# Propulsion Systems Design

- Rocket engine basics
- Solid rocket motors
- Liquid rocket engines
  - Monopropellants
  - Bipropellants
  - Propellant feed systems
- Hybrid rocket engines
- Auxiliary propulsion systems

# 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

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

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

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

# 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

$$T = \dot{m}c \Rightarrow c = V_e + \left(p_e - p_{amb}\right)\frac{A_e}{\dot{m}} \qquad \left(I_{sp} = \frac{c}{g_0}\right)$$

- Expansion ratio

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

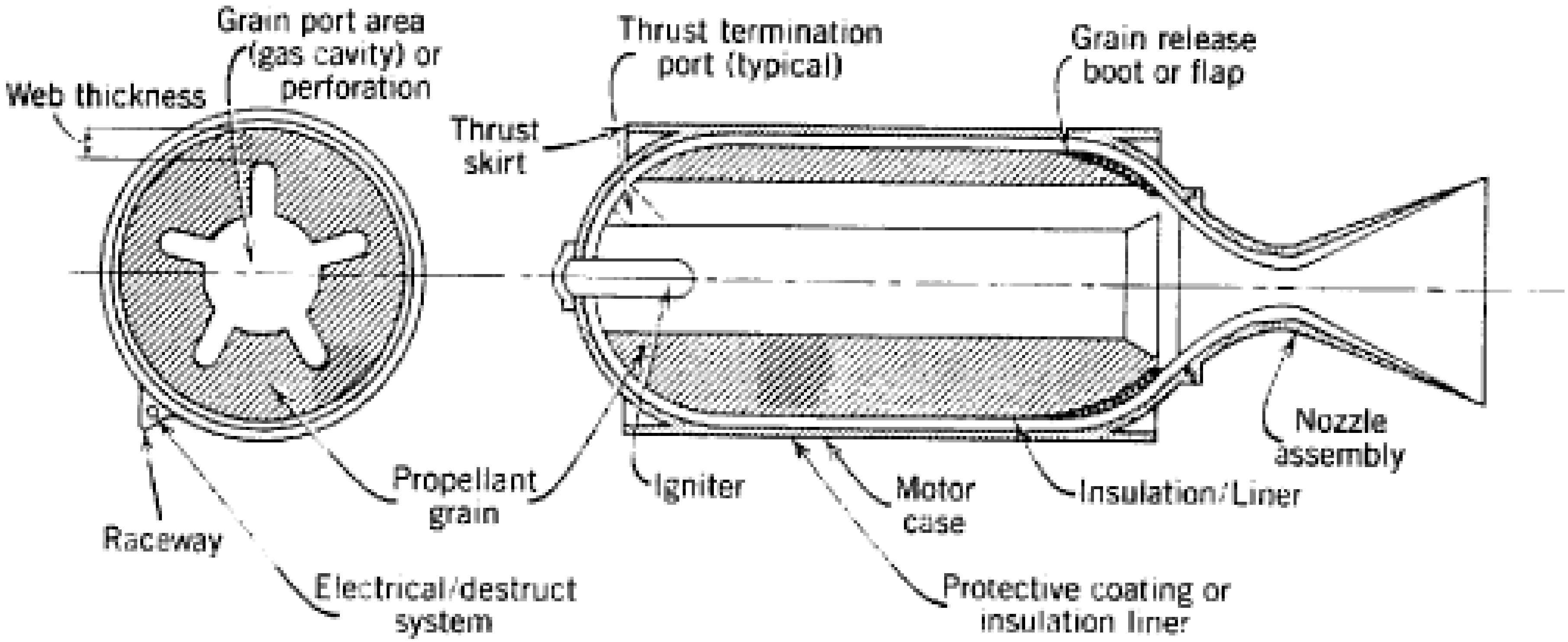
# Nozzle Design

- 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 $V_e$

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

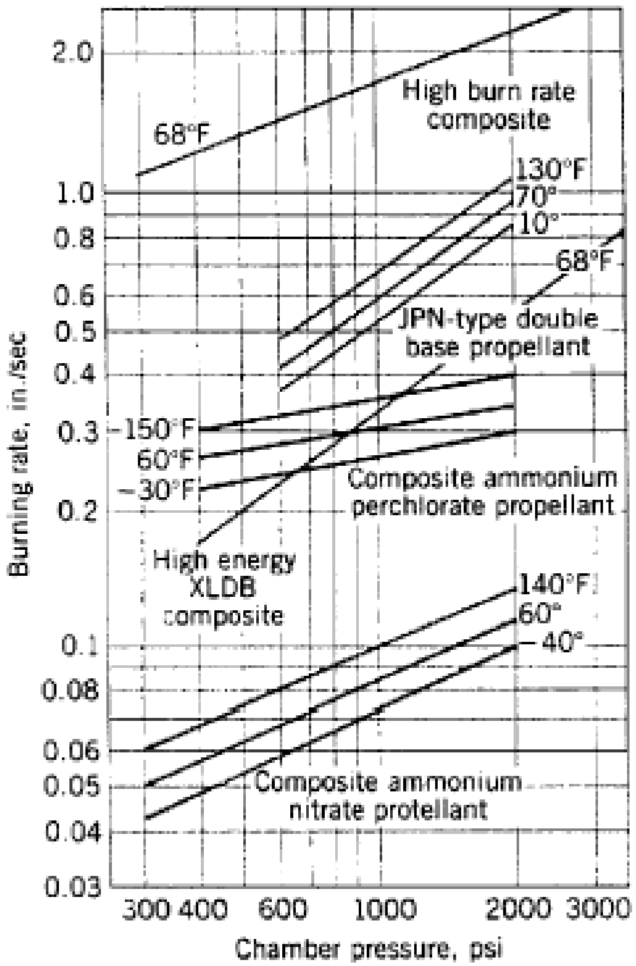
- $I_{sp,vacuum}$=455 sec --> $I_{sp,sl}$=333 sec

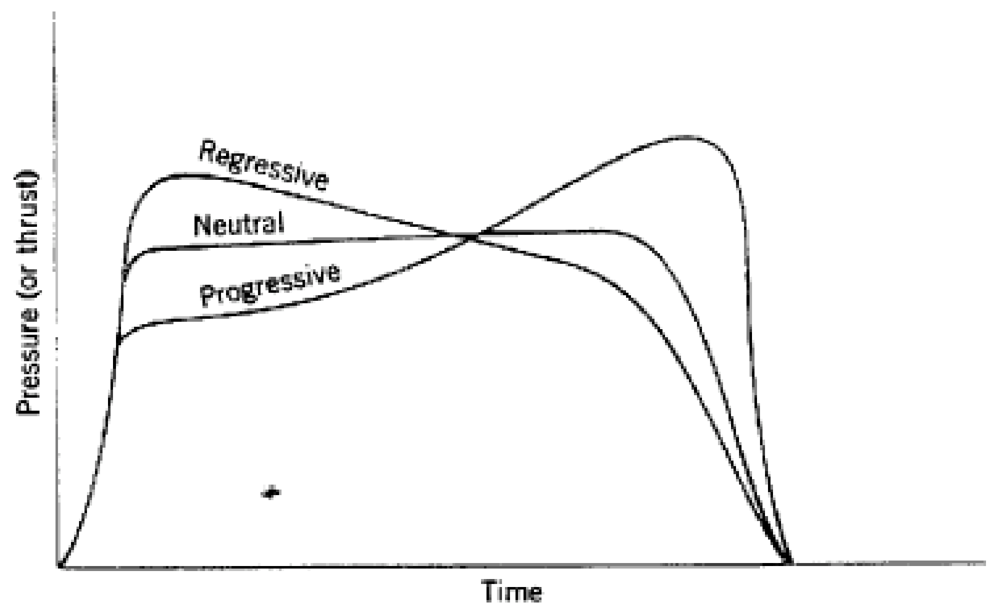
# Solid Rocket Motor




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

# Solid Propellant Combustion Characteristics

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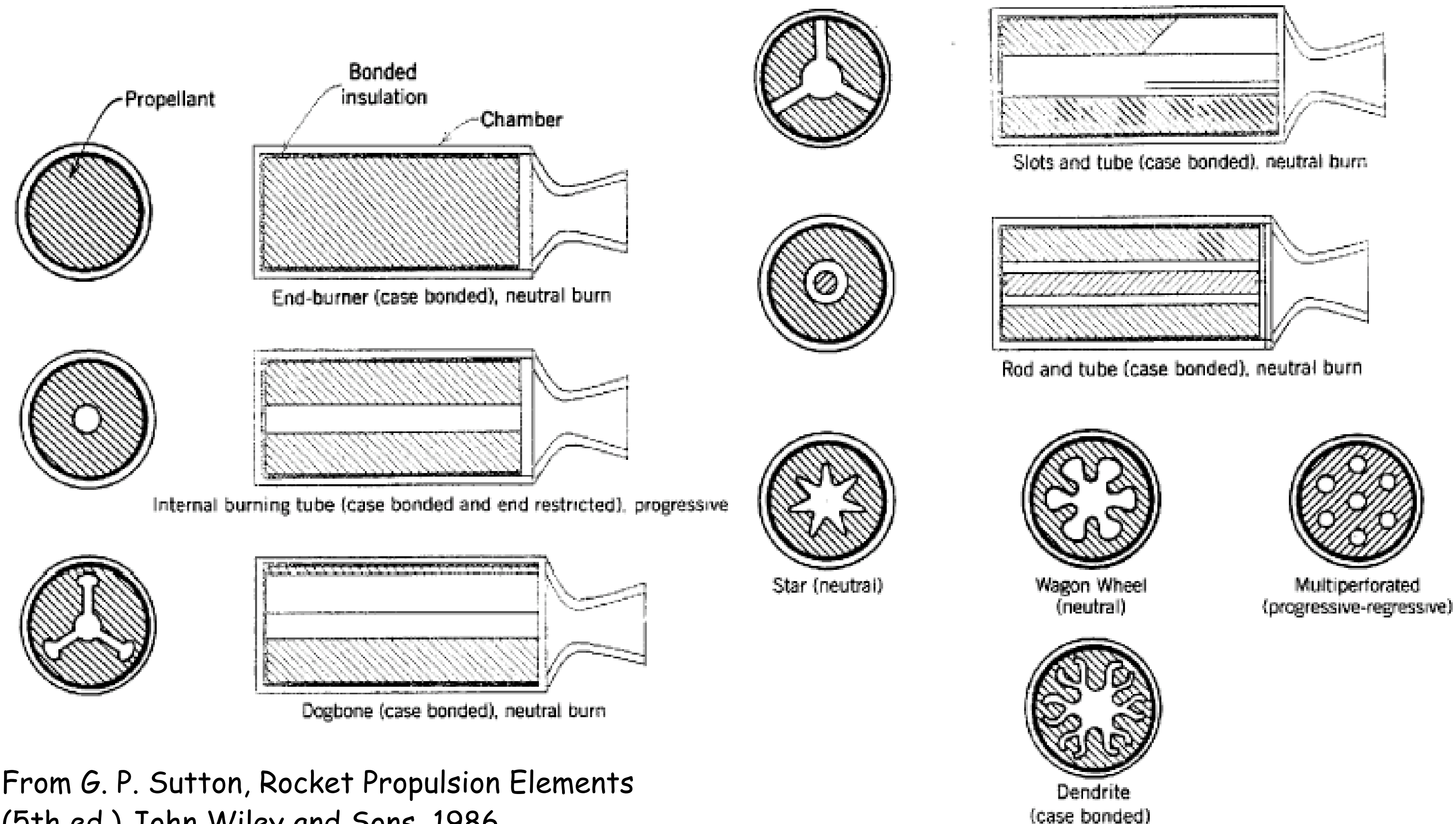
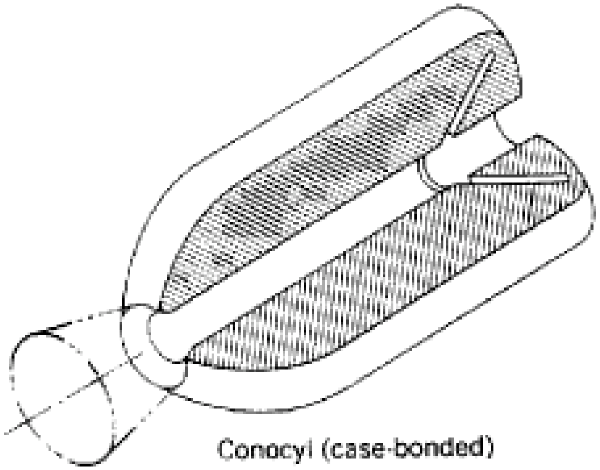
# Solid Grain Configurations



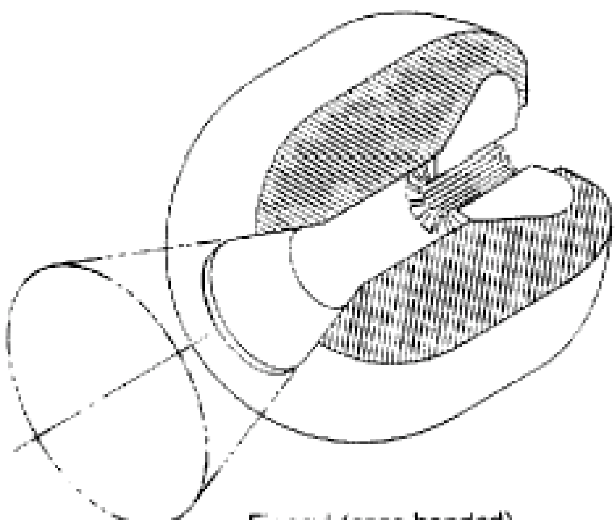

From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

# Short-Grain Solid Configurations

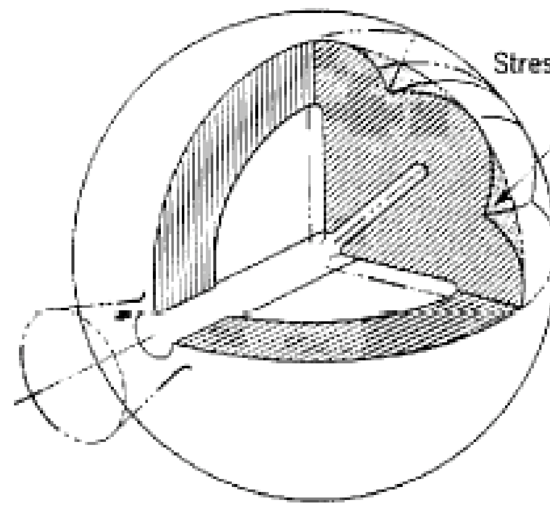


Conocyl (case-bonded)

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



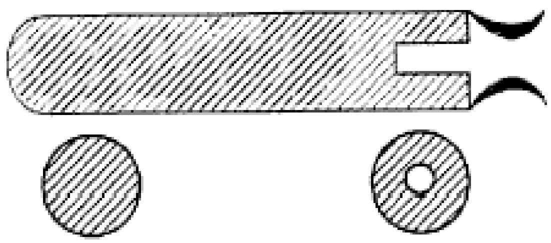
Finocyl (case-bonded)

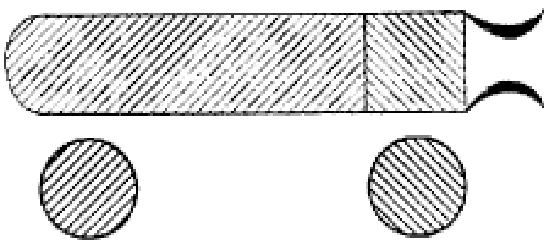


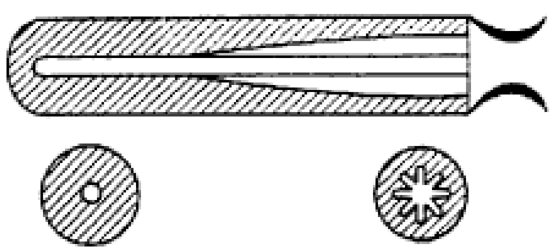

Spherical (case-bonded) with slots and cylinder

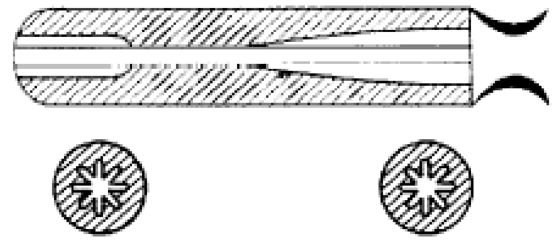
# Advanced Grain Configurations

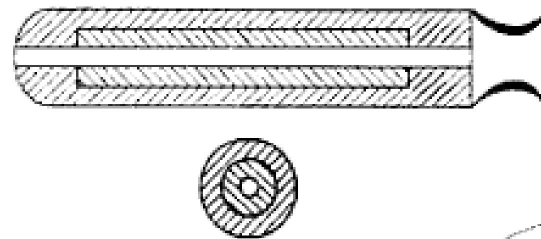
From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

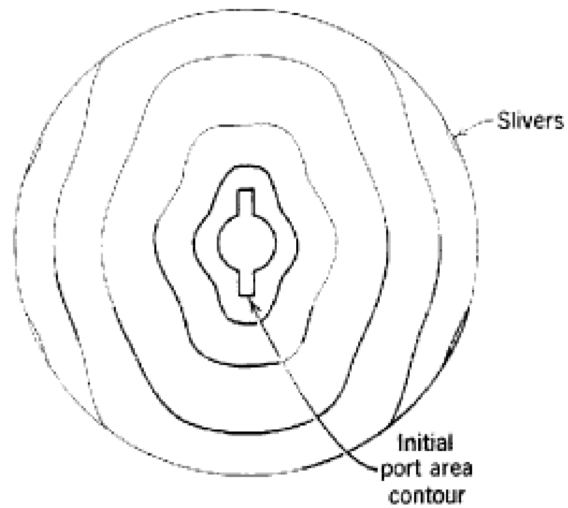


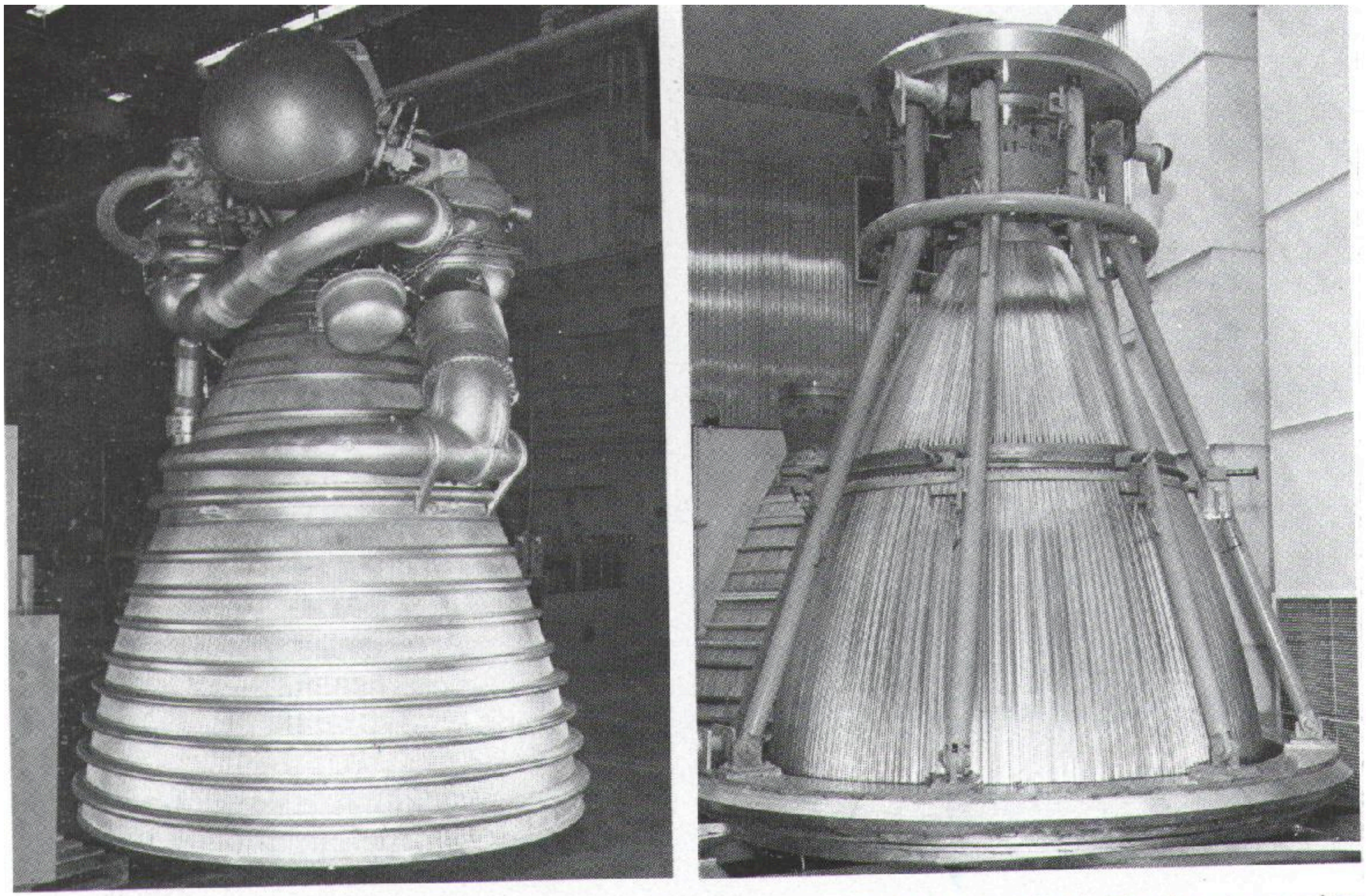



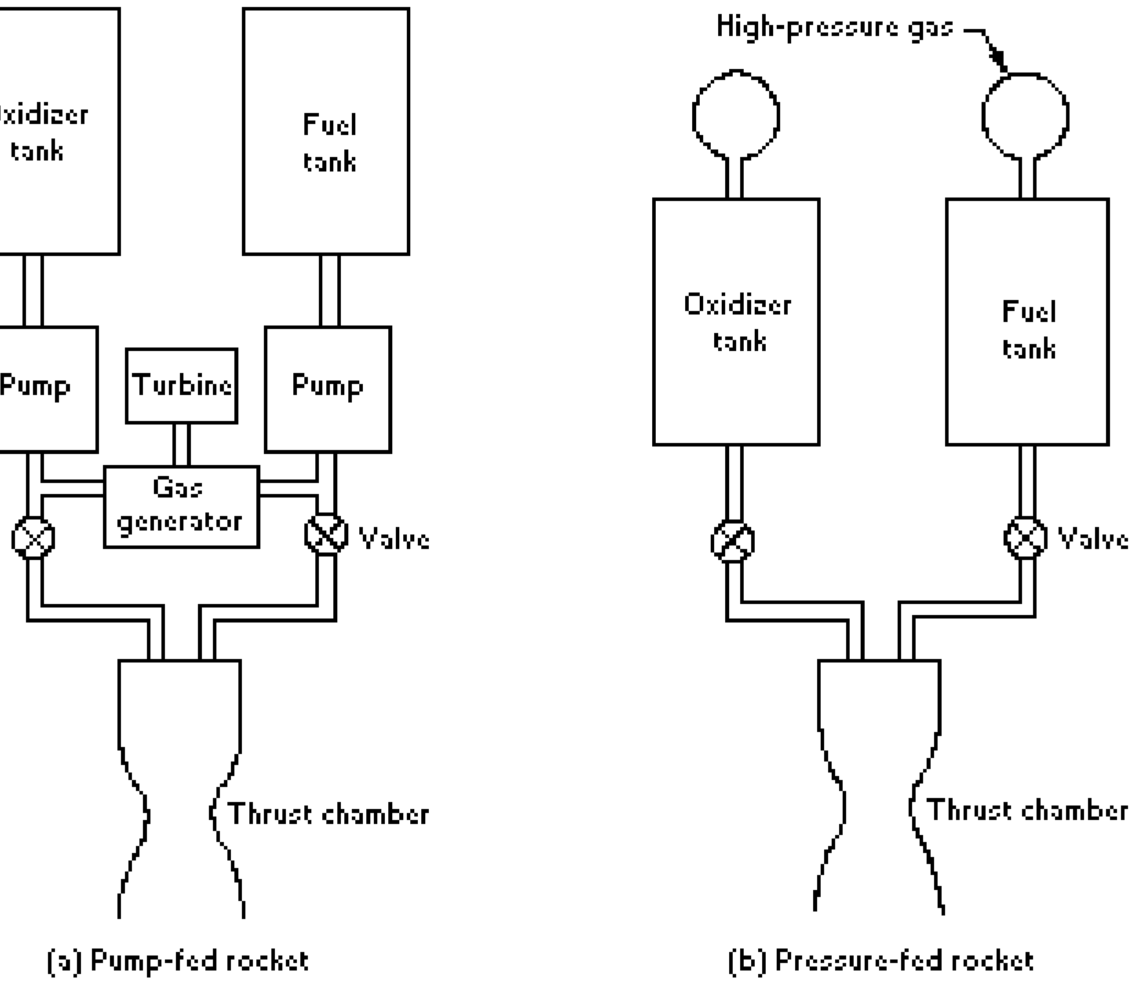



# Liquid Rocket Engine



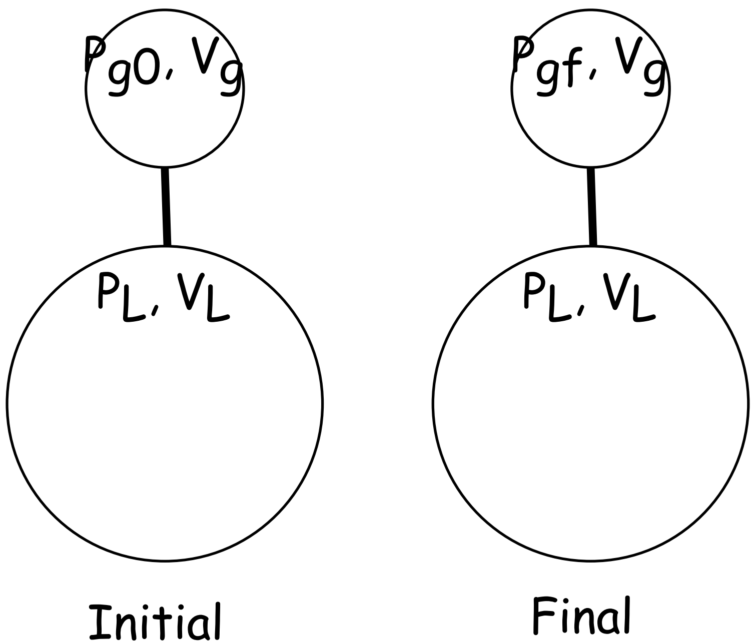
A completed J-2 rocket engine (left), with its pumps and lines installed. The basic engine structure is built up from a series of hollow tubes (right).

# Liquid Propellant Feed Systems



# 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

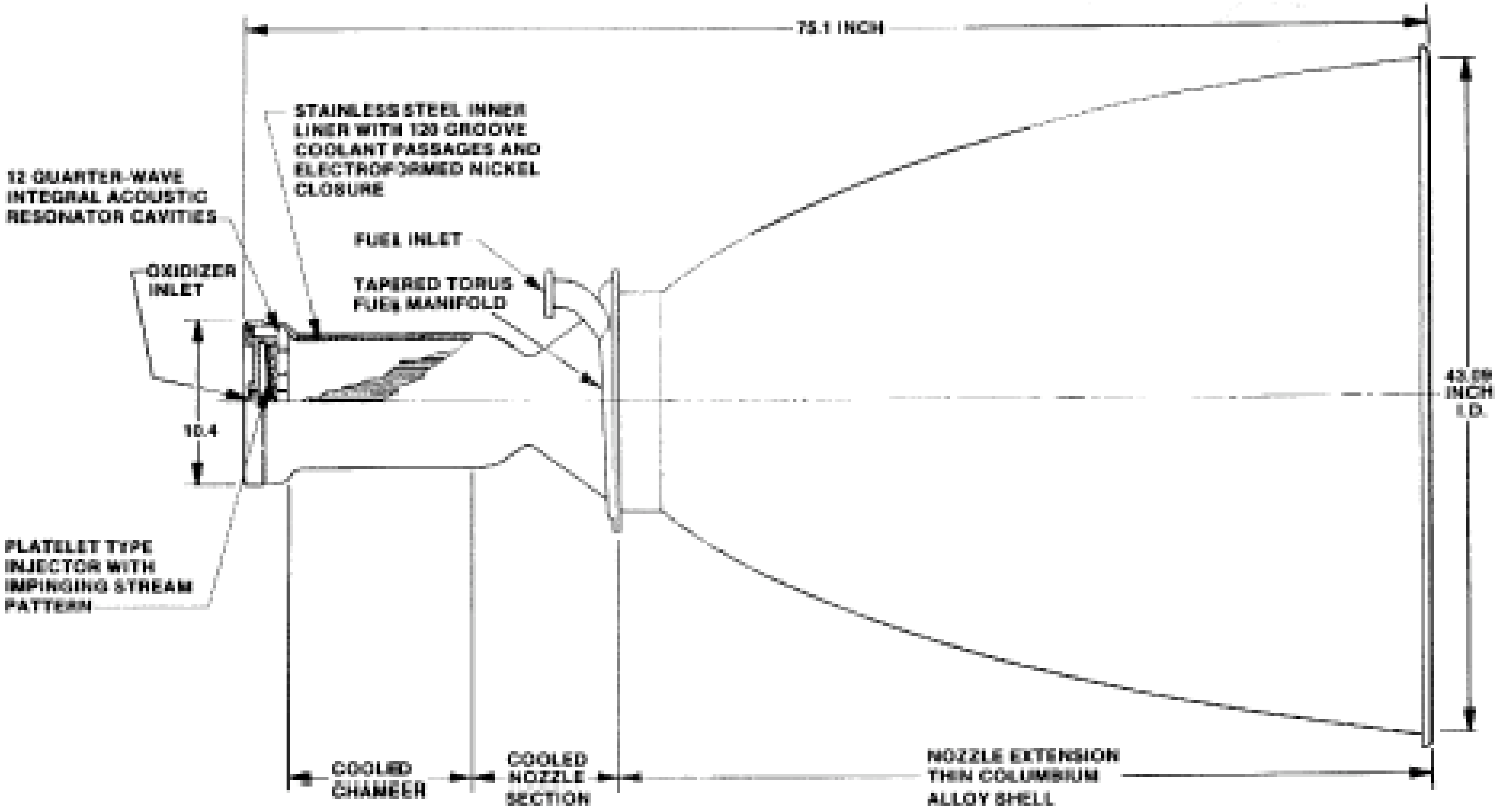
$$(4500\ psi)V_g^{1.67} = (388\ psi)V_g^{1.67} + (312\ psi)\left(17.39\ m^3\right)^{1.67}$$

- $V_g = 3.713\ m^3$
- Ideal gas: T=300°K -->

$$\rho_{He} = \frac{p_{g,0}\overline{M}}{\Re T_0}$$

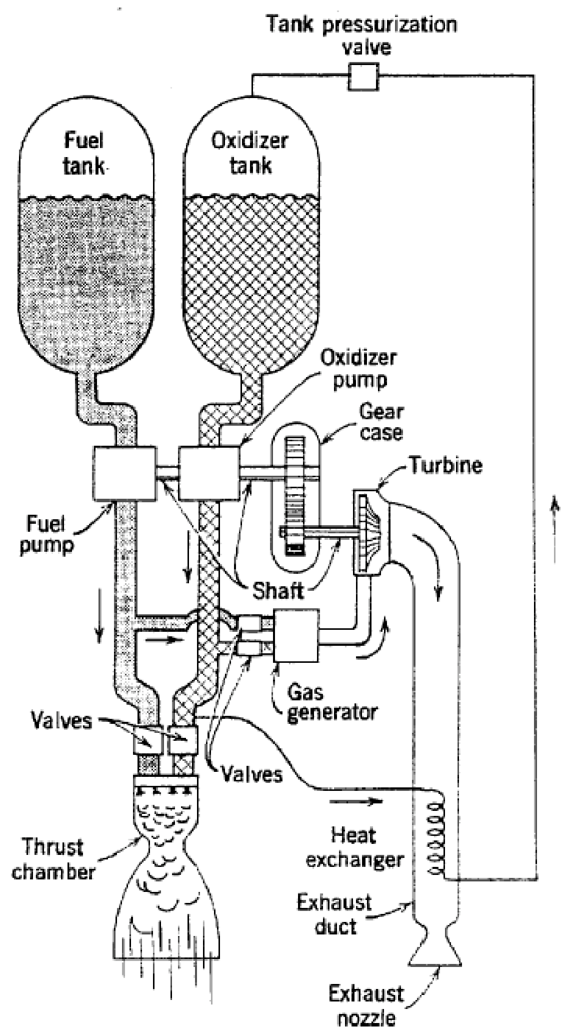
$\rho$=49.7 kg/m$^3$ (300 psi = 31.04 MPa) $M_{He}$=185.1 kg

# Space Shuttle OMS Engine




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

# Turbopump Fed Liquid Rocket Engine

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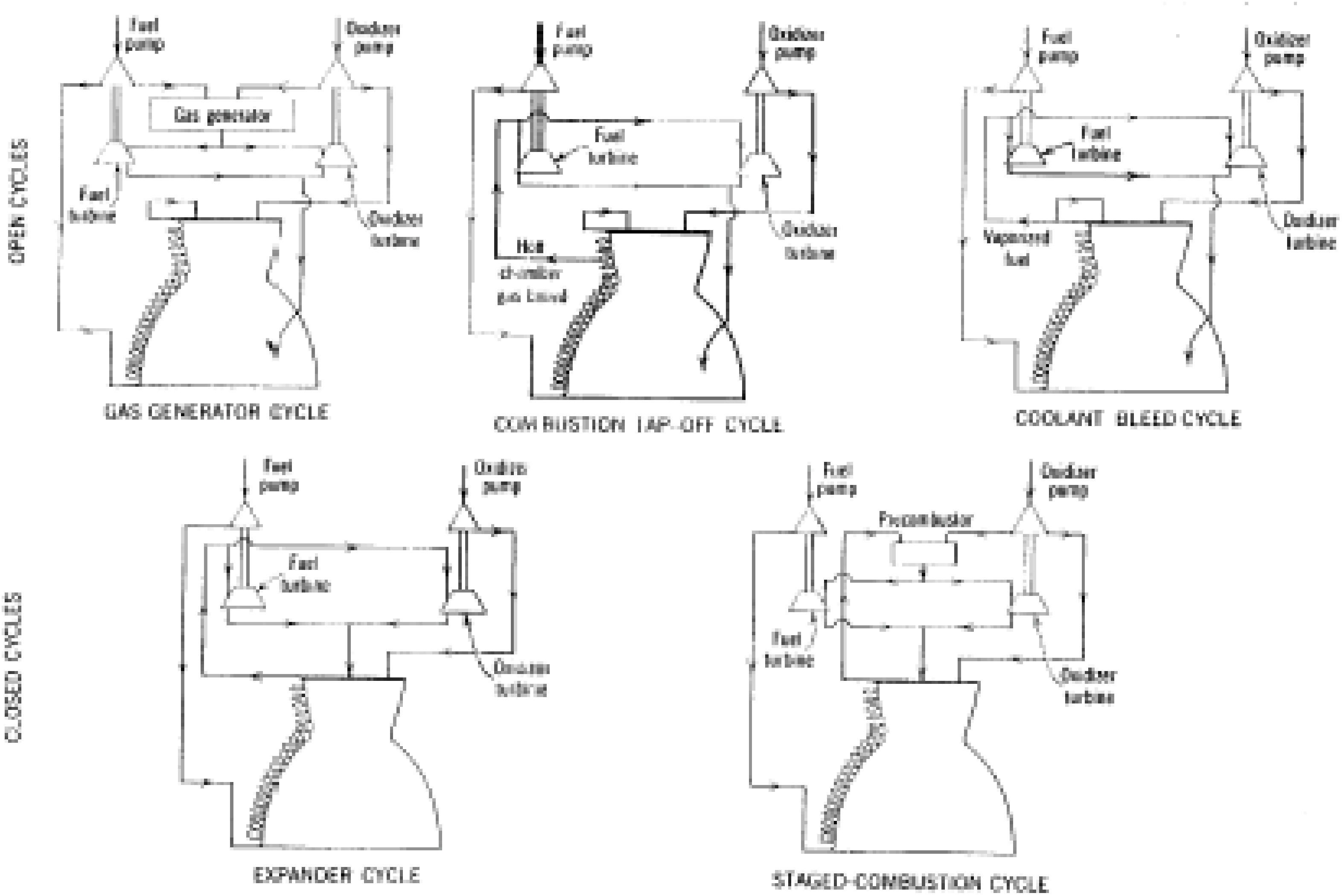
# Sample Pump-fed Engine Cycles

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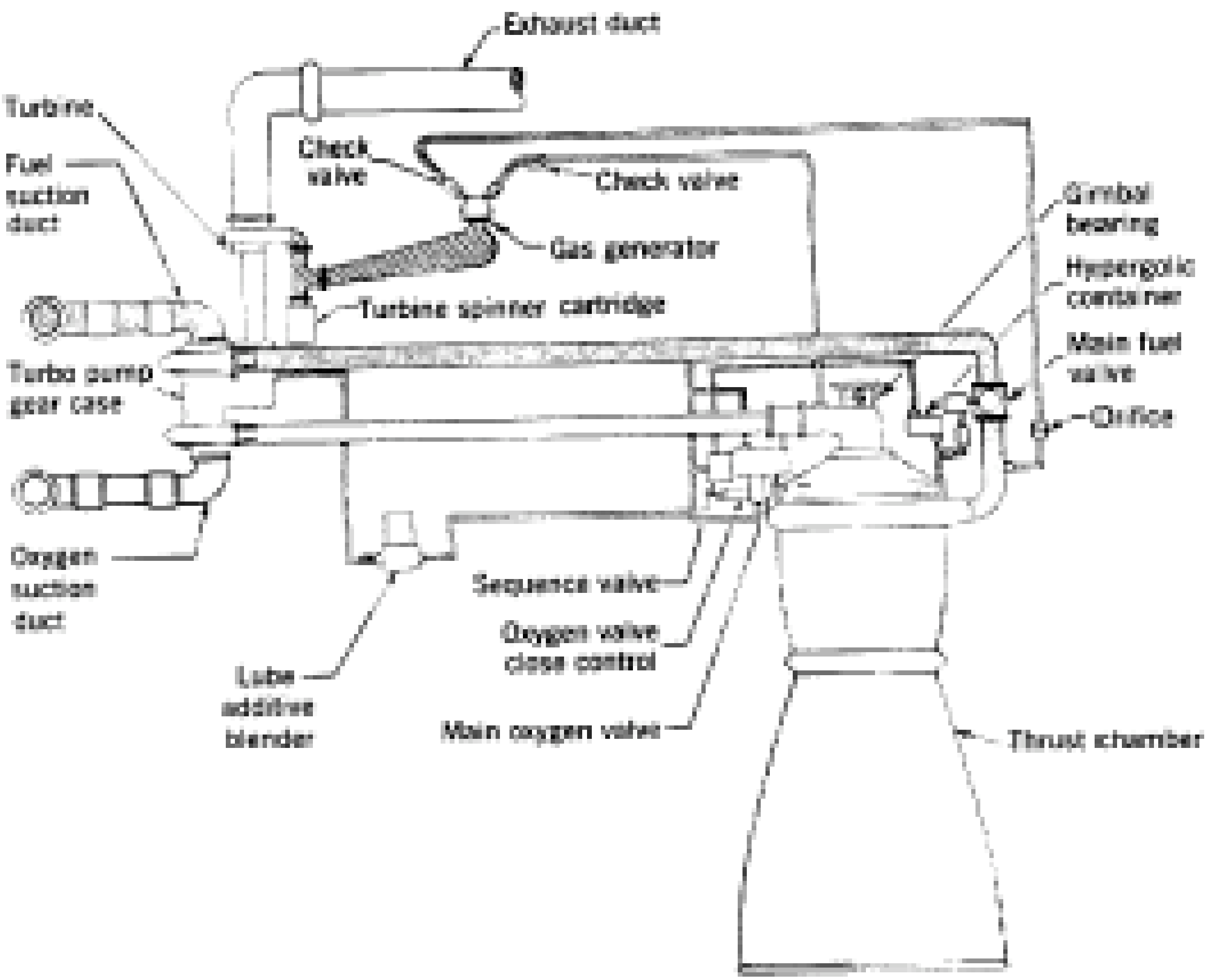
# Gas Generator Cycle Engine

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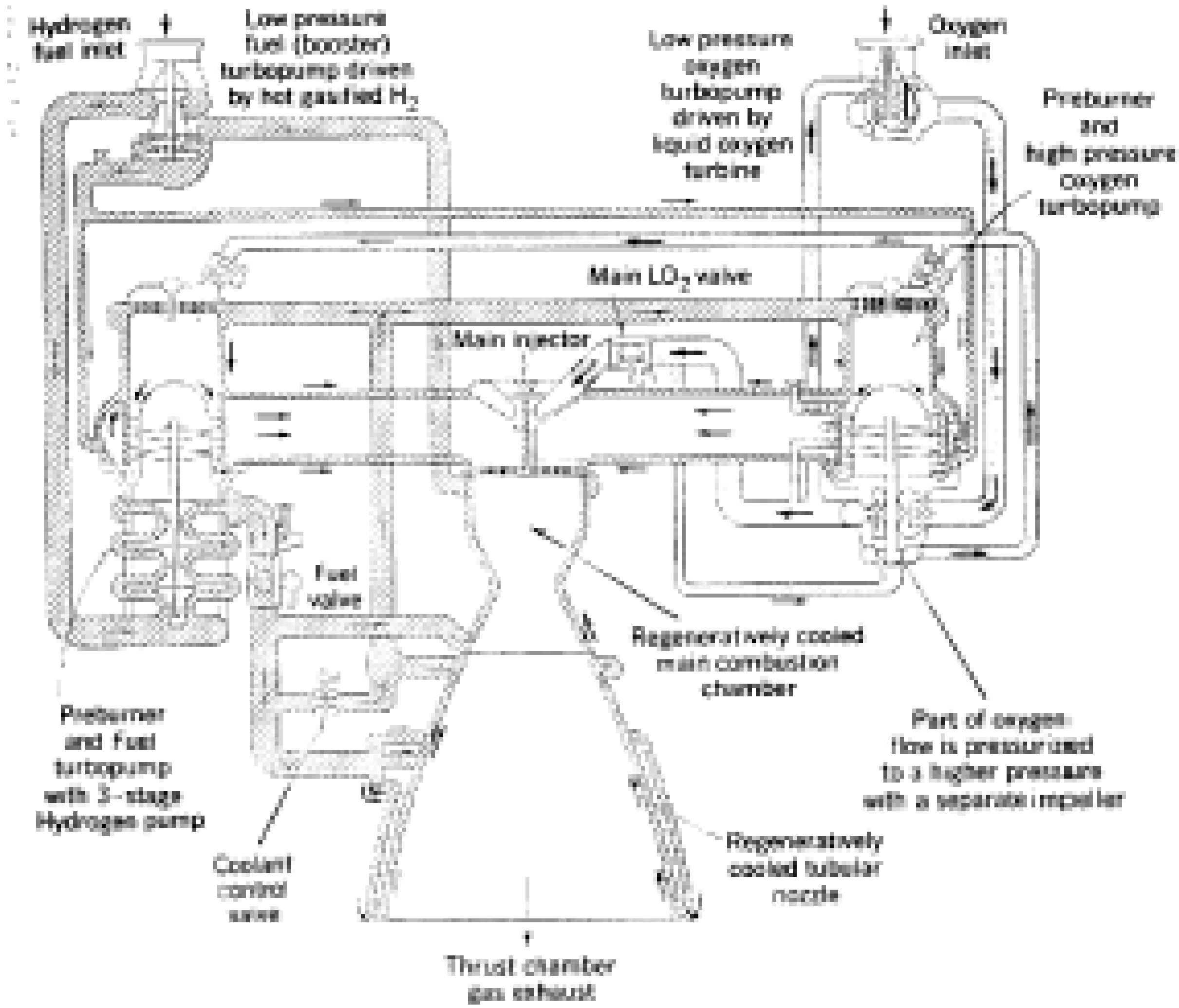
# SSME Engine Cycle

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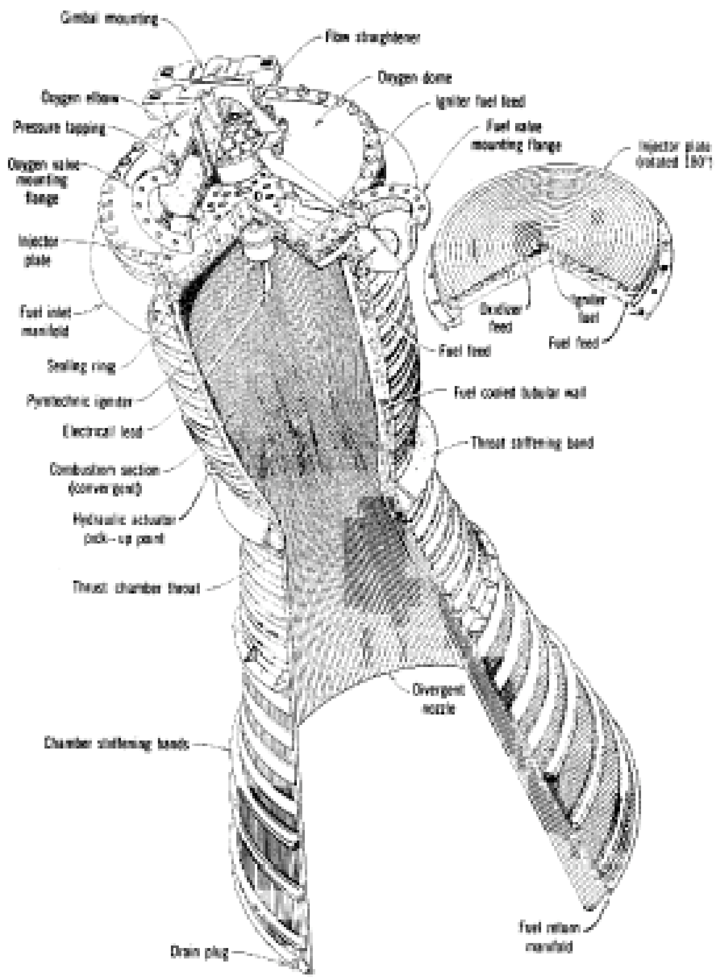
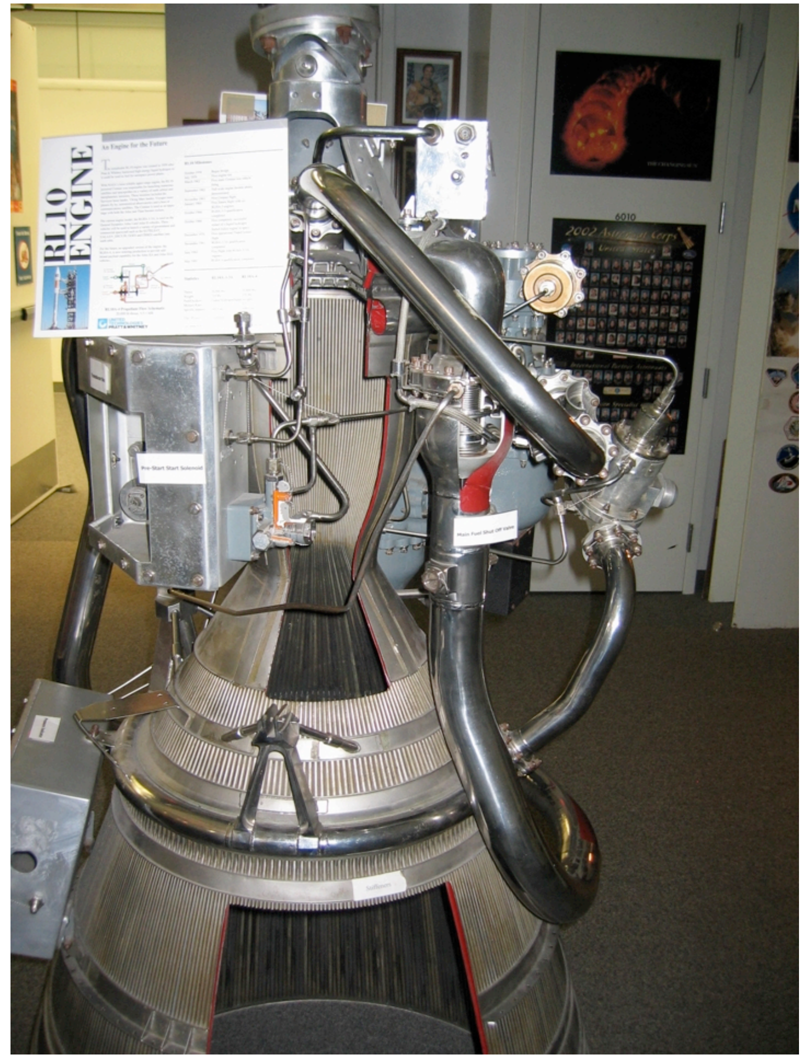
# Liquid Rocket Engine Cutaway

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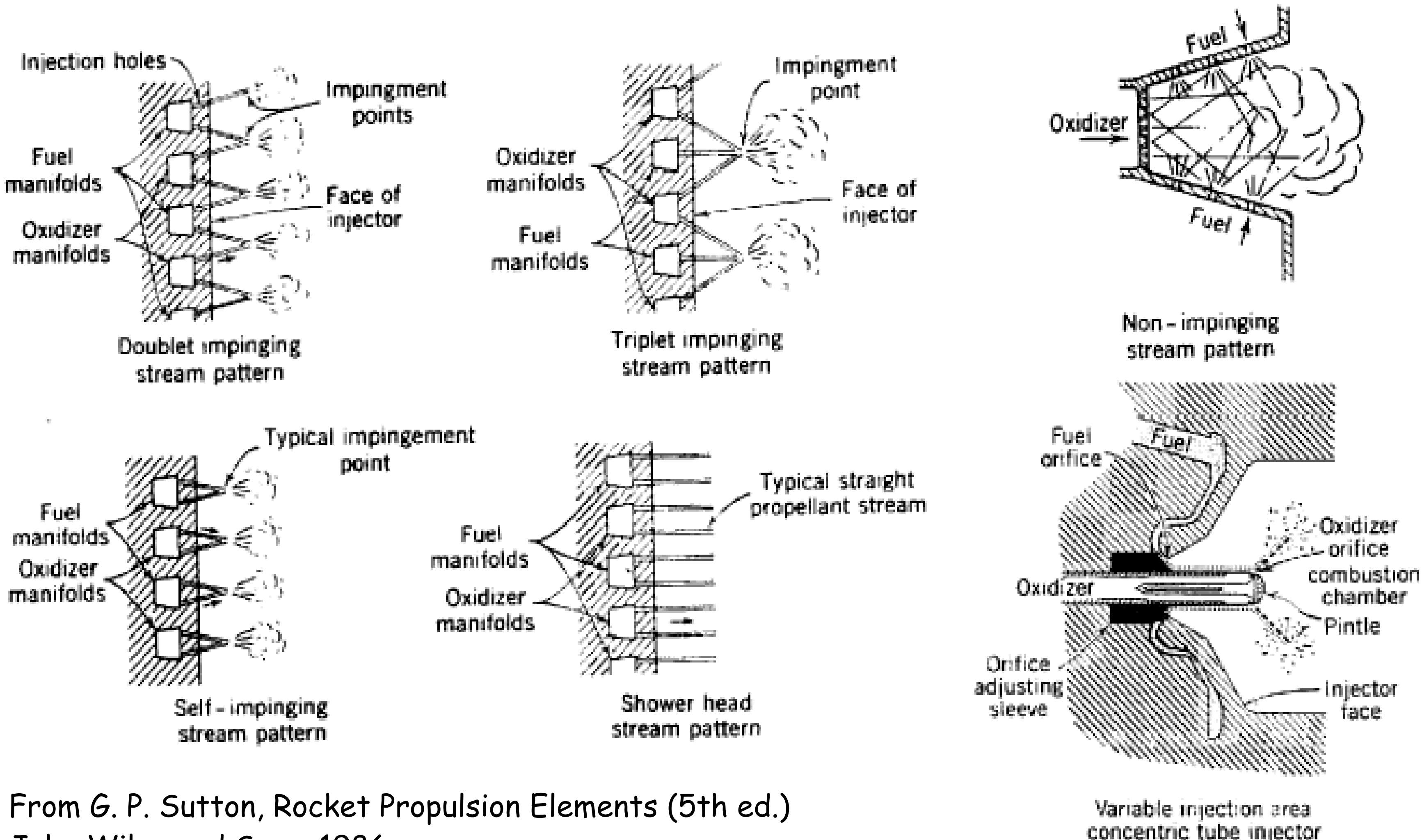
# H-1 Engine Injector Plate

# Injector Concepts

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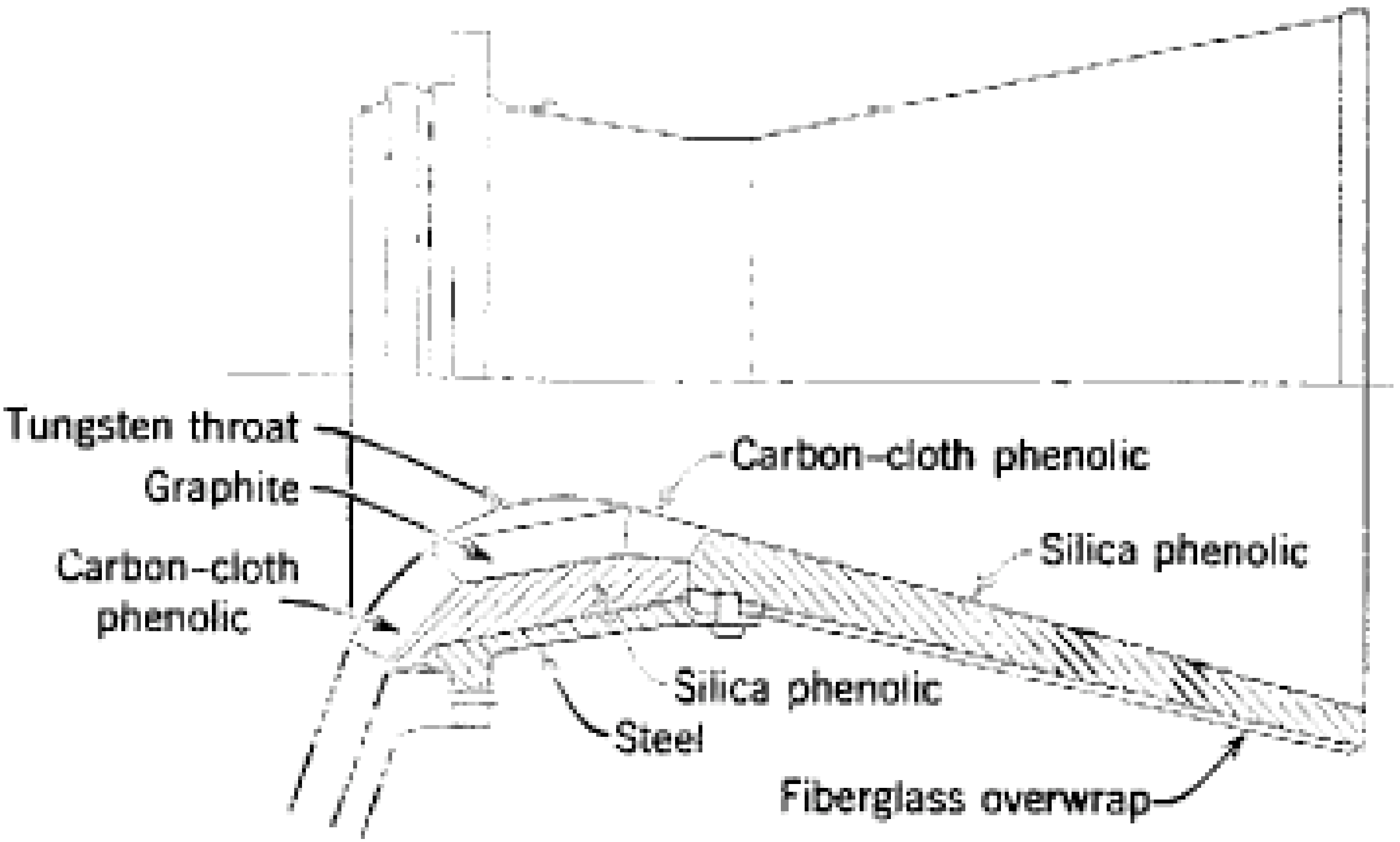
# Solid Rocket Nozzle (Heat-Sink)

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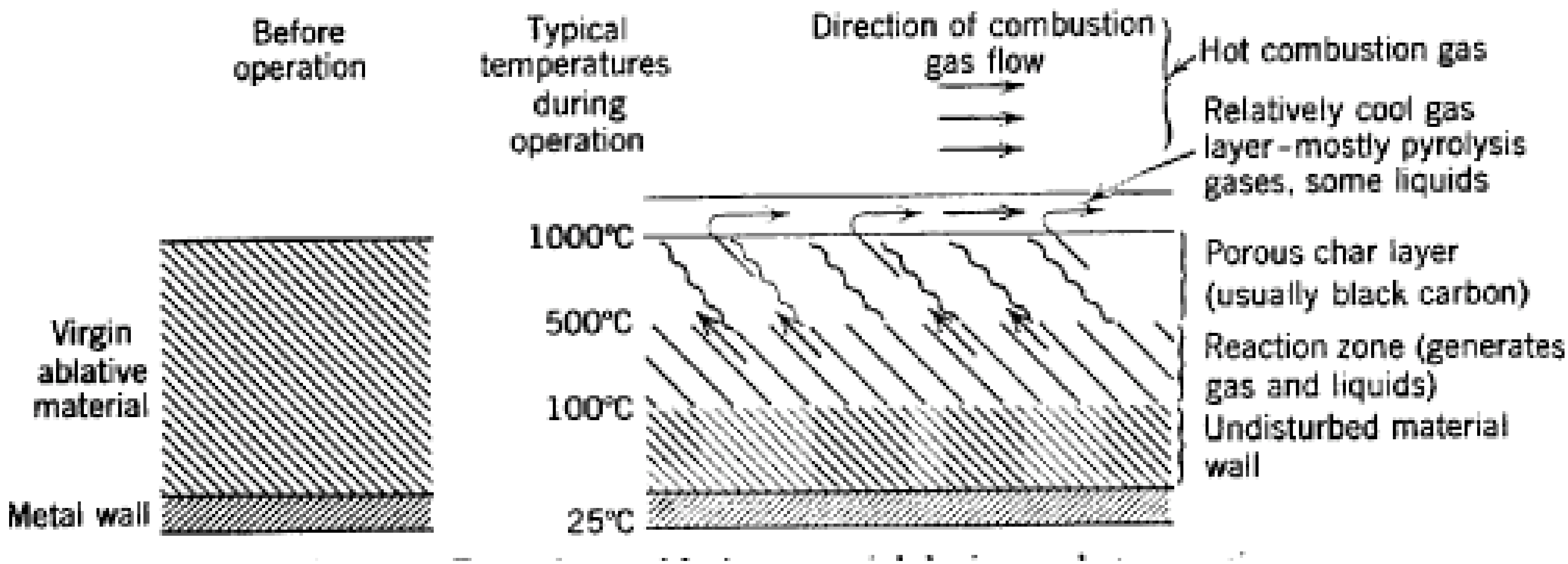
# Ablative Nozzle Schematic

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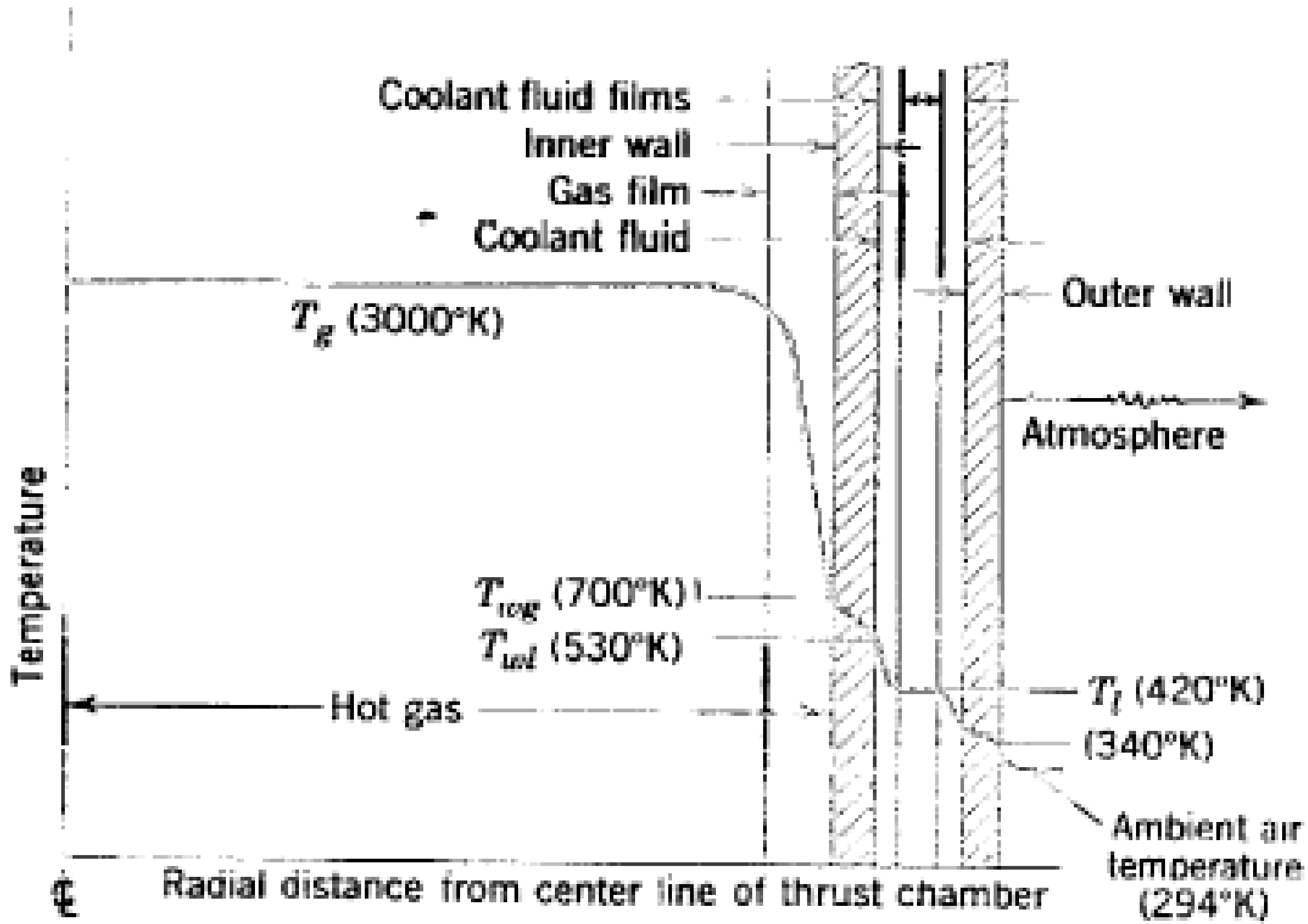
# Active Chamber Cooling Schematic

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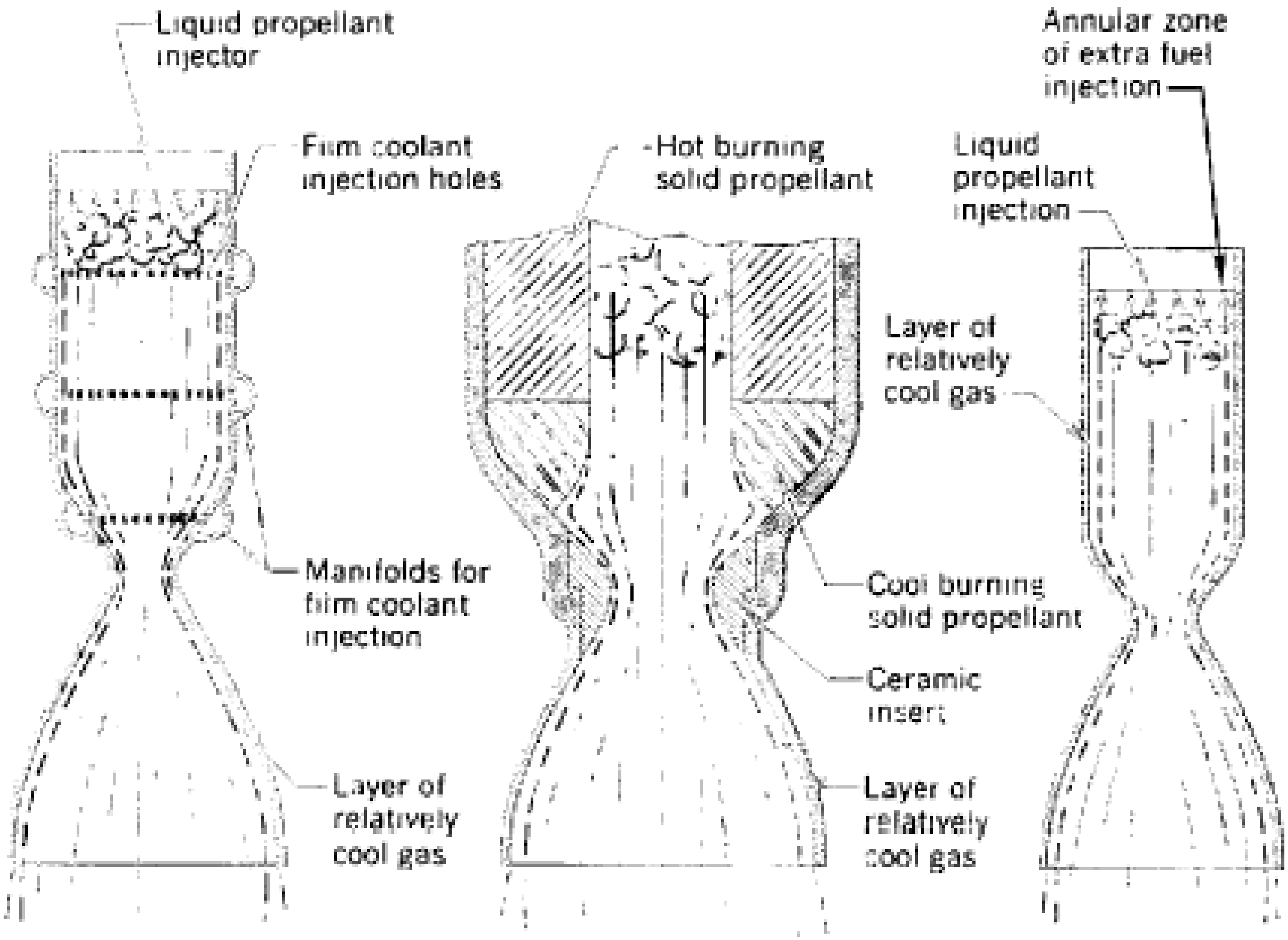
# Boundary Layer Cooling Approaches

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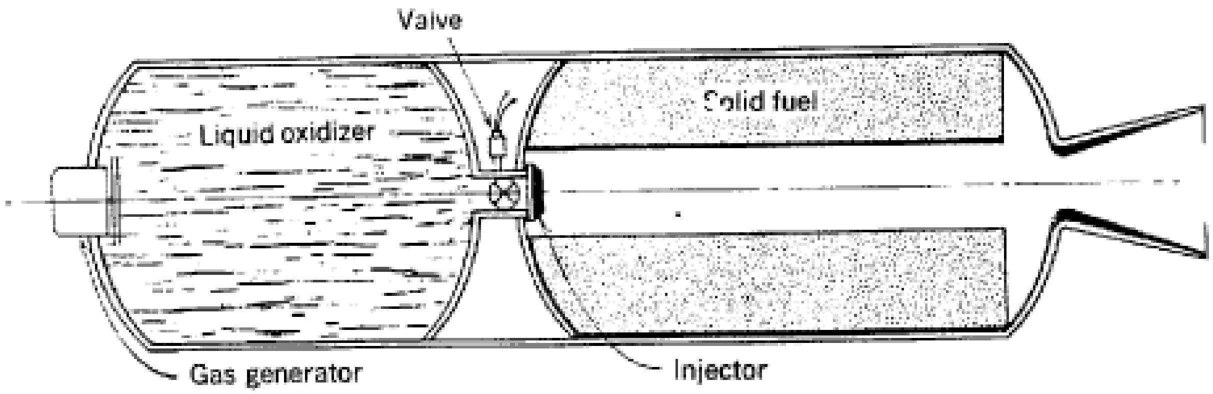
# Hybrid Rocket Schematic

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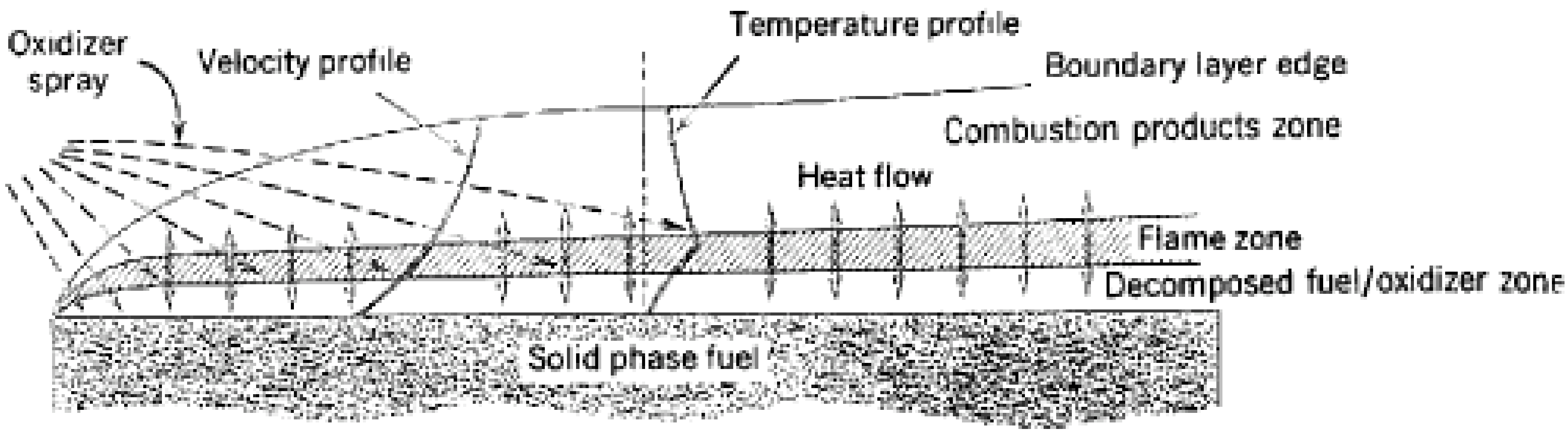
# Hybrid Rocket Combustion

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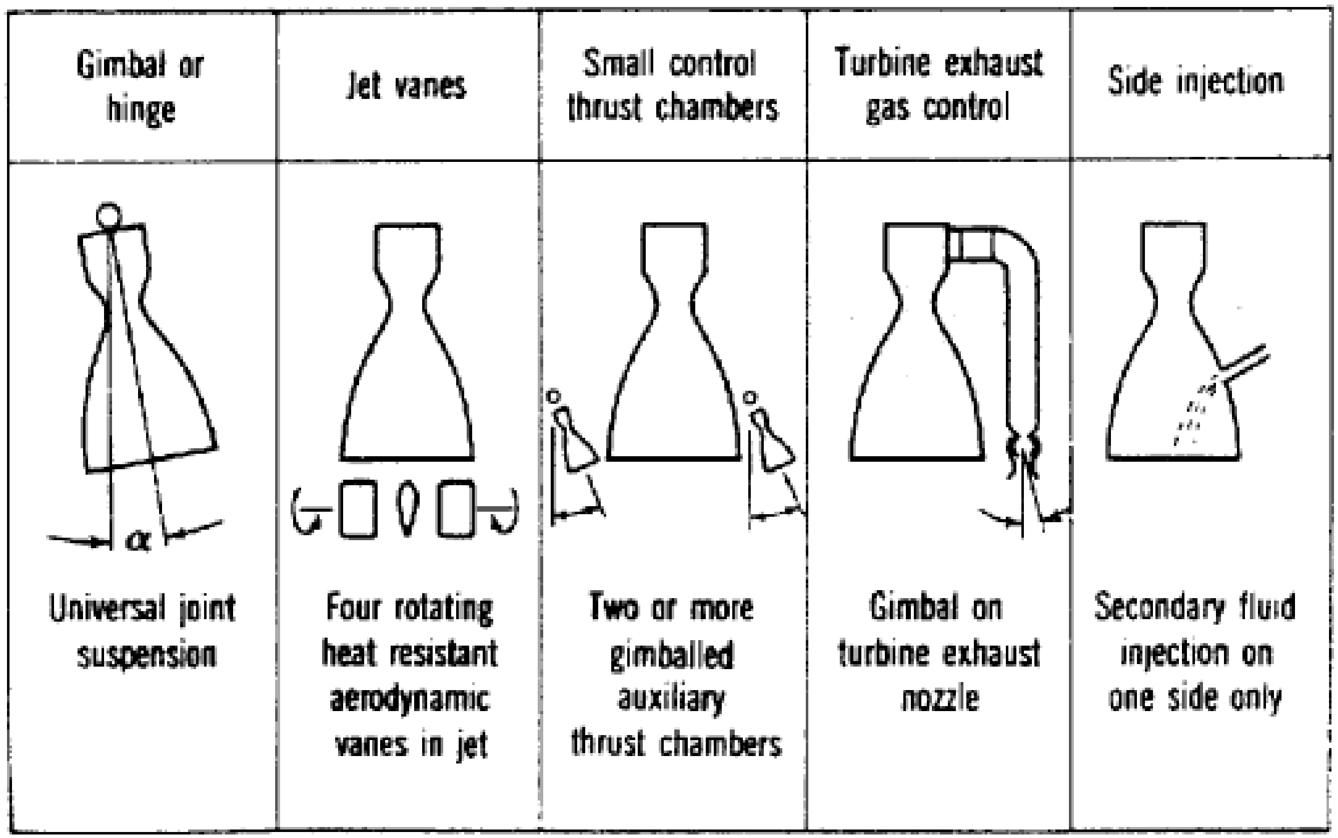
# Thrust Vector Control Approaches

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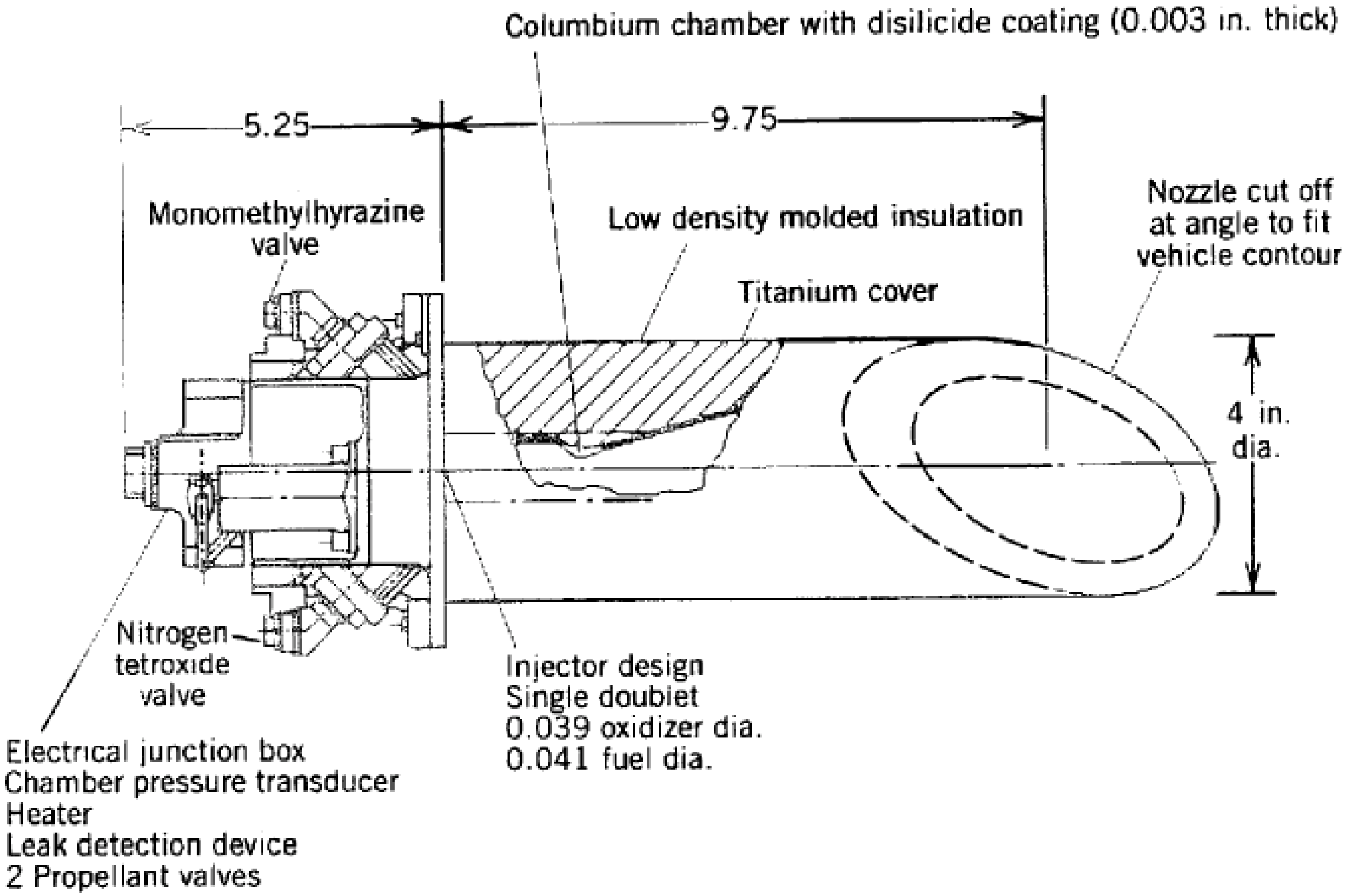
# Apollo Reaction Control System Thrusters

# Space Shuttle Primary RCS Engine

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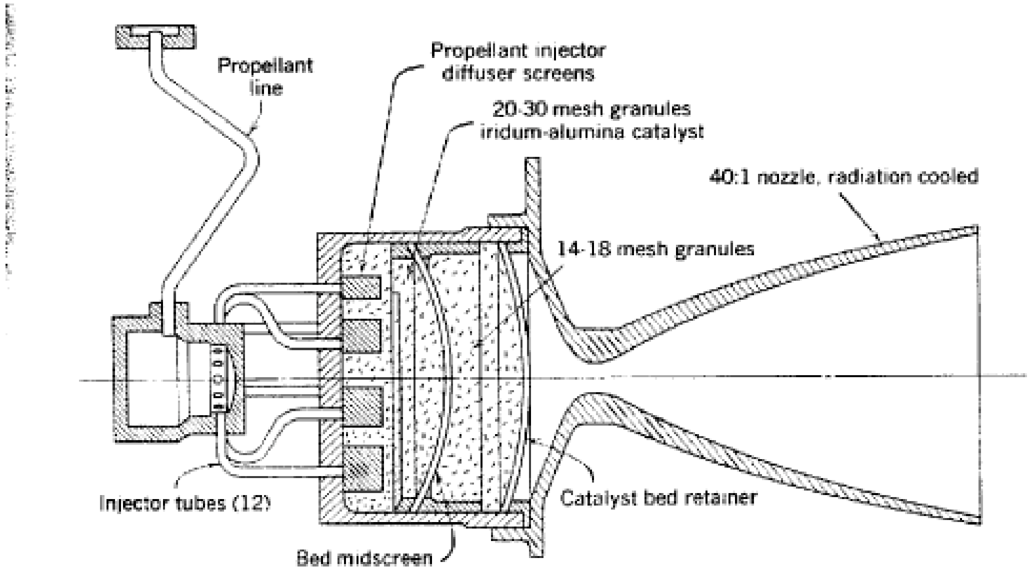
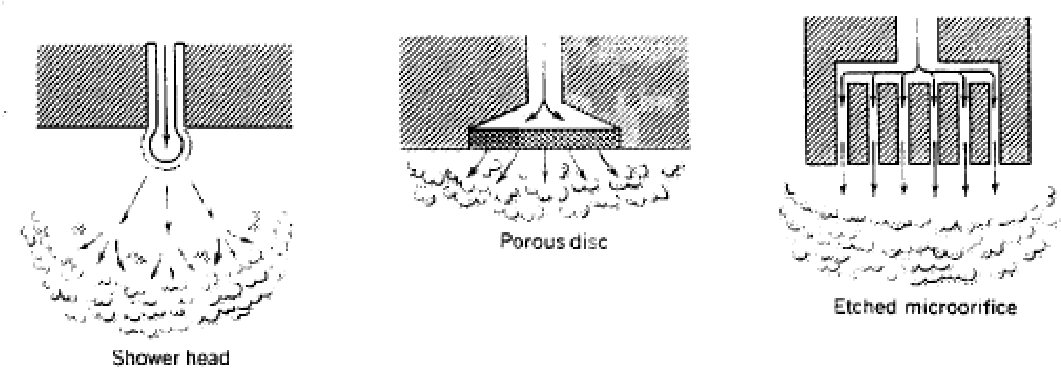
# Monopropellant Engine Design

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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.

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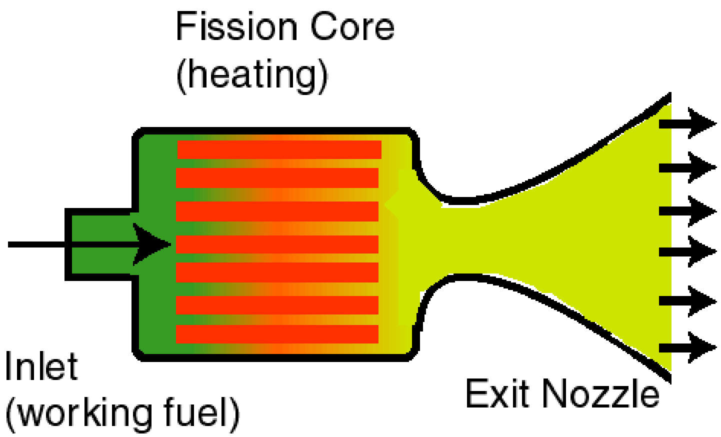
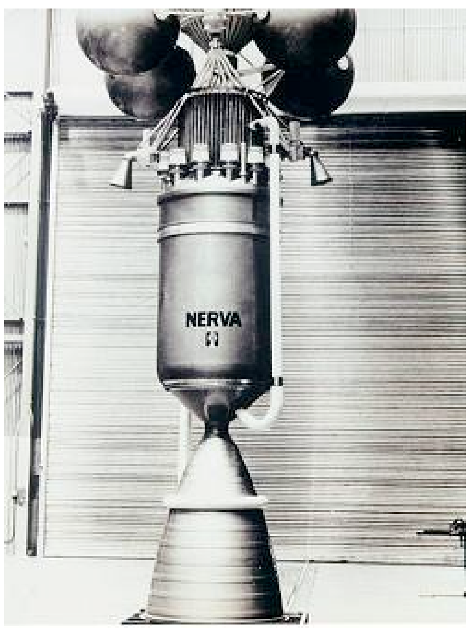
# Nuclear Thermal Rockets




- 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