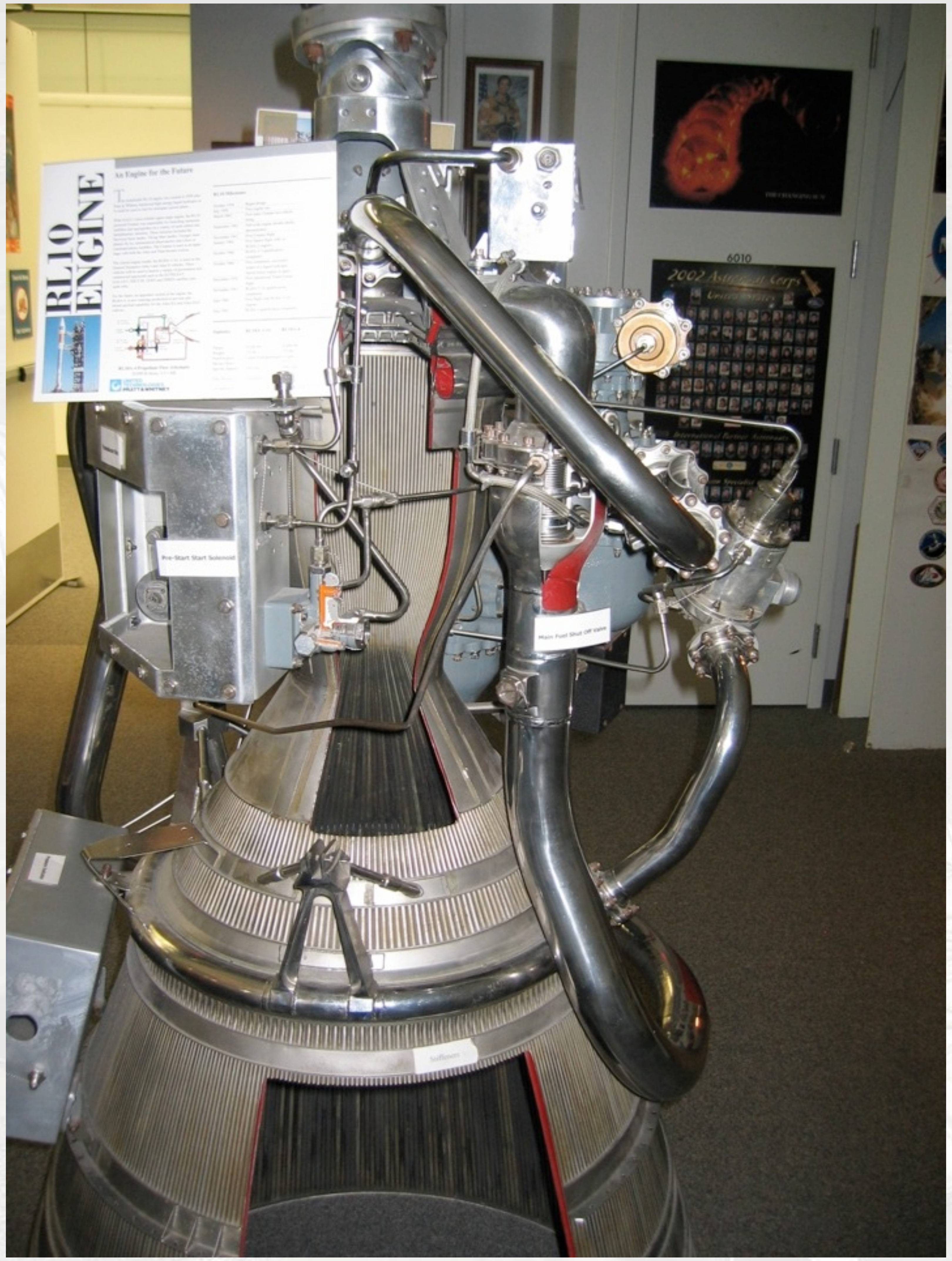


From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

SpaceX SuperHeavy Raptor Engines

Liquid Rocket Engine Cutaway





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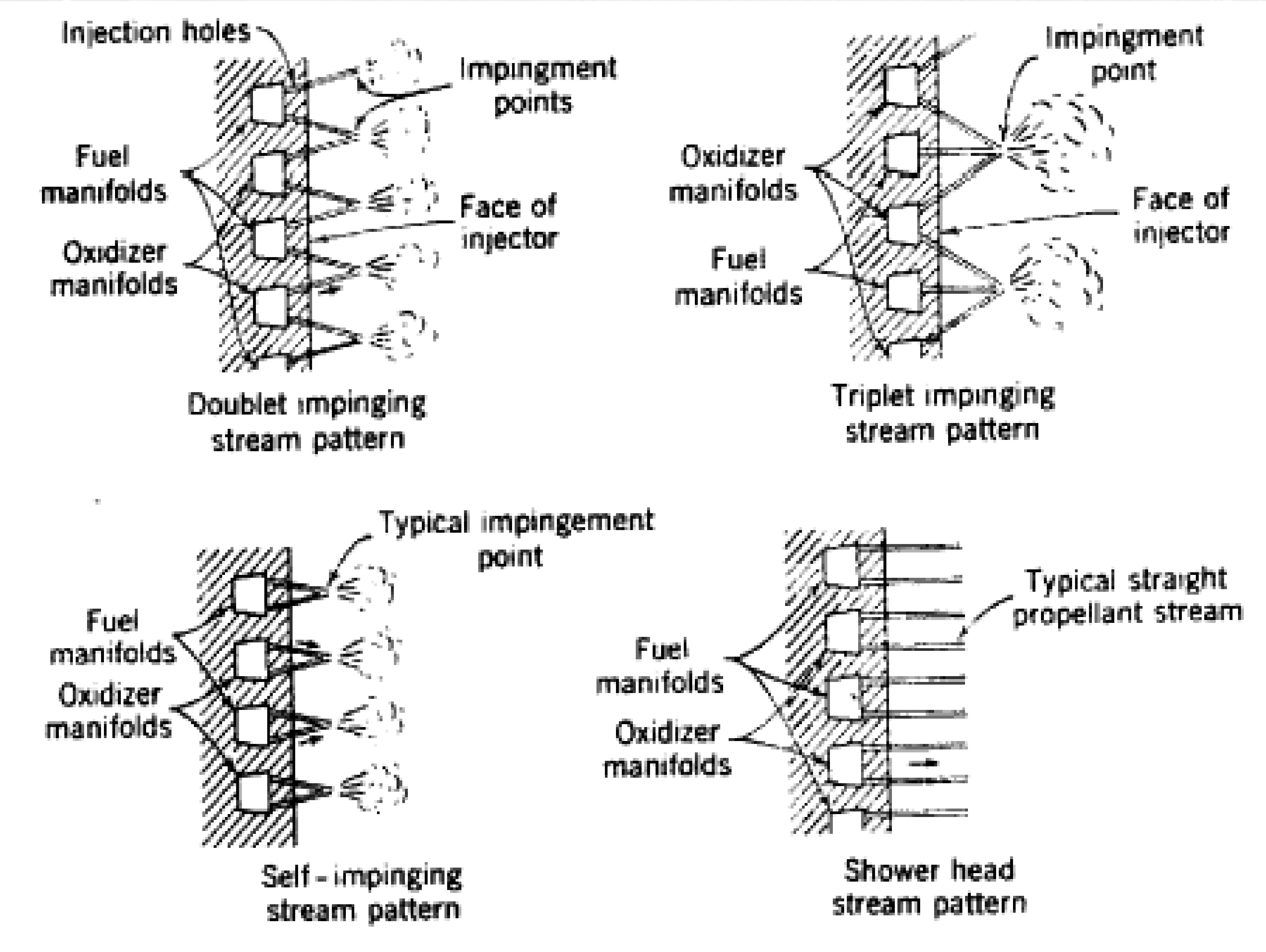
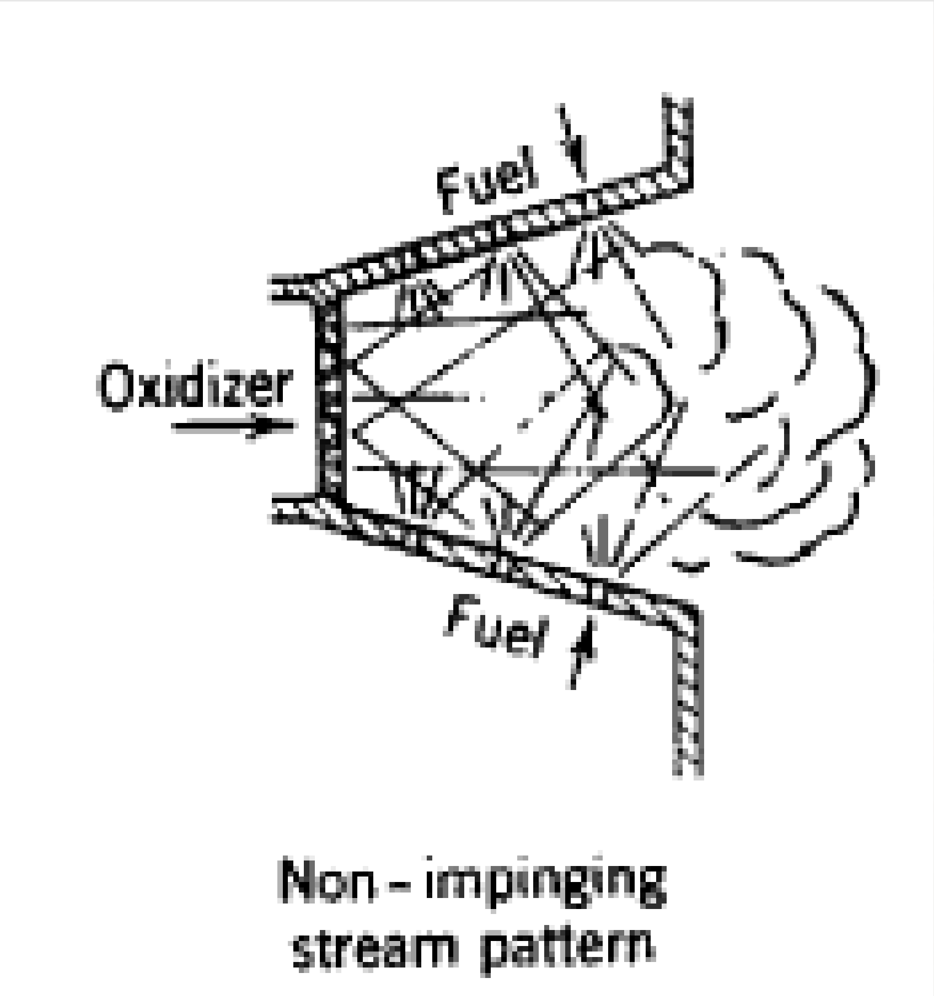
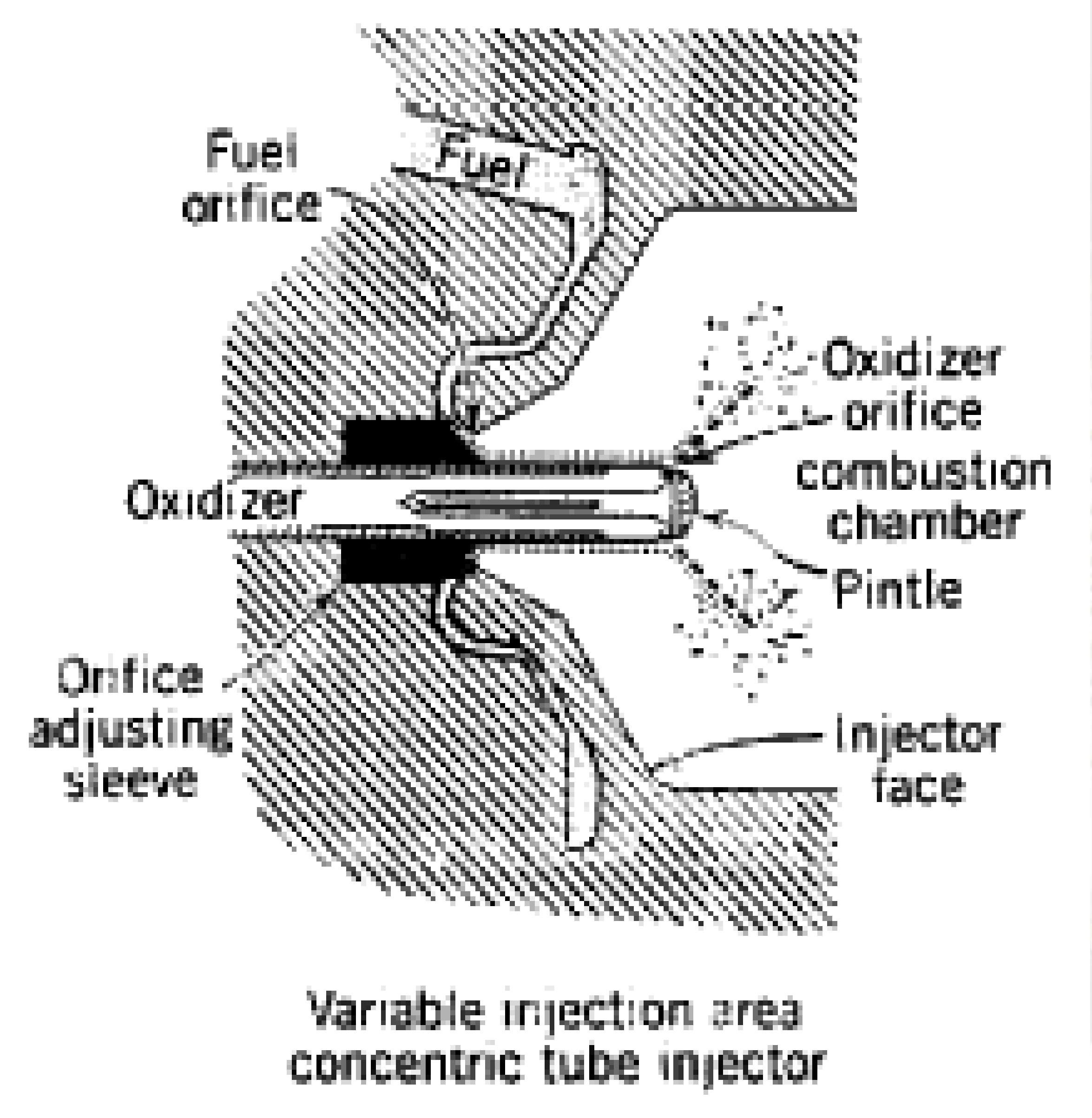
H-1 Engine Injector Plate

Injector Concepts



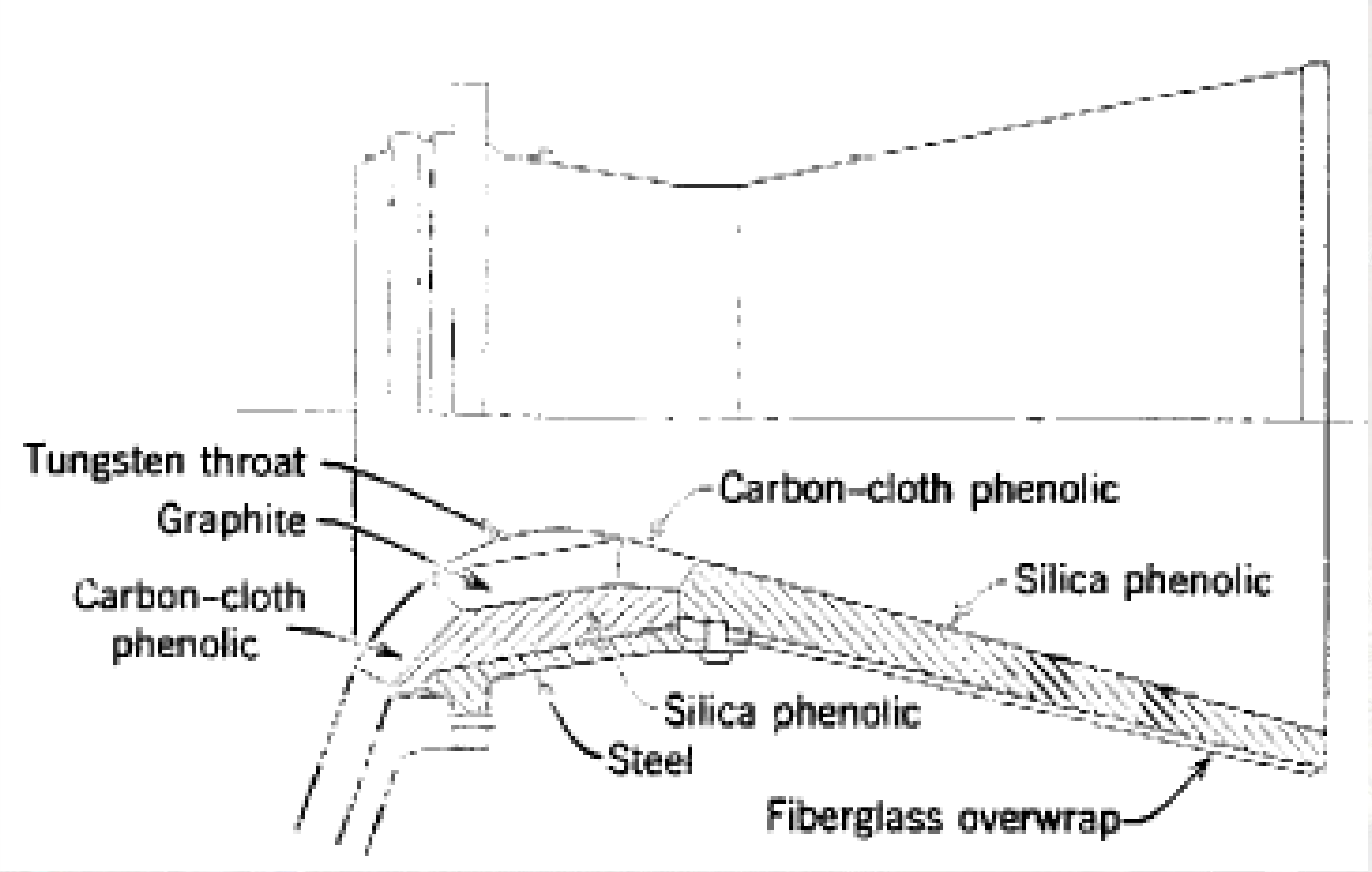




From G. P. Sutton, Rocket Propulsion Elements (5th ed.)
John Wiley and Sons, 1986

TR-201 Engine (LM Descent/Delta)

Solid Rocket Nozzle (Heat-Sink)



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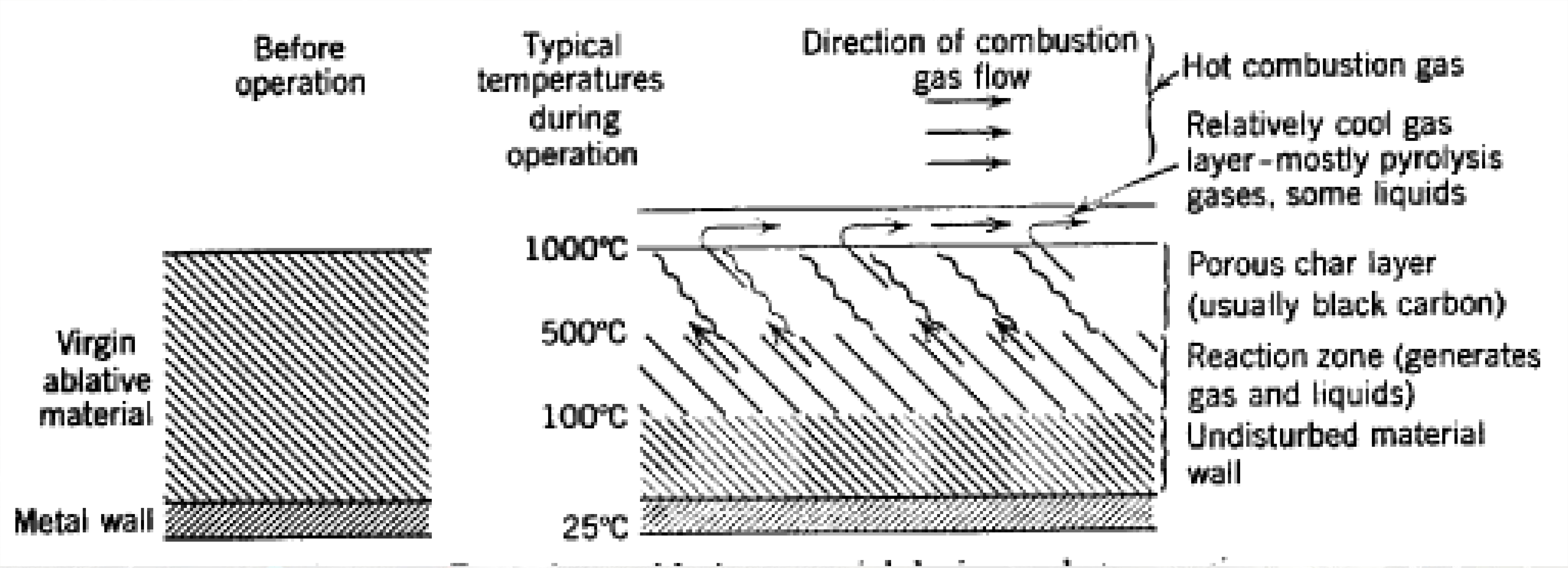
Ablative Nozzle Schematic



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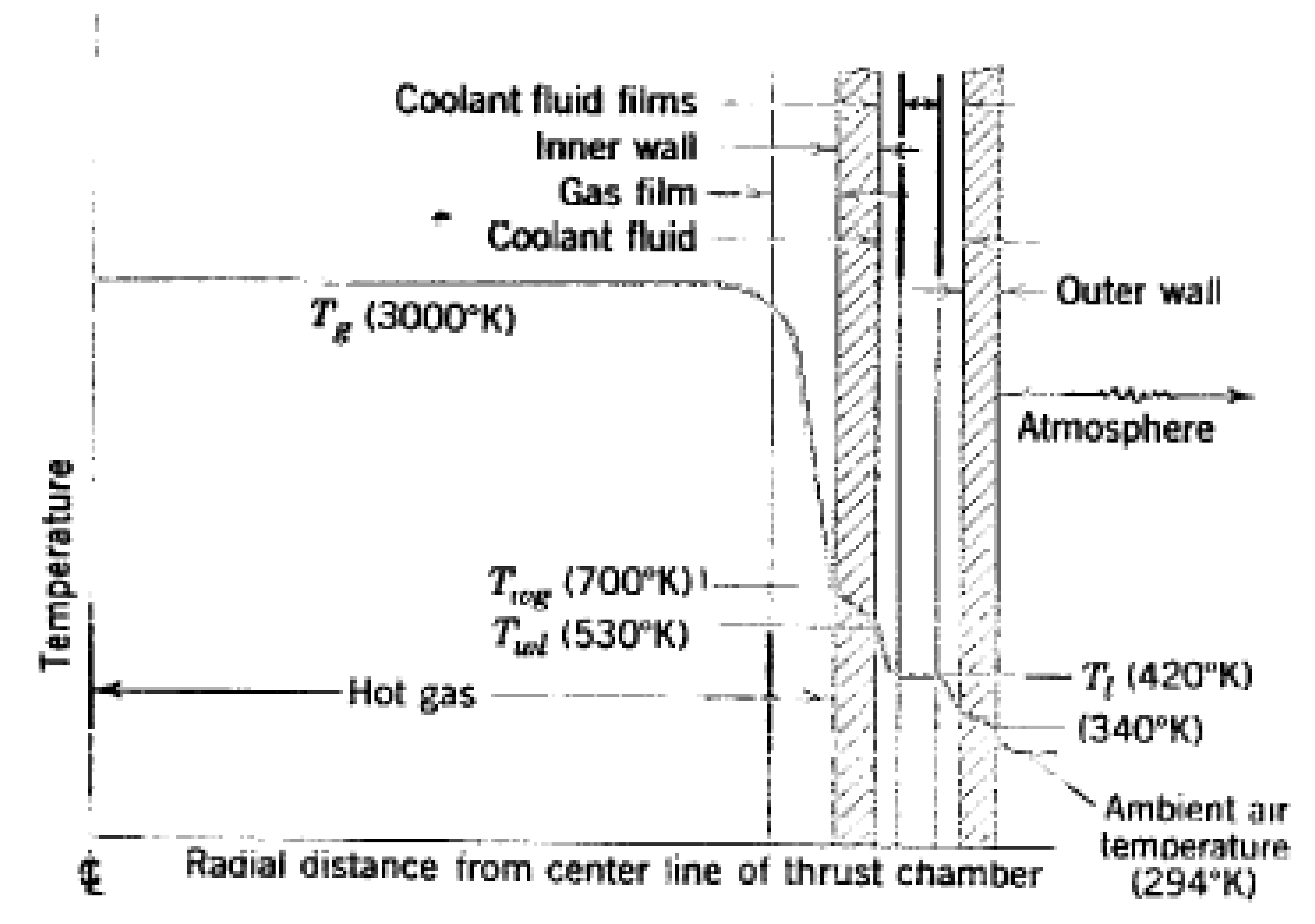
Active Chamber Cooling Schematic



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

Hybrid Rocket Schematic

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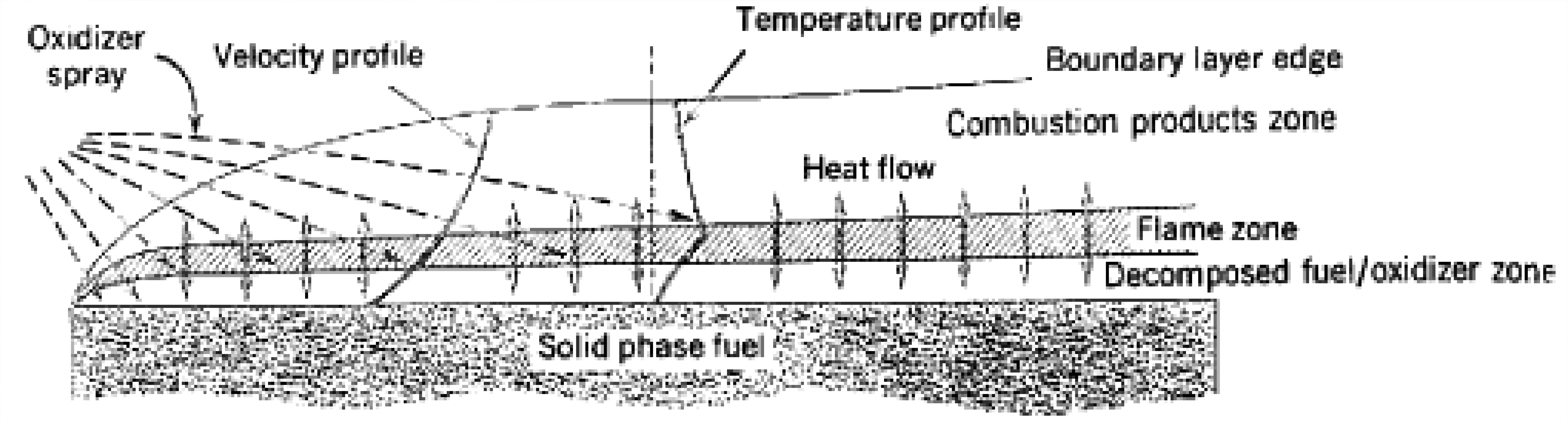
Hybrid Rocket Combustion



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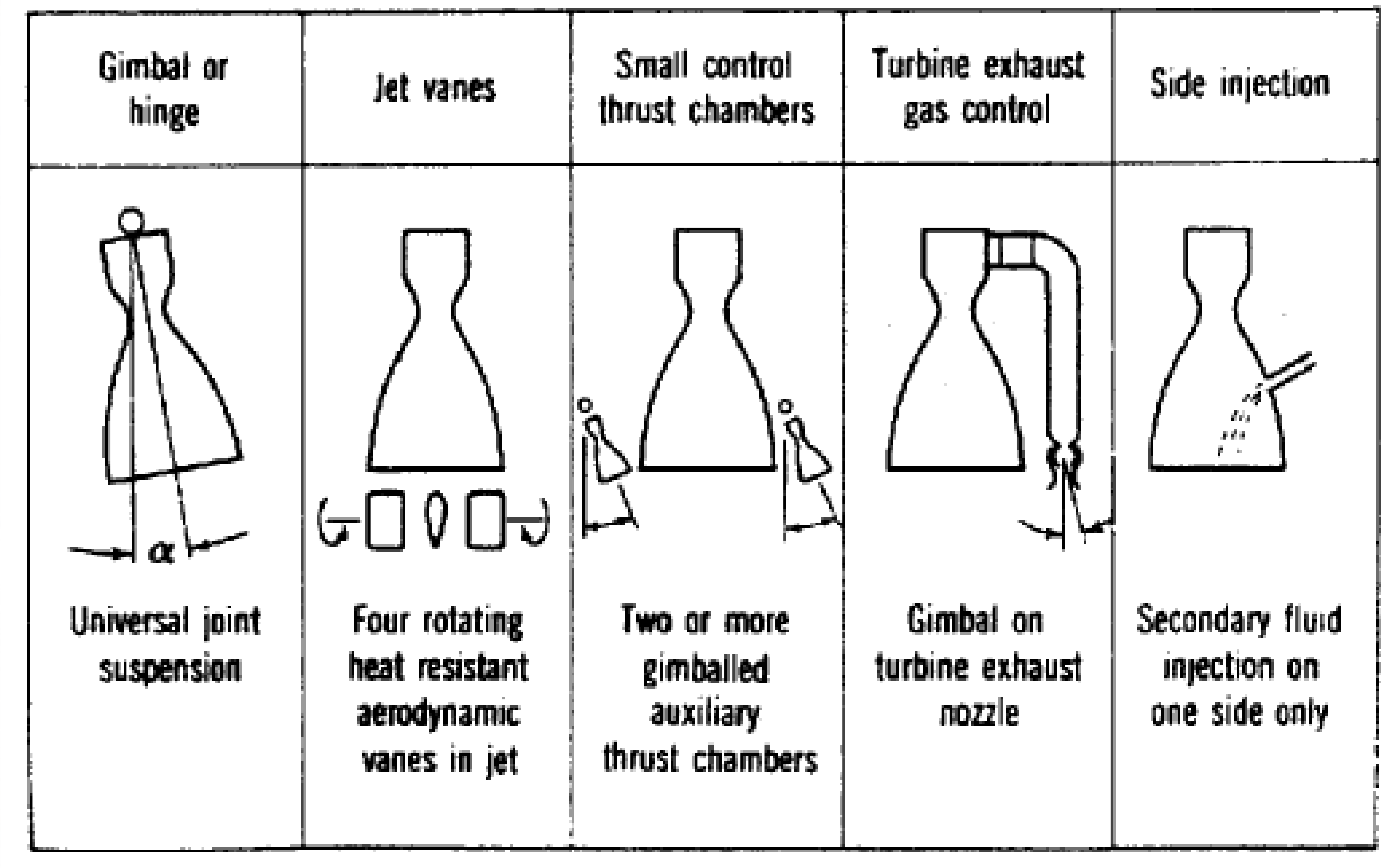
Thrust Vector Control Approaches



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Reaction Control Systems

- Thruster control of vehicle attitude and translation
- "Bang-bang" control algorithms
- Design goals:
  - Minimize coupling (pure forces for translation; pure moments for rotation)except for pure entry vehicles
  - Minimize duty cycle (use propellant as sparingly as possible)
  - Meet requirements for maximum rotational and linear accelerations

Single-Axis Equations of Motion

$$\tau = I\ddot{\theta}$$

$$\frac{\tau}{I}t = \dot{\theta} + C_1$$

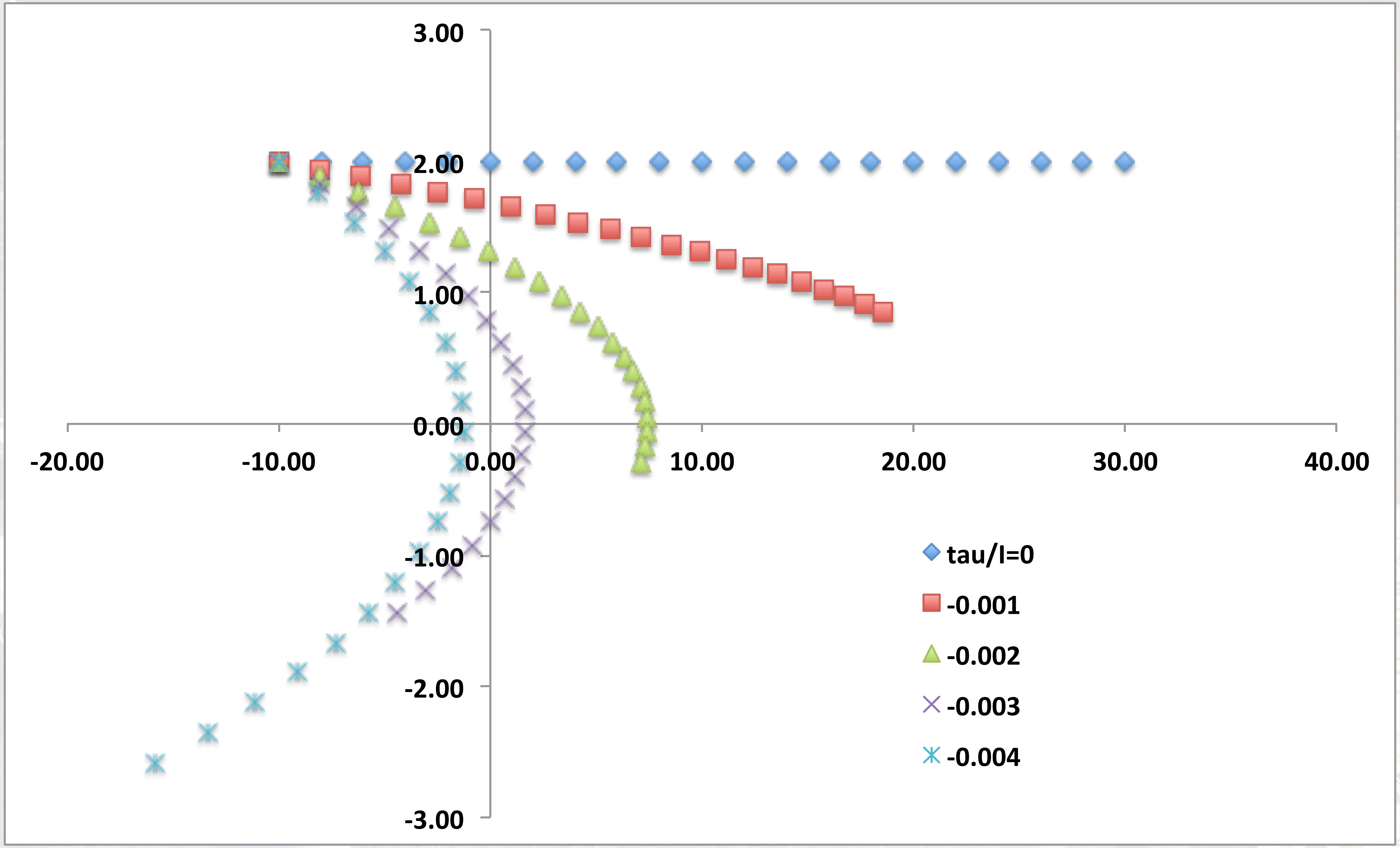
$$\text{at } t = 0,\ \dot{\theta} = \dot{\theta}_o \implies \frac{\tau}{I}t = \dot{\theta} - \dot{\theta}_o$$

$$\frac{1}{2}\frac{\tau}{I}t^2 + \dot{\theta}_o t = \theta + C_2$$

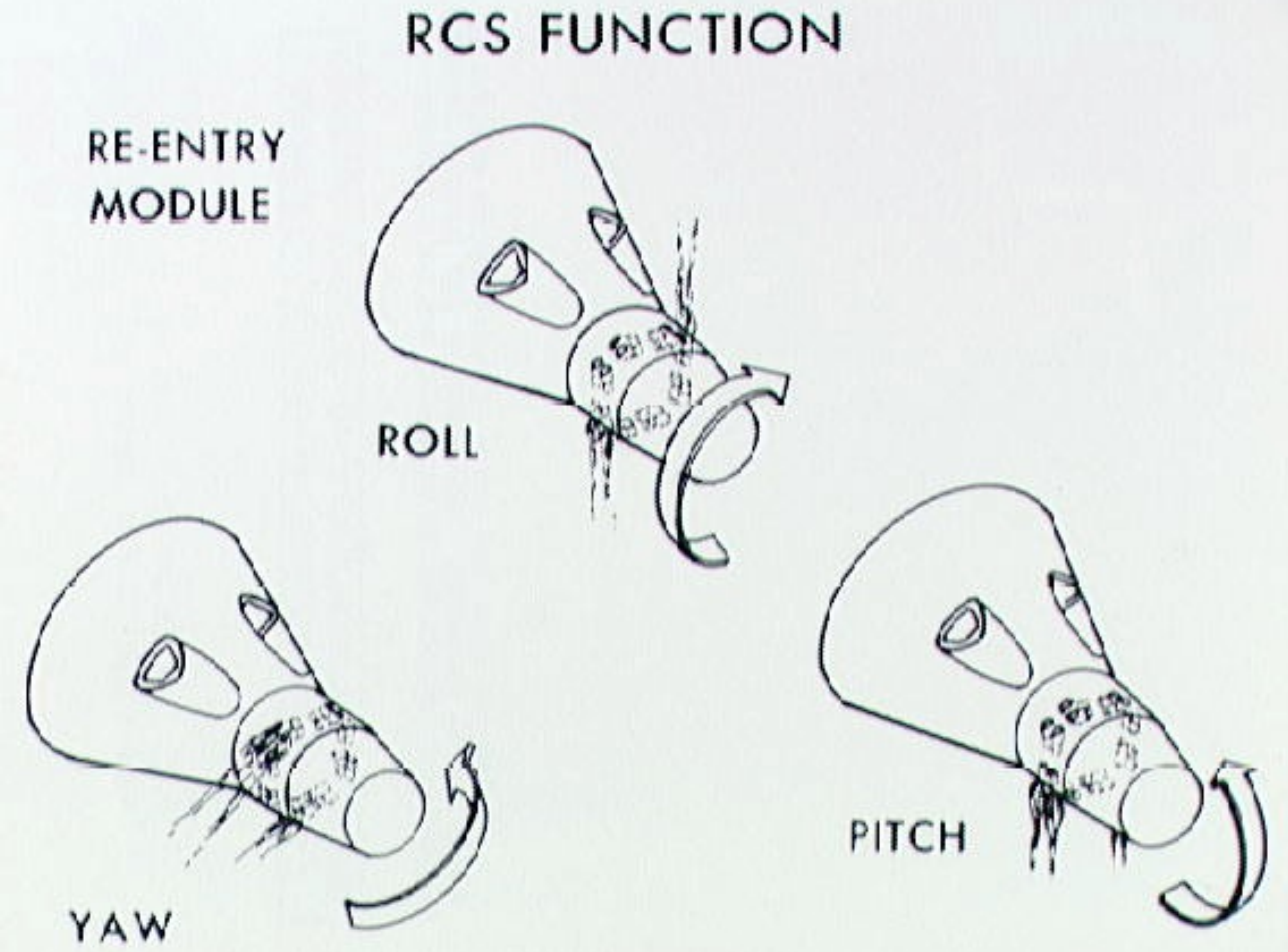
$$\text{at } t = 0,\ \theta = \theta_o \implies \frac{1}{2}\frac{\tau}{I}t^2 + \dot{\theta}_o t = \theta - \theta_o$$

$$\frac{1}{2}\left(\dot{\theta}^2 - \dot{\theta}_o{}^2\right) = \frac{\tau}{I}\left(\theta - \theta_o\right)$$

Attitude Trajectories in the Phase Plane



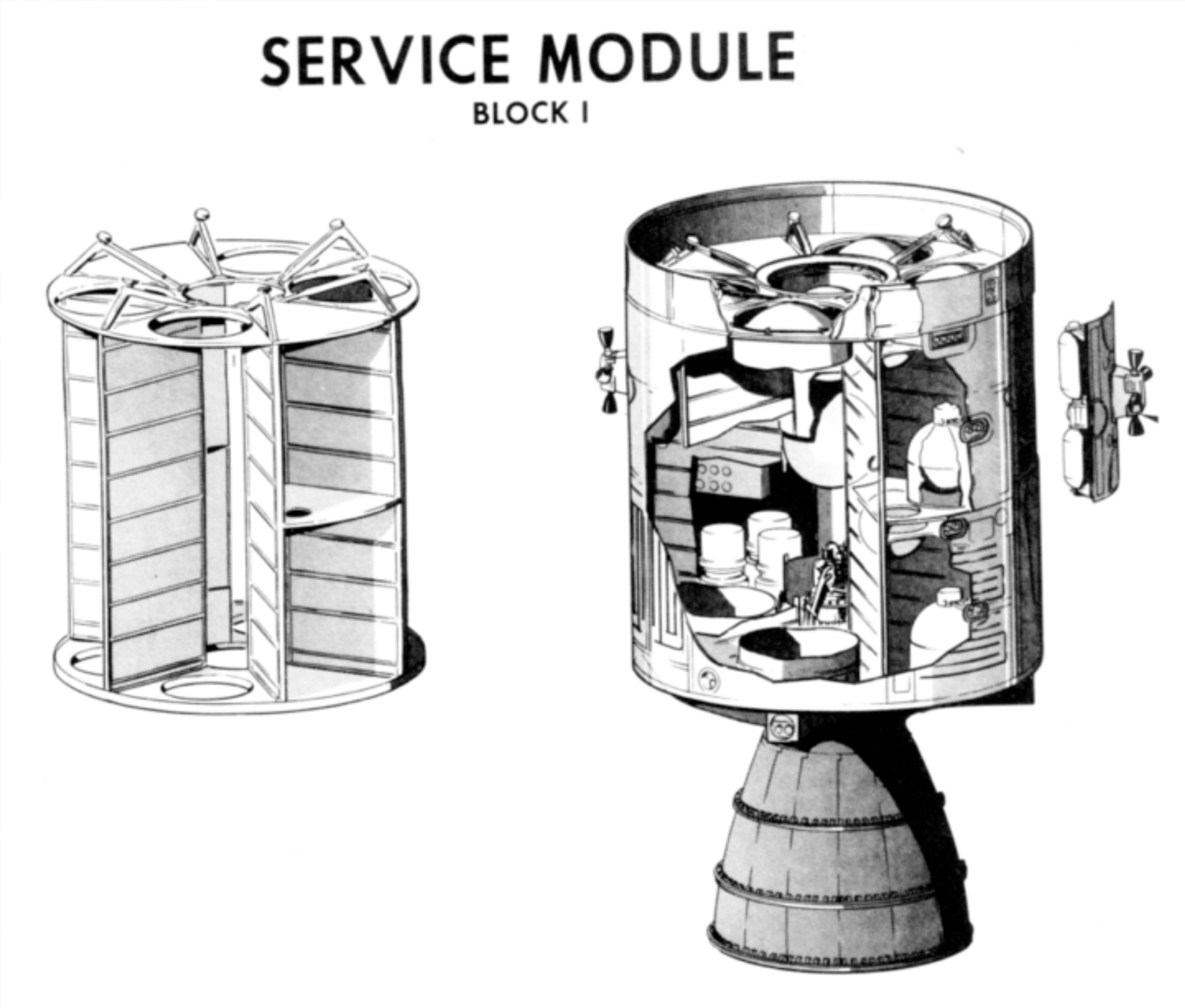
Gemini Entry Reaction Control System

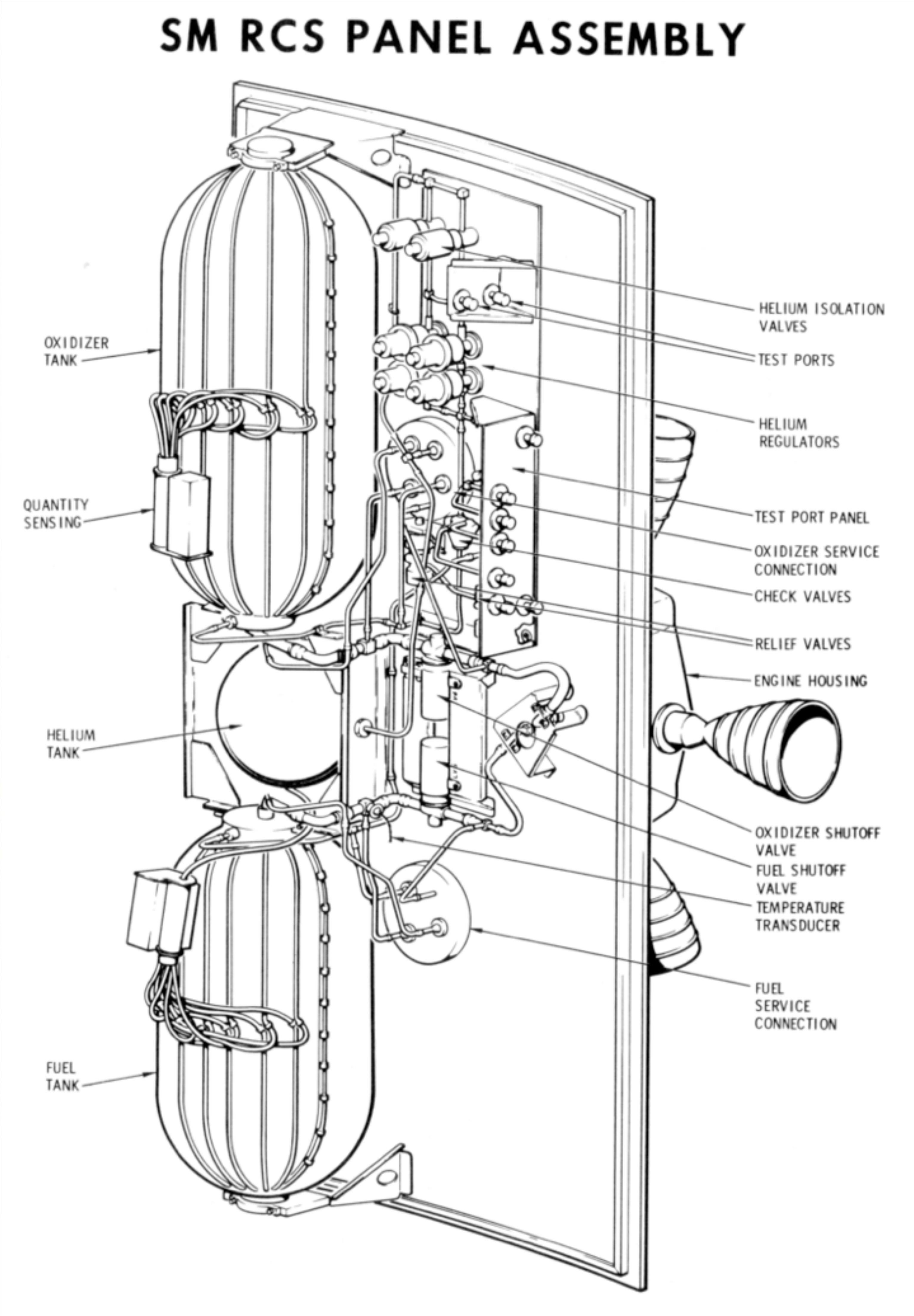


Apollo Reaction Control System Thrusters

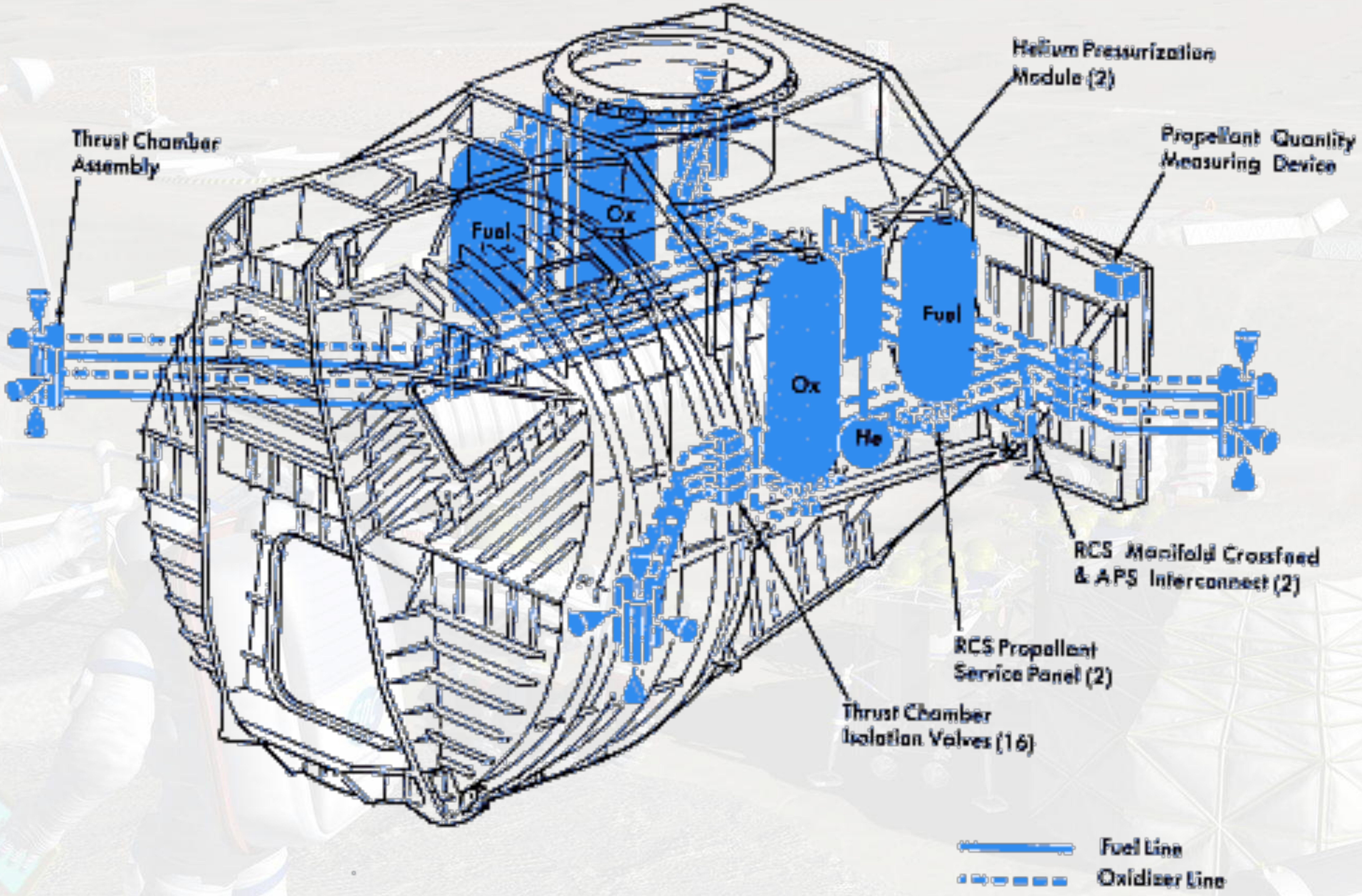
RCS Quad

Apollo CSM RCS Assembly

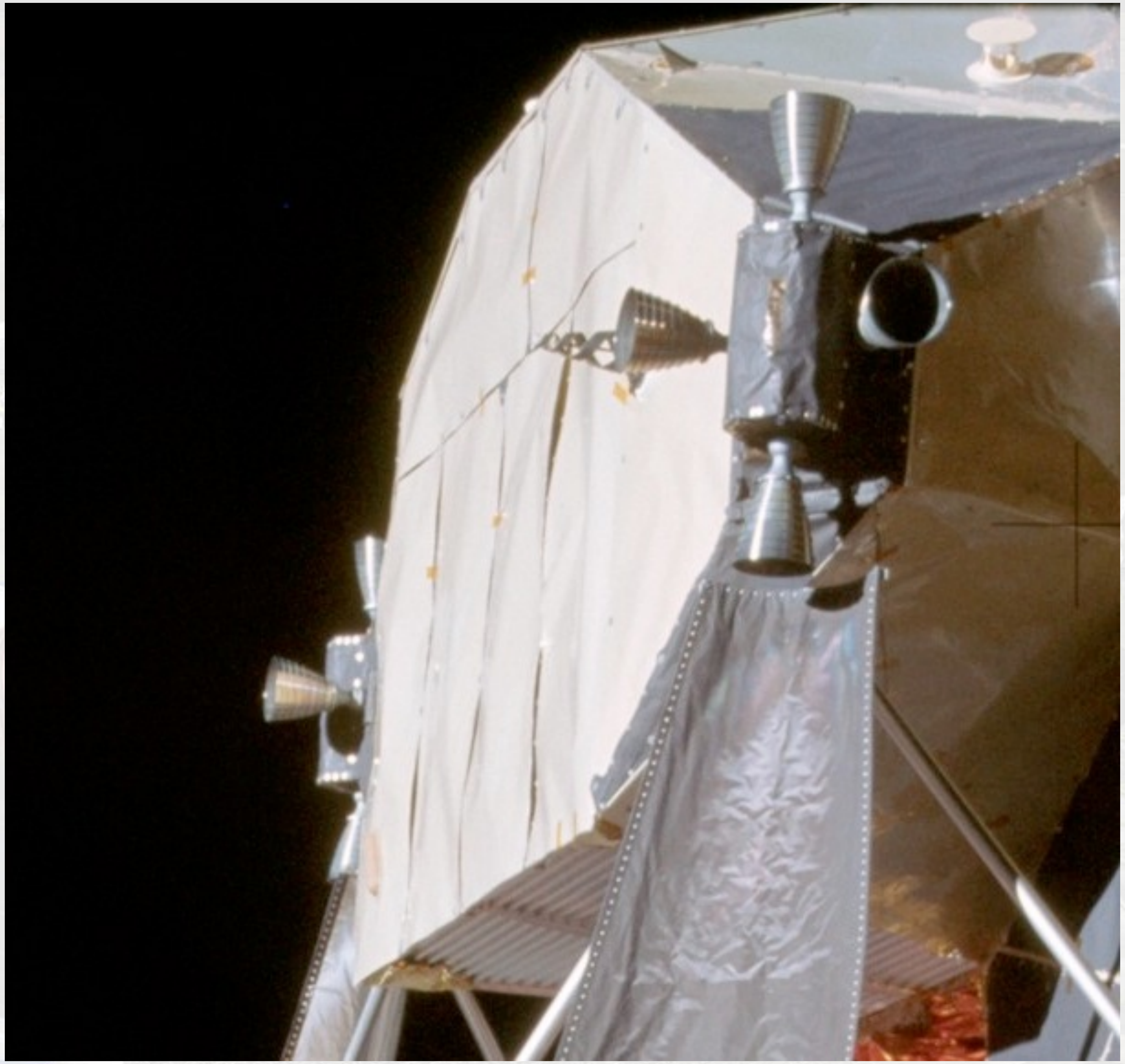




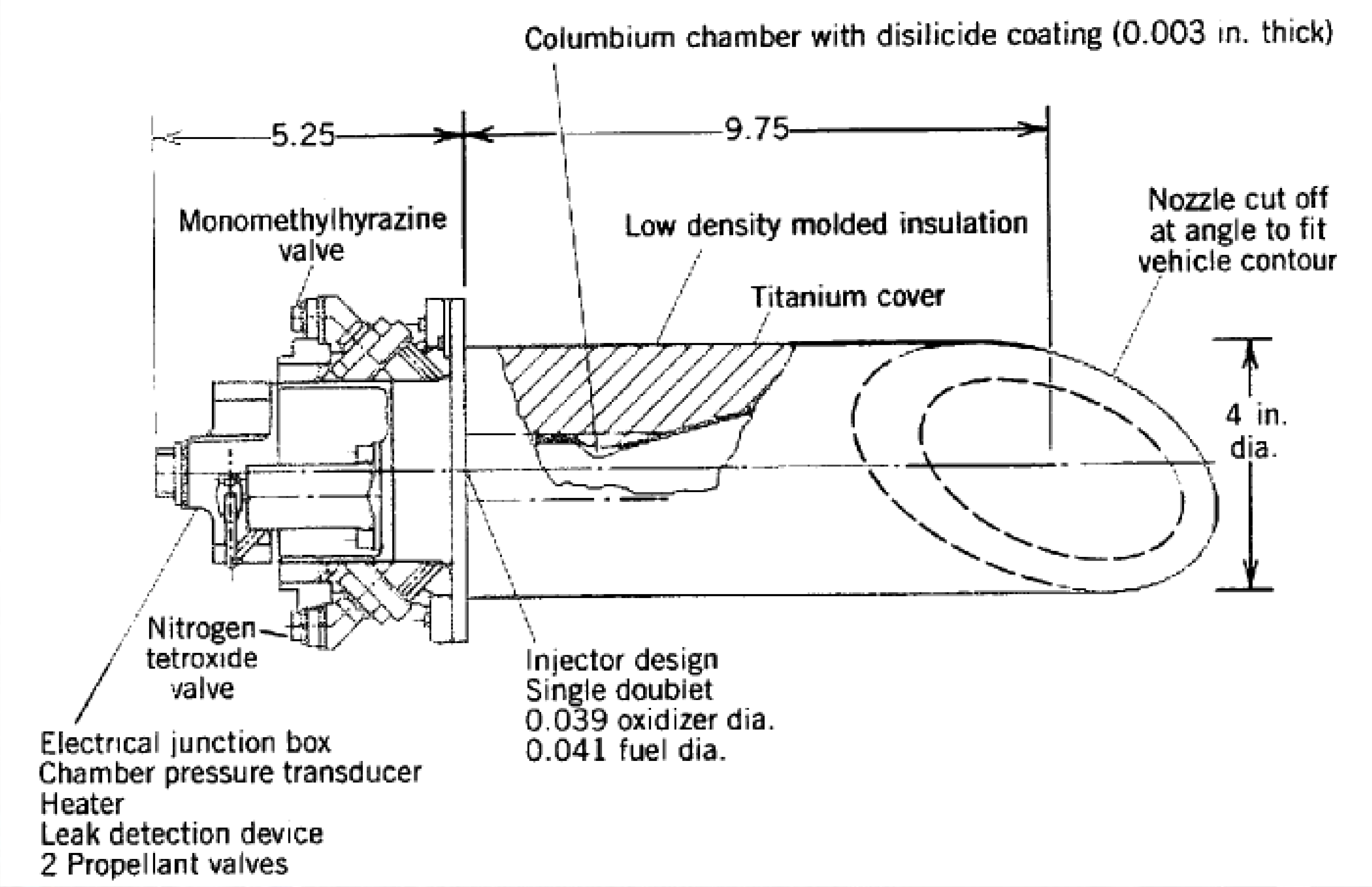
Lunar Module Reaction Control System



LM RCS Quad



Space Shuttle Primary RCS Engine



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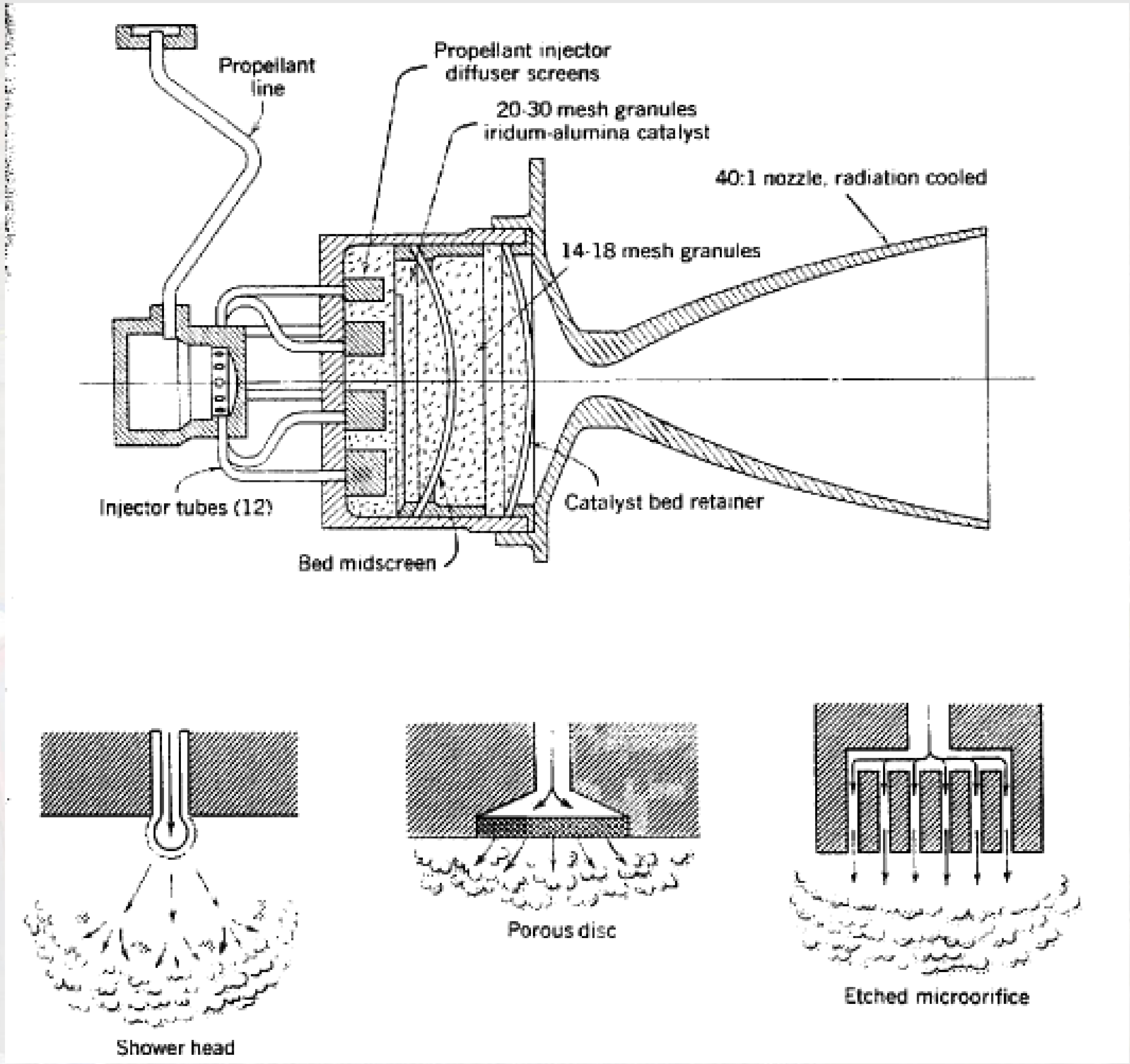
Monopropellant Engine Design



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Cold Gas Thruster Exhaust Velocity

Assume nitrogen gas thrusters

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

$$\bar{M} = 28 \qquad p_0 = 300\ psi$$

$$T_0 = 300\ K \qquad p_e = 2\ psi$$

$$\Re = 8314.3 \qquad \gamma = 1.4$$

$$V_e = \sqrt{\frac{2(1.4)}{1.4 - 1}\frac{8314.3(300)}{28}\left[1 - \left(\frac{2}{300}\right)^{\frac{1.4-1}{1.4}}\right]} = 689\ \frac{m}{sec}$$

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>

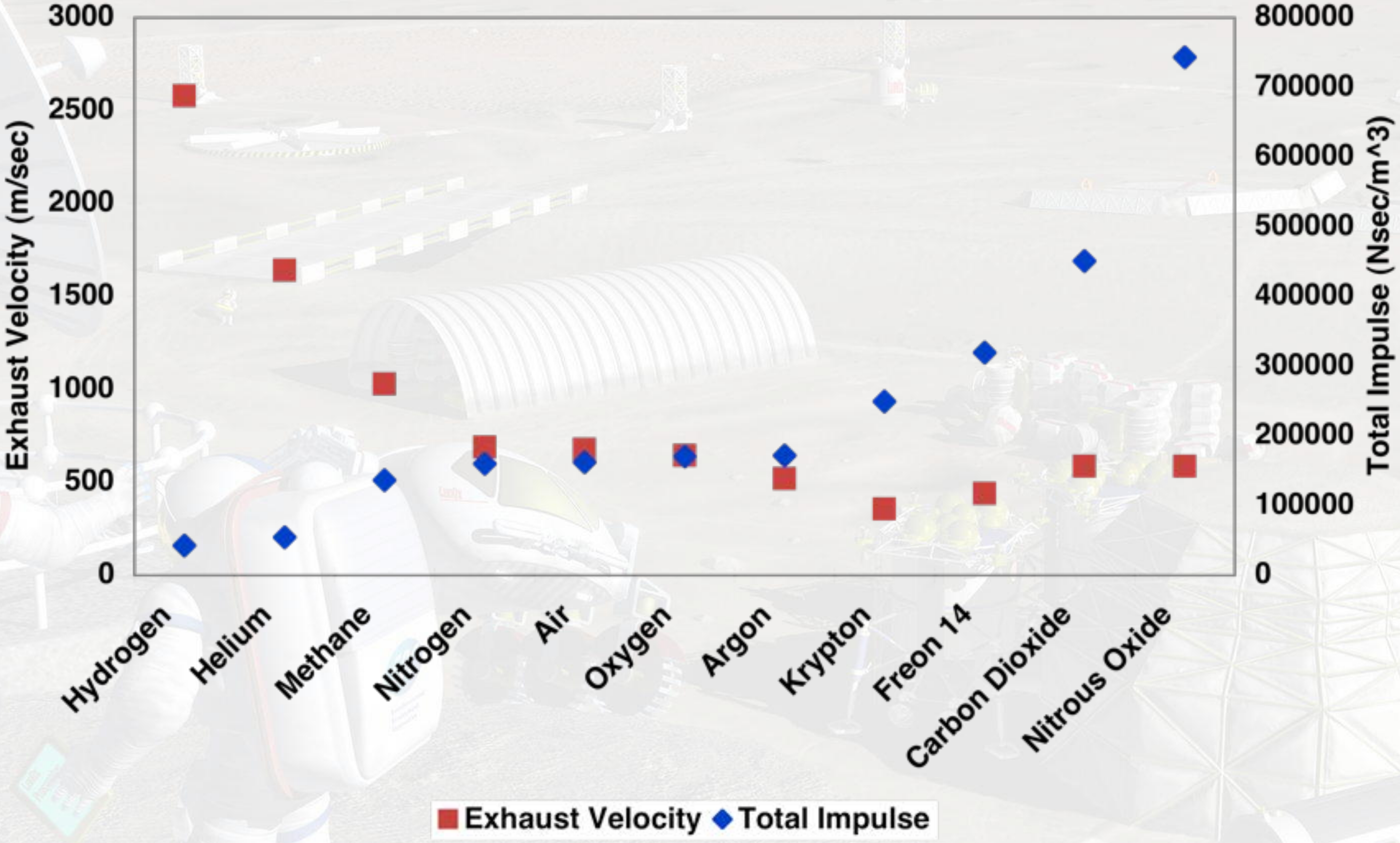
$$I_t = Tt = \dot{m} v_e t$$

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

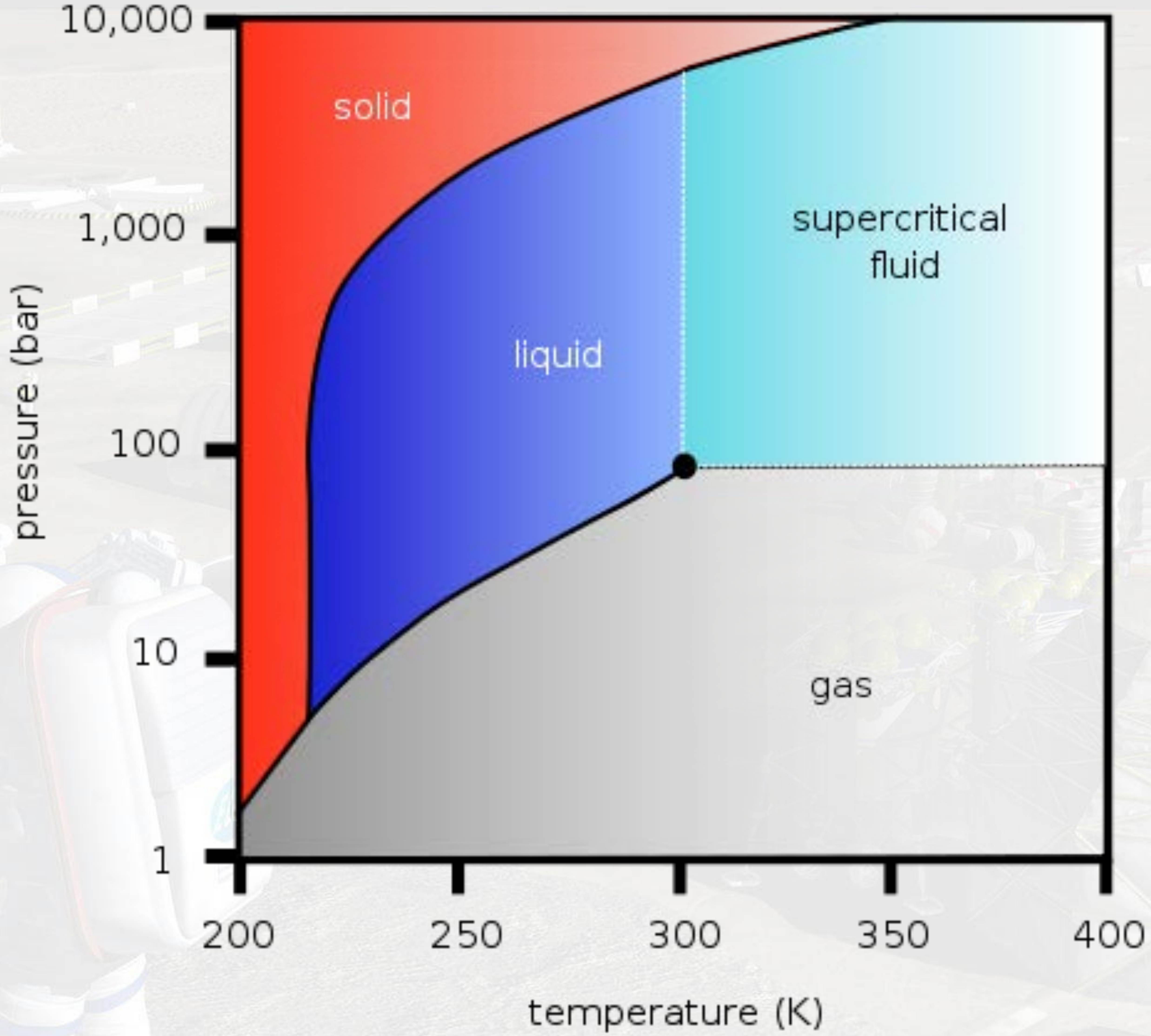
$$I_t = \rho V v_e$$

- To assess cold-gas systems, we can examine total impulse per unit volume of propellant storage $$\frac{I_t}{V} = \rho v_e$$

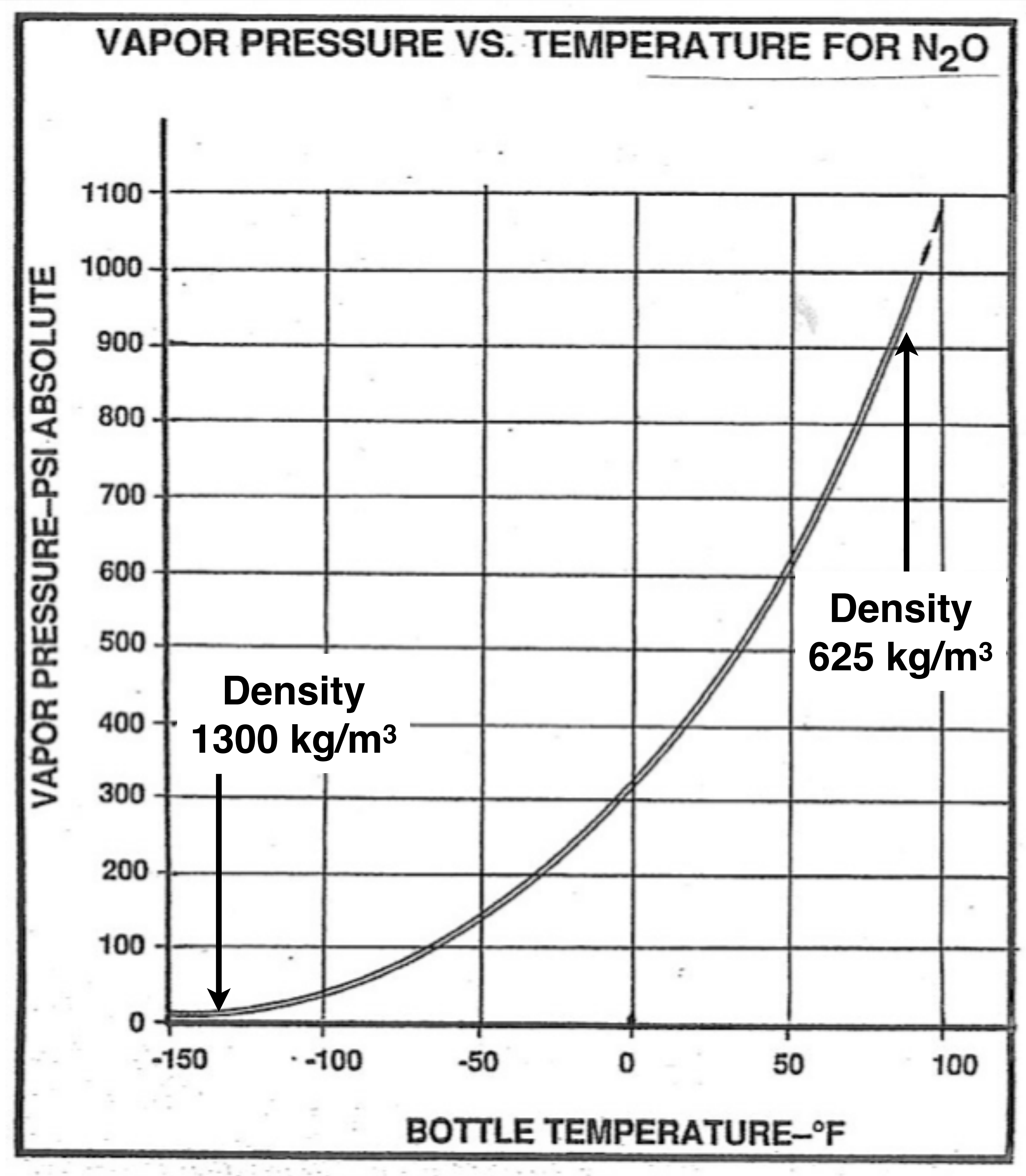
Performance of Cold-Gas Systems



Self-Pressurizing Propellants (CO_2)



Self-Pressurizing Propellants (N_2O)

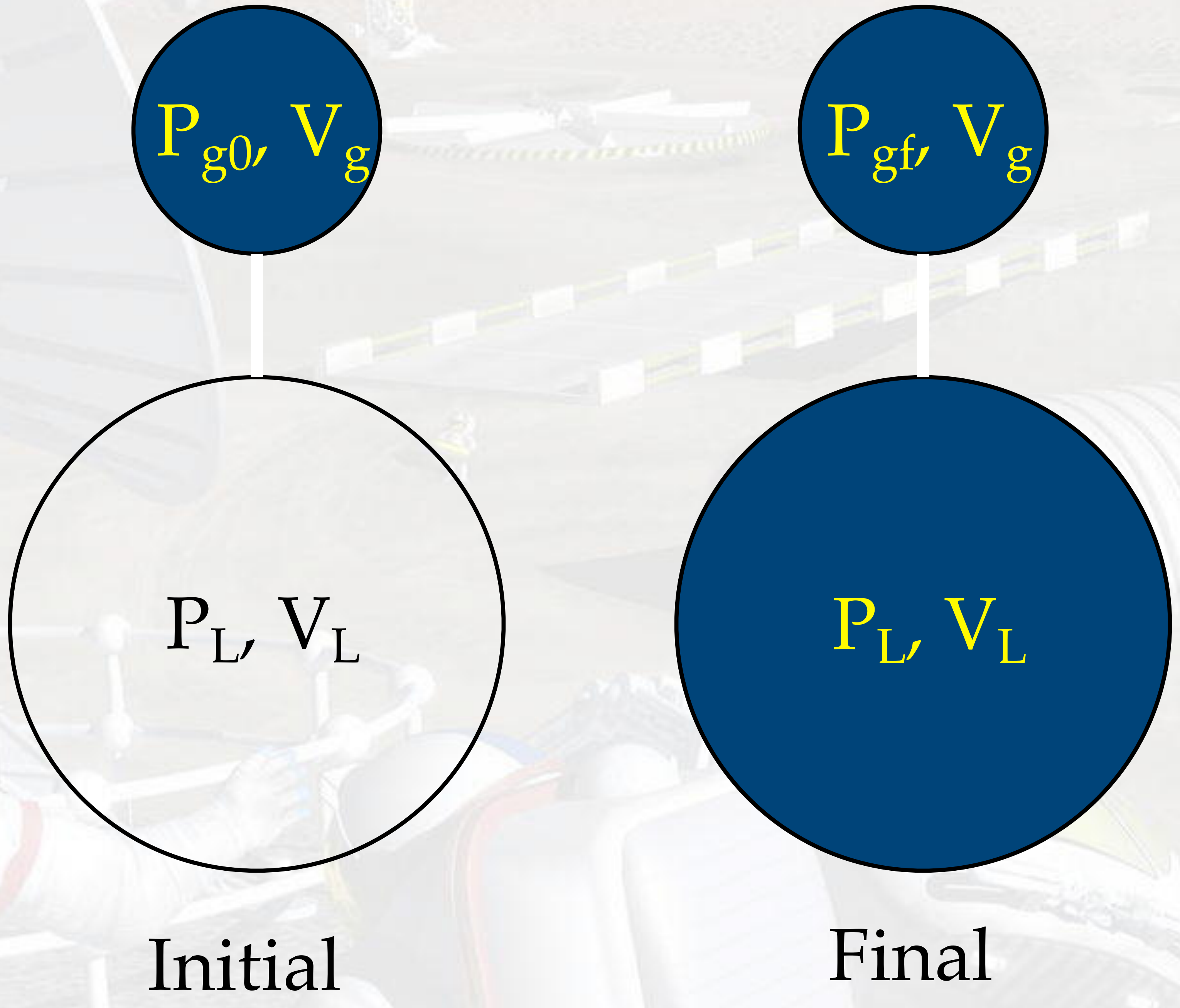


N_2O Performance Augmentation

- Nominal cold-gas exhaust velocity ~600 m/sec
- N_2O dissociates in the presence of a heated catalyst
  engine temperature ~1300°C $2N_2O \rightarrow 2N_2 + O_2$
  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=31.04 MPa
- Conversion factor 1 psi = 6892 Pa
- Ratio of specific heats for He = 1.67

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

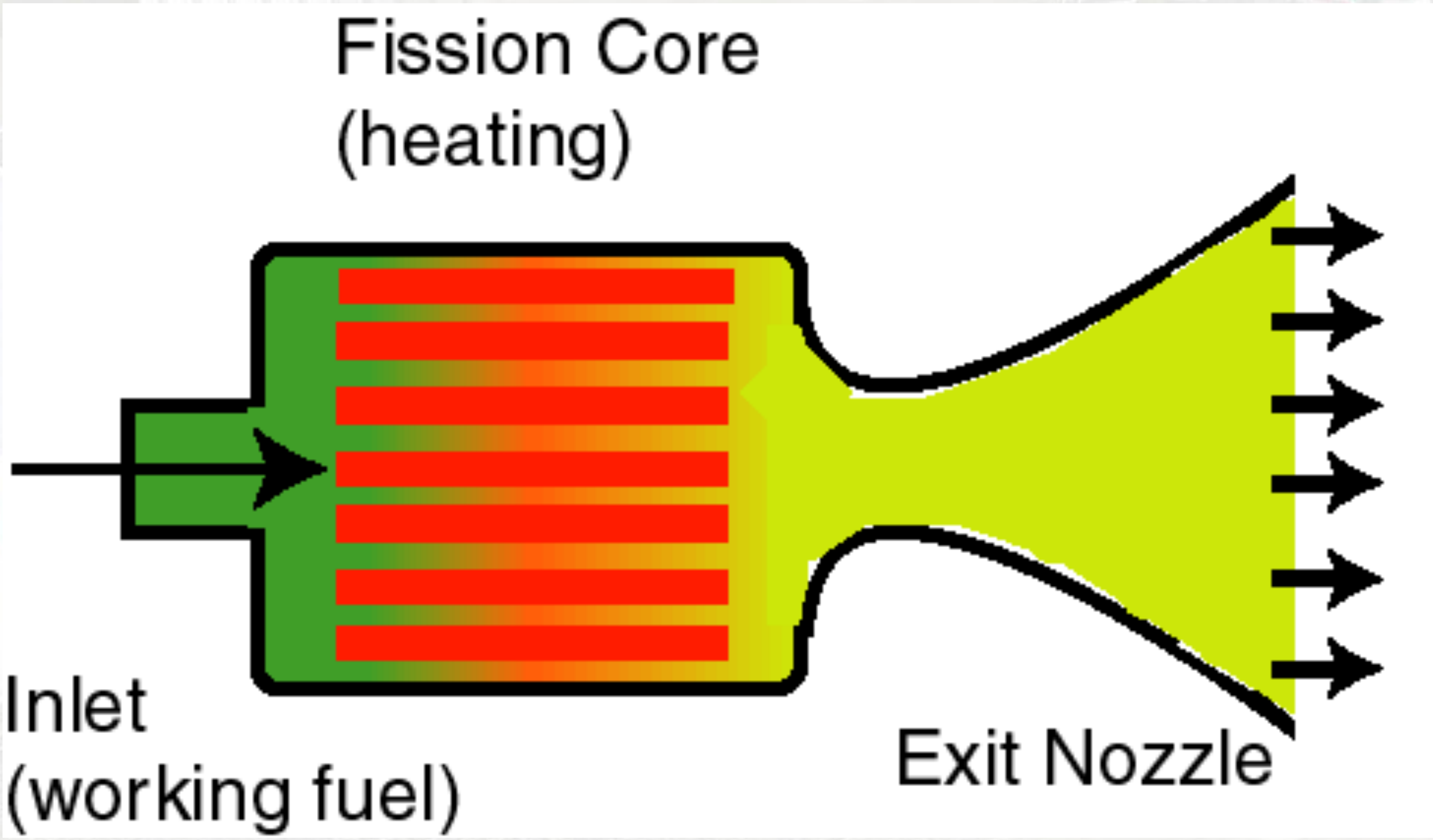
- $V_g = 3.713\ m^3$
- Ideal gas: T=300°K $\Rightarrow \rho_{He} = \dfrac{p_{g,0}\bar{M}}{\Re T_0}$ =49.7 kg/m^3 $\Rightarrow M_{He}$=185.1 kg

Autogenous Pressurization

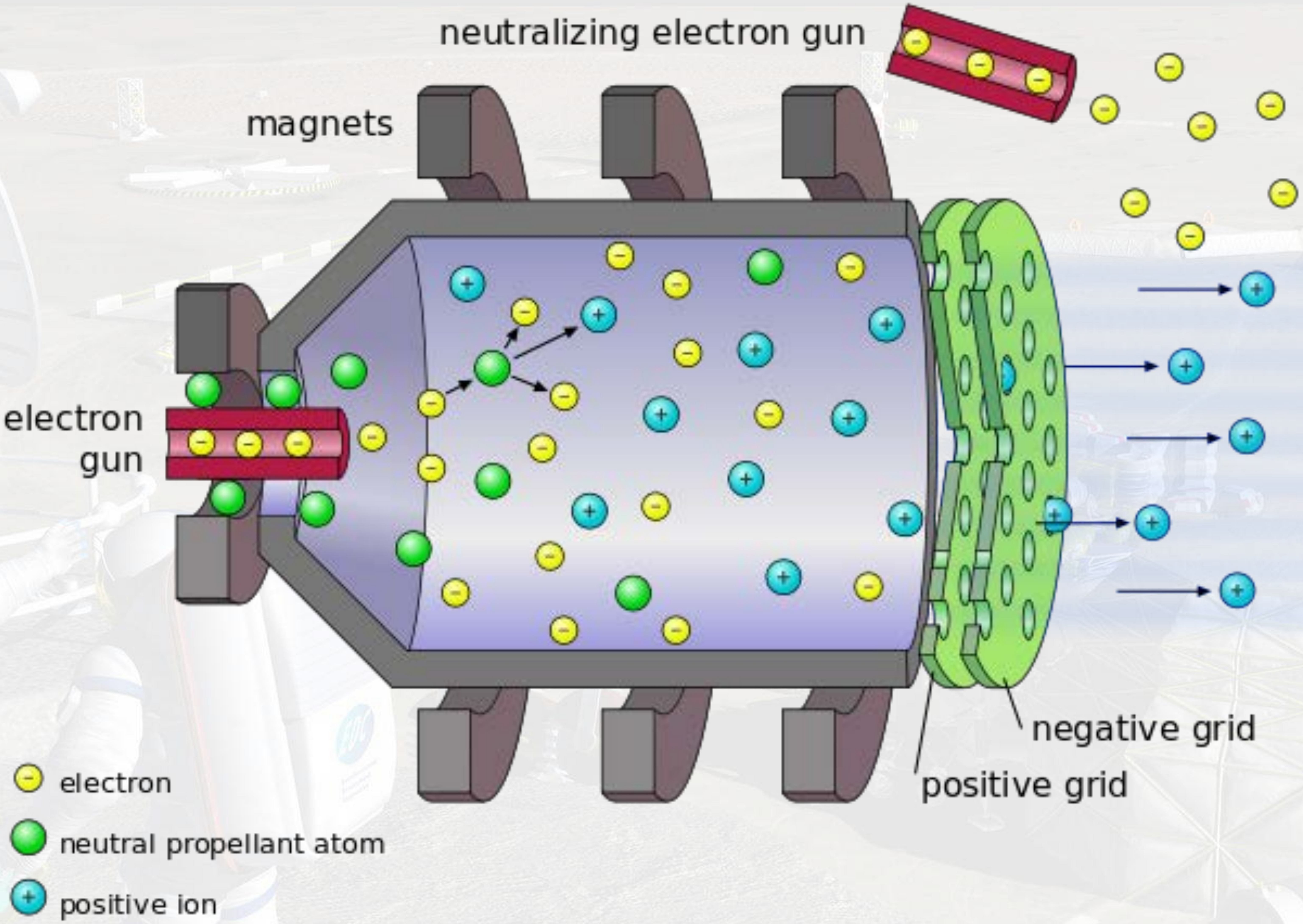
- Use gaseous propellants to pressurize tanks with liquid propellants
- Heat exchanger to gasify and warm propellants, then route back into ullage volume
- Eliminates need for pressurized gases for ullage and high-pressure storage bottles (e.g., Falcon 9 failures)
- Issue: start-up transient

Nuclear Thermal Rockets

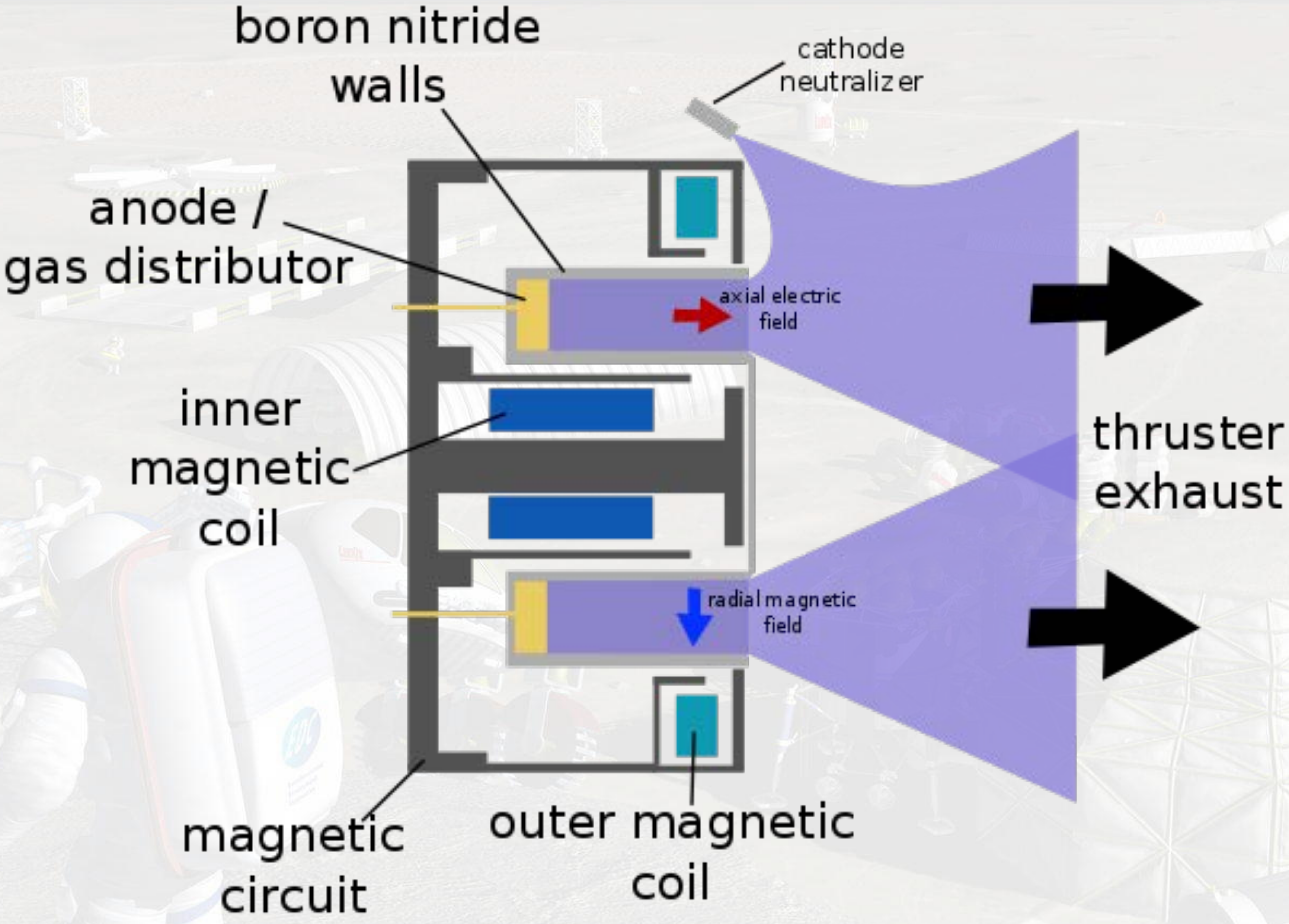
- Heat propellants by passing through nuclear reactor
- Isp limited by temperature limits on reactor elements (~900 sec for H2 propellant)
- Mass impacts of reactor, shielding
- High thrust system



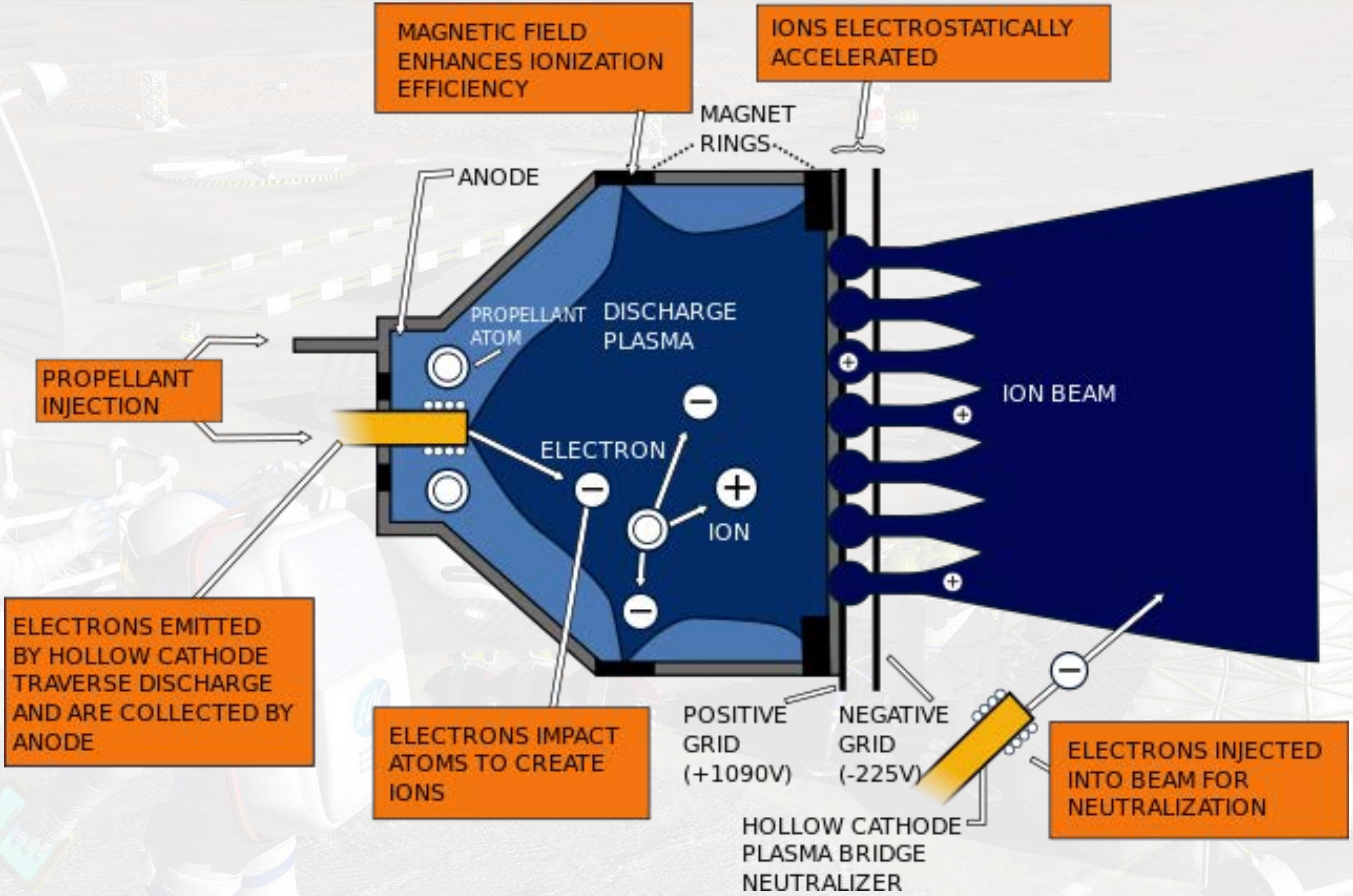
Electrostatic Ion Thruster



Hall-Effect Thruster



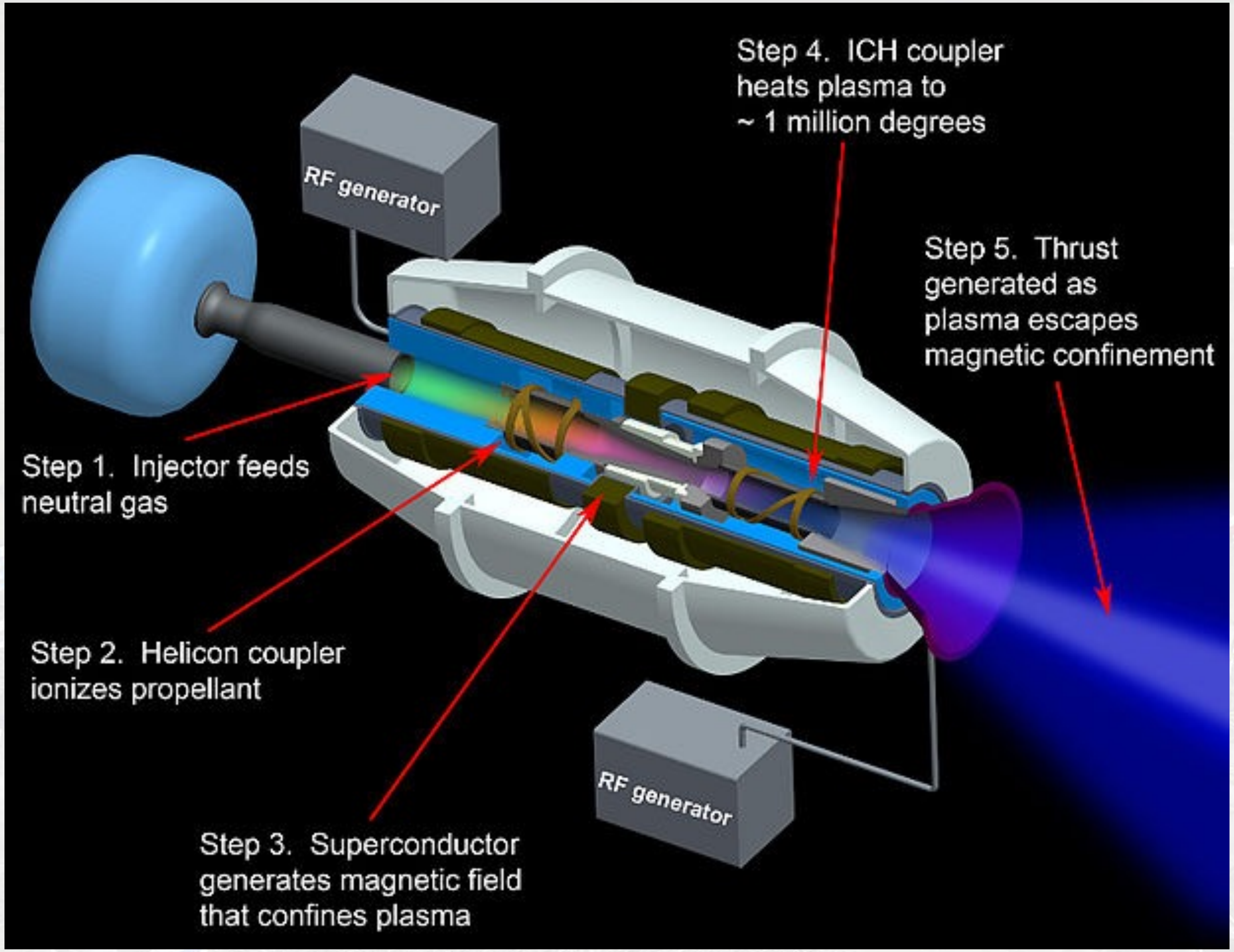
Ion Engine Schematic



Ion Engines Existing/In Development

- NSTAR (NASA Solar Technology Application Readiness)
  - DS1 and Dawn
  - 30 cm, 2.3 kW, 92 mN, 3120 sec
- NEXT (NASA Evolutionary Xenon Thruster)
  - Available 2019
  - 6.9 kW power, 236 mN thrust, Isp 4190 sec
- HiPEP (High Power Electric Propulsion)
  - TRL 3

Variable Specific Impulse Magnetoplasma Rocket



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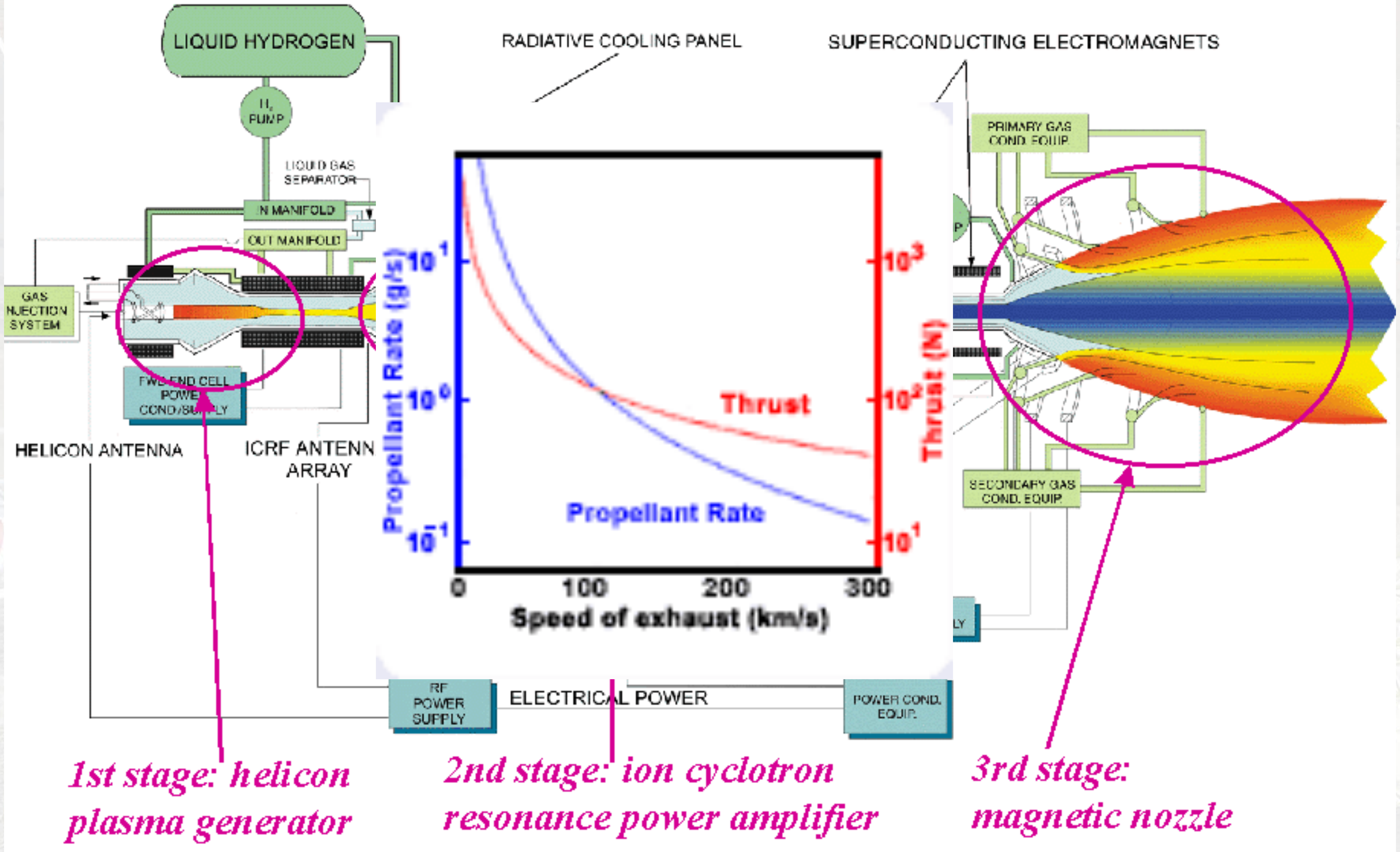
VASIMR Engine Concept



VASIMR Operating Specifications

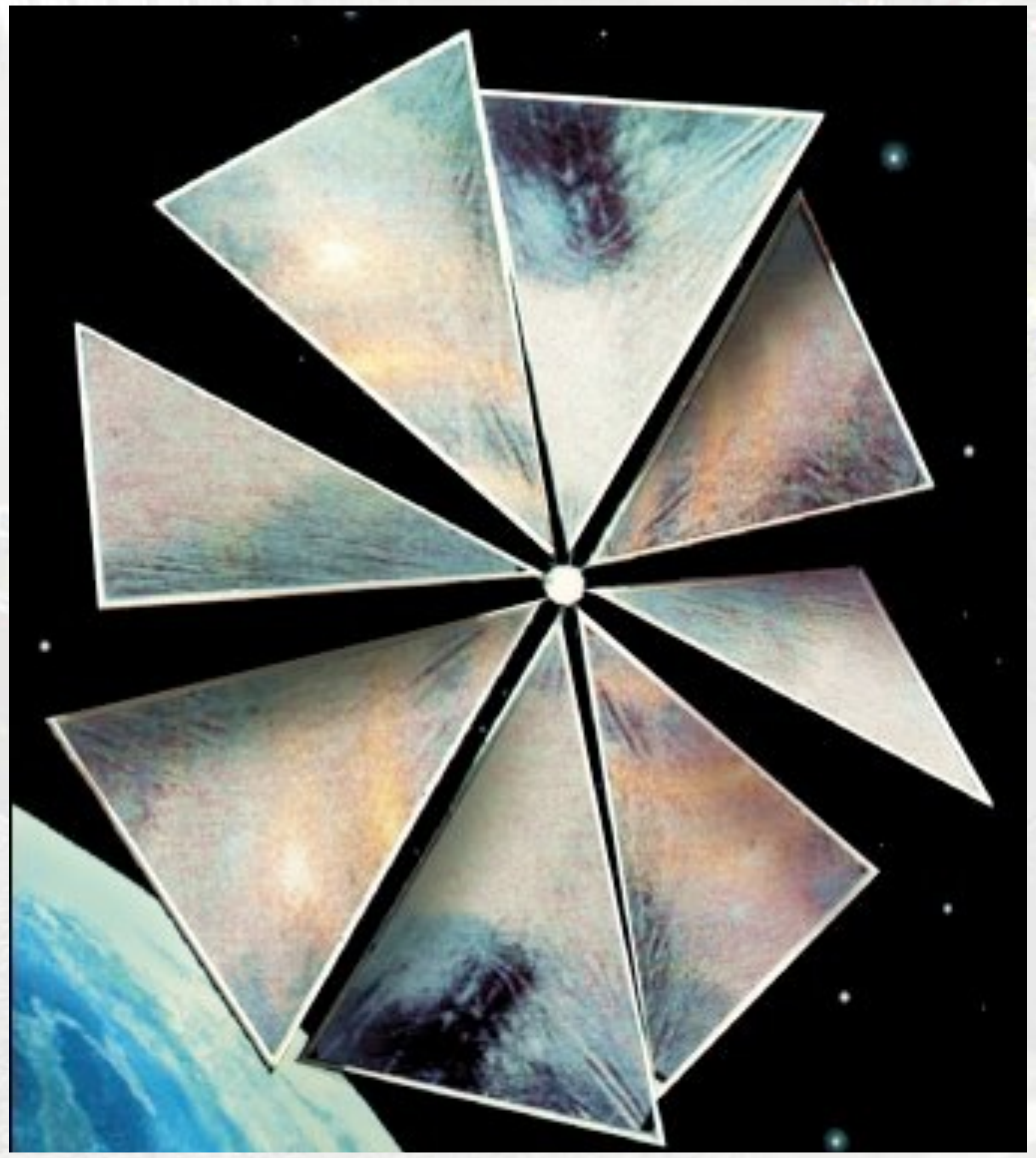
- Optimum operating point
  - Isp 5000 sec
  - Thrust 5.7 N
  - Power 200 kW
- Can be derated for higher thrust at lower Isp
- Compare to ion engines at equivalent power
  - 87 NSTAR thrusters: 8 N, 3120 sec Isp
  - 29 NEXT thrusters: 6.8 N, 4190 sec Isp

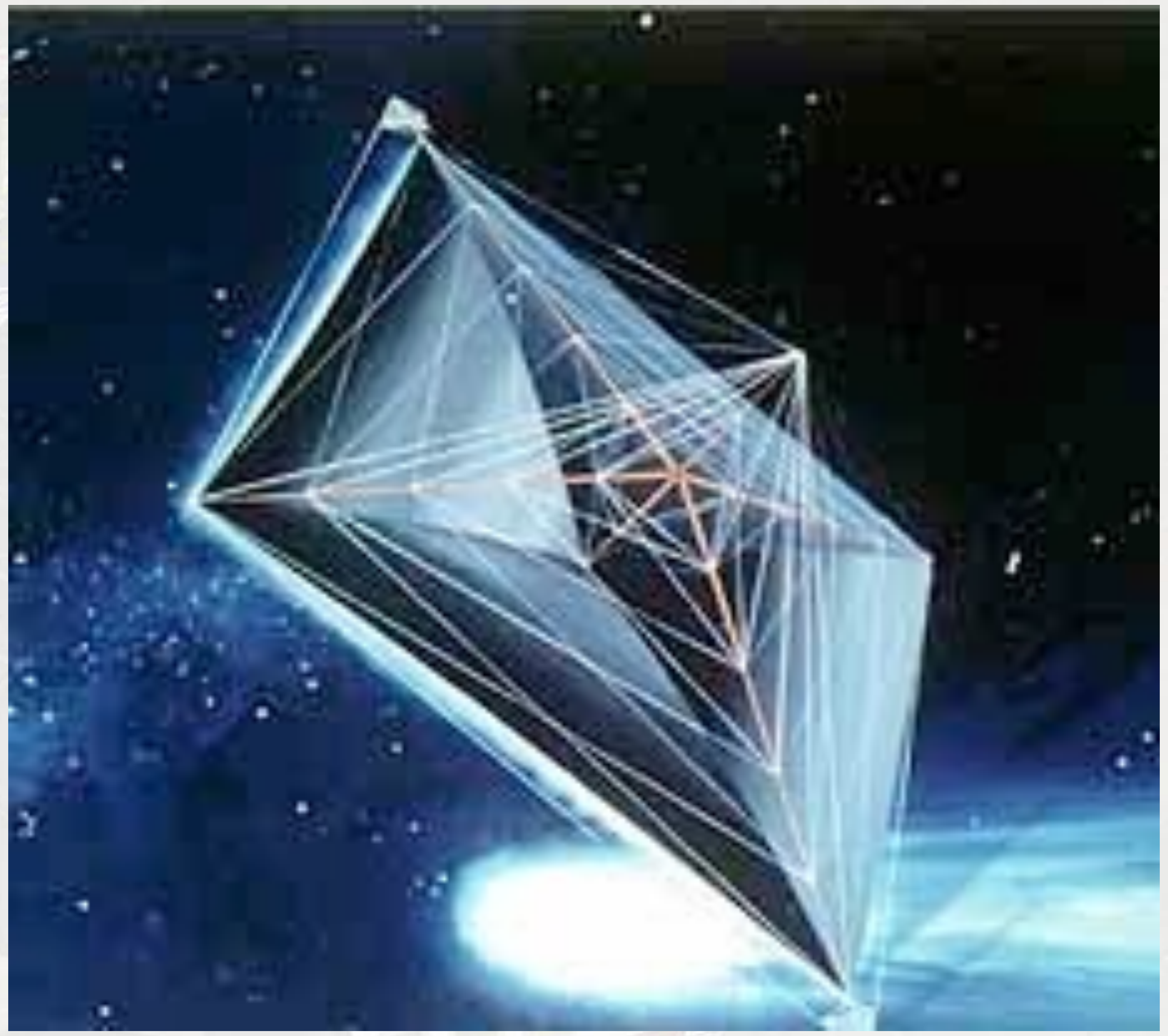
Solar Sails

- Sunlight reflecting off sail produces momentum transfer

$$E = mc^2 \Rightarrow m = \frac{E}{c^2} \Rightarrow \dot{m} = \frac{E}{t}\frac{1}{c^2} = \frac{P}{c^2}$$

$$T = 2\dot{m}V = 2\dot{m}c$$





- At 1 AU, P=1394 W/m^2
- c=3x10^8 m/sec
- T=9x10^{-6} N/m^2

Propulsion Taxonomy