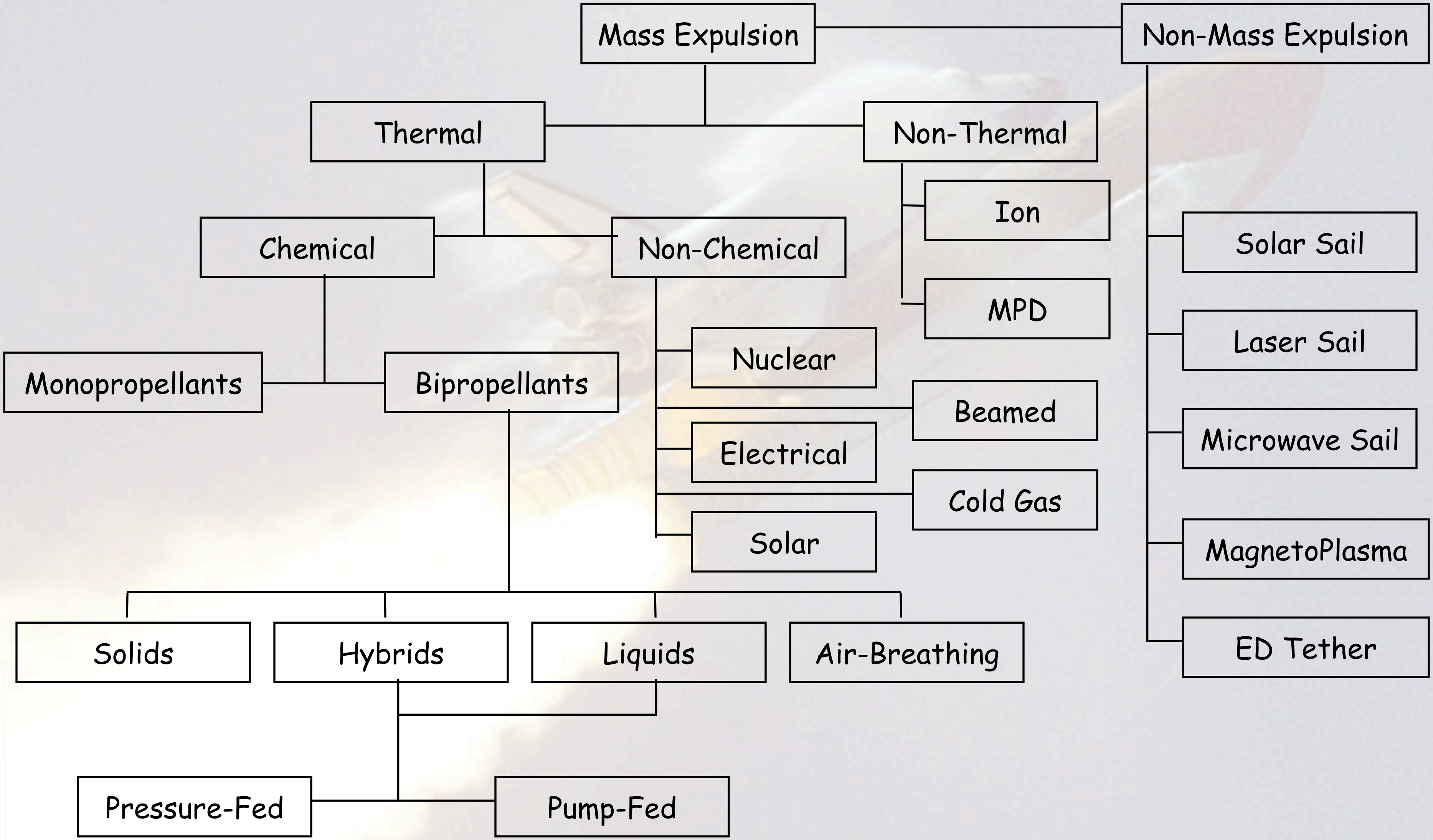
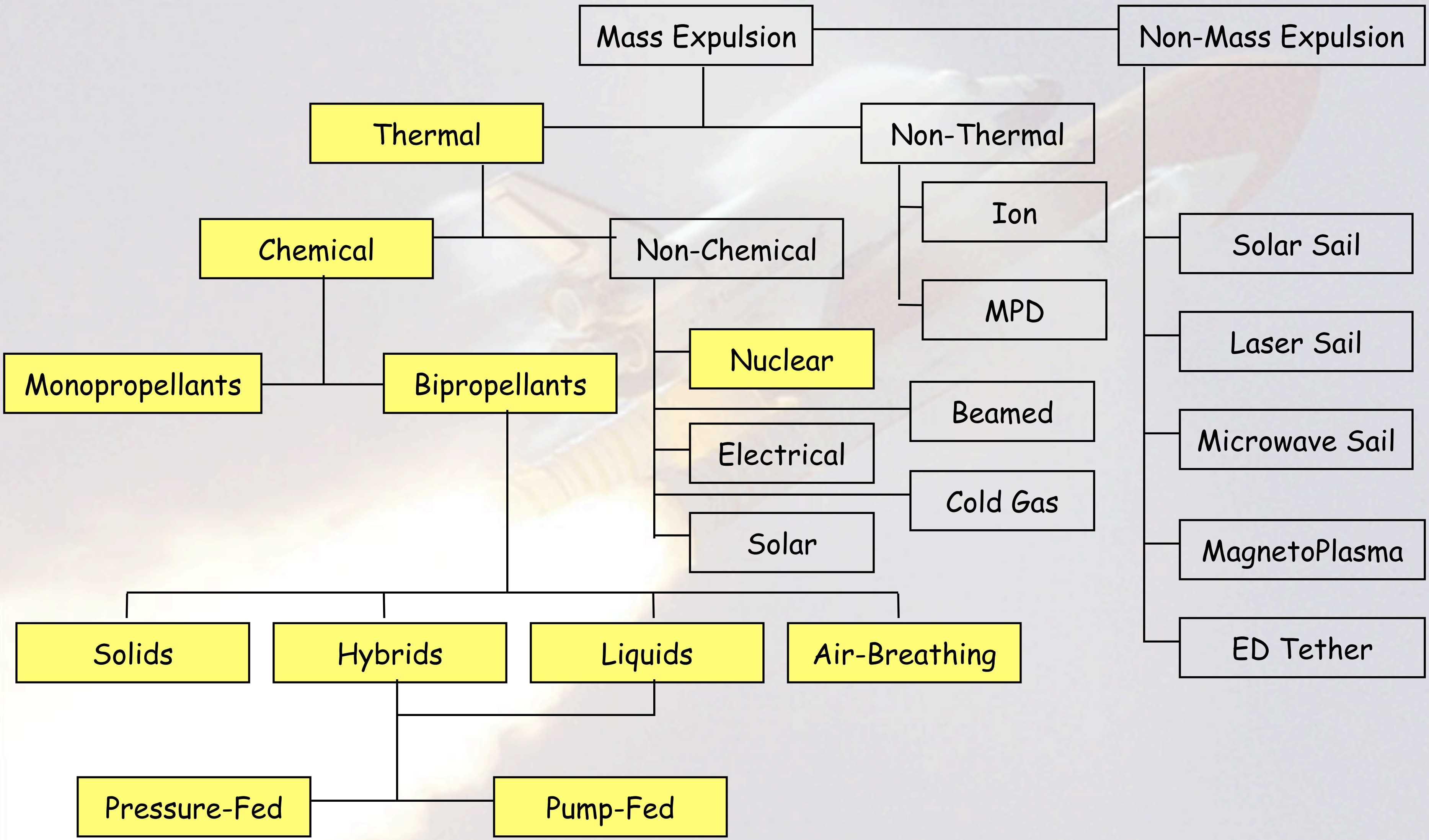
Propulsion Systems Design

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

Propulsion Taxonomy



Propulsion Taxonomy



Thermal Rocket Exhaust Velocity

- Exhaust velocity is

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

where

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

$\Re \equiv$ universal gas constant $= 8314.3 \frac{Joules}{mole\ {}^oK}$

$\gamma \equiv$ ratio of specific heats ≈ 1.2

$p_e \equiv$ pressure at nozzle exit plane

$p_o \equiv$ combustion chamber pressure

Ideal Thermal Rocket Exhaust Velocity

- Ideal exhaust velocity is

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

- 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

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

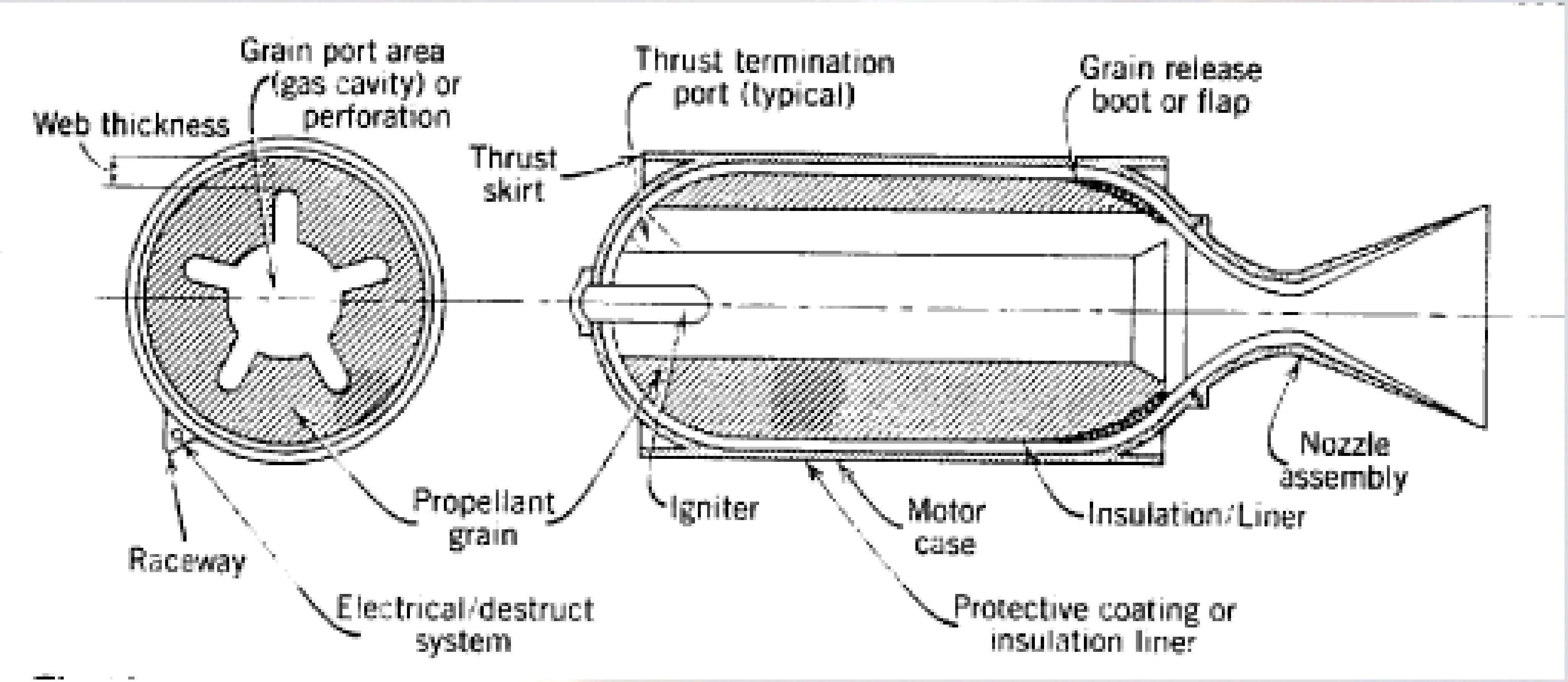
Nozzle Design

- 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

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

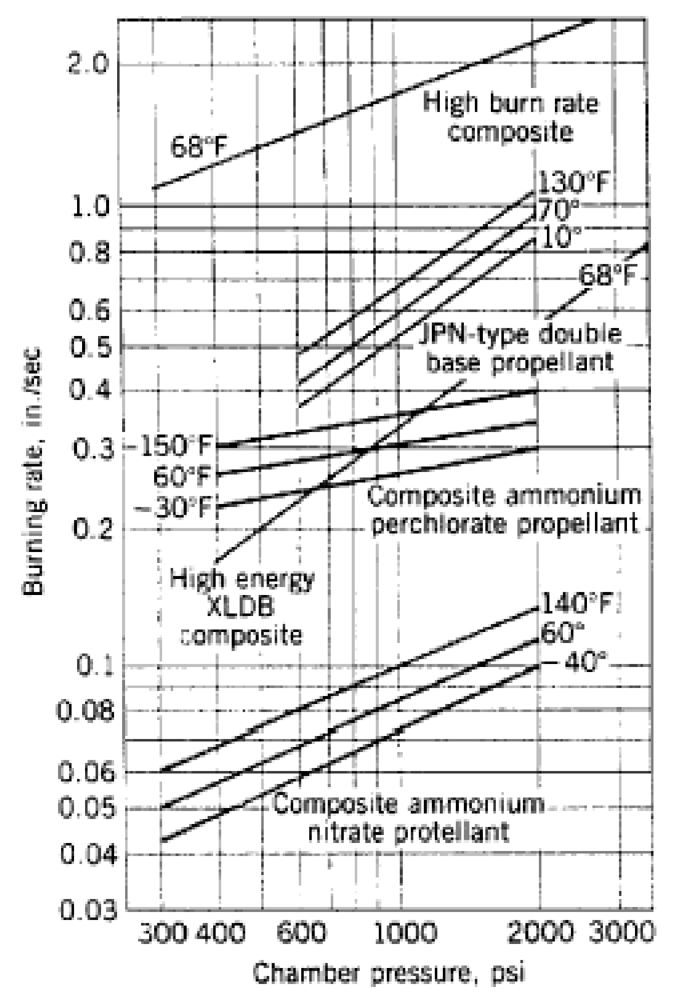
- $I_{sp,vacuum}$=455 sec --> $I_{sp,sl}$=333 sec

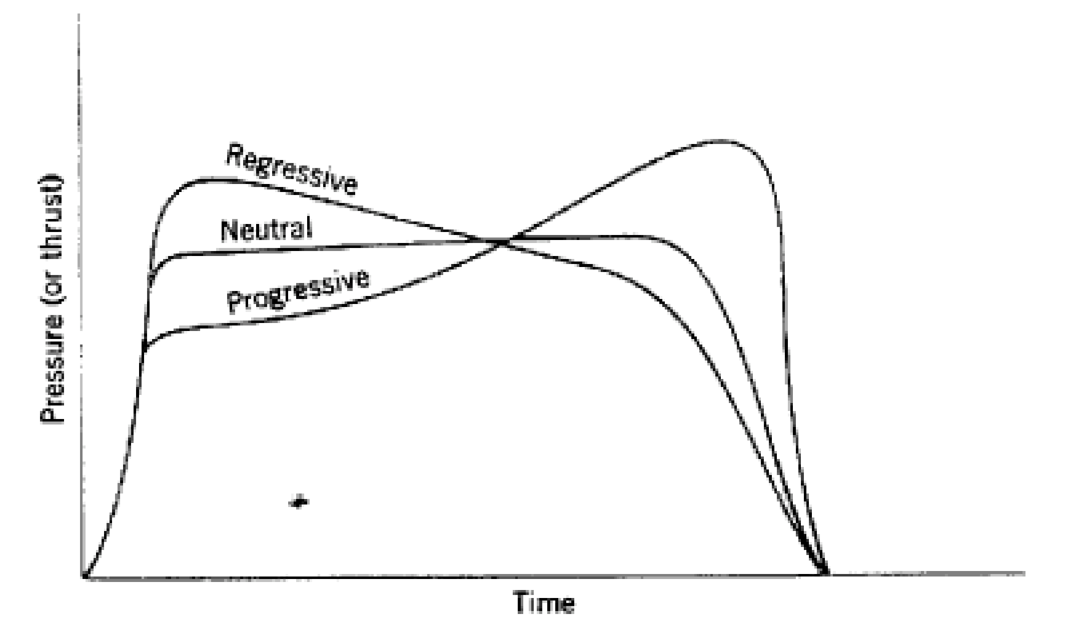
Solid Rocket Motor



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

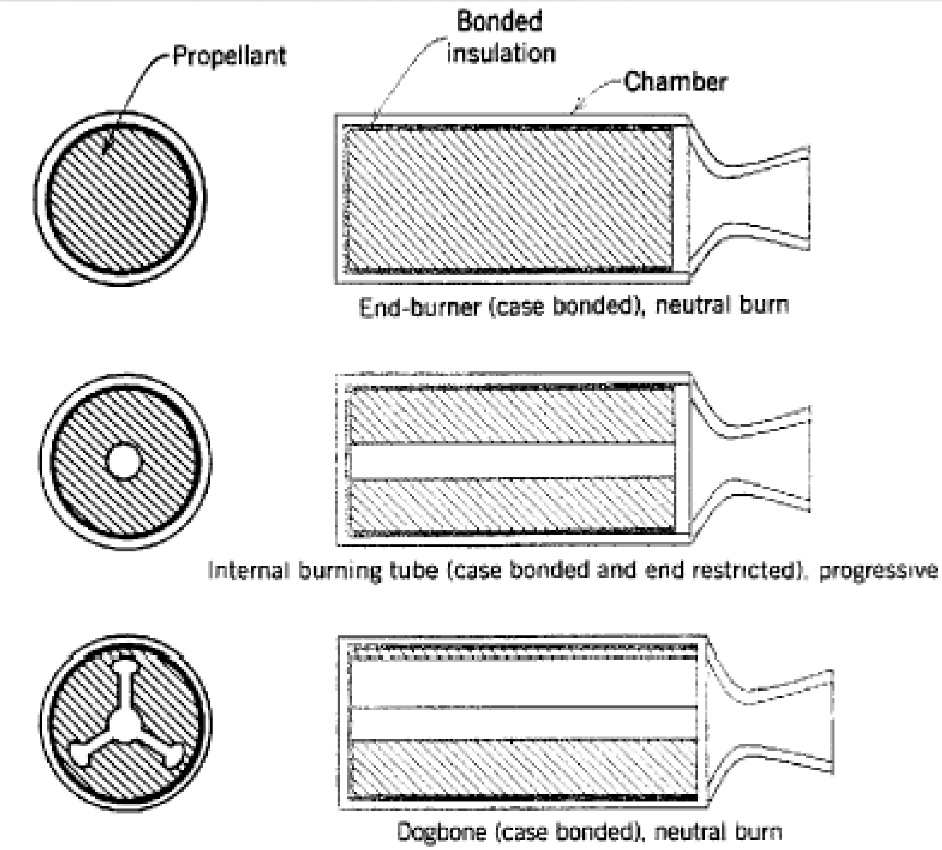
Solid Propellant Combustion

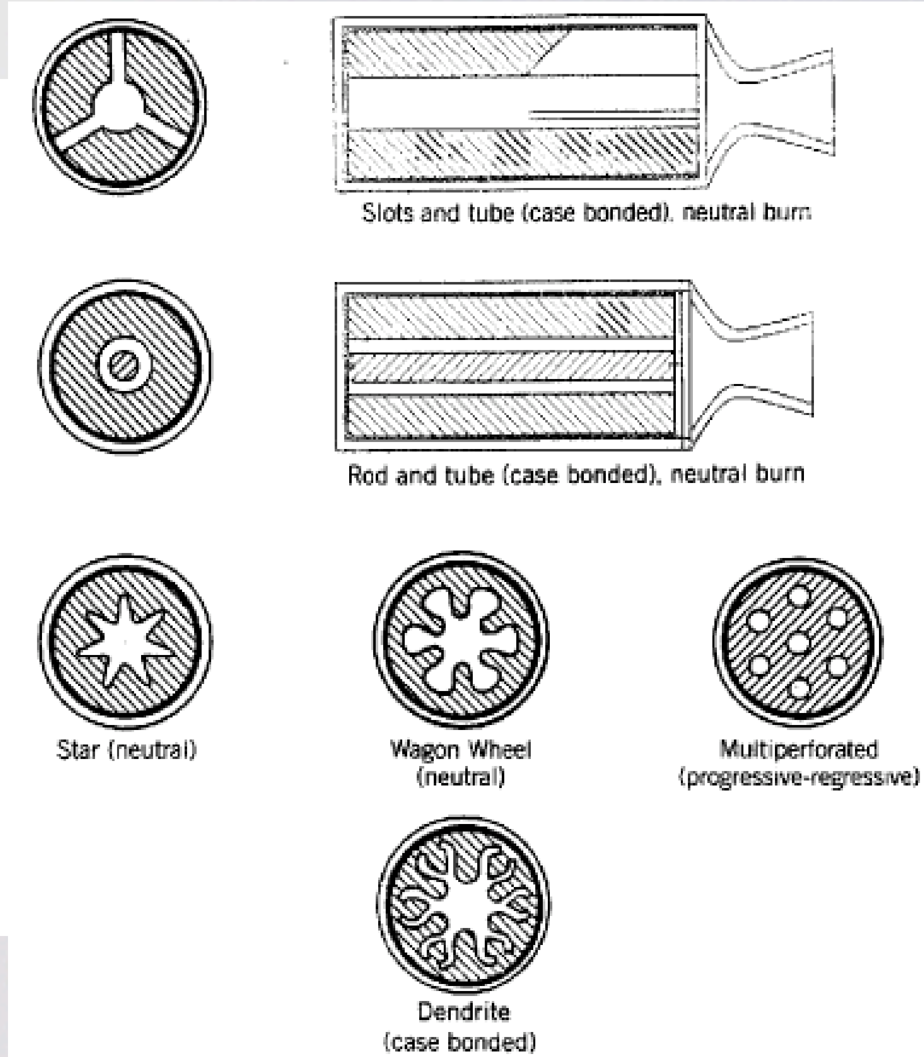




From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

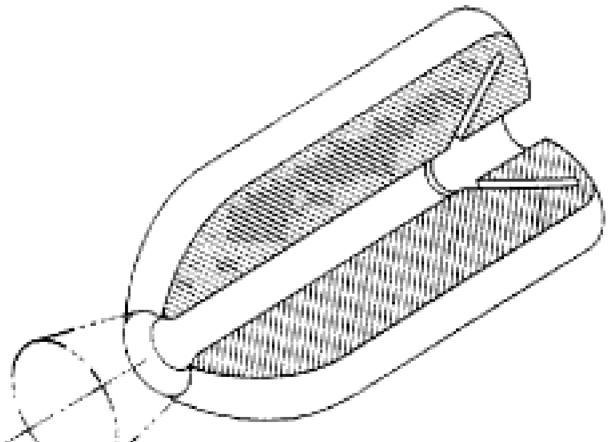
Solid Grain Configurations





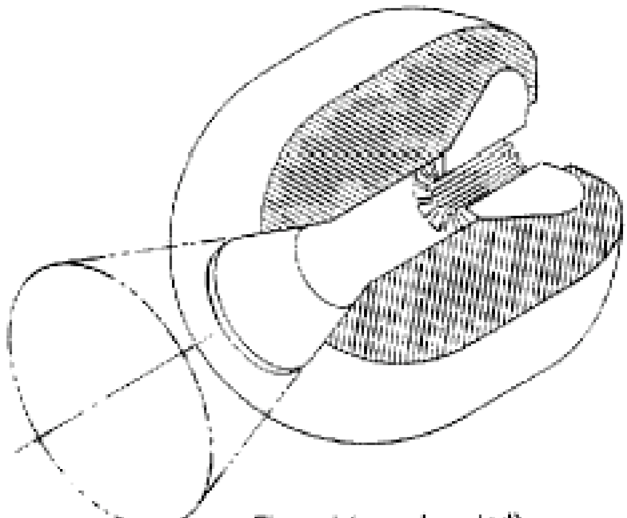
From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

Short-Grain Solid Configurations

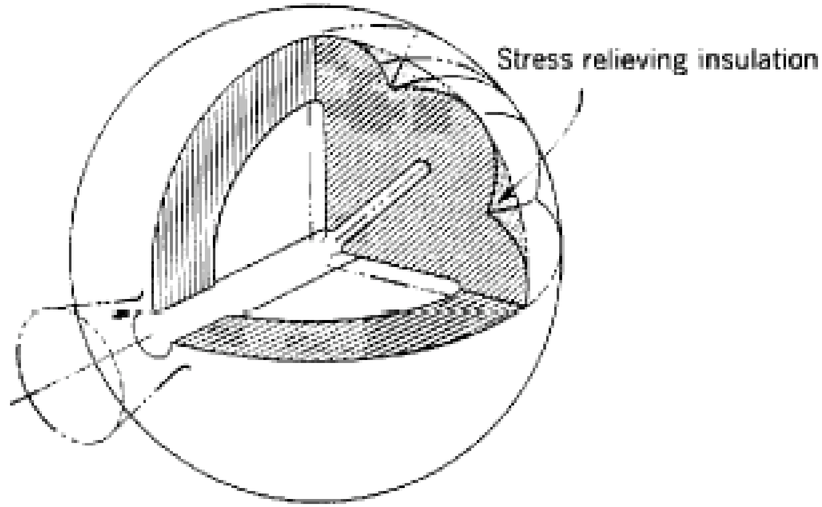


Conocyl (case-bonded)

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



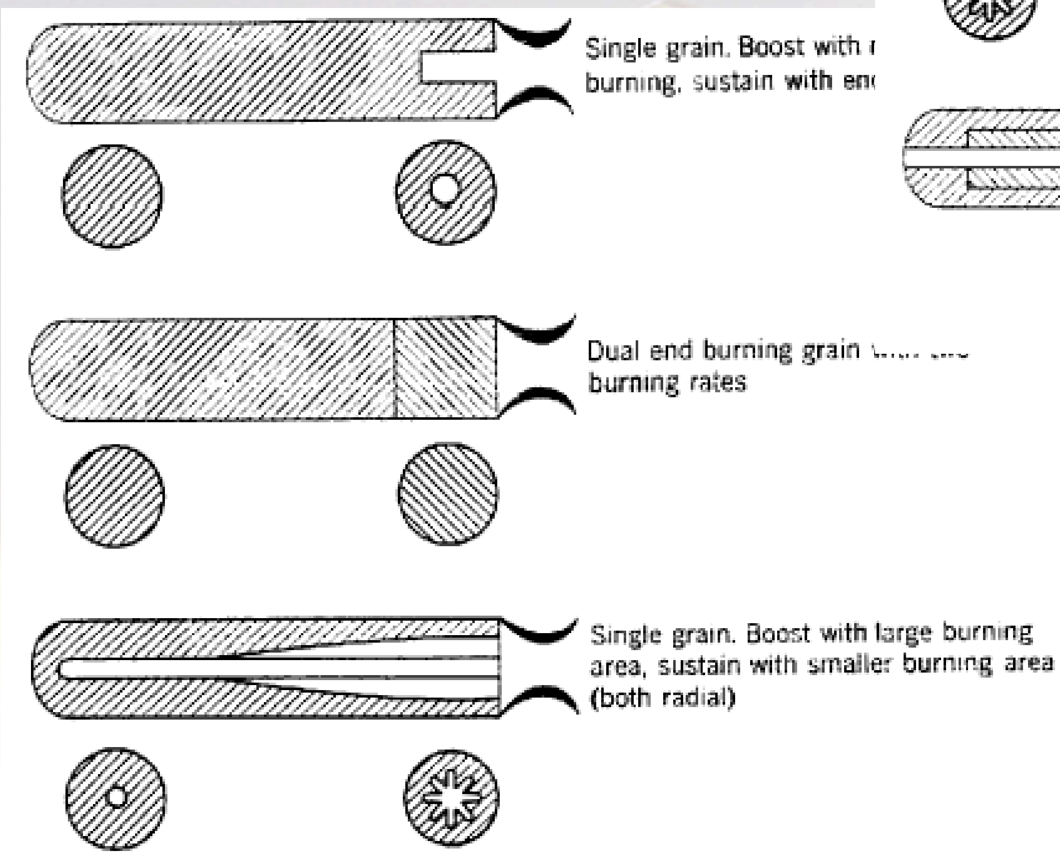
Finocyl (case-bonded)

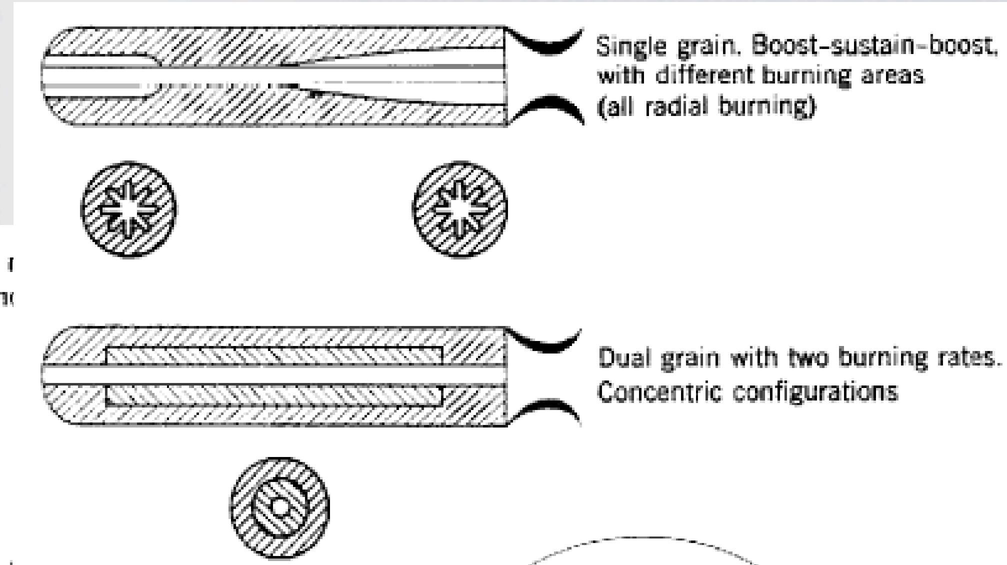


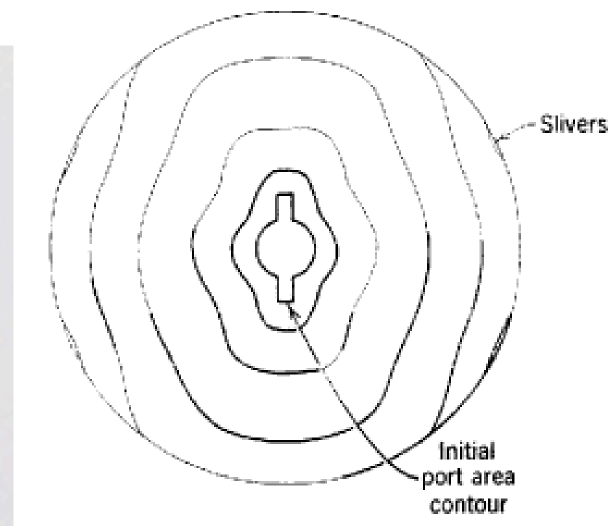
Spherical (case-bonded) with slots and cylinder

Advanced Grain Configurations

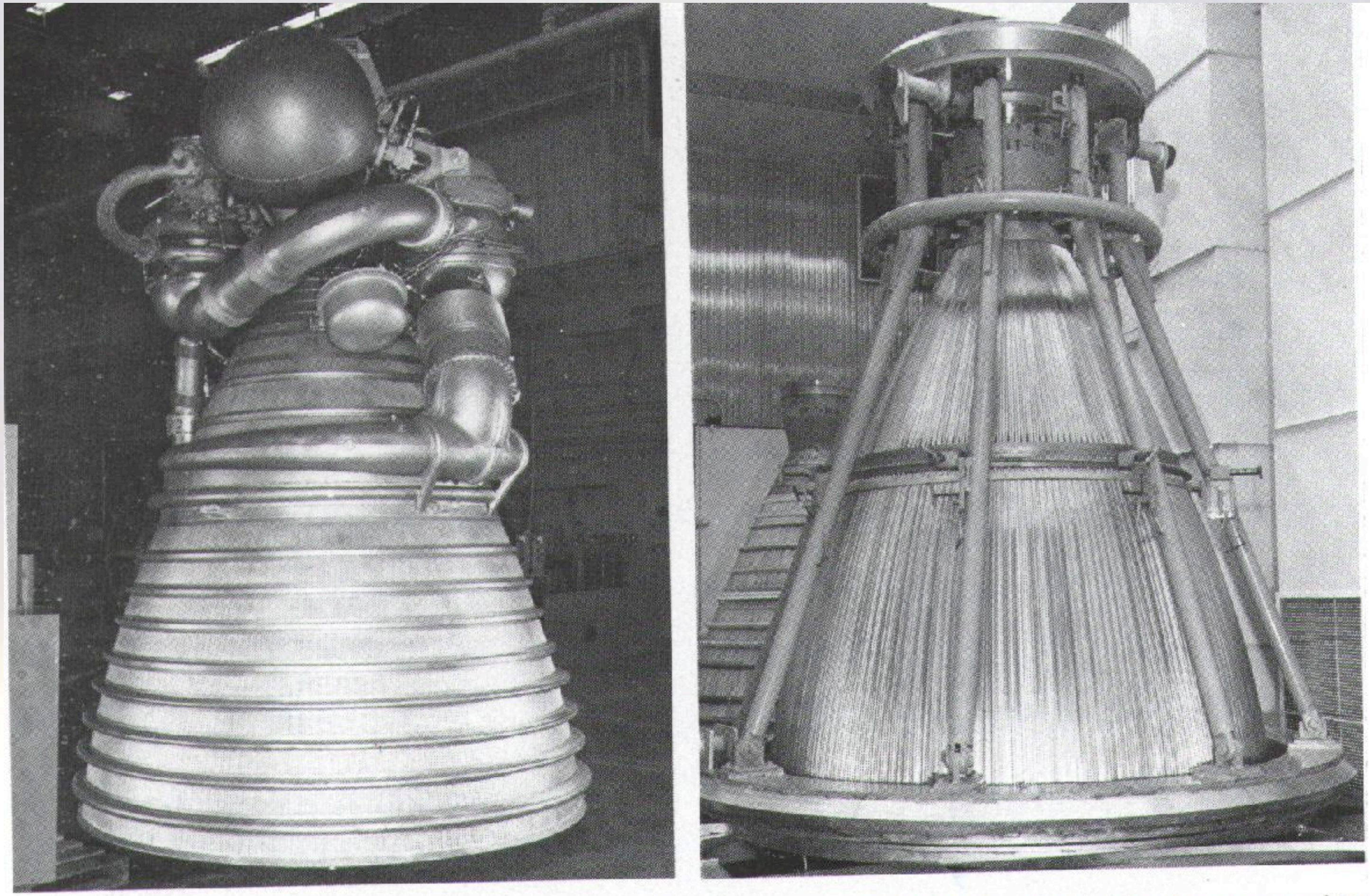
From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986





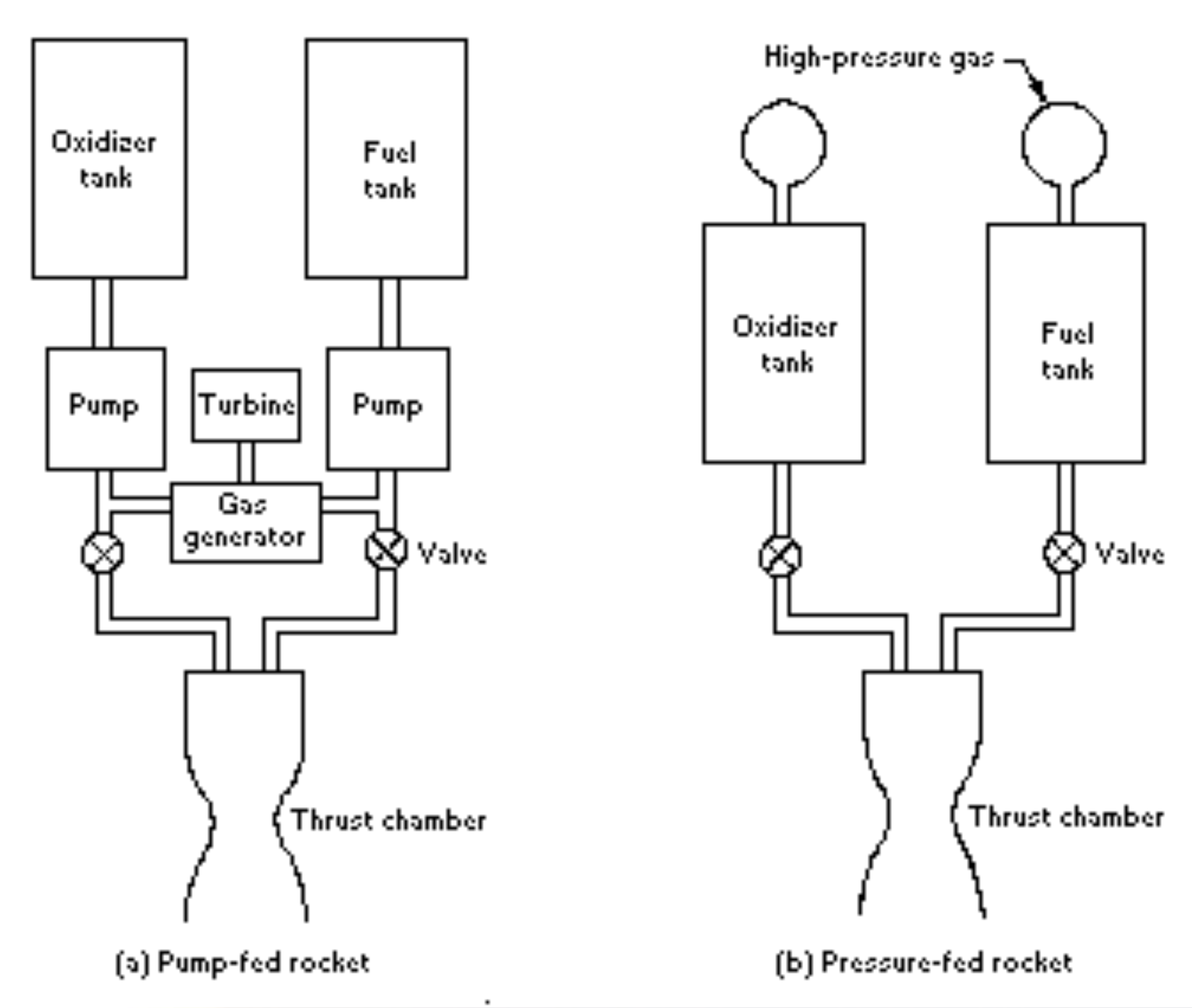


Liquid Rocket Engine

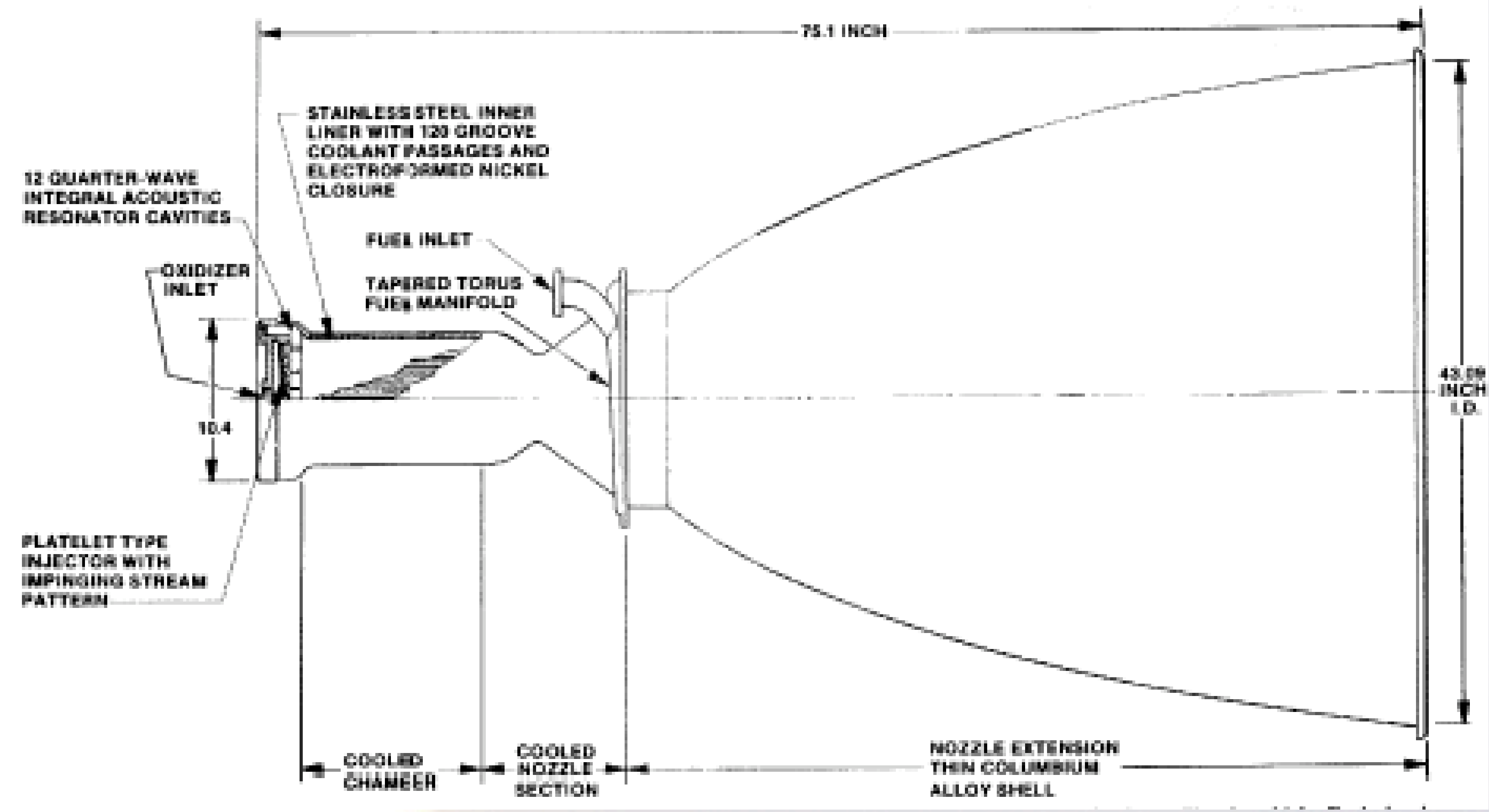


A completed J-2 rocket engine (left), with its pumps and lines installed. The basic engine structure is built up from a series of hollow tubes (right).

Liquid Propellant Feed Systems

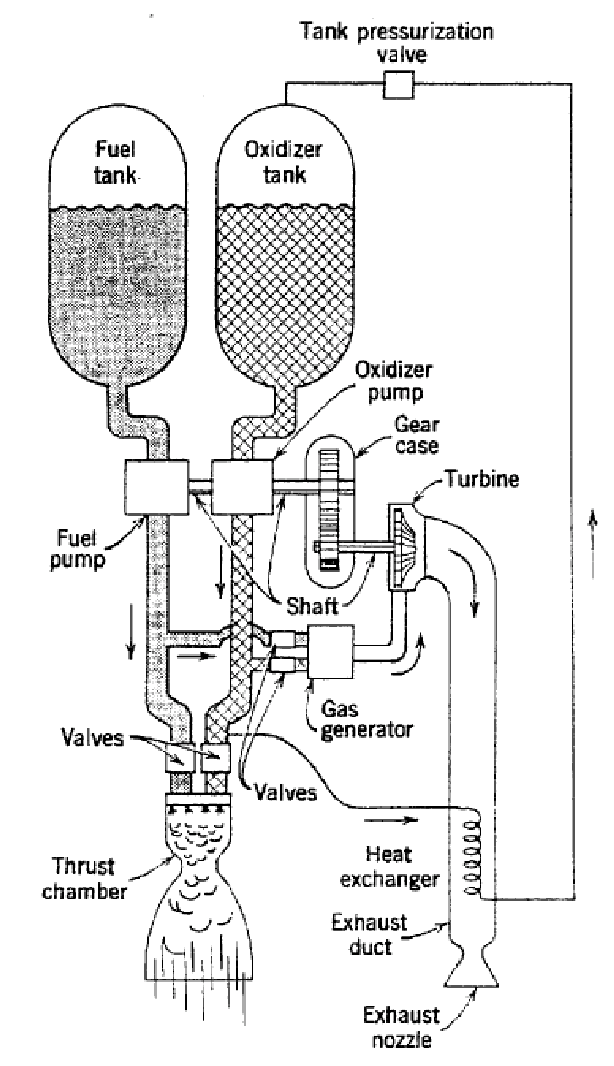


Space Shuttle OMS Engine



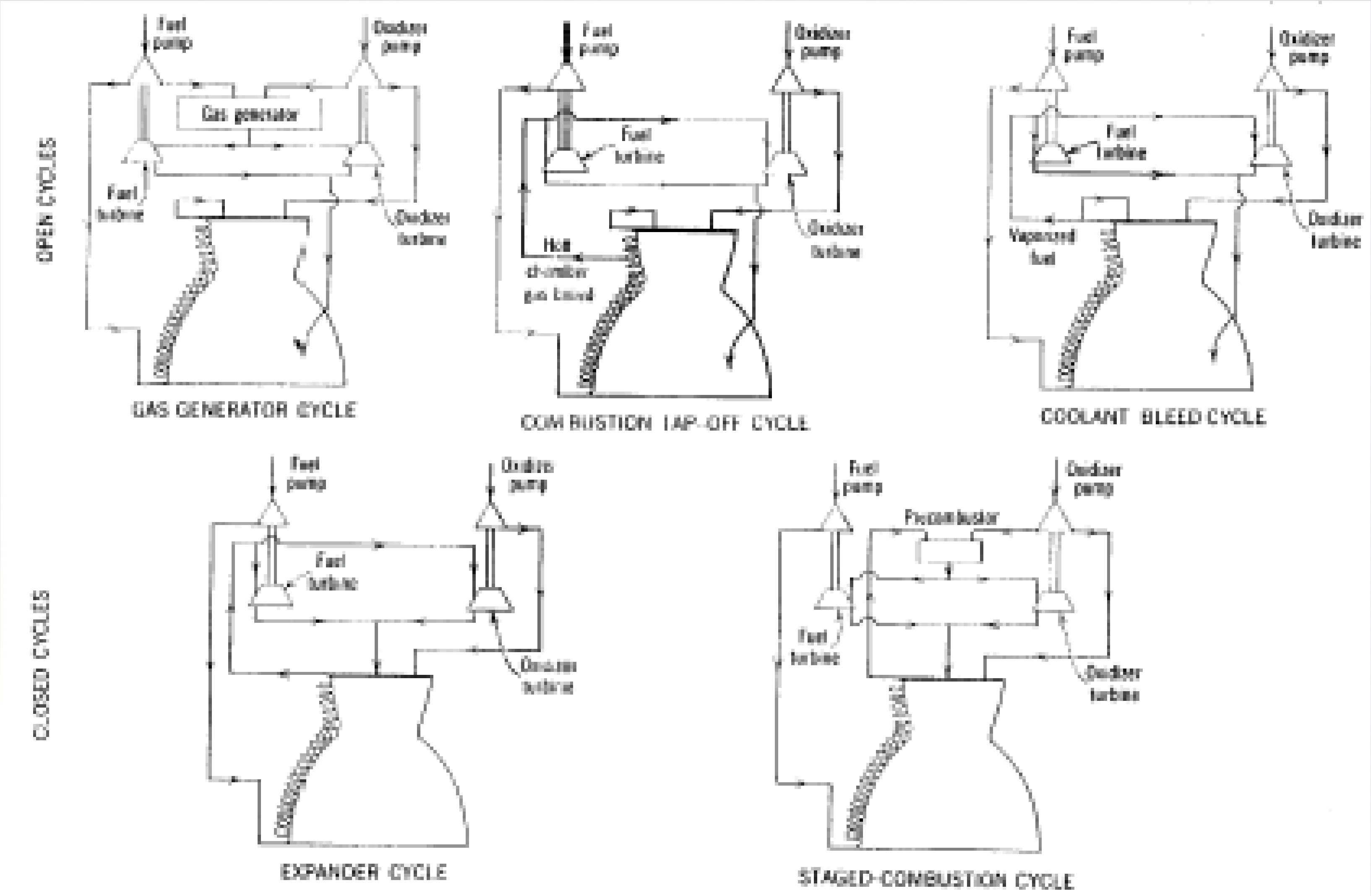
From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

Turbopump Fed Liquid Rocket Engine



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

Sample Pump-fed Engine Cycles



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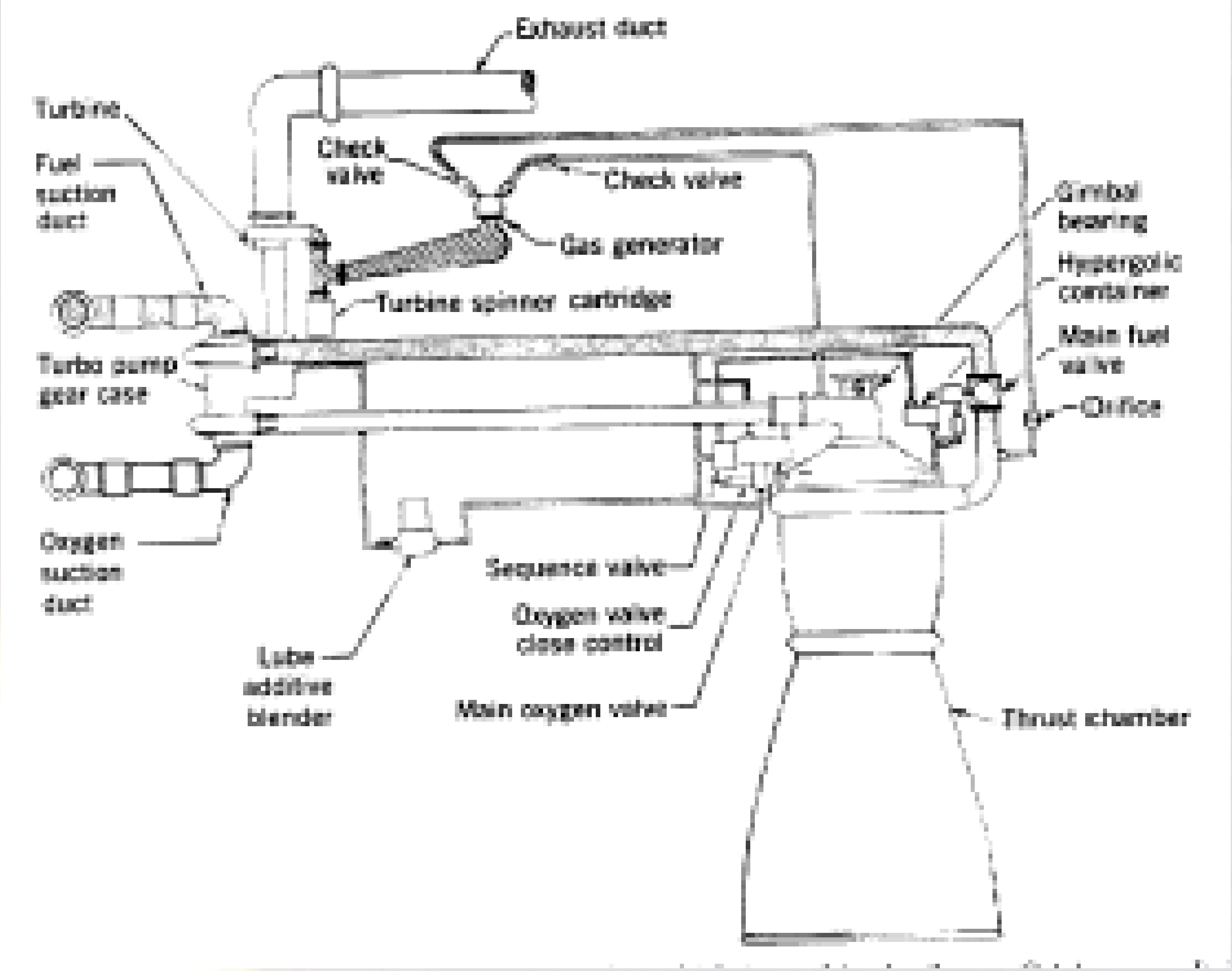
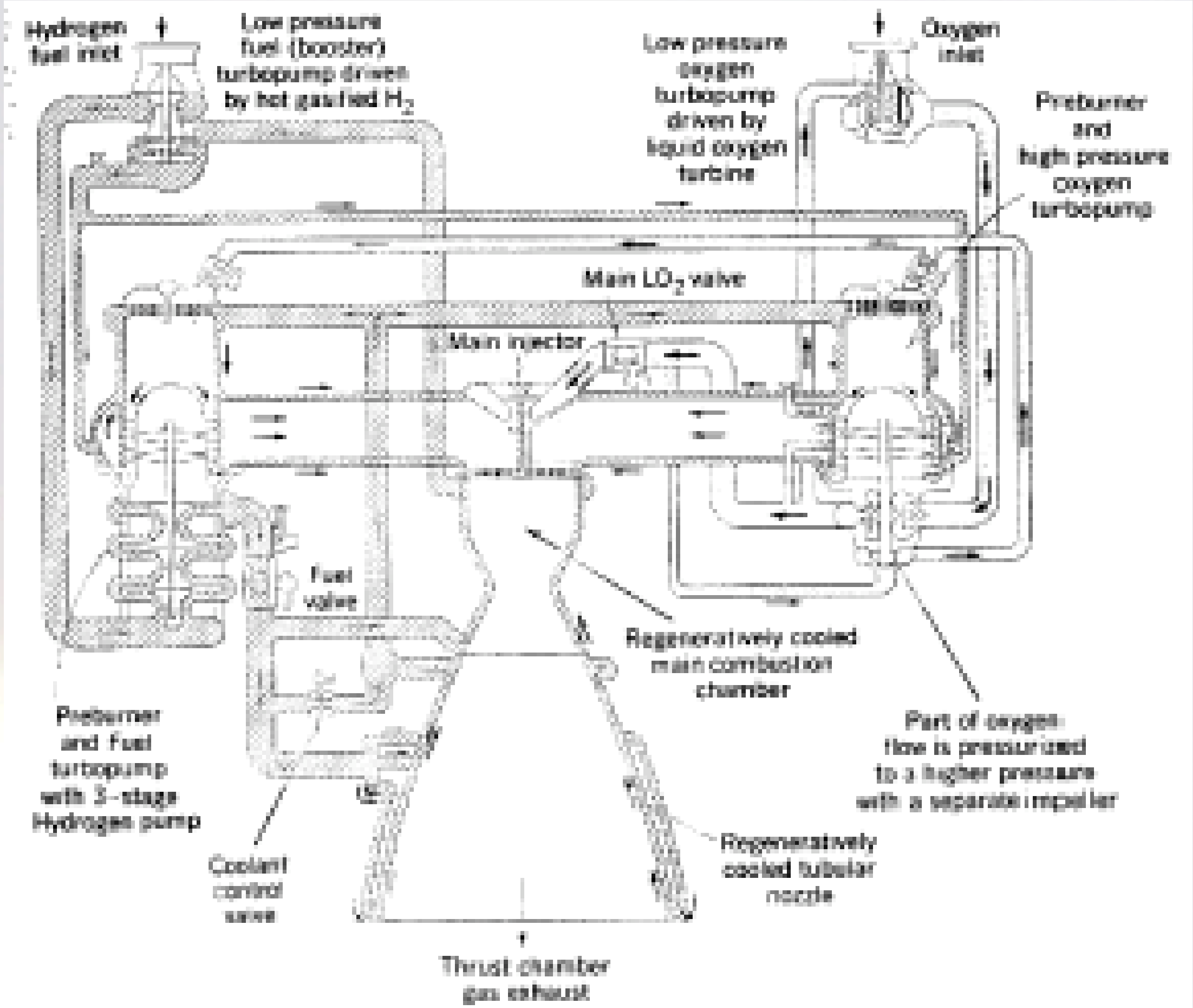
Gas Generator Cycle Engine



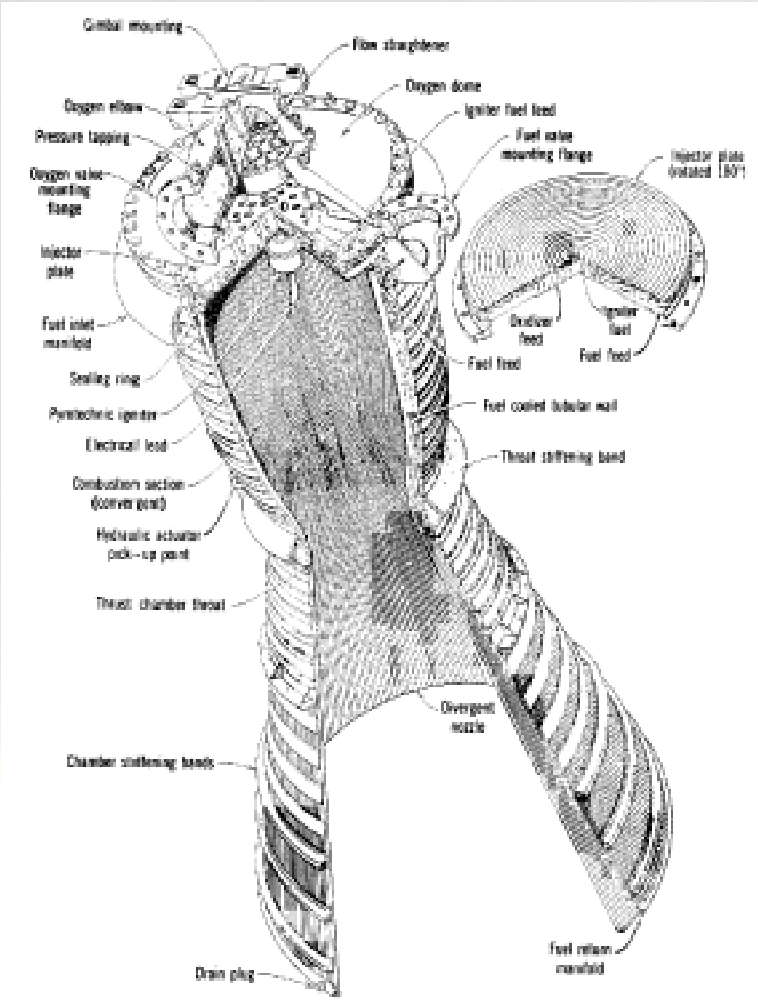
From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

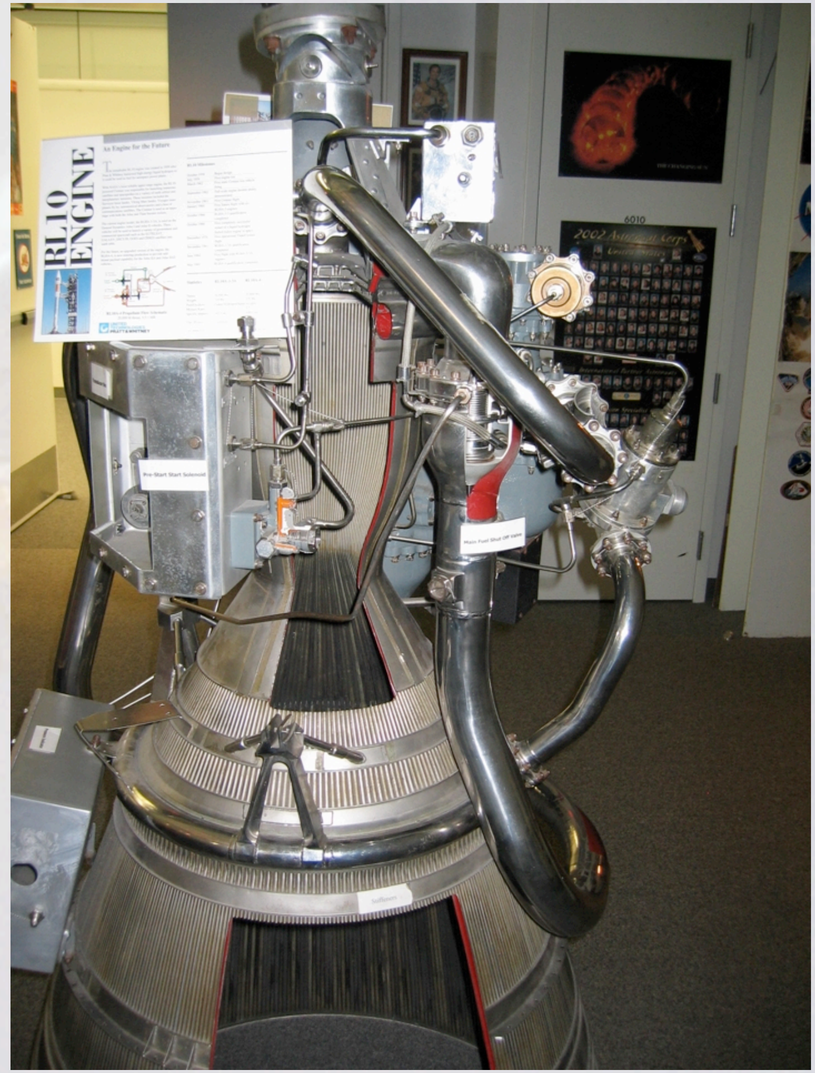
SSME Engine Cycle



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

Liquid Rocket Engine Cutaway

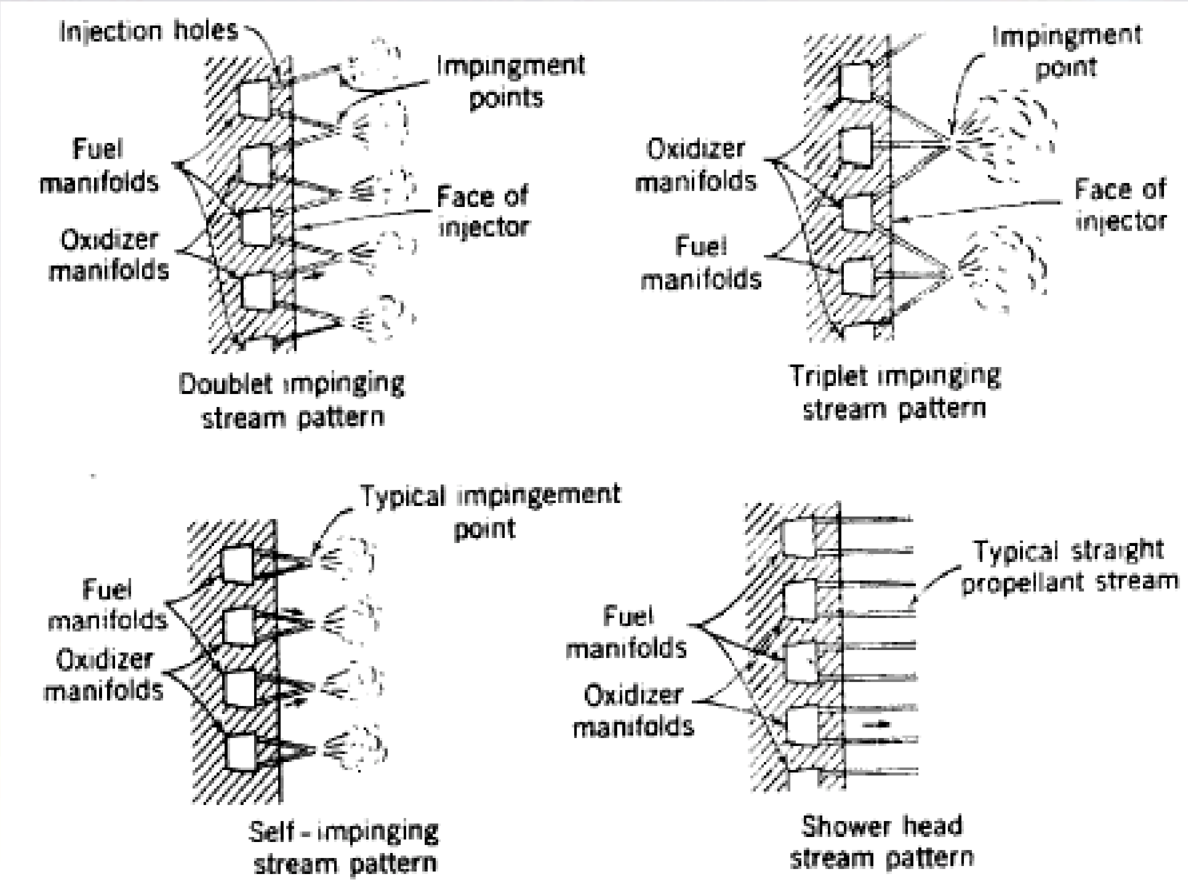




From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

H-1 Engine Injector Plate

Injector Concepts

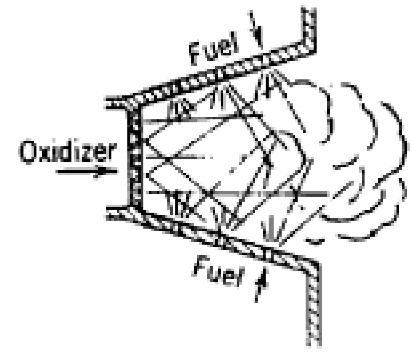


Doublet impinging stream pattern

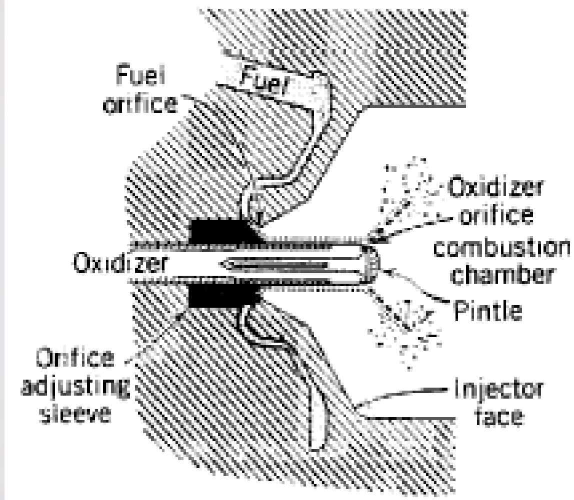
Triplet impinging stream pattern

Self-impinging stream pattern

Shower head stream pattern



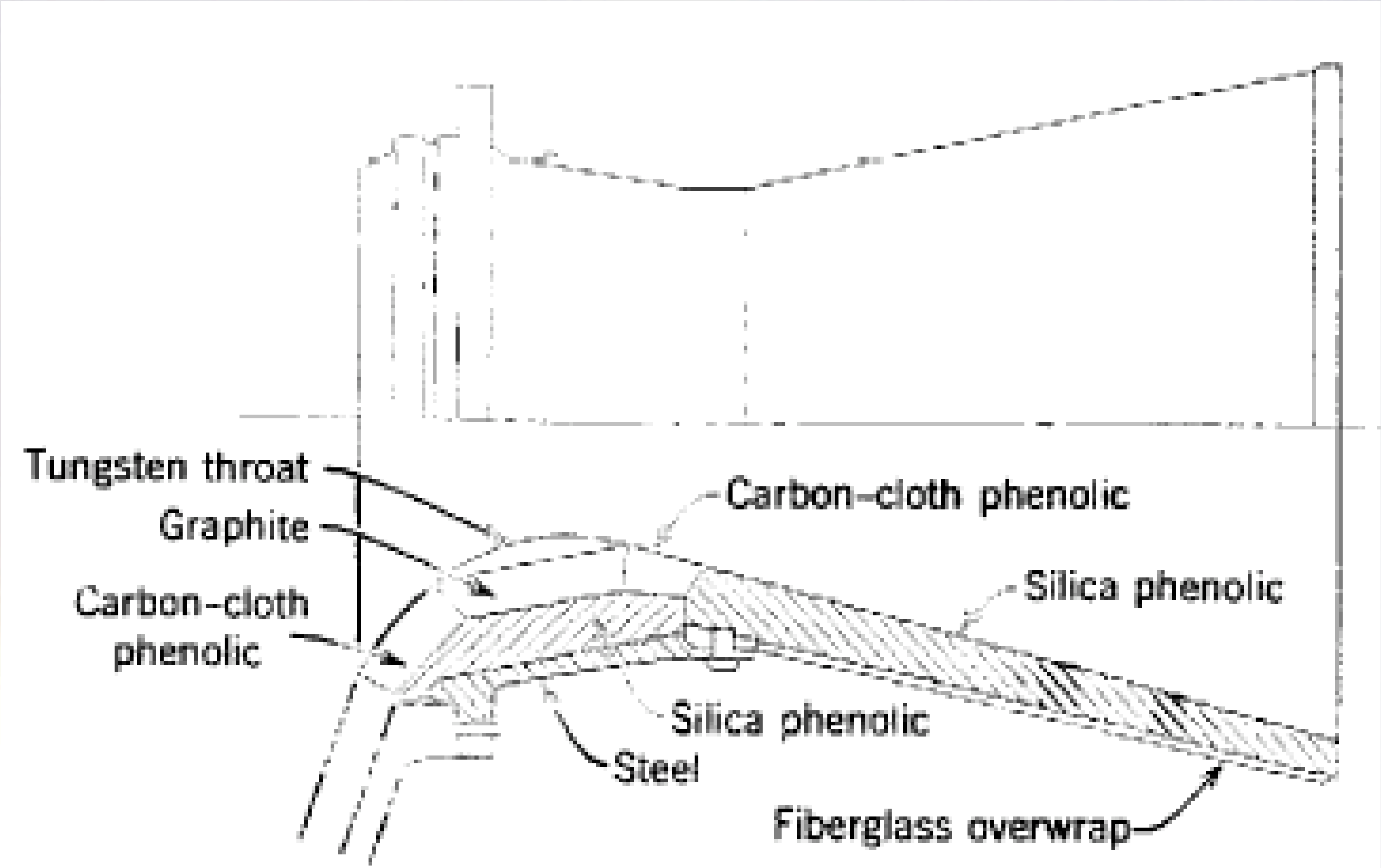
Non-impinging stream pattern



Variable injection area concentric tube injector

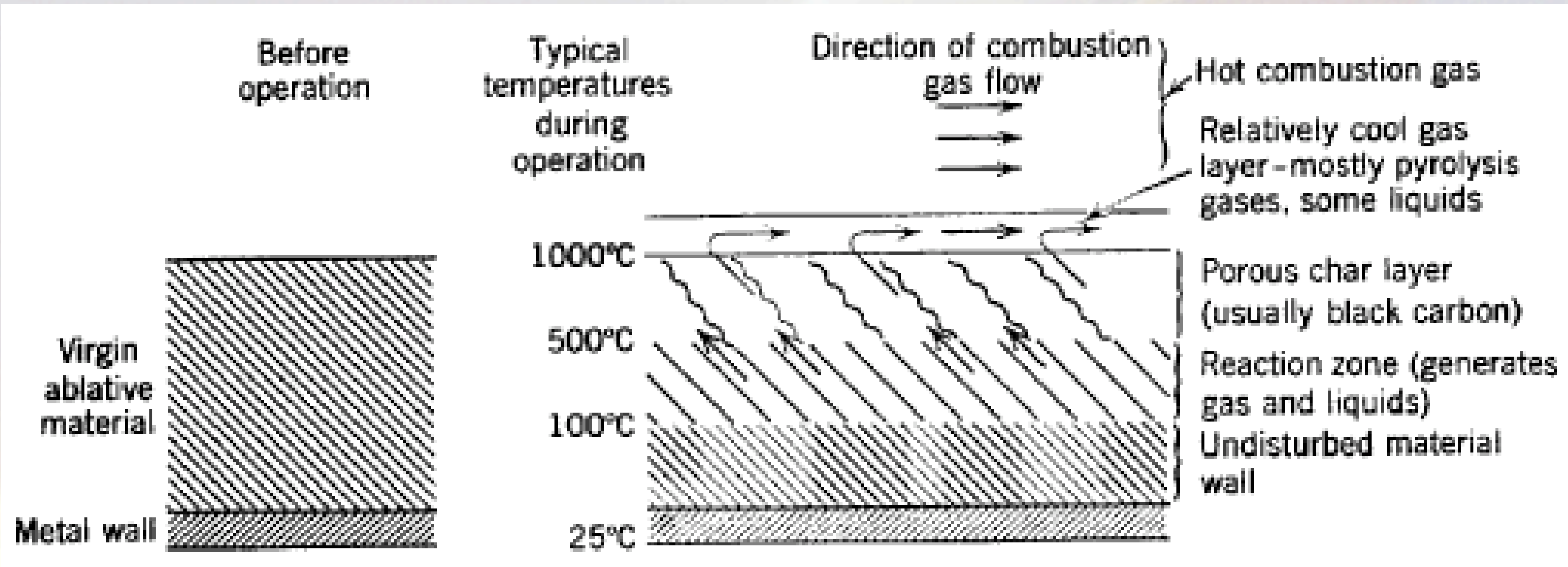
From G. P. Sutton, Rocket Propulsion Elements (5th ed.)
John Wiley and Sons, 1986

Solid Rocket Nozzle (Heat-Sink)



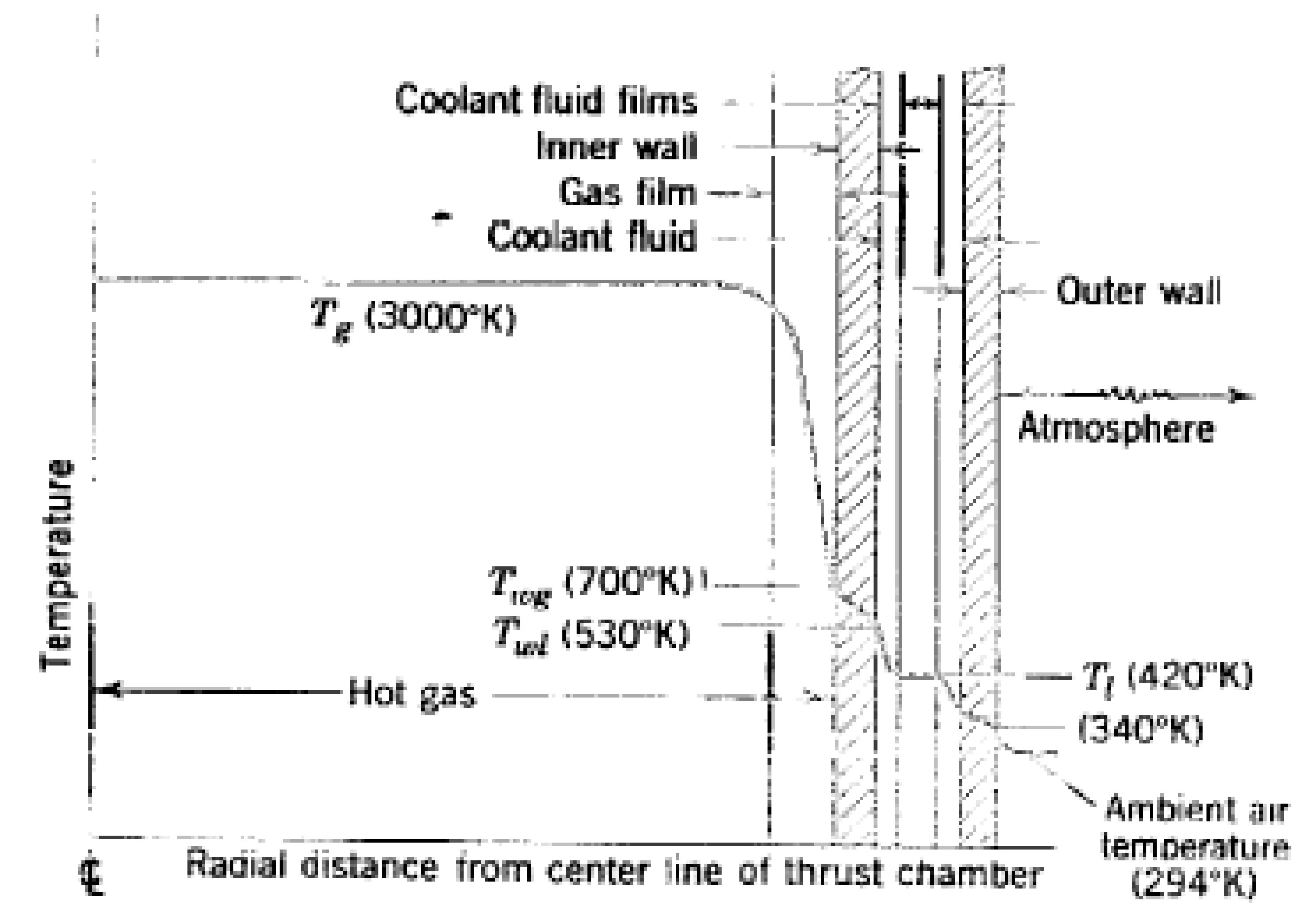
From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

Ablative Nozzle Schematic



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

Active Chamber Cooling Schematic



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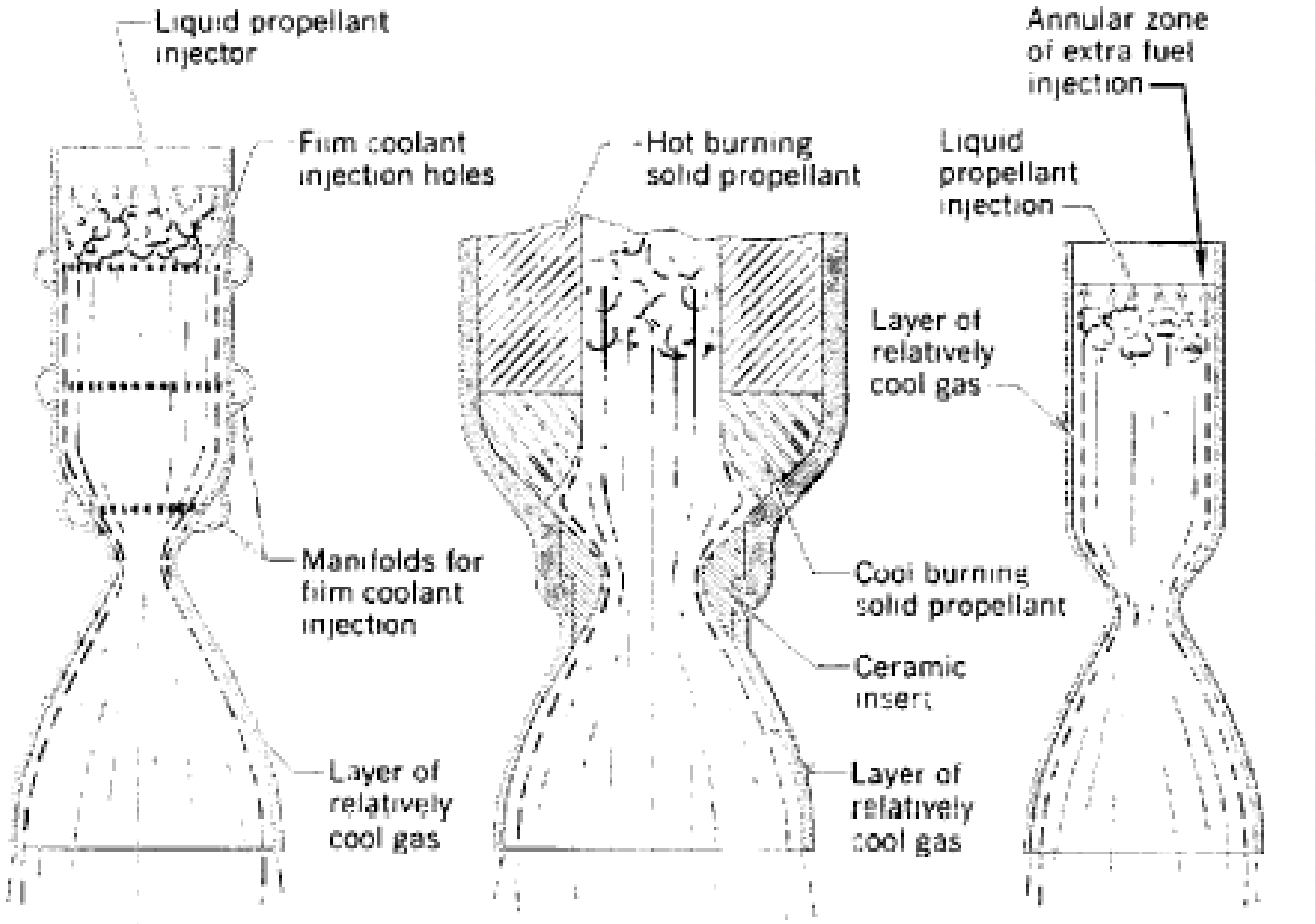
Boundary Layer Cooling Approaches



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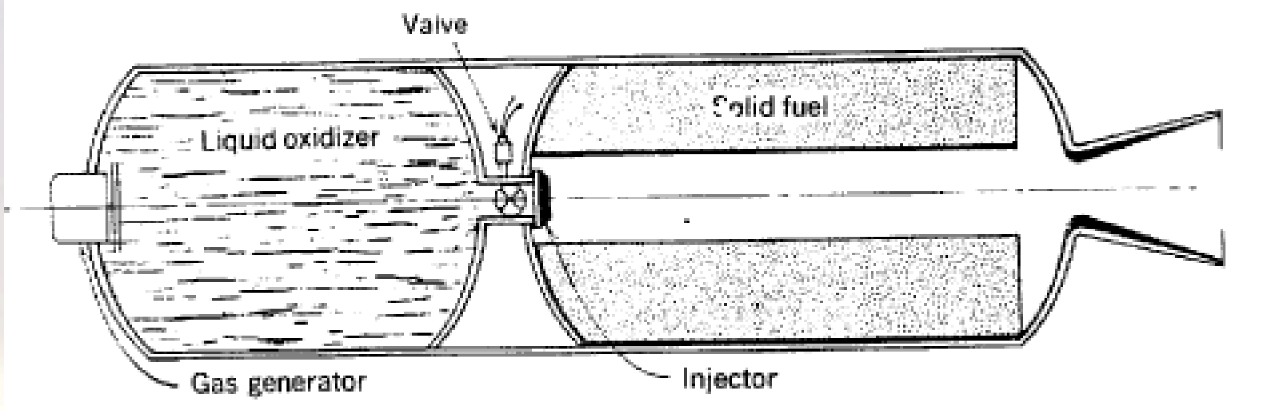
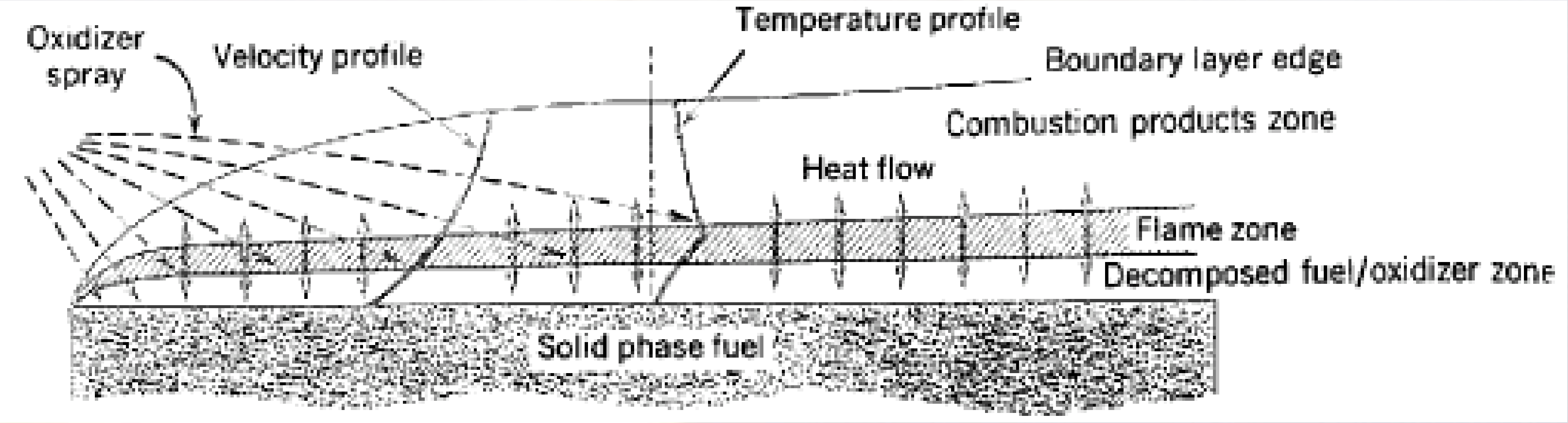
Hybrid Rocket Schematic



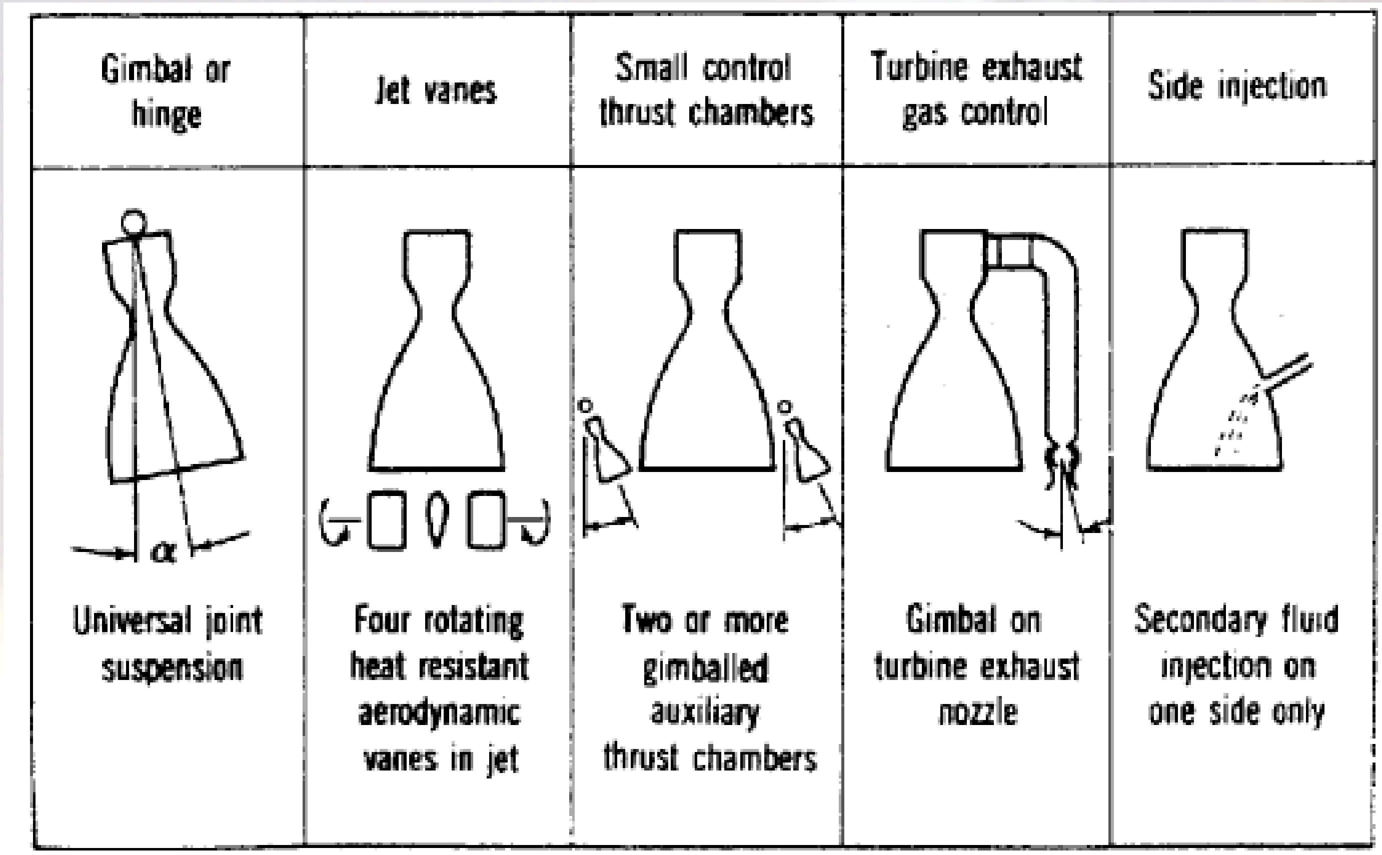
From G. P. Sutton, Rocket Propulsion Elements (5th ed.) John Wiley and Sons, 1986

Hybrid Rocket Combustion



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

Thrust Vector Control Approaches



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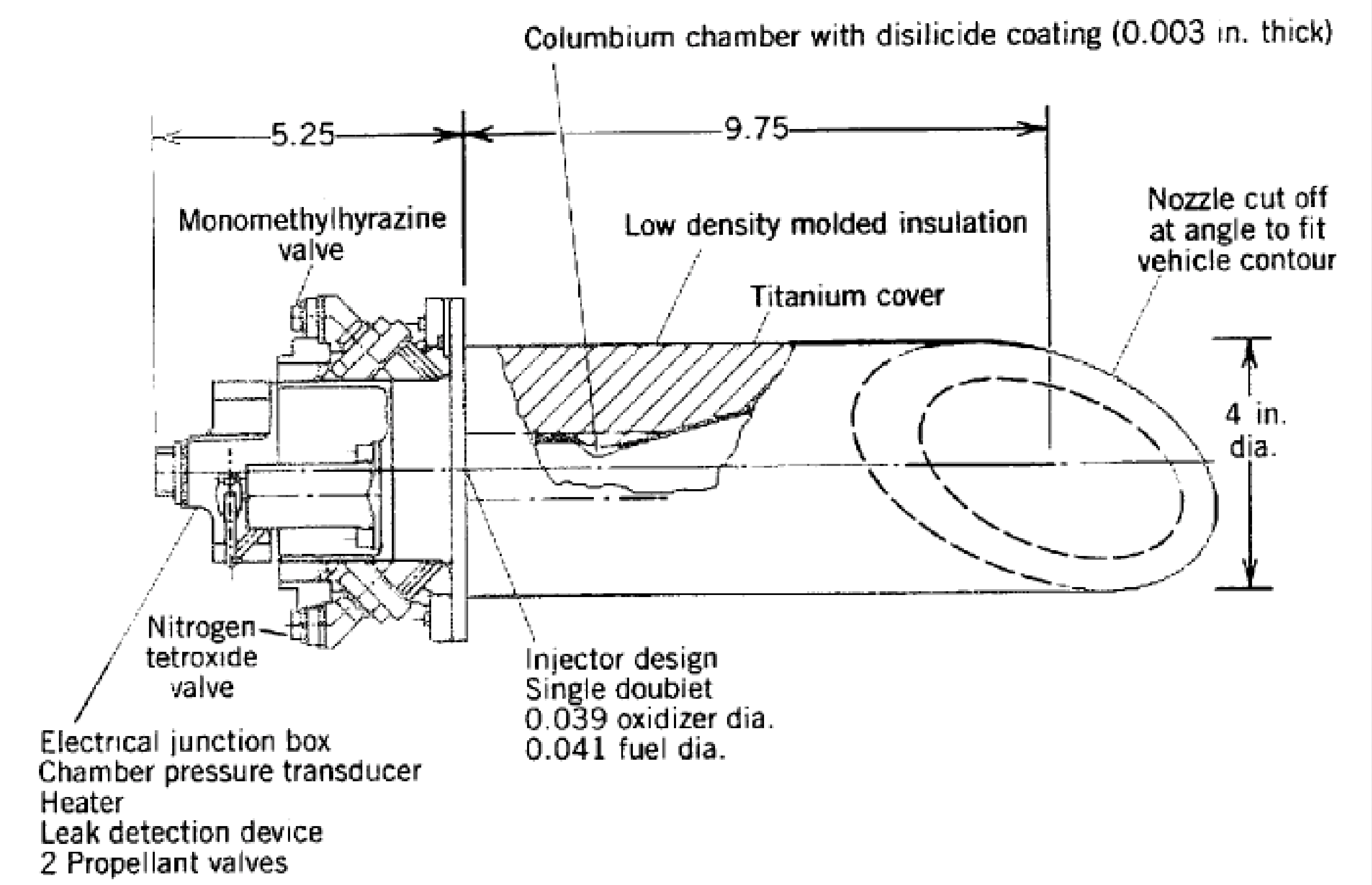
Apollo Reaction Control System Thrusters

Space Shuttle Primary RCS Engine



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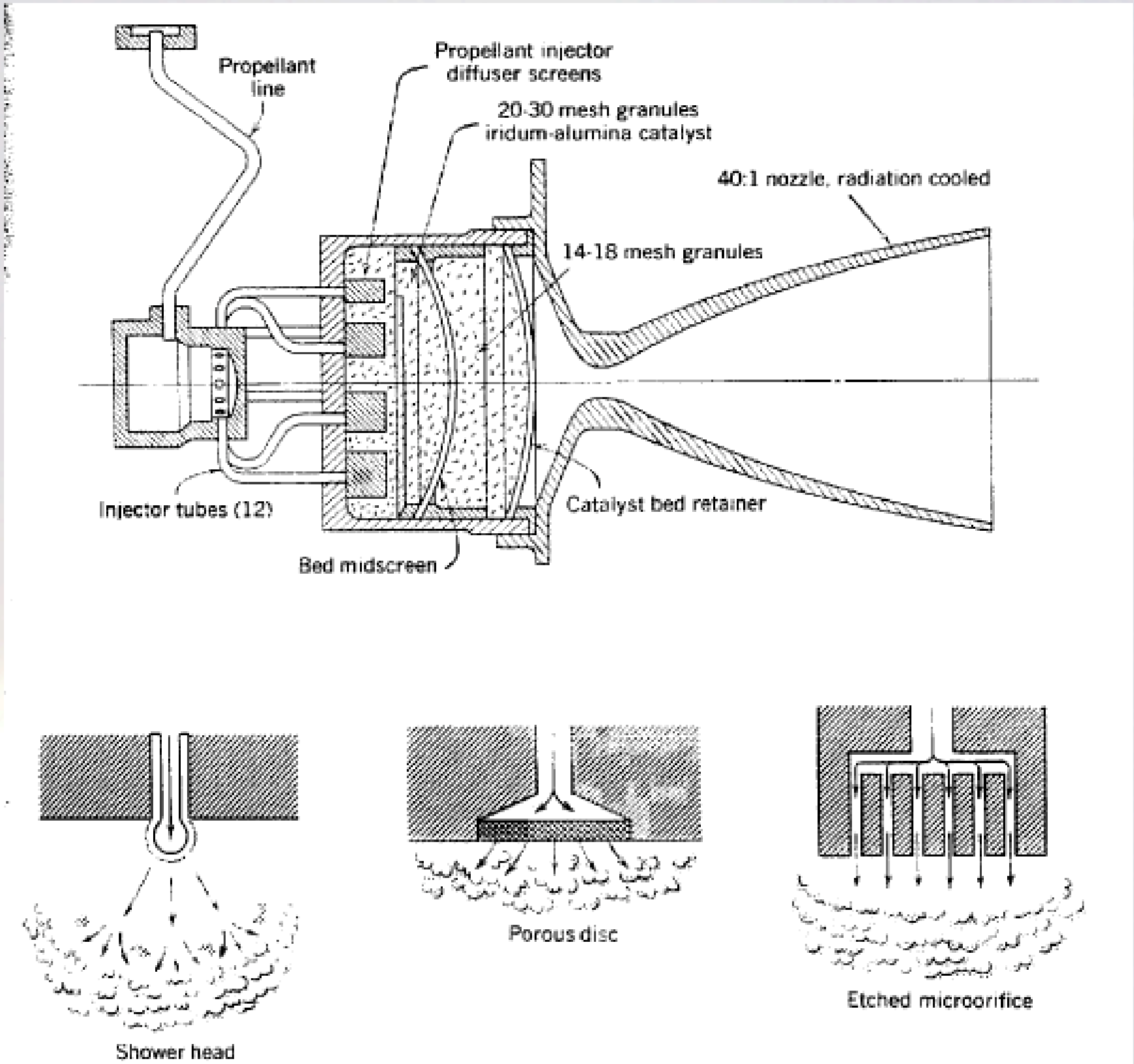
Monopropellant Engine Design



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Cold-gas Propellant Performance

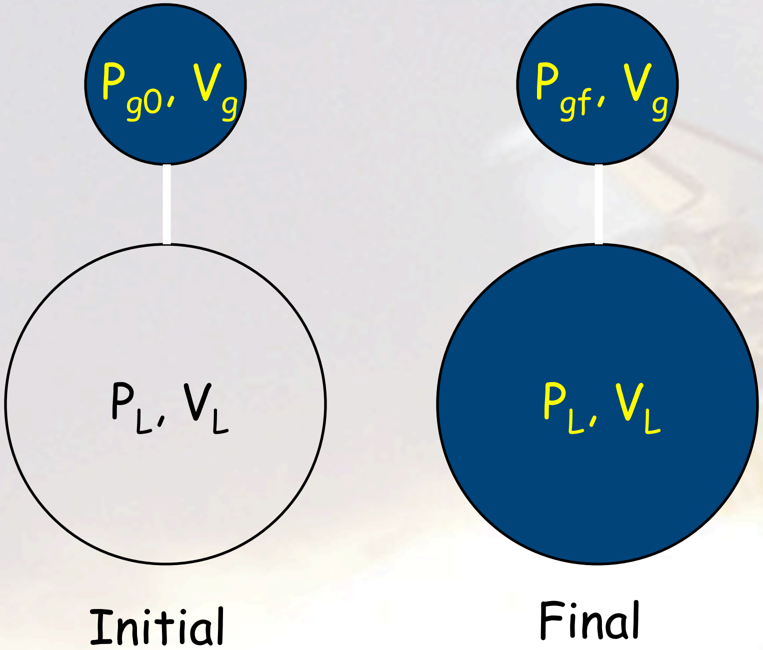
Propellant	Molecular Mass	Density[a] (lb/ft^3)	Theoretical Specific Impulse (sec)
Hydrogen	2.0	1.21	296
Helium	4.0	2.37	179
Methane	16.0	12.10	114
Nitrogen	28.0	17.37	80
Air	28.9	19.3	74
Argon	39.9	27.60	57
Krypton	83.8	67.20	39
Freon 14	88.0	60.01	55
Carbon dioxide	44.0	Liquid	67

[a] At 3500 psia and 0°C.

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

Pressurization System Analysis

Adiabatic Expansion of Pressurizing Gas



$$p_{g,0}V_g^{\gamma} = p_{g,f}V_g^{\gamma} + p_{\ell}V_{\ell}^{\gamma}$$

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

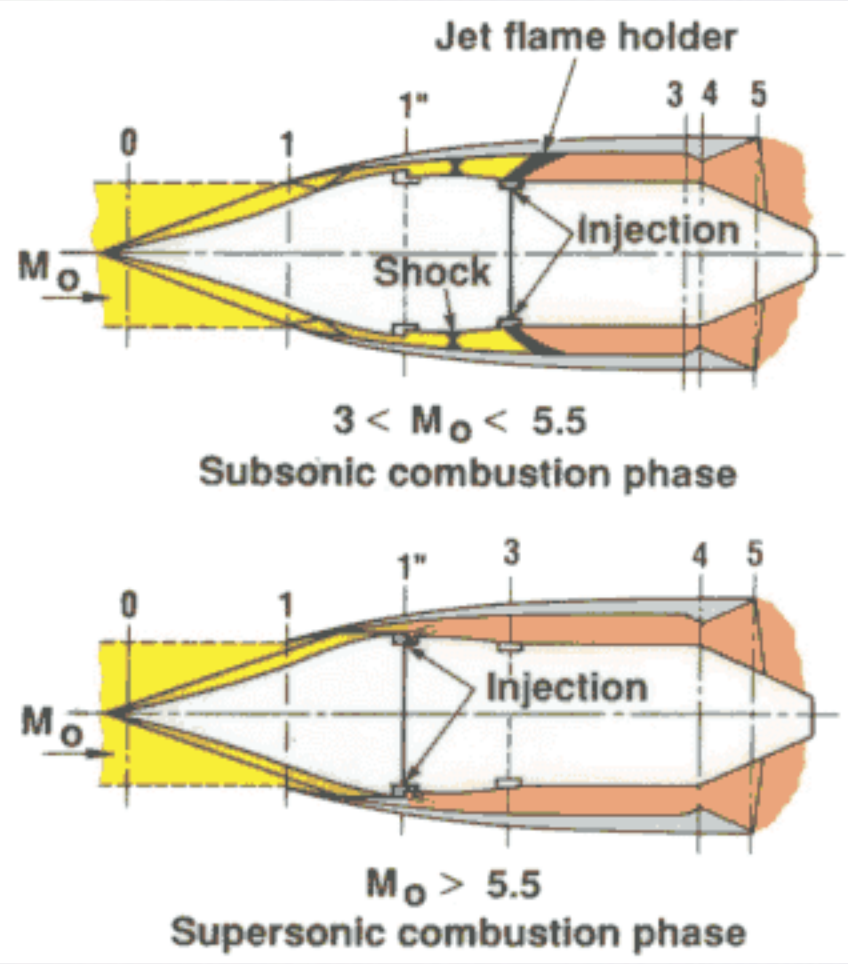
$$(4500\ psi)V_g^{1.67} = (388\ psi)V_g^{1.67} + (312\ psi)\left(17.39\ m^3\right)^{1.67}$$

- V_g = 3.713 m^3
- Ideal gas: T=300°K --> $\rho_{He} = \frac{p_{g,0}\overline{M}}{\Re T_0}$
  ρ=49.7 kg/m^3 (300 psi = 31.04 MPa) M_{He}=185.1 kg

Air-Breathing Propulsion

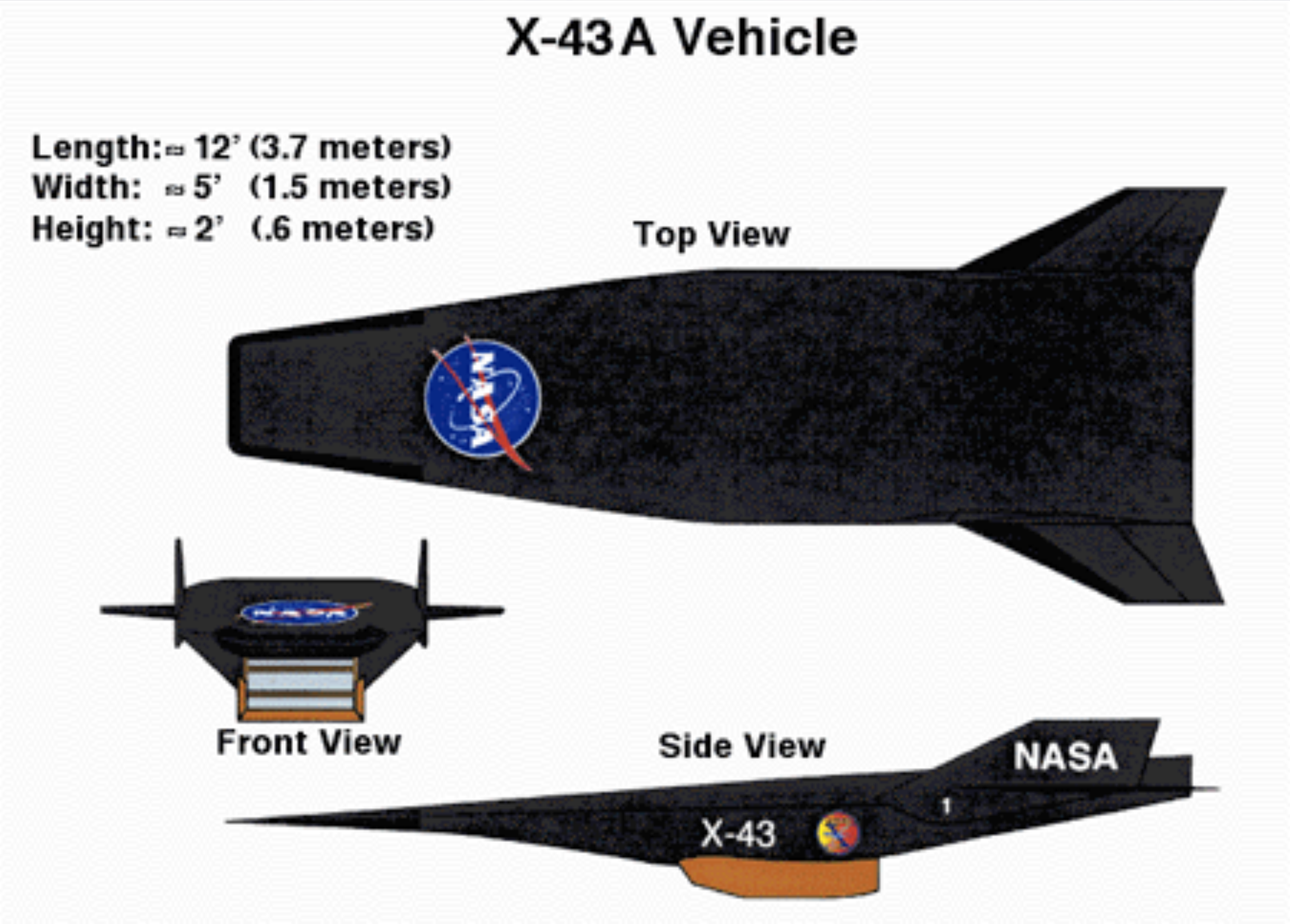
- Much of the total energy required is to accelerate propellants for the last phases of launch
- Typical (LOX/LH2) propellants are 86% oxygen by mass $(I_{sp} = 450\ sec \Rightarrow I_{sp,fuel} = 3150\ sec)$
- Logical conclusion: take in oxygen from ambient air during early portion of launch trajectory

Hypersonic Research Engine



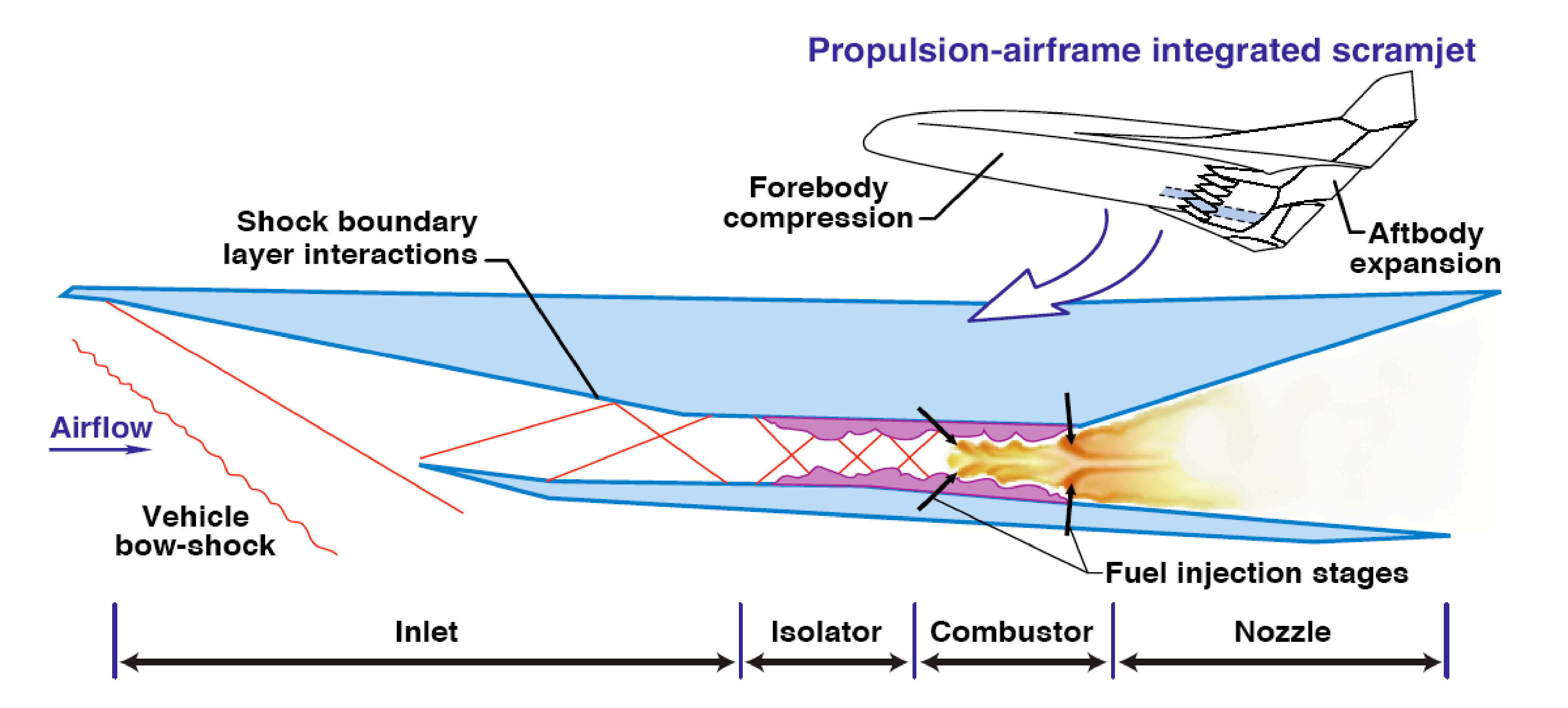
http://www.onera.fr/conferences-en/ramjet-scramjet-pde/#scramjet

X-43A Hypersonic Flight Test



http://hapb-www.larc.nasa.gov/Public/Projects/Hyperx.html

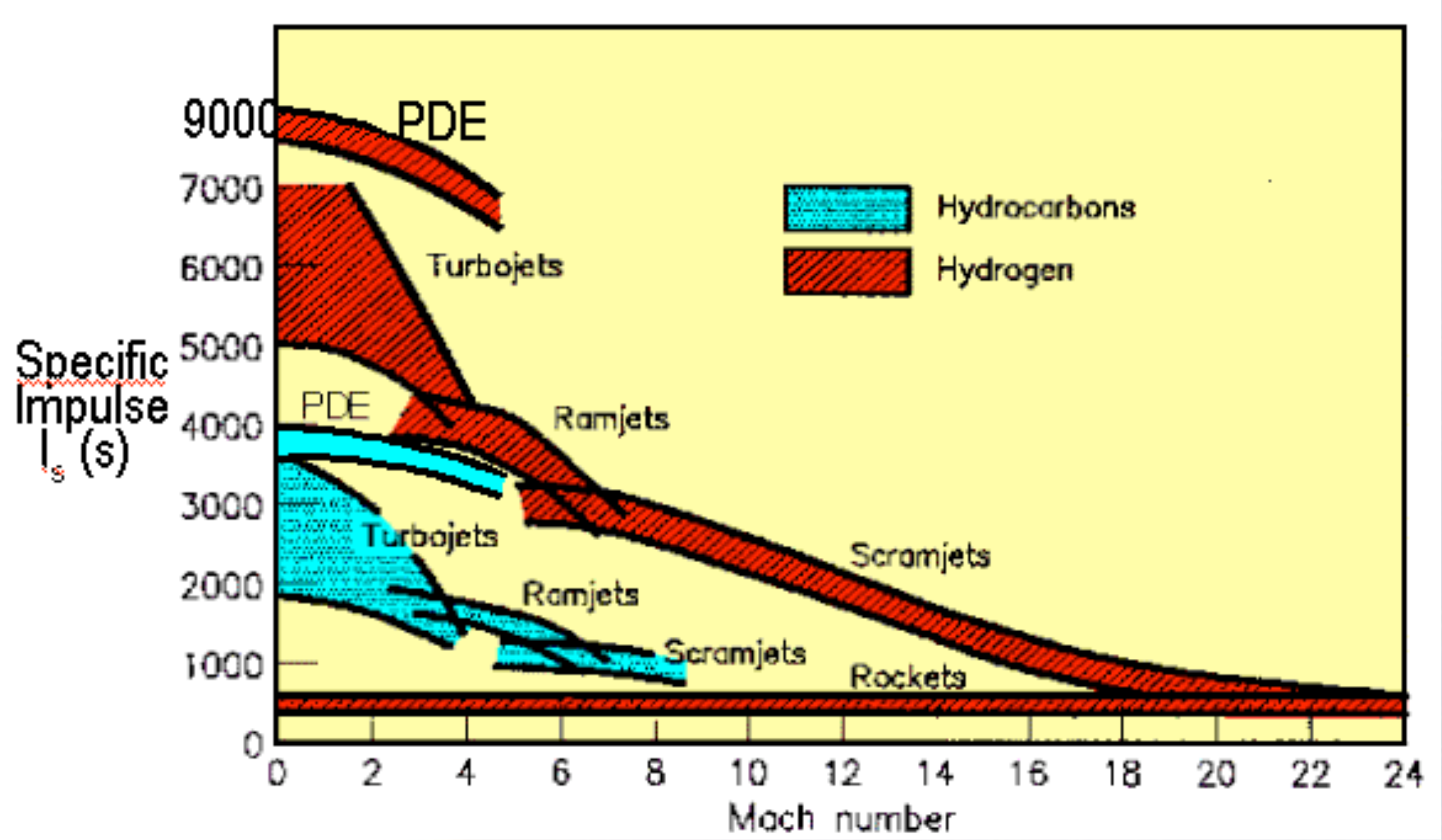
Scramjet-Airframe Integration



“NASA Hyper-X Program Demonstrates Scramjet Technologies” NASA Facts FS-2004-10-98-LaRC

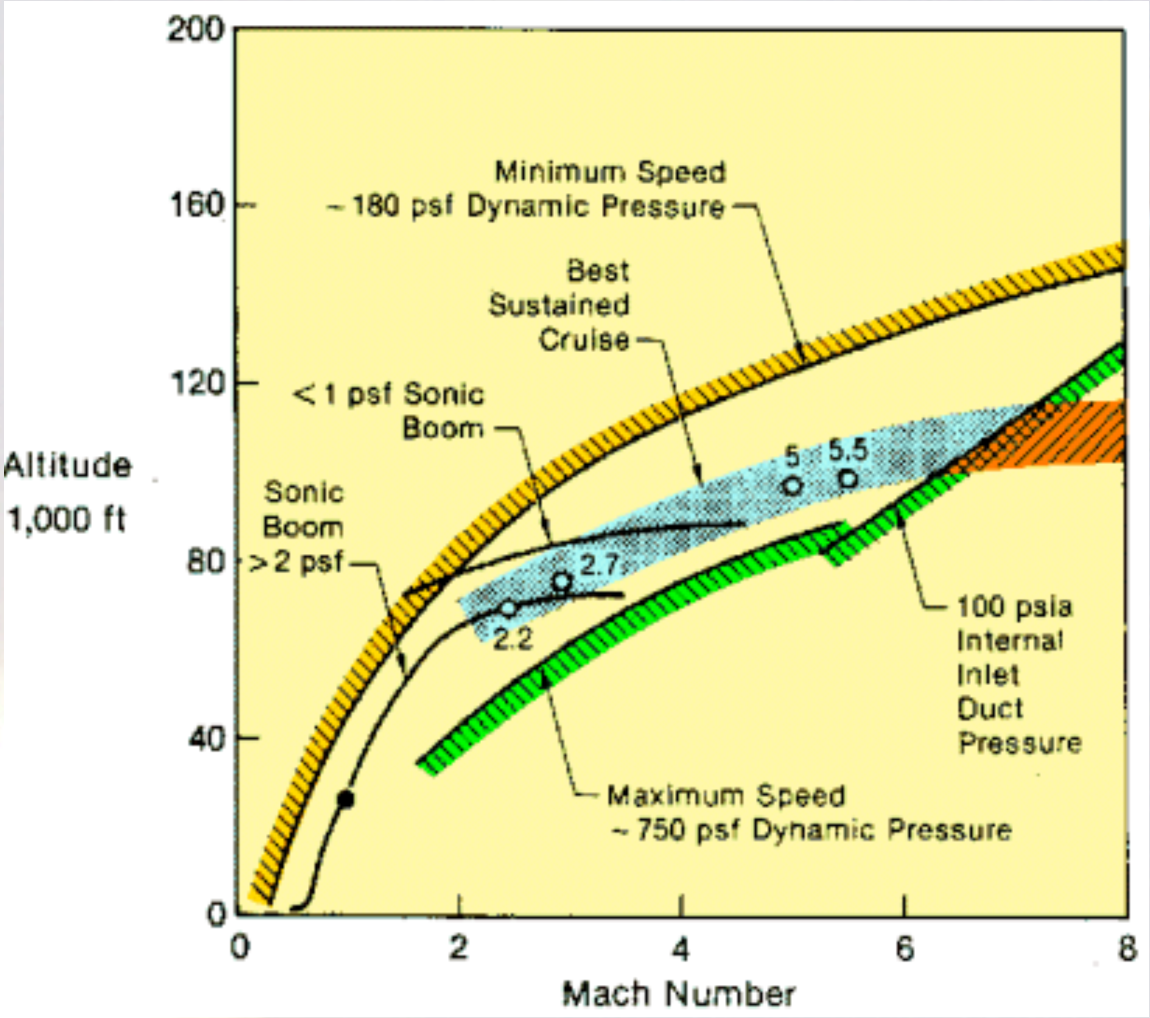
X-43A Film Clip

Performance of Airbreathing Engines



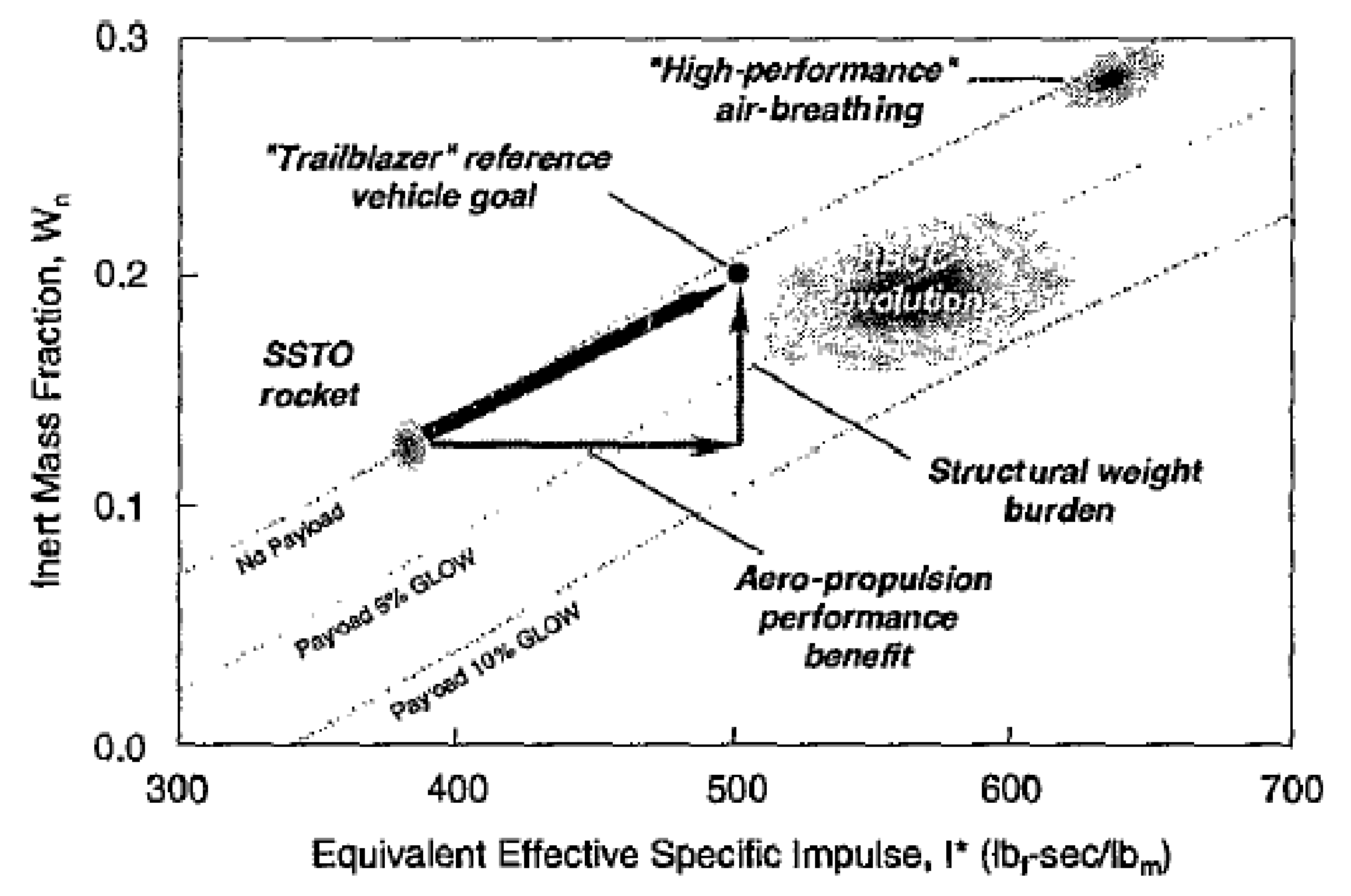
http://www.onera.fr/conferences-en/ramjet-scramjet-pde/#scramjet

Constraints on Airbreathing Trajectories



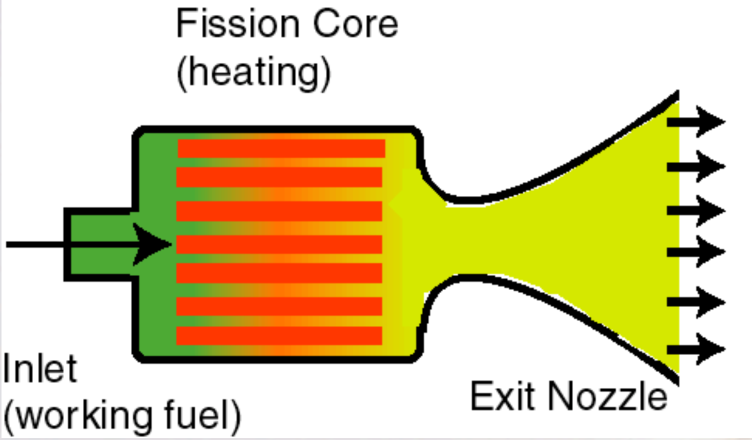
http://www.onera.fr/conferences-en/ramjet-scramjet-pde/#scramjet

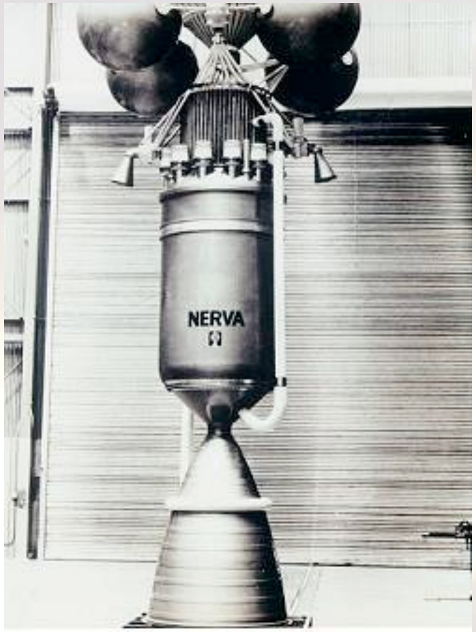
Design Trends of Air-Breathing Propulsion



C. Trefny, "An Air-Breathing Launch Vehicle Concept for Single-Stage-to-Orbit" AIAA 99-2730

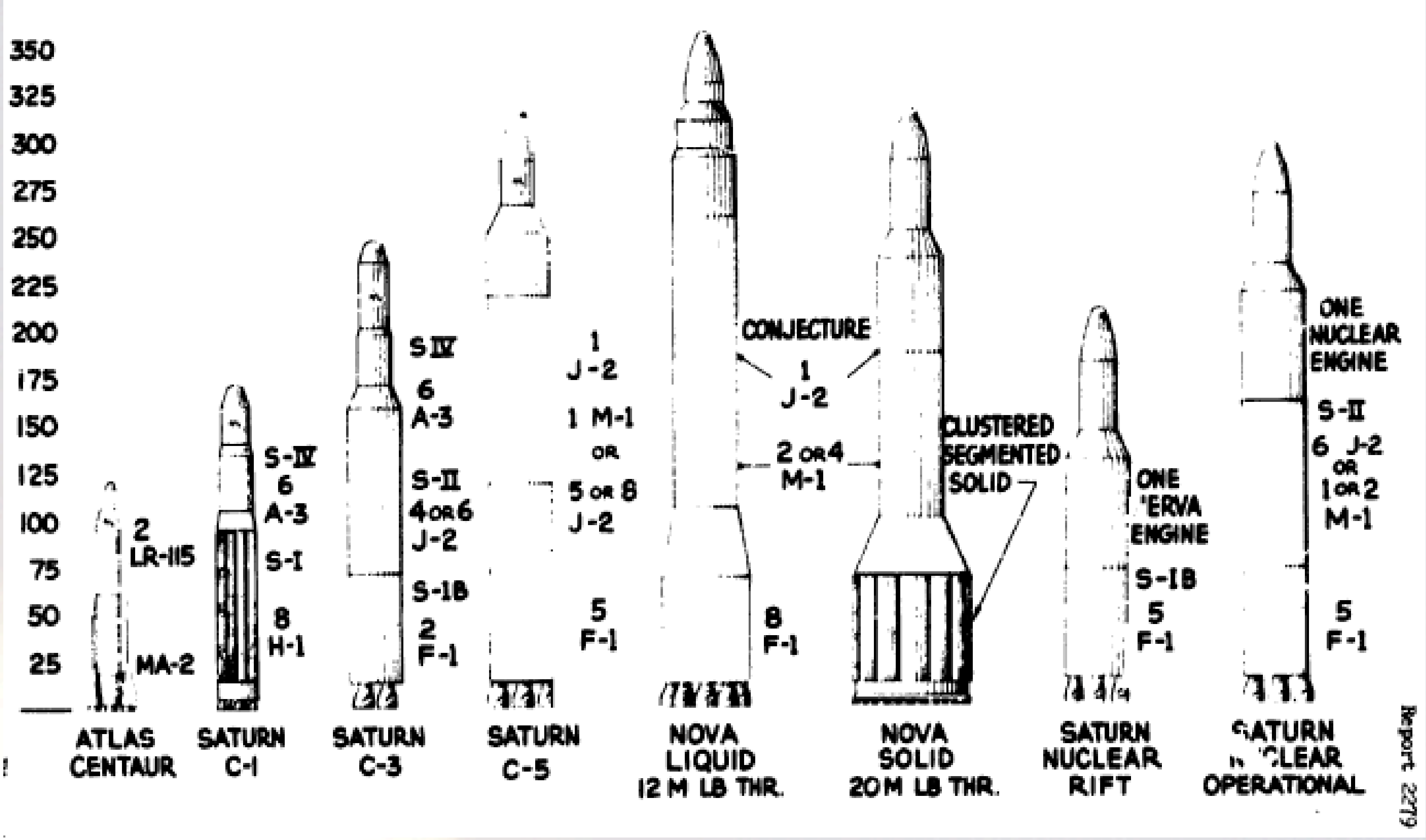
Nuclear Thermal Rockets





- 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

Speculative Designs Including Nuclear



Aerojet General, "Payload, Cost, and Reliability Analysis of Saturn C-5 and Nova with NERVA or Chemical Third Stages" AGC-2279, June 1962