FUNDAMENTALS OF ROCKET PROPULSION

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1. INTRODUCTION

- **TRANSPARENCIES AVAILABLE ON WEBSITE:**
  dma.dima.uniroma1.it:8080/STAFF2/lentini.html (under *Lecture Notes*)
- **diego.lentini@uniroma1.it, tel. ++39–0644585281**
- **CLASSICAL TEXTBOOK:**
  G.P. Sutton and O. Biblarz, *Rocket Propulsion Elements*, now 9th ed., Wiley, 2016

1.2 GOALS OF SPACE PROPULSION

- ***PROPELLERE*** **(Latin) = 'PUSH FORWARD' IN ORDER TO:**
  - **ATTAIN VERY HIGH SPEED**
  - **OVERCOME GRAVITY/AERODYNAMIC FORCES**
  - **MANOUVRES**
  - **ATTITUDE ADJUSTMENT**
  - **DECELERATE (RE–ENTRY, SOFT LANDING)**
- **THRUST GENERATED AS REACTION TO EXPULSION OF A *PROPULSIVE FLUID***
- **KEY REQUIREMENT: LIGHTWEIGTH (BOTH ENGINE AND PROPELLANTS)**

1.3 THRUST GENERATION

- **REACTION TO EJECTION OF PROPULSIVE FLUID MOMENTUM:**
  - **ROCKET ENGINE:**
    PROPELLANTS STORED ON BOARD
    $\rightarrow$ ***PRODUCTS***

1.4 ENGINEERING GOAL

- **ATTAIN GIVEN TARGET *AT MINIMUM COST***
- **COSTS OF A SPACE MISSION:**
  - **LAUNCHER**
  - **(PAYLOAD)**
  - **LAUNCH OPERATIONS**
  - **MONITORING**
  - **INSURANCE**
  - **PROPELLANTS**
  - **...**

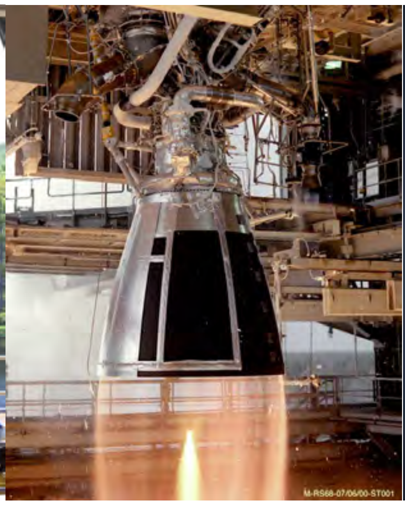
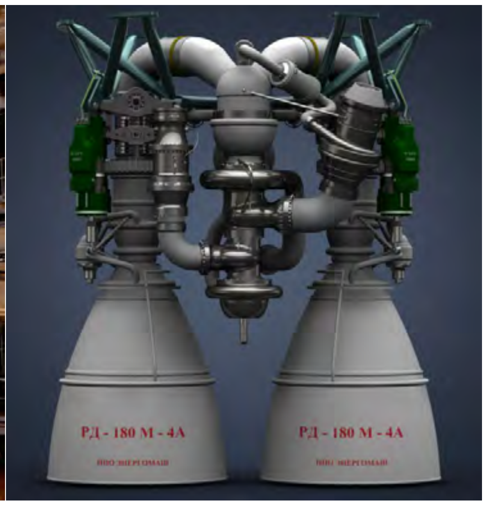
1.5 PERFORMANCE AND PRICE OF SOME ROCKET ENGINES

Engine	thrust sea level/vacuum kN	adopted by	price M$
Vulcain 2	960 / 1 359	Ariane 5	12 (2017)
RS–25	1 860 / 2 279	Space Shuttle	50 (2011)
RS–68	2 950 / 3 137	Delta IV	15 (2006)
RD–180	3 830 / 4 150	Atlas V	23.5 (2017)

– SEARCH FOR SIMPLICITY (TO REDUCE COSTS AND IMPROVE RELIABILITY)

→ CONTAIN *PART COUNT*



1.6 ARIANE 5, RD–170

1.7 CLASSIFICATION BASED ON *ENERGY SOURCE*

– **EXHAUST JET MOMENTUM GENERATED BY MEANS OF ENERGY...**
  * **CHEMICAL: OXIDIZER/FUEL, MONO-PROPELLANT (THERMAL)**
  * **SOLAR: THERMAL, ELECTRIC**
  * **(ELECTRIC): SOLAR, NUCLEAR, CHEMICAL**
  * **NUCLEAR: THERMAL**

1.8 CLASSIFICATION BASED ON *PROPULSIVE PRINCIPLE*

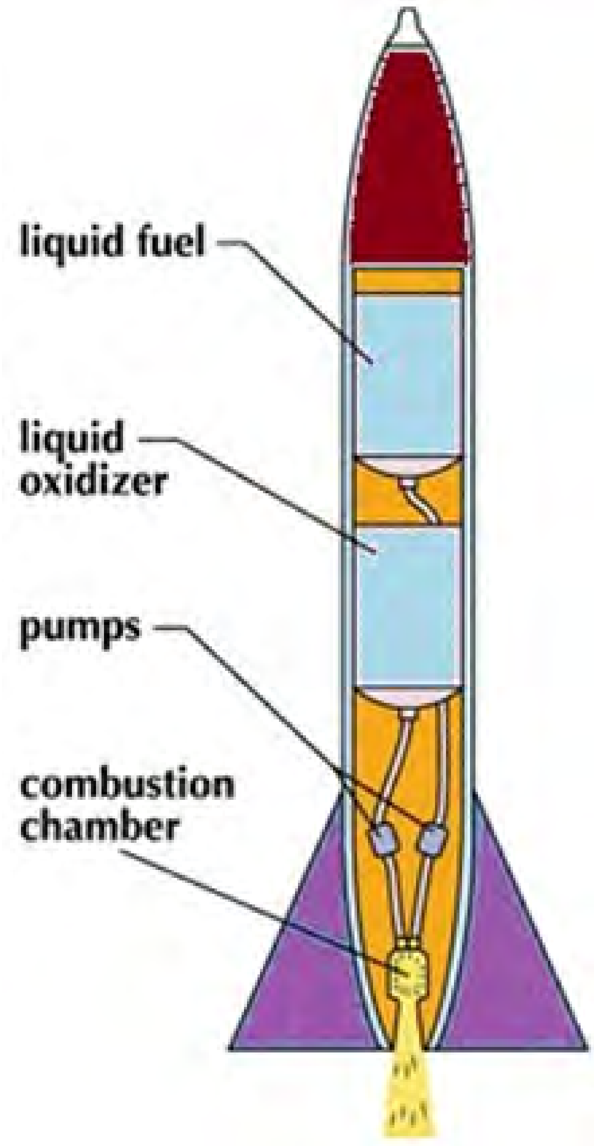
- **TRASFORMATION OF PRIMARY ENERGY INTO JET KINETIC ENERGY:**

  1. ***THERMAL*: CHEMICAL, NUCLEAR, SOLAR**
  2. ***ELECTROSTATIC* (IONS):**
     (ELECTRICAL) ← SOLAR
  3. ***ELECTROMAGNETIC* (PLASMA):**
     (ELECTRICAL) ← SOLAR

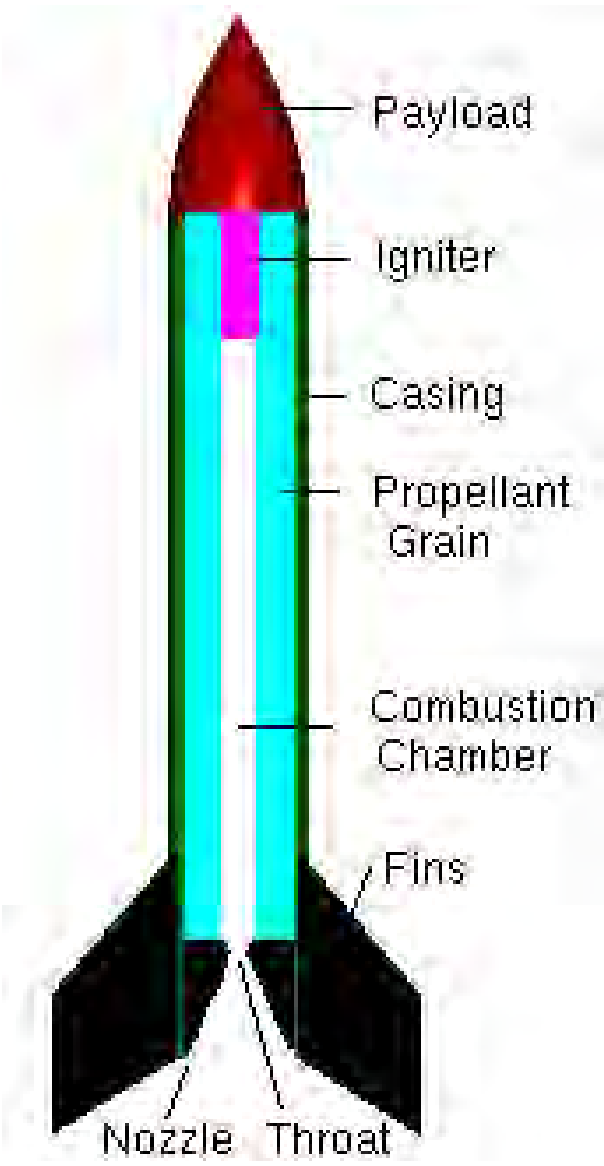
  - **2and3 GIVE LOW RATIOS THRUST/WEIGHT (CANNOT LIFT–OFF FROM THE GROUND)**

1.9 CHEMICAL ROCKET ENGINES (THERMAL)

LIQUID ROCKET ENGINES (LRE)

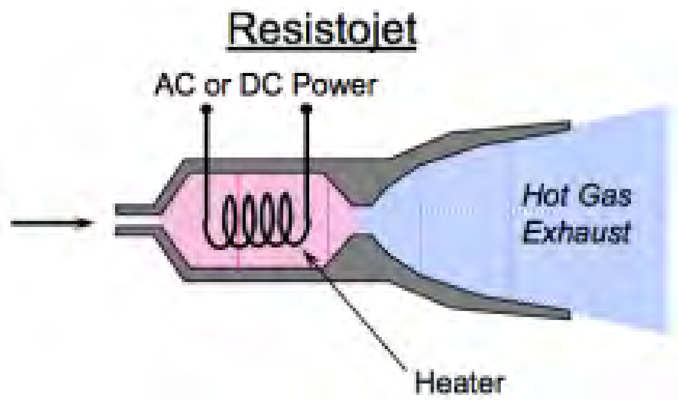


SOLID ROCKET MOTORS (SRM)

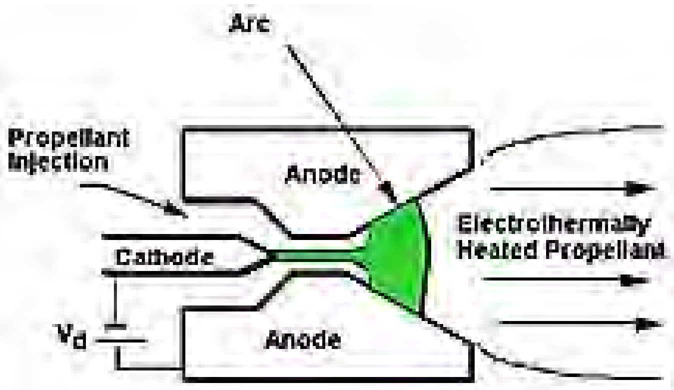


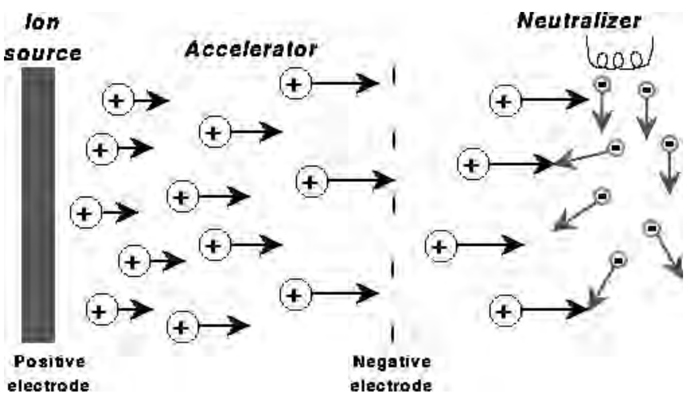
1.10 ELECTRIC THRUSTERS

ELECTROTHERMAL RESISTOJET

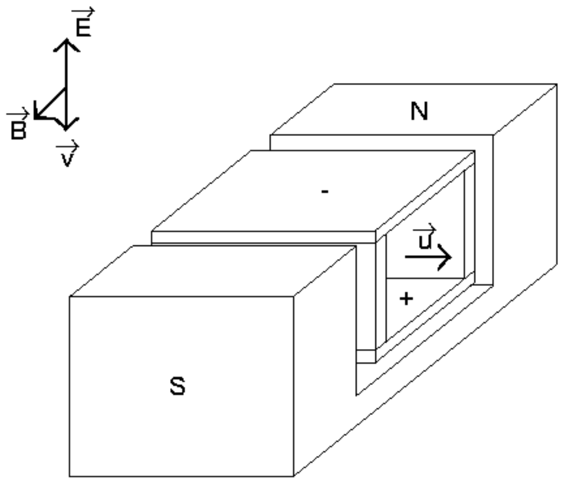


ELECTROTHERMAL ARCJET





ELECTROSTATIC (IONS)



ELECTROMAGNETIC (PLASMA)

2.1 FUNDAMENTALS OF COMPRESSIBLE FLOWS

- **THERMAL ROCKETS USE A FLUID, CARRYING ENERGY IN THE FORM OF HEAT**
- **HEAT TRANSFORMED INTO KINETIC ENERGY IN THE *NOZZLE* TO GENERATE *THRUST***
- **LAWS FOR *COMPRESSIBILE* FLOWS (HIGH SPEED $u \rightarrow$ VARIABLE ρ):**
  - **MASS CONSERVATION**
  - **MOMENTUM CONSERVATION**
  - **ENERGY CONSERVATION**

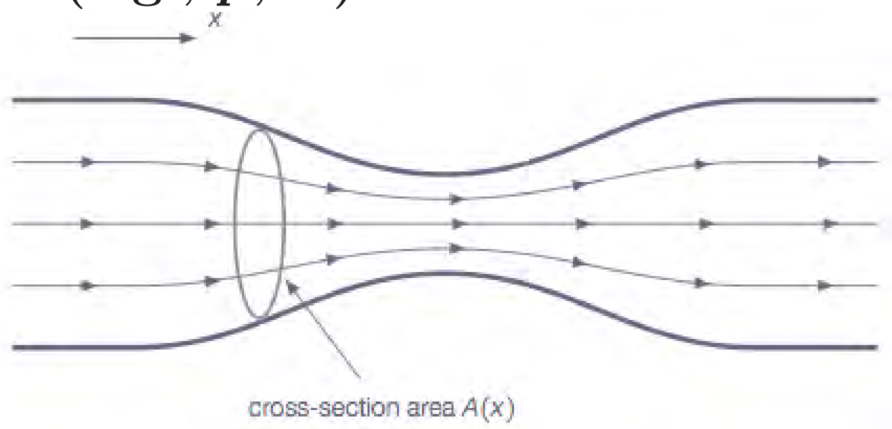
2.2 SOUND SPEED, MACH NUMBER

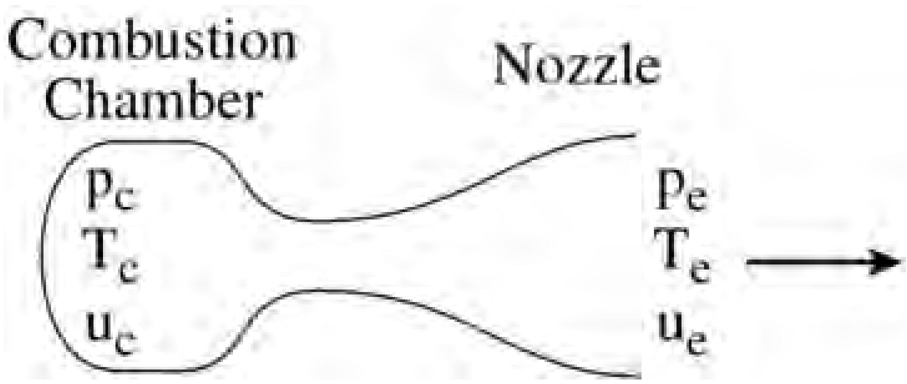
- $a = \sqrt{\gamma R T} = \sqrt{\gamma \frac{p}{\rho}}$

- $M = u/a$

2.3 EQS. OF MOTION: QUASI 1–D FLOW

- $u, p, T = f(x)$, $A = f(x)$
- **1 QUANTITY TO IDENTIFY STATE OF MOTION (e.g., VELOCITY COMPONENT u, or M)**
- **2 QUANTITIES TO IDENTIFY THERMODYNAMIC STATE (e.g., p, T)**





2.4 FLOW EVOLUTION IN A DUCT SET MOTION EQS.

- **3 EQS. (MASS, MOMENTUM, ENERGY) in u, p, T**
- **IF DETERMINANT OF COEFFICIENT MATRIX $\neq 0 \rightarrow 1$ SOLUTION**
- **IF DETERMINANT OF COEFFICIENT MATRIX $= 0 \rightarrow M = 1 \rightarrow$ *MORE* THAN 1 SOLUTION**

2.5 FLOW WITH AREA CHANGE (*ISENTROPIC*)

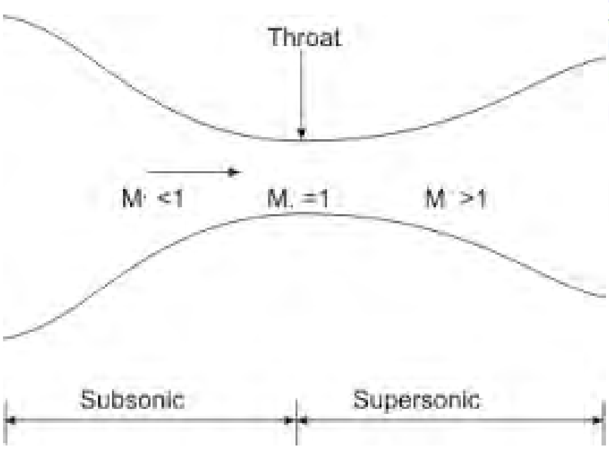
- CRITICAL SECTION (UNITY MACH no.) <u>MUST</u> BE AT THE NOZZLE THROAT ($dA = 0$)
- TO ATTAIN $M = 1$ AT THROAT, $p_c \geq 2 \cdot p_a$

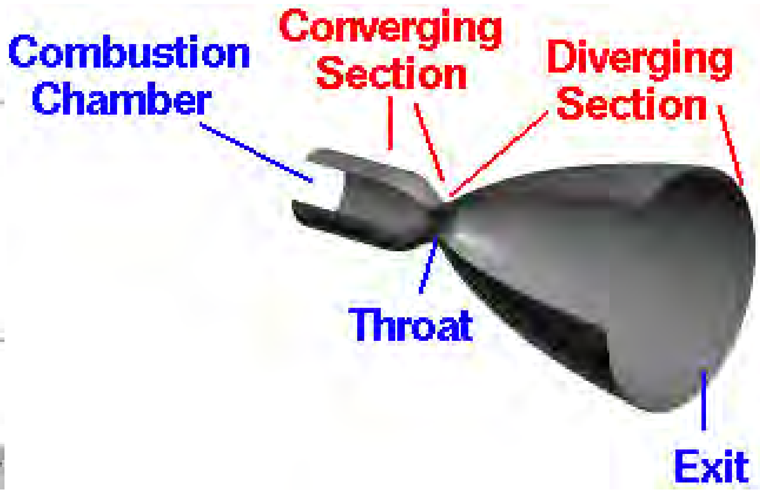
∘ *CONVERGING* DUCTS ($dA < 0$):

$$du \begin{cases} > 0 & \text{for } M < 1 \\ < 0 & \text{for } M > 1 \end{cases} ; \; dp \begin{cases} < 0 & \text{for } M < 1 \\ > 0 & \text{for } M > 1 \end{cases} ; \; dT \begin{cases} < 0 & \text{for } M < 1 \\ > 0 & \text{for } M > 1 \end{cases}$$

∘ *DIVERGING* DUCTS ($dA > 0$):

$$du \begin{cases} < 0 & \text{for } M < 1 \\ > 0 & \text{for } M > 1 \end{cases} ; \; dp \begin{cases} > 0 & \text{for } M < 1 \\ < 0 & \text{for } M > 1 \end{cases} ; \; dT \begin{cases} > 0 & \text{for } M < 1 \\ < 0 & \text{for } M > 1 \end{cases}$$



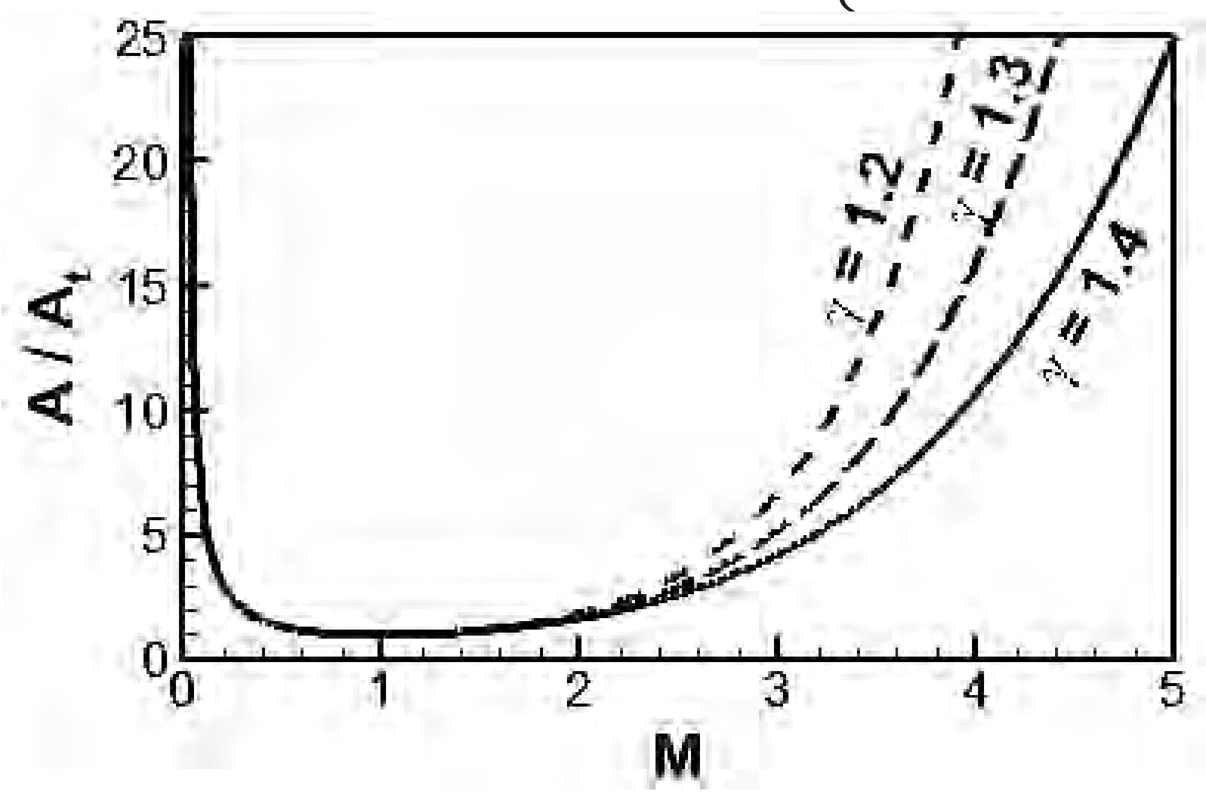


2.6 CHOCKED FLOW AT THE THROAT ($M_t = 1$) (AREA–MACH no. LAW)

- **SECTION** $1 = t$ ($M = 1$), $2 =$ **GENERIC ONE**

$$\boxed{\frac{A}{A_t} = \frac{1}{M}\left(\frac{1 + \frac{\gamma-1}{2}M^2}{\frac{\gamma+1}{2}}\right)^{\frac{\gamma+1}{2(\gamma-1)}}} \qquad (1)$$

- **PER EACH VALUE OF** A/A_t $\begin{cases} \text{SUBSON. SOLUT.} \\ \text{SUPERSON. SOLUT} \end{cases}$



2.7 MASS FLOW RATE FOR ISENTROPIC, CHOCKED FLOW

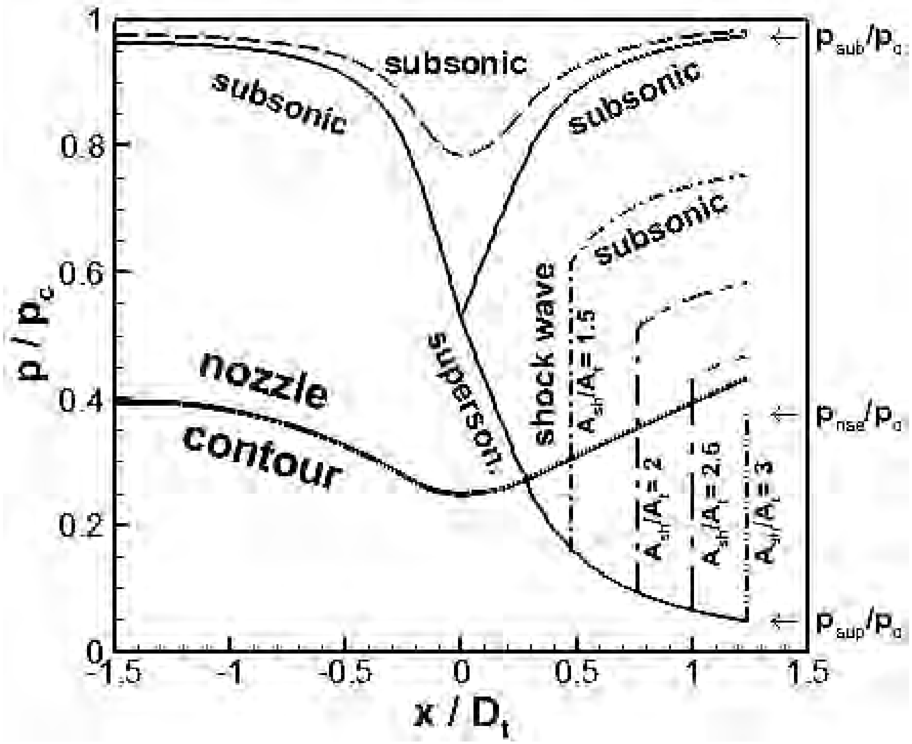
- $M_t = 1 \;\rightarrow$
$$\boxed{\dot{m} \;=\; \Gamma \, \frac{p_c}{\sqrt{R\,T_c}}\, A_t} \qquad (2)$$

- **GIVEN THE VALUES UPSTREAM OF THE NOZZLE (COMBUSTION CHAMBER) T_c, p_c and A_t $\rightarrow$ $\dot{m}$ <u>FIXED</u>**

2.8 ISENTROPIC/NON–ISENTR. SOLUTIONS

$$\frac{A}{A_t} = \frac{1}{M}\left(\frac{1+\frac{\gamma-1}{2}M^2}{\frac{\gamma+1}{2}}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \qquad \text{(CHOCKED THROAT)} \quad (3)$$

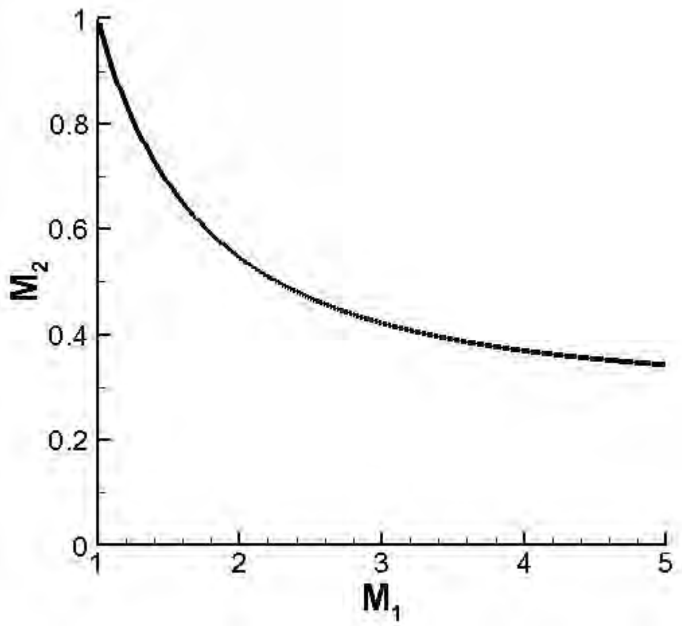
- **SINGLE VALUE A/A_t PER EACH M, BUT...**
- **2 VALUES M PER EACH VALUE A/A_t**
- **WHICH SOLUTION WILL PREVAIL?**
  IT DEPENDS UPON RATIO p_a/p_c (ambient/chamber)

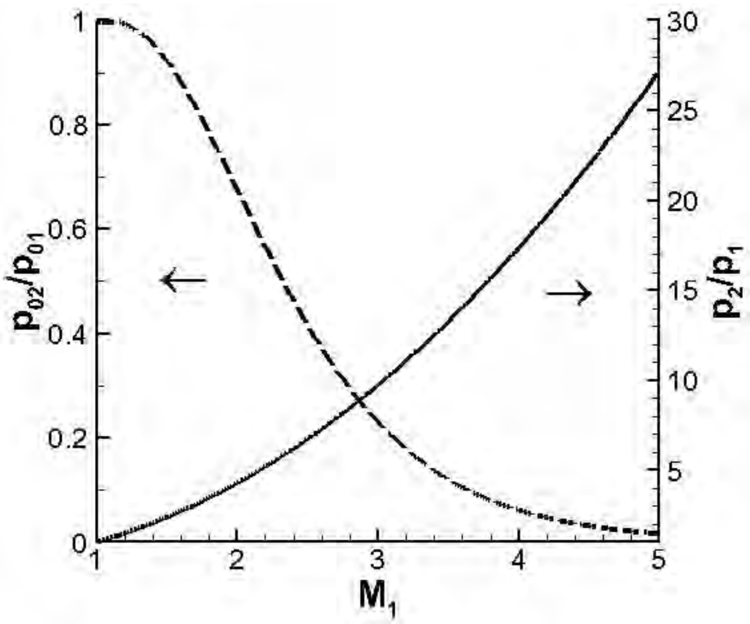


(p_{sub}, p_{sup} exit pressure of subsonic/supersonic *isentropic* solutions

A_{sh} area of cross–section where normal shocks takes place)

2.9 NORMAL SHOCK RELATIONSHIPS





1 → CONDITIONS UPSTREAM OF SHOCK

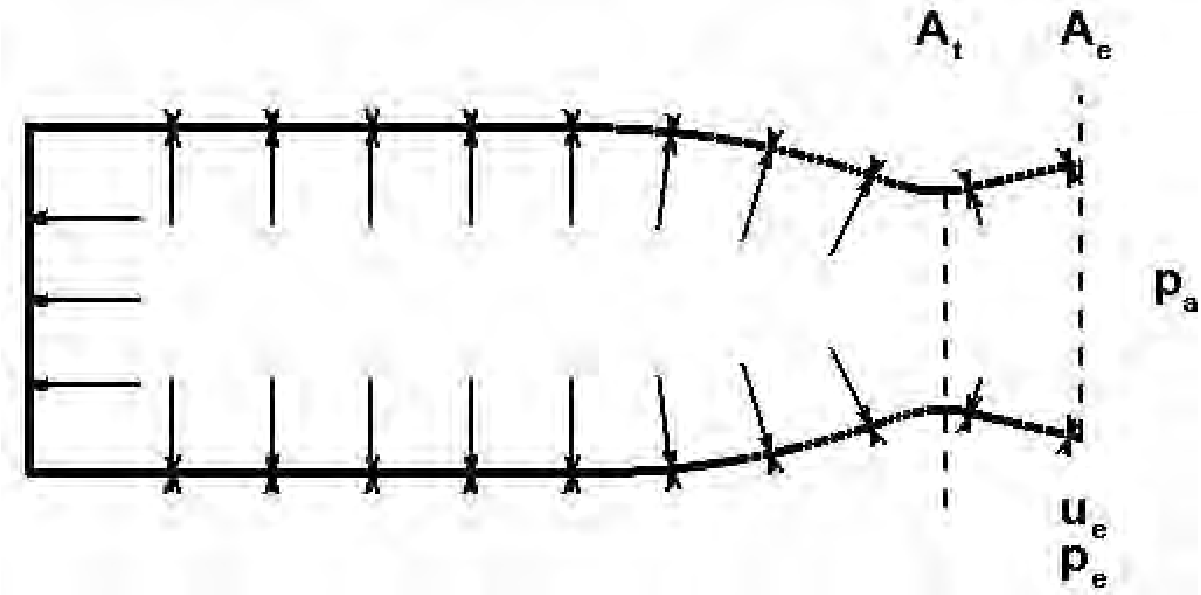
2 → CONDITIONS DOWNSTREAM OF SHOCK

3.1 PERFORMANCE INDICES

- **IDENTIFY PERFORMANCE INDICES (THRUST, EXHAUST VELOCITY, SPECIFIC IMPULSE, PROPELLANT CONSUMPTION, ...)**
- **THRUST GENERATION**

$\rightarrow$ **APPLY MOMENTUM EQ.**

3.2 ROCKET ENGINE THRUST



$$\boxed{\boldsymbol{F = \dot{m}\, u_e + A_e\,(p_e - p_a)}} \qquad (4)$$

- **THRUST =**
  IMPULSIVE THRUST + PRESSURE THRUST

3.3 CONDITIONS FOR OPTIMAL THRUST

$$\boldsymbol{F = \dot{m}\, u_e + A_e\,(p_e - p_a)} \tag{5}$$

- **GIVEN $A_t, T_c, p_c \rightarrow \dot{m}$ (FOR CHOCKED FLOW)**
- **SEEK FOR MAXIMUM BY DIFFERENTIATION wrt $A_e \rightarrow dA_e,\ du_e,\ dp_e$**

$$\begin{aligned} dF &= \dot{m}\, du_e + A_e\, dp_e + (p_e - p_a)\, dA_e = \\ &= A_e(\rho_e\, u_e\, du_e + dp_e) + (p_e - p_a)\, dA_e = (p_e - p_a)\, dA_e \end{aligned} \tag{6}$$

- $\boxed{p_e = p_a}$ **(ADAPTED NOZZLE)**
- p_a **VARIES WITH ALTITUDE**
- **PRESSURE TERM $A_e(p_e - p_a)$ USUALLY SMALL**

3.4 EFFECTIVE EXHAUST VELOCITY

- **FOR AN ADAPTED NOZZLE:**

$$\boldsymbol{F = \dot{m}\, u_e} \tag{7}$$

- **OTHERWISE:**

$$\boldsymbol{F = \dot{m}\, u_{eff}} \tag{8}$$

- **BY DEFINING**

$$\boxed{\boldsymbol{u_{eff} = \frac{F}{\dot{m}}}} \boldsymbol{= u_e + \frac{(p_e - p_a)\, A_e}{\dot{m}}} \tag{9}$$

3.5 ROCKET PERFORMANCE INDICES

- **EFFECTIVE EXHAUST VELOCITY** (in m/s)

$$\boxed{u_{eff} = \frac{F}{\dot{m}}} \tag{10}$$

- **SPECIFIC IMPULSE** (in s; more used on historical grounds)

$$\boxed{I_{sp} = \frac{F}{\dot{w}}} \tag{11}$$

$$I_{sp} = \frac{F}{g_0 \, \dot{m}} \quad \rightarrow \quad \boxed{I_{sp} = \frac{u_{eff}}{g_0}} \tag{12}$$

$$g_0 = 9.80665 \, \frac{\mathrm{m}}{\mathrm{s}^2} \tag{13}$$

- **ENGINE THRUST/WEIGHT RATIO**

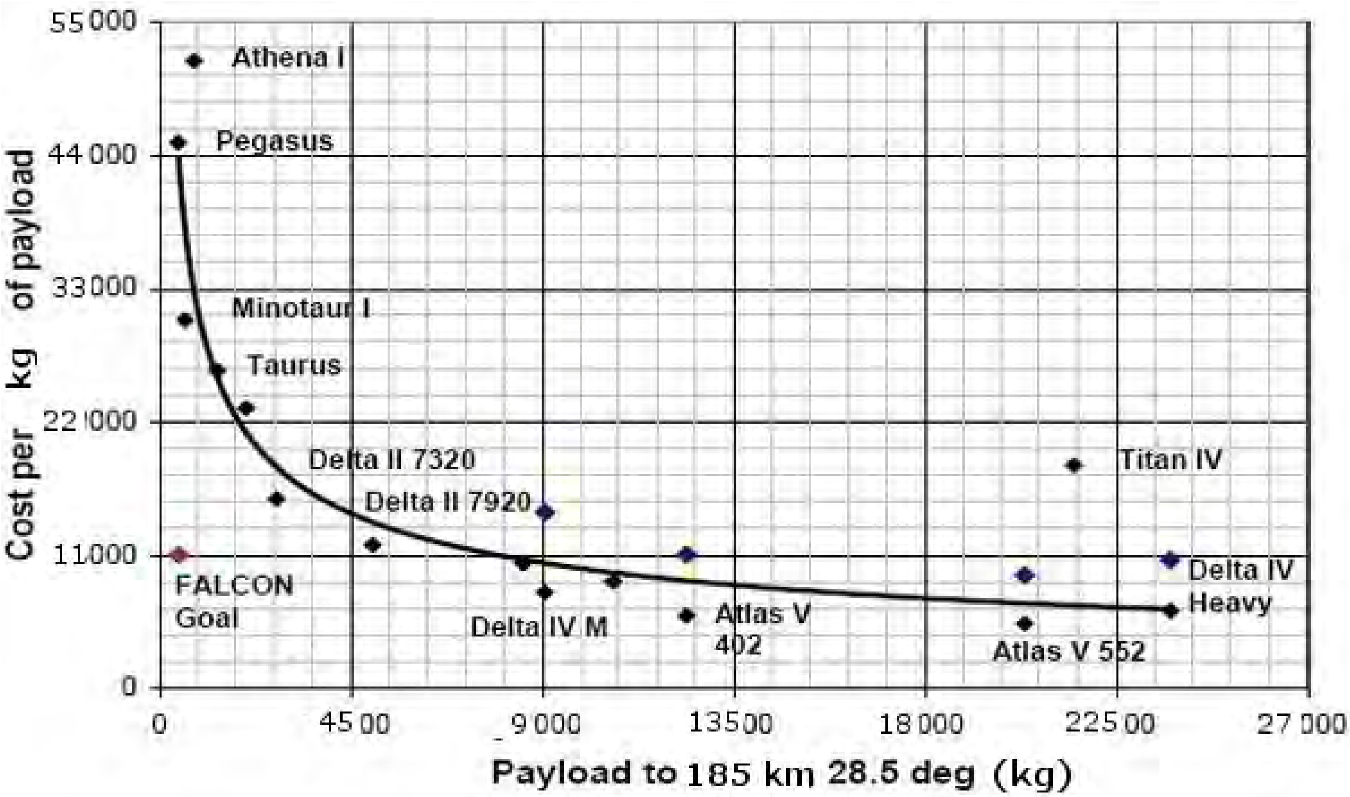
$$\frac{F}{w_{eng}} = \frac{F}{g_0 \, m_{eng}} \tag{14}$$

3.6 TYPICAL VALUES OF ROCKET PERFORMANCE INDICES

Propulsion system	u_{eff} m/s	F/w_{eng}
Chemical, liquid propellants Chemical, solid propellants	$2500 \div 4600$ $2000 \div 3000$	$50 \div 100$
Resistojet Arcjet	$2000 \div 8000$ $4000 \div 20000$	0.06 0.01
Electrostatic Electromagnetic	$25000 \div 34000$ $3000 \div 12000$	0.001 0.0001

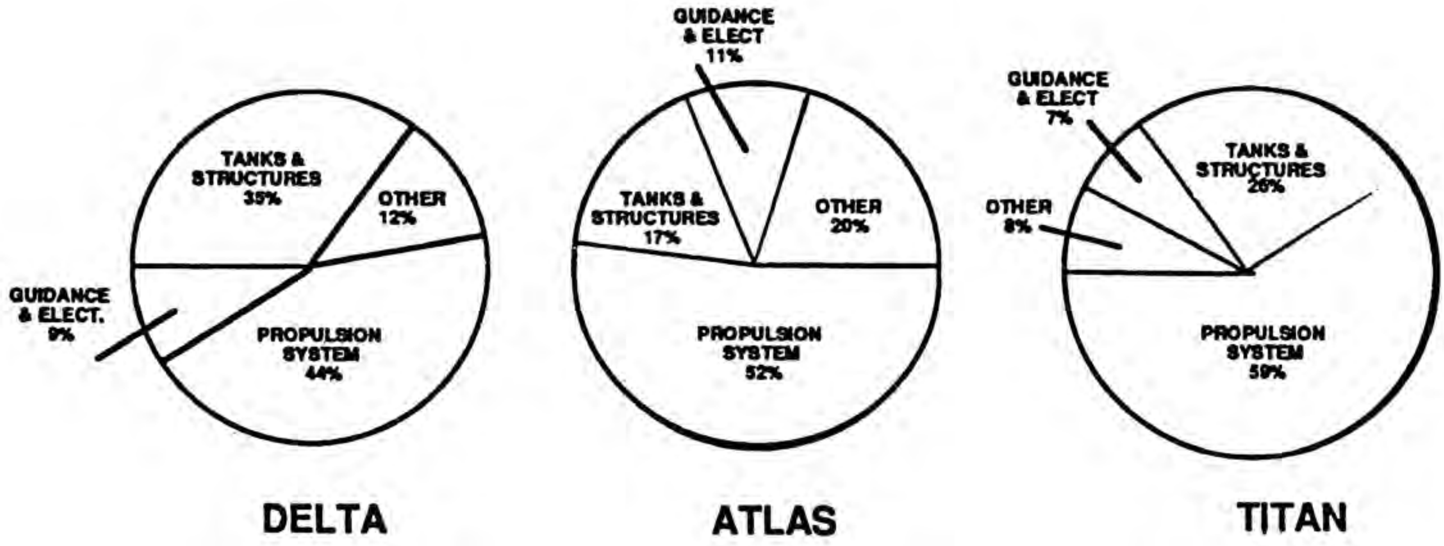
4.1 SPACE MISSIONS REQUIREMENTS (Δv)

- **PROPELLANT MASS >80%**
  → TIGHT COUPLING BETWEEN DESIGN OF LAUNCHER AND PROPULSION SYSTEM
- **HIGH COST ORBIT INJECTION:**
  - **LEO(*LOW EARTH ORBIT*)**
  - **GEO(*GEOSTATIONARY EQUATORIAL ORBIT***
  - **COST GEO ≃ COST LEO · (5÷10)**



4.2 BREAKDOWN OF LAUNCHER COST

- **PROPULSION SYSTEM DOMINATING: 44% TO 59%; > 70% INCLUDING TANKS**
- **PLUS COSTS LAUNCH OPERATIONS, TRACKING, INSURANCE...**



4.3 WHY USING ROCKETS?

- **SPACE MISSIONS REQUIRE:**
  - **VERY HIGH SPEED**
  - **OPERATING OUTSIDE THE ATMOSPHERE**

JET ENGINE $\quad F \simeq \dot{m}_a (u_e - v)$

ROCKET $\quad F = \dot{m}\, u_e + A_e (p_e - p_a)$

- **THRUST JET ENGINE LIMITED BY** $u_e - v$
- **ROCKET THRUST INDEPENDENT OF** v
  $\rightarrow$ **ANY** v **CAN BE ATTAINED (IN PRINCIPLE)**
- **(**u_e **jet exit velocity,** v **aircraft/rocket speed)**

4.4 CIRCULAR ORBIT VELOCITY

- m **SATELLITE MASS**
- M_E **EARTH MASS**
- G **UNIVERSAL GRAVITATIONAL CONST.**
- r **ORBIT RADIUS**
- μ_E **EARTH SOURCE POTENTIAL CONST.**

$$\frac{m\,v^2}{r} = G\,\frac{m\,M_E}{r^2} \qquad (15)$$

$$\mu_E = G\,M_E = 3.986 \cdot 10^{14}\,\frac{\mathrm{m}^3}{\mathrm{s}^2} \qquad (16)$$

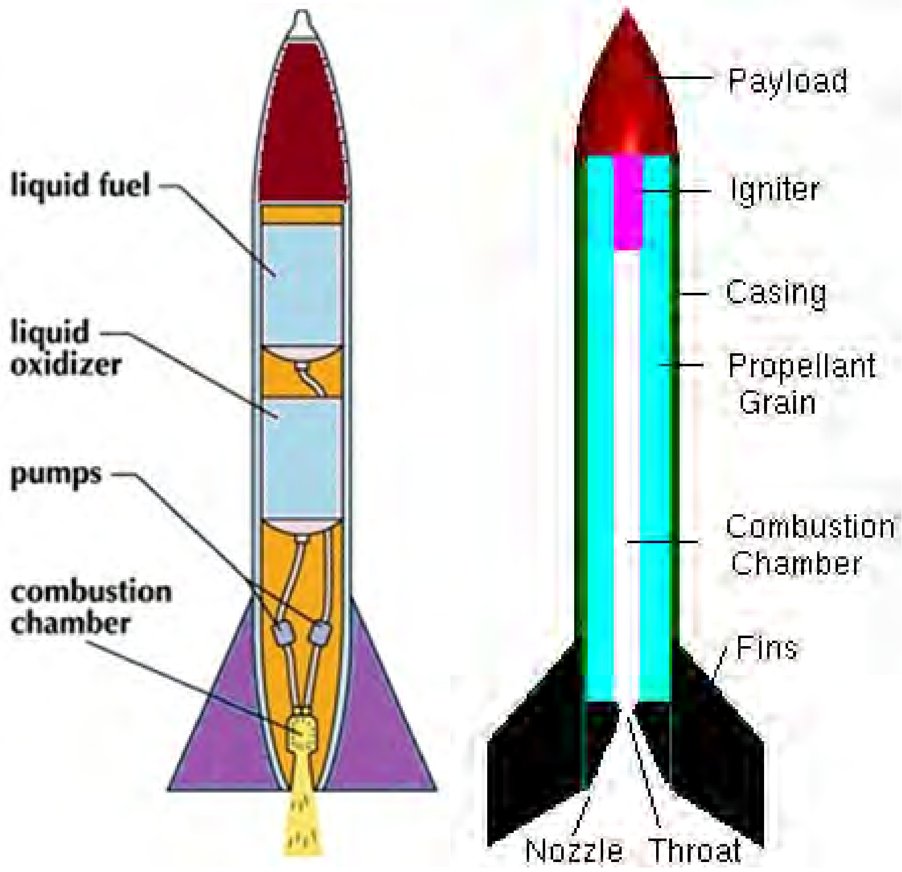
$$v_c = \sqrt{\frac{\mu_T}{r}} \qquad (17)$$

- **FOR AN ORBIT OF ALTITUDE** h**=0** $\rightarrow r = R_{E,equa}$

$$v_c = \sqrt{\frac{3.986 \cdot 10^{14}}{6378 \cdot 10^3}} = 7905\,\frac{\mathrm{m}}{\mathrm{s}} \qquad (18)$$

- **FOR ORBIT at** h**=200 km** $\rightarrow v_c \simeq$ **7800 m/s**
- Δv **<u>IDEAL</u>**

4.5 DEFINITION OF LAUNCHER MASSES



- m_{pl} **PAYLOAD MASS**
- m_p **PROPELLANT MASS**
- m_s **STRUCTURAL (INERT) MASS**
- m_0 **LAUNCHER INITIAL MASS**
- m_f **LAUNCHER FINAL MASS**

$$m_0 = m_{pl} + m_p + m_s \tag{19}$$

$$m_f = m_{pl} + m_s \tag{20}$$

$$m_f = m_0 - m_p \tag{21}$$

4.6 DEFINITION OF MASS RATIOS

- **MASS RATIO (FINAL/INITIAL)**

$$MR = \frac{m_f}{m_0} \tag{22}$$

- **PAYLOAD RATIO:**

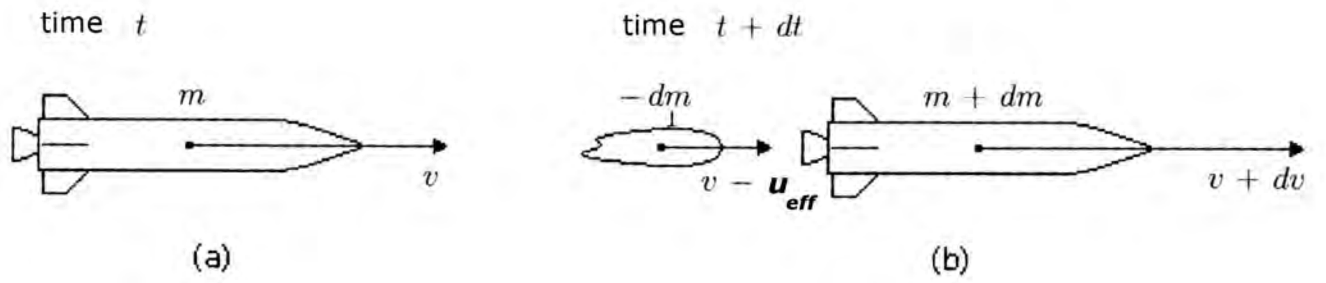
$$\lambda = \frac{m_{pl}}{m_0} \tag{23}$$

- **STRUCTURAL COEFFICIENT**

$$\kappa_s = \frac{m_s}{m_p + m_s} \tag{24}$$

4.7 TSIOLKOVSKY EQUATION

- **RELATES PROPELLANT MASS to REQUIRED Δv**
- **HP: 1. THRUST F ONLY FORCE ON ROCKET;**
  2. THRUST ALIGNED WITH SPEED
  3. EFF. EXHAUST VELOC. $u_{eff} = F/\dot{m}$=const;



$$\boxed{\Delta v = u_{eff} \log_e \frac{m_0}{m_f}} \tag{25}$$

$$m_p = m_0 - m_f \tag{26}$$

4.8 OVERCOMING TSIOLKOVSKY's EQ. LIMITS

- $\boldsymbol{F}$ **ONLY FORCE, ALIGNED WITH ROCKET** $\vec{v}$
- **BUT THERE ARE ALSO WEIGHT+AEROD. DRAG**
- $\boldsymbol{F}$ **NOT ALIGNED WITH** $\vec{v}$ **DURING ASCENT**

→ **PROPULSIVE LOSSES** Δv_{losses}

$$\Delta v_{real} = \Delta v_{ideal} - \Delta v_{losses} \qquad (27)$$

- **LET'S INTRODUCE AN** ***EFFECTIVE*** Δv

$$\Delta v_{eff} = \Delta v_{ideal} + \Delta v_{losses} \qquad (28)$$

- **USE TSIOLKOVSKY WITH** Δv_{eff}
- Δv_{losses} **DEPENDS ON** ***SPECIFIC*** **TRAJECTORY, ANYWAY RATHER WELL IDENTIFIED:**
- **e.g., LAUNCH TO LEO** Δv_{losses}**=1600 – 2000 m/s**

→ $\Delta v_{LEO} \simeq$ **9500 m/s**

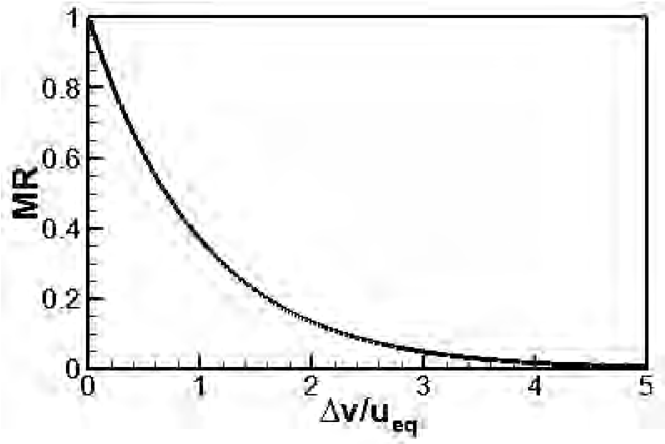
- **UP TO 465 m/s CAN BE RECOVERED FROM EQUATORIAL LAUNCH DUE EAST**
- u_{eff} **NOT CONSTANT, ESPECIALLY FIRST STAGE AND BLOWDOWN SATELLITES**

4.9 Δv REQUIREMENTS OF SOME MISSIONS

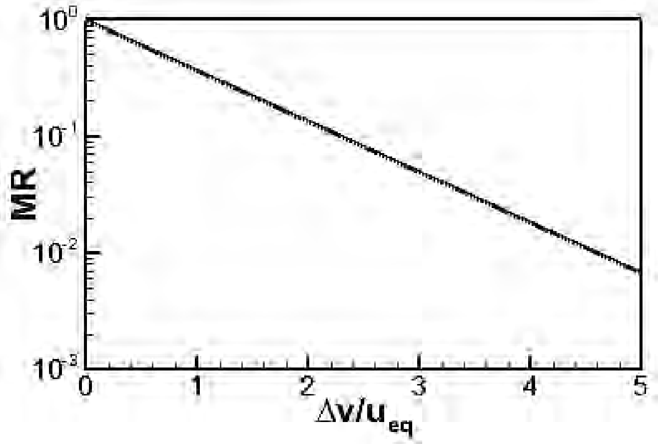
InterContinental Ballistic Missiles (ICBM)	**8000**	**m/s**
Low Earth Orbit (LEO)	**9000**	**m/s**
escape from Earth's gravity; Moon impact	**12500**	**m/s**
Geostationary Equatorial Orbit (GEO)	**13000**	**m/s**
soft landing on the Moon	**15000**	**m/s**
round trip to the Moon	**18000**	**m/s**
round trip to Venus/Mars	**from 18000**	**m/s**
	to 27000	**m/s**
compensation of orbital perturbations	**50**	$\frac{\text{m/s}}{\text{year}}$

- **GOAL OF THE PROPULSION SYSTEM: DELIVER A VELOCITY *INCREMENT* Δv**

4.10 MASS RATIO AS A FUNCTION OF REDUCED Δv



(linear scale)



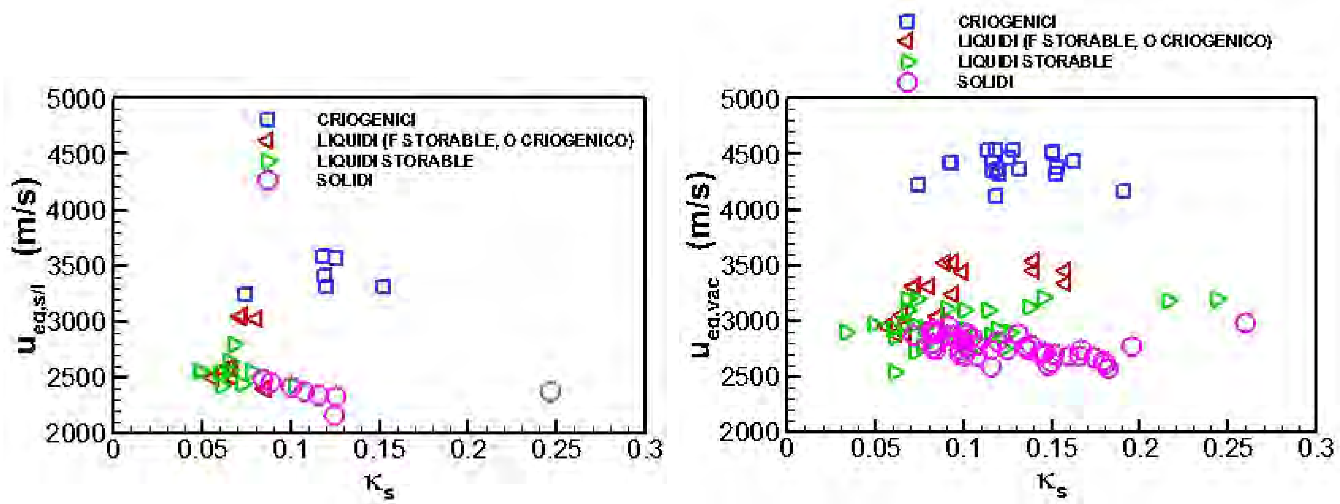
(log scale)

- **BUT WHAT REALLY MATTERS IS THE *PAYLOAD RATIO* $\lambda = m_{pl}/m_0$**

4.11 PAYLOAD RATIO

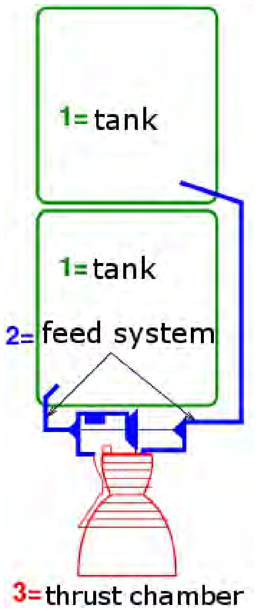
$$\boxed{\lambda = 1 - \frac{1 - MR}{1 - \kappa_s}} \qquad (29)$$

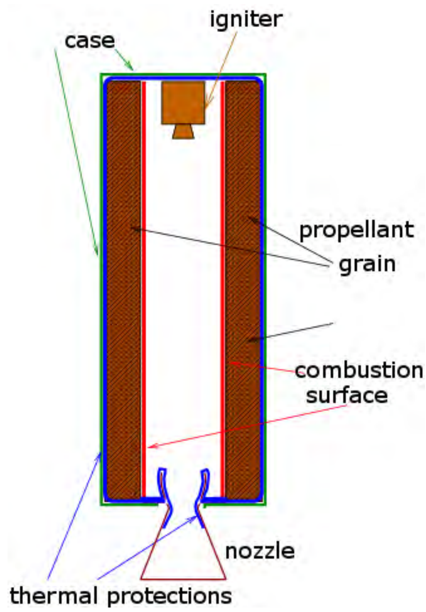
- **WE WISH HIGH u_{eff} AND LOW κ_s, BUT...**

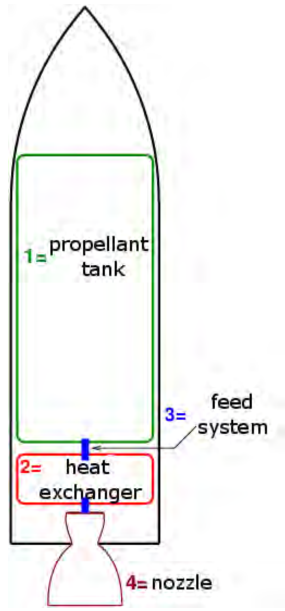


5.1 THERMAL ROCKET ENGINES

- **THERMAL ENERGY CONVERTED INTO KINETIC ENERGY BY NOZZLE**







LIQUID PROP. **SOLID PROP.** **ELECTROTHERMAL**
SOLAR THERMAL
NUCLEAR THERMAL

5.2 CHARACTERISTIC INDICES

$$F = \dot{m}\, u_e + (p_e - p_a)\, A_e$$

$$F = p_c\, A_t \left\{ \sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{p_e}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \overset{\overset{\varepsilon}{\downarrow}}{\frac{A_e}{A_t}} \left(\frac{p_e}{p_c} - \frac{p_a}{p_c}\right) \right\}$$

- $u_{eff} = F/\dot{m}$ EFFECT. EXHAUST VEL. (m/s)
- CAN BE SPLIT IN PRODUCT 2 COEFFICIENTS:
  - CHARACTERISTIC VELOCITY c^*, DETERMINED BY COMBUSTION CHAMBER
  - THRUST COEFFICIENT C_F,

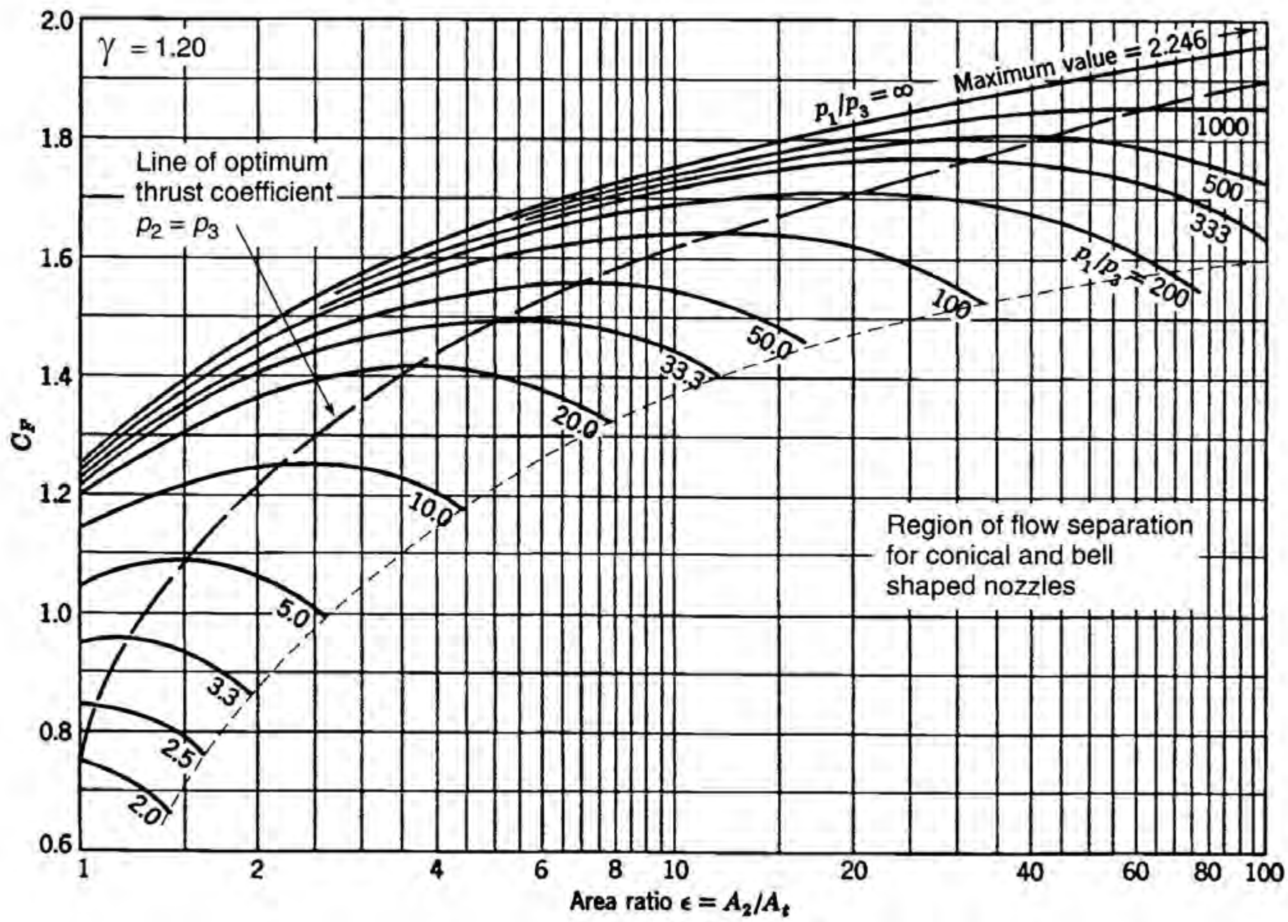
    DETERMINED BY NOZZLE $\varepsilon = A_t/A_e$ EXPANSION RATIO

$$\boxed{c^*} = \frac{p_c\, A_t}{\dot{m}} = \frac{1}{\Gamma}\sqrt{\frac{\mathcal{R}\, T_c}{\mathcal{M}}}; \qquad \boxed{C_F} = \frac{F}{p_c\, A_t}$$

$$\boxed{\rightarrow \qquad u_{eff} = c^* \cdot C_F}$$

5.3 THRUST COEFFICIENT

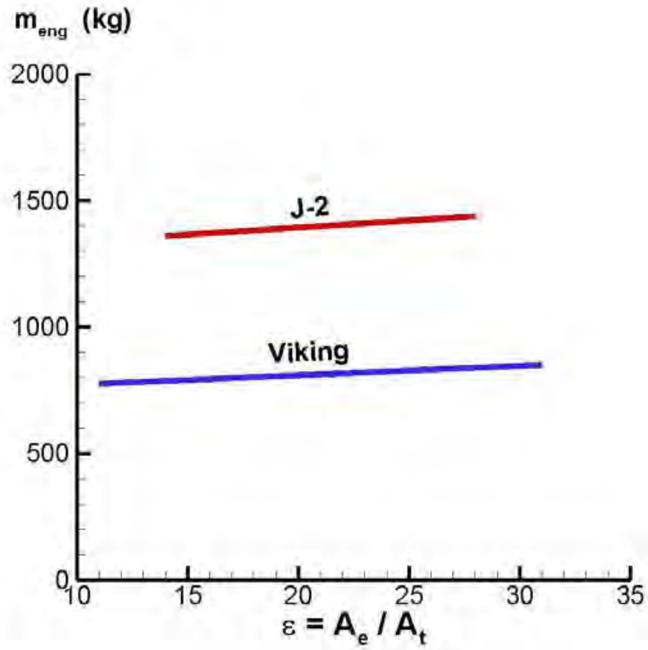
$$C_F = f\left(\gamma, \varepsilon, p_c/p_a\right)$$

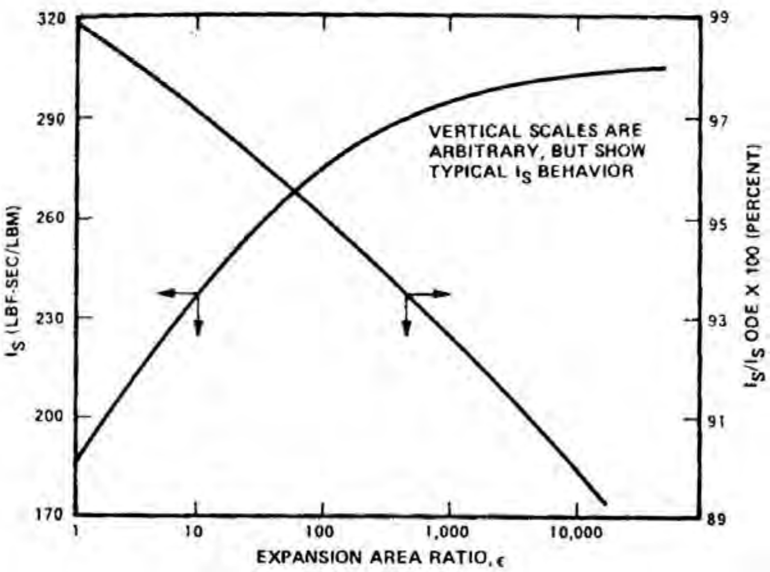


SEPARATION FOR $p_e < (0.25 \div 0.4)\, p_a$

- **HIGHER RATIOS** $p_c/p_a \rightarrow$ **HIGHER** C_F **BUT...**
  REQUIRE HIGHER $\varepsilon = A_e/A_t \rightarrow$
  HIGHER NOZZLE WEIGHT, FRICTION LOSSES

5.4 EFFECT OF EXPANSION RATIO ε ON NOZZLE WEIGHT, EFFICIENCY η_n AND I_{sp}





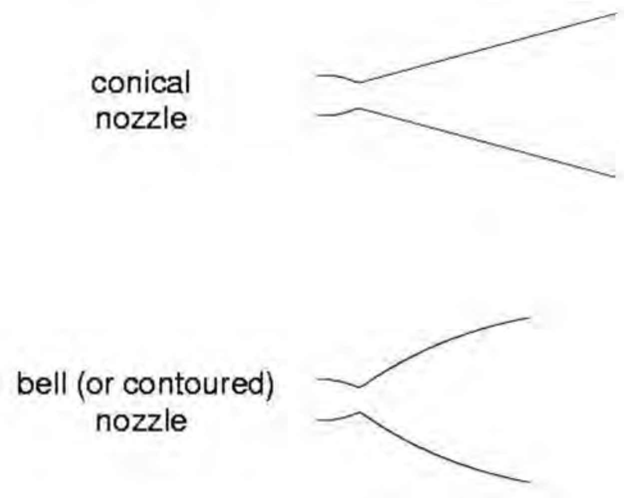
5.5 NOZZLE GEOMETRY: CONICAL *vs.* BELL

- **$12^o \le \alpha \le 18^o$ FOR CONICAL NOZZLES**

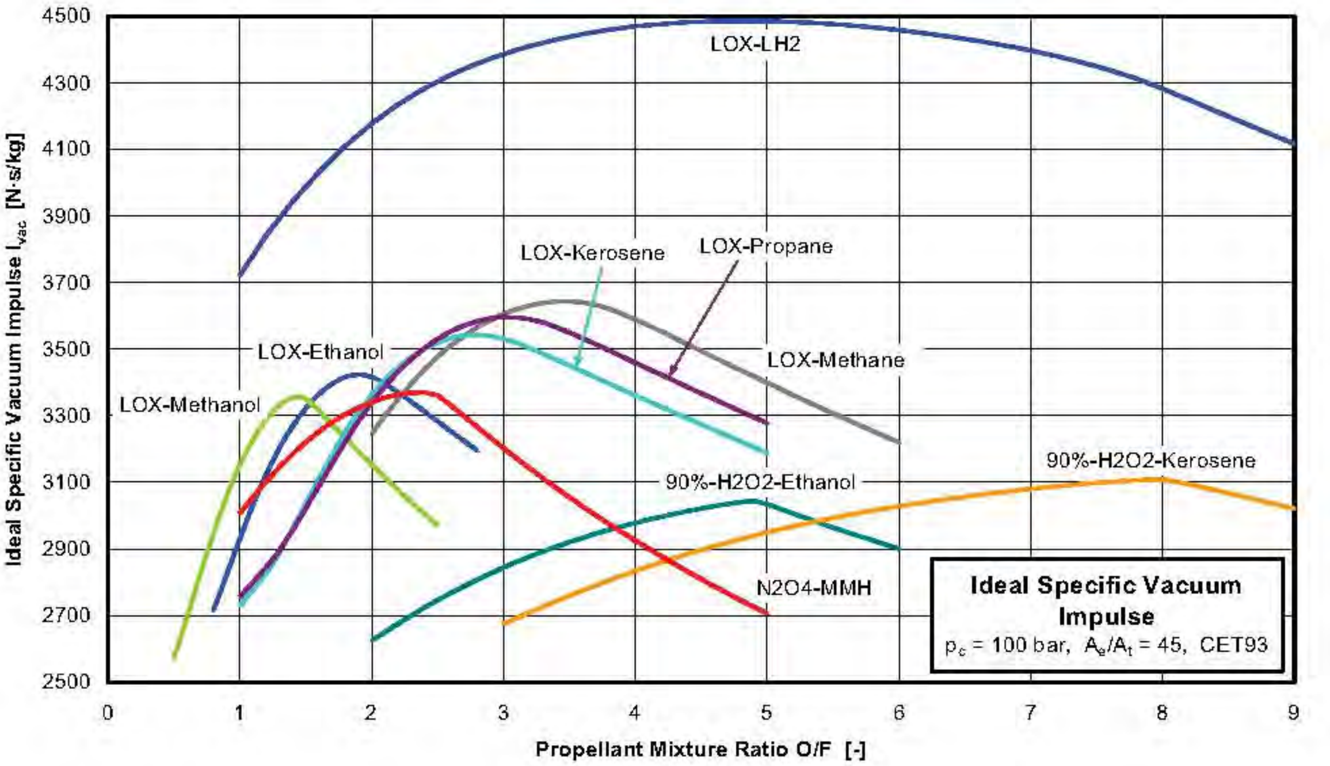
- **FOR BELL NOZZLES:**
- **SEMI–ANGLE IMMEDIATELY DOWNSTREAM OF THROAT $\sim 30^o \div 60^o$, AT EXIT $2^o \div 8^o$**

→ SMALLER DIVERGENCE LOSSES

→ SHORTER (LIGHTER)

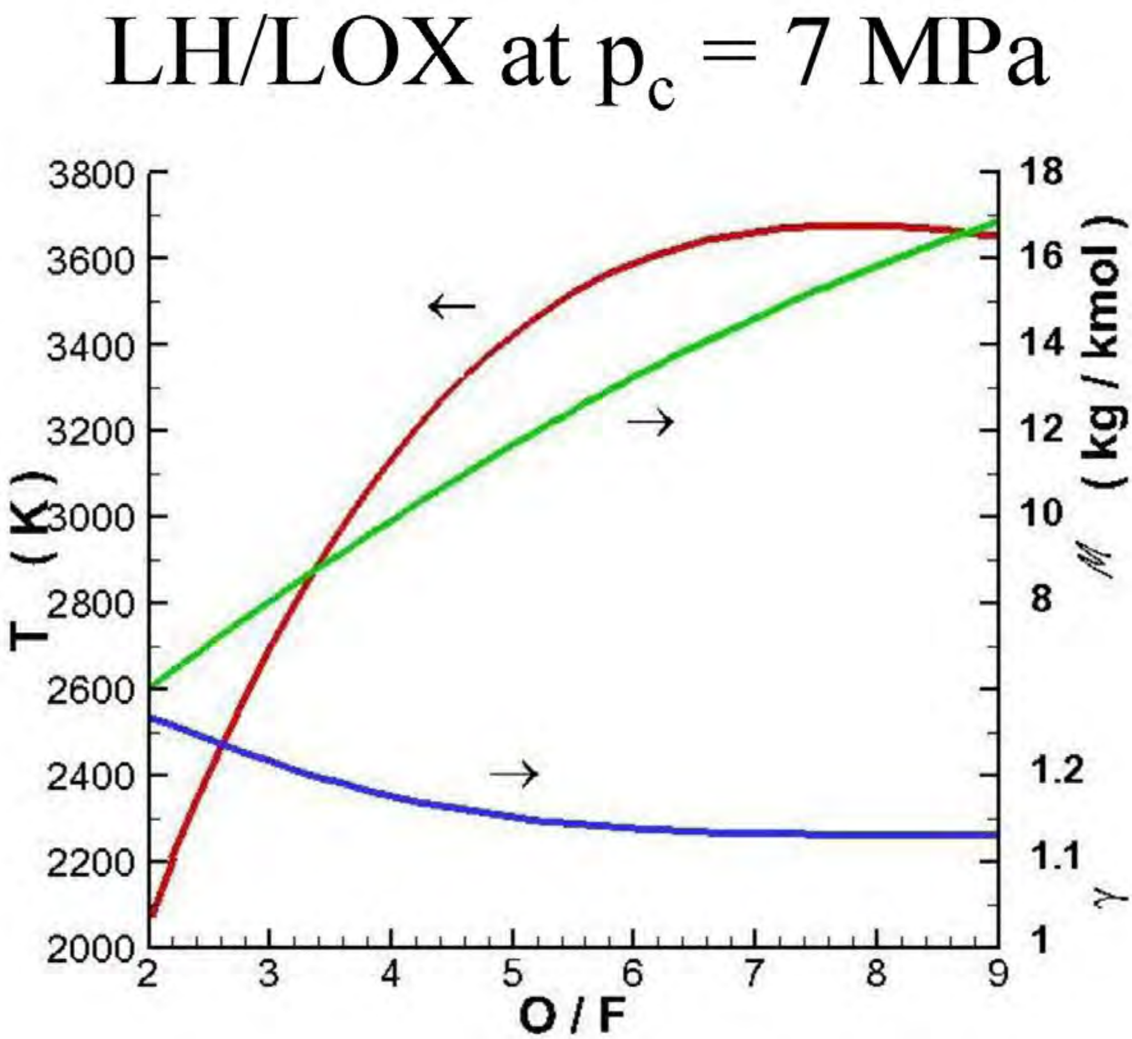


5.6 PERFORMANCE IN VACUUM OF DIFFERENT PROPELLANT COMBINATIONS



- $p_c = 10$ MPa, $\varepsilon = 45$

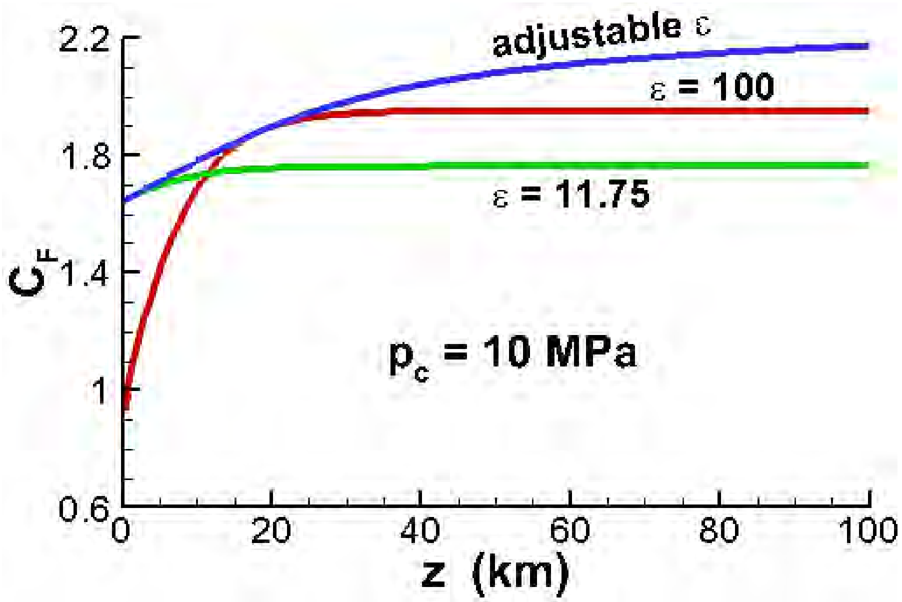
5.7 COMBUSTION CHAMBER CONDITIONS EFFECT OF OXIDIZER/FUEL MASS RATIO O/F (MIXTURE RATIO)

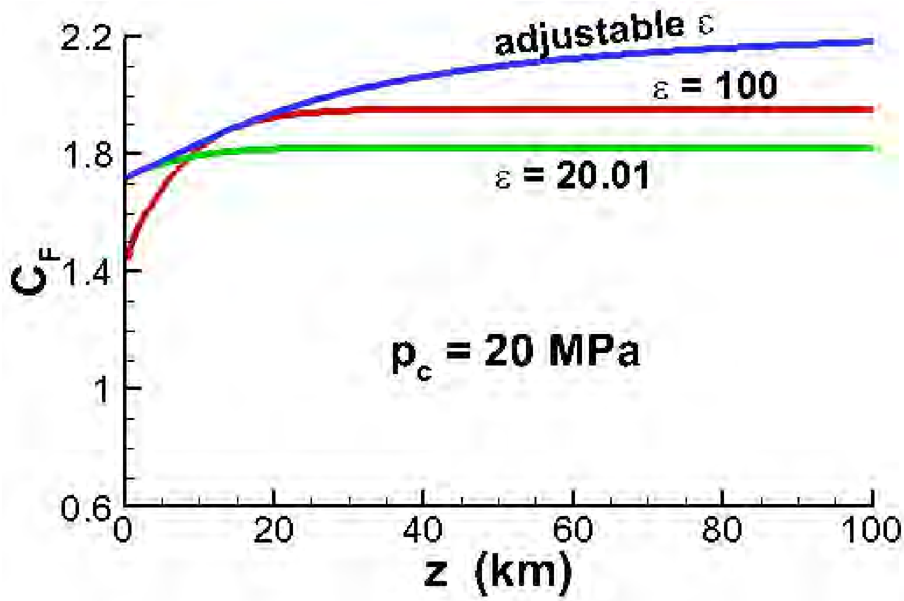


- **BUT LIQUID HYDROGEN DENSITY IS VERY LOW, INCREASING FUEL MASS → HEAVY TANKS**

5.8 NOZZLE BEHAVIOUR FOR VARYING ALTITUDE

- **TO ADAPT NOZZLE TO DECREASING p_a DURING ASCENT, WE'D NEED A VARIABILE ε**





6.1 LIQUID–PROPEL. ROCKET ENGINES (LRE)

1. **PROPELLANT TANKS**
2. **FEED SYSTEM, with SUBSYSTEMS:**
   – **TURBOPUMPS or PRESSURE GAS**
   – **FEED LINES**
   – **VALVES**
3. **THRUST CHAMBER, with SUBSYSTEMS:**
   – **INJECTORS**
   – **COMBUSTION CHAMBER**
   – **NOZZLE**
   – **COOLING SYSTEM**
   – **IGNITION SYSTEM**

- **PROS/CONS:**
  – **HIGH PERFORMANCE** (u_{eff})
  – **RE–IGNITABLE, THROTTLEABLE**
  – **COMPLEX, COSTLY**
  – **CANNOT BE EASILY SCALED UP/DOWN**

6.2 OXIDIZERS

OXYGEN (O_2)

CRYOGENIC **[T_b=90K(1 atm)], ρ=1140kg/m^3**

NITROGEN TETROXIDE (N_2O_4)

STORABLE **[T_{fr} = 262 K, T_b = 294 K (1 atm)], ρ = 1450 kg/m^3, TOXIC, CORROSIVE**

6.3 FUELS

HYDROCARBONS (HC; C_nH_m, CH_x)
CHEAP, WIDELY AVAILABLE:
<u>KEROSENE</u> ($\sim CH_{1,96}$), STORABLE, $\rho \sim 800$ kg/m^3, (USA: RP–1, MINIMAL *FOULING*);
<u>METHANE</u> (CH_4), CRYOGENIC (T_b=112 K), $\rho = 425$ kg/m^3;

<u>HYDROGEN</u> (H_2)
CRYOGENIC (T_b=20 K), VERY LOW ρ=70 kg/m^3

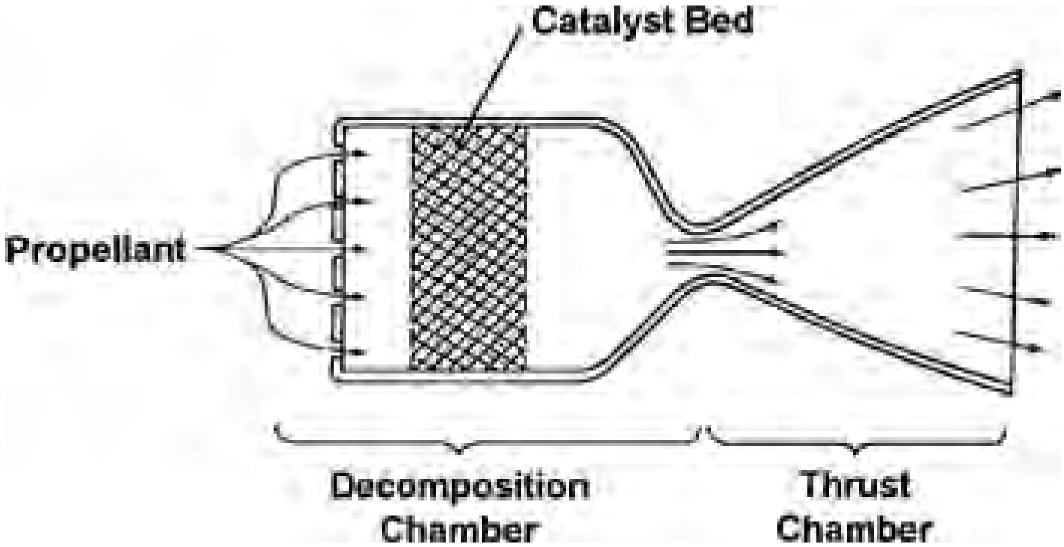
<u>HYDRAZINE</u> (N_2H_4)
STORABLE (T_{fr}=275K, T_b=387K), ρ=1020 kg/m^3, TOXIC, CORROSIVE, CARCINOGEN

<u>UNSYMMETRICAL DIMETHILHYDRAZINE</u> (UDMH
$(CH_3)_2NNH_2$ $\sim$ HYDRAZINE BUT MORE STABLE
(T_{fr}=216 K, T_b=336 K), ρ=850 kg/m^3

<u>MONOMETHILHYDRAZINE</u> (MMH), CH_3NHNH_2
$\sim$ HYDRAZINE, MORE STABLE
(T_{fr}=221 K, T_b=361 K), ρ=875 kg/m^3

AEROZINE : MIXTURE 50% UDMH, 50% N_2H_4

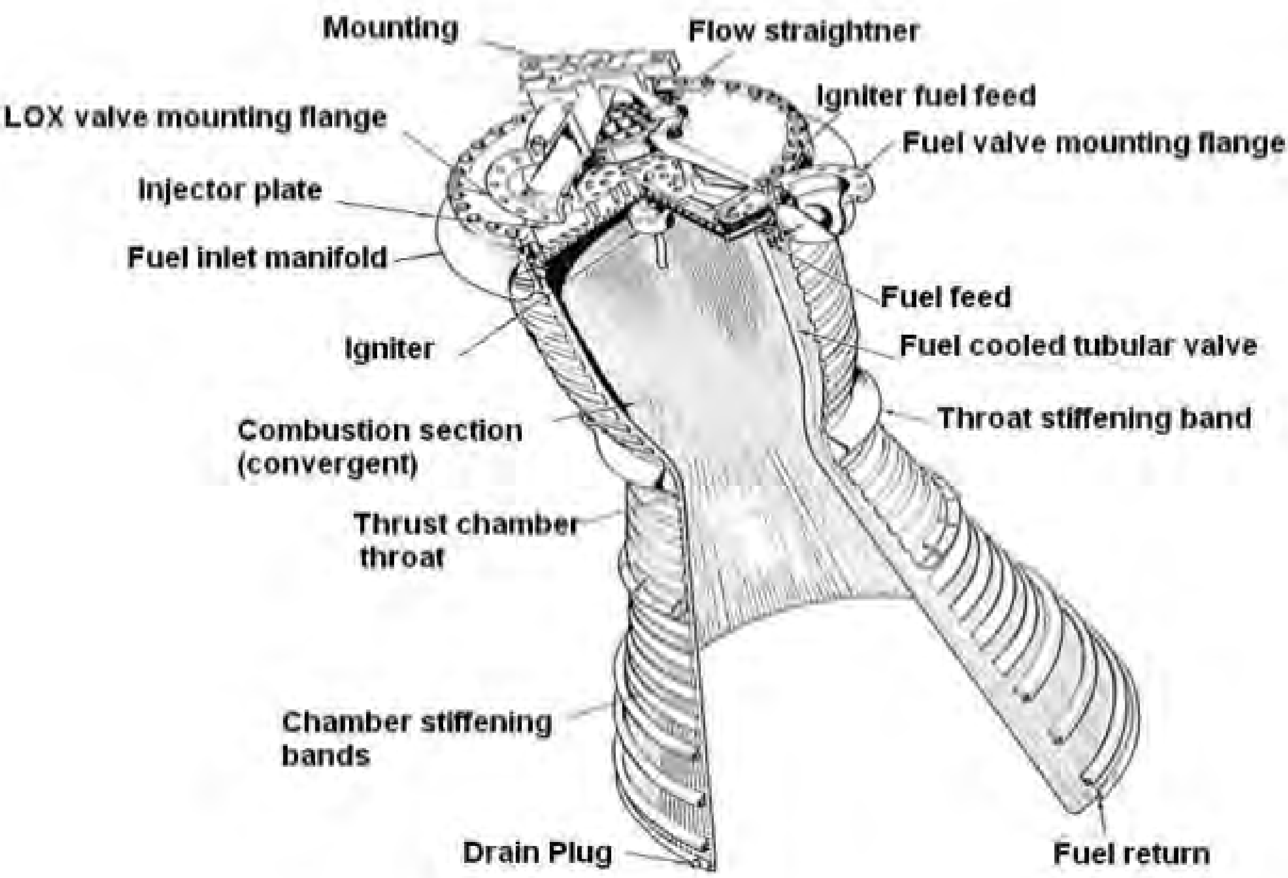
6.4 MONOPROPELLANTS



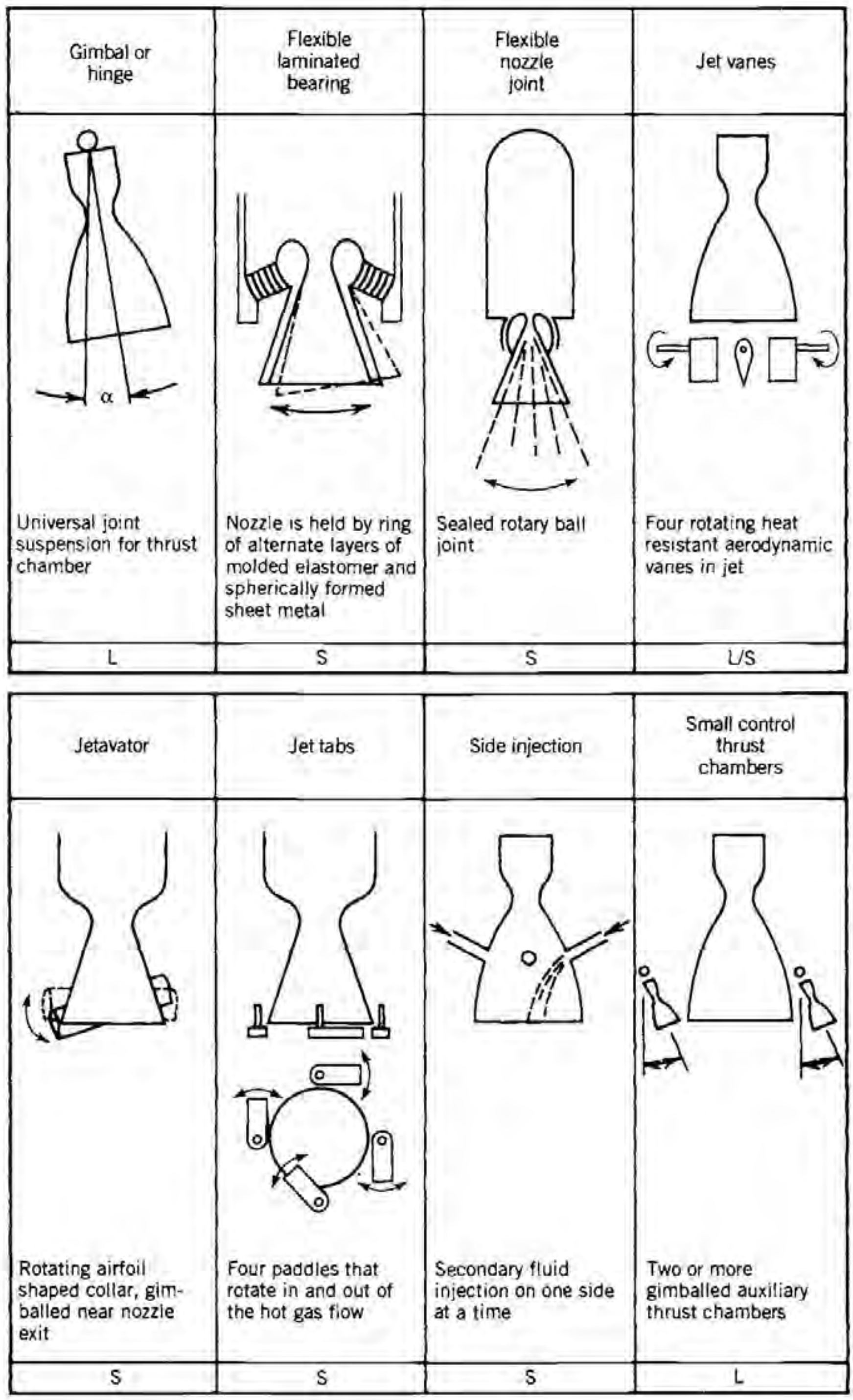
- **<u>HYDRAZINE</u> (N_2H_4), $u_{eff,vac}$ UP TO 2300 m/s**
- **DECOMPOSITION BY CATALYST (T)**
- **(UDMH and MMH TOO MUCH STABLE)**

6.5 THRUST CHAMBER

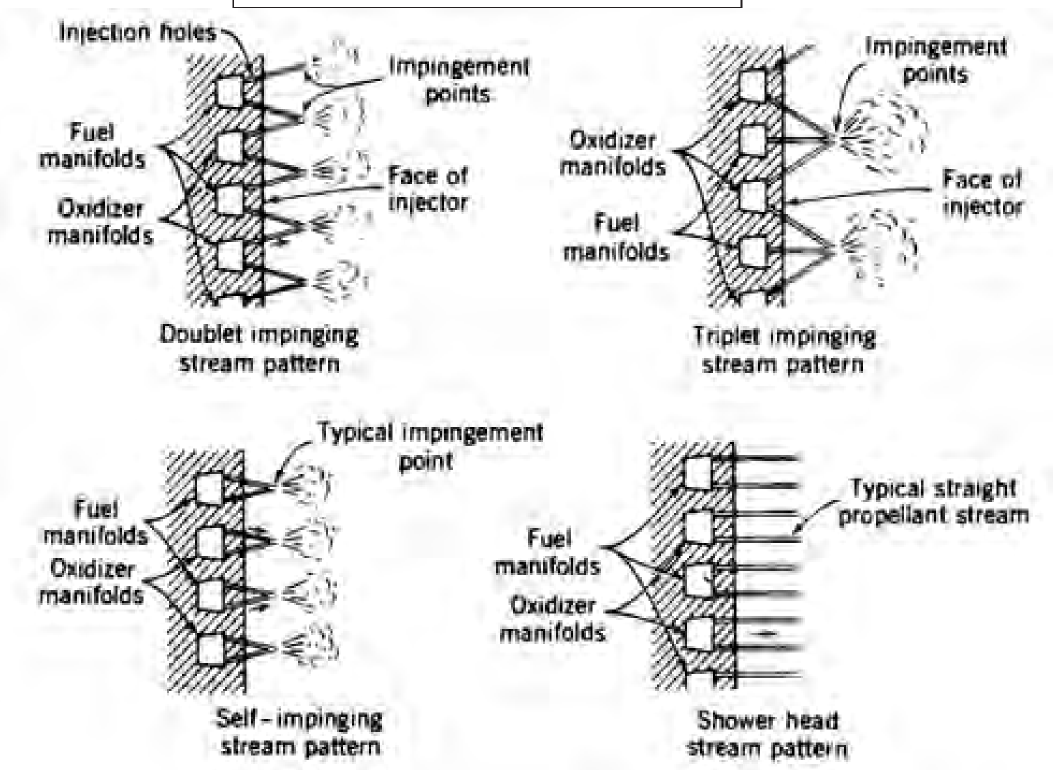
- **COMPONENTS:**
  - **INJECTION PLATE**
  - **COMBUSTION CHAMBER**
  - **NOZZLE**
  - **COOLING SYSTEM**
  - **IGNITION SYSTEM**

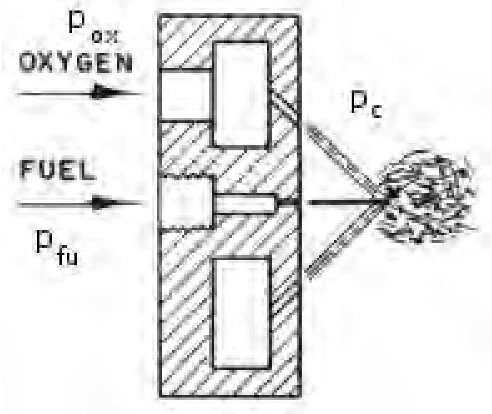


6.6 THRUST VECTOR CONTROL (TVC)



6.7 INJECTORS





$\dot{m}_i$	injector mass flow rate
ρ_p	liquid density
A_i	injector cross–section
$\Delta p_i = p_{in} - p_c$	Δp through injector
C_d	discharge coefficient

$$\dot{m}_i = C_d A_i \sqrt{2\,\rho_p\,\Delta p}$$

1. **Δp SMALL TO CONTAIN $P_{PUMPING}$**
2. **Δp LARGE FOR ATOMIZATION, AND CONTROL OF COMBUSTION INSTABILITY**
3. **D_i from 30 to 800 μm (AFFECTS *INSTABILITY*)**

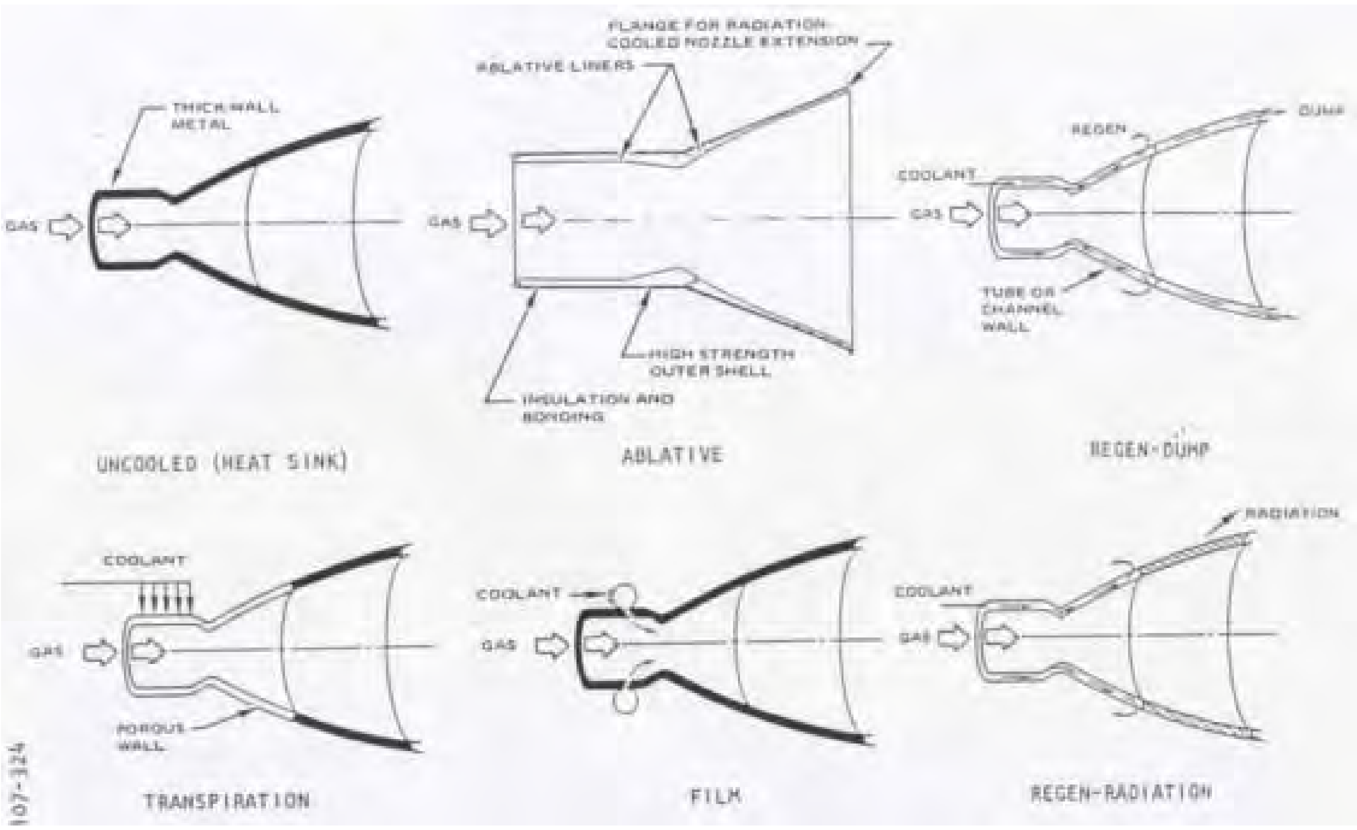
6.8 INJECTION PLATE


6.9 COOLING SYSTEM

- **ACTIVE METHODS: FORCED CONVECTION (COOLANT: FUEL)**
- **PASSIVE METHODS: RADIATION, THERMAL PROTECTIONS**

1. **REGENERATIVE (ACTIVE)**
2. **FILM (ACTIVE)**
3. **RADIATION (PASSIVE)**

CRUCIAL SECTION: THROAT



6.10 COOLING FLUID: FUEL

- **THRUST CHAMBER MADE UP OF HUNDREDS OF CHANNELS**
- **DUE TO STRONG THERMAL/MECHANICAL STRESS, A CRACK CAN OCCUR**

$\rightarrow$ **COOLANT SQUIRTS INTO CHAMBER**

- **COMBUSTION PRODUCTS ARE <u>FUEL</u>–RICH**
- **IF USING OXIDIZER AS A COOLANT, IT WILL REACT WITH EXCESS FUEL**

$\rightarrow$ **FURTHER REACTION $\rightarrow$ HEAT RELEASE $\rightarrow$ CRACK WIDENS**

- **FURTHER, WALL MATERIAL (Cu) CAN BURN WITH OXYGEN**

$\rightarrow$ **USE <u>FUEL</u> AS A COOLANT**

6.11 FEED SYSTEM

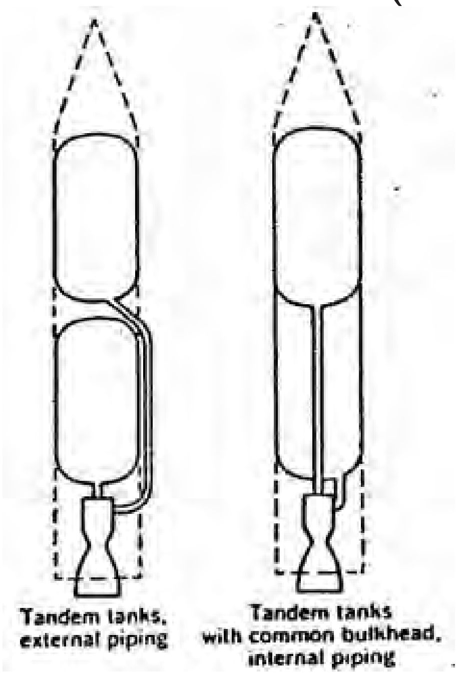
- **CRUCIAL COMPONENT**

1. **PRESSURE–GAS:**
   - **ONLY APPLICABLE FOR LOW p_c (LARGE m_{tank}); OK FOR MULTIPLE IGNITIONS**
2. **TURBOPUMPS:**
   - **LIGHT; COMPLEX AND COSTLY, LIMITED no. IGNITIONS**
3. **(ELECTRIC PUMPS):**
   - **FED BY CELLS; RECENTLY PROPOSED FOR UPPER STAGES AND SPACECRAFTS, MULTIPLE IGNITIONS**

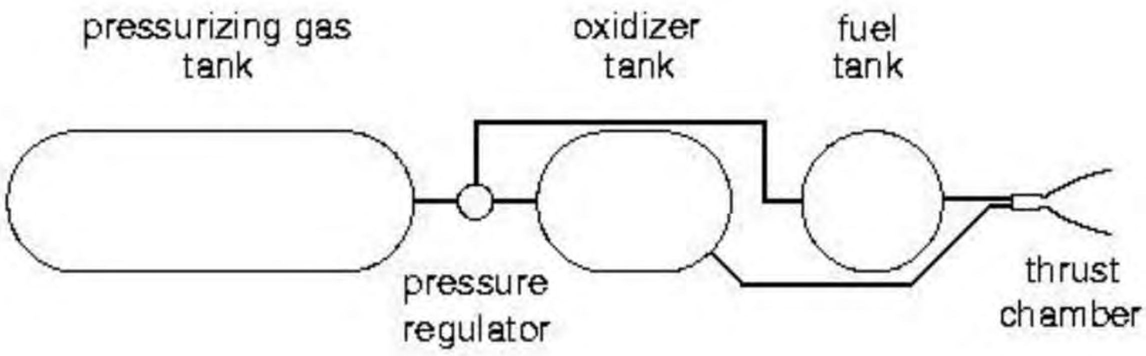
6.12 TANKS

- **HIGH PRESSURE (20–40 MPa): PRESSURANT TANKS, USUALLY SPHERICAL**
- **MEAN PRESSURE (2–6 MPa): PROPELLANT TANKS IN PRESSURE–GAS SYST.**
- **LOW PRESSURE (0,1–0,6 MPa) PROPELLANT TANKS IN TURBOPUMP SYST.**

- **THERMAL INSULATION (CRYOGENIC PROPEL.)**

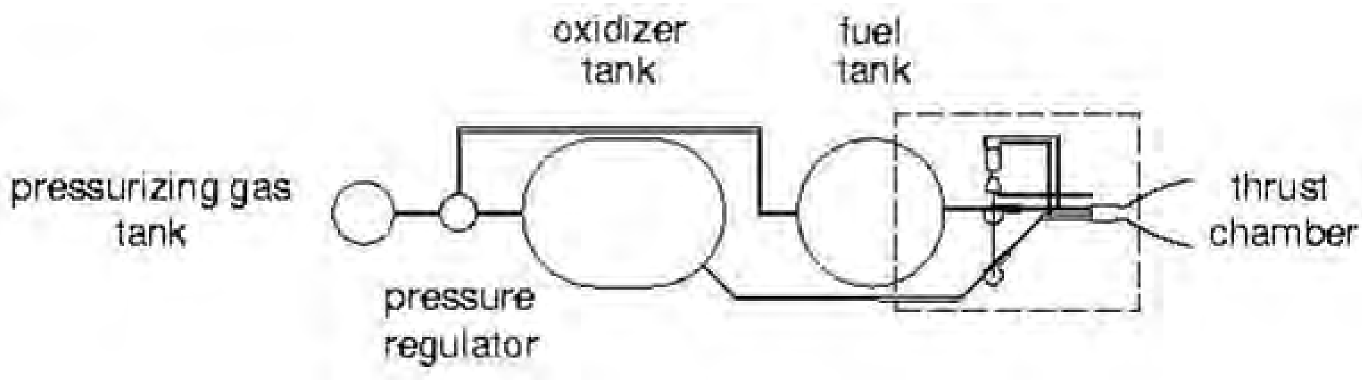


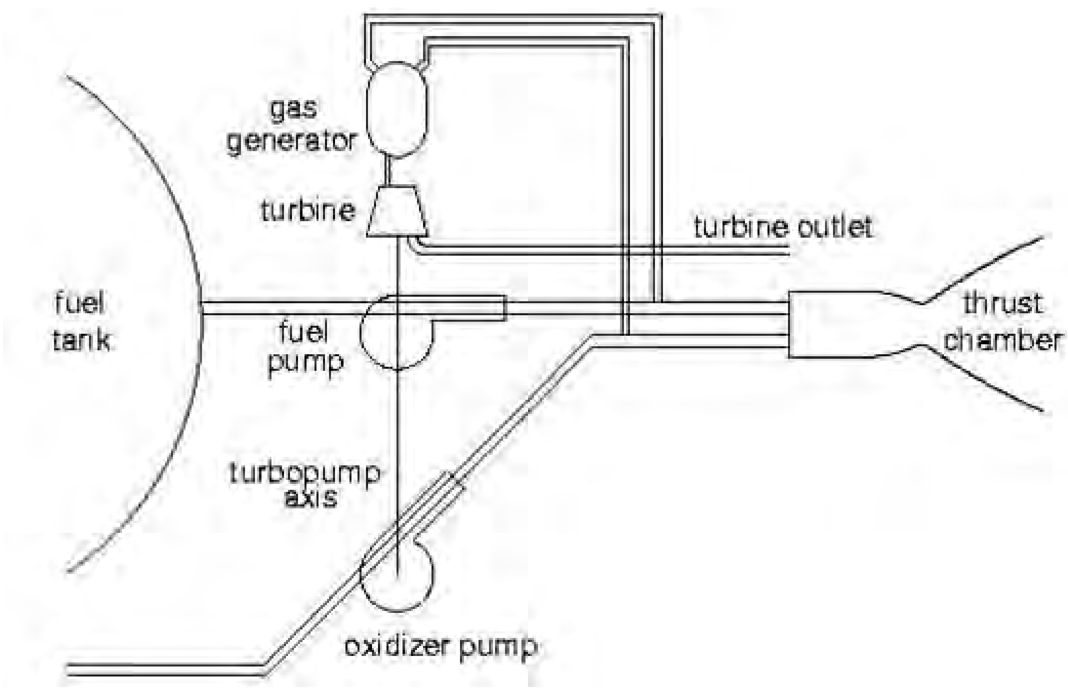
6.13 PRESSURE–GAS FEED SYSTEM



- **TANKS *UNDER PRESSURE* (HEAVY)**
- **PRESSURANT:**
  - **AIR: CHEAP, BUT MAY REACT WITH FUEL**
  - **NITROGEN (N_2): INERT, BUT SOLUBLE IN O_2, N_2O_4 (NTO)**
  - **HELIUM (He): INERT, LIGHT, COSTLY**

6.14 TURBOPUMP FEED SYSTEM



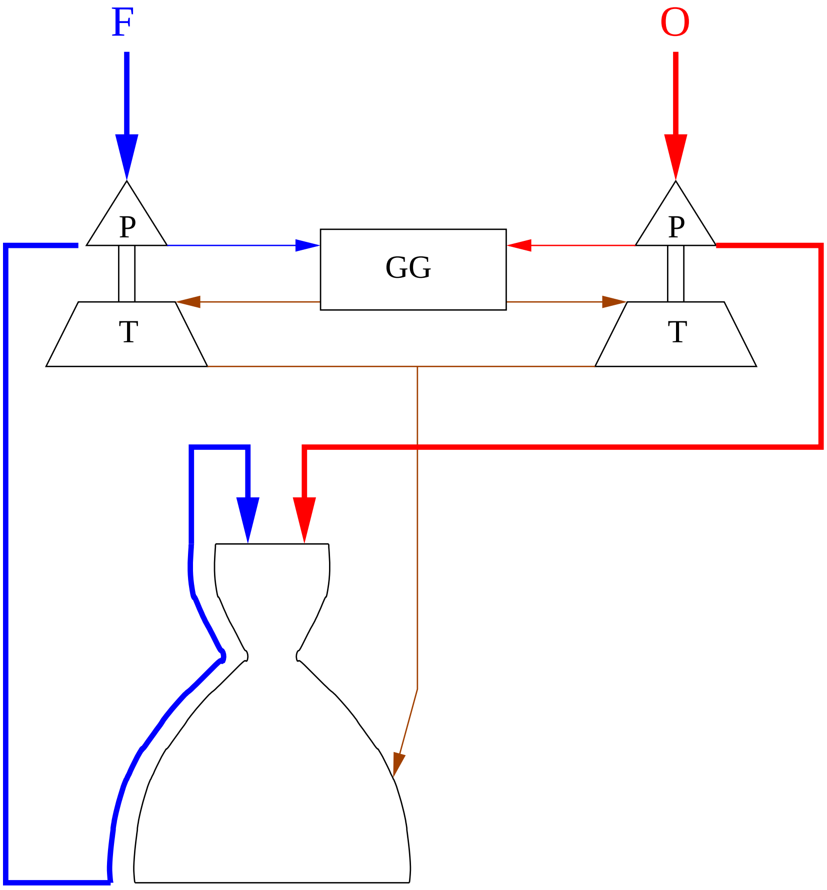


6.15 TURBOPUMP FEED SYSTEM: OPTIONS

1. **OPEN CYCLE:**
   TURBINE GASES EXHAUSTED BY SEPARATE NOZZLE, OR RE–INJECTED INTO DIVERGENT

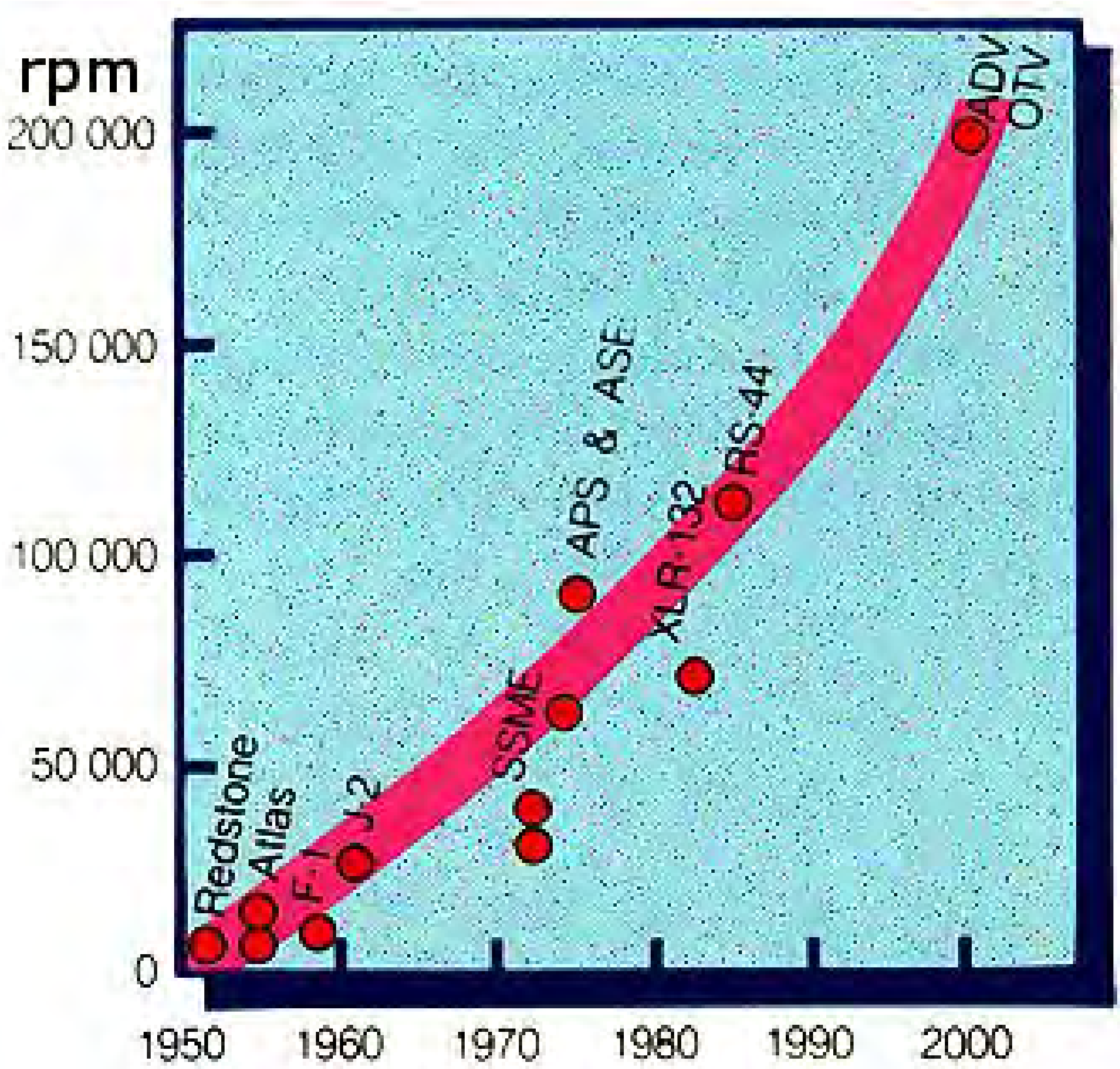
2. **CLOSED CYCLE:**
   TURBINE GASES RE–INJECTED INTO COMBUSTION CHAMBER

6.16 GAS–GENERATOR CYCLE (OPEN)



- **TURBINE TEMPERATURE $\simeq 800 \div 900$ K (O/F VERY FAR FROM STOICHIOMETRIC)**
- **GAS–GENERATOR MASS FLOW 1–5% OF $\dot{m}$**
- **ALTERNATIVE: COMBUSTION GASES SPILL**

6.17 TURBOPUMP ROTATIONAL SPEED



6.18 COMBUSTION INSTABILITY (COMBUSTION PRESSURE OSCILLATIONS

a. LONGITUDINAL MODES

b. TANGENTIAL MODES (*STANDING* or *SPINNING*

c. RADIAL MODES

- **MAY LEAD TO CHAMBER FAILURE**

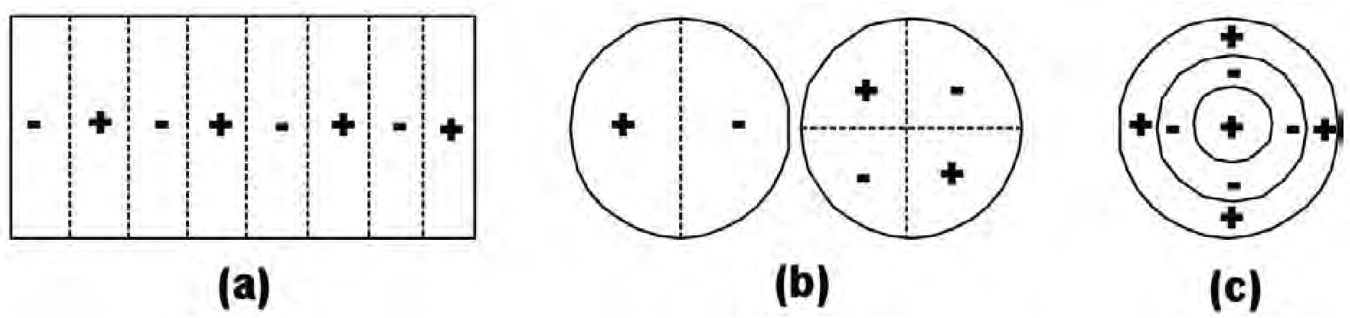


Figure 1: 8th order 1st/2nd order 2nd order

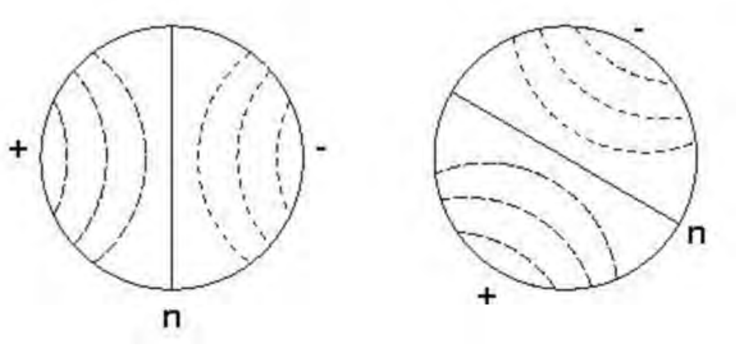
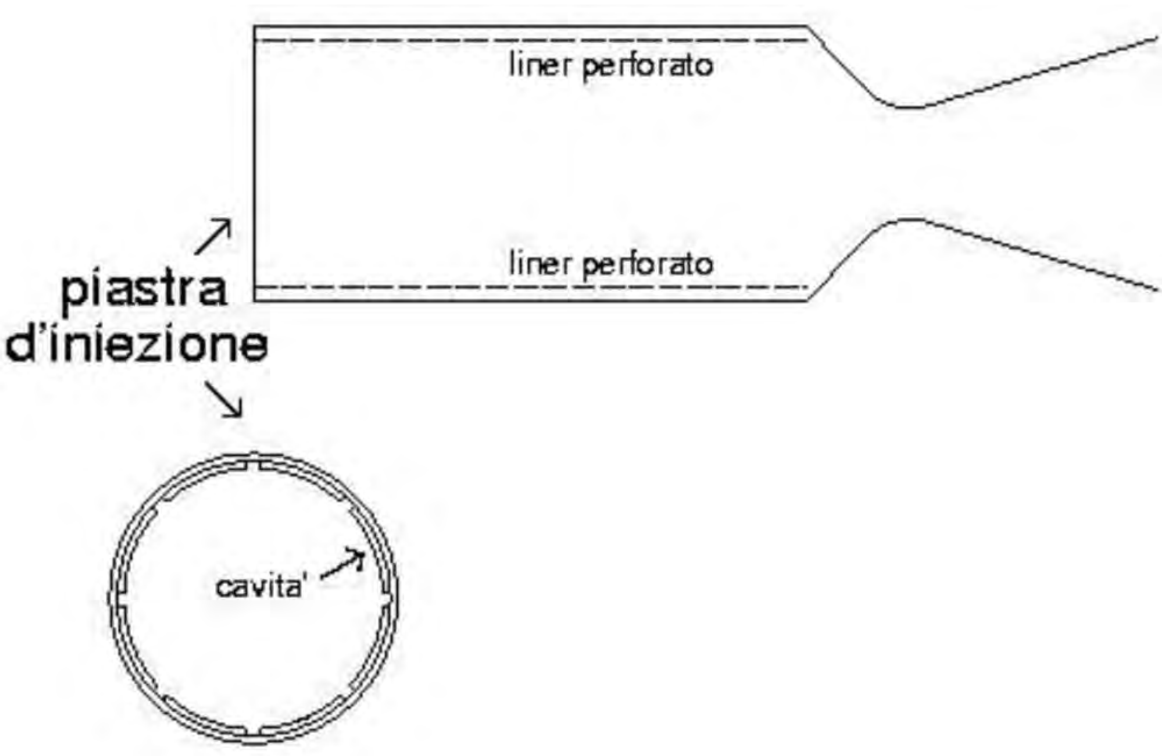


Figure 2: Spinning tangential mode, 1st order.

6.19 EXPEDIENTS TO COUNTER COMBUSTION INSTABILITY IN *LRE*

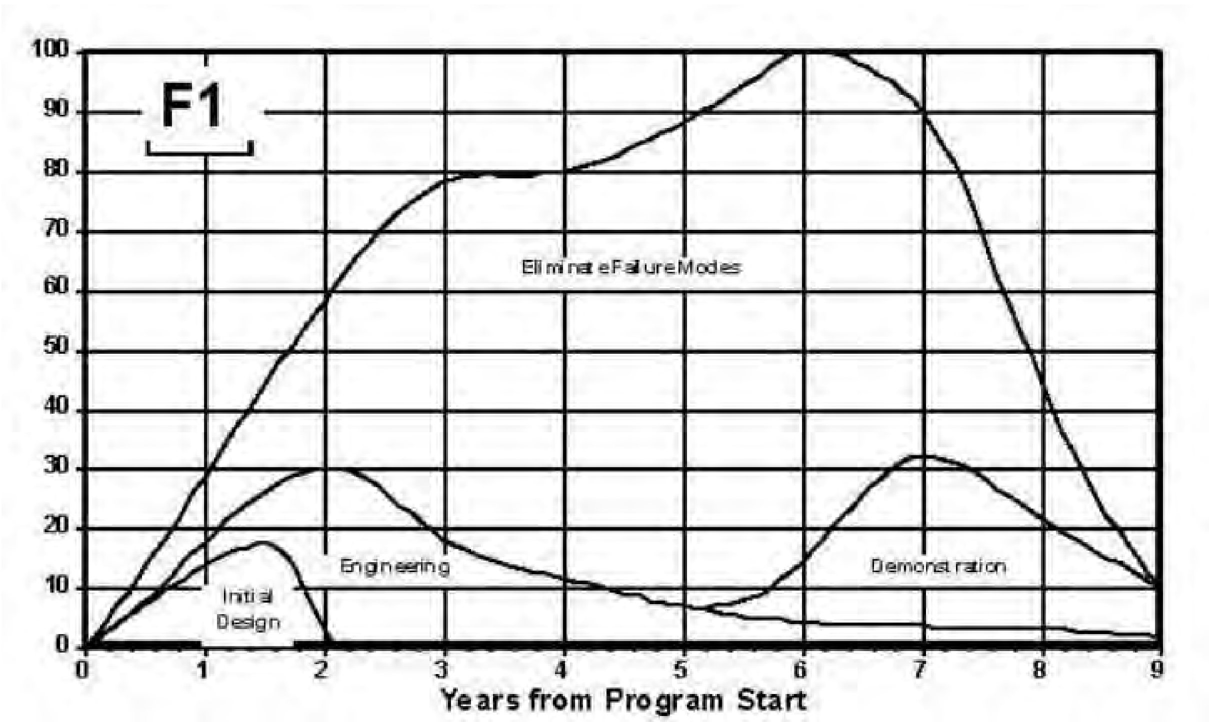
- **EMPIRICAL:**
- **HIGH INJECTOR Δp**
- **CAVITIES IN INJECTOR PLATE**
- **LINERS PERFORATED LINERS**
- **BAFFLES**



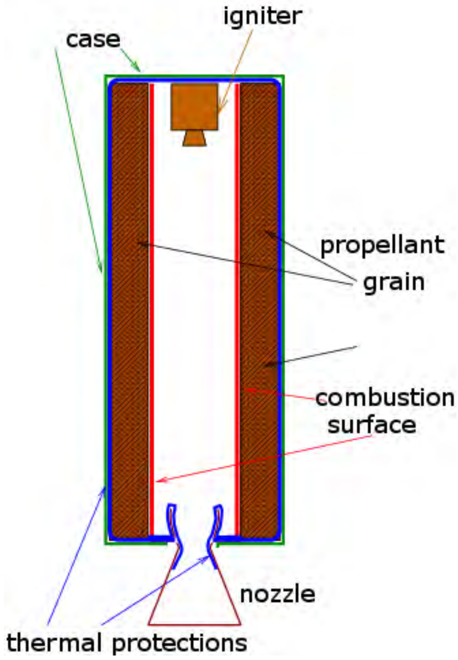
- **INSTABILITY ENHANCED BY HIGH p_c**
- **CRITICAL ISSUE: DISTANCE INJECTION PLATE–FLAME FRONT**

6.20 COMPONENTS OF DEVELOPMENT COST OF A (VERY LARGE) LRE

- F–1 (1st STAGE SATURN V)
- IMPORTANT COMPONENT: SUPPRESSION COMBUSTION INSTABILITY



7.1 SOLID PROPELLANT MOTORS (SRM)



- **PROS/CONS:**
  - \+ **SIMPLE**
  - \+ **STORABLE**
  - \+ **EASILY SCALABLE DESIGN**
  - – **LOWER PERFORMANCE (u_{eff})**
  - – **NON RE–IGNITABLE, NON THROTTLEABLE**
- **MAIN COMPONENTS:**
  - ◦ **COMBUSTION CHAMBER**
  - ◦ **PROPELLANT GRAIN**
  - ◦ **IGNITER**
  - ◦ **NOZZLE**
- **USES: BOOSTERS, UPPER STAGES, TACTICAL MISSILES, GAS GENERATORS**

7.2 SOLID PROPELLANTS

- **DOUBLE BASE**
- **COMPOSITES**

7.3 DOUBLE BASE PROPELLANTS

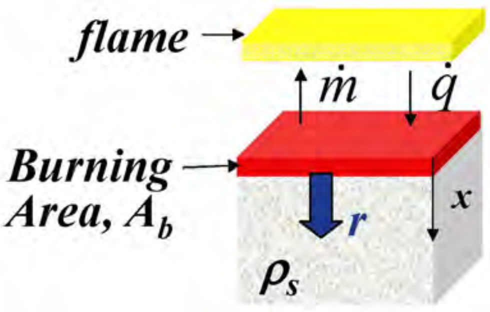
- **NITROGLYICERIN ABSORBED ON NITROCELLULOSE**
- **SMOKELESS**
- **LOW COST, NON–TOXIC PRODUCTS**
- **LOW PERFORMANCE ($u_{eff} \simeq 2200$ m/s, s/l)**
- **LOW DENSITY ($\rho_p \simeq 1600$ kg/m^3)**
- **HIGHLY DANGEROUS**

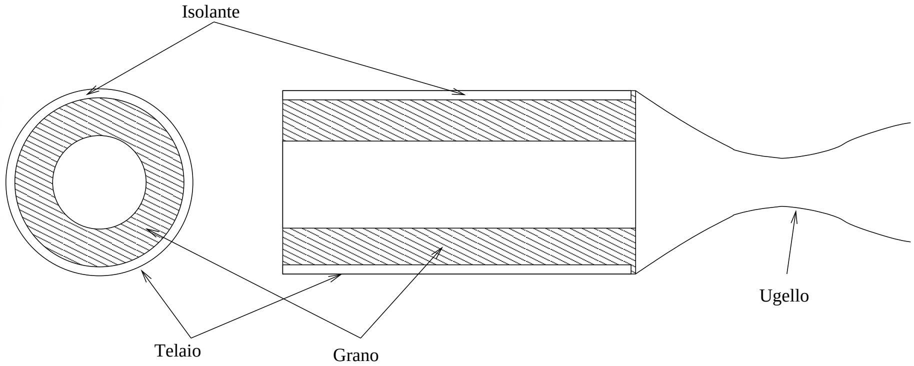
→ MILITARY APPLICATIONS

7.4 COMPOSITE PROPELLANTS

- **OXIDIZERS:**
  - **AMMONIUM PERCHLORATE NH_4ClO_4 (AP)**
  - **AMMONIUM NITRATE NH_4NO_3 (AN)**
- **FUELS:**
  - **BINDER: ASPHALT OR RUBBER**
    - **POLIBUTADIENE (HTPB, PBAN, ...), ASPHALTS, ...**
  - **METAL POWDER ($\leq$ 20%):**
    - **Al, Be (CARCINOGEN)**

7.5 GRAIN REGRESSION VELOCITY





- **DEPENDING UPON:**
  1. **GRAIN COMPOSITION**
  2. **CHAMBER PRESSURE** $r = a\, p_c^n$
  3. **GRAIN INITIAL TEMPERATURE**
  4. **GAS VELOCITY (|| GRAIN)**

7.6 EQUILIBRIUM CHAMBER PRESSURE

NOZZLE FLOW RATE: $\dot{m}_{out} = \Gamma \dfrac{p_c\,A_t}{\sqrt{R\,T_c}}$

FLOW RATE COMBUST. PRODUCTS: $\dot{m}_{in} = r\,\rho_p\,A$

AT EQUILIBRIUM: $\dot{m}_{in} = \dot{m}_{out} \quad \Rightarrow$

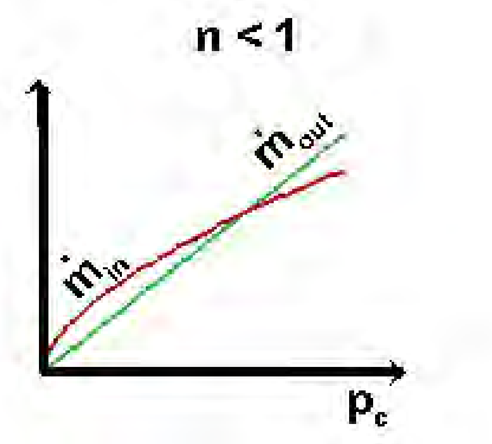
$$p_c = \left(a\,c^*\,\rho_p\,K\right)^{\frac{1}{1-n}}$$

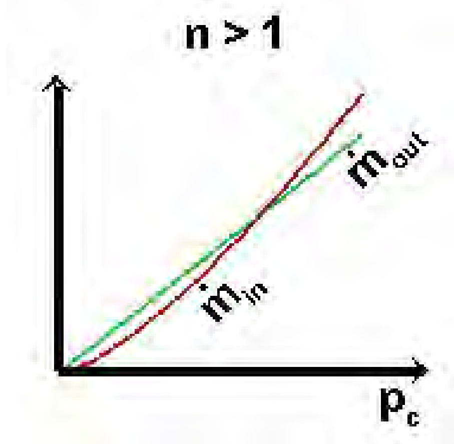
- $K = A_b/A_t$, **'KLEMMUNG'**
  (RATIO BURNING AREA/THROAT AREA)

7.7 OPERATING POINT STABILITY

NOZZLE FLOW RATE: $\dot{m}_{out} \propto p_c$

COMBUSTION PRODUCTS FLOW RATE: $\dot{m}_{in} \propto p_c^n$





$\Delta p_c > 0: \quad \Delta \dot{m}_{out} > \dot{m}_{in} \qquad \Delta \dot{m}_{out} < \dot{m}_{in}$

$\Delta p_c < 0: \quad \Delta \dot{m}_{out} < \dot{m}_{in} \qquad \Delta \dot{m}_{out} > \dot{m}_{in}$

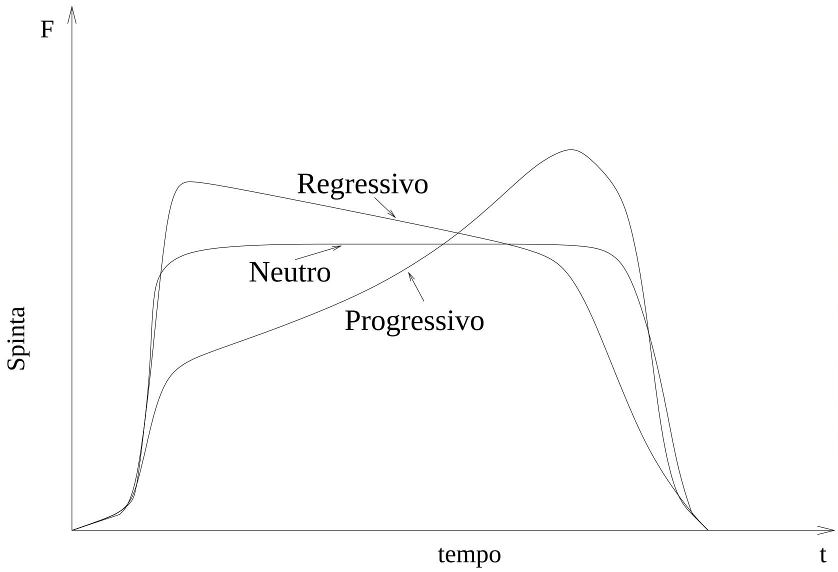
$n < 1$ **STABLE,** $\qquad n > 1$ **UNSTABLE**

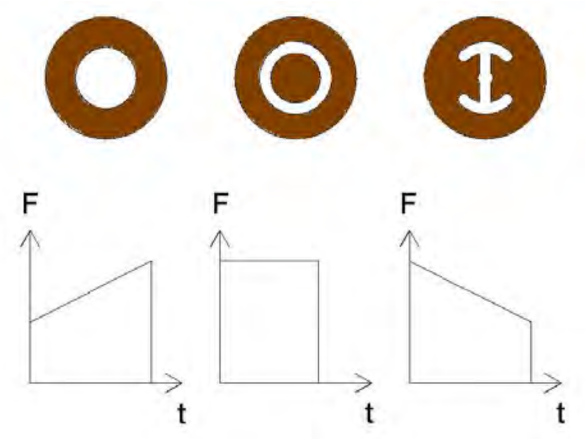
7.8 GRAIN GEOMETRY (1)

- **CAST OR EXTRUDED**

$$p_c = \left(a\, c^*\, \rho_p\, K \right)^{\frac{1}{1-n}}$$

$$F(t) \; = \; F(0) \; \left[\frac{A_b(t)}{A_b(0)} \right]^{\frac{1}{1-n}}$$

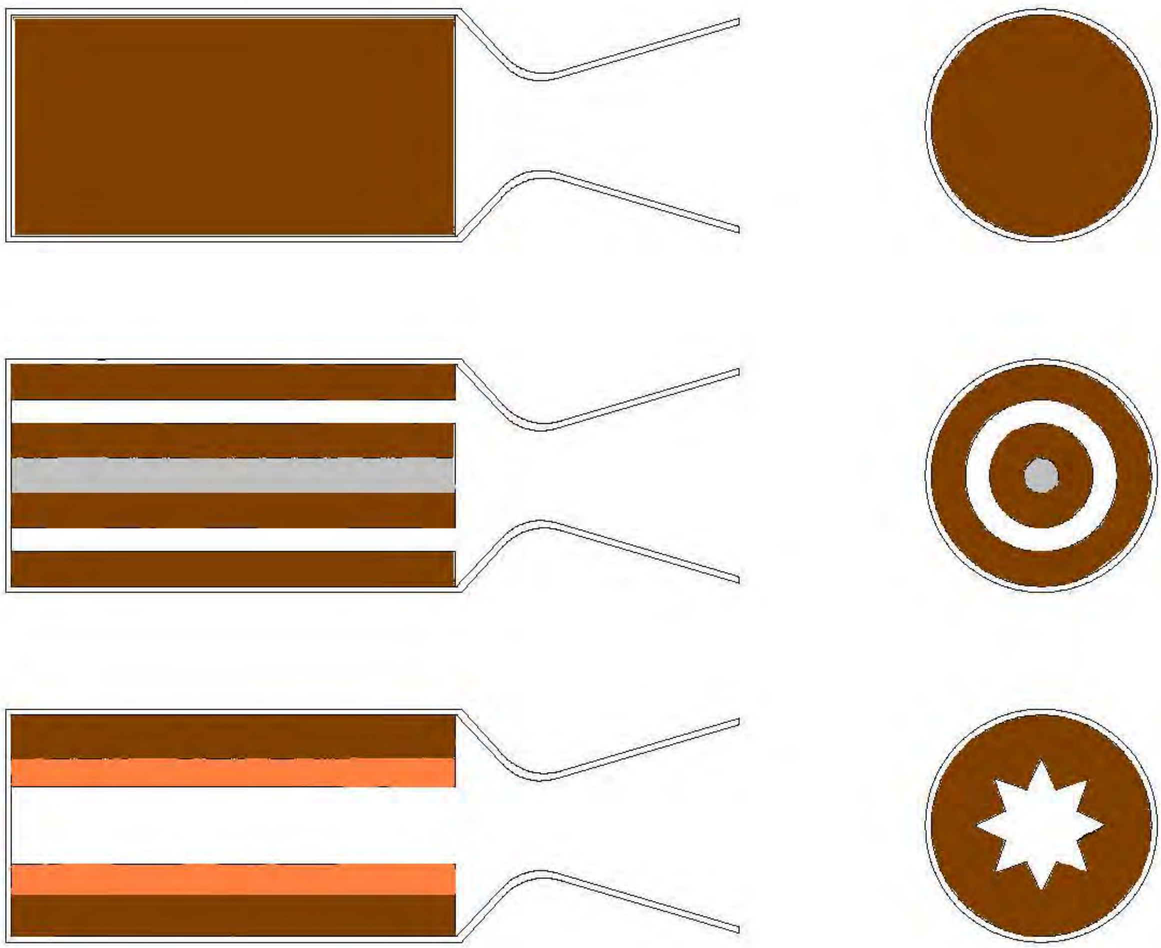




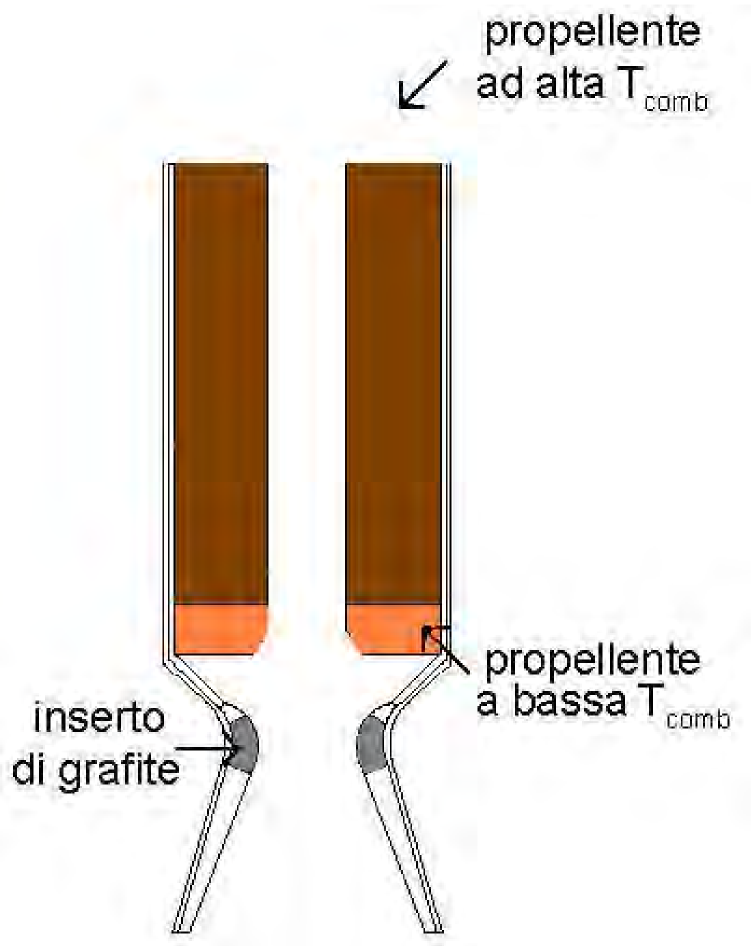
- **PROGRESSIVE** $dA_b/dt > 0$, F **INCREASING**
- **NEUTRAL** $dA_b/dt = 0$, F **CONSTANT**
- **REGRESSIVE** $dA_b/dt < 0$, F **DECREASING**

7.9 GRAIN GEOMETRY (2)

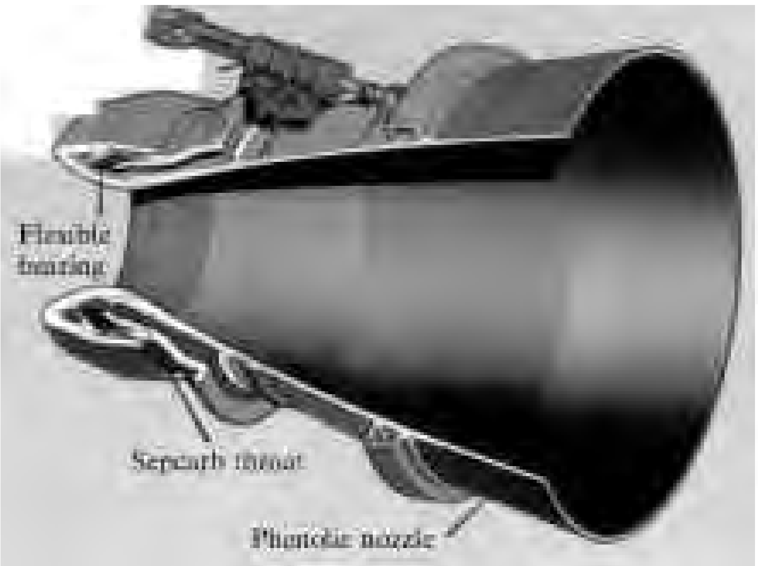
- **TWO–DIMENSIONAL GRAINS**
  - **SIGARETTE BURNING (LOW K)**
  - **RADIAL BURNING (TUBULAR, STAR)**

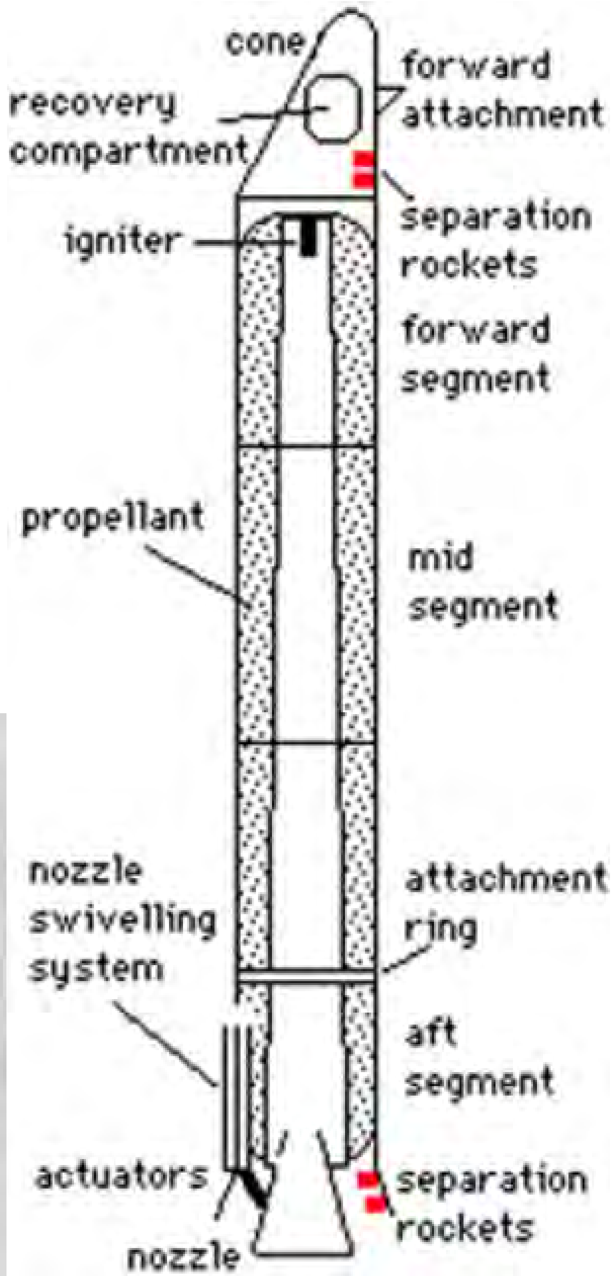


7.10 THERMAL PROTECTIONS



7.11 *CONICAL* NOZZLES

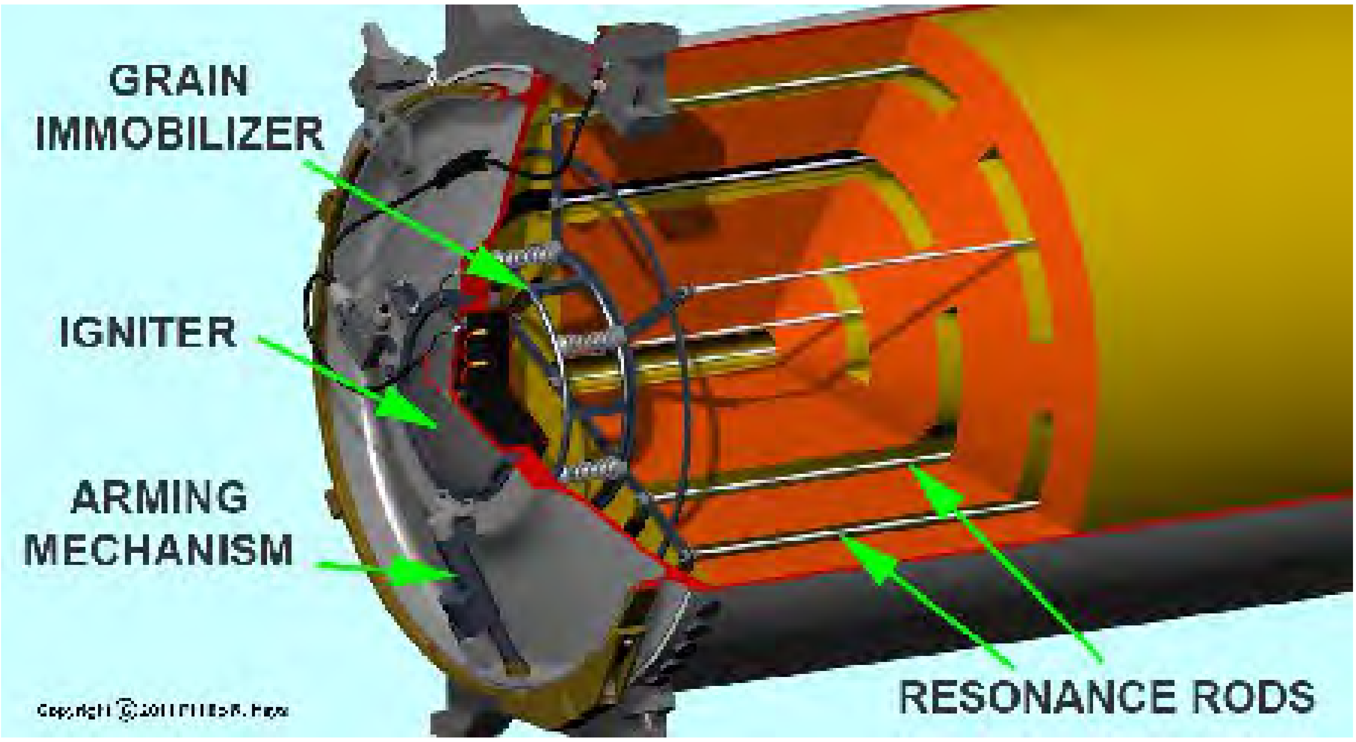


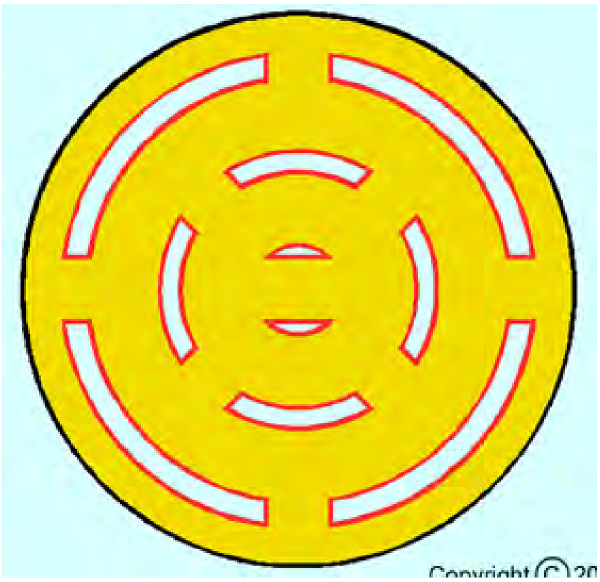


7.12 COMBUSTION INSTABILITY IN SRMs

- **LESS SERIOUS THAN IN LREs, AS:**
  - **VISCOELASTIC GRAIN;**
  - **VISCOUS COUPLING WITH SOLID PARTICLES (particle diameter must be tailored to size)**
  - **CHAMBER DIMENSION VARIES → NATURAL f's NOT CONSTANT**
  - **NO FEED SYSTEM**

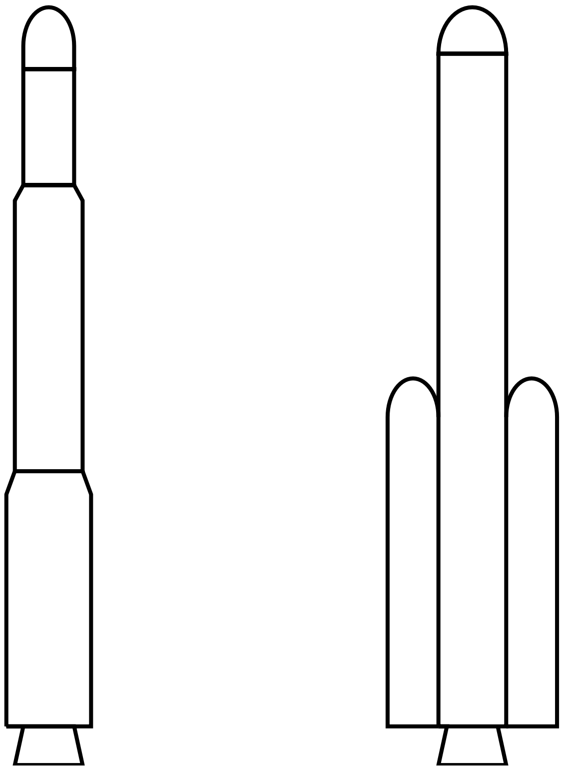
7.13 RESONANCE RODS





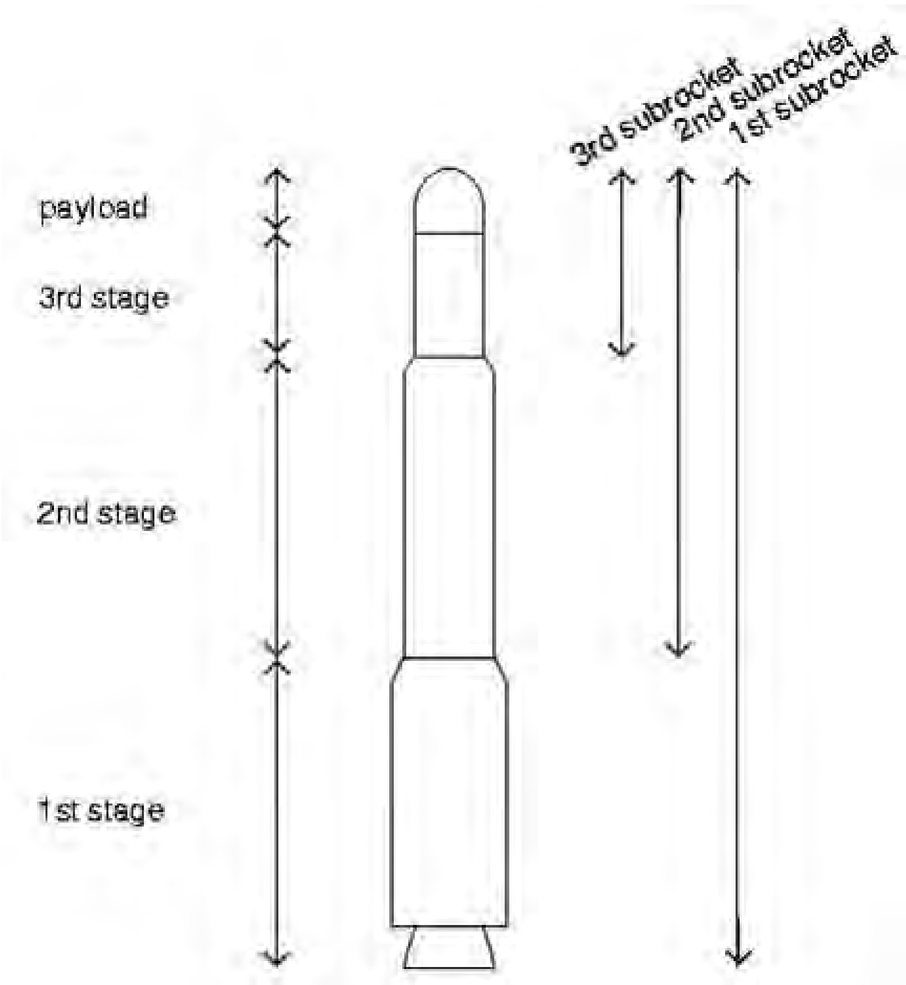
8.1 STAGING

- **TANDEM** **PARALLEL**



8.2 TANDEM STAGING

- ***STAGES*** **AND** ***SUBROCKETS***
- $MR_i = m_{fi}/m_{0i}$; $\lambda_i = m_{pl,i}/m_{0i}$
- MR_i, λ_i **RELATIVE TO** ***SUBROCKETS***
- u_{eff} **and** $\kappa_{s,i}$ **RELATIVE TO** ***STAGES***
- **ALSO CONTAINS FINAL ACCELERATION**



8.3 PAYLOAD RATIO λ_i INDIVIDUAL SUBROCKET

$$\boxed{\lambda_i = 1 - \frac{1 - MR_i}{1 - \kappa_{si}} = 1 - \frac{1 - \exp(-\Delta v_i / u_{eff,i})}{1 - \kappa_{si}}} \tag{30}$$

VELOCITY INCREMENT OF INDIVIDUAL SUBROCKET

$$MR_i = \lambda_i (1 - \kappa_{si}) + \kappa_{si} \tag{31}$$

$$\Delta v_i = -u_{eq,i} \log [\lambda_i (1 - \kappa_{si}) + \kappa_{si}] \tag{32}$$

8.4 VELOCITY INCREMENT AND PAYLOAD RATIO OF THE WHOLE LAUNCHER

$$\boxed{\Delta v = \sum_{i=1}^{N} \Delta v_i = -\sum_{i=1}^{N} u_{eff,i} \log \left[\lambda_i \left(1 - \kappa_{si}\right) + \kappa_{si}\right]} \tag{33}$$

$$\boxed{\lambda = \prod_{i=1}^{N} \lambda_i} \tag{34}$$

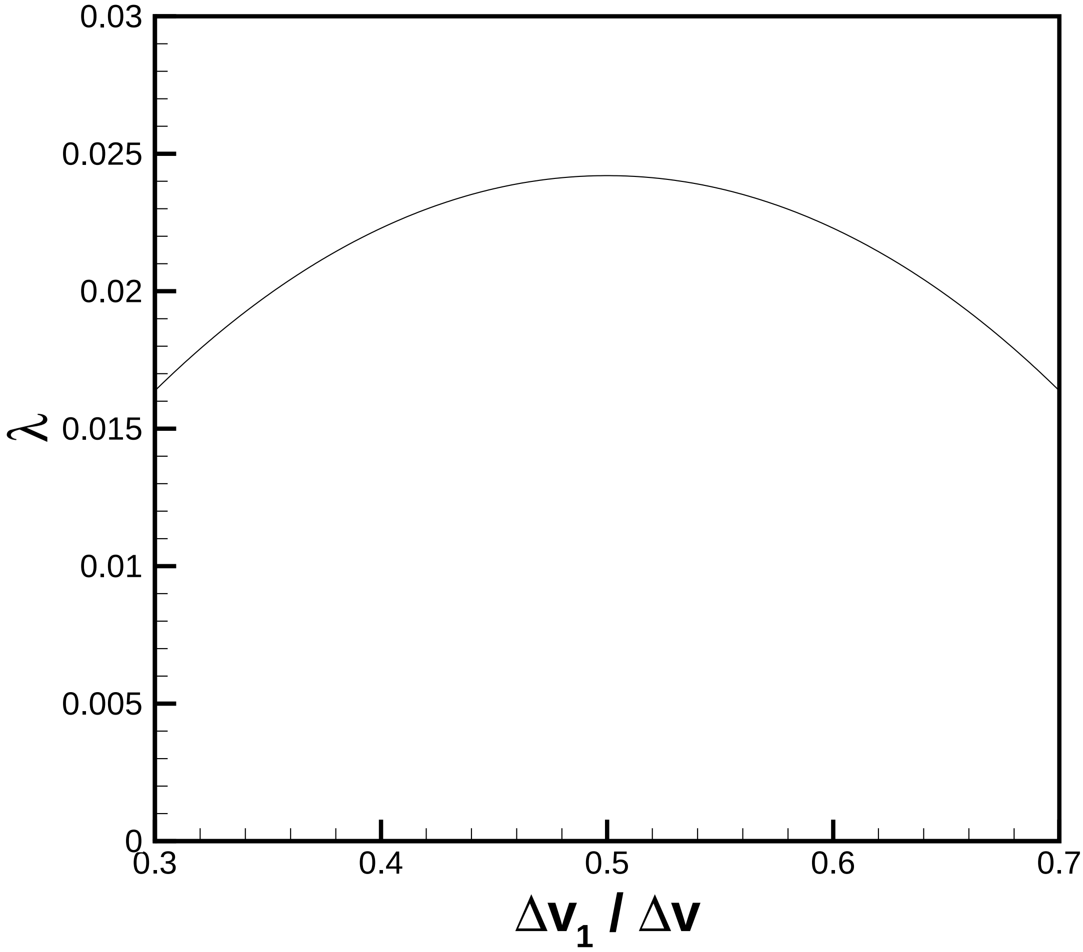
$$\lambda = \frac{m_{pl}}{m_0} = \frac{m_{pl,3}}{m_{01}} = \frac{m_{pl,1}}{m_{01}} \frac{m_{pl,2}}{m_{02}} \frac{m_{pl,3}}{m_{03}} \tag{35}$$

8.5 PARTITIONING THE VELOCITY INCREMENT AMONG THE N SUBROCKETS

$$\boxed{\Delta v_i = \frac{\Delta v}{N}} \tag{36}$$

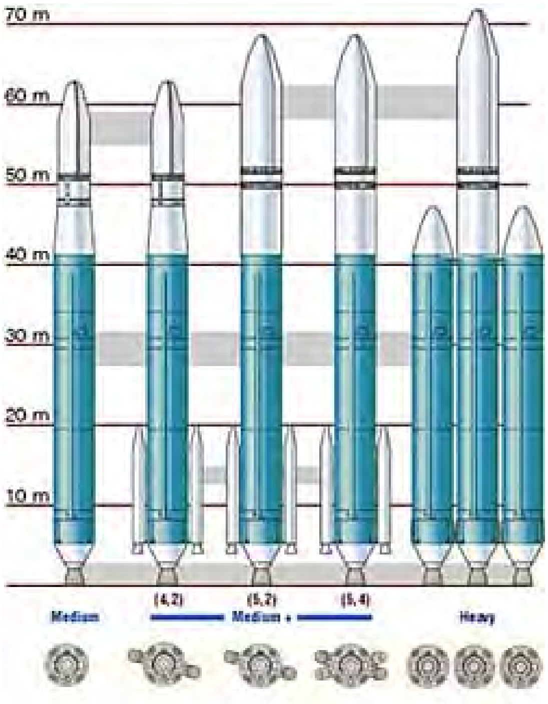
- **STRICTLY VALID ONLY FOR $u_{eff,i}$ and $\kappa_{s,i}$ EQUAL FOR ALL STAGES**

8.6 PAYLOAD RATIO AS A FUNCTION OF Δv PARTITIONING



8.7 LAUNCHER ‘FAMILIES’

- **NON–OPTIMAL Δv PARTITIONING**
- **BUT LOWER DEVELOPMENT COSTS**



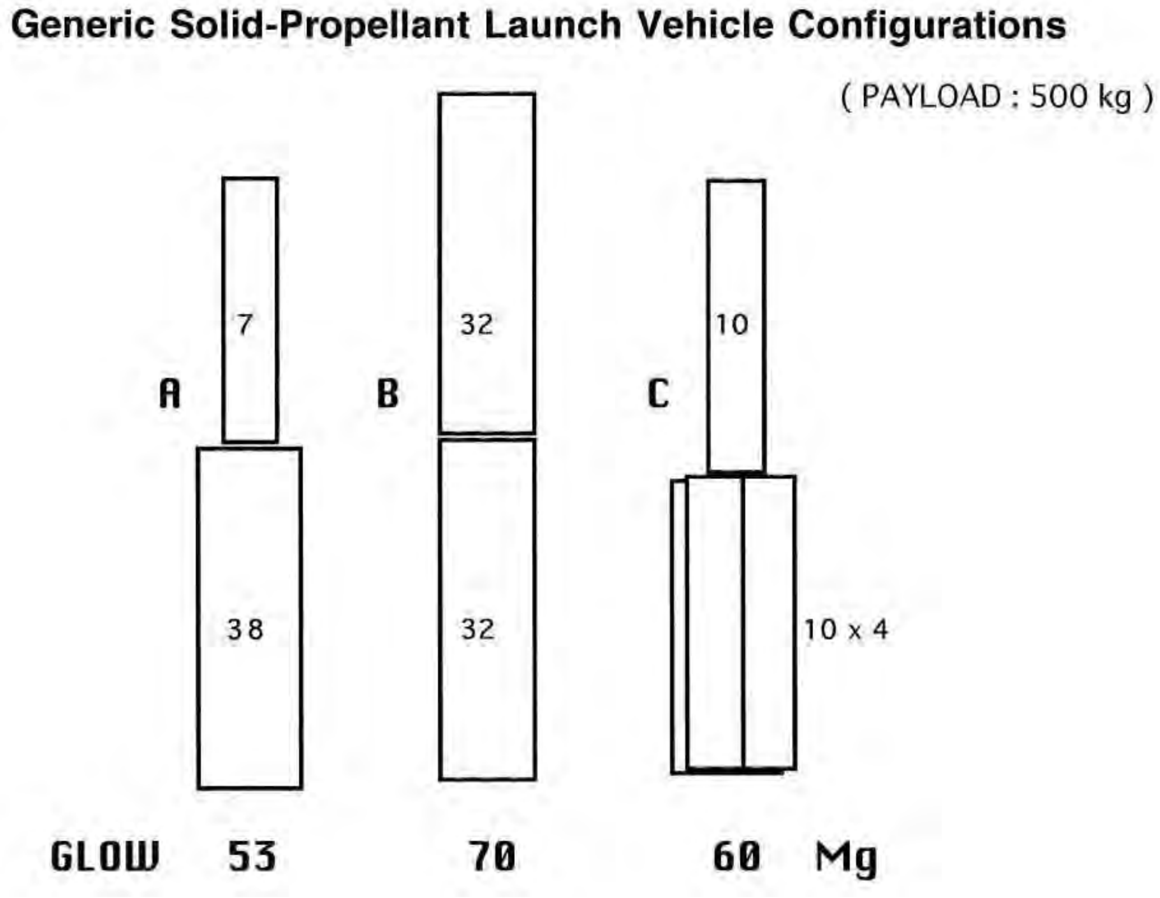
8.8 TRADE–OFF BETWEEN DEVELOPMENT AND PRODUCTION COSTS

A. MAX DEVELOPMENT COSTS, MIN. PRODUCTION COSTS

B. MIN. DEVELOPMENT COSTS, MAX PRODUCTION COSTS

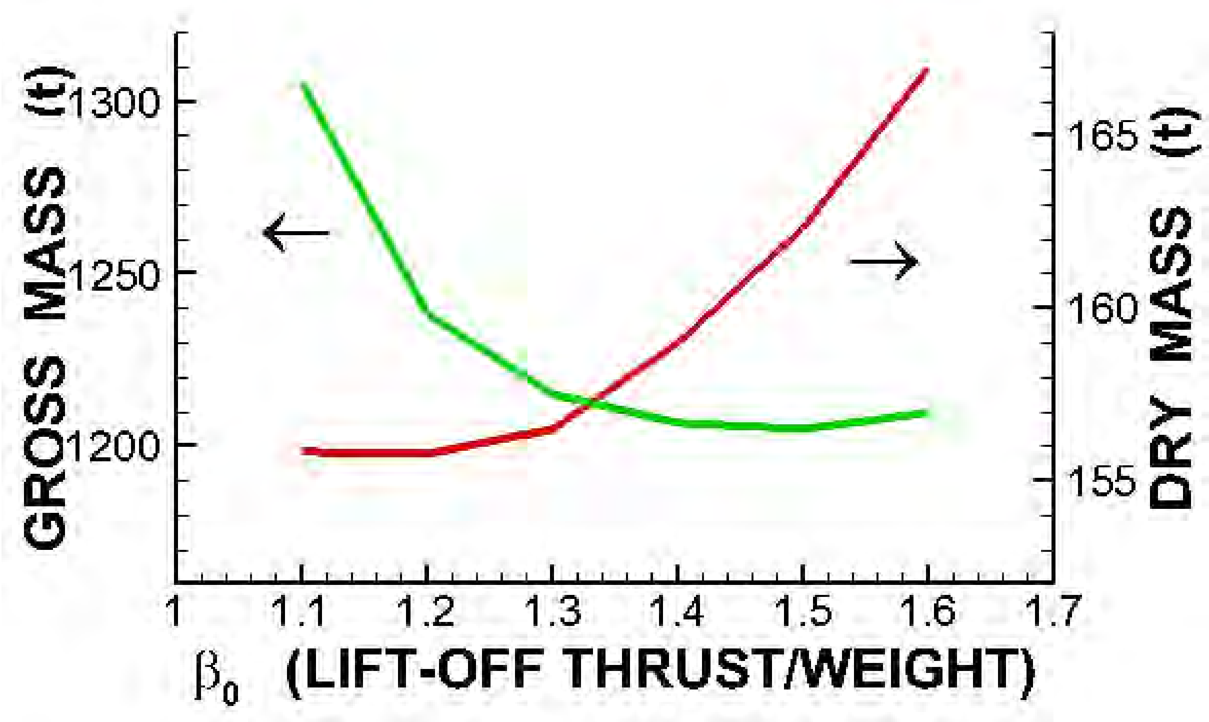
C. INTERMEDIATE DEVELOPMENT AND PRODUCTION COS

- **CHOICE DEPENDS OF no. UNITS TO BE BUILT**

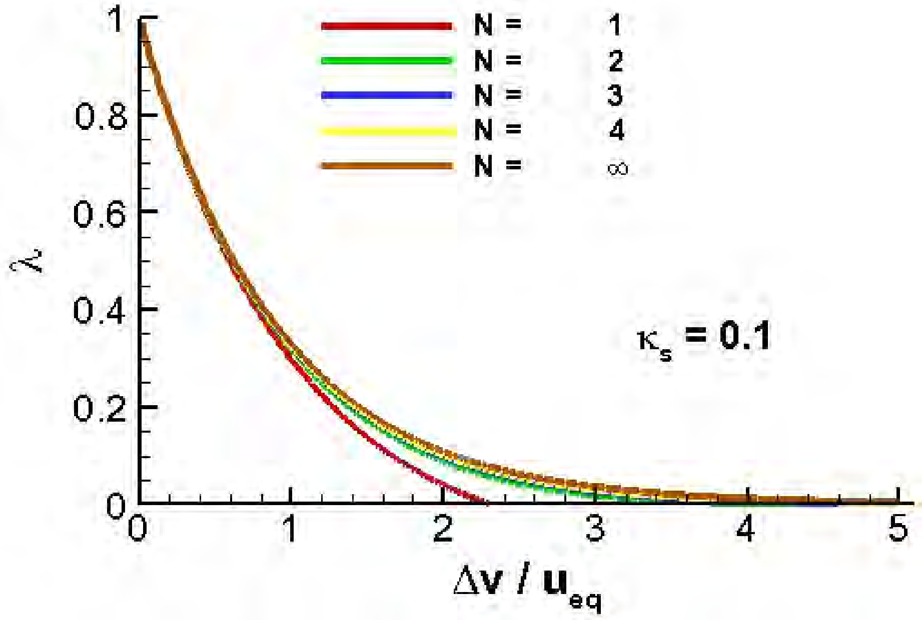


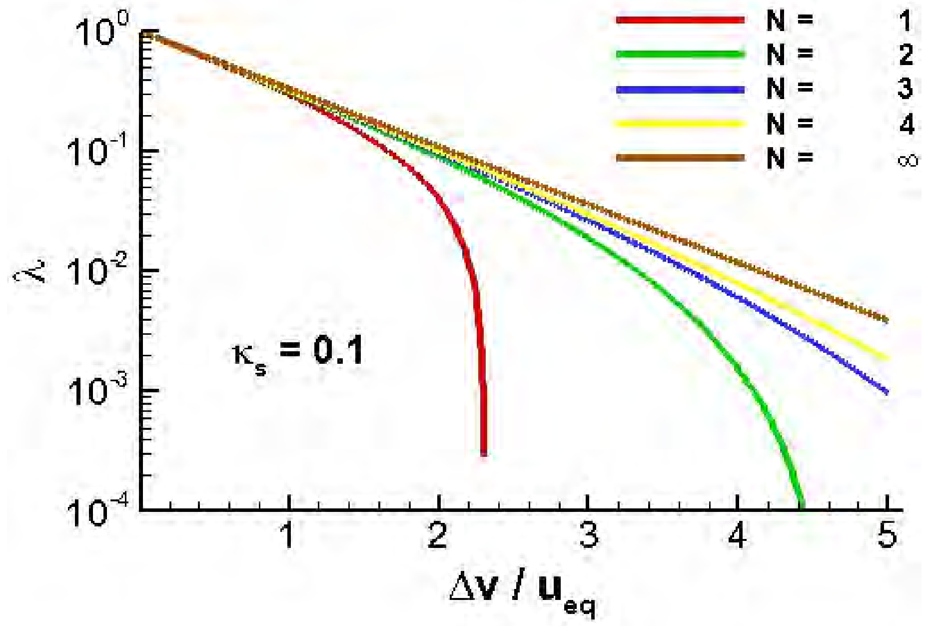
8.9 MINIMIZING LAUNCHER COST

- **HIGHER INITIAL ACCELERATION → LOWER GRAVITATIONAL LOSSES**
- **BUT ENGINE MASS INCREASES (VERY COSTLY)**



8.10 PAYLOAD RATIO λ FOR AN N–STAGE LAUNCHER





- **N LIMITED BY COSTS, RELIABILITY**
- **N HIGHER FOR HIGH–Δv MISSIONS; HIGH u_{eff} ALSO INDICATED**

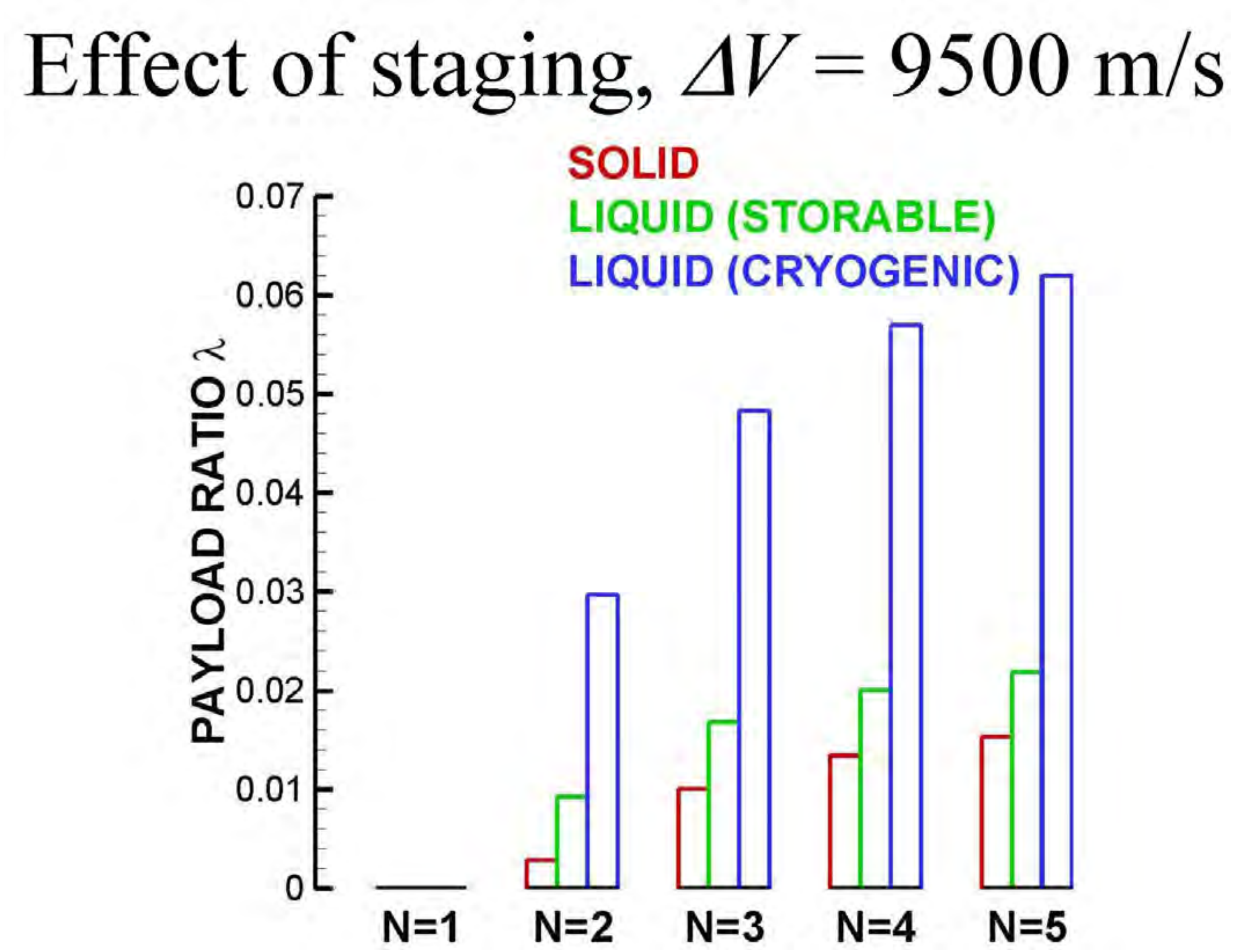
8.11 HEAT SHIELD (or FAIRING) JETTISON

- **PROTECTING PAYLOAD FROM THERMAL AND DYNAMIC STRESSES**
- **RELEASED AROUND 80 – 100 km ALTITUDE**
- **FURTHER, JETTISON OF INTERSTAGES**

8.12 PROPELLANT CONTINGENCIES

- **LIQUID PROPELLANTS:**
  - *** UP TO A 3–4% TRAPPED IN FEED LINES**
  - *** 1% UNUSABLE (NON–NOMINAL O/F RATIOS)**
  - *** 0.8% RESERVE**
- **SOLID PROPELLANTS:**
  - *** 1% UNBURNT RESIDUAL (SLIVER)**
  - *** 0.8% RESERVE**

8.13 λ FOR LEO MISSIONS UNDER DIFFERENT PROPULSIVE ASSUMPTIONS



- **(λ QUITE OPTIMISTIC DUE TO NEGLECTED EFFECTS)**

8.14 λ FOR GEO MISSIONS UNDER DIFFERENT PROPULSIVE ASSUMPTION