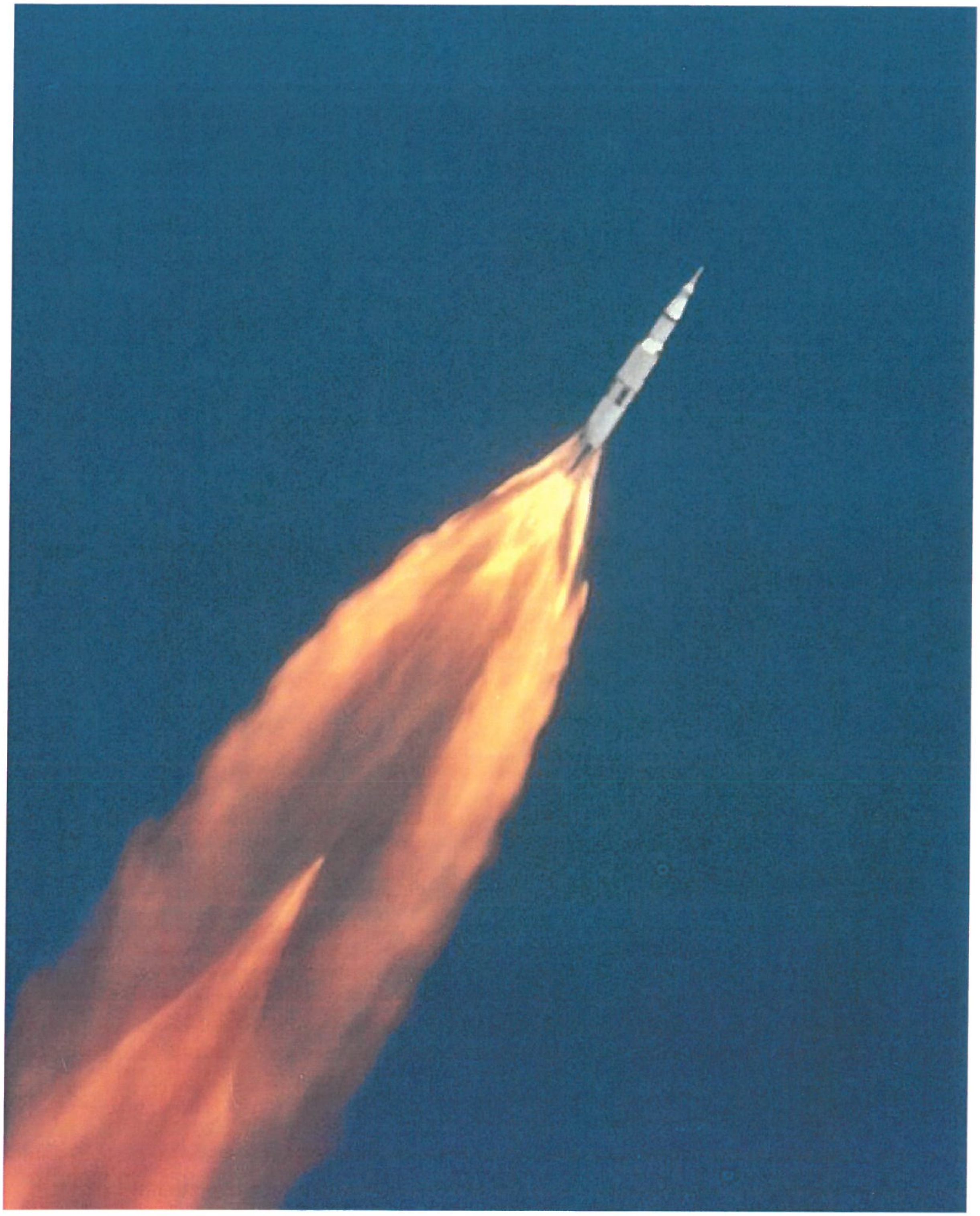
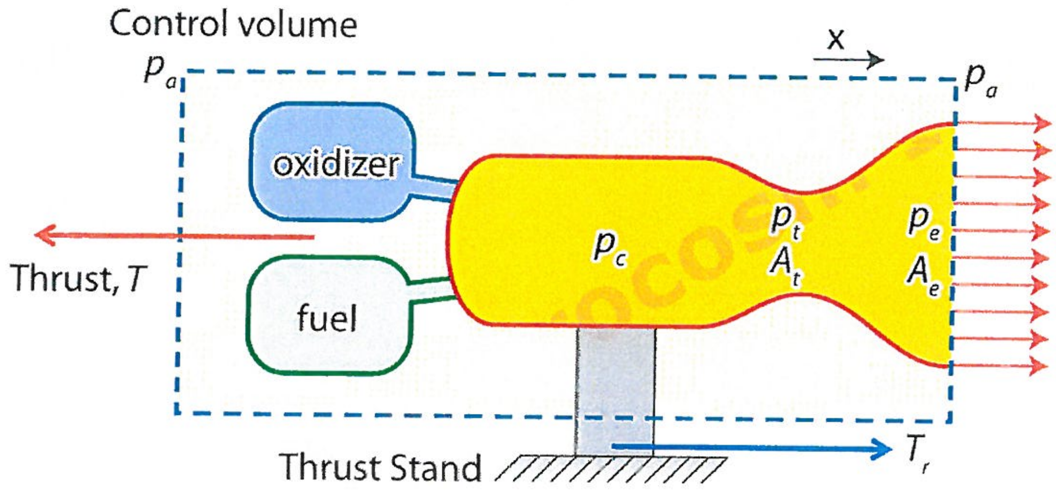
Propulsion Systems



Text 18.1: Basic Rocket Equations



$\dot{m}_e = \rho_e V_e A_e$

(the structure holding the rocket in place feels a reaction force equal in magnitude to the thrust but in the opposite direction)

- Rocket engines produce thrust, T.

  We'd like to understand which variables influence T.

- Newton's 2nd Law: $\sum \vec{F} = \frac{d}{dt}(m\vec{V})$

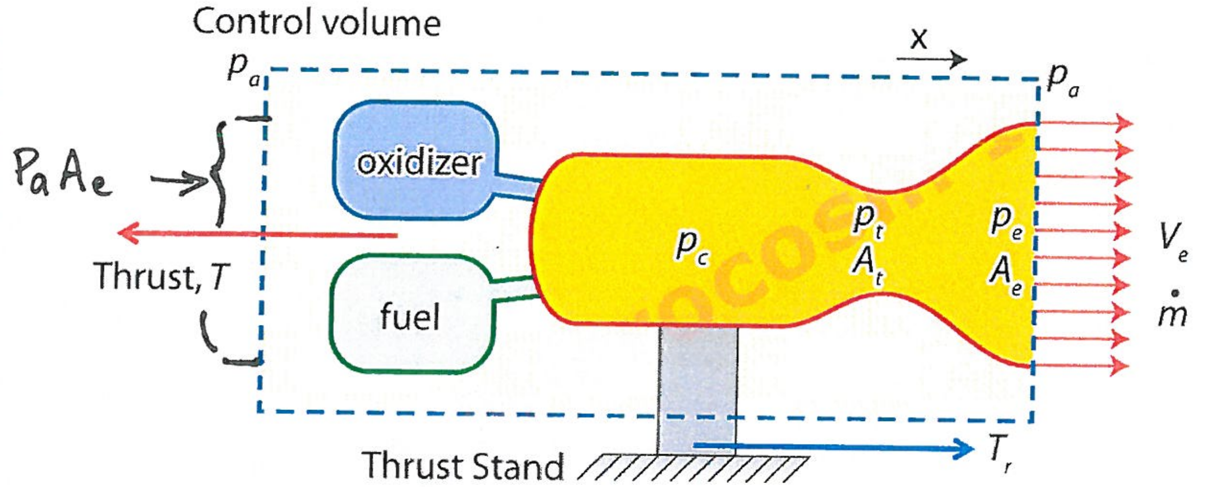
  for a set of mass particles

- Transform to Control Volume using Reynolds Transport Theorem:

$$\sum \vec{F} = \frac{d}{dt}(m\vec{v}) = \underbrace{\frac{d}{dt}\left(\int_{cv} \vec{V}\rho \, d\forall\right)}_{=0 \text{ steady}} + \underbrace{\int_{cs} \vec{V}\rho(\vec{V}\cdot\vec{n})\,dA}_{\text{discrete, 1D outlet}}$$

$$= (\dot{m}\vec{V})_{out} - \cancel{(\dot{m}\vec{V})_{in}} \rightarrow 0$$

Text 18.1: Basic Rocket Equations



(the structure holding the rocket in place feels a reaction force equal in magnitude to the thrust but in the opposite direction)

- Continue x-momentum balance:

$$\sum F_x = (\dot{m} V_x)_{out}$$ where $\dot{m}$ = exhaust mass-flow rate ~~[illegible]~~
$= \rho_e V_e A_e$

$$T_r + P_a A_e - P_e A_e = \dot{m} V_e$$

$$T_r = \dot{m} V_e + A_e (P_e - P_a)$$ acting in $+x$ direction

or $$\boxed{T = |T_r| = \underbrace{\dot{m} V_e}_{\text{momentum thrust}} + \underbrace{A_e (P_e - P_a)}_{\text{pressure thrust}}}$$ acting in $-x$ direction

generally, $T_{press} \ll T_{mom}$

<u>note</u>: V_e = velocity of exhaust relative to rocket

$P_a \Rightarrow 0$ outside atmosphere

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- So Thrust $T = \dot{m} V_e + A_e (P_e - P_a)$

How to maximize T?

- Exit flow is a supersonic nozzle, so could increase A_e → increase V_e

  but decrease P_e

  (don't want $(P_e - P_a) < 0$)

  So large A_e OK in vacuum (2^{nd} stage)

  Also, larger nozzle → weight!

- For fixed geometry and $\dot{m}$, could increase V_e by increasing chamber temp and pressure by changing propellant properties/mixture.

  But higher P_c and T_c require more strength (weight) and more cooling (cost, complexity, efficiency reduced).

- Could increase $\dot{m}$, but go through fuel more quickly.

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Measures of rocket performance

- "Effective exhaust velocity"

$$\boxed{V_{eq} \equiv \frac{T}{\dot{m}} = V_e + \left(\frac{P_e - P_a}{\dot{m}}\right) A_e}$$

One measure of how efficiently the engine converts propellant mass to thrust

- "Characteristic exhaust velocity"

$$\boxed{C^* \equiv \frac{P_c A_t}{\dot{m}}}$$

P_c = chamber pressure

A_t = throat area

$\dot{m}$ = prop. mass flow rate

A measure of combustion efficiency.

Independent of nozzle design.

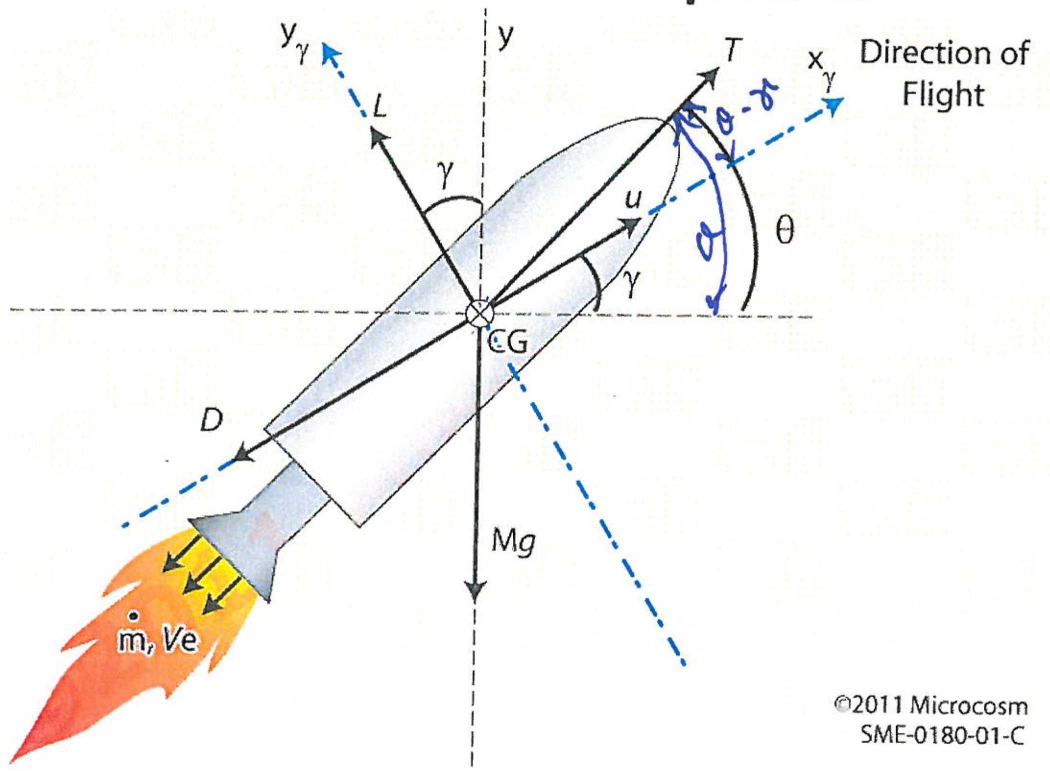
Depends mostly on the chemistry of the particular propellants.

$$\frac{C^*_{experiment}}{C^*_{theory}} \sim 90-98\%$$

$C^*_{experiment}$ → from ground experiments

C^*_{theory} → from thermochemistry theory

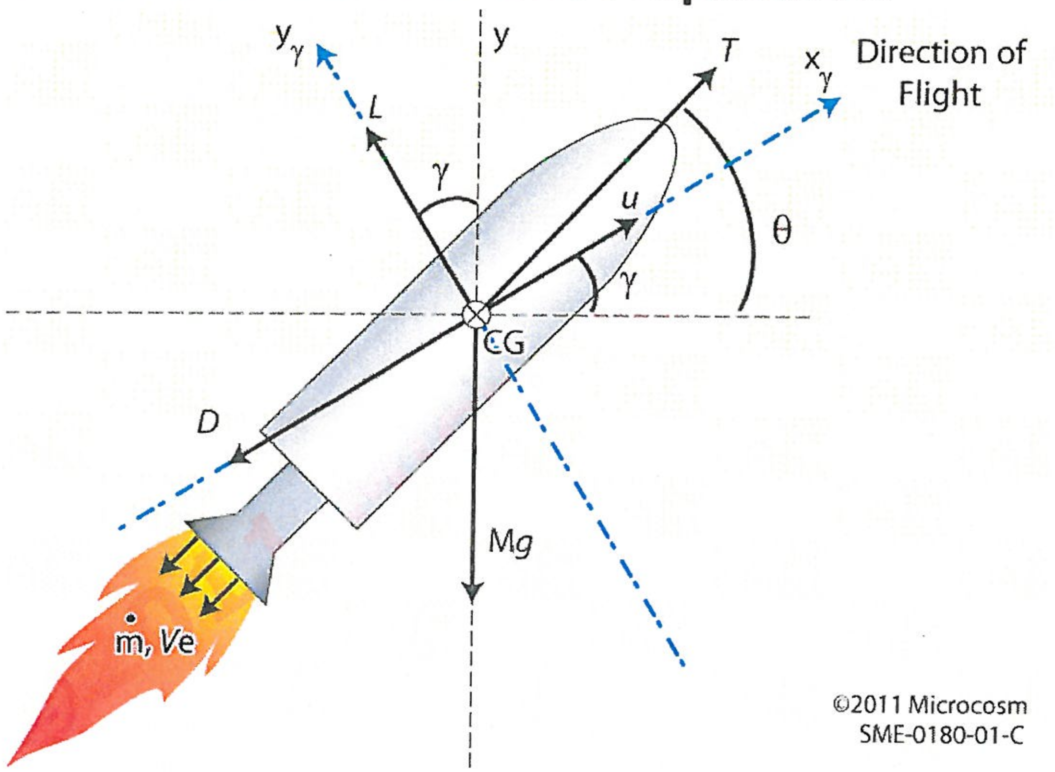
Text 18.1: Basic Rocket Equations



Derive the Ideal Rocket Equation

- Most general case: T: thrust (acting along rocket axis)
  Mg: weight of rocket (+ propellant on board)
  D: atmospheric drag
  L: atmospheric lift
- Note that direction of flight ≠ direction of thrust
- All variables are dynamic functions of time, so force/momentum balance is instantaneous
- For real rockets launching from Earth, want $\frac{T}{mg} \gtrsim 1.3$

Text 18.1: Basic Rocket Equations



- Newton's 2^{nd} Law in direction of flight (γ):

$$\Sigma F_\gamma = M \frac{dV}{dt} = \underbrace{T \cos(\theta - \gamma)}_{\text{thrust in direction of flight}} - \underbrace{D}_{\text{drag}} - \underbrace{Mg \sin \gamma}_{\text{weight component}}$$

(note lift term is $\perp$ to direction of flight.)

where $T = \underbrace{\dot{m} V_e}_{\text{momentum thrust}} + \underbrace{A_e (P_e - P_a)}_{\text{pressure thrust}}$

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- Calculate velocity (V_f) attained by a rocket after a given time, integrate momentum eqn from an initial time (t_o) to a final time (t_f):

$$\frac{dV}{dt} = \frac{T}{M} - \frac{T}{M}\left[1-\cos(\theta-\gamma)\right] - \frac{D}{M} - g\sin\gamma$$

$$\int_{V_o}^{V_f} dV = \int_{t_o}^{t_f} \frac{T}{M}\,dt - \int_{t_o}^{t_f} \frac{T}{M}\left[1-\cos(\theta-\gamma)\right]dt - \int_{t_o}^{t_f} \frac{D}{M}\,dt - \int_{t_o}^{t_f} g\sin\gamma\,dt$$

or:

$$\Delta V = \underbrace{\Delta V_{prop}}_{\text{ideal velocity gain due to rocket thrust}} \underbrace{- \Delta V_{steering} - \Delta V_{drag} - \Delta V_{gravity}}_{\text{thrust losses due to angle of attack, drag, and gravity}}$$

$\downarrow$

$\cong$ 1300 - 1700 m/s for LEO

launches to

Text 18.1: Basic Rocket Equations

- Simplify ΔV equation:

Substitute $V_{eq} \equiv \frac{T}{\dot{m}}$

and realize $\underbrace{-\frac{dM}{dt}}_{\text{rate at which rocket loses mass}} = \underbrace{\dot{m}}_{\text{mass flow rate leaving rocket as exhaust}}$

Also, let $\theta = \gamma$

neglect drag D

neglect gravity

So now,

$$\Delta V = \int_{t_0}^{t_f} \frac{T}{M}\, dt - 0 - 0 - 0$$

$$= V_{eq} \int_{t_0}^{t_f} \frac{\dot{m}}{M}\, dt$$

$$= V_{eq} \int_{M_0}^{M_f} \frac{dM}{M} \longrightarrow$$

Text 18.1: Basic Rocket Equations

$$\Delta V = V_{eq} \int_{M_0}^{M_f} \frac{1}{M} \, dM$$

$$\boxed{\begin{aligned} \Delta V &= -V_{eq} \ln\left(\frac{M_f}{M_0}\right) = V_{eq} \ln\left(\frac{M_0}{M_f}\right) \\ &= V_{eq} \ln\left(\frac{M_0}{M_0 - M_p}\right) \end{aligned}}$$

<u>Ideal</u> <u>Rocket</u> <u>Equation</u>

- where $V_{eq} \equiv \frac{T}{\dot{m}} = V_e + \left(\frac{P_e - P_a}{\dot{m}}\right) A_e$

M_0 = initial mass of rocket + propellant, at t_0

M_f = final = $M_0 - M_p$, at t_f

M_p = mass of propellent burned between t_0 and t_f

- recall assumptions:

weight = negligable (far from large body)

drag = negligable (out of atmosphere)

- Define mass ratio = $\frac{M_f}{M_0}$

Text 18.1: Basic Rocket Equations

Pause - Where are we?

- For changing orbits, we framed our equations in terms of ΔV required.

  Now we can calculate how much ΔV our rocket can give us, in terms of mass properties and engine performance.

- $$\Delta V = V_{eq} \ln\left(\frac{M_0}{M_f}\right) = V_{eq} \ln\left(\frac{M_0}{M_0 - M_p}\right)$$

  where $V_{eq} = \frac{T}{\dot{m}}$

  So for example, if rocket is

  50% propellant, $\Delta V_{avail} = 0.69\ V_{eq}$

  90% propellant, $\Delta V_{avail} = 2.30\ V_{eq}$

Text 18.1: Basic Rocket Equations

"Specific Impulse" of a Rocket or Engine (like mpg)

- Total Impulse defined as:

$$I \equiv \int_0^t T\,dt \quad \text{for } T = f(t) \text{ (non-constant)}$$

= proportional to energy provided by engine(s)

$= T\,t_{burn}$ for constant T over burn time t_{burn}

- Specific Impulse defined as: $\dfrac{\text{Total Impulse}}{\text{weight of propellants}}$

$$I_{sp} \equiv \frac{\int_0^t T\,dt}{g_0 \int_0^t \dot{m}\,dt} \quad (\dot{m} = \text{rate of propellant burn})$$

for T = constant and $\dot{m}$ = constant

$$\boxed{I_{sp} = \frac{T}{\dot{m}\,g_0} = \frac{V_{eq}}{g_0}}$$

where

$$V_{eq} \equiv V_e + \left(\frac{P_e - P_a}{\dot{m}}\right) A_e$$

$$= T/\dot{m}$$

where T = constant thrust

$\dot{m}$ = rate of burn of propellants = constant

g_0 = grav constant at Earth surface

$= 9.8\ m/s^2$

Text 18.1: Basic Rocket Equations

More on Specific Impulse

- Isp is a measure of how efficiently we produce thrust. The higher the Isp, the less propellant used to achieve a given ΔV.

- For launch propulsion, thrust T is somewhat more important than Isp, because we <u>must</u> lift up through atmosphere against gravity.

- For in-space propulsion, Isp is most important to minimize the propellant required to achieve the required orbit/velocity.

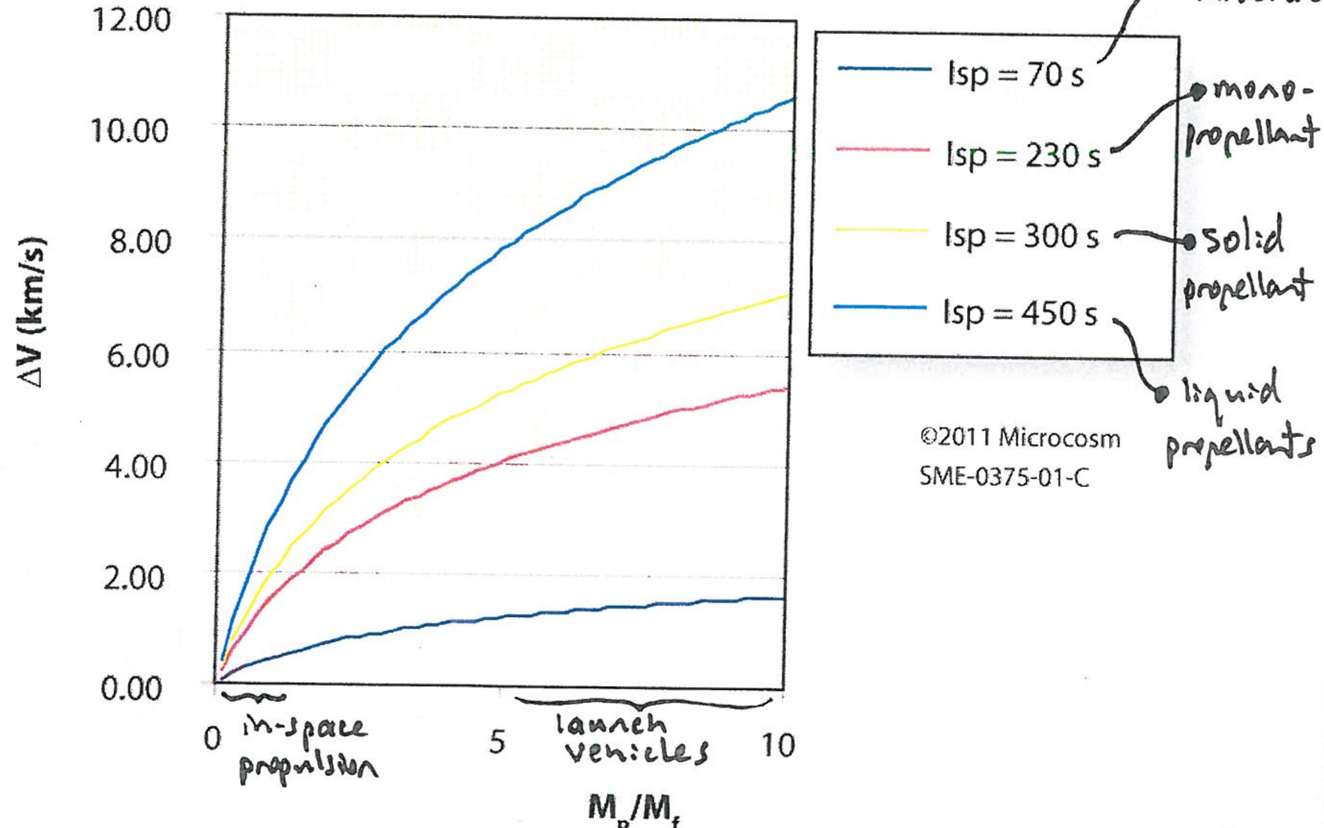
- Substitute Isp equation into Rocket Eqn:

$$\Delta V = V_{eq} \ln\left(\frac{M_0}{M_f}\right) = I_{sp}\, g_0 \ln\left(\frac{M_0}{M_f}\right)$$

So $\Delta V \propto I_{sp}$ → improvements in Isp give large effect on ΔV

various engine types

Text 18.1: Basic Rocket Equations



- Alternate forms of Rocket Equation in terms of propellant mass (M_p):

$$M_p = M_f\left(e^{\frac{\Delta V}{V_{eq}}} - 1\right) = M_f\left[e^{\left(\frac{\Delta V}{I_{sp} g_0}\right)} - 1\right] \quad \left(\text{plotted above}\right)$$

$$M_p = M_0\left(1 - e^{\frac{-\Delta V}{V_{eq}}}\right) = M_0\left[1 - e^{\left(\frac{-\Delta V}{I_{sp} g_0}\right)}\right]$$

18.1

Text ~~18.2~~: ~~Staging~~

Propulsion Function	Typical ΔV and Other Requirements
Orbit Transfer to GEO (orbit insertion)	
Perigee Burn	2,400 m/s
Apogee Burn	1,500 (low inclination) to 1800 m/s (high inclination)
Initial Spin Up	1–60 rpm
LEO to Higher Orbit Raising ΔV	
Drag makeup ΔV	60–500 m/s
Controlled Re-entry	120–150 m/s
Acceleration to Escape Velocity from LEO Parking Orbit	3,600–4,000 m/s into a heliocentric orbit
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 at GEO	3 to 6 m/s per year
North-South stationkeeping ΔV at GEO	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, 1K to 10K pulses, 0.01 to 5.0 sec pulse width
On-orbit normal mode control with 3-axis stabilization, limit cycle	100K to 200K 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, 1K to 10K pulses, 0.02 to 0.2 sec pulse width
Momentum management (wheel unloading)	5 to 10 pulse trains every few days, 0.02 to 0.10 s pulse width
3-axis control during ΔV	on-off pulsing, 10K to 100K pulses, 0.05 to 0.20 s pulse width

Table 18-2. Typical Functions and Requirements for Upper Stages and In-Space Propulsion

Text ~~18.3~~ 18.1: Chemical Propulsion Systems

©2011 Microcosm Inc. Propulsion System	Orbit Insertion		Orbit Maintenance and Maneuvering	Attitude Control	Typical Range of I_{sp} (s)
	Perigee	Apogee			
Cold Gas			X	X	45–73
Solid	X	X			290–304
Liquid					
Monopropellant			X	X	200–235
Bipropellant	X	X	X	X	274–467
Electric		X	X	X	500–3,000

Table 18-3. Overview of Common Applications for Different Propulsion Systems.

- Real-World I_{sp} :

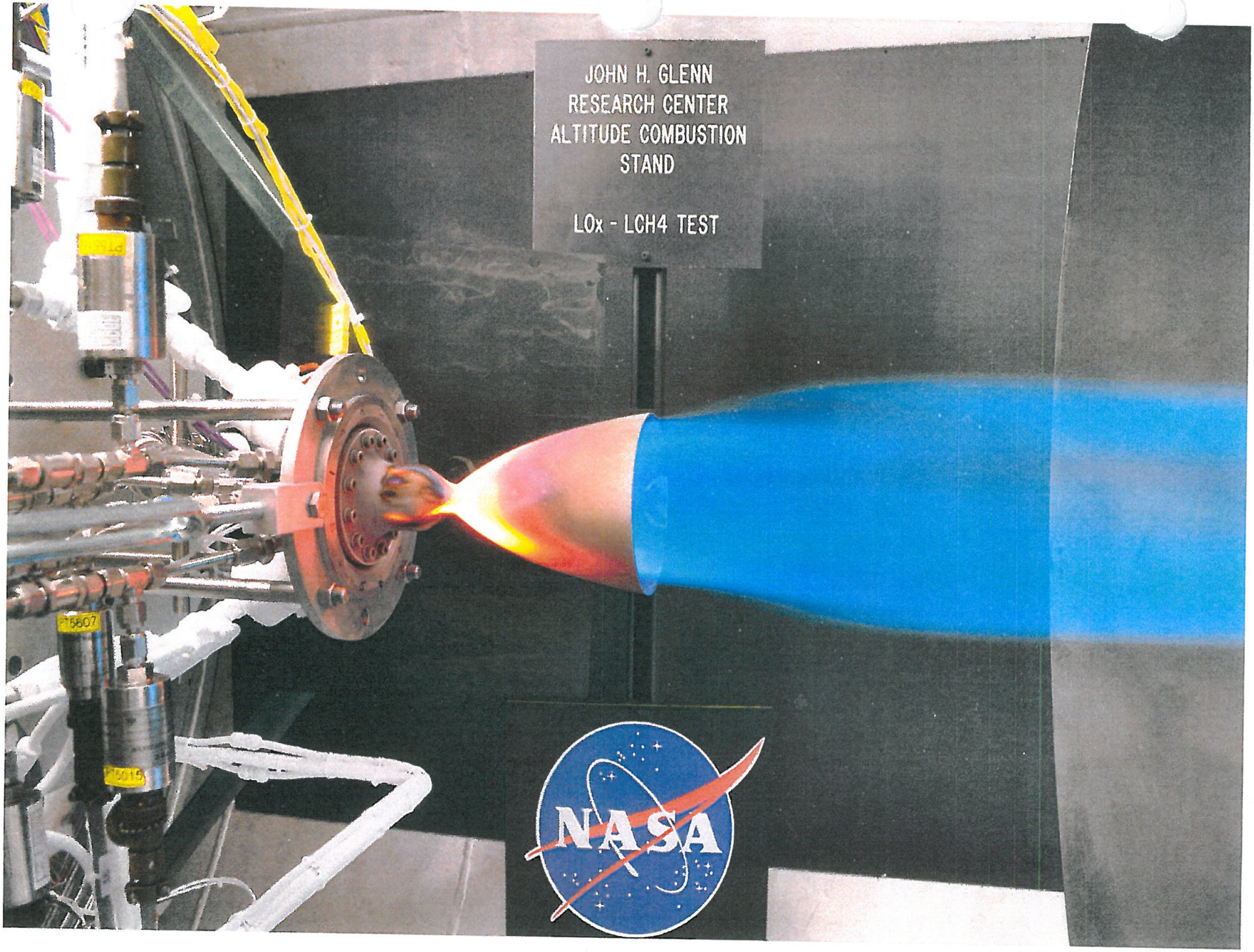
$$I_{sp} = \frac{T}{\dot{m}\, g_0}$$

$$= \frac{\dot{m} V_e + \overbrace{A_e (P_e - P_a)}^{\text{pressure thrust}}}{\dot{m}\, g_0}$$

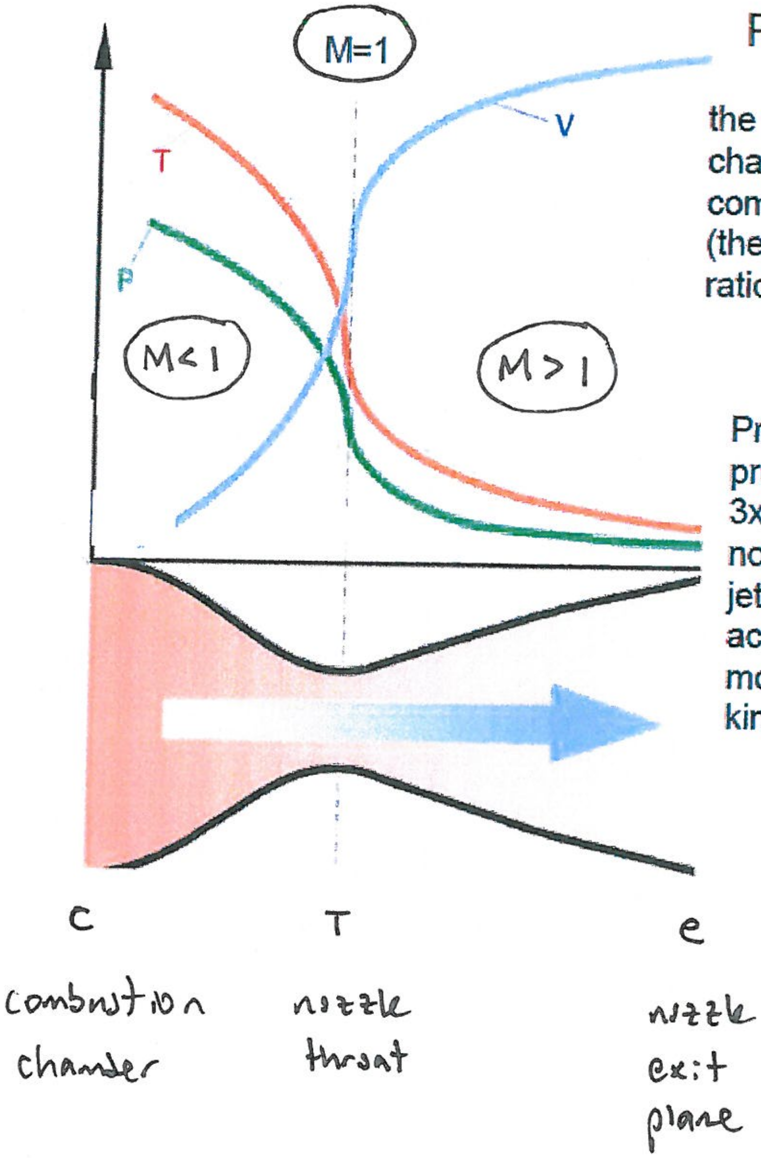
pressure thrust → increases with altitude until $P_a = 0$ (vac.)

$$\therefore \; I_{sp_{vacuum}} > I_{sp_{sealevel}}$$

A rocket gets more efficient as it leaves atmosphere!
(nothing to do with drag)


JOHN H. GLENN
RESEARCH CENTER
ALTITUDE COMBUSTION
STAND
LOx - LCH4 TEST
NASA

Text 18.1: Basic Rocket Equations

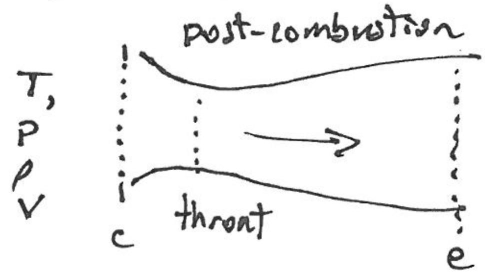


Rocket nozzles

the hot gas produced in the combustion chamber is permitted to escape from the combustion chamber through an opening (the "throat"), within a high expansion-ratio 'de Laval nozzle'.

Provided sufficient pressure is provided to the nozzle (about 2.5-3x above ambient pressure) the nozzle *chokes* and a supersonic jet is formed, dramatically accelerating the gas, converting most of the thermal energy into kinetic energy.

Nozzle Performance



Want thermodynamic relation for V_e

- Assumptions:
  - combustion products constant composition & homogeneous
  - Perfect gas $P = \rho \frac{R_o}{W} T$; R_o = univ. gas constant, W = molecular weight
  - constant C_p, C_v
  - 1-D, steady, isentropic flow, adiabatic

- Cons. Mass: $\dot{m} = \rho V A$

Cons. Energy: $\underbrace{\frac{1}{2} V^2}_{\text{kinetic}} + \underbrace{C_p T}_{\text{internal}} = \text{constant}$

- Adiabatic relations:

$$\frac{P}{\rho^\gamma} = \text{constant} \quad ; \quad \frac{T}{T_c} = \left(\frac{\rho}{\rho_c}\right)^{\gamma-1} = \left(\frac{P}{P_c}\right)^{\frac{\gamma-1}{\gamma}}$$

- Want an expression for V_e, to see influences on thrust.

Start by maximizing $\dot{m}$ through nozzle, so need expression for $\dot{m}$ ...

• Energy eqn from c :

$$\frac{1}{2}\cancel{V_c^2}^{\to 0} + C_p T_c = \frac{1}{2} \underset{\uparrow \left(\frac{\dot{m}}{\rho A}\right)^2}{V^2} + C_p T$$

$$\left(\frac{\dot{m}}{\rho A}\right)^2 = C_p (T_c - T)$$

$$\frac{\dot{m}}{A} = \left[\rho\, C_p\, \underbrace{(T_c - T)}\right]^{1/2}$$

$$T_c\left(1 - \frac{T}{T_c}\right)$$

$$T_c\left(1 - \frac{p}{p_c}^{\frac{\gamma-1}{\gamma}}\right)$$

$$\frac{p}{\rho^\gamma} = \frac{p_c}{\rho_c^\gamma}$$

$$\rho^\gamma = \frac{p}{p_c}\rho_c^\gamma$$

$$\rho = \left[\frac{p}{p_c}\rho_c^\gamma\right]^{+\frac{1}{\gamma}}$$

• Collect :

$$\frac{\dot{m}}{A} = \left\{\frac{2\gamma}{\gamma-1} p_c \rho_c \left(\frac{p}{p_c}\right)^{\frac{2}{\gamma}} \left[1 - \left(\frac{p}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]\right\}^{1/2} \qquad 6.26$$

recall that $\dot{m}_{max}$ when throat velocity is <u>sonic</u> : $V_t = a$

- Choked nozzle : sonic throat conditions

$$\frac{P_t}{P_c} = \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma}{\gamma-1}}$$

$$\frac{T_t}{T_c} = \frac{2}{\gamma+1}$$

- Plug into 6.16 to get

$$\left.\frac{\dot{m}}{A}\right|_{max} = \left(\gamma P_t \rho_t\right)^{1/2} = \left(\rho_t V_t\right)$$

where V_t = "critical" (sonic) throat velocity

$$= a_t$$

$$= \left(\gamma \frac{P_t}{\rho_t}\right)^{1/2} = \left(\gamma \frac{R_0}{W} T_t\right)^{1/2}$$

- So $\dot{m}$ determined solely by throat conditions :

$$\boxed{\dot{m} = \sqrt{\gamma}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \frac{P_c A_t}{\sqrt{R_0 T_c / W}}} \quad 6.17$$

nozzle mass-flow rate in terms of throat area and combustion chamber P and T

- Expression for V_e :

Energy : $$\frac{1}{2} V_e^2 + C_p T_e = \frac{1}{2} \cancel{V_c^2}^{\,V_c \cong 0} + C_p T_c$$

$$V_e = \sqrt{2 C_p (T_c - T_e)}$$

$$\boxed{V_e = \sqrt{\frac{2 \gamma R_0 T_c}{(\gamma - 1) W} \left[1 - \left(\frac{P_e}{P_c} \right)^{\frac{\gamma-1}{\gamma}} \right]}} \quad 6.20$$

- To maximize V_e (and therefore thrust):

  - increase $\frac{P_c}{P_e}$

  - increase T_c

  - decrease W molecular weight

- Common shorthands : "characteristic velocity" C^*

$$\dot{m} = P_c A_t / C^*$$

where $$C^* = \sqrt{\frac{R_0 T}{w}} \Big/ \left[\sqrt{\gamma}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}\right]$$ 6.18

- "Characteristic thrust coefficient" : C_F^0

$$V_e = C^* C_F^0$$

where

$$C_F^0 = \sqrt{\left[\gamma\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{(\gamma-1)}}\right]\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$ 6.21

- Exit-to-throat area ratio :

$$\frac{A_e}{A_t} = \overset{\text{continuity}}{\frac{\rho_t V_t}{\rho_e V_e}} = \frac{\gamma\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\left(\frac{P_c}{P_e}\right)^{\frac{1}{\gamma}}}{C_F^0}$$ 6.22

Nozzle Design - Real World

- Thrust eqn:

$$F = \dot{m} V_e + A_e \left(P_e - P_a \right)$$

- A_e for max thrust?

(P_e = const)

$$\delta F = \dot{m}\, \delta V_e + \delta A_e \left(P_e - P_a \right) + A_e\, \delta P_e$$

for constant $\dot{m}$

Momentum:

$$\dot{m}\, \delta V_e + A_e\, \delta P_e = 0$$

Solve for

$$\frac{\delta F}{\delta A_e} = \dot{m} \frac{\delta V_e}{\delta A_e} + \left(P_e - P_a \right) + A_e \frac{\delta P_e}{\delta A_e}$$

$$= \left(P_e - P_a \right) + \frac{1}{\delta A_e} \underbrace{\left(\dot{m}\, \delta V_e + A_e\, \delta P_e \right)}_{0 \text{ Mom.}}$$

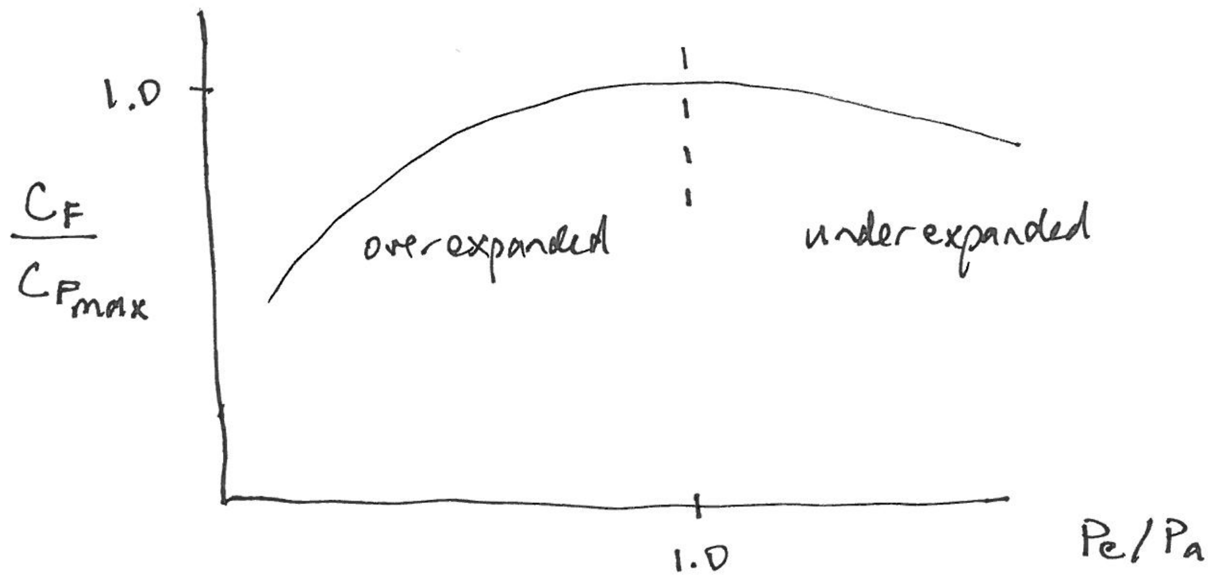
$$\boxed{\text{So max thrust when } P_e = P_a}$$

ideal expansion aft of nozzle

• Effect of non-ideal expansion:

define thrust coefficient

$$C_F = \frac{F}{P_c A_T} = C_F^0 + \frac{A_e}{A_t}\left(\frac{P_e}{P_c} - \frac{P_A}{P_c}\right) \quad 6.23$$



• somewhat worse to overexpand

• adaptive geometry nozzles: complex, thrust-efficient

Aerospike engine

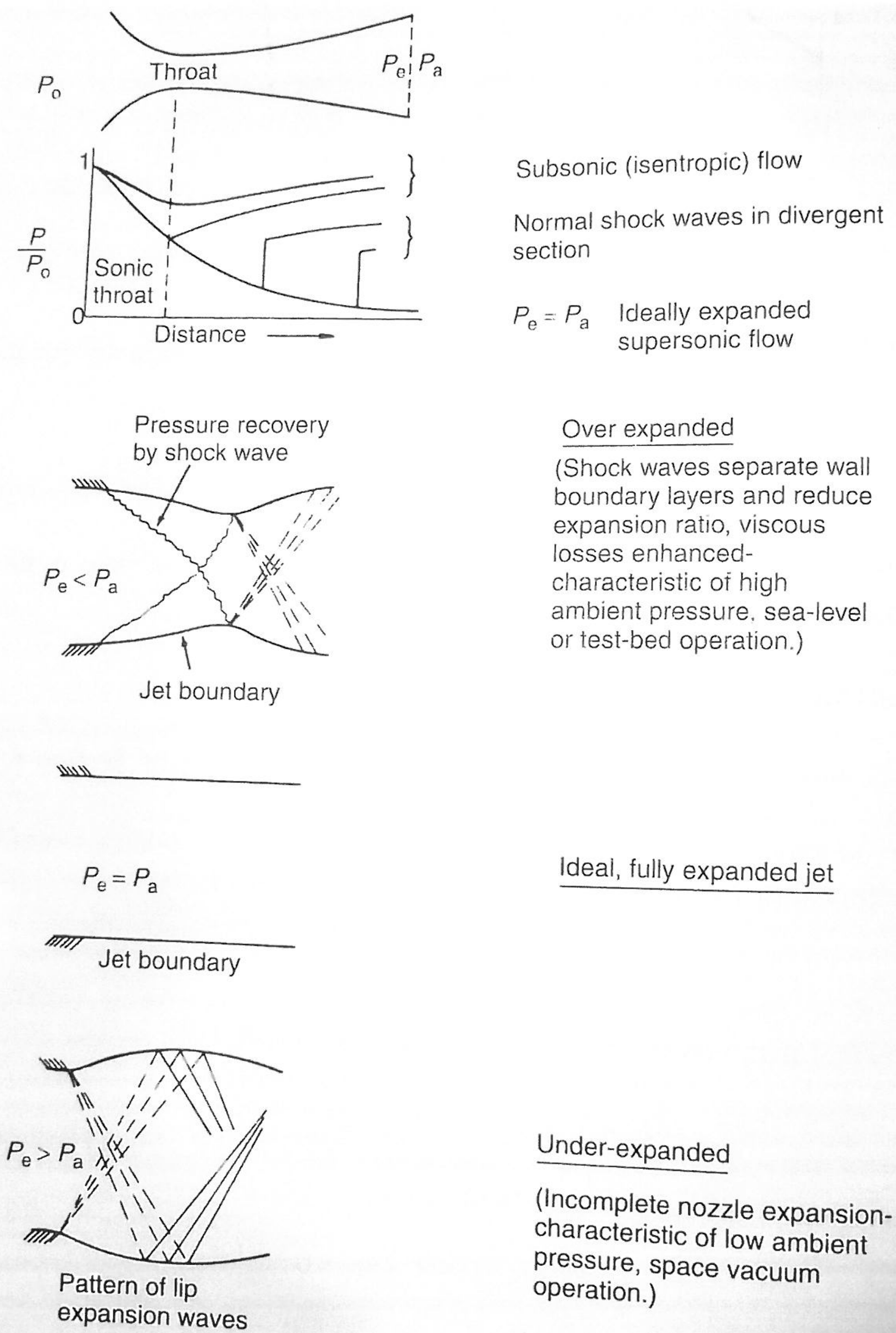


Figure 6.6 Nozzle flows: non-ideal expansion

18.1

Text ~~18.2: Staging~~

- Recall

$$I_{sp} = \frac{T}{\dot{m}\, g_0}$$

$$= \frac{\overbrace{\dot{m} V_e}^{\text{momentum}} + \cancel{A_e (P_e - P_a)}}{\dot{m}\, g_0}$$

generally much less than momentum thrust

$$\cong \frac{V_e}{g_0}$$

- From previous page:

$$\boxed{I_{sp} \cong \frac{1}{g_0}\sqrt{2\,\frac{R_u}{MW}\left(\frac{k}{k-1}\right) T_c \left[1 - \left(\frac{P_e}{P_c}\right)^{\frac{k-1}{k}}\right]}}$$

(same as text eqn 18-17, but without pressure thrust term)

R_u = universal gas constant = $8.314 \; \frac{J}{mol\,K}$

MW = exhaust molecular weight

T_c = combustion temp

k = C_p / C_v

P_e = nozzle exit pressure

P_c = combustion pressure

For high I_{sp}, want:

low exhaust MW

high combustion temp

high press. ratio $\frac{P_c}{P_e}$

Rocket Engine Efficiency: ISP Considerations

- Chemical propellants: $175 < ISP < 400$
- For high ISP:
  - high exhaust-gas temp: large heat of combustion
  - Low molecular-density combustion products: H, C, O, Fl, Al, Be, Li, etc
- Propellant density:
  - lower density = higher tank volume and mass
- Coolant properties
- Safety & handling:
  - Corrosive, flammable, toxic
  - All three!
- Availability, storage
- Environmental concerns

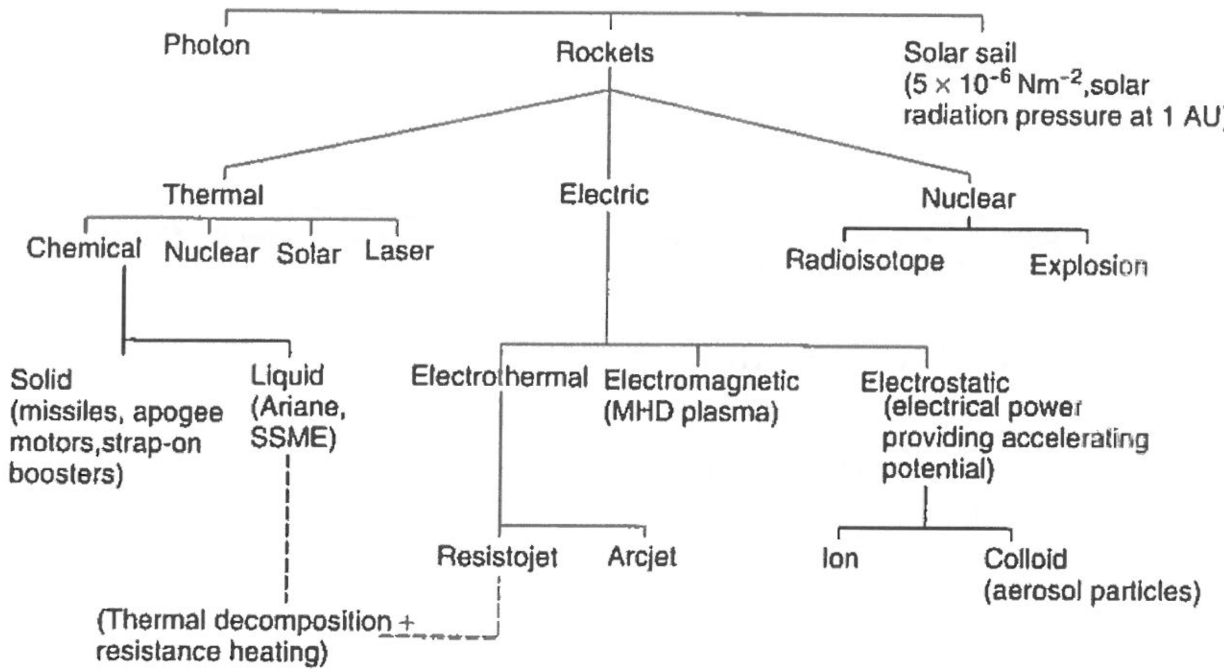


Figure 6.1 Propulsion systems classification

Propellants

- Exothermic heat ~~[illegible]~~ release in nozzle gas flow

    specific energy content
    rate of heat release
    storage, handling
    plumbing
    failure modes
    toxicity (before and after combustion)
    demands on system mat'l properties

Liquid Propellants

- bi-propellant : 2 liquids

    amb-combust : hypergolic

    monopropellant : exothermic decomposition

- requires pumping system, tanks, plumbing

    throttling
    stop/start

- May require cryogenic storage – time limit

- Recall : want high T_c and low W for max V_e 6.20

- Common : LOX/LH_2 and LOX/RP1 (kerosene) – Large thrust

    MMH/N_2O_4 – Lower thrust

Table 6.1 Liquid propellants

Fuel	Oxidizer	Molecular weight of products	Combustion temperature T_c (K)	Ideal specific impulse (s)	Mean density kg/m^3
H_2 (hydrogen)	O_2 (oxygen)	10	2980	390	280
	F_2 (fluorine)	12.8	4117	410	460
Kerosine	O_2	23.4	3687	301	1020
	F_2	23.9	3917	320	1230
	RFNA (red fuming nitric acid)	25.7	3156	268	1355
	N_2O_4 (nitrogen tetroxide)	26.2	3460	276	1260
	H_2O_2 (hydrogen peroxide)	22.2	3008	278	1362
N_2H_4 (hydrazine)	O_2	19.4	3410	313	1070
	HNO_3 (nitric acid)*	20	2967	278	1310
UDMH	O_2	21.5	3623	310	970
$(CH_3)_2NNH_2$ (unsymmetrical dimethyl hydrazine)	HNO_3^*	23.7	3222	276	1220
	* hypergolic				
Monopropellants					
N_2H_4		10.3	966	199	1011
H_2O_2		22.7	1267	165	1422

*Note: All quoted values are for $p_c = 7$ MPa with an ideal expansion to $p_e = 0.1$ MPa. Higher chamber pressures admit increases in I_{SP}—for example, at 20 MPa, LOX/LH$_2$ yields a specific impulse of ~460 s.

Solid Propellants

- Solid Propellants :
  - Composite: fuel & oxidizer mixed & solidified with binder
  - Most (up to 80%) of the volume is oxidizer
  - Often used as low-altitude add-ons for liquid rockets

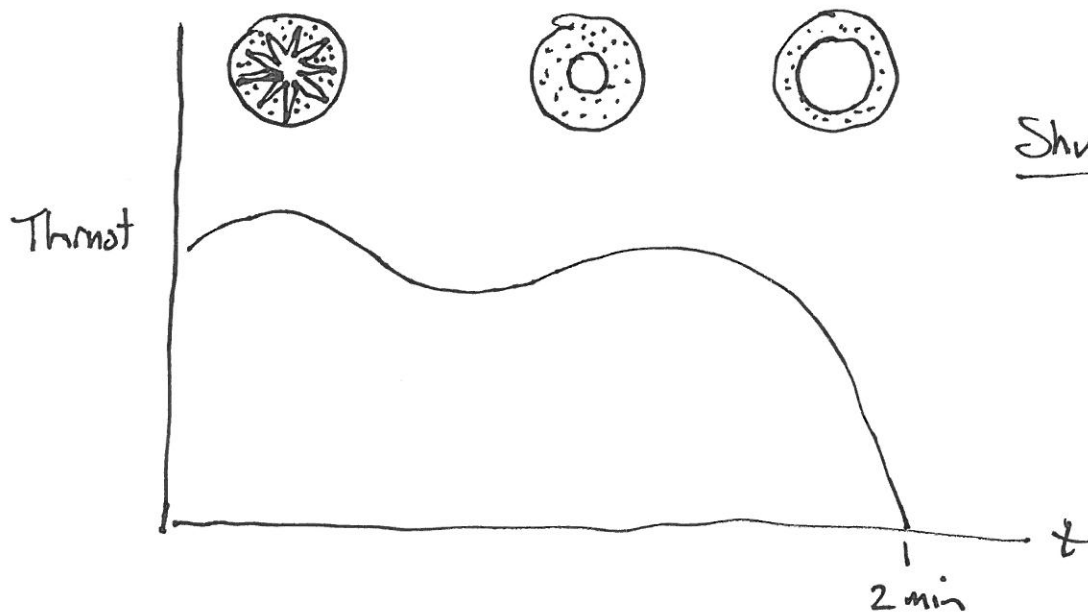
- Solid Propellants Pros:
  - Simple, few moving parts, no propellant feed systems
  - compact (higher density than some liquid props)
  - transportable, safe, low-cost
  - Scalable – massive thrust possible (SRB 3Mlbf)

- Solid Propellants Cons:
  - Entire stage is the combustion chamber – structural penalties & segment seals (shuttle Challenger)
  - Case heats during burn – more structural penalties
  - Propellant itself must load-bearing within the case (rocket body)
  - No stop or restart, no throttling (until recently)
  - Lower efficiency than liquid (lower exhaust velocities)
  - Thrust-vectoring requires complex gimbal seals

Solid Propellants

- Lower ISP than liquid
- Throttle, stop/start almost impossible
- Poured or extruded, then cured – rubberlike
- Thrust-history tailoring:

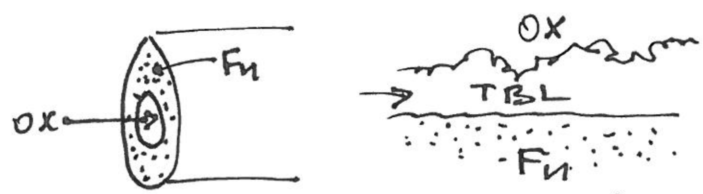
grain-pattern evolves via combustion:



Shuttle SRB

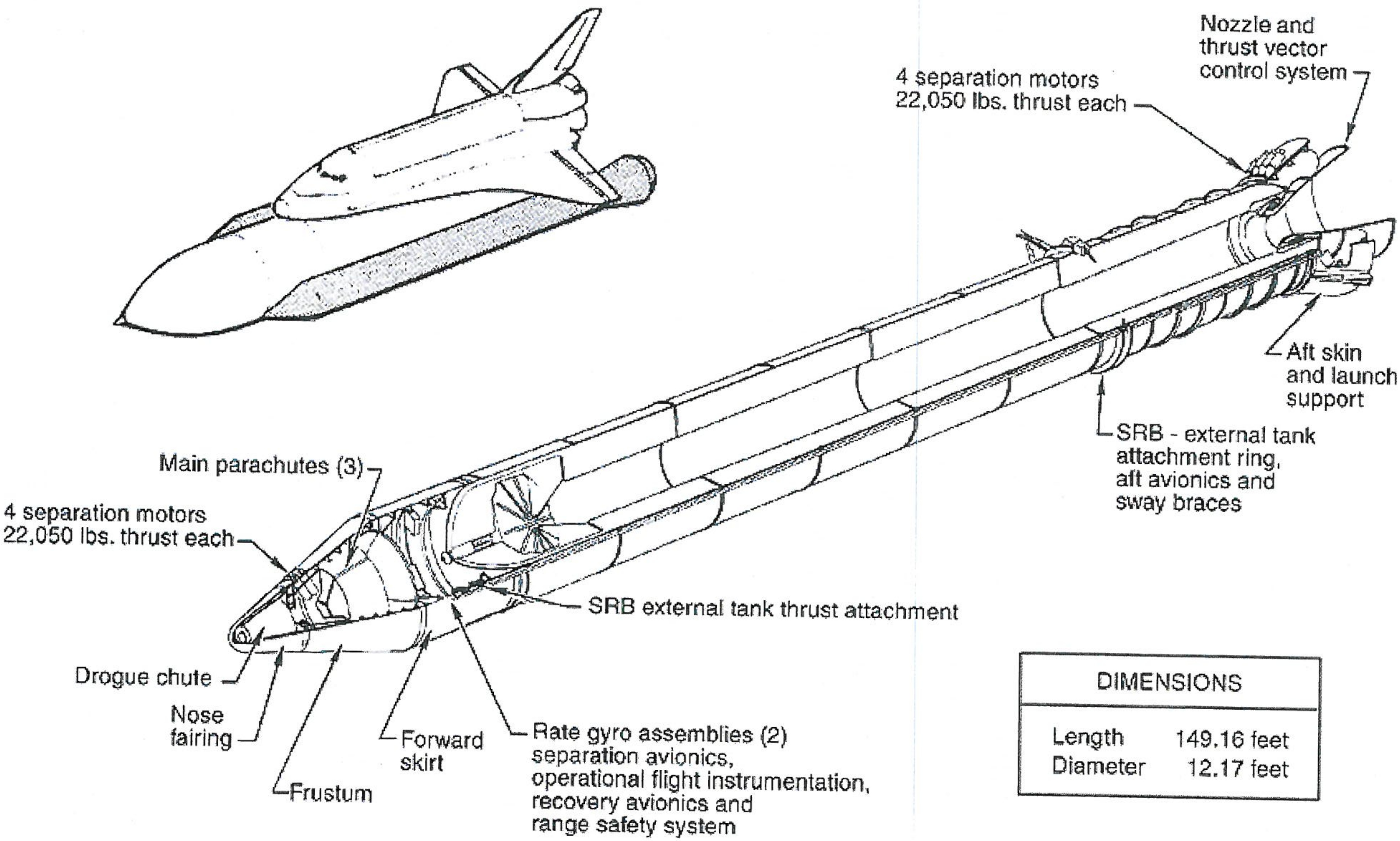
Hybrid Propellants

- Liquid ox, solid fuel
- Throttle, stop/start
- $\frac{1}{2}$ plumbing, storage complexity of liquid bi-propellants
- Fluid-solid interface combustion limits ISP



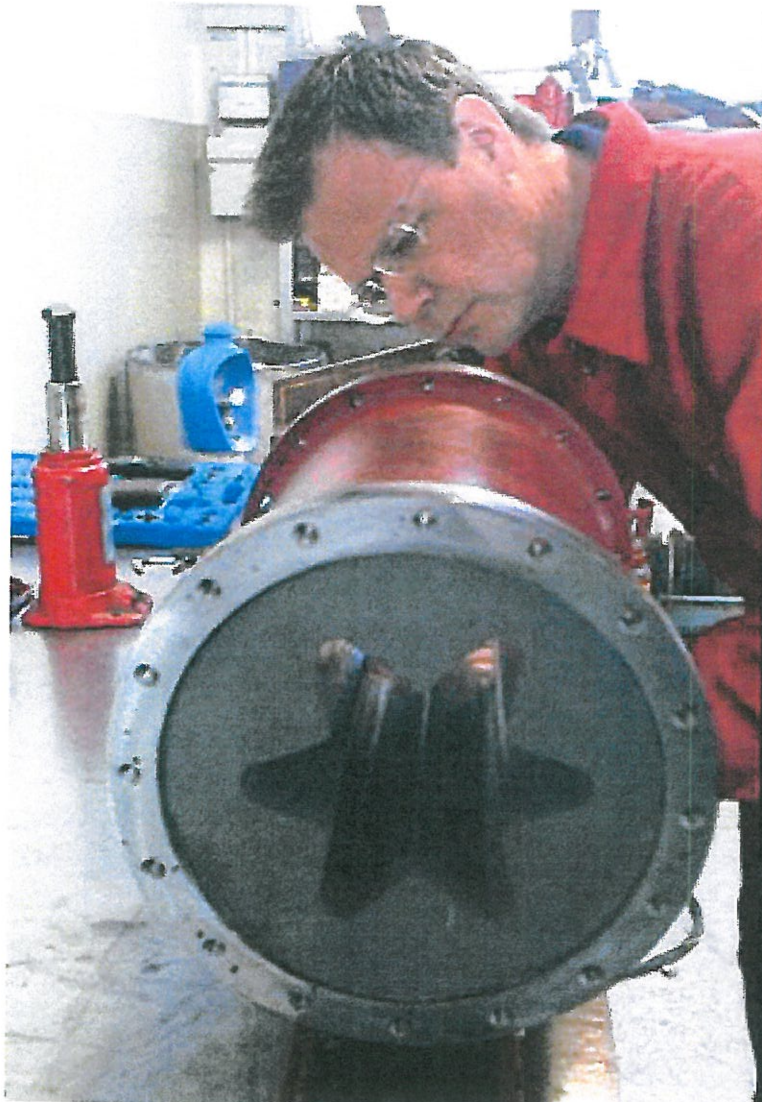
combustion interface

Space Shuttle Solid Rocket Boosters

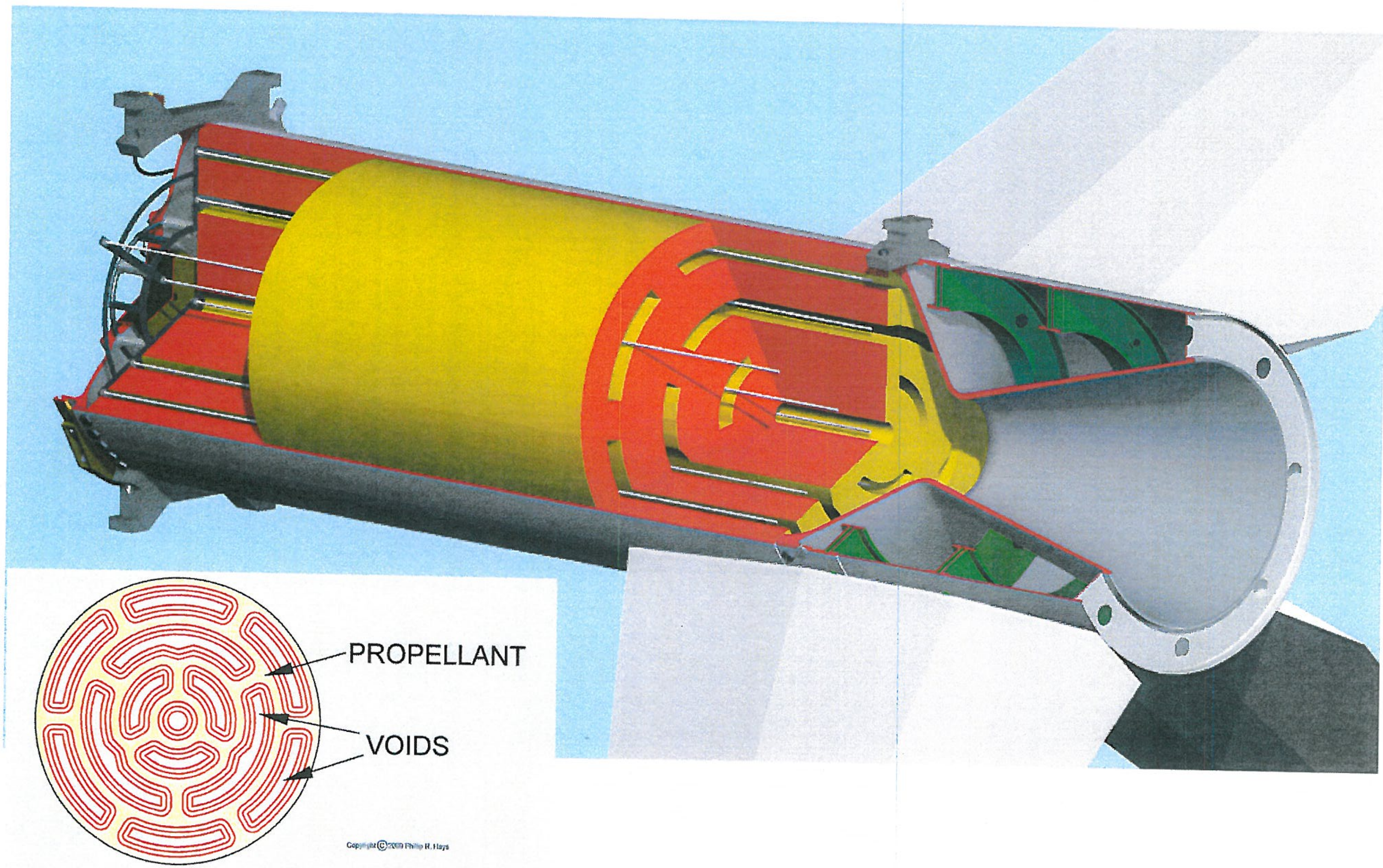


DIMENSIONS	
Length	149.16 feet
Diameter	12.17 feet

Solid Rocket Grain Patterns



Solid Rocket Grain Patterns



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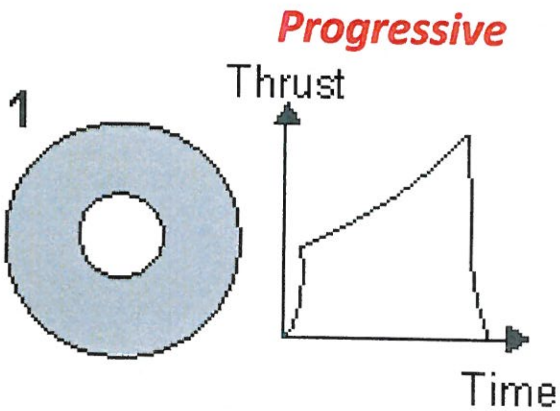
Solid Rocket Grain Patterns

1



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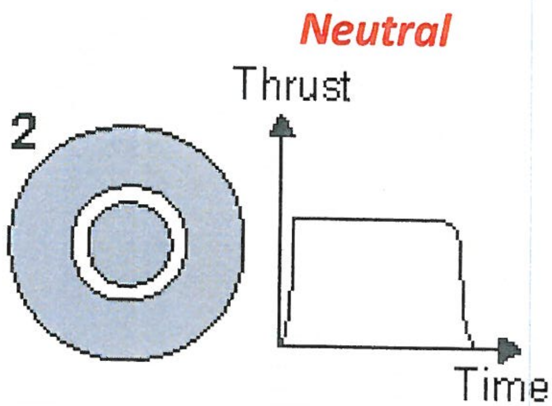

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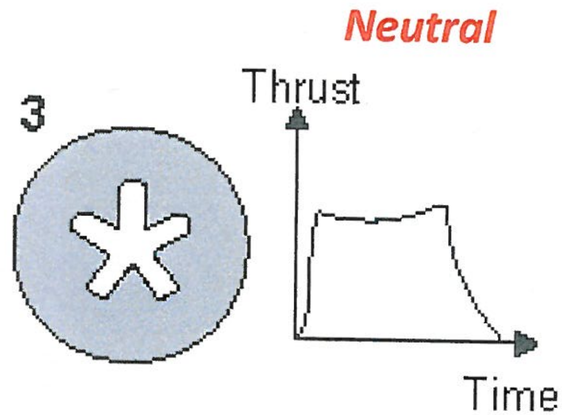

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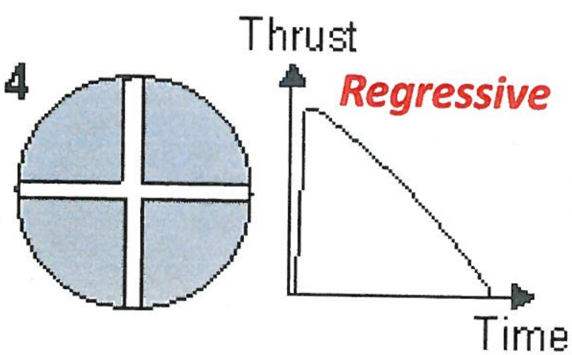

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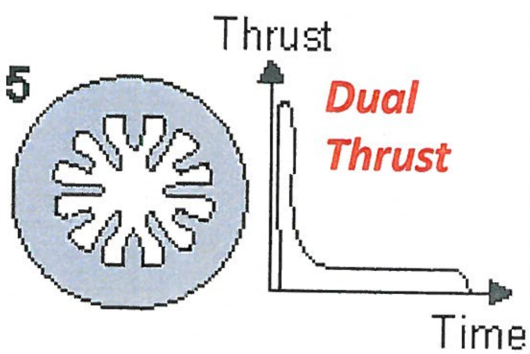

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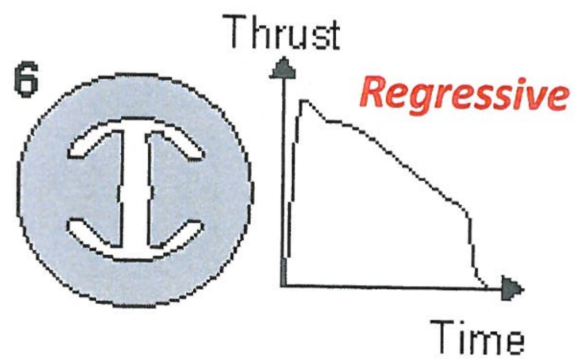
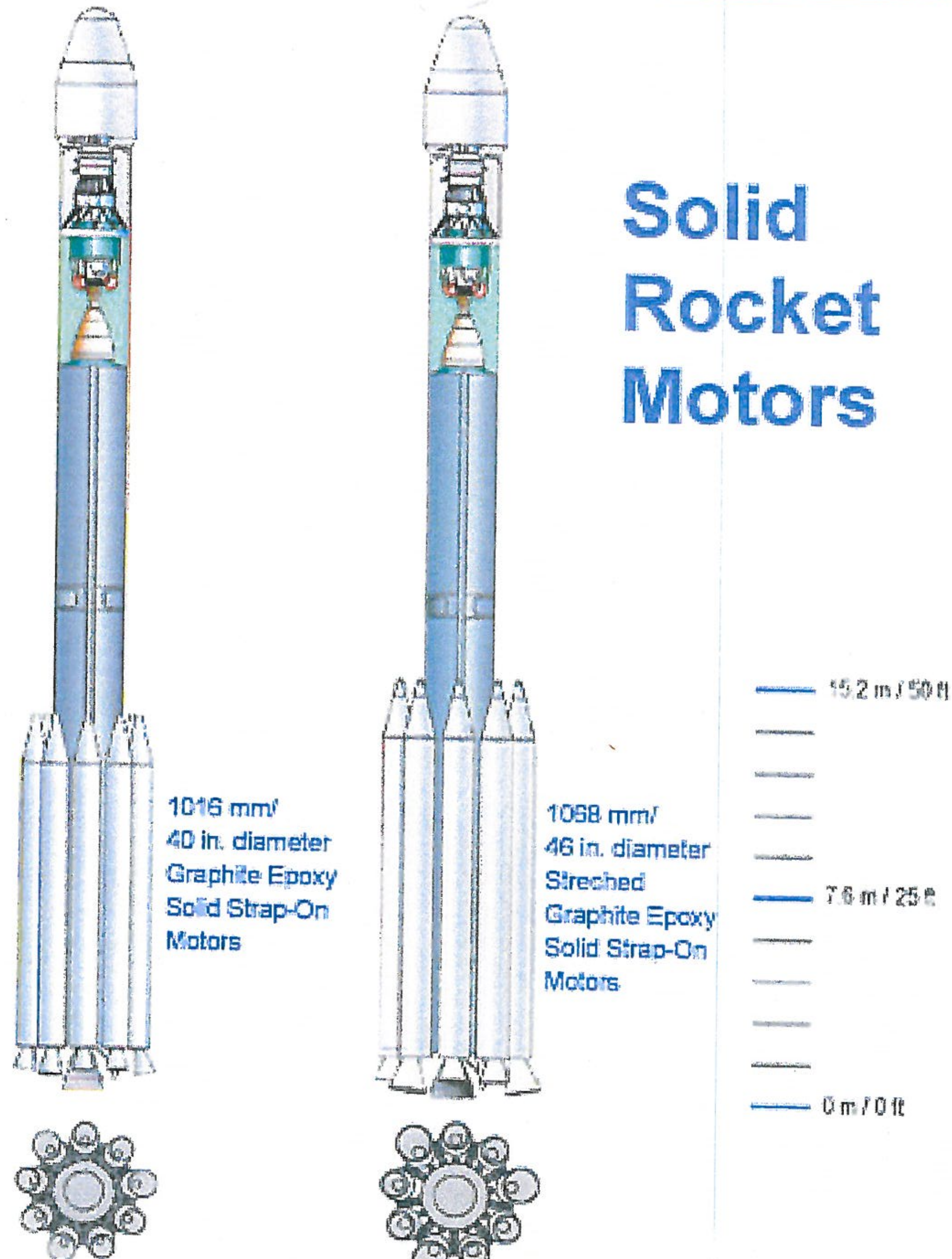


Solid Rocket Motors



Delta II 7925

Delta II 7925-H

Open-Cycle Design:

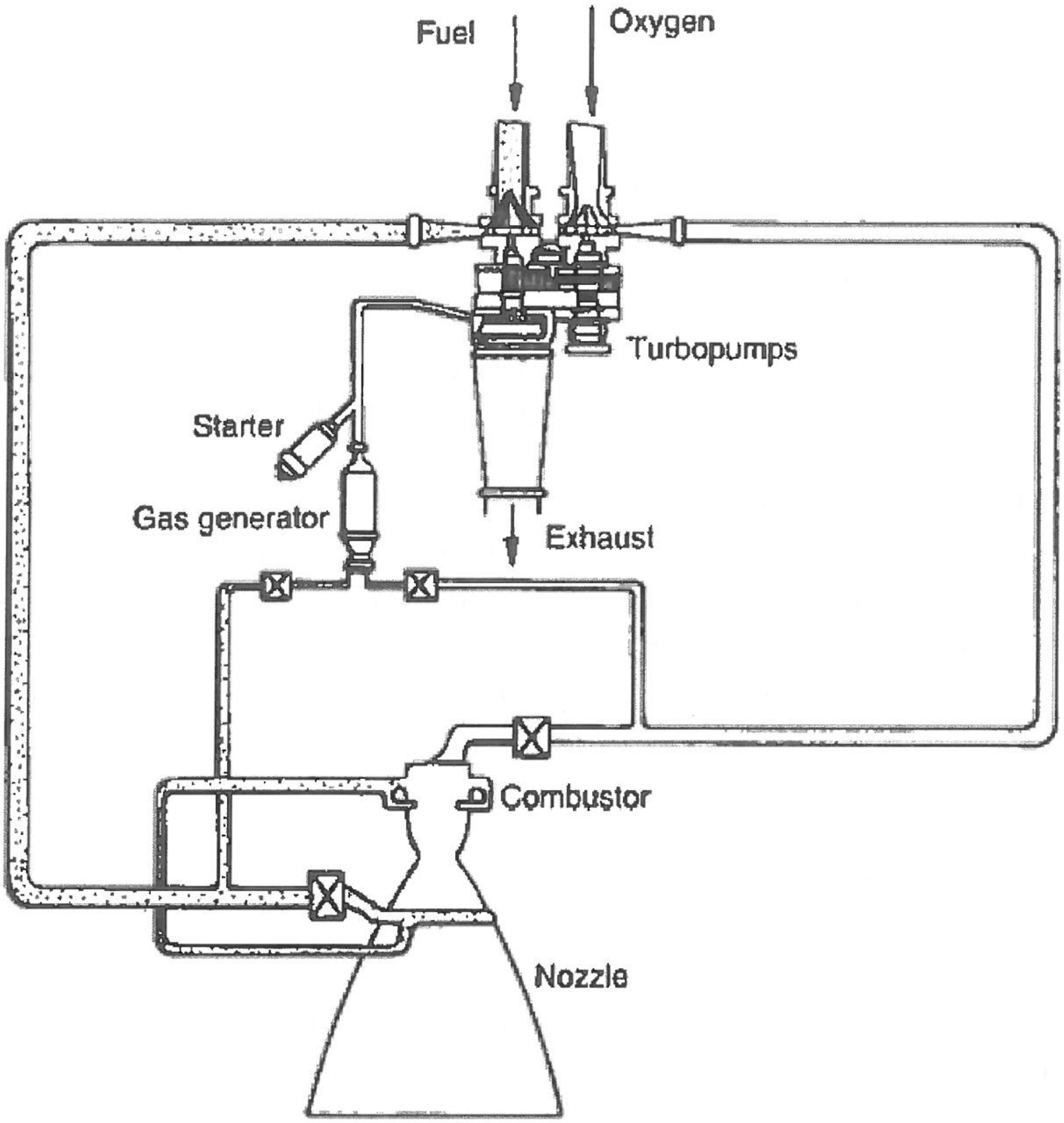
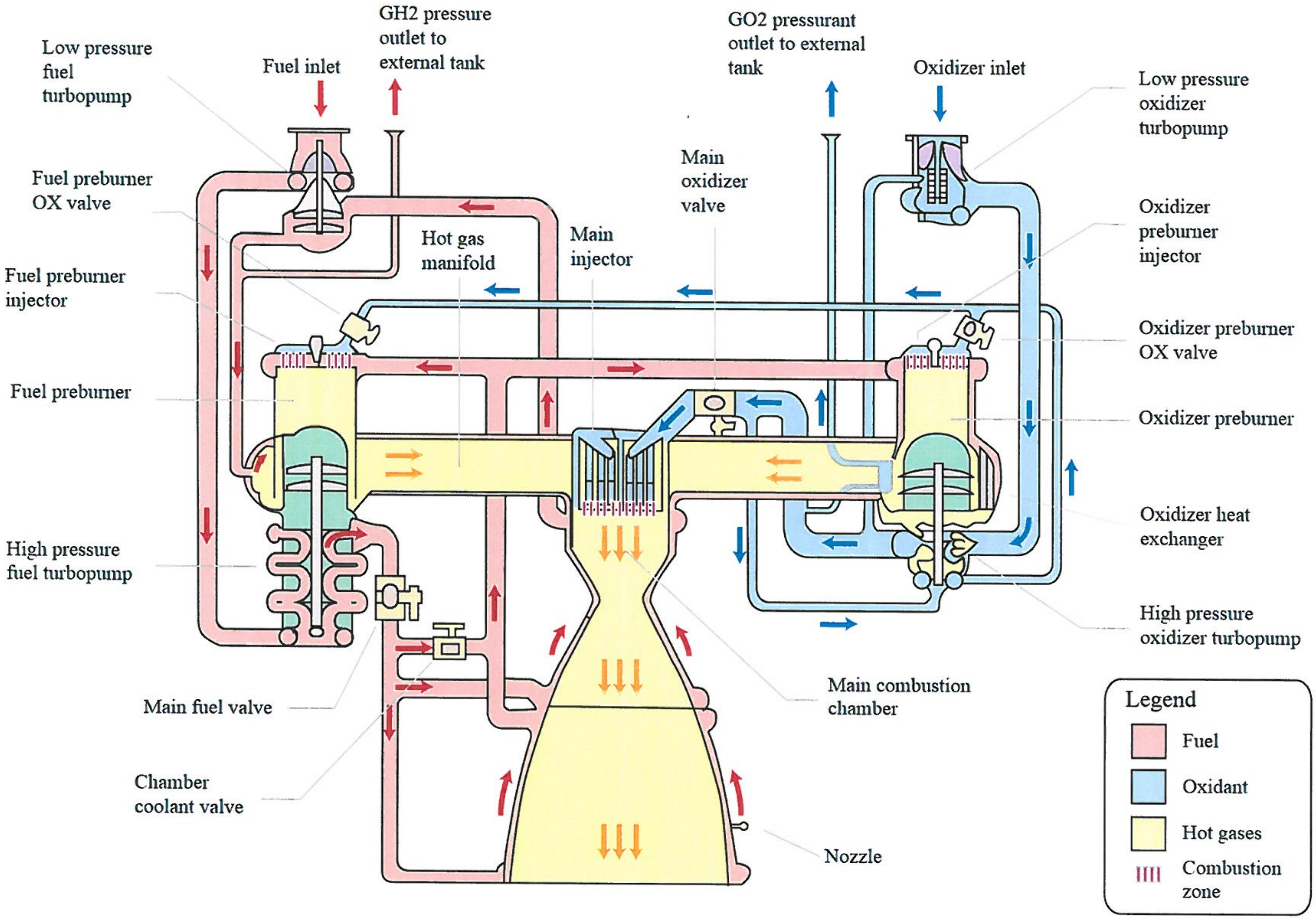


Figure 6.10 Schematic of a liquid rocket motor

Closed Cycle Design: Shuttle Main Engine

UC Davis EAE-243a Prof. S.K. Robinson



Closed (SSME) vs Open (Vulcain):

Table 6.2 Illustrative comparison of closed- and open-cycle engines

	SSME	Vulcain (Ariane 5)
Thrust (kN):		
Vacuum	2090	1390
Sea level	1700	960
Specific impulse (s):		
Vacuum	455	432
Sea Level	363	310
Mixture ratio (stoichiometric 8:1 $2H_2 + O_2 \rightarrow 2H_2O$)	6:1	5.3:1
Chamber pressure (bar)	207	108
Nozzle area ratio	77	45
Flowrates (kg/s)	468 (engine)	270
	248 (pre-combustor)	10 (gas generator)
Pump discharge pressure (bar)	309 (LOX)	125 (LOX)
	426 (LH_2)	150 (LH_2)
Burn time (s)	480	540
Mass (kg)	3022	1650

Propellant Management:

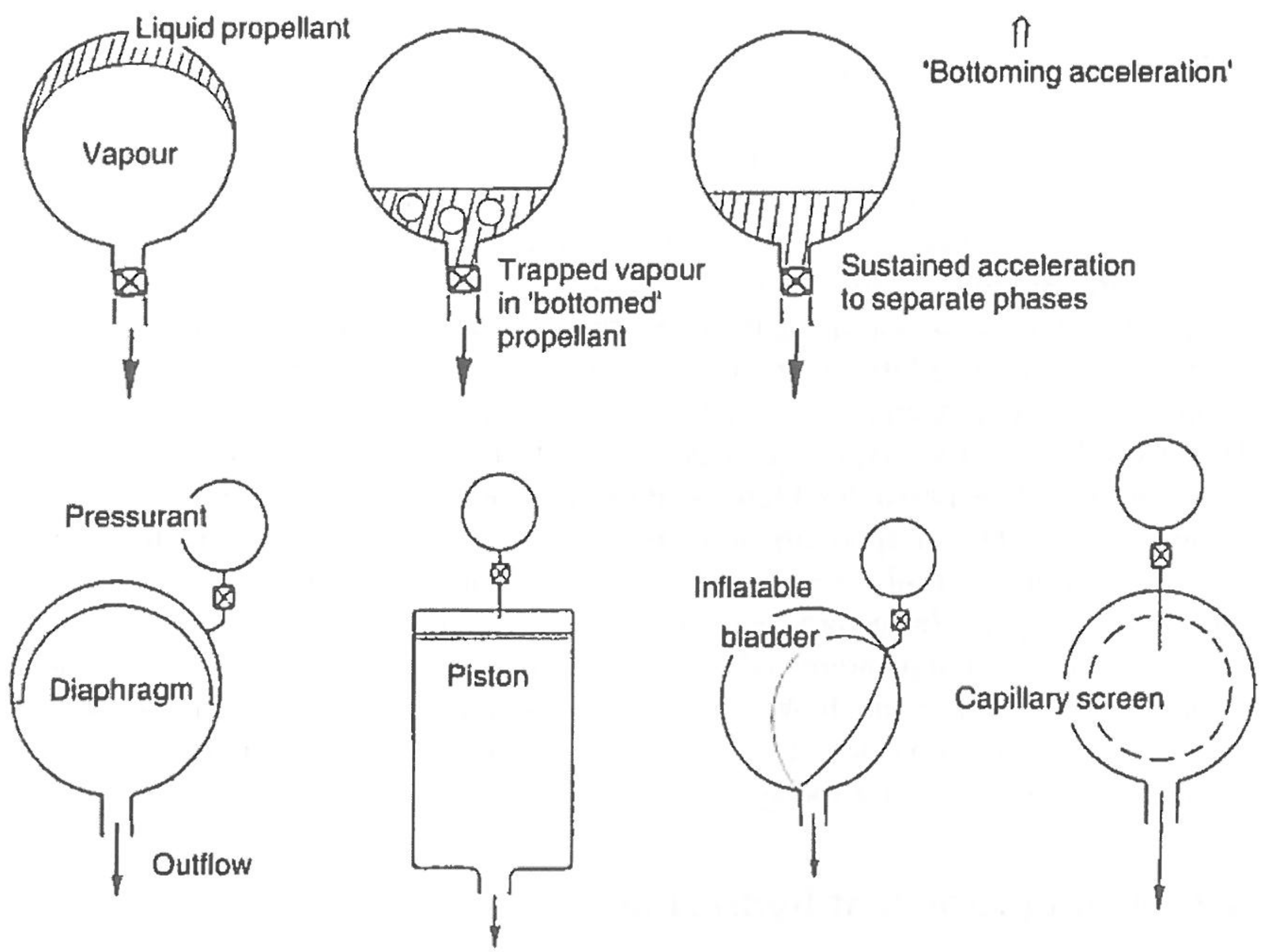
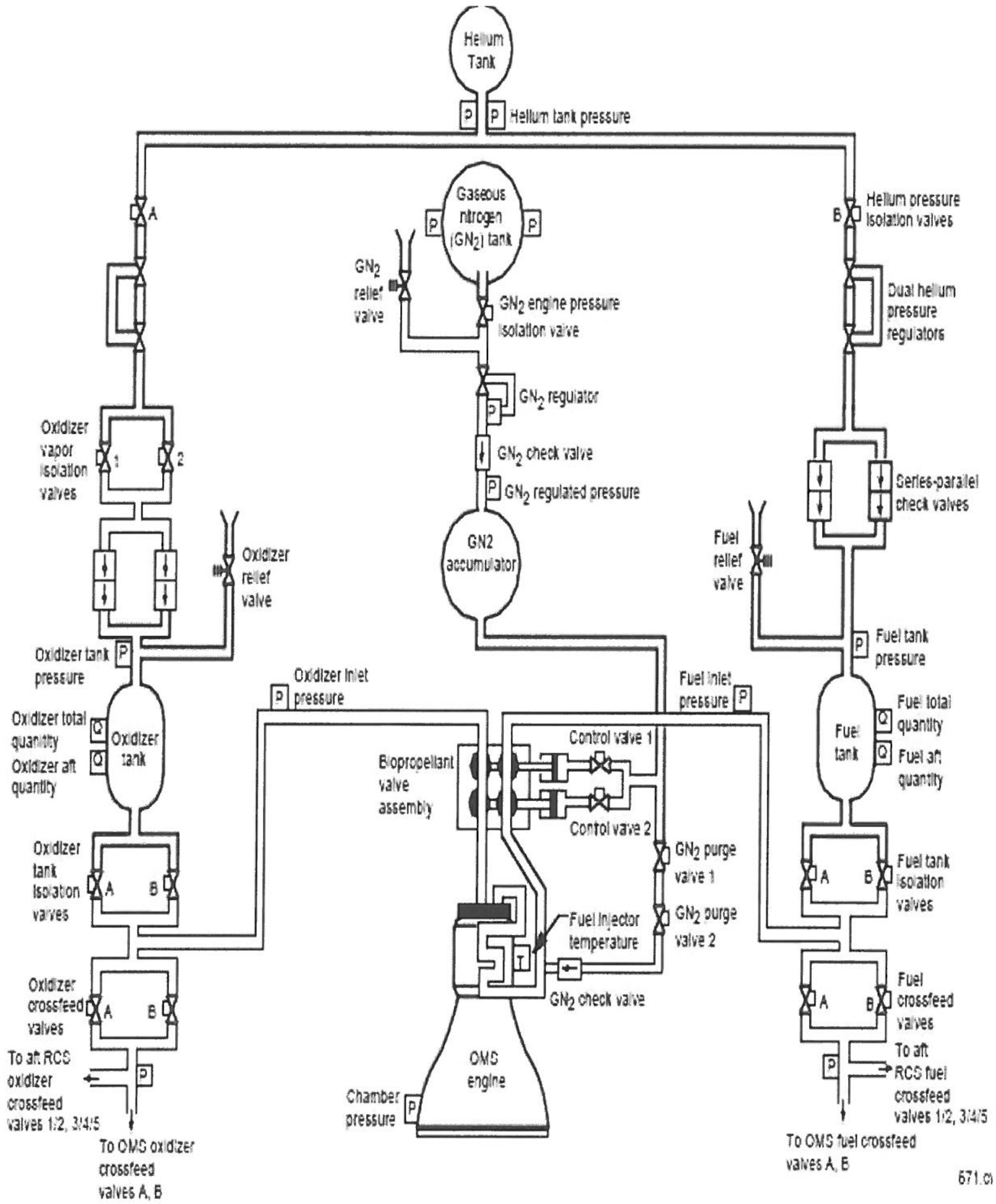


Figure 6.17 Illustrative propellant storage and delivery systems

Propellant Management: Shuttle Orbital Maneuvering System (OMS) Engine (T = 6Klbf)



Orbital Maneuvering System Pressurization and Propellant Feed System for One Engine (other Engine Identical)

Biprop Engine Design

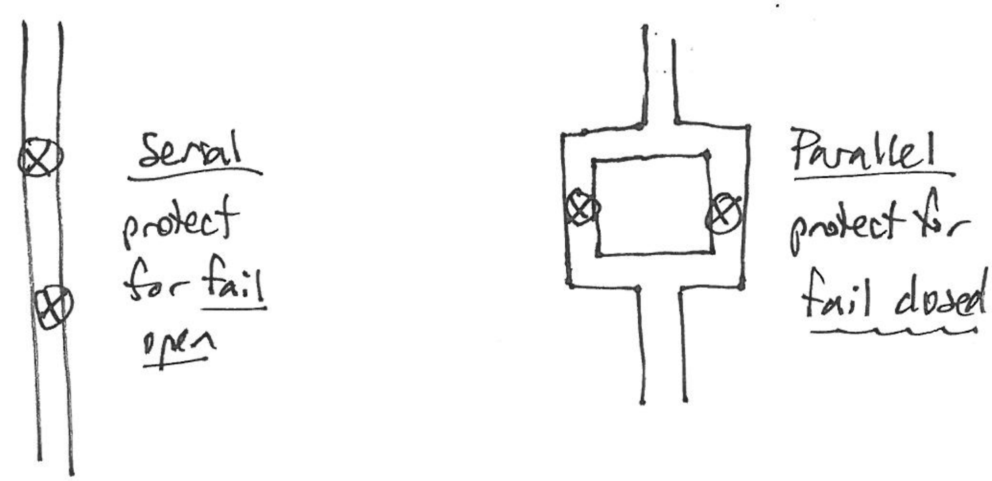
- Open vs closed
  - ↓ SSME
  - ↓ FA?

- ~~Fig~~ Table 6.2 : SSME vs Vulcain

- Prop management

  OMS diagram

- Parallel vs Serial valve design



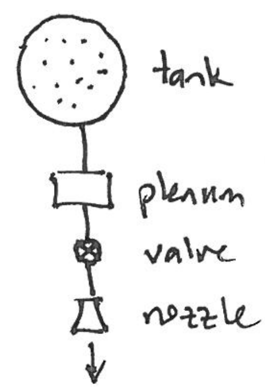
- Fig 6.17

Spacecraft Propulsion

- Tasks: final orbit acquisition after launcher drop-off
  orbit maintenance, station keeping
  relative maneuvering
  attitude control
  reboost
  de-orbit

Cold Gas

- pressurized inert gas: N_2, Ar, freon, hydrocarbon (propane, etc)



- Liquid or gas form storage
- Exhaust plume → sensors ?
- $T = 10^{-4}\ N - 10^{-2}\ N$
- $I_{sp} \sim 50$ sec (low efficiency)
- CubeSats, SAFER, etc

Monopropellant Hydrazine

- $N_2H_4 \xrightarrow[\text{catalytic}]{\text{thermal}} N_2, NH_3, H_2$ + heat
- ISP 200-250 s
- Liquid-form storage, with inert pressurant (N_2 or He)
- $T \sim 10$ N

Bi-Propellant ; MMH + N_2O_4 (hypergolic)

- ISP ~ 300 s

(figs 6.20, 6.21)

Mono-propellant System

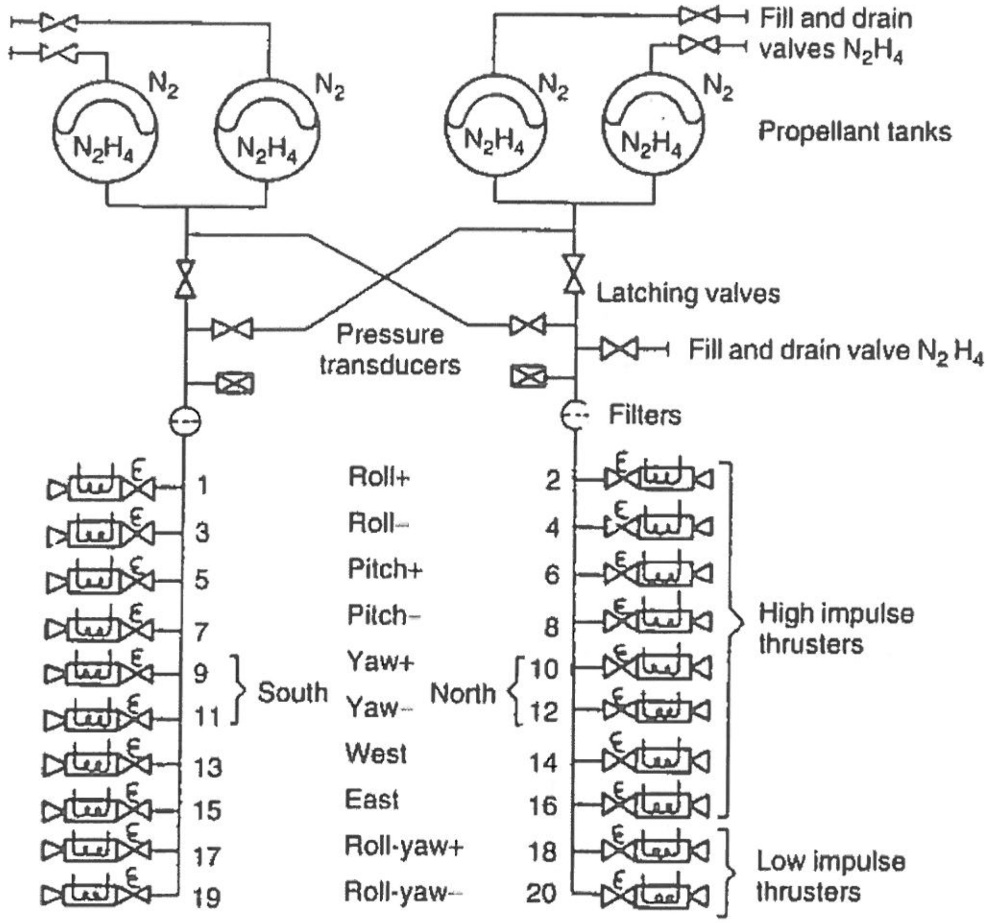


Figure 6.20 Illustrative spacecraft propulsion system using monopropellant thrusters

Bi-propellant Hypergolic System

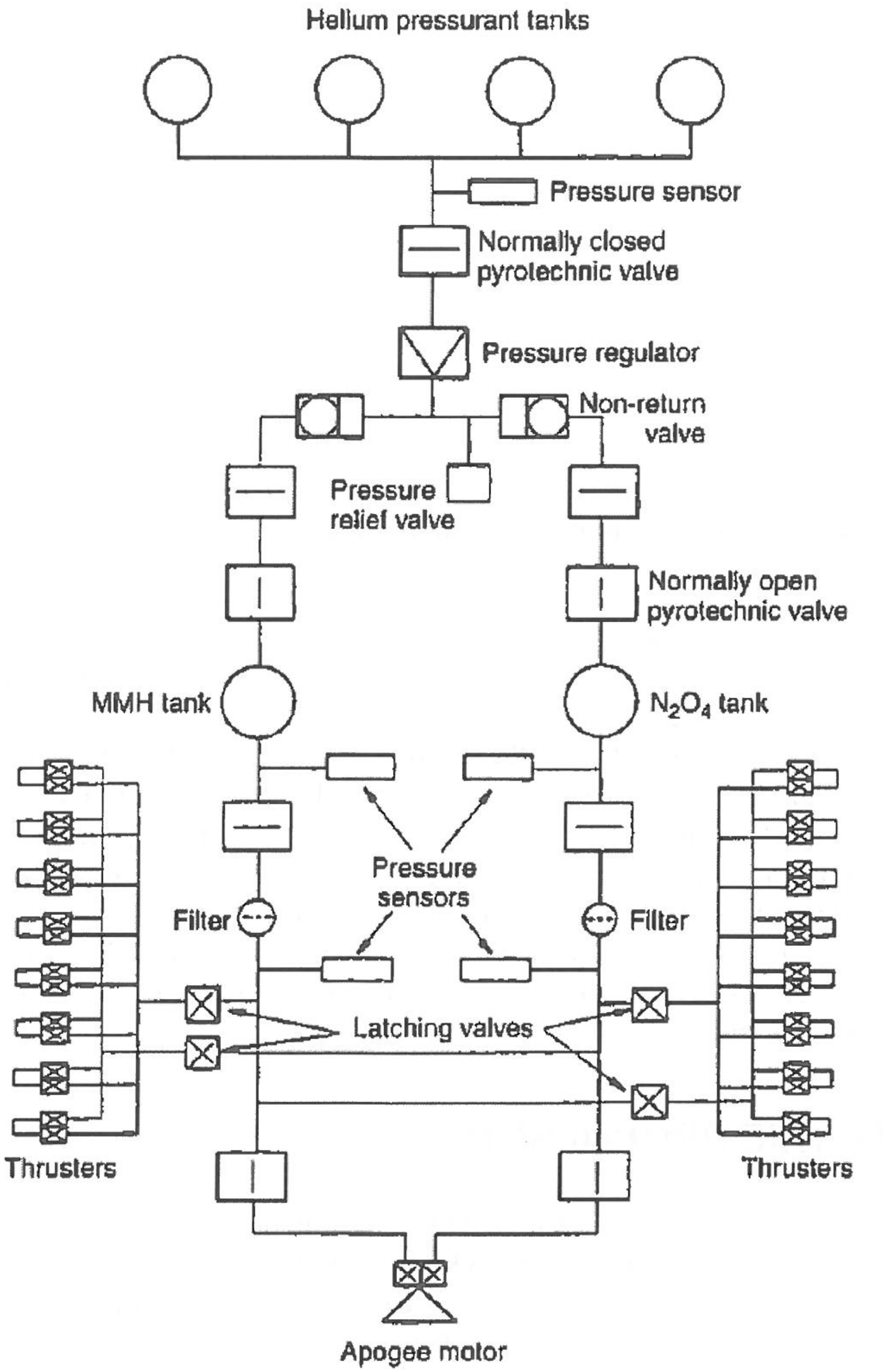
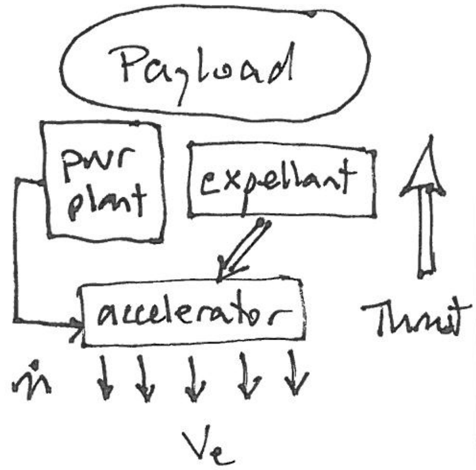


Figure 6.21 Schematic of a typical bi-propellant propulsion system

Electric Propulsion



- Let M_W = powerplant mass

  M_e = expellant mass

  M_p = payload mass

- Power-plant output:

$$W = \frac{1}{2}\dot{m}V_e^2 \quad \text{(power)}$$

expellent

- Assume power-plant output prop. to mass:

$$M_W = \alpha W$$

"inverse specific power"

- Assume $\dot{m}$ constant throughout burn time t_b:

$$\dot{m} = \frac{M_e}{t_b}$$

- Define total rocket mass

$$M_0 = M_W + M_e + M_p$$

- Solve for M_e :

$$M_e = \dot{m} t_b = \frac{2W}{V_e^2} t_b = \frac{2 M_w}{V_e^2} \frac{t_b}{\alpha}$$

$$= \frac{M_0 - M_p}{1 + \left(V_e^2 / \frac{2 t_b}{\alpha} \right)}$$ 6.28

- Similarly,

$$M_w = \frac{M_0 - M_p}{1 + \left(\frac{2 t_b}{\alpha} / V_e^2 \right)}$$ 6.29

- Use rocket equation to get ΔV :

$$\Delta V = V_e \ln \left(\frac{M_{initial}}{M_{final}} \right)$$

$$\boxed{\Delta V = V_e \ln \left[\frac{1 + (V_e/V_c)^2}{\frac{M_p}{M_0} + (V_e/V_c)^2} \right]}$$ 6.30

(Fig 6.24)

Note that there is an <u>optimum</u> V_e

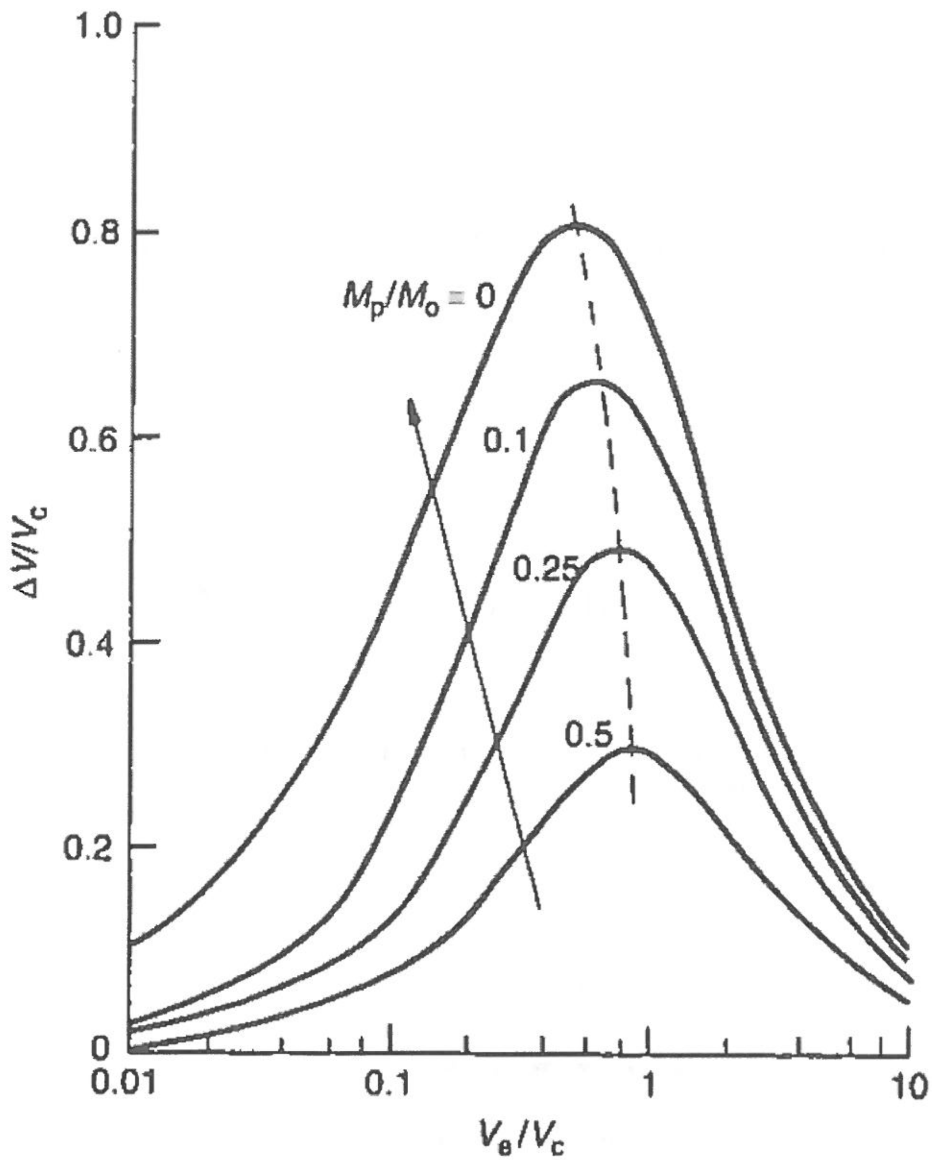


Figure 6.24 Separately powered electric rocket performance

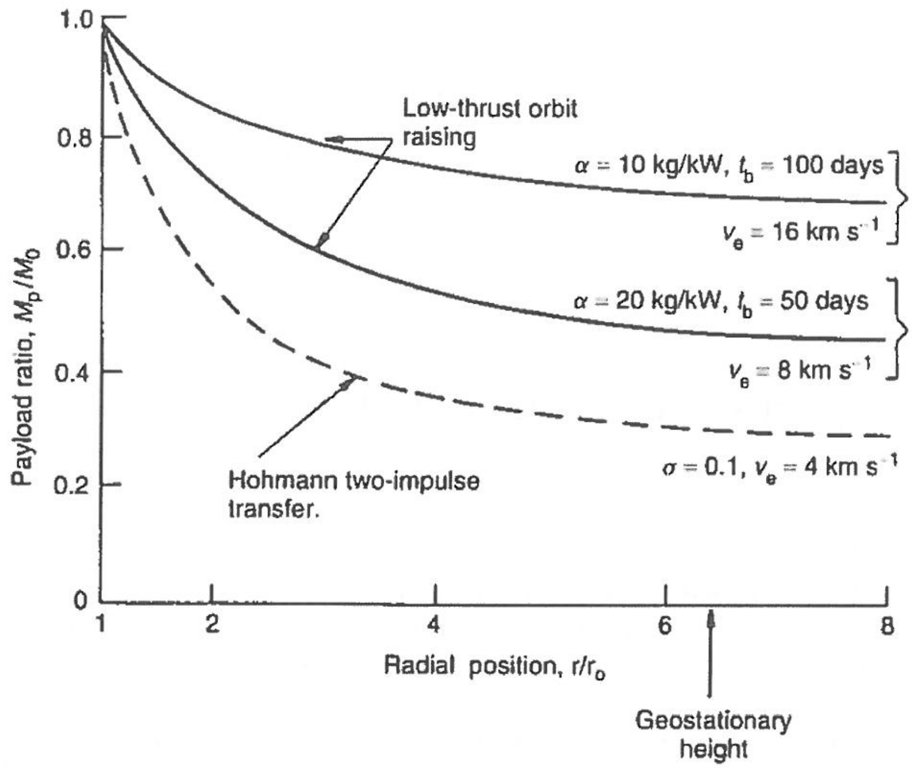


Figure 6.25 Comparative performance: chemical impulse versus low-thrust orbital transfer

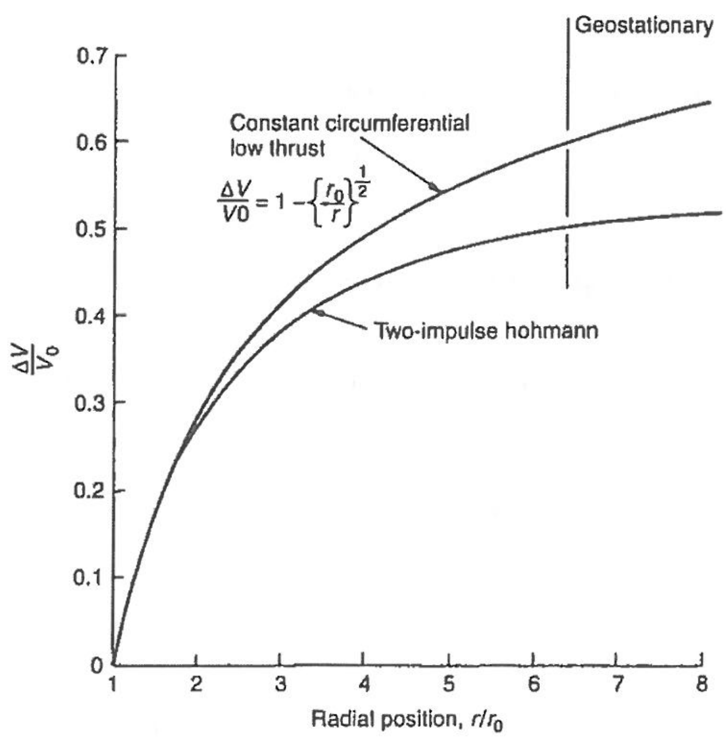


Figure 6.26 Comparative ΔV requirement for transfer between circular orbits r_0 to r