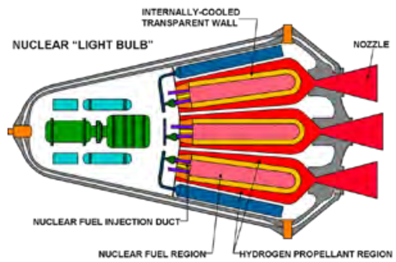
Chemical/Nuclear Propulsion

Space System Design, MAE 342, Princeton University
Robert Stengel

- Thermal rockets
- Performance parameters
- Propellants and propellant storage



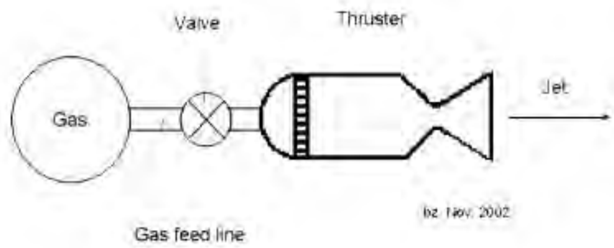
http://www.princeton.edu/~stengel/MAE342.html

Chemical (Thermal) Rockets

- Liquid/Gas Propellant
  - –Monopropellant
    - Catalytic ignition / chemical decomposition
    - Cold gas
  - –Bipropellant
    - Separate oxidizer and fuel
    - Hypergolic (spontaneous) ignition
    - External ignition
    - Storage
      - – Ambient temperature and pressure
      - – Cryogenic
      - – Pressurized tank
  - –Throttlable
  - –Start/stop cycling
- Solid Propellant
  - –Mixed oxidizer and fuel
  - –External ignition
  - –Burn to completion
- Hybrid Propellant
  - –Liquid oxidizer, solid fuel
  - –Throttlable
  - –Start/stop cycling

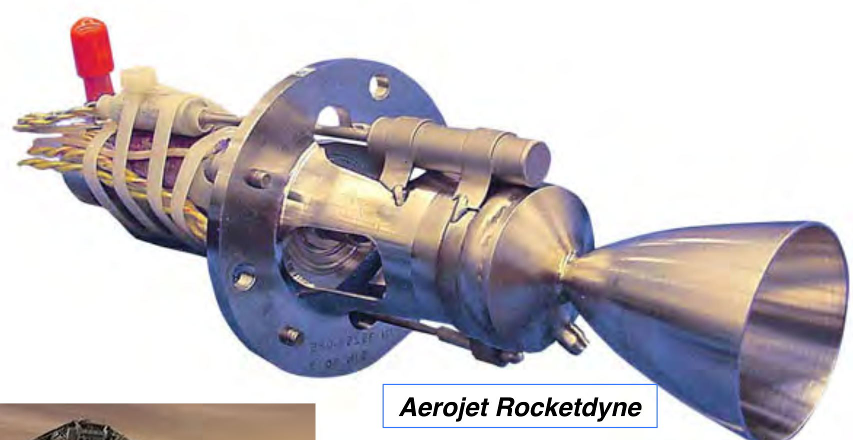
Cold Gas Thruster

(used with inert gas)

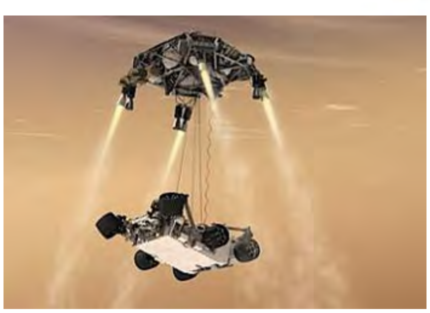


Moog Divert/Attitude Thruster
and Valve

Monopropellant Hydrazine Thruster



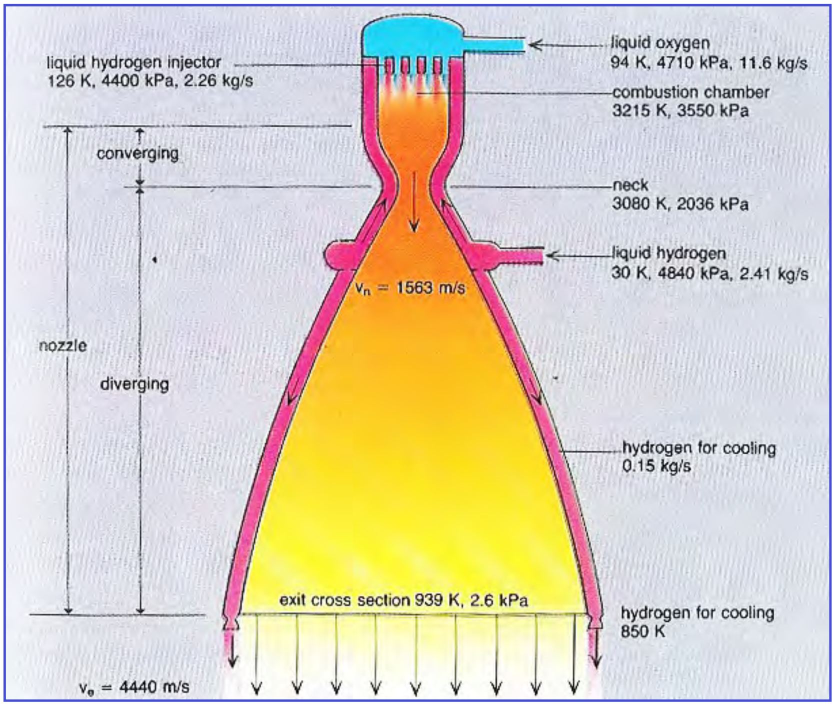
Aerojet Rocketdyne



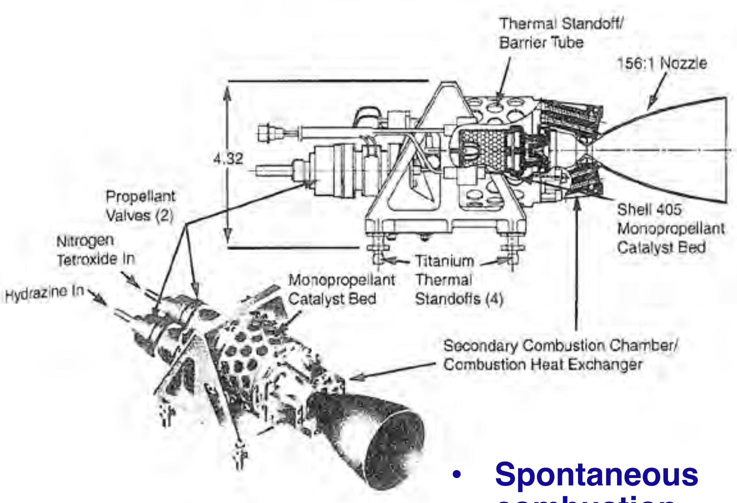
- **Catalytic decomposition produces thrust**
- **Reliable**
- **Low performance**
- **Toxic**

Bi-Propellant Rocket Motor

Thrust / Motor Weight ~ 70:1



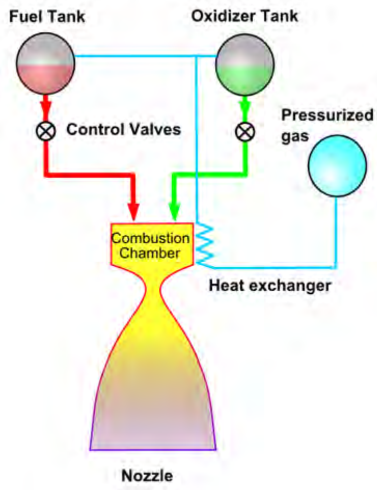
Hypergolic, Storable Liquid-Propellant Thruster



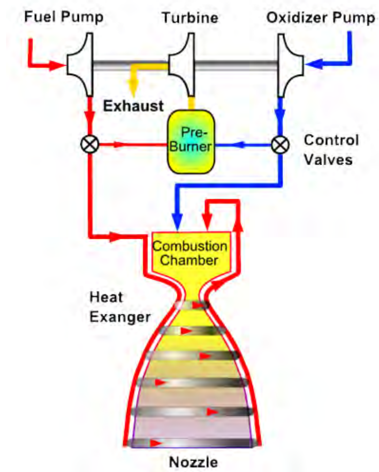
- **Spontaneous combustion**
- **Reliable**
- **Corrosive, toxic**

Pressure-Fed and Turbopump Engine Cycles

Pressure-Fed Rocket Cycle

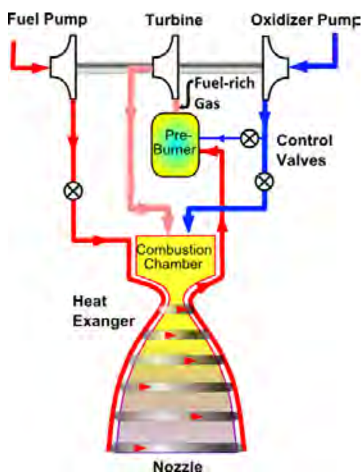


Gas-Generator Rocket Cycle, with Nozzle Cooling

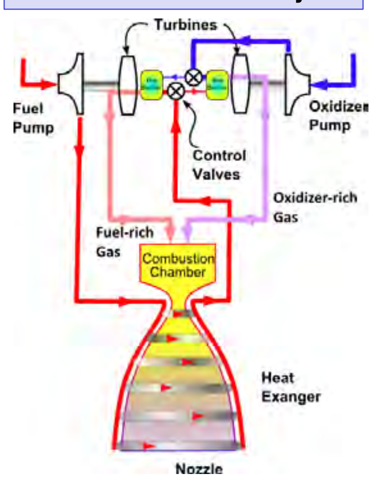


Staged Combustion Engine Cycles

Staged Combustion Rocket Cycle

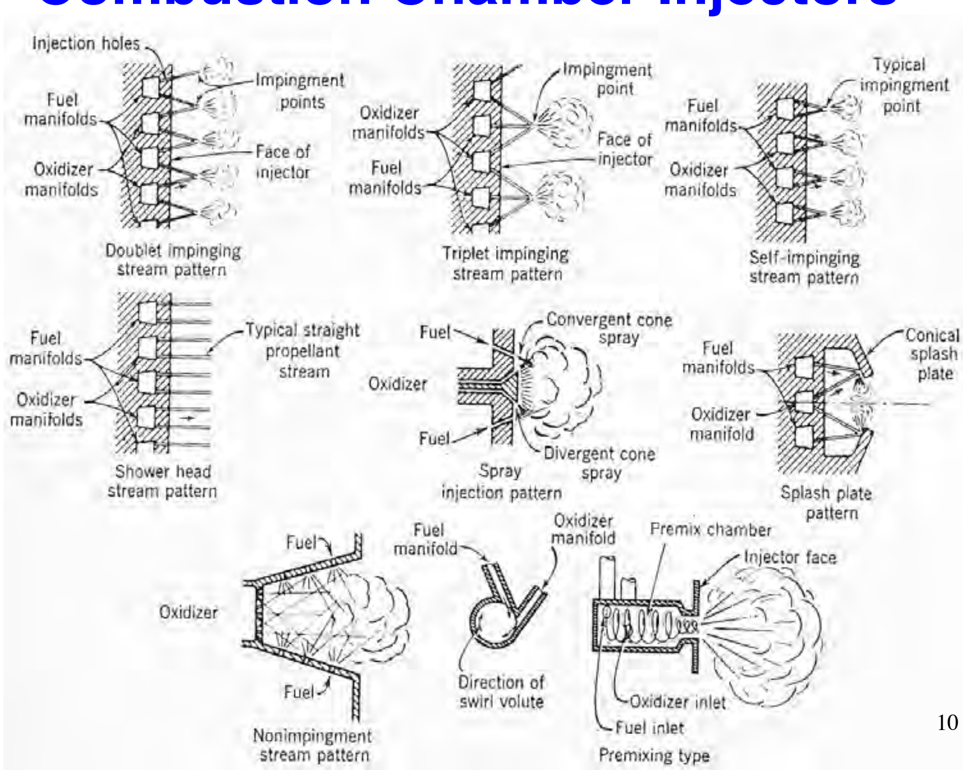


Full-Flow Staged Combustion Rocket Cycle

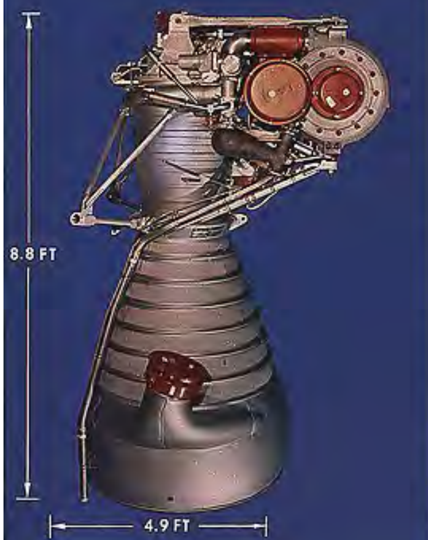


German V-2 Rocket Motor, Fuel Injectors, and Turbopump

Combustion Chamber Injectors



H-1 ENGINE



	VEHICLE EFFECTIVITY	
	SA-201 THRU SA-205	SA-206 & SUBSEQUENT
THRUST (SEA LEVEL)	200,000 LB	205,000 LB
THRUST DURATION	155 SEC	155 SEC
SPECIFIC IMPULSE (LB-SEC/LB)	260.5 MIN	261.0 MIN
ENGINE WT DRY (INBD)	1,830 LB	2,100 LB
(OUTBD)	2,100 LB	2,100 LB
ENGINE WT BURNOUT (INBD)	2,200 LB	2,200 LB
(OUTBD)	2,200 LB	2,200 LB
EXIT-TO-THROAT AREA RATIO	8 TO 1	8 TO 1
PROPELLANTS	LOX & RP-1	LOX & RP-1
MIXTURE RATIO	2.23±2%	2.23±%

CONTRACTOR: NAA/ROCKETDYNE
VEHICLE APPLICATION
SATURN IB/S-IB STAGE (EIGHT ENGINES)

IND B1408F

11

F-1 ENGINE

	VEHICLE EFFECTIVITY	
	SA-501 THRU SA-503	SA-504 & SUBSEQUENT
THRUST (SEA LEVEL)	1,500,000 LB	1,522,000 LB
THRUST DURATION	150 SEC	165 SEC
SPECIFIC IMPULSE (LB-SEC/LB)	260 SEC MIN	263 MIN
ENGINE WEIGHT DRY	18,416 LB	18,500 LB
ENGINE WEIGHT BURNOUT	20,096 LB	20,180 LB
EXIT-TO-THROAT AREA RATIO	16 TO 1	16 TO 1
PROPELLANTS	LOX & RP 1	LOX & RP 1
MIXTURE RATIO	2.27±2%	2.27±2%

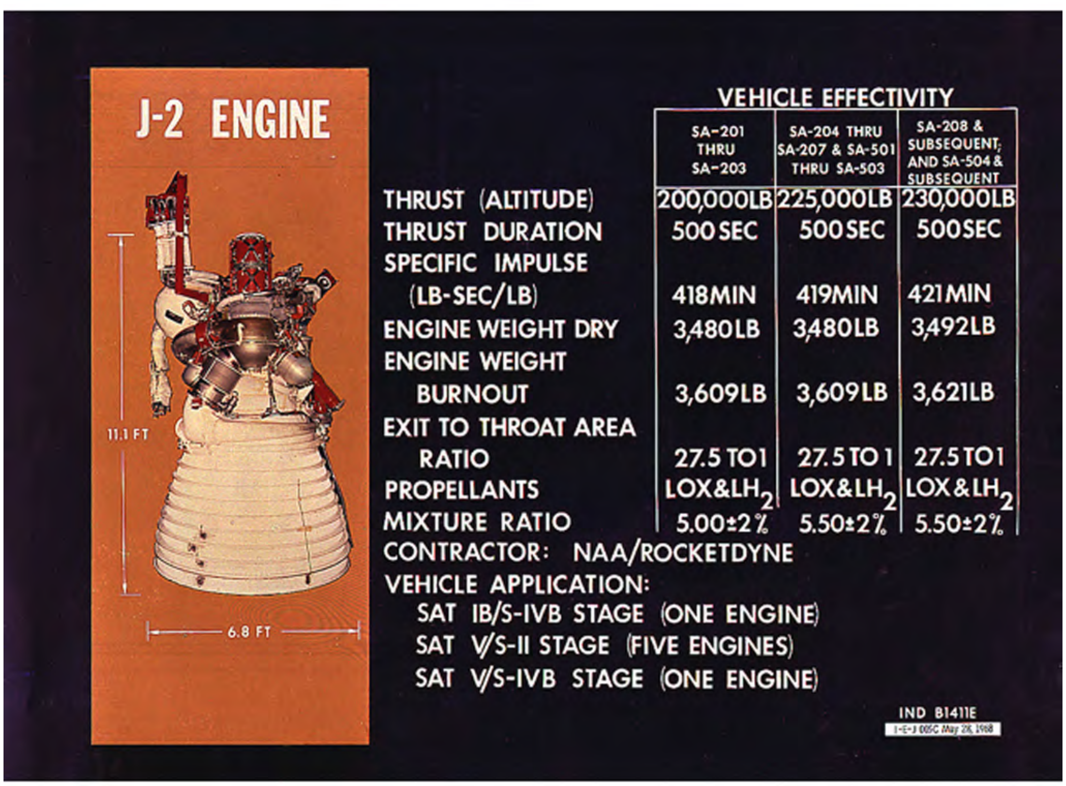
CONTRACTOR: NAA/ROCKETDYNE
VEHICLE APPLICATION:
SATURN V/S-IC STAGE (FIVE ENGINES)

IND B1413D

12

Origins of the F-1

- Air Force legacy (1955)
  - Design undertaken before vehicle or mission were identified
- Big engine, big problems
  - 16:1 nozzle expansion
  - 6.67 MN thrust
- F-1 turbopumps
  - Oxygen: 24,811 gal/min
  - RP-1: 15,741 gal/min
- F-1 injector
- Combustion instability
  - Significant theoretical work by Luigi Crocco and David Harrje, Princeton



J-2 ENGINE

11.1 FT

6.8 FT

	VEHICLE EFFECTIVITY		
	SA-201 THRU SA-203	SA-204 THRU SA-207 & SA-501 THRU SA-503	SA-208 & SUBSEQUENT; AND SA-504 & SUBSEQUENT
THRUST (ALTITUDE)	200,000LB	225,000LB	230,000LB
THRUST DURATION	500 SEC	500 SEC	500SEC
SPECIFIC IMPULSE (LB-SEC/LB)	418MIN	419MIN	421MIN
ENGINE WEIGHT DRY	3,480LB	3,480LB	3,492LB
ENGINE WEIGHT BURNOUT	3,609LB	3,609LB	3,621LB
EXIT TO THROAT AREA RATIO	27.5 TO 1	27.5 TO 1	27.5 TO 1
PROPELLANTS	LOX&LH_2	LOX&LH_2	LOX&LH_2
MIXTURE RATIO	5.00±2%	5.50±2%	5.50±2%

CONTRACTOR: NAA/ROCKETDYNE

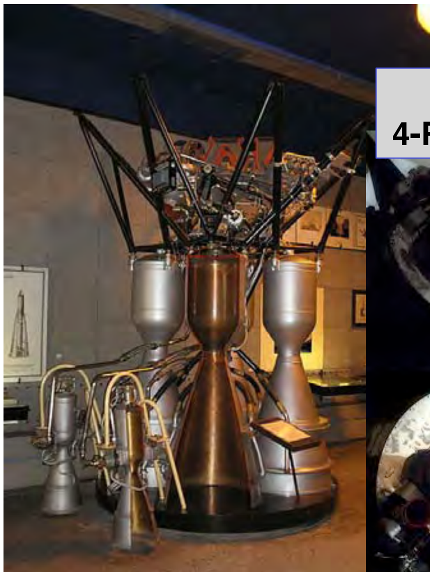
VEHICLE APPLICATION:
SAT IB/S-IVB STAGE (ONE ENGINE)
SAT V/S-II STAGE (FIVE ENGINES)
SAT V/S-IVB STAGE (ONE ENGINE)

IND B1411E

USSR RD-107/8 Rocket Motors

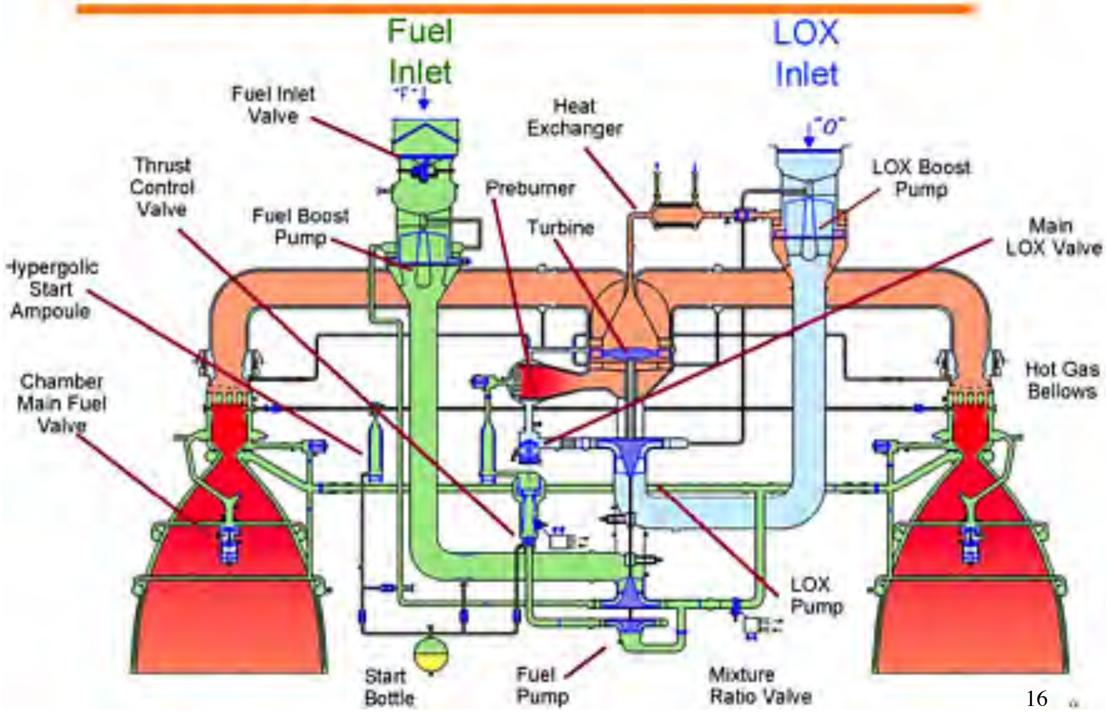
RD-107
4 combustion chambers, 2 verniers

RD-108
4 combustion chambers, 4 verniers

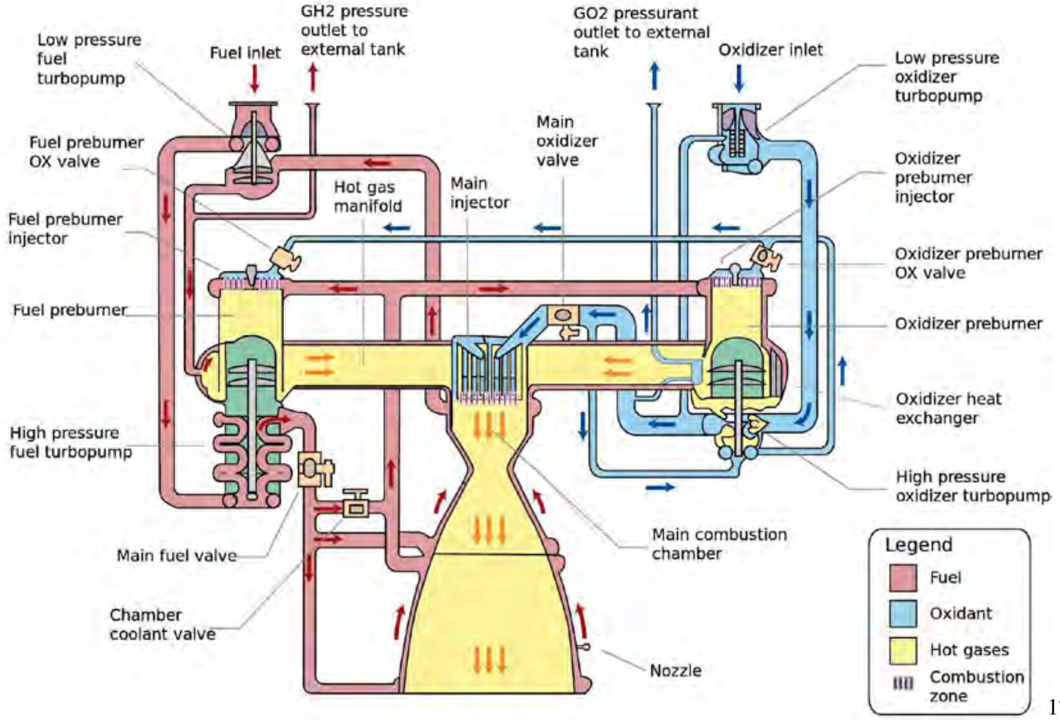


R-7 Base
4-RD-107, 1-RD-108

RD-180 Engine Schematic (used on Atlas V)



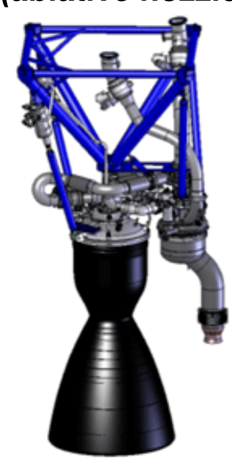
Special Shuttle Main Engine (RS-25)



17

SpaceX Merlin Family

Merlin 1A
(ablative nozzle)

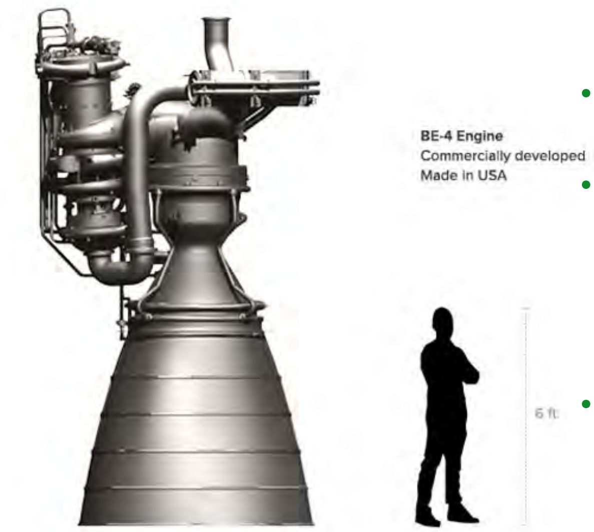


Roll control from turbine exhaust

Merlin 1C
(vacuum nozzle)

Merlin 1D
(throttlable)

Blue Origin BE-4



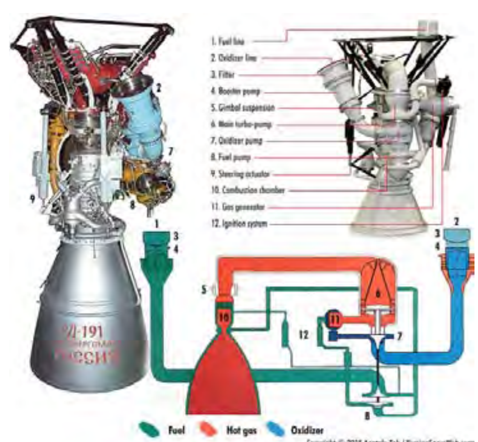
- **LOX/Liquefied natural gas**
- **United Launch Alliance has chosen as motor for the Vulcan launch vehicle**
- **Thrust = 2.5 MN (550,000 lb)**

RD-181 and RD-191

RD-181

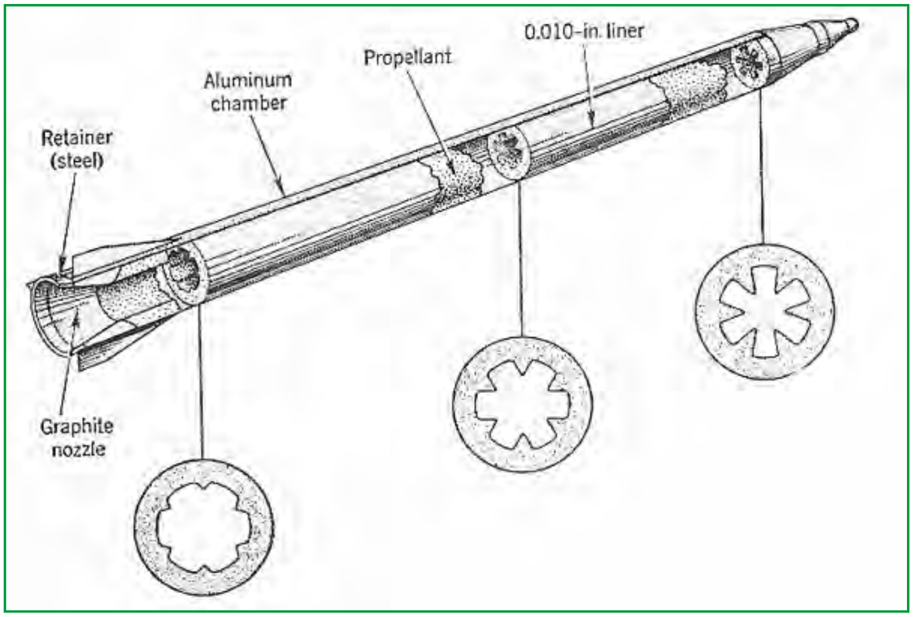
to be used on Orbital-ATK Antares

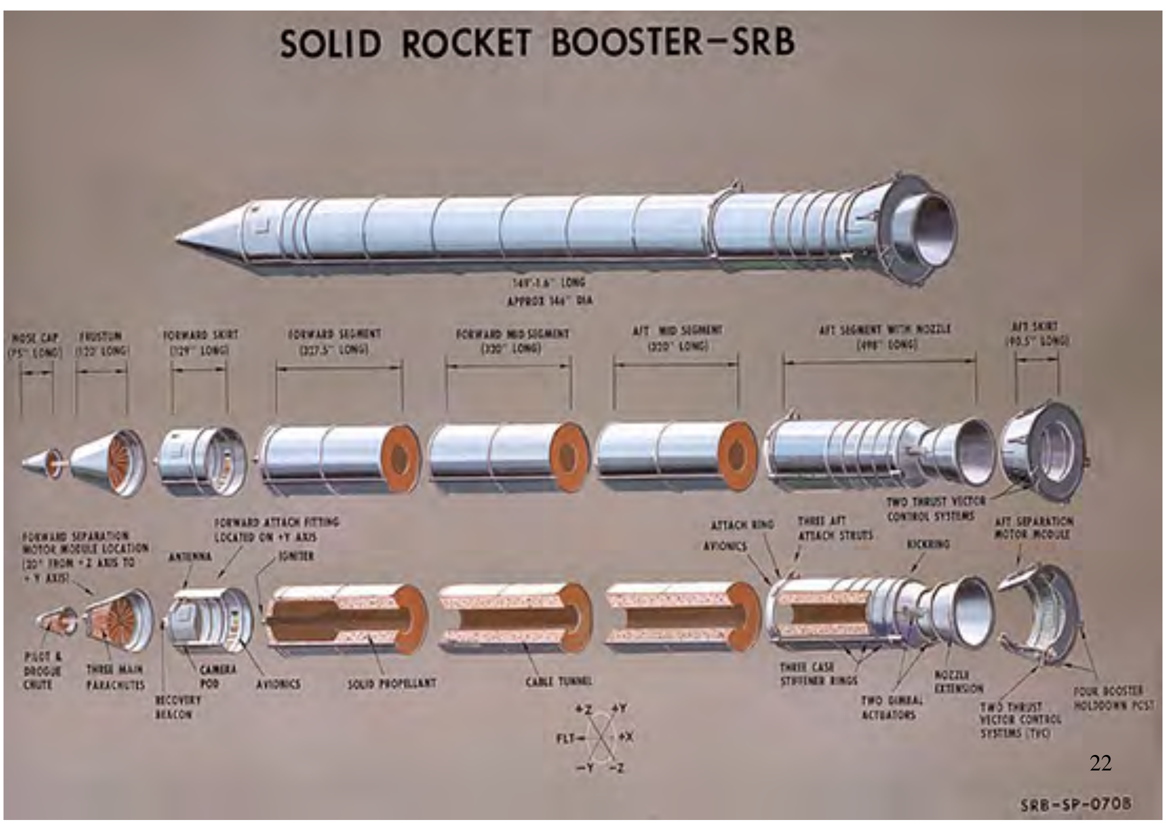
RD-191



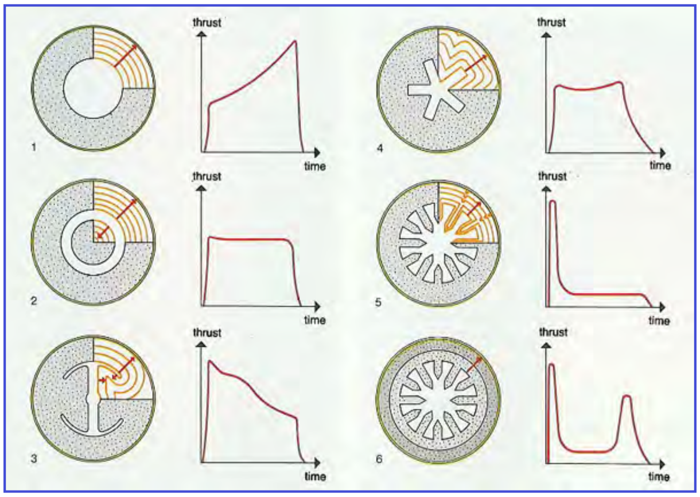
to be used on NPO Energomash Angara

Solid-Fuel Rocket Motor



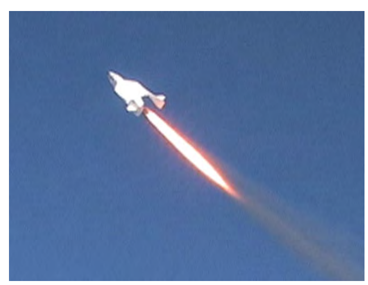


Solid-Fuel Rocket Motor

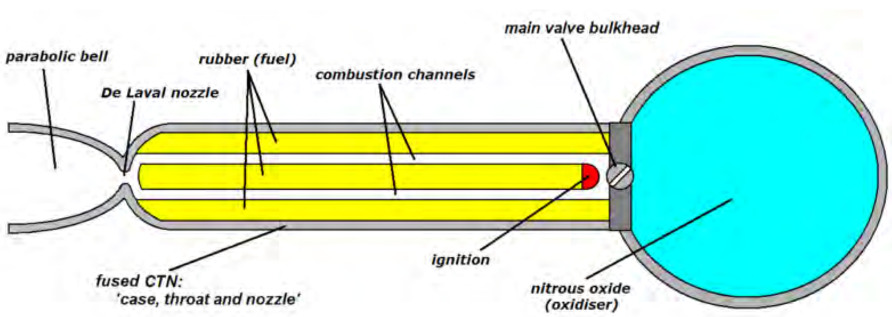


Thrust is proportional to burning area
Rocket grain patterns affect thrust profile
Propellant chamber must sustain high pressure and temperature
Environmentally unfriendly exhaust gas

Hybrid-Fuel Rocket Motor



- **SpaceShipOne motor**
  - **Nitrous oxide**
  - **Hydroxy-terminated polybutadiene (HTPB)**
- **Issues**
  - **Hard start**
  - **Blow back**
  - **Complete mixing of oxidizer and fuel toward completion of burn**

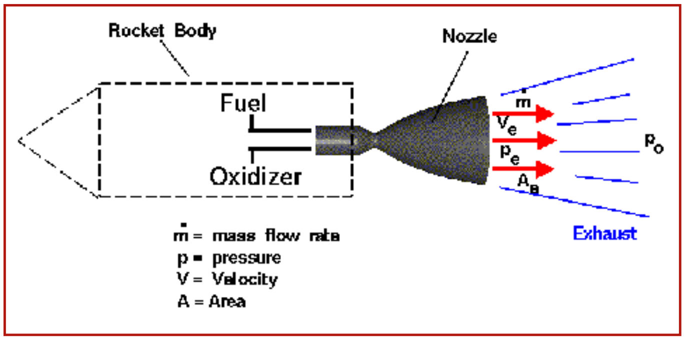


Rocket Thrust

$$\text{Thrust} = \dot{m}_{propellant} V_{exhaust} + A_{exit}\left(p_{exit} - p_{ambient}\right) \equiv \dot{m}\, c_{eff}$$

$$c_{eff} = \frac{\text{Thrust}}{\dot{m}} = \text{Effective exhaust velocity}$$

$$\dot{m} \equiv \text{Mass flow rate of on-board propellant}$$



Specific Impulse

$$I_{sp} = \frac{\text{Thrust}}{\dot{m}\, g_o} = \frac{c_{eff}}{g_o}, \quad \text{Units} = \frac{m/s}{m/s^2} = \text{ seconds}$$

$$g_o \equiv \text{Gravitational acceleration at earth's surface}$$

- **g_o is a normalizing factor for the definition**
- **Chemical rocket specific impulse (vacuum)**
  - **Solid propellants: < 295 s**
  - **Liquid propellants: < 510 s**

- **Space Shuttle Specific Impulses**
  - **–Solid boosters: 242-269 s**
  - **–Main engines: 455 s**
  - **–OMS: 313 s**
  - **–RCS: 260-280 s**

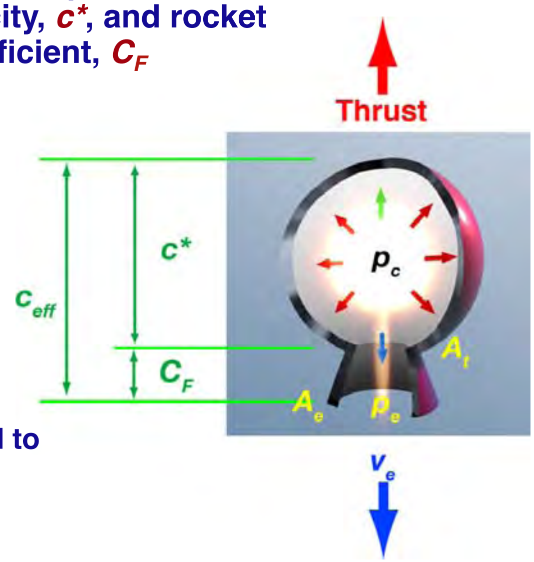
Specific Impulse

Specific impulse is a product of characteristic velocity, *c, and rocket thrust coefficient, C_F**

$$I_{sp} = \frac{\text{Thrust}}{\dot{m}\, g_o} = \frac{c_{eff}}{g_o}$$

$$= C_F\, c^* / g_o$$

$$= \frac{V_{exhaust}}{g_o} \quad \text{when } C_F = 1,\ p_e = p_{ambient}$$



- **Characteristic velocity is related to**
  - combustion chamber performance
  - propellant characteristics
- **Thrust coefficient is related to**
  - nozzle shape
  - exit/ambient pressure differential

The Rocket Equation

Ideal velocity increment of a rocket stage, ΔV_I (gravity and aerodynamic effects neglected)

$$\frac{dV}{dt} = \frac{\text{Thrust}}{m} = \frac{\dot{m}\, c_{eff}}{m} = -\frac{{dm}/{dt}\; I_{sp} g_o}{m}$$

$$\int_{V_i}^{V_f} dV = -I_{sp} g_o \int_{m_i}^{m_f} {dm}/{m} = -I_{sp} g_o \ln m \Big|_{m_i}^{m_f}$$

$$\left(V_f - V_i\right) \equiv \Delta V_I = I_{sp} g_o \ln\left(\frac{m_i}{m_f}\right) \equiv I_{sp} g_o \ln \mu$$

Volumetric Specific Impulse

Specific impulse

$$\Delta V_I = I_{sp} g_o \ln \mu = I_{sp} g_o \ln\left(\frac{m_{final} + m_{propellant}}{m_{final}}\right) = I_{sp} g_o \ln\left(1 + \frac{m_{propellant}}{m_{final}}\right)$$

$$= I_{sp} g_o \ln\left(1 + \frac{Density_{propellant} \bullet Volume_{propellant}}{m_{final}}\right)$$

$$\approx g_o I_{sp}\left(\frac{\rho_{propellant} \bullet Vol_{propellant}}{m_{final}}\right) = g_o\left(I_{sp}\rho_{propellant}\right)\frac{Vol_{propellant}}{m_{final}}$$

Volumetric specific impulse

$$I_{sp_{vol}} \triangleq VI_{sp} = I_{sp}\rho_{propellant}$$

Volumetric Specific Impulse

- For fixed volume and final mass, increasing volumetric specific impulse increases ideal velocity increment

	Density, g/cc	Isp, s, SL	VIsp, s (g/cc), SL	Isp, s, vac	VIsp, s (g/cc), vac
LOX/Kerosene	1.3	265	345	304	395
LOX/LH2 (Saturn V)	0.28	360	101	424	119
LOX/LH2 (Shuttle)	0.28	390	109	455	127
Shuttle Solid Booster	1.35	242	327	262	354

- Saturn V Specific Impulses, vacuum (sea level)
  - 1st Stage, 5 F-1 LOX-Kerosene Engines: 304 s (265 s)
  - 2nd Stage, 5 J-2 LOX-LH2 Engines: 424 s (~360 s)
  - 3rd Stage, 1 J-2 LOX-LH2 Engine: 424 s (~360 s)

Typical Values of Chemical Rocket Specific Impulse

- Chamber pressure = 7 MPa (low by modern standards)
- Expansion to exit pressure = 0.1 MPa

Liquid-Fuel Rockets

Monopropellant		Isp, s	VIsp, kg-s/m^3 x 10^3
Hydrogen Peroxide		165	238
Hydrazine		199	201
Nitromethane		255	290

Bipropellant

Fuel	Oxidizer	Isp, s	VIsp, kg-s/m^3 x 10^3
Kerosene	Oxygen	301	307
	Flourine	320	394
	Red Fuming Nitric Acid	268	369
Hydrogen	Oxygen	390	109
	Flourine	410	189
UDMH	Nitrogen Tetroxide	286	339

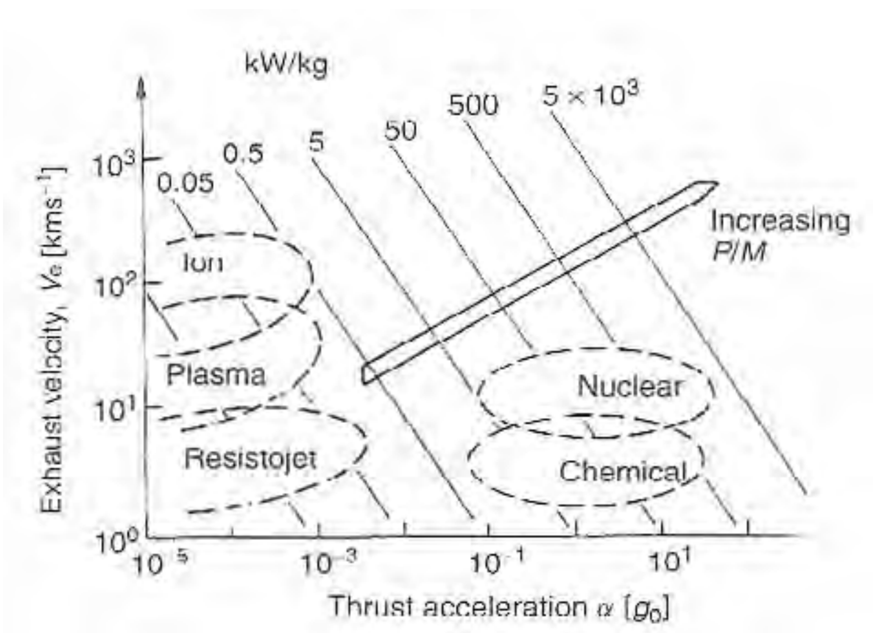
Solid-Propellant Rockets

Double-Base	Isp, s	VIsp, kg-s/m^3 x 10^3
AFU	196	297
ATN	235	376
JPN	250	405
Composite		
JPL 540A	231	383
TRX-H609	245	431
PBAN (SSV)	260	461

Hybrid-Fuel Rocket

Fuel	Oxidizer	Isp, s
HTPB	N2O	250

Exhaust Velocity vs. Thrust Acceleration



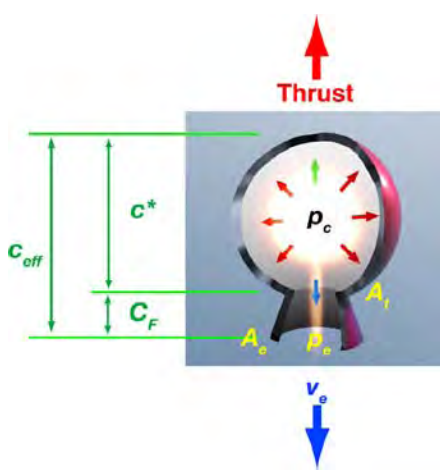
Rocket Characteristic Velocity, *c**

$$c^* = \frac{1}{\Gamma}\sqrt{\frac{R_o T_c}{M}}, \quad \text{where} \quad \Gamma = \sqrt{\gamma}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}$$

$$R_o = \text{universal gas constant} = 8.3\times10^3 \text{ kg m}^2/\text{s}^2\ ^\circ\text{K}$$
$$T_c = \text{chamber temperature}, ^\circ\text{K}$$
$$M = \text{exhaust gas mean molecular weight}$$
$$\gamma = \text{ratio of specific heats } (\sim 1.2\text{-}1.4)$$

Rocket Characteristic Velocity, *c**

$$c^* = \frac{p_c A_t}{\dot{m}} = \text{exhaust velocity if } C_F = 1$$

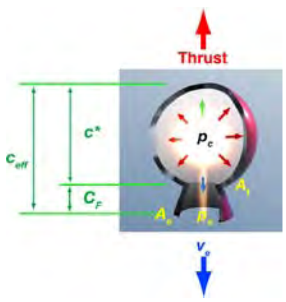


Rocket Thrust Coefficient, C_F

$$C_F = \frac{\text{Thrust}}{p_c A_t} = \lambda \Gamma \sqrt{\left(\frac{2\gamma}{\gamma-1}\right)\left[1-\left(\frac{p_e}{p_c}\right)^{(\gamma-1)/\gamma}\right]} + \left(\frac{p_e - p_{ambient}}{p_c}\right)\frac{A_e}{A_t}$$

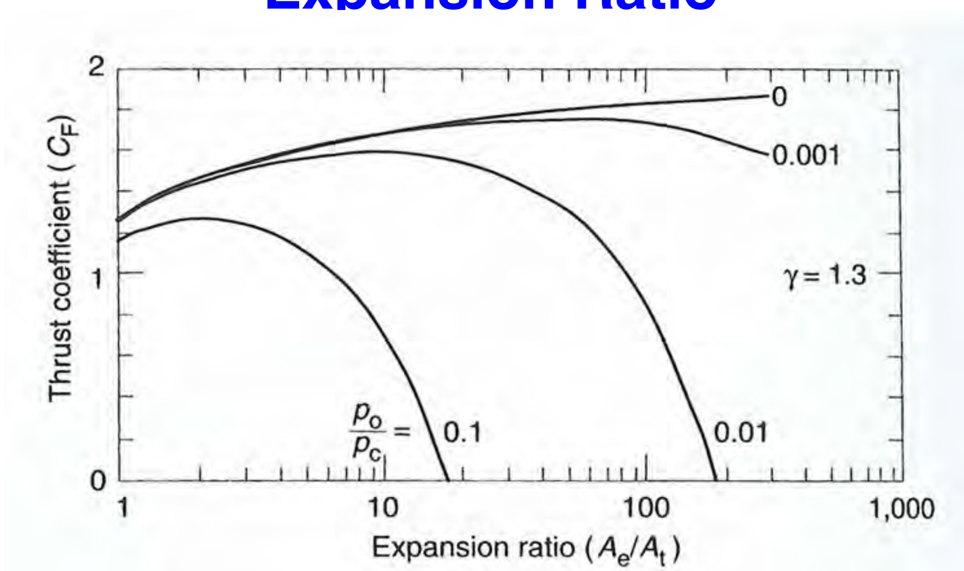
$$\textbf{Thrust} = \lambda\, \dot{m}\, v_e + A_e\left(p_e - p_{ambient}\right)$$

λ : reduction ratio (function of nozzle shape)



C_F typically 0.5 - 2

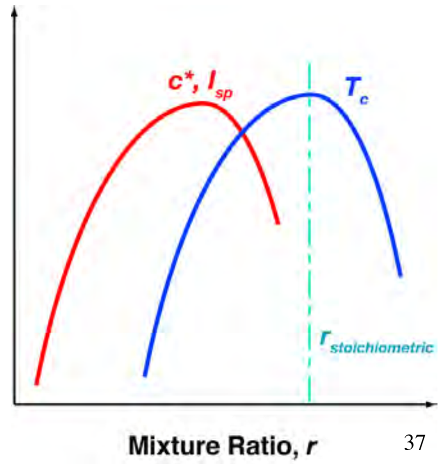
Thrust Coefficient, C_F, vs. Nozzle Expansion Ratio



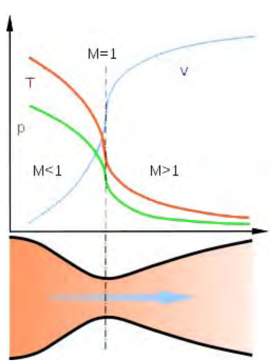
Mixture Ratio, *r*

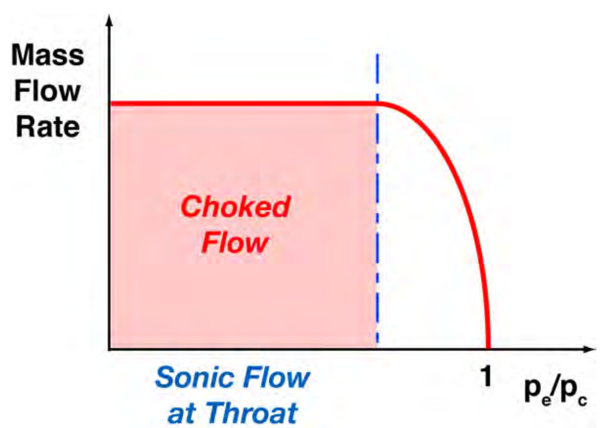
$$r = \frac{\dot{m}_{oxidizer}}{\dot{m}_{fuel}}; \quad \dot{m}_{fuel} = \frac{\dot{m}_{total}}{1+r}; \quad \text{"leaner"} < r < \text{"richer"}$$

- **Stoichiometric mixture: complete chemical reaction of propellants**
- **Specific impulse maximized with lean mixture ratio, *r* (i.e., below stoichiometric maximum)**



Effect of Pressure Ratio on Mass Flow





In choked flow, mass flow rate is maximized

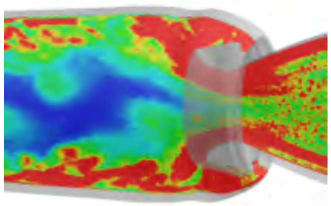
$$\dot{m} = \frac{\Gamma p_c A_t}{\sqrt{R_o T_c / M}}$$

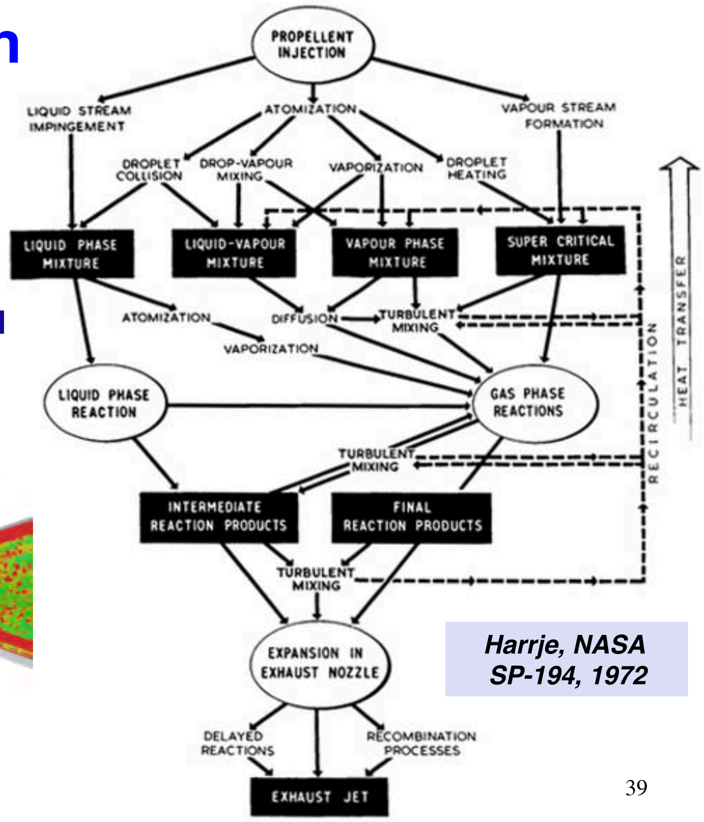
Choked flow occurs when

$$\frac{p_e}{p_c} \le \left(\frac{2}{\gamma+1}\right)^{\gamma/\gamma-1} \approx 0.53$$

Combustion Instability

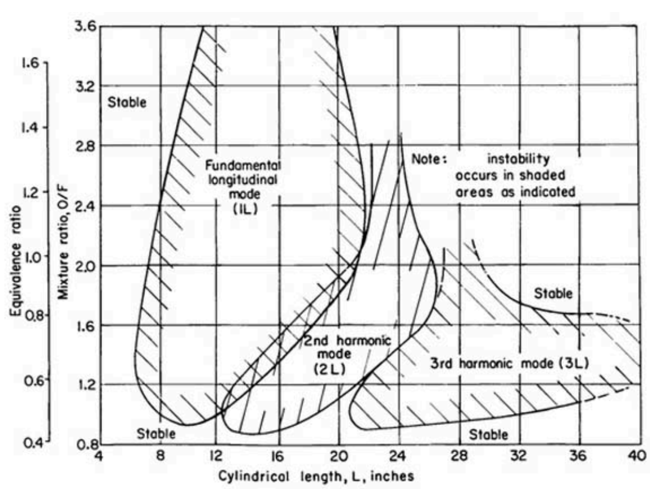
- Complex mix of species, phases, pressures, temperatures, and flows
- Cavity resonance





Harrje, NASA SP-194, 1972

Combustion Instability



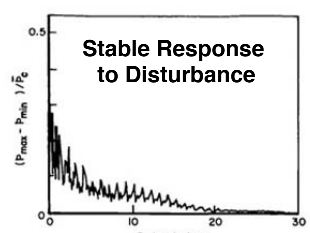
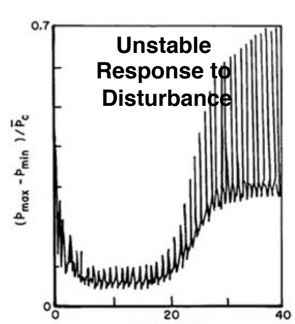


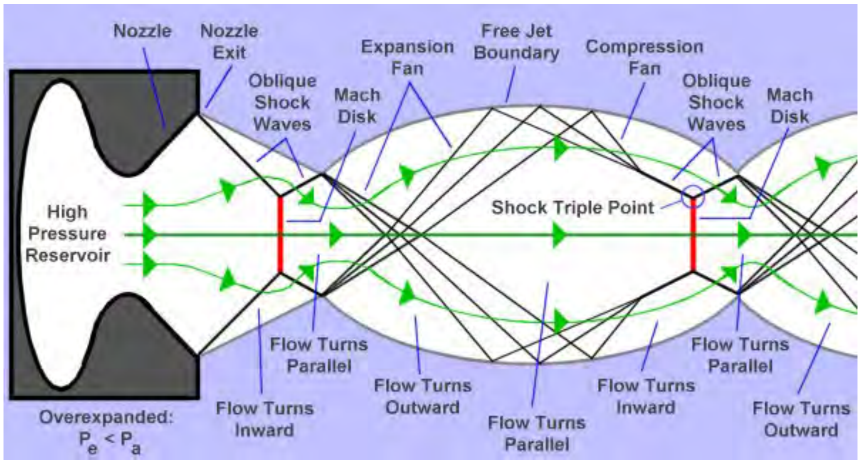
FIGURE 6.4.1b.—Computer plot of $(p_{max} - p_{min})/p_c$ versus reduced time. Stable operation.



Harrje, NASA SP-194, 1972

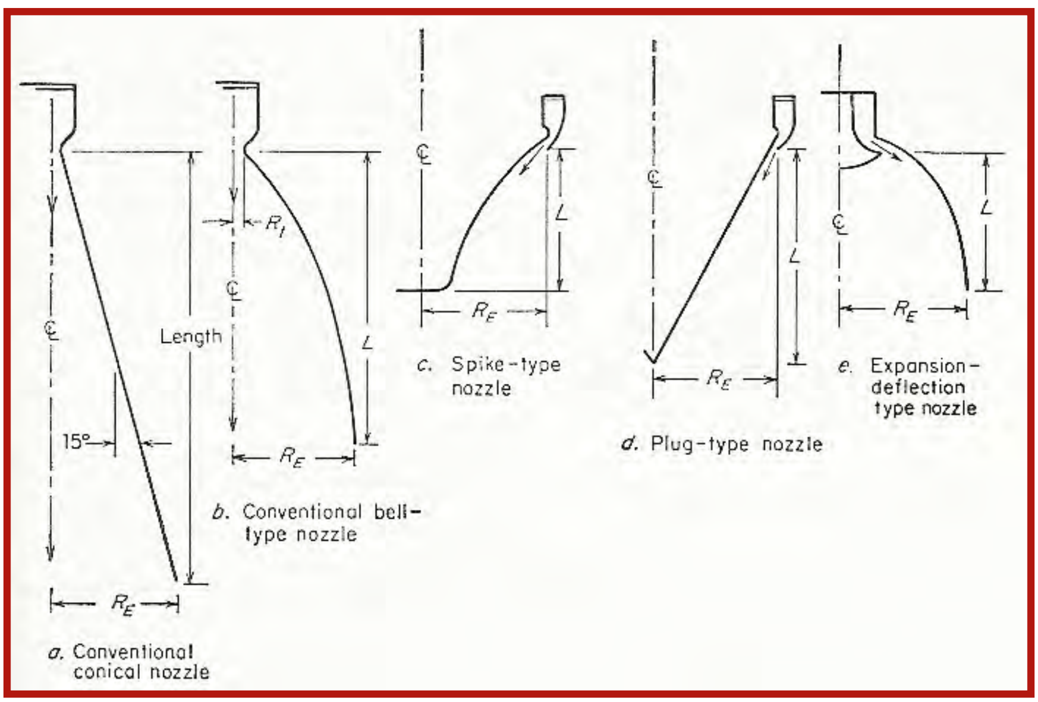
Shock Diamonds

When $p_e \neq p_a$, exhaust flow is over- or underexpanded
Effective exhaust velocity < maximum value



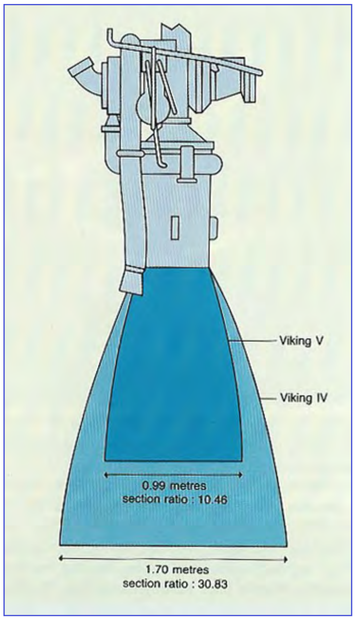
https://www.youtube.com/watch?v=qiMSko4HBe8

Rocket Nozzles

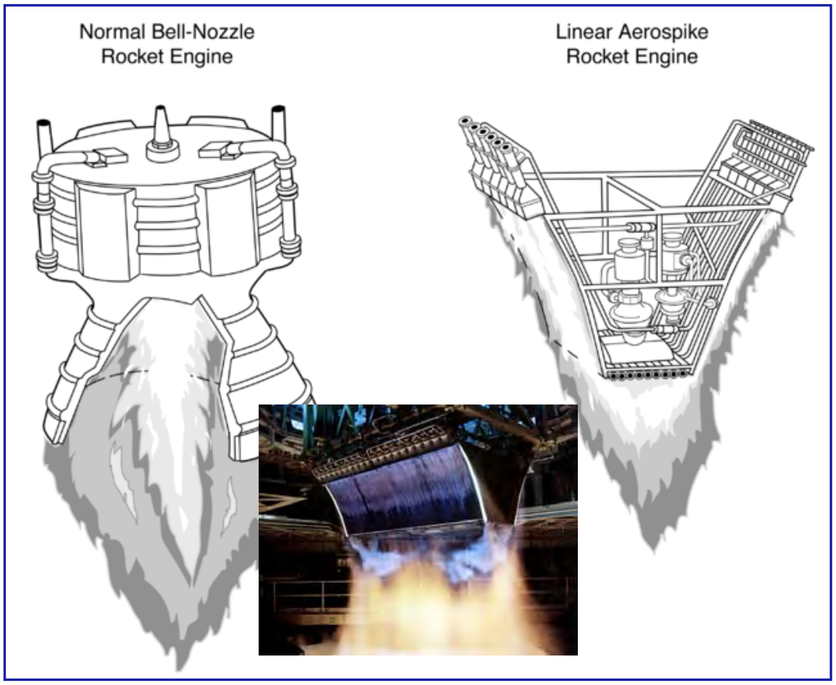


Rocket Nozzles

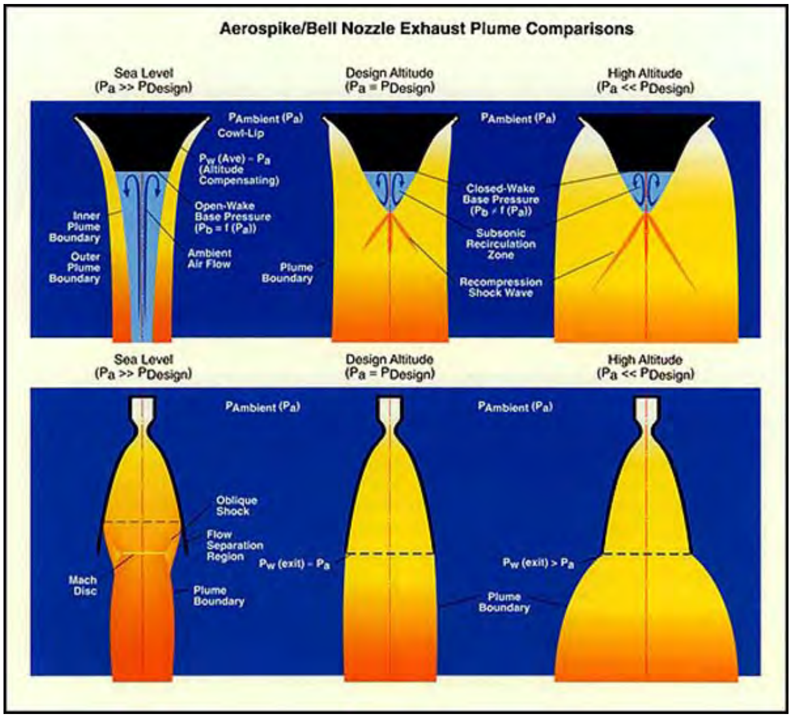
- **Expansion ratio, A_e/A_t, chosen to match exhaust pressure to average ambient pressure**
  - Ariane rockets: Viking V for sea level, Viking IV for high altitude
- **Rocket nozzle types**
  - DeLaval nozzle
  - Isentropic expansion nozzle
  - Spike/plug nozzles
  - Expansion-deflection nozzle



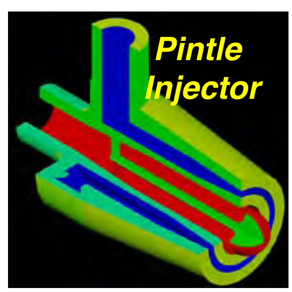
Rocket Nozzles

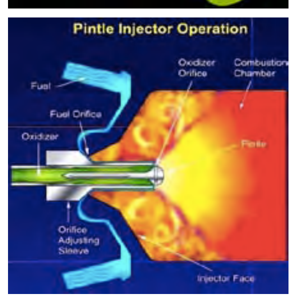


Linear Spike/Plug Nozzles



Throttling, Start/Stop Cycling



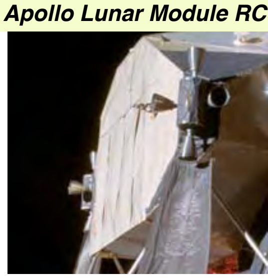


Reaction Control Thrusters

- Direct control of angular rate
- Unloading momentum wheels or control-moment gyros
- Reaction control thrusters are typically on-off devices using
  - Cold gas
  - Hypergolic propellants
  - Catalytic propellant
  - Ion/plasma rockets
- Thrusters commanded in pairs to cancel velocity change

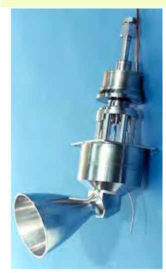
- **Issues**
  - Specific impulse
  - Propellant mass
  - Expendability

Apollo Lunar Module RCS

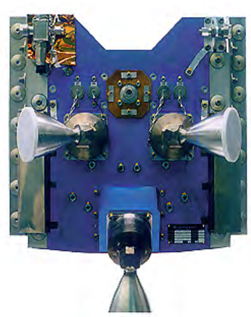


Space Shuttle RCS

RCS Thruster



Divert and Attitude Control Thrusters



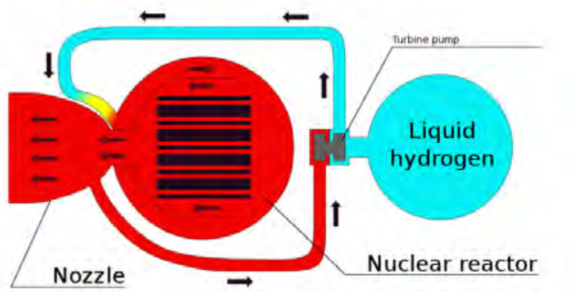
https://www.youtube.com/watch?v=W8efpDBvTDE

https://www.youtube.com/watch?v=71qgl6bddM8

https://www.youtube.com/watch?v=KBMU6l6GsdM

https://www.youtube.com/watch?v=JURQYH669_g

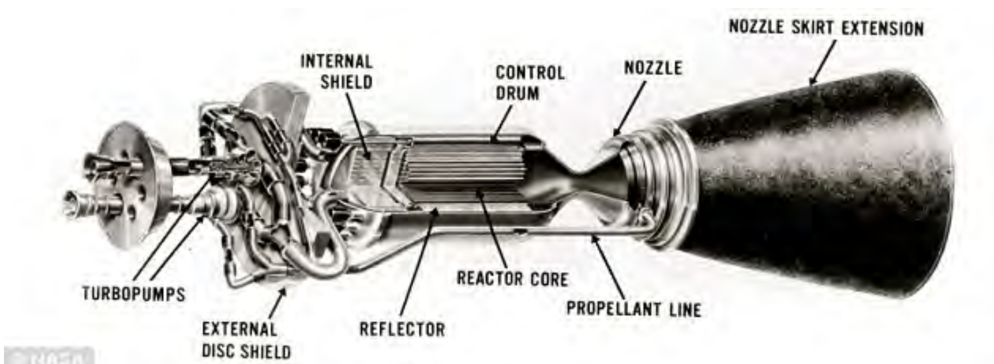
Nuclear Propulsion



$$c^* = \frac{1}{\Gamma}\sqrt{\frac{R_o T_c}{M}}$$

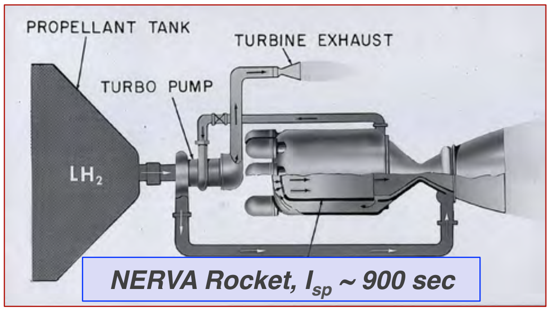
- **Nuclear reaction produces thermal energy to heat inert working fluid**
  - **Solid core**
  - **Liquid core**
  - **Gaseous core**
- **High propellant temperature leads to high specific impulse**
- **Working fluid chosen for low molecular weight and storability**

Solid-Core Nuclear Rocket



- **Operating temperature limited by**
  - **melting point of reactor materials**
  - **cracking of core coating**
  - **matching coefficients of expansion**
- **Possible propellants: hydrogen, helium, liquid oxygen, water, ammonia**
- **I_{sp} = 850 – 1,000 sec**
- **T / W ~ 7:1**

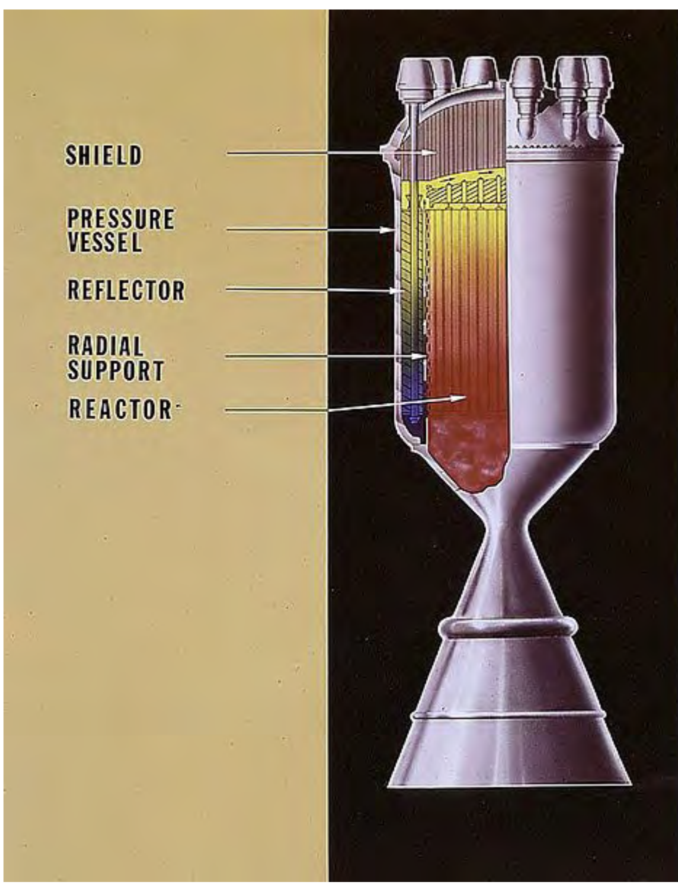
Project Rover, 1955-1972

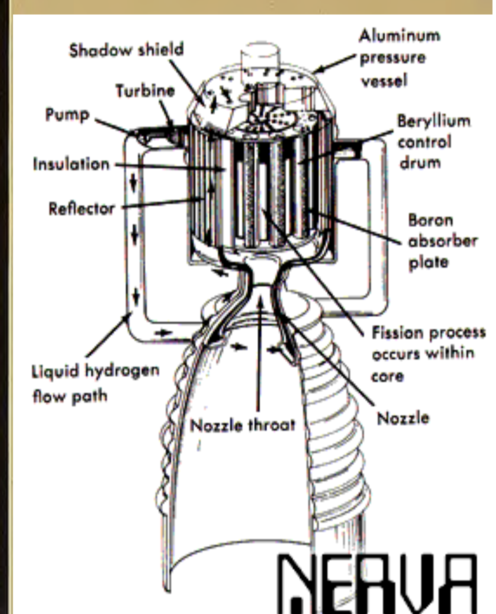


NERVA Rocket, I_{sp} ~ 900 sec

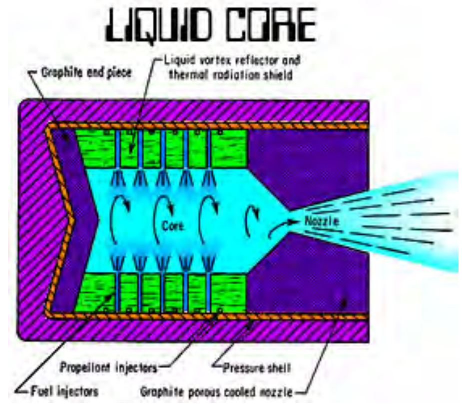
NERVA-Powered Mars Mission

Kiwi-B4-A Reactor/Rocket



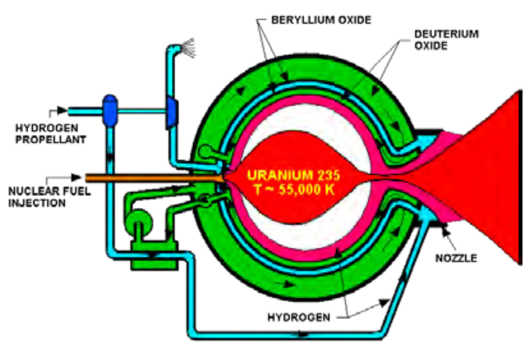


Liquid/Particle-Core Nuclear Rocket



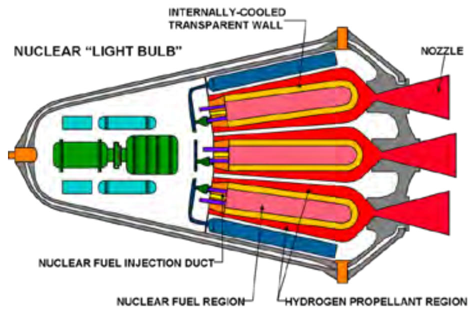
- Nuclear fuel mixed with working fluid
- In principle, could operate above melting point of nuclear fuel
- I_{sp} ~ 1,300 – 1,500 sec
- Conceptual
- Massive radioactive waste

Open-Cycle Gas Core Nuclear Rocket



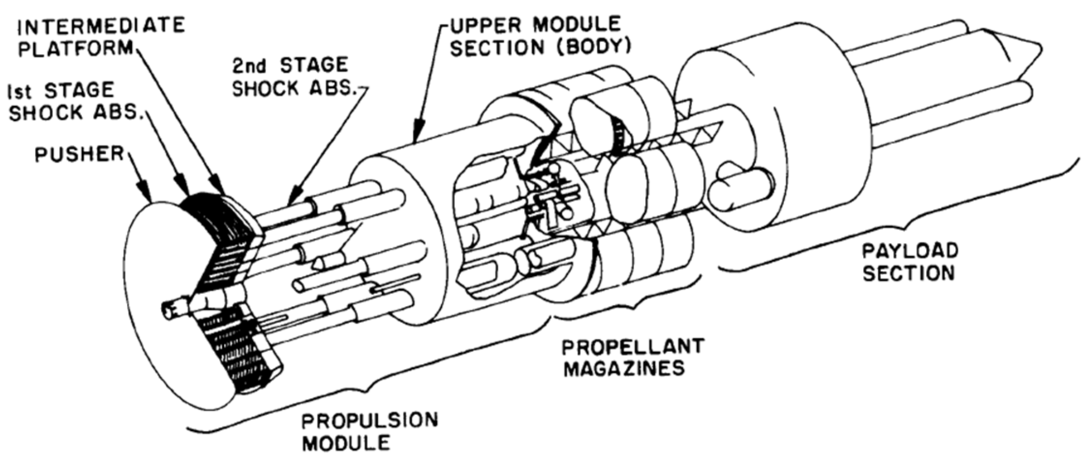
- Toroidal circulation of working fluid confines nuclear fuel to center
- Fuel does not touch the wall
- Conceptual
- Massive radioactive waste
- I_{sp} ~ 3,000 – 5,000 sec

Closed-Cycle Gas Core Nuclear Rocket



- "Nuclear light bulb"
- Nuclear fuel contained in quartz container
- I_{sp} ~ 1,500 – 2,000 sec
- Conceptual

Nuclear-Pulse ("Explosion") Rocket - Project Orion



"Physics packages" ejected behind the pusher plate

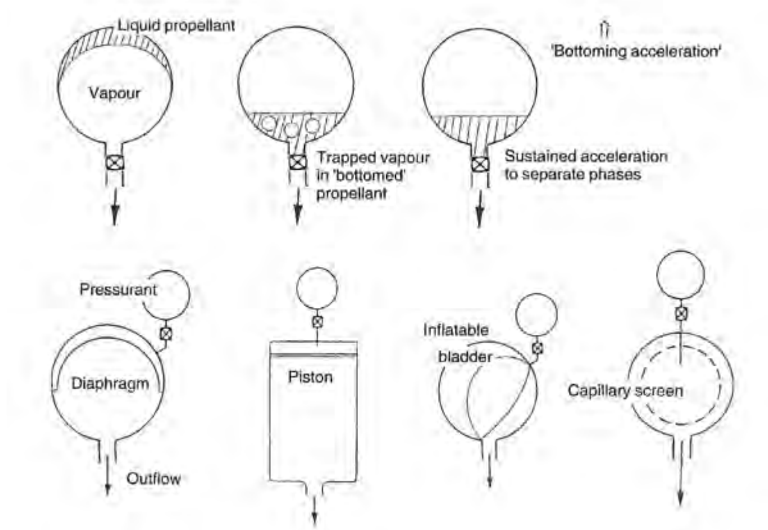
https://en.wikipedia.org/wiki/Project_Orion_(nuclear_propulsion)

Next Time:
Launch Vehicles

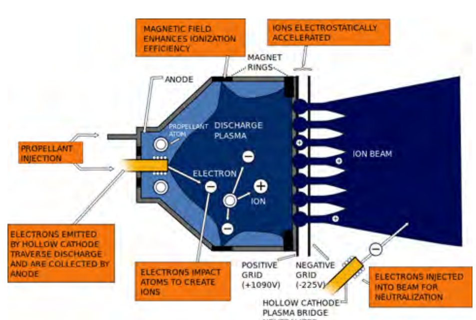
SUPPLEMENTAL MATERIAL

Propellant Tanks

Propellant must be kept near the exit duct without bubbles during thrusting



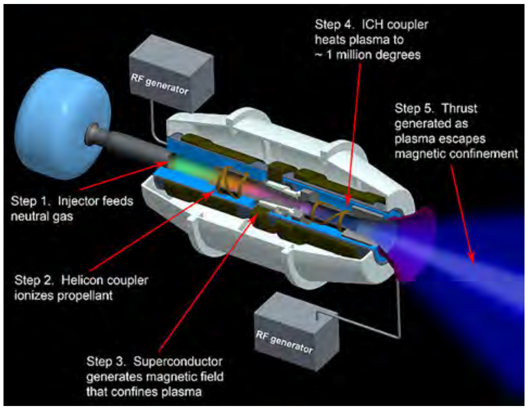
Ion/Plasma Thrusters

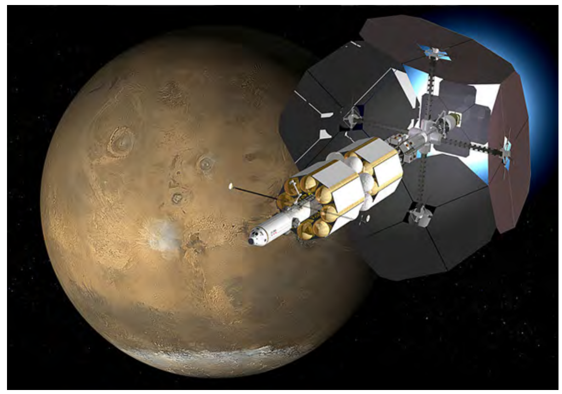


Engine	Propellant	Required power kW	Specific impulse s	Thrust mN
NSTAR	Xenon	2.3	3,300 to 1,700	92 max
NEXT[	Xenon	6.9	4,300	236 max
HiPEP	Xenon	20–50	6,000–9,000	460–670
Hall effect	Xenon	25	3,250	950
FEEP	Liquid Cesium	6×10–5–0.06	6,000–10,000	0.001–1
VASIMR	Argon	200	3,000–12,000	~5,000
DS4G	Xenon	250	19,300	2,500 max

Variable Specific Impulse Magnetoplasma Rocket (VASIMR)

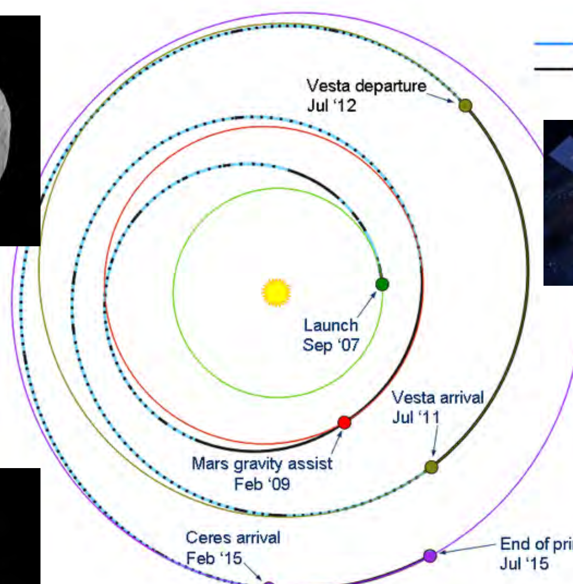
Propellant	Required power kW	Specific impulse s	Thrust mN
Argon	200	3,000–12,000	~5,000

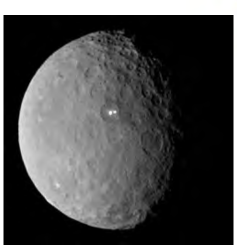




61

DAWN Spacecraft





Engine	Propellant	Required power kW	Specific impulse s	Thrust mN
NSTAR	Xenon	2.3	3,300 to 1,700	92 max