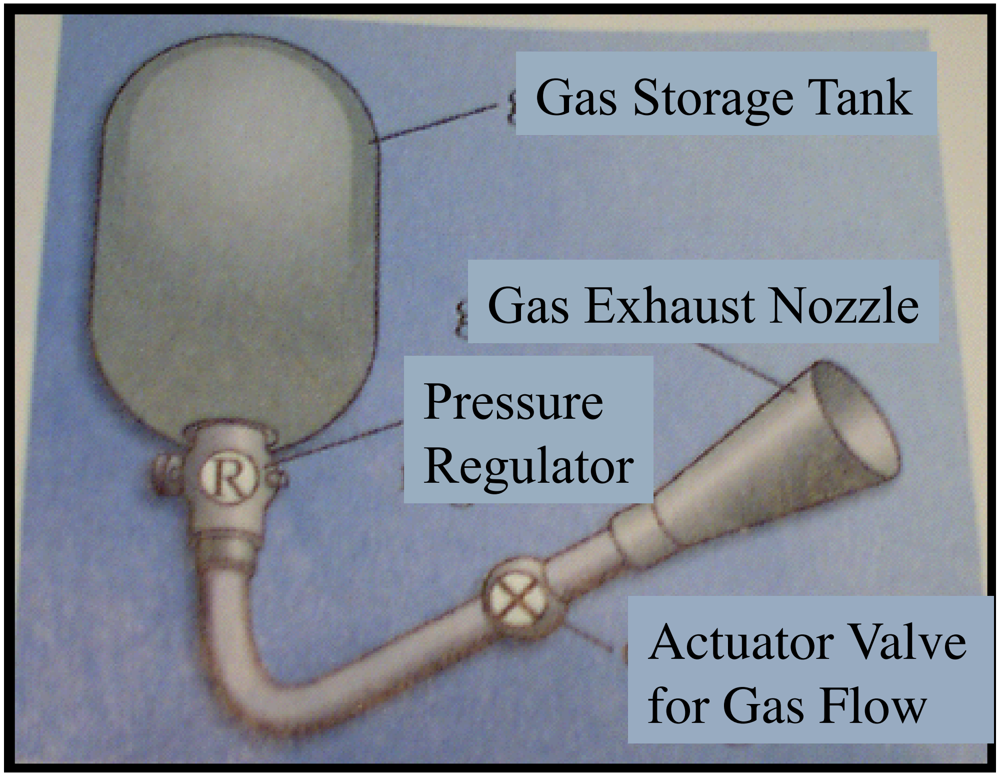
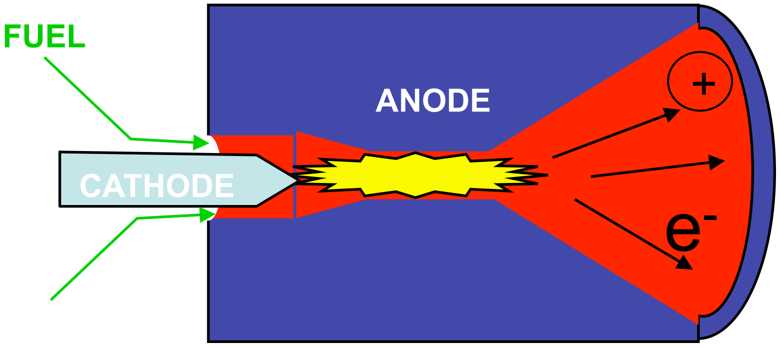
Types of Propulsion Systems: A Quick Overview



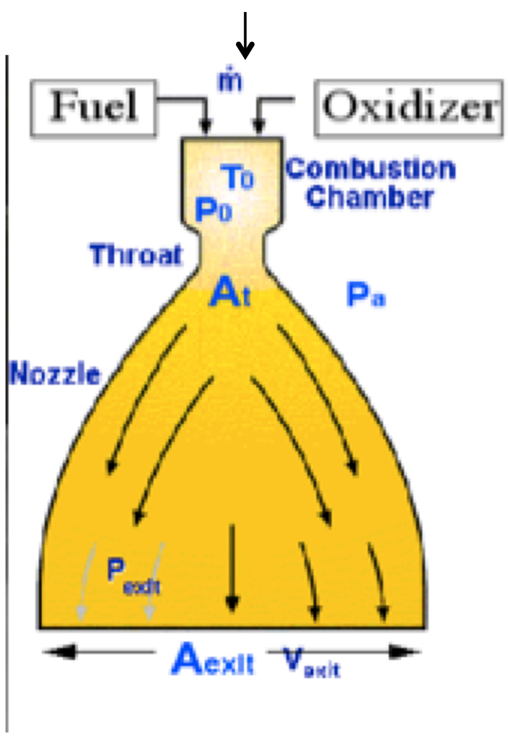


TAYLOR CHAPTER 5

Review: Thrust Equation

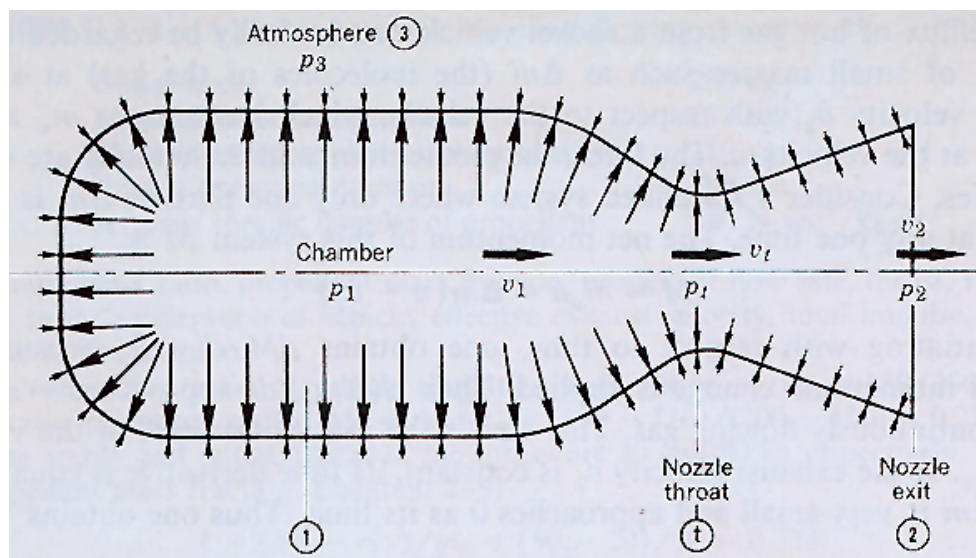
$$\boxed{F = \dot{m}_e V_e + \left(p_e A_e - p_\infty A_e\right)}$$

$$\dot{m}_i = 0$$



• Thrust + Oxidizer enters combustion Chamber at ~0 velocity, combustion Adds energy … High Chamber pressure Accelerates flow through Nozzle

Resultant pressure forces produce thrust



Review: Specific Impulse

$$I_{sp} = \frac{\mathrm{I}_{mpulse}}{g_0 M_{propellant}} = \frac{\int_0^t F_{thrust} dt}{g_0 \int_0^t \dot{m}_{propellant}\, dt} = \frac{F_{thrust}}{g_0 \dot{m}_{propellant}} \rightarrow$$

$$I_{sp} = \frac{1}{g_0}\left[V_e + \frac{p_e A_e - p_\infty A_e}{\dot{m}_e}\right] \equiv \frac{C_e}{g_0}$$

"Units ~ seconds"

- *Effective Exhaust Velocity*

Specific Impulse (Revisited)

Typical I_{sp}'s

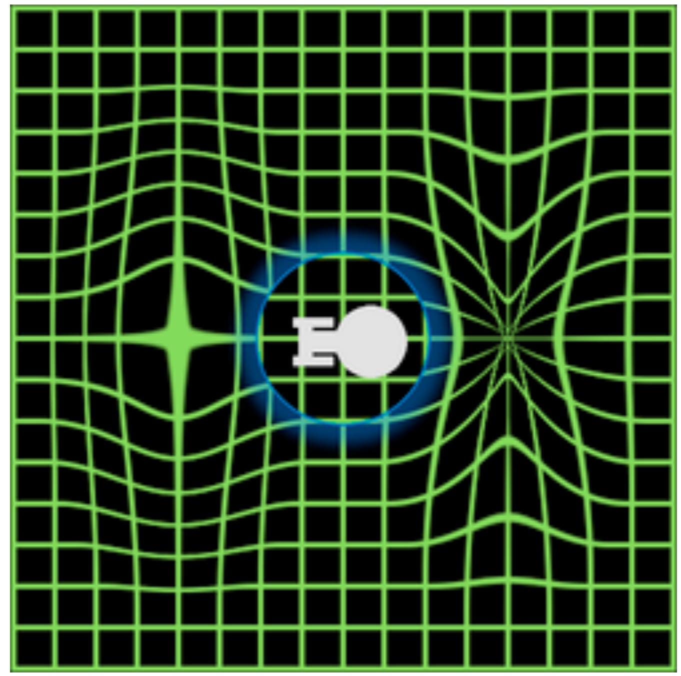
Cryogenic:
400 to 440 seconds

Hypergolics:
260 to 290 seconds

Electric (Ion):
2,500-10,000 seconds

Nuclear:
10^2 to 10^3 seconds

Antimatter:
10^7 seconds



Warp Speed!

Required Delta V Summary

Velocity increment ΔV

- **Required Velocity Change ≡ Velocity increment Δv (m/s) of launch/space maneuvers.**

  (Change in velocity imparted to the launcher/spacecraft by the propulsion system to complete space maneuvers; data listed are indicative only)

START	DESTINATION	TYPICAL ΔV REQUIREMENT m/s	REMARKS
Kourou (French Guiana)	LEO GTO	9 300 11 443	Equatorial (Δv-gain by earth rotation: 465 m/s)
Cap Canaveral (USA)	LEO GEO	9 500 13 600	Equatorial
LEO	GEO	4 260	Change of inclination: 28°, Cap Canaveral
GTO	GEO	1 500 1 800	Change of inclination: 9°, Kourou Change of inclination: 28°, Cap Canaveral
GEO →	GEO	North/South station-keeping: ≈ 50 /year East/West station-keeping: ≈ 3 –6 /year Attitude control: ≈ 3% of total propellant budget	On orbit operations (orbit maintenance requirements per year)
LEO (parking)	Earth orbit escape Lunar orbit Mars orbit	3 200 3 900 5 700	Into planetary trajectory

Legend: LEO = Low Earth Orbit (≈300 km)
GTO = Geostationary Transfer Orbit (Apogee: ≈36 000 km; Perigee: ≈200 km)
GEO = Geostationary Earth Orbit (≈36 000 km)

Types of Propulsion Systems

- **Chemical Propulsion Systems**

  The energy to produce thrust is stored in the propellant, which is released by chemical reactions and the propellant is then accelerated to a high velocity by expanding it in form of gas through a nozzle.

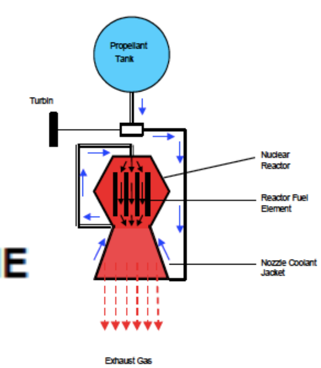
- **Electric Propulsion Systems**

  The energy to produce thrust is not stored in the propellant but has to be supplied from outside by an extra power source, e.g. nuclear, solar radiation receivers or batteries.

  Thrust is produced by:

  - **expansion of hot gas (which is heated by electric current)**
  - **accelerating of charged particles in electric or magnetic fields to high expulsion velocities.**

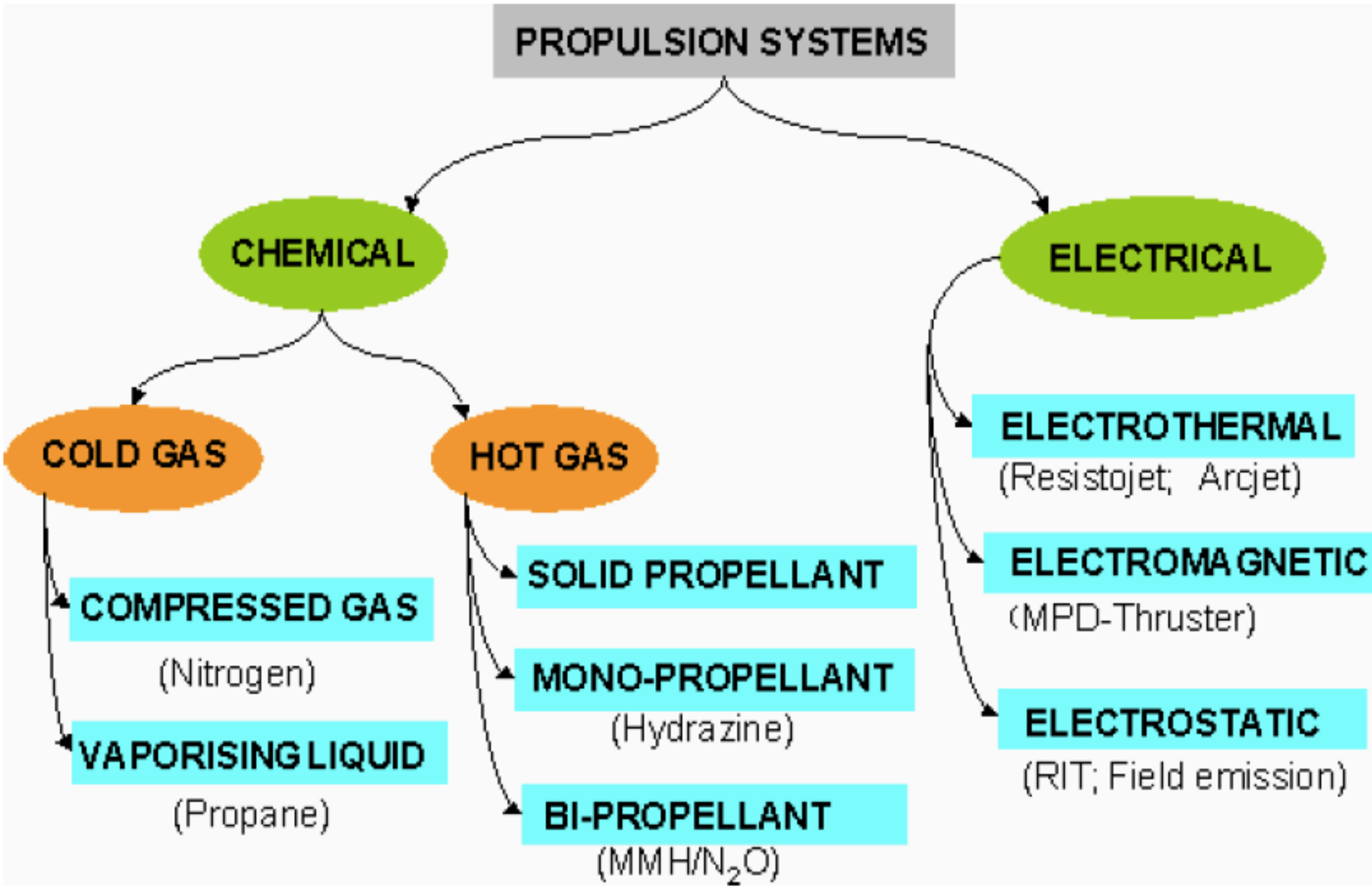
SCHEMATIC OF A THERMO-NUCLEAR ROCKET ENGINE



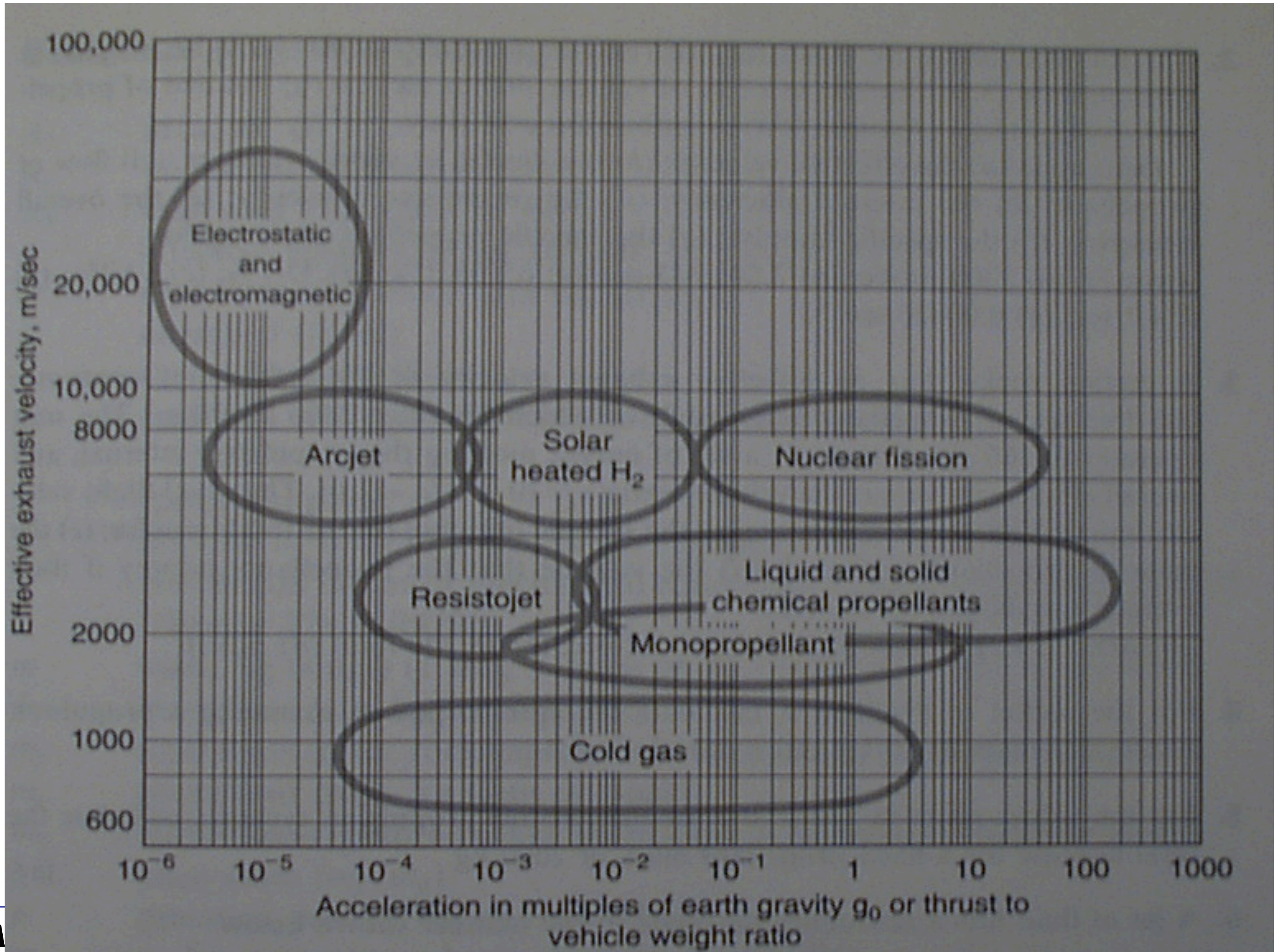
- Nuclear Propulsion .. More on this topic later

SCHEMATIC OF A THERMO-NUCLEAR ROCKET ENGINE

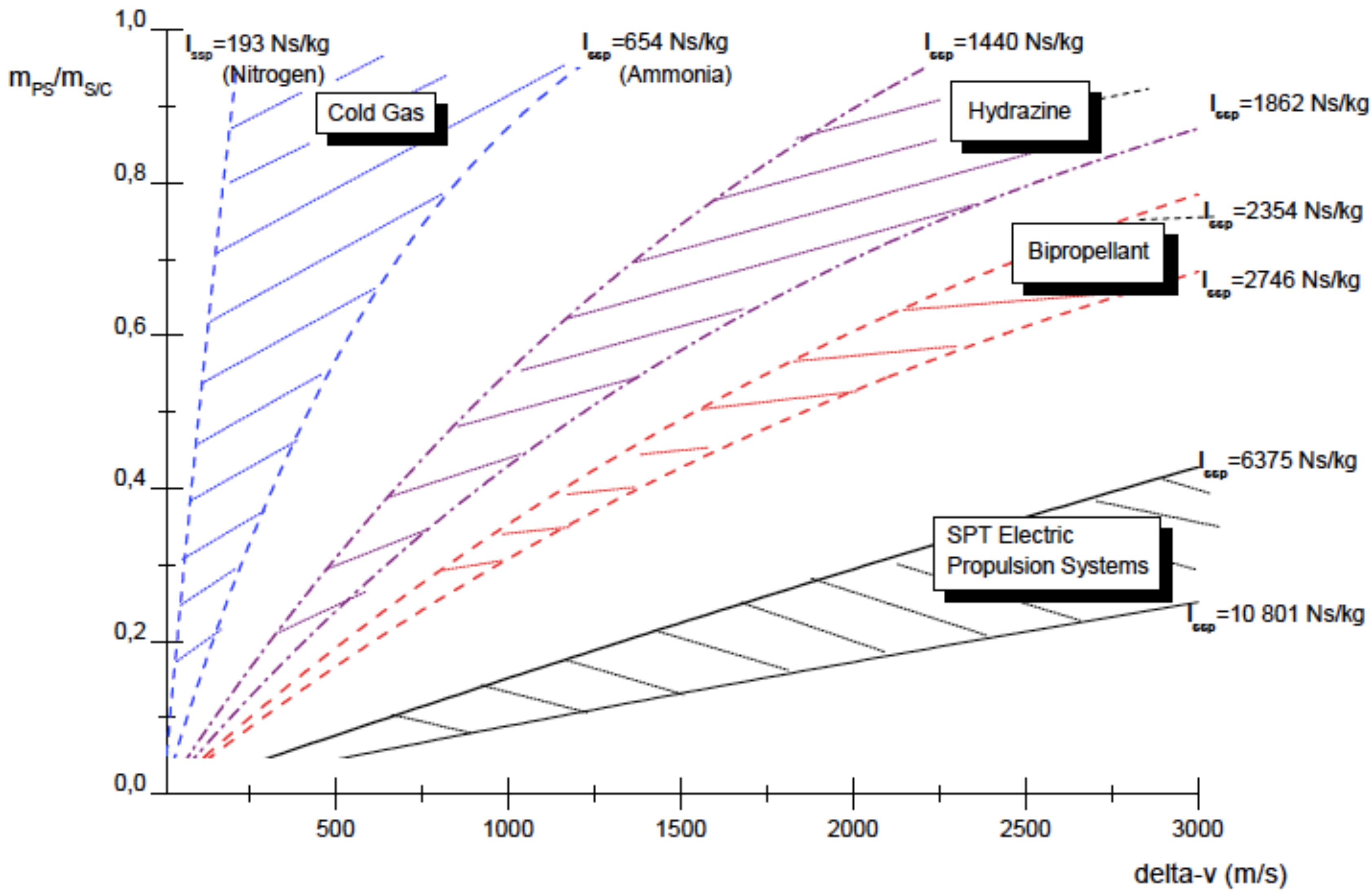
Types of Propulsion Systems



Thrust Vs Isp



Δv-Performance Range of Built Spacecraft Propulsion System Concepts (Examples)



Chemical Propulsion Systems

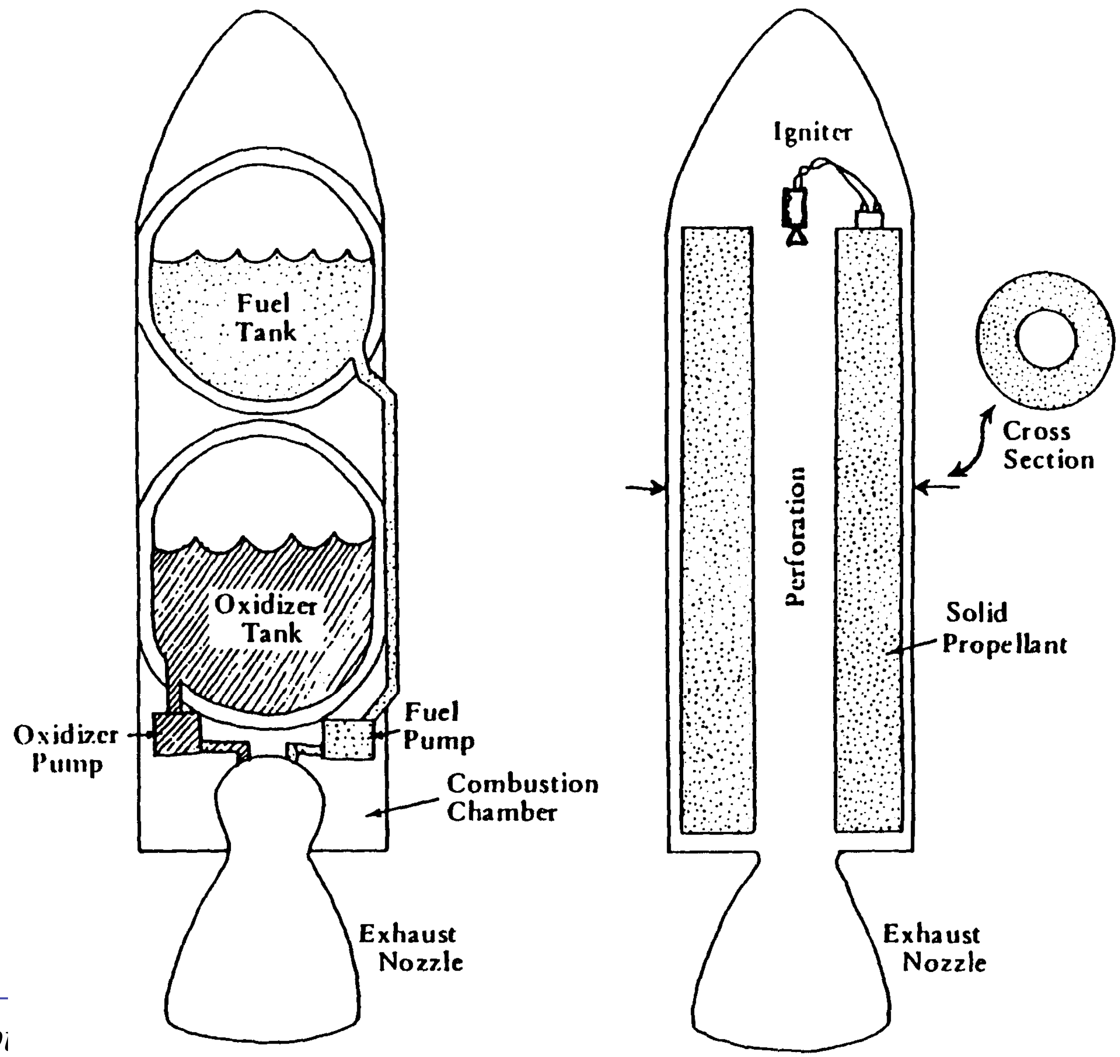
Boosters if they are big, Thrusters if they are small.

Material from Spacecraft Propulsion, Charles D. Brown, ISBN 1-56347-128-0

Desirable Characteristics for Launch Vehicle Stages (Boosters)

- high velocity increment capability (7 - 11.5 km/s)
- very high thrust levels (ratio thrust/launch vehicle weight: >1.3)
- low fraction of take-off mass of launch vehicle for payload mass (1 - 5%) of the launching vehicle
- powerful chemical rockets

Typical Launch Vehicle (Booster) Rocket Stage Systems



Desirable Characteristics for In-Space Propulsion Systems (Thrusters)

In order to fulfill attitude and orbit operational requirements of spacecraft, propulsion systems are characterized in particular by:

MOderate to low

- ~~Very high~~ velocity increment capability (many km/s)
- Low thrust levels (1 mN to 500 N) with low acceleration levels
- Continuous operation mode for orbit control
- Pulsed operation mode for attitude control
- Predictable, accurate and repeatable performance (impulse bits)
- Reliable, leak-free long time operation (storable propellants)
- Minimum and predictable thrust exhaust impingement effects

Desirable Characteristics for In-Space Propulsion Systems (Thrusters) (2)

Impulse bits

- ***Impulse bit*** **is the smallest change in momentum required to allow for e.g. fine attitude and orbit control of a spacecraft.**

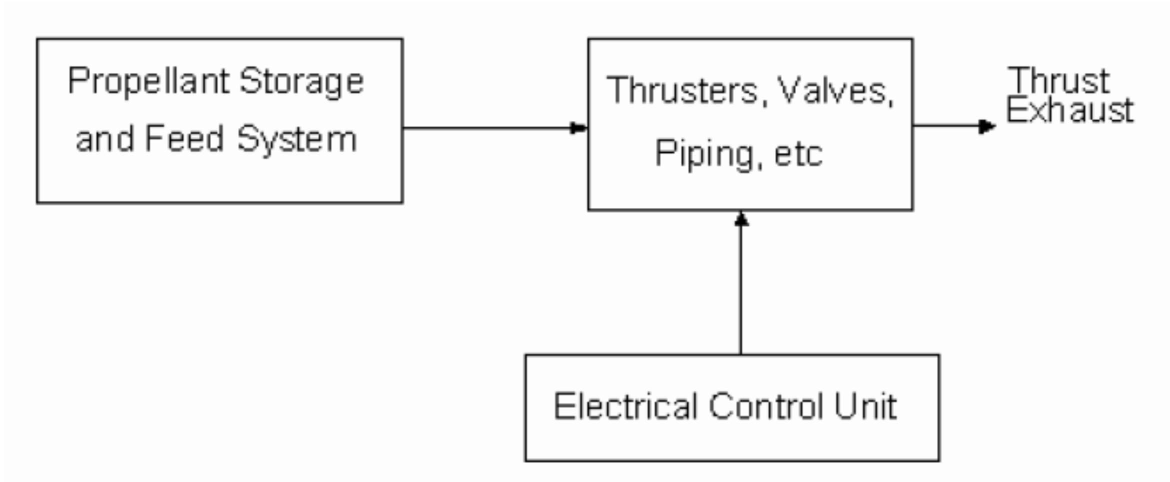
Storable propellants

- **Storable Propellants are liquid (or gaseous) at ambient temperature and can be stored for long periods in sealed tanks, e.g. monopropellant hydrazine (see chapter S1B8C3).**

- **In contrast, cryogenic propellants, which are liquefied gases at low temperature, such as liquid oxygen (-147 °C) or liquid hydrogen (-253 °C) are difficult to be used for long space flight missions.**

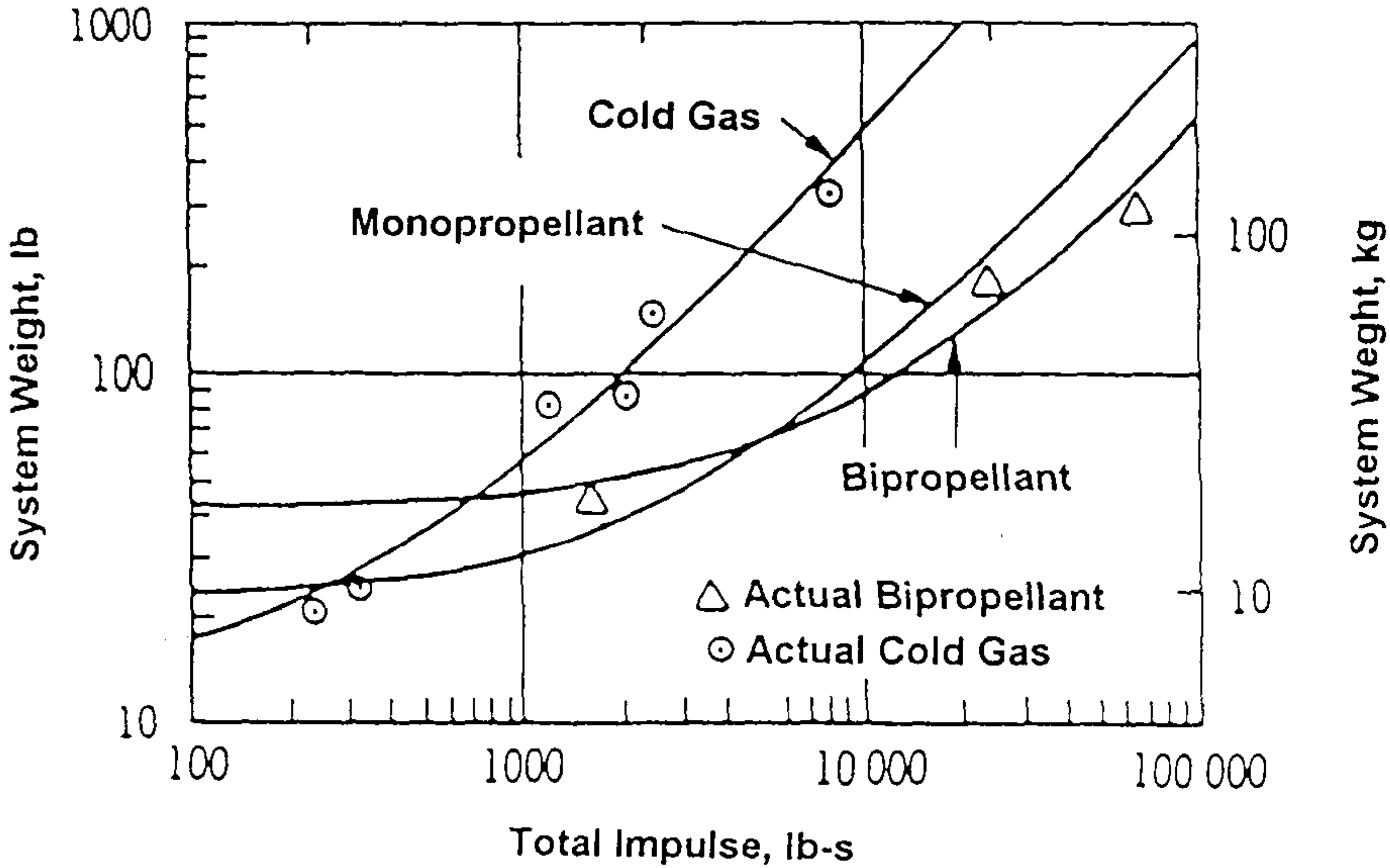
5 Types of Chemical Thrusters

- ***Cold Gas***
- ***Monopropellant***
- ***Bipropellant***
- ***Solid***
- ***Hybrid***



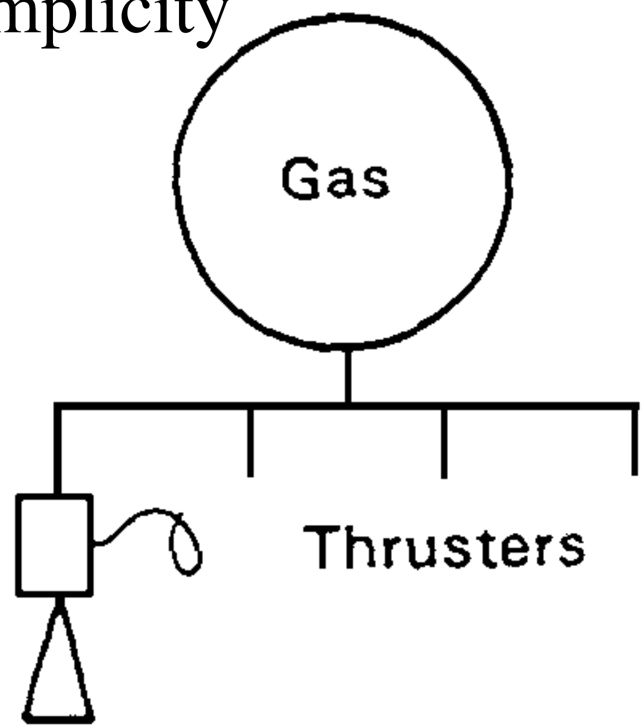
- Chemical Propulsion systems comprise the following main components
  - Storage and feed system that stores and feeds the propellant to the thrusters
  - Valves, piping which connects the propellant storage system with the thruster
  - Electric control unit to operate electrically the valves and thrusters

System Weight Comparison

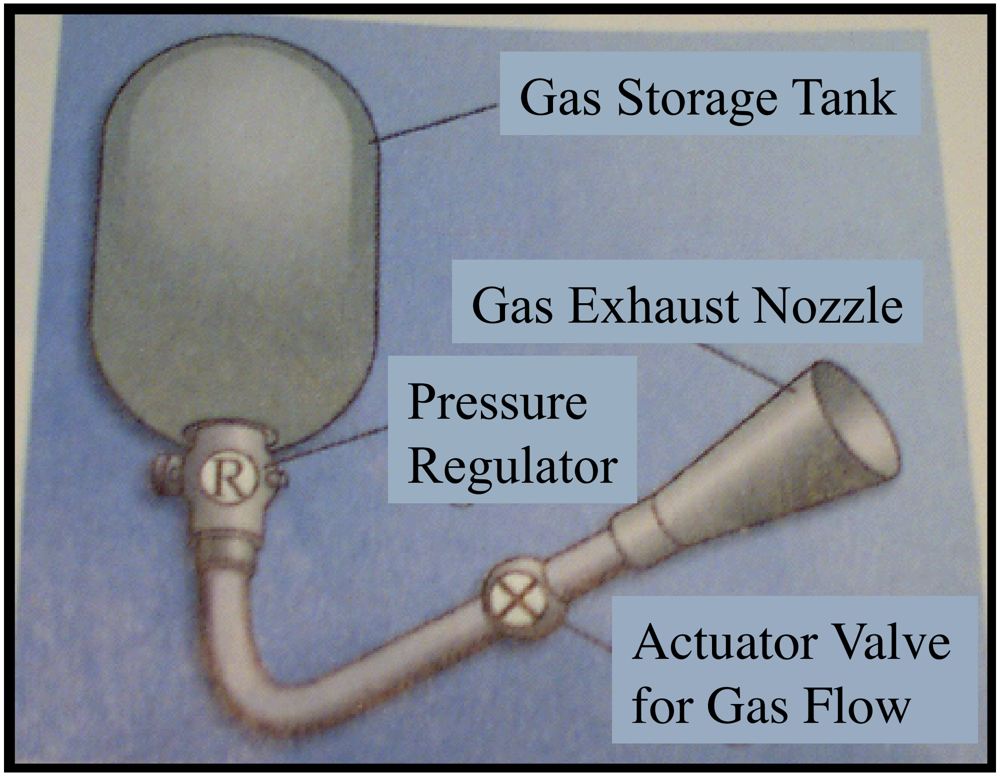


Cold Gas Thrusters

- The balloon model: A big tank of gas, a valve, and a nozzle.
- Used on early satellites for simplicity
- I_{sp} of 50 seconds
- thrust less than a pound



Cold Gas Thrusters



- No Combustion
- Thrust provided by expansion of gas through Nozzle
- Low I_{sp}
- Simple Mechanism

Cold Gas Thrusters (2)

CHARACTERISTICS OF SOME CANDIDATE COLD GAS PROPELLANTS

Name	Formula	Mol. Mass M (kg/kmol)	Crit. Temp (°C)	Crit. Pressure (bar)	Compr. Factor z (200 bar, 20°C)	Exhaust Velocity v_e [1] (20°C) (m/s)	Remarks 'System-spec Impulse' I_{ssp} (Ns/kg ≡ m/s)
Compressed Gas							
Hydrogen	H_2	2	- 240	20	1.18	2668	≈ 194 [6]
Methane	CH_4	16	-83	46	0.86	1030	≈ 454 [6]
Nitrogen	N_2	28	-147	33	1.13	706	≈ 378 [6]
Argon	A	39.9	-122	49	1.02	490	≈ 319 [6]
Freon 14	CF_4	88	-45	41	0.68 [2]	441	[2] at 110 bar, 20°C; ≈ 377
Xenon	Xe	131.3	+17	58	0.3 [3]	275	[3] at 75 bar, 25°C; ≈ 262
Vaporising Liquids		Density ρ $(kg/m^3)10^3$			Max. Oper. Press. P_{op} (30°C) (bar)		
Ammonia	NH_3	0.62	+132	119	12	950	≈ 700 [7]
Propane	C_3H_8	0.53	+97	42	11	608 [4]	[4] X-4, UK-5 S/C; ≈ 486 [7]
Carbon dioxide	CO_2	0.8	+31	73	71	598 [5]	≈ 340 [7]

[1] Holcomb, L.B.; Satellite auxiliary-propulsion selection techniques. JPL Technical Report 32-1505, 1970

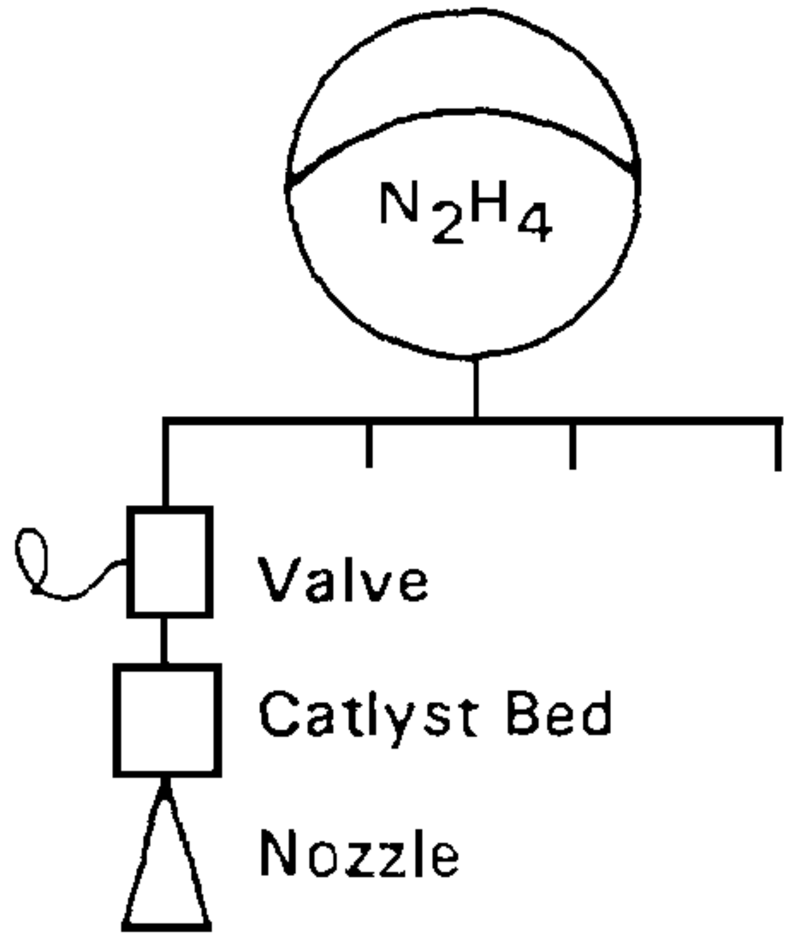
[5] Sackheim, R.L. et.al; The Next Generation of Spacecraft Propulsion Systems, 15th Joint Propulsion Conference June 1979

[6] for gas storage tanks made of fibre composite material

[7] for gas storage tank made of aluminium with heat exchanger

Monopropellant systems

- Often used for spacecraft RCS system



Monopropellant Thrusters

- An unstable chemical that will decompose exothermically in the presence of a catalyst.
- The chemical needs to be unstable, but not too unstable.
- V2 used hydrogen peroxide, but it decomposes in storage, leading to overpressures and water.
- Current systems use Hydrazine, which decomposes into Hydrogen, nitrogen, and ammonia in the presence of iridium. I_{sp} is on the order of 230, and total thrust can reach hundreds of lbs.

Monopropellant Thrusters (2)

- The hydrazine propellant is decomposed in a thruster by a catalyst and the resulting hot gas is expelled through a nozzle, thus generating thrust force on the spacecraft.
- A typical monopropellant system uses nitrogen or helium gas to expel the propellant from a diaphragm tank into the chamber catalyst beds of the thrusters. Since the pressuring gas is stored (at a pre-selected but relatively low pressure, e.g. 22 bar) in the propellant tank, the propellant pressure varies with propellant usage.
- A typical selection of the ullage volume of 25% filled with pressuring gas (thus containing 75% propellant) will results in a propellant feed pressure decay, and thus in a thrust decay of 4:1.
- This mode of operation is also referred to as the ***blow-down mode***, in contrast to the ***pressure constant mode***, which requires the storage of a high-pressure gas in a tank external to the propellant tank (see bipropellant systems).
- In a hydrazine gas generator system, the hydrazine decomposition gases are exhausted into a gas storage tank for later gas expulsion.
- The catalytic thruster and gas generator systems have identical propellant feed systems consisting typically of propellant tank(s) with a diaphragm expulsion device(s), propellant and gas fill valves, eventually latch valves (start valves), line pressure transducers and filters.

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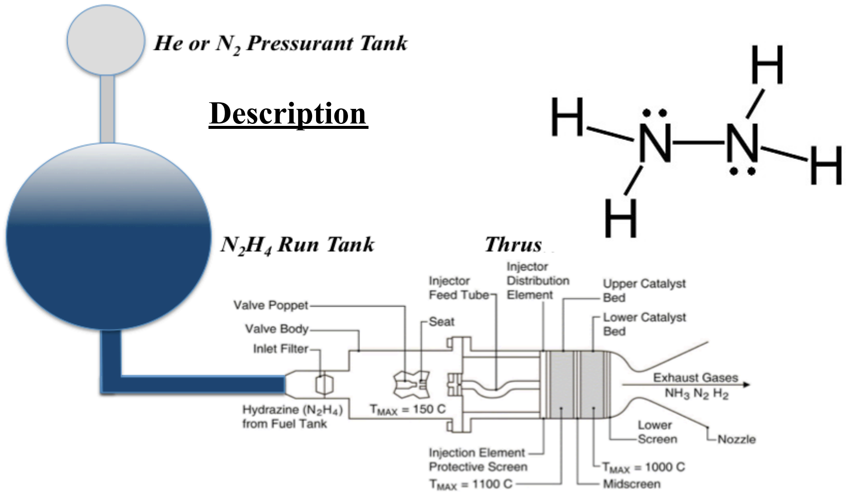
Description



Hydrazine is by far the most commonly used monopropellant for primary spacecraft propulsion and attitude control thrusters.

Mission Applications/Benefits

Monopropellant thrusters using hydrazine (N_2H_4) are commonly used on a variety of spacecraft and satellites. However, hydrazine is a highly toxic and dangerously unstable substance. Although procedures are in place to allow hydrazine to be managed safely on tightly controlled military and NASA-owned flight experiments; the toxicity and explosion potential of hydrazine requires extreme handling precautions. Increasingly, with a growing regulatory burden, infrastructure requirements associated with hydrazine transport, storage, servicing, and clean up of accidental releases are becoming cost prohibitive. The use of hydrazine as a propellant has been compromised by stringent laws to protect personnel who have to work with substances which are highly toxic and carcinogenic.

Recently, much more benign, low toxicity (“green”) storable liquid propellants have attracted significant attention as possible replacements for hydrazine.

Subsystem State of the Art

Hydrazine is a very effective and reliable monopropellant. Hydrazine thrusters are simple, versatile, and dependable. In monopropellant form hydrazine has a low vapor pressure and requires a “top pressurant” for rocketry use. A typical arrangement will include a pressurant tank, propellant tank, electric solenoid or pneumatic valve, catalyst bed, post-combustion chamber, and exit nozzle. The industry standard catalyst bet is composed of alumina pellets impregnated with iridium. The catalytic material was originally commercially developed under the brand name Shell-405® (Shell Chemical, Houston, TX.) In 2002 commercial production shifted from Shell Chemical to the Aerospace Corporation, Redmond WA. The product is now commercially available as Aerojet S-405.

Technical/Development Hurdles

Hydrazine is a highly energetic and is listed by the US DOT as a shock-sensitive chemical prone explode when struck, vibrated, or otherwise agitated. Material compatibility is a critical concern. Hydrazine spontaneously explodes upon contact with calcium oxide, barium oxide, iron oxides, copper oxide, chromate salts, and many others.

Hydrazine poses serious biological environmental concerns and is extremely destructive to living tissues, and a known human carcinogen. Dimethyl-hydrazine, has caused colon cancer in laboratory animals following a single exposure. Hydrazine is corrosive to eyes, skin, and mucous membranes. Caustic burns to the skin are the immediate result of contact with the liquid. Exposure produces a variety of adverse systemic effects including damage to liver, kidneys, nervous system, and red blood cells.

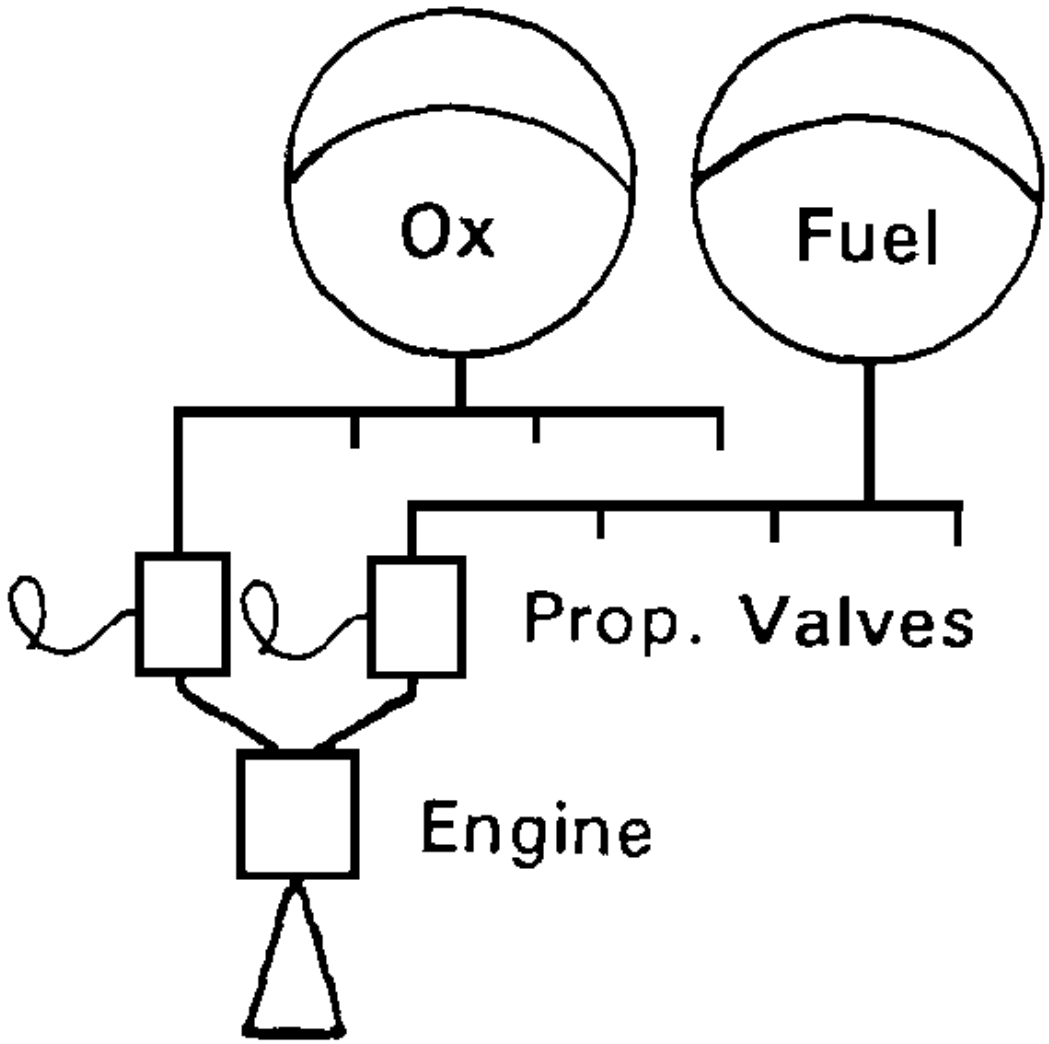
is

Recommendations

With a growing regulatory burden, infrastructure requirements associated with hydrazine transport, storage, servicing, and clean up of accidental releases are becoming cost prohibitive. Extreme handling precautions generally do not favor hydrazine as a propellant for secondary payloads. A non-toxic, stable propellant alternative for hydrazine is clearly desirable. NASA should discourage the use of, and gradually phase out, hydrazine and replace this propellant with non-toxic propellants, target to the mission application. NASA should aggressively pursue multiple options for hydrazine replacement alternatives.

Hydrazine Monopropellant

Bi-propellant Rocket Systems

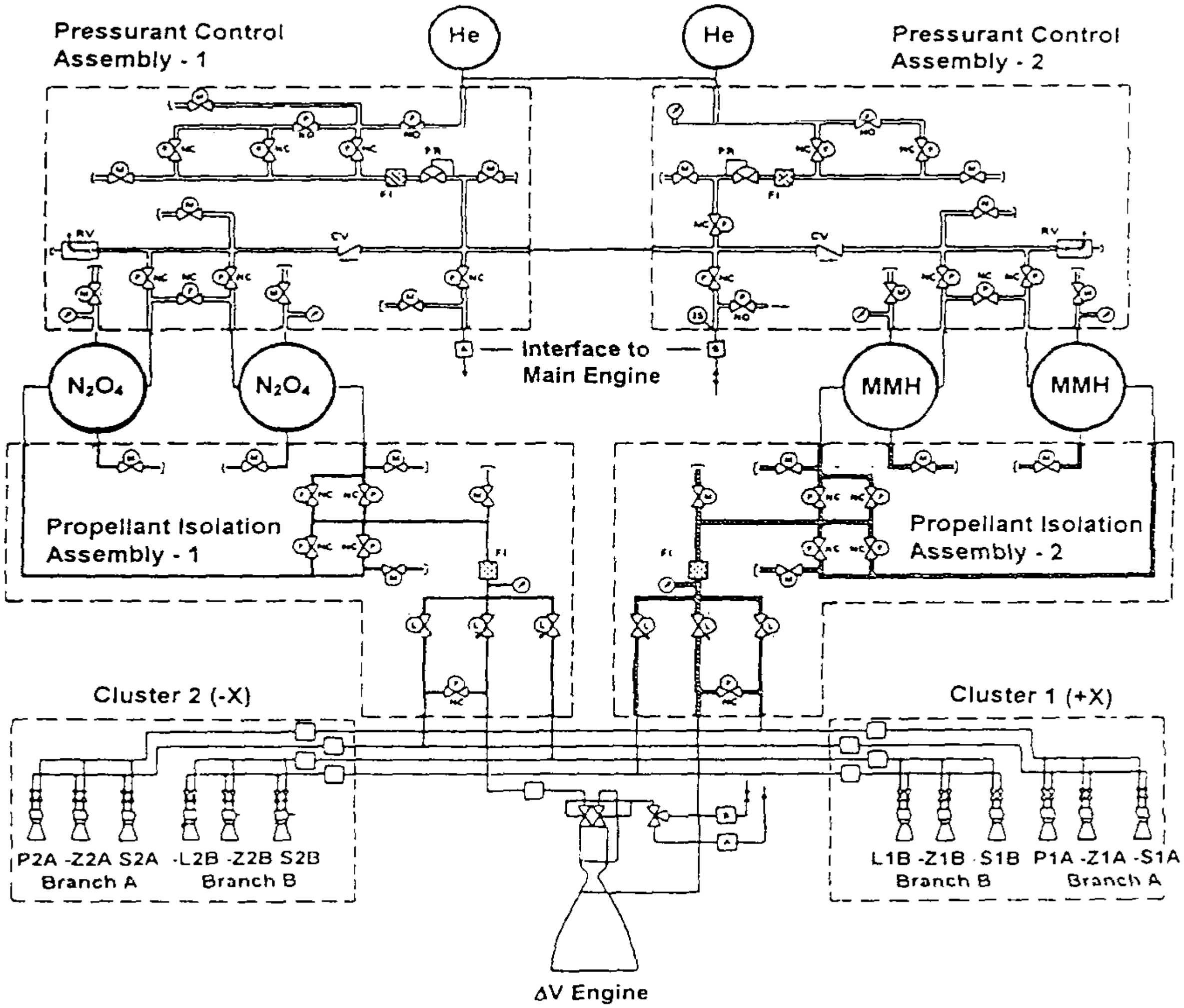


- Bi-prop offers the most performance (I_{sp} as high as 450 sec) and the most versatility.

- They also offer the most failure modes and the highest price tags.

- Almost all first stage liquid rockets are Bi-prop.

Galileo Plumbing and Instrumentation Diagram (P&ID)



Bi-propellant Rocket Systems (2)

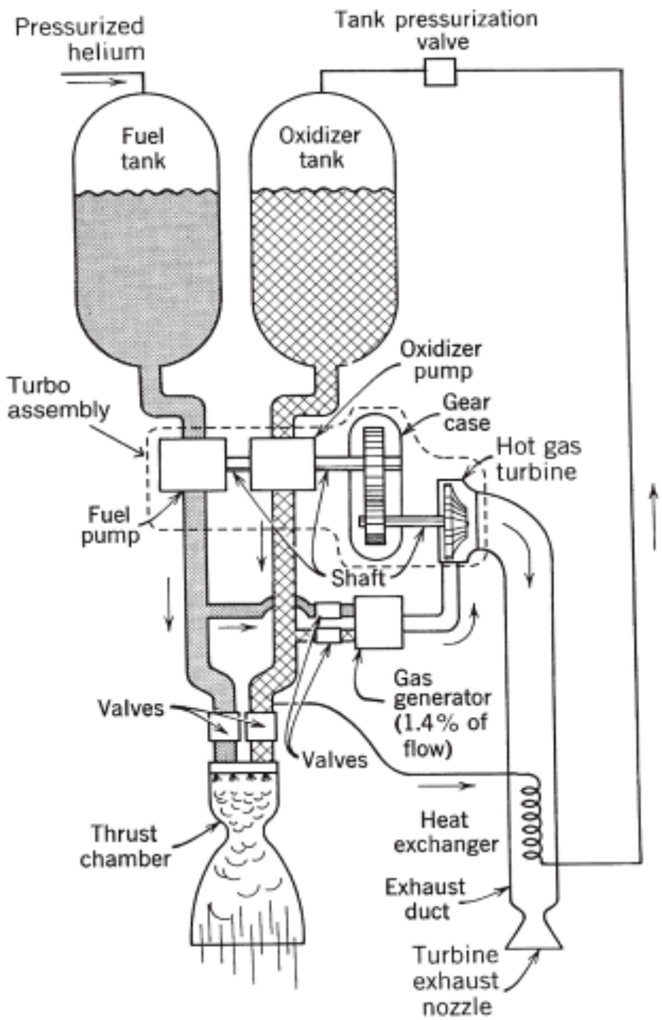
- Bipropellant systems are characterized by the combustion of two (Bi) propellants, a fuel (e.g. MMH) and an oxidizer (e.g. N_2O_4) to produce thrust. The propellants are injected separately into the bipropellant thruster combustion chamber where they react spontaneously (hypergolic propellant) to perform high-temperature, low molecular weight combustion products, which are the expelled through a nozzle.

- The system basically consists of a pressurizing-gas system, propellant tanks (with surface tension propellant management devices), propellant lines and thrusters. Unlike hydrazine thrusters, bipropellant thrusters accept only a limited range of propellant inlet pressure variation of ≤ 2. Therefore, the high-pressure gas, generally nitrogen or helium is regulated to the desired tank pressure, e.g. 17 bar. This mode of operation is also referred to as the pressure constant mode.

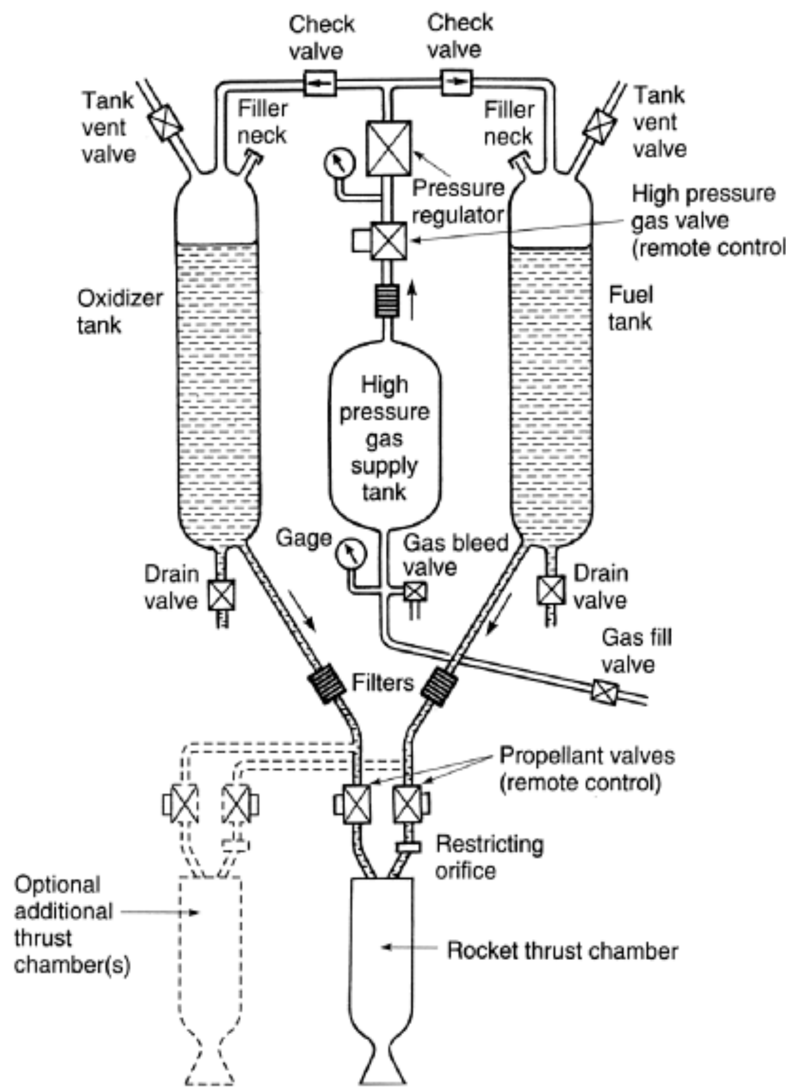
- The system contains check valves upstream of the propellant tanks to prevent possible back-flow, mixing, and combustion of the propellant vapors in the common pressuring gas line. Relieve valves are incorporated in the system upstream of the propellant tanks to prevent system rupture in the event of a pressure regulator failure. Filters are provided in the propellant lines directly upstream of the thruster valves to prevent clogging of the injector or damage of the valve seat by entrained foreign material. Finally, the system contains pyro- or latch valves, line pressure transducers, fill and drain valves and various test ports for system check out.

Bi-Prop Combustion Cycle Alternatives

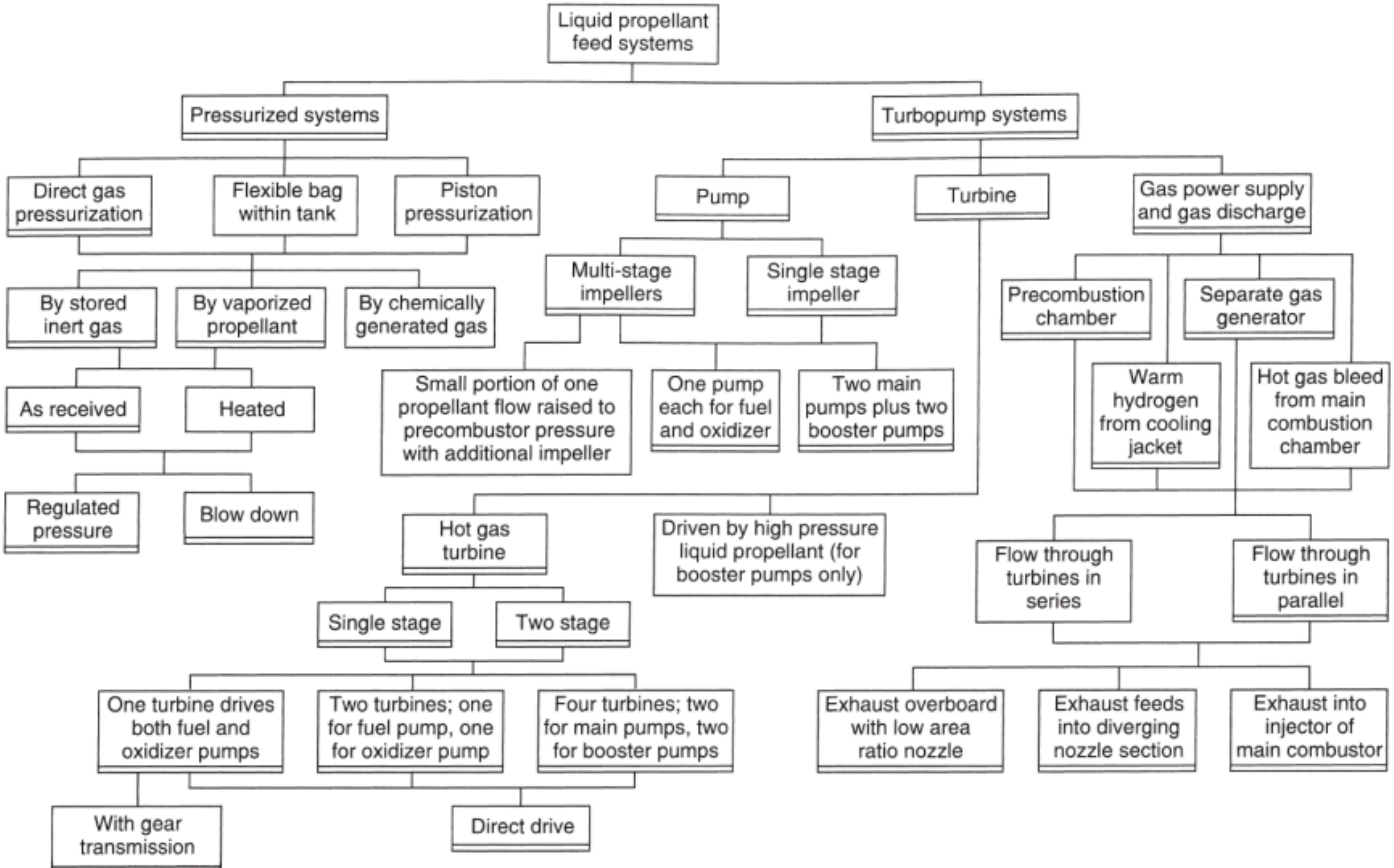
- Turbine Fed Bi-Prop System (Gas generation cycle)



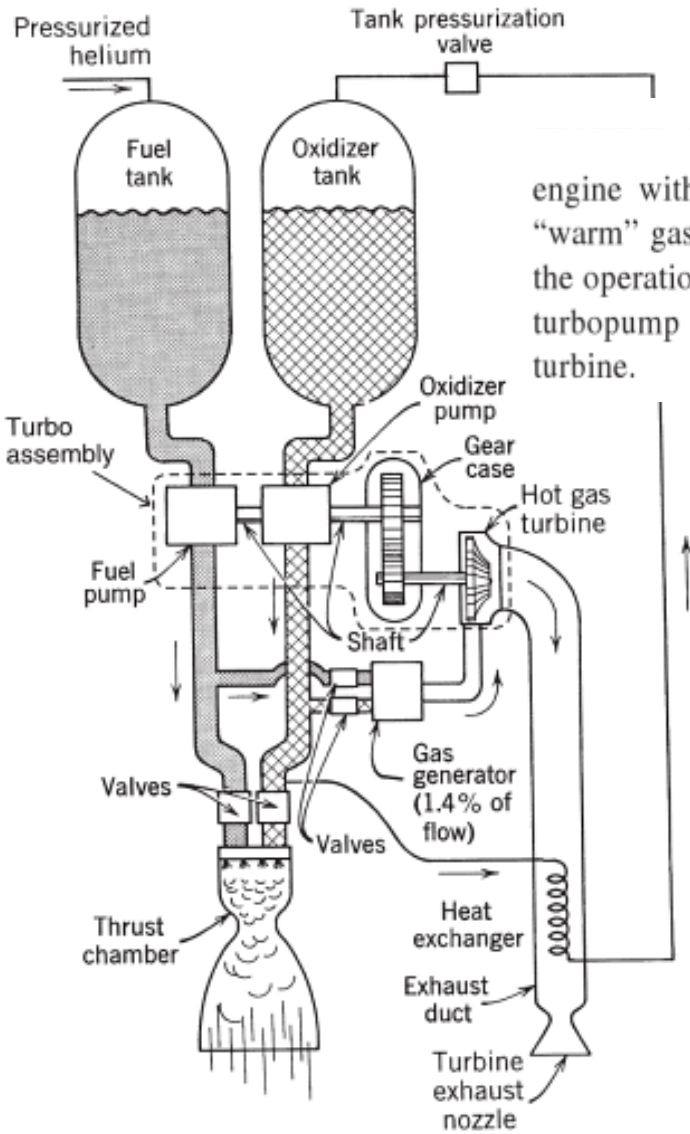
- Pressure Fed Bi-Prop System



Bi-Prop Combustion Cycle Alternatives



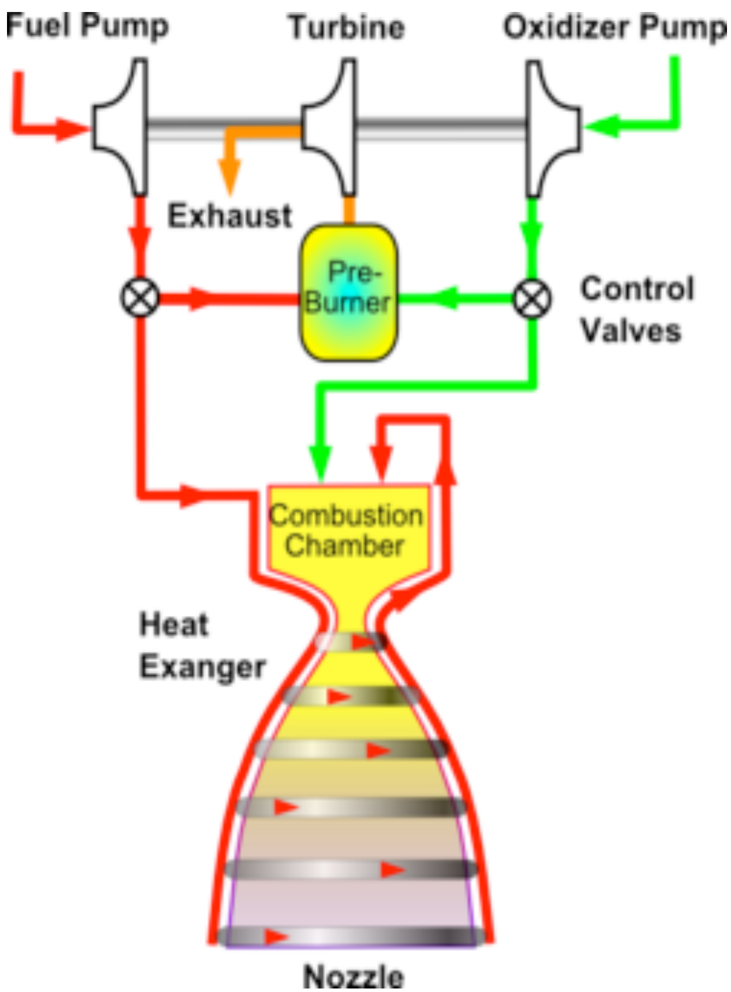
Gas Generation Cycles for BiProps



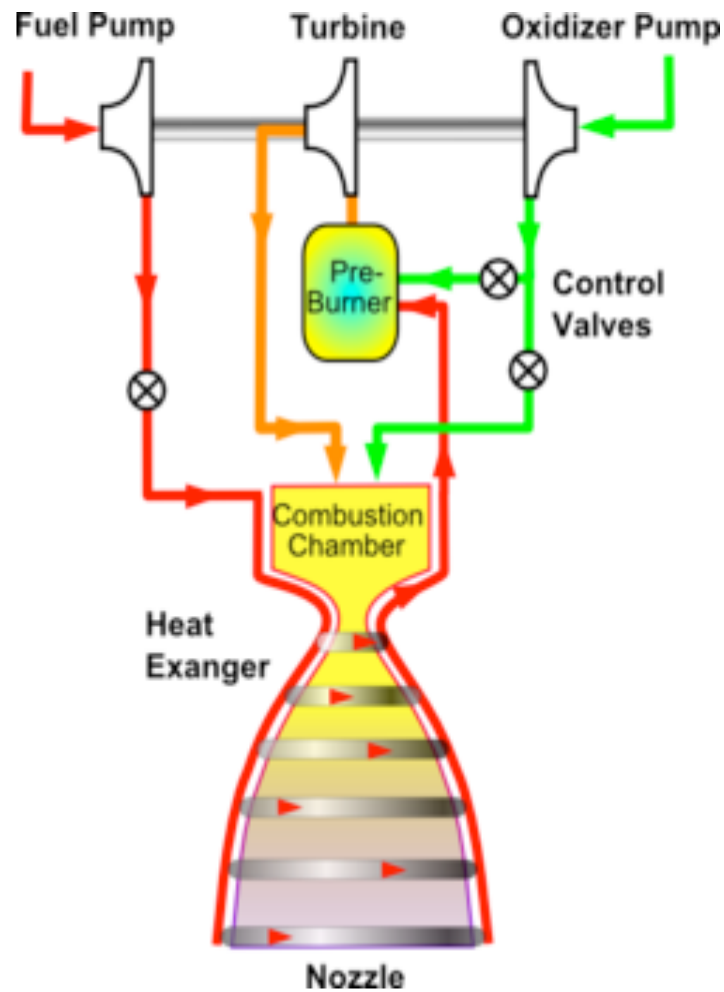
Simplified schematic diagram of one type of liquid propellant rocket engine with a turbopump feed system and a separate gas generator, which generates "warm" gas for driving the turbine. Not shown are components necessary for controlling the operation, filling, venting, draining, or flushing out propellants, filters or sensors. This turbopump assembly consists of two propellant pumps, a gear case, and a high speed turbine.

Credit: Sutton and Biblarz

Gas Generation Cycles for BiProps

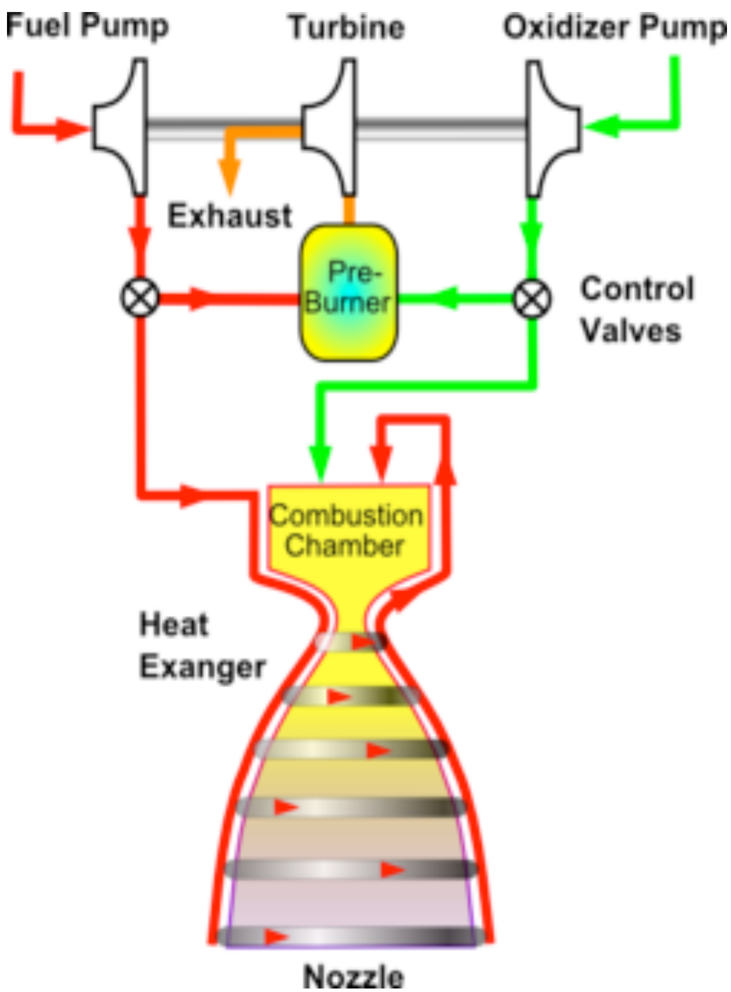


Open Gas Generator Combustion Cycle.



Staged Combustion (Closed) Cycle.

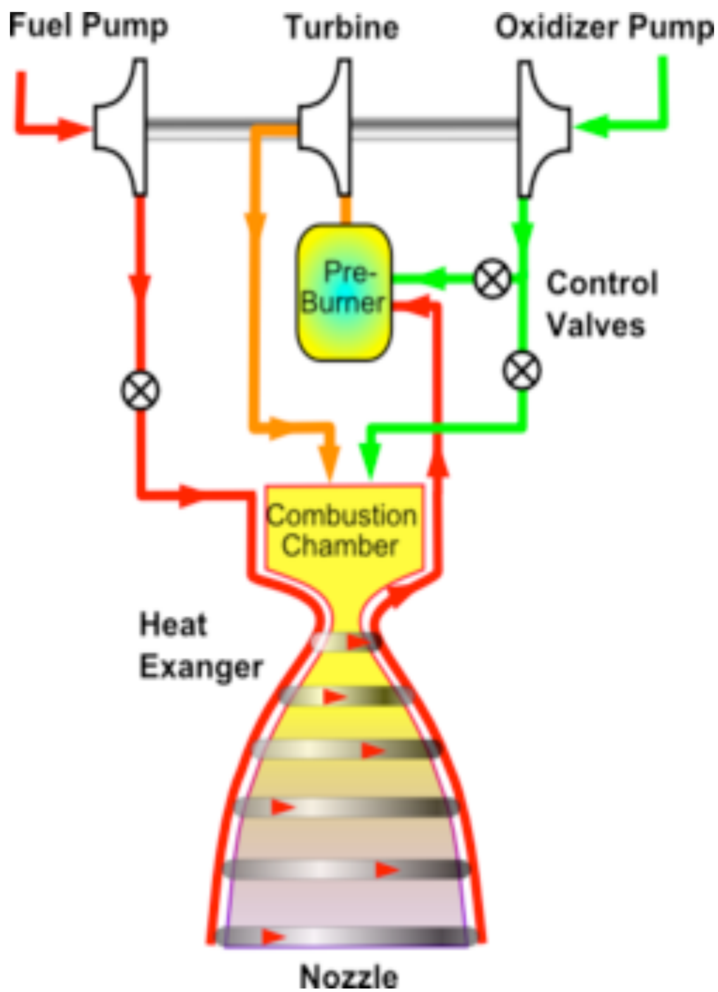
Gas Generation Cycles for BiProps (2)



Open Gas Generator Combustion Cycle.

- Fuel and oxidizer is burned separately to power the turbo-pumps and then discarded.
- Most gas-generator engines use the fuel for nozzle cooling.
- Complex design, but delivers very precise fuel/oxidizer flow rates
- Can be throttled significantly
- Loss in I_{sp} due to discarded propellant.

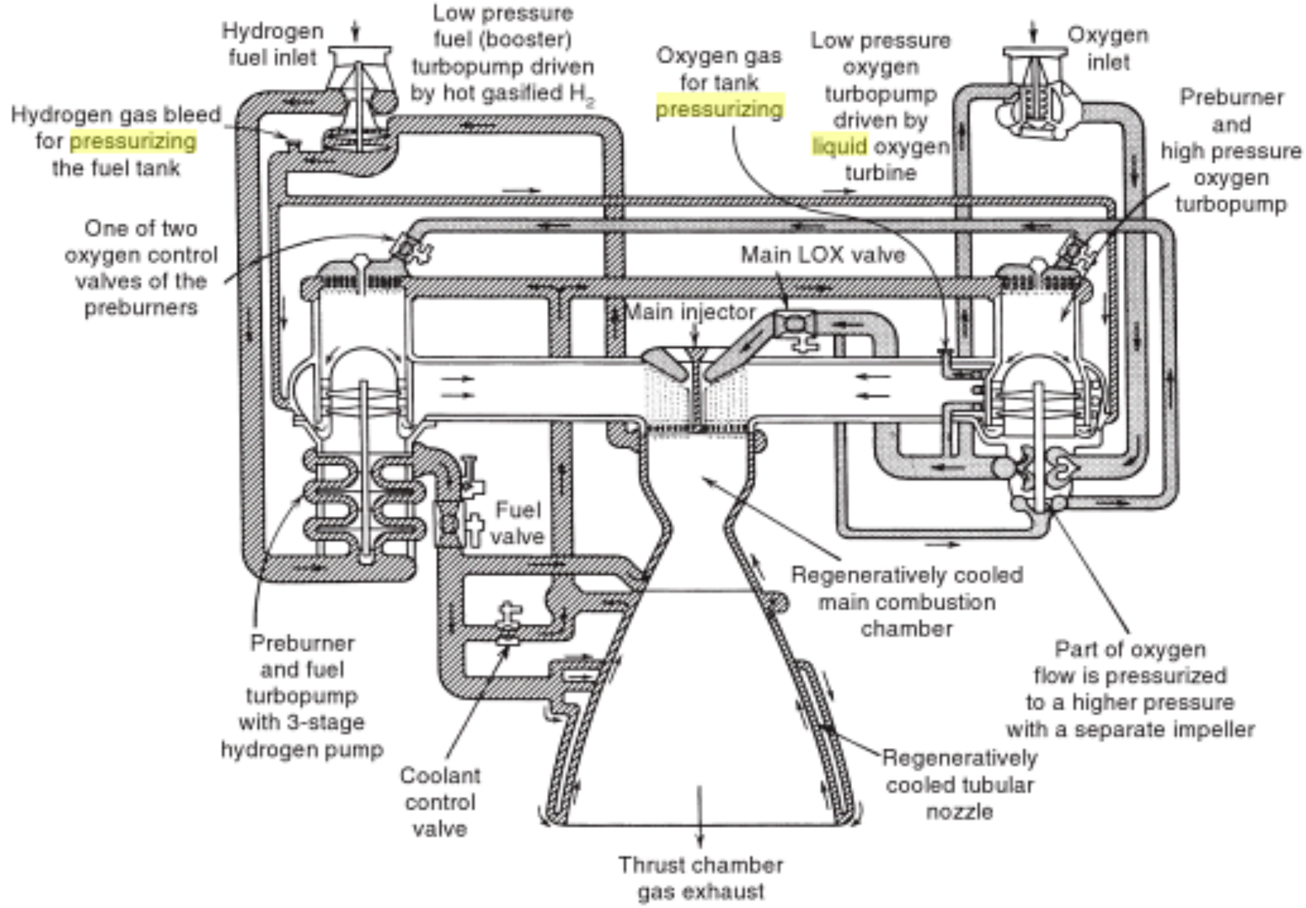
Gas Generation Cycles for BiProps (2)



Staged Combustion (Closed) Cycle.

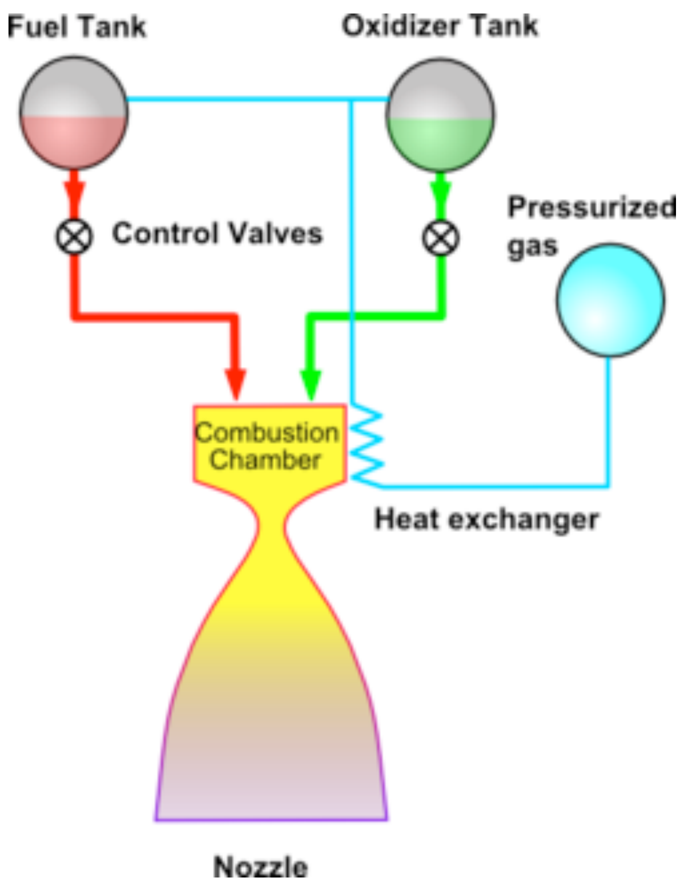
- All fuel and portion of the oxidizer are fed through the fuel-rich pre-burner (fuel rich) to generate gas for pumps.
- Because of fuel rich mixture, Preburned gas is recovered and remaining fuel is burned in chamber
- Significantly higher Isp, more complex plumbing than open cycle
- Delivers very precise fuel/oxidizer flow rates and can be throttled significantly
- Combustion rich gases generally corrosive

Gas Generator Bi-Prop System Components



Flow diagram illustrating the staged combustion cycle of the Space Shuttle Main Engine (SSME) using liquid oxygen and liquid hydrogen fuel. (Courtesy of Pratt & Whitney Rocketdyne and NASA.)

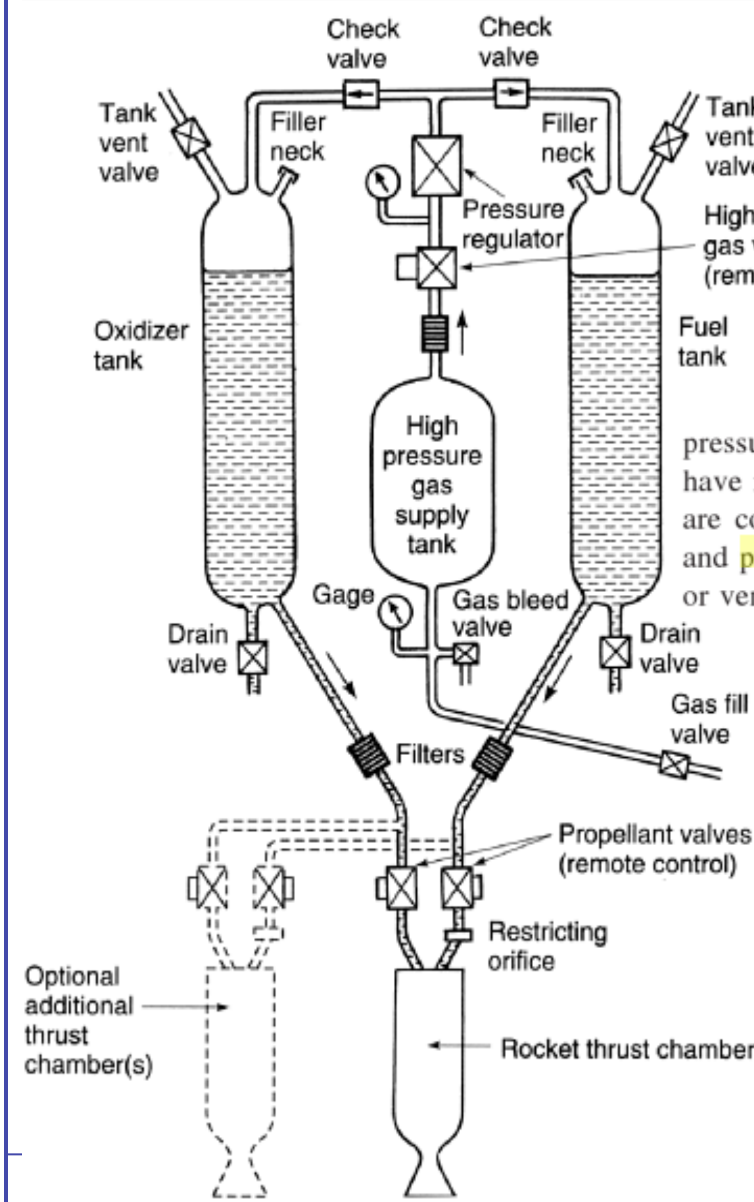
Pressure-Fed Bi-Prop Systems



Pressure fed Bi-Prop System

- Pressurized gas used to control feed of propellant through system
- Heat exchanger often required to keep pressurant gas from dropping pressure
- Significantly less complex system
- Higher degree of variability in system flow rates
- System cannot be throttled
- Pressurant gas limits flow rates

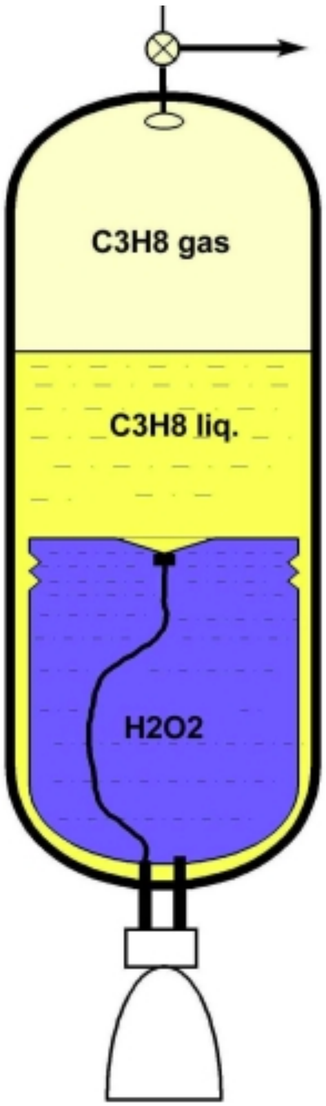
Pressure-Fed Bi-Prop Systems



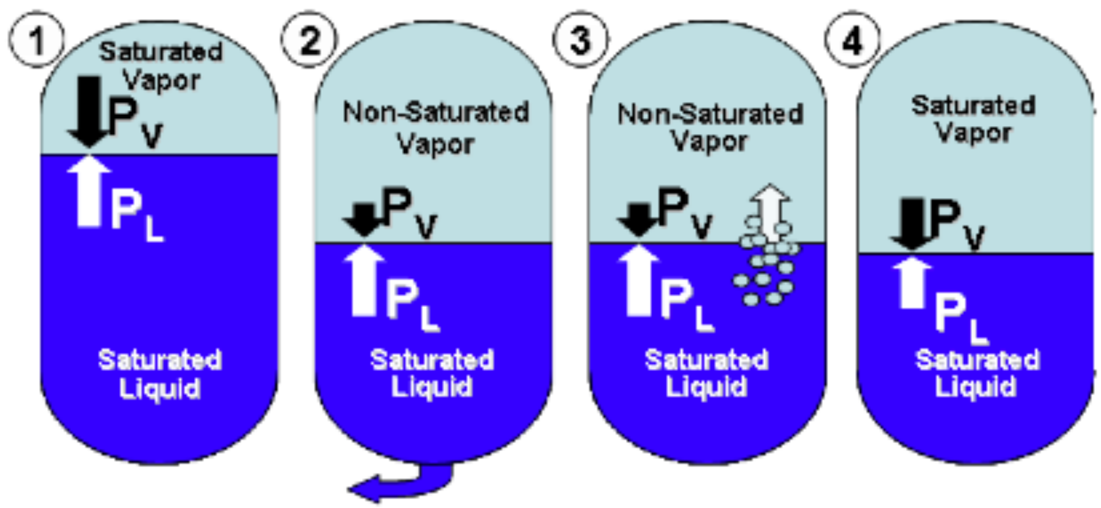
. Schematic flow diagram of a liquid propellant rocket engine with a gas pressure feed system. The dashed lines show a second thrust chamber, but some engines have more than a dozen thrust chambers supplied by the same feed system. Also shown are components needed for start and stop, controlling tank pressure, filling propellants and pressurizing gas, draining or flushing out remaining propellants, tank pressure relief or venting, and several sensors.

Credit: Sutton and Biblarz

Self-Pressurized Bi-Prop Systems



- Relies on one or more saturated liquid propellants with high natural vapor pressure
- Sometimes referred to as "Blowdown Systems"
- Extremely Simple System Plumbing
- Highly Variable massflows
- Potentially dangerous if propellant blow back occurs



1. $P_L=P_V$ (fluids are in equilibrium)
2. Liquid is removed, vapor expands, $P_L > P_V$
3. Liquid boils, repressurizes vapor
4. $P_L=P_V$ (fluids are in equilibrium with $P_4<P_1$, cycle repeats)

TABLE 6–3. Comparison of Two Types of Gas Pressurization Systems

Type	Regulated Pressure	Blowdown
Pressure/thrust	Stays essentially constant	Decreases as propellant is consumed
Gas storage	In separate high-pressure tanks	Gas is stored inside propellant tank with large ullage volume (30–60%)
Required components	Needs regulator, filter, gas valve, and gas tank	Larger, heavier propellant tanks
Advantages	Constant-pressure feed gives essentially constant propellant flow and approximately constant thrust, constant I_s and r Better control of mixture ratio	Simpler system. Less gas required Can be less inert mass
Disadvantages	Slightly more complex	Thrust decreases with burn duration
	Regulator introduces a small pressure drop	Somewhat higher residual propellant due to less accurate mixture ratio control
	Gas stored under high pressure Shorter burning time	Thruster must operate and be stable over wide range of thrust values and modest range of mixture ratio Propellants stored under pressure; slightly lower I_s toward end of burning time

Future Development in Liquid Propellant Technologies

<u>Need</u>:

- **Environmentally friendly, safer propellants.**
  - **Current spacecraft and satellite users and manufacturers are looking for more environmentally friendly, safe propellants. These can reduce cost by eliminating the need for *self-contained atmospheric protective ensemble* (SCAPE) suits that are needed for toxic propellants.**
  - **Moreover, extensive and prohibitive propellant safety precautions, and isolation of the space vehicle from parallel activities during propellant loading operations can be minimized or eliminated. If used on these satellites, the costs for operating the vehicles will be lowered, in some cases dramatically.**

Future Development in Liquid Propellant Technologies (2)

Under Development:

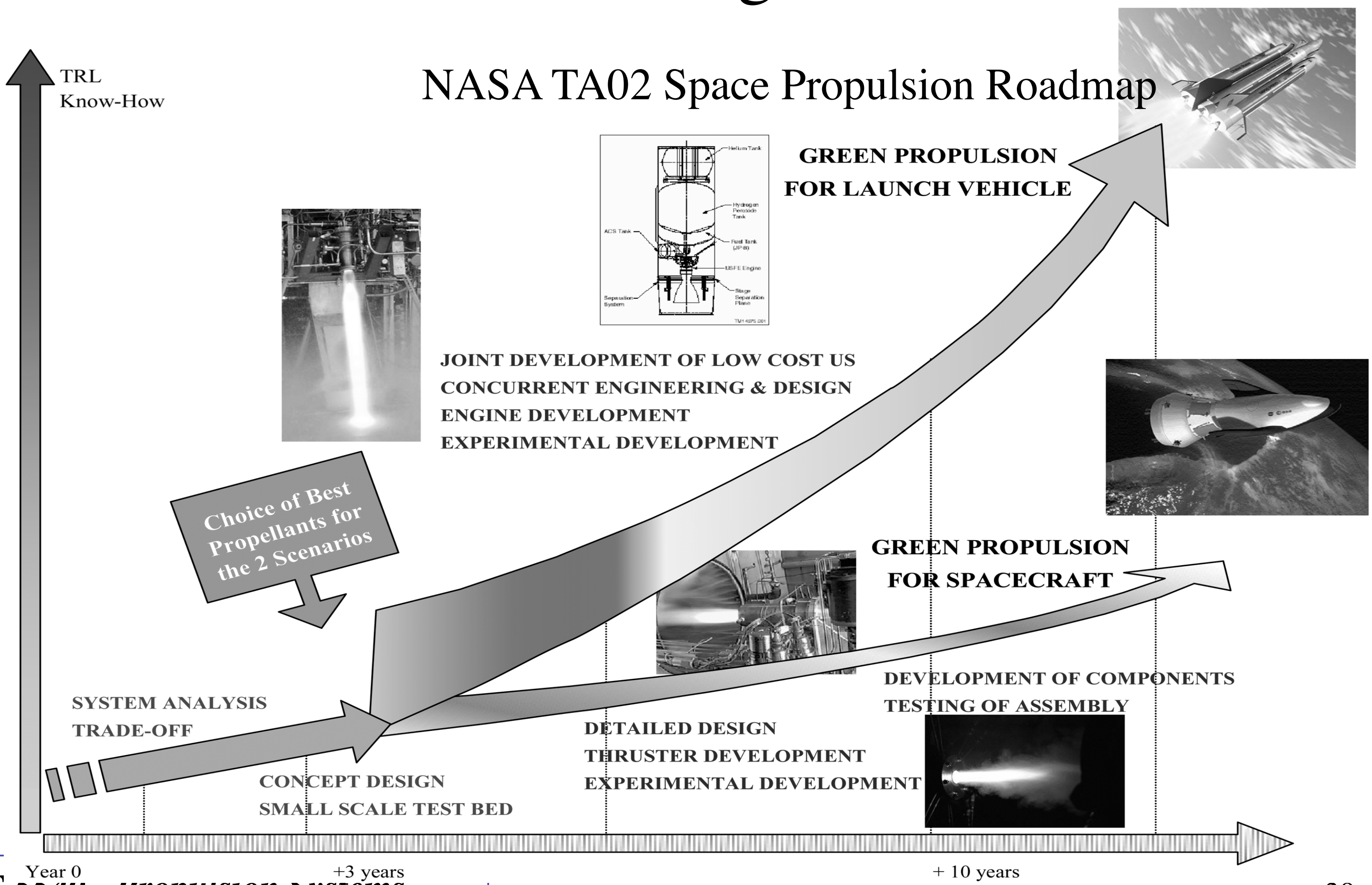
- **A new family of environmentally friendly monopropellants has been identified as an alternative to hydrazine. These new propellants are based on blends of e.g. hydroxyl ammonium nitrate (HAN), ammonium dinitramide (AND), hydrazinium nitroformate (HNF), nitrous oxide (N_2O), and hydrogen peroxide (H_2O_2)**

- **When compared to hydrazine, e.g. HAN blends have a range of specific impulse (I_{sp}) which can exceed that of hydrazine. Testing of HAN based propellants has begun to show promise and could soon be adopted for on-board propulsion systems of LEO satellites and constellations.**

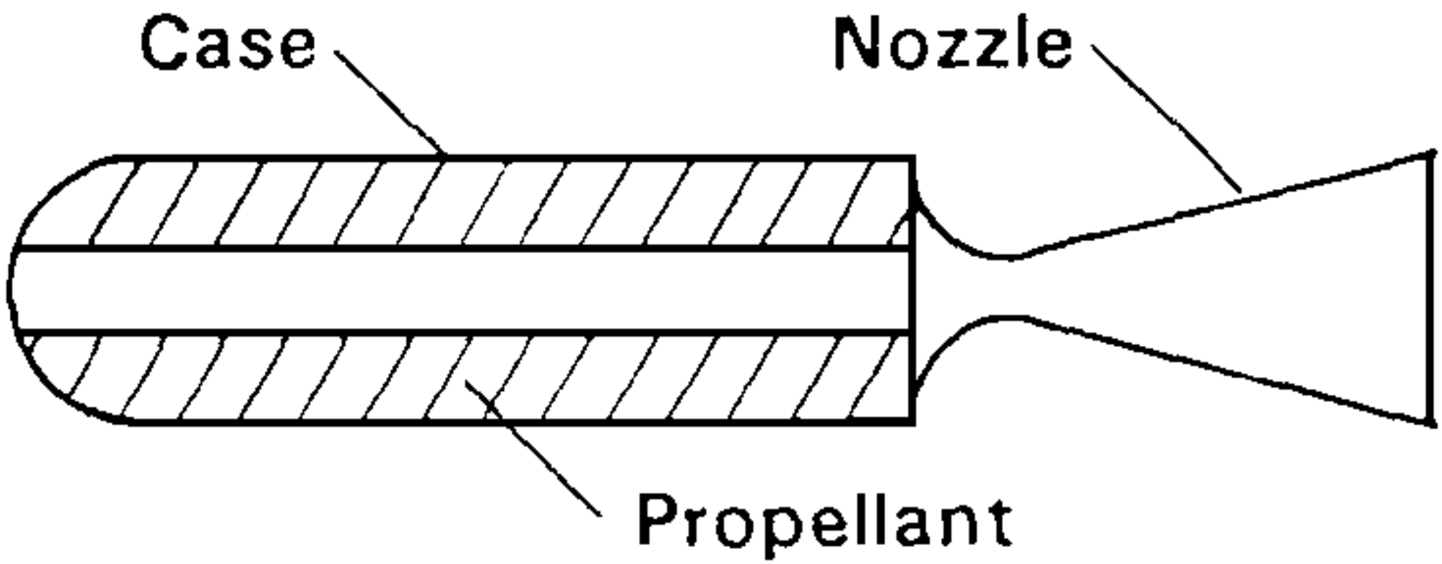
Advantages:

- **Safer propellants (also called '*green*' or '*reduced hazard propellants*') reduce costs by:**
    - **Eliminating the need for self-contained atmospheric protective ensemble (SCAPE) suits needed for toxic propellants.**
    - **No extensive and prohibitive propellant safety precautions and isolation of the space vehicle from parallel activities during propellant loading operations.**

Future Development in Liquid Propellant Technologies (2)

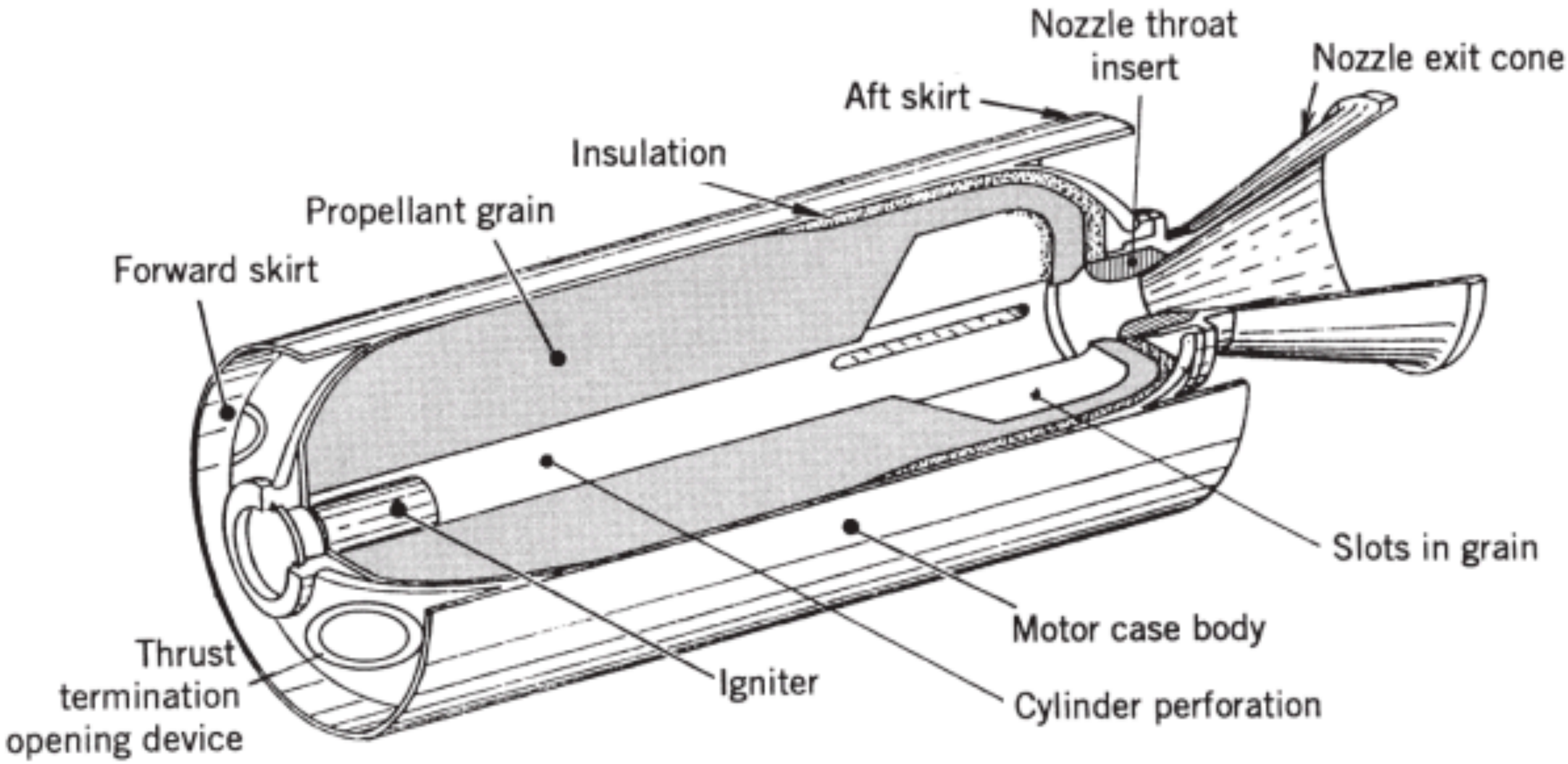


Solid Rocket Motors



- The oxidizer and fuel are stored in the combustion chamber as a mechanical mixture in solid form
- Two conditions for use:
    The total Impulse is known accurately in advance
    Restart is not required
- Elements include: Case, Igniter, Grain, Nozzle, liner/insulation

Solid Rocket Motors



Simplified perspective three-quarter section of a typical solid propellant rocket motor with the propellant grain bonded to the case and the insulation layer and with a conical exhaust nozzle. The cylindrical case with its forward and aft hemispherical domes form a pressure vessel to contain the combustion chamber pressure.

Solid Propellants

- There are two principal types of propellants:
  - *Homogeneous propellants*, which are composed of fuels that contain enough chemically bonded oxygen to sustain the propellant burning process,
  - *Composite propellants*, which are composed of organic fuel binders and oxidizers.
- Most common is the use of *composite propellants*, usually based on solid aluminium powder held in e.g. a hydroxyl terminated polybutadiene (HTPB) synthetic rubber binder and stable solid oxidizer (ammonium perchlorate or nitro-cellulose, -double based). The propellant is premixed and batch loaded into lightweight simple motors.
- Typical solid propellant mixtures are listed below:

Double-based Propellant (fuel and oxidant chemically mixed)		Composite Propellant (fuel and oxidant mechanically mixed)	
	%		%
Nitrocellose	51.4	Ammonium perchlorate (NH_4ClO_4)	62.0
Nitroglycerine	42.9	Binding material (fuel also)	21.9
Additives	5.7	Aluminium powder (fuel)	15.0
		Additives	1.1
Total	100		100

Ref: L.J. Carter, SPACEFLIGHT, Vol. 36, June 1994

Solid Propellants (2)

Main Performance of Solid Propellant Motors

- Thrust level: 50 N (for e.g. spin-up/down of small satellites) ≤ 50 000 N typical for satellite orbit transfer applications; up to $5 \cdot 10^6$ N for launcher/spacecraft application.
- Delivered impulse: ~10 Ns (F =50 N, e.g. spin-up/down of small satellites) ≤ 10^7 Ns for satellite orbit transfer applications
- Motor-spec. Impulse: ~ 2400 Ns/kg for $F \leq 50$ N; ≤ 3000 Ns/kg for $F \leq 50\,000$ N
- System-spec. Impulse: 2300 ÷ 2700 Ns/kg (~120 Ns/kg for $F \leq 50$ N)

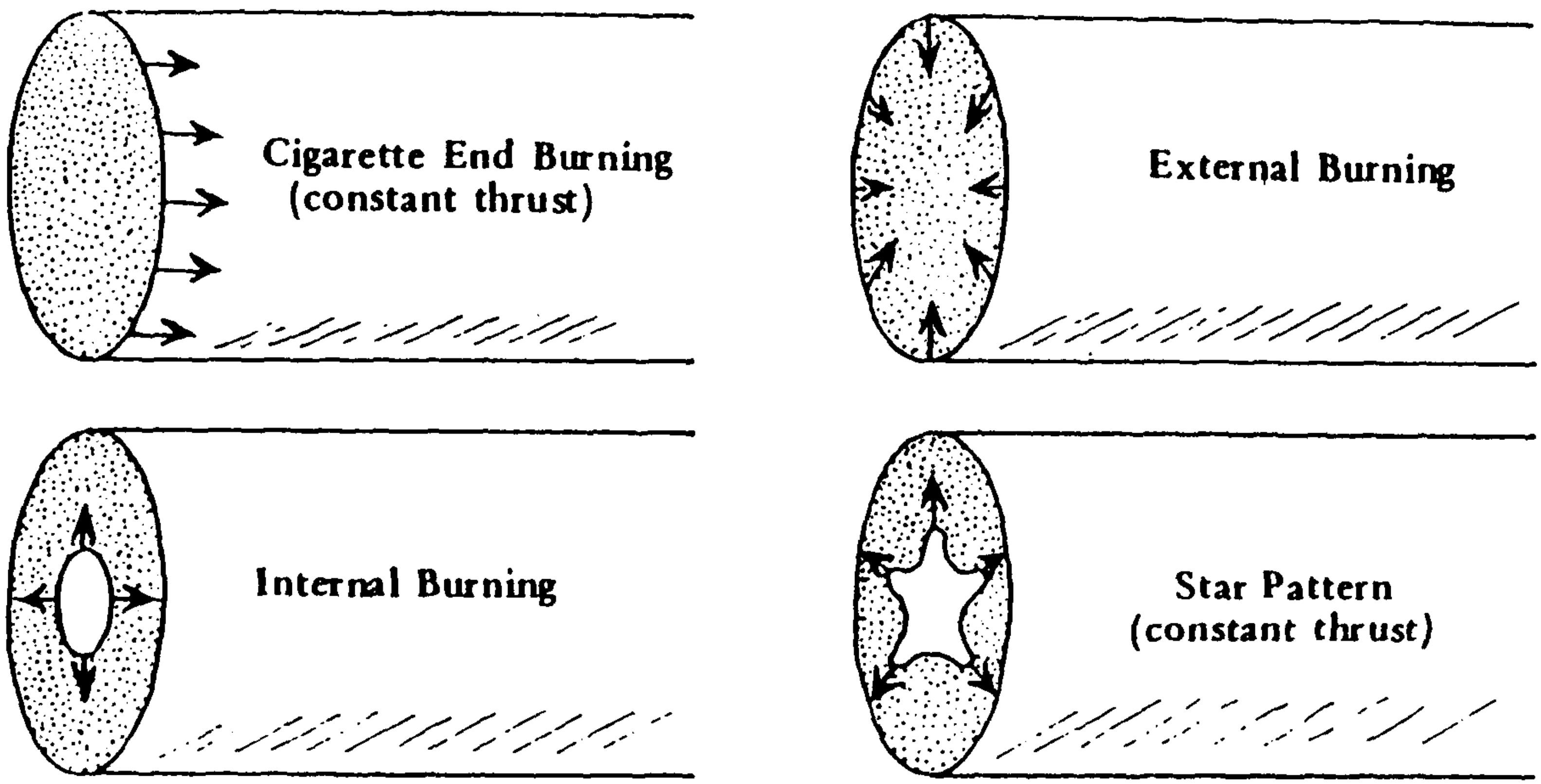
Advantages

- Relatively simple operation
- Very high mass fraction, excellent bulk density and packaging characteristics
- Good long-term storage characteristics

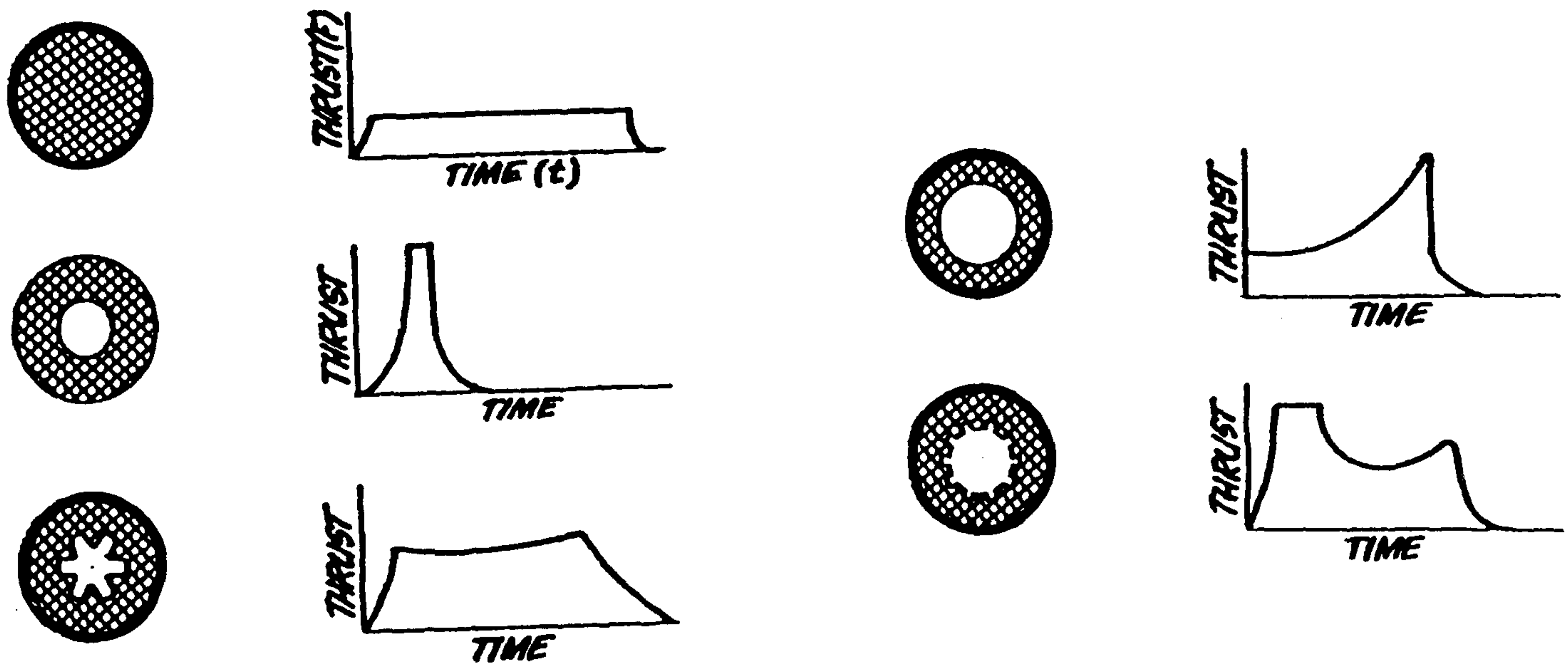
Disadvantages

- Not readily tested and checked-out prior to flight
- Very difficult to stop and restart, throttle, pulse, etc. (hybrid)
- Limited I_{sp} performance (2400 – 3000 Ns/kg)
- Limited redundancy with associated reliability and safety issues

Burning Patterns

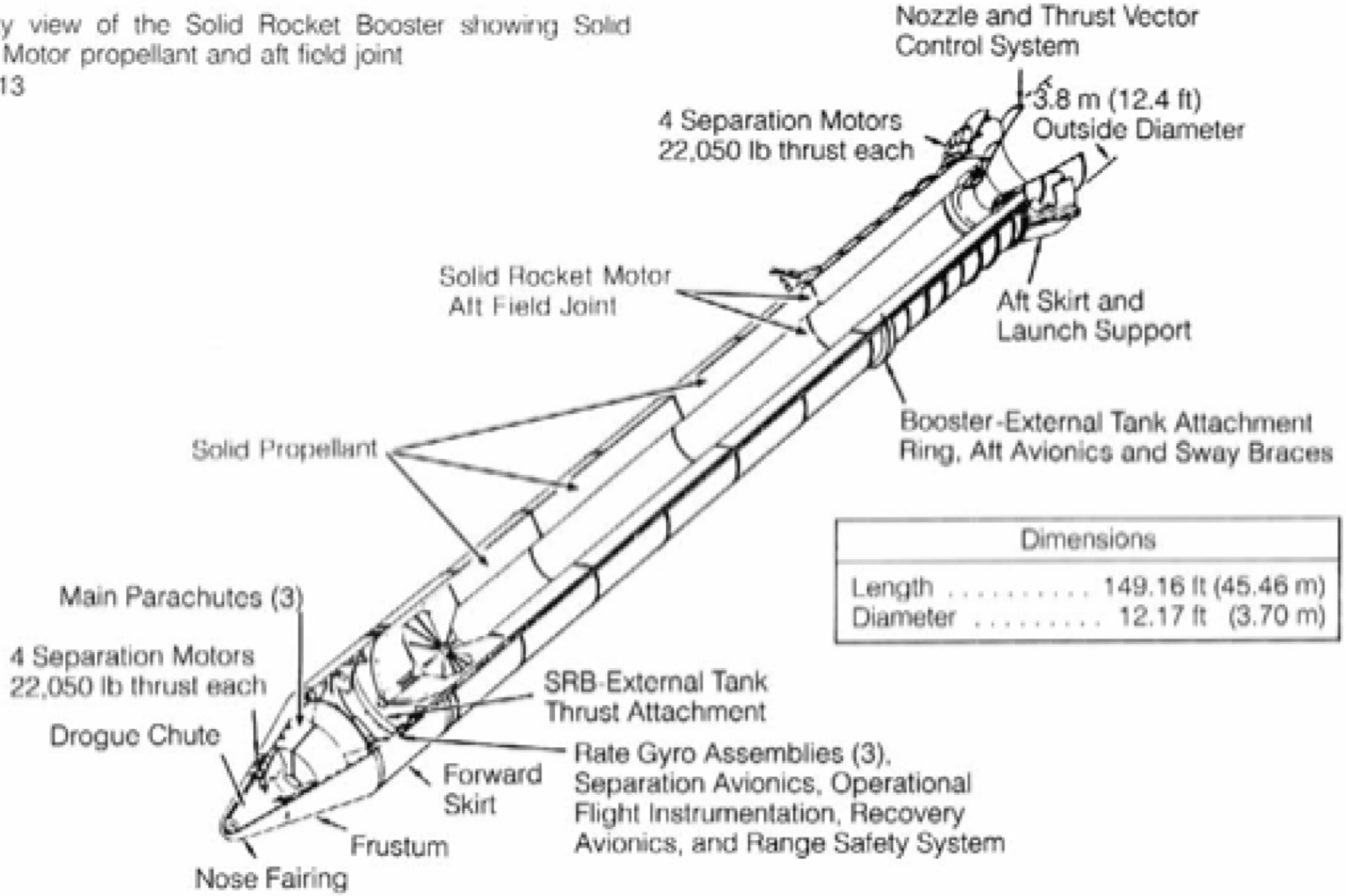


Thrust Profiles

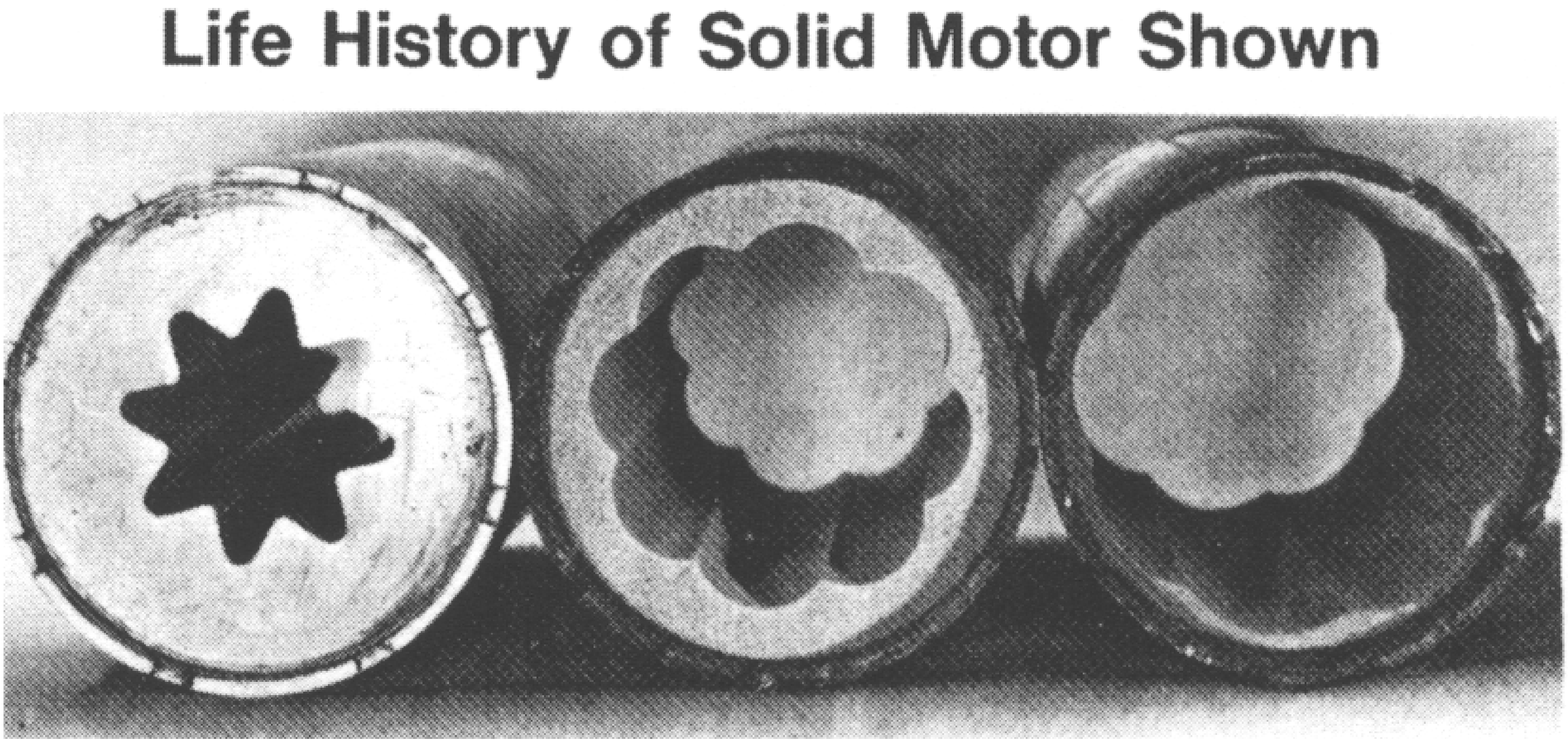


Example: Space Shuttle Solid Booster

Cutaway view of the Solid Rocket Booster showing Solid Rocket Motor propellant and aft field joint
Figure 13

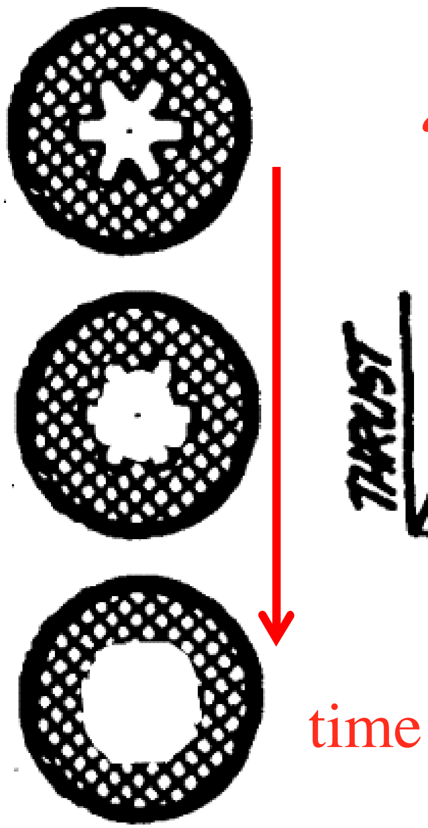


SHAPE OF PROPELLANT GRAINS QUENCHED AT DIFFERENT TIMES



Start condition **Quenched at 1.5 s** **Quenched at 2.5 s**

Burn Area Revisited (cont'd)

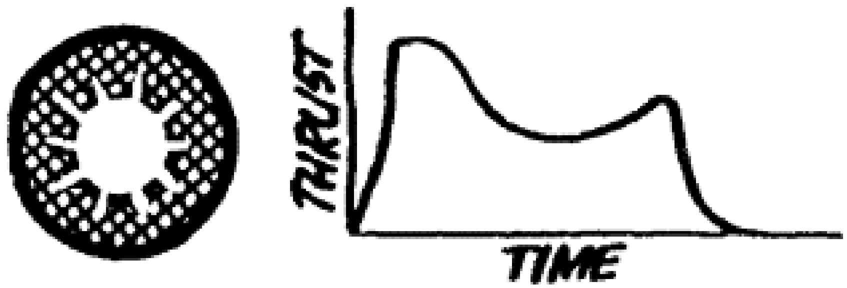


"tips burn first"

- Burn Area stays relatively constant
- Burn Volume Goes Down
- Ratio of Burn Area to Chamber Volume goes ….. *Down! Fast!*
- *Result is a more shaped burn profile*

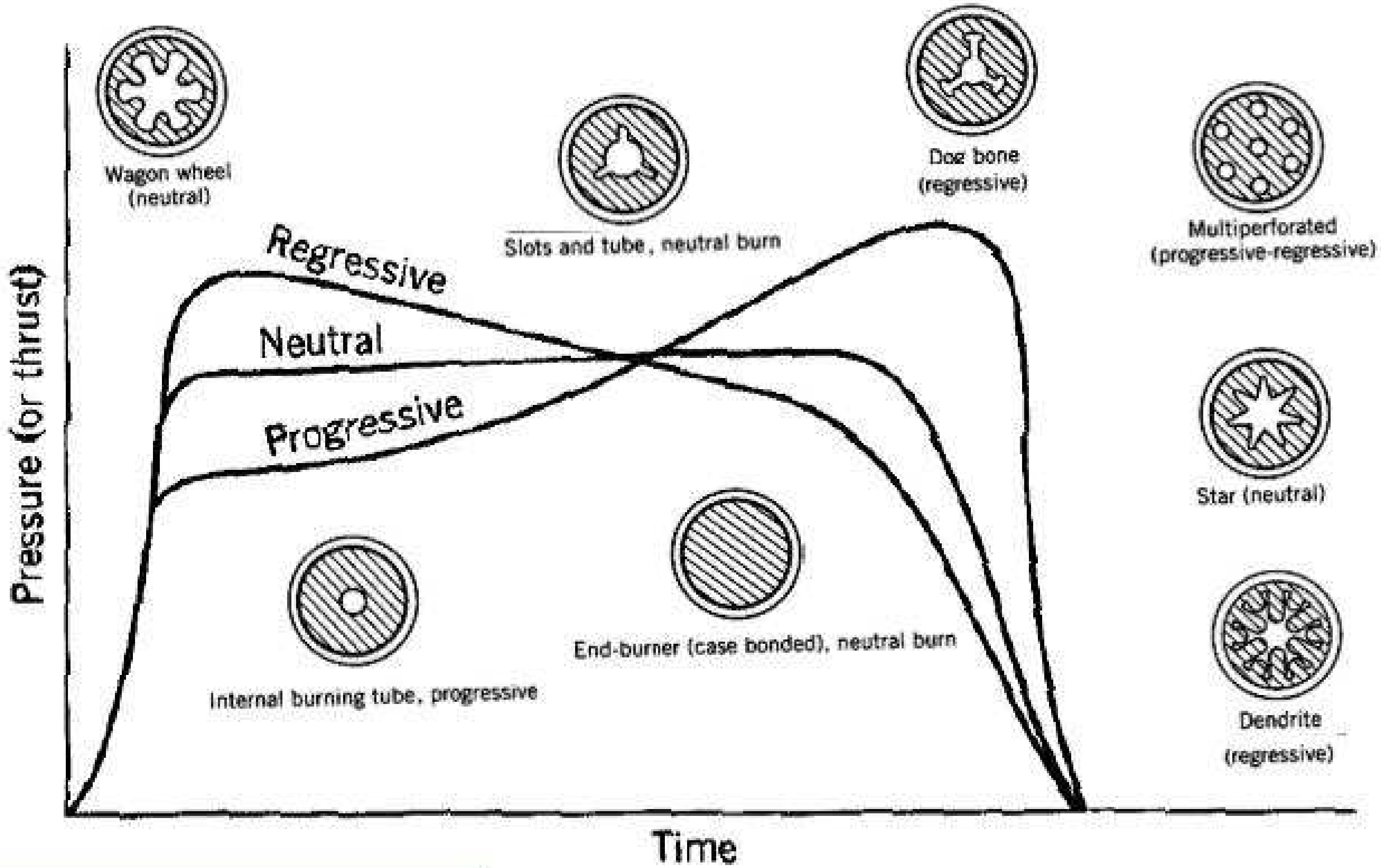
$$\frac{\partial P_0}{\partial t} = \frac{A_{burn} a P_o^{\,n}}{V_c}\left[\rho_p R_g T_0 - P_0\right] - P_0\left[\frac{A^*}{V_c}\sqrt{\gamma R_g T_0\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{(\gamma-1)}}}\right]$$

Finally Look at grain pattern



- "*Regressive Grain pattern*" … Burn surface area actually shrinks As propellant is burned

Solid Rocket Burn Summary



Hybrid Rocket Motors

• Possess features of both liquid and solid rockets.

• Hybrid consists of a solid fuel grain made from a polymeric material.

• Oxidizer is stored in a tank separate from the fuel grain, which is stored in a combustion chamber.

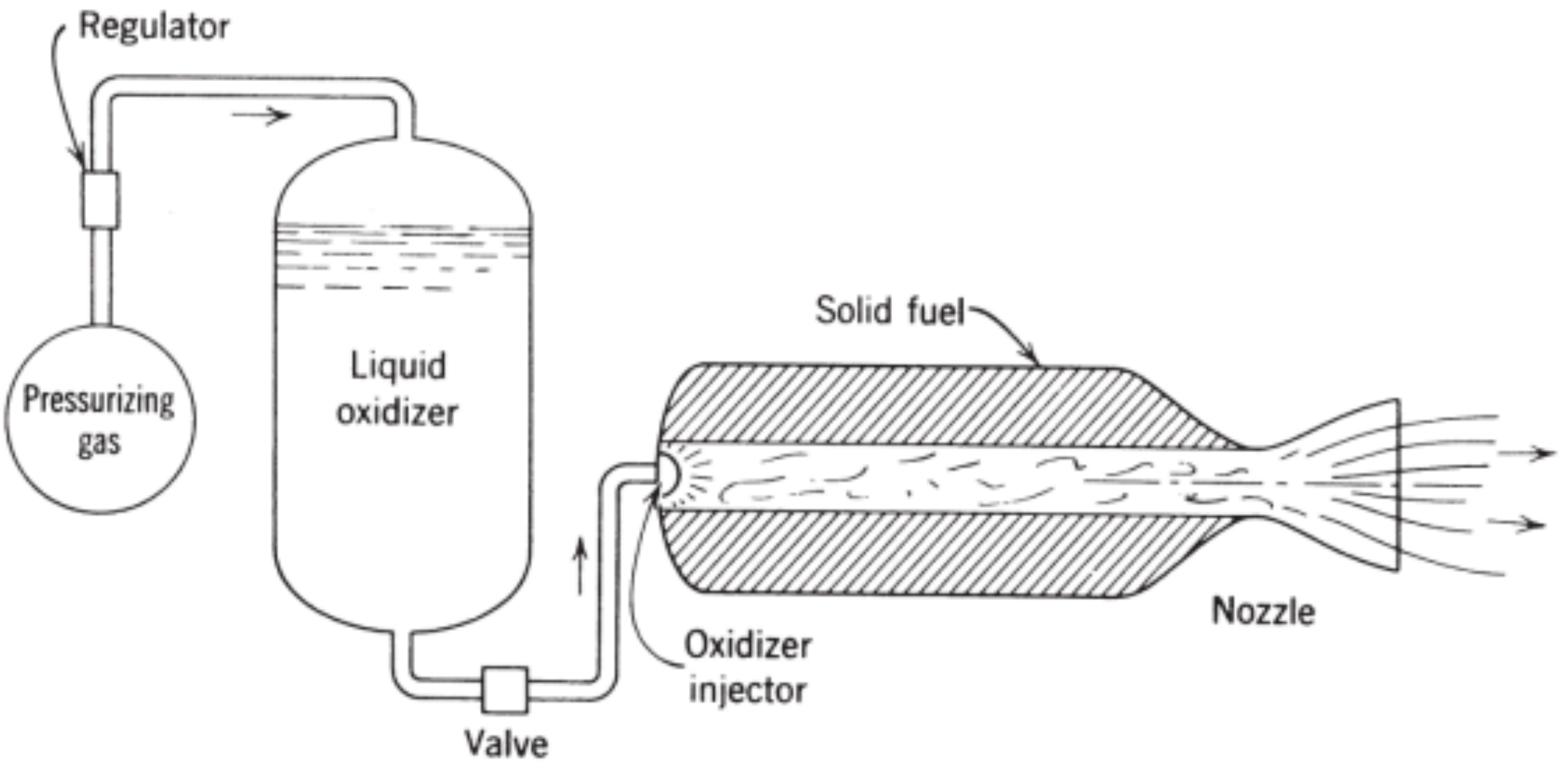
• Both propellants are inert and only combust when the fuel is converted to gaseous state and mixed with oxidizer in the combustion chamber.

• Limited explosion potential

•Like liquid rockets, hybrid rockets can potentially be throttle, stopped, and restarted.

Hybrid Rocket Motor

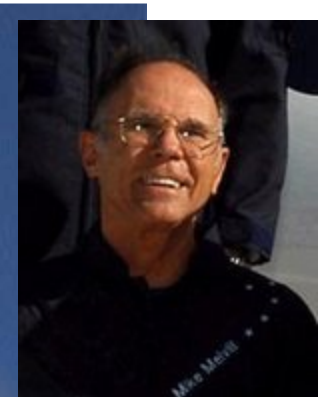
Hybrid Rocket Motors



Simplified schematic diagram of a typical hybrid rocket engine. The relative positions of the oxidizer tank, high-pressure gas tank, and the fuel chamber with its nozzle depend on the particular vehicle design.

Credit: Sutton and Biblarz

Space Dev® Hybrid Powered "Spaceship 1" (cont'd)



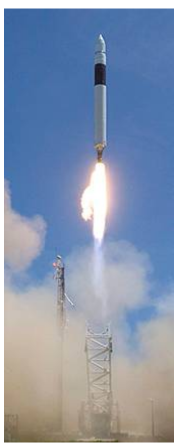
- Built by Burt Rutan (Scaled Composites®) with Paul Allen's (Apple co founder) Money in Mojave CA SS1 wrote history, when the first private suborbital spaceflight was conducted on June 21, 2004 (with pilot Mike Melvill).

- SS1 won the X-Prize with flights on 29.09.2004 (Melville) and a follow up flight on 04.10.2004. (Brian Binneie)

- Powered by a 16700 lbf thrust Hybrid Motor (SpaceDev)

Hybrids Rockets are a Potentially Enabling Technology for the Emerging Commercial Spaceflight Industry

- NASA is contracting with commercial space hardware and launch service companies to fill the void left by the retirement of the Space Shuttle fleet.
- Well funded firms like Virgin Galactic, SpaceX, Blue Origin, Sierra Nevada Corp, Bigelow Aerospace, and others are pioneering a new era in spaceflight and space exploration.


Space X Falcon 9 Medium Lift Launcher


SNC Dream Chaser Powered by SNC Hybrid Rocket Motor


Masten Engineering's Winning Lunar X-prize Entry


Danish Suborbital's Tycho Brahe Spacecraft powered by Hybrid HEAT Rocket


Bigelow Aerospace Space Station Module

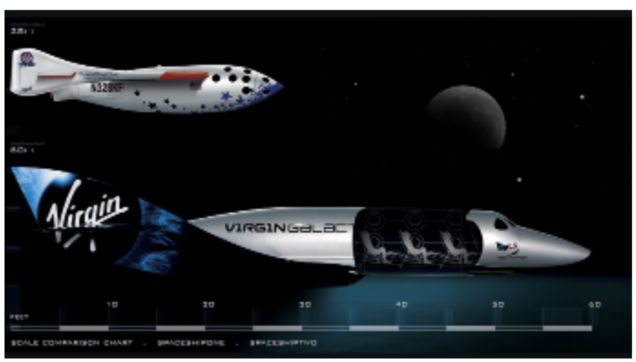

Spaceship One^{TM} Hybrid Rocket Firing During Ansari X-prize Flight


Virgin Galactic VSS Enterprise Powered by SNC Hybrid Rocket Motor

Birth of a New Transportation Industry?

Virgin Galactic, the British company created by entrepreneur Sir Richard Branson to send tourists into space and the State of New Mexico have entered into an agreement for the State to build a $225 million spaceport. Virgin Galactic has also revealed that 38,000 people from 126 countries have expressed interest its commercial suborbital flights. A core group of 100 "founders" have paid the full initial $200,000 ticket price and an additional 300 intrepid passengers have placed deposits.

Virgin Galactic was cleared for civil airspace operations in 2008 and is expected to initiate passenger services beginning in 2012.



Propellant Comparisons

	Cold gas	Monoprop	Biprop	Solid
Specific impulse, s	50	225	310	290
Thrust range, lb	0.01–0.02	0.1–600	>2	>300
Impulse range, lb-s	$<10^3$	10^3–10^5	10^4–10^8	10^4–10^8
Min impulse bit, lb-s	0.0002	0.003	0.03	N
Complexity	Least	Midrange	Most	Midrange
S/C contamination	N	N	Y	Y
Restart	Y	Y	Y	N
Pulsing	Y	Y	Y	N
Throttling	N	Y	Y	N

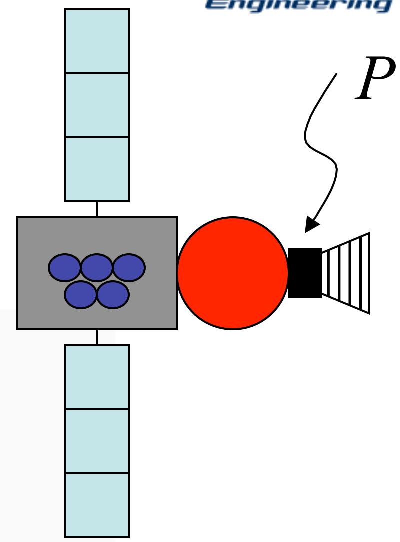
Propellant Comparisons

Type	Propellant	Vacuum I_{sp}	Thrust Range (lb_f)	Avg Density (gm/cm^3)
Cold Gas	N_2, NH_3, Freon, He	50-75	0.01-50	0.28-0.96
Solid Motor	Various	280-300	$10 - 10^6$	1.8
Mono prop	H_2O_2, N_2H_4	150-225	0.01-0.1	1.44, 1.0
Bi-Prop	O_2 and RP-1	350	$1 - 10^6$	1.14 and 0.80
Bi-Prop	O_2 and H_2	450	$1 - 10^6$	1.14 and 0.07
Bi-Prop	N_2O_4 and MMH	300-340	$1 - 10^6$	1.43 and 0.86
Bi-Prop	F_2 and N_2H_4	425	$1 - 10^6$	1.5 and 1.0
Bi-Prop	OF_2 and B_2H_6	430	$1 - 10^6$	1.5 and 0.44
Bi-Prop	CIF_5 and N_2H_4	350	$1 - 10^6$	1.9 and 1.0

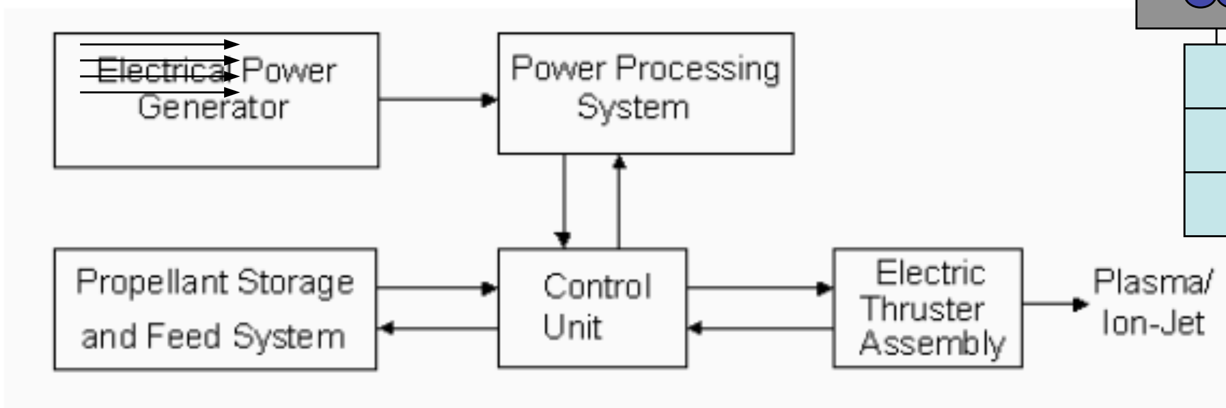
Chemical Rocket Comparison

Feature	Liquid, Bipropellant	Solid	Hybrid
Safety	Potential for combustion instability, can explode, volatile propellants	Highly flammable significant explosion potential, DOT 1.1	**Inert propellants, low explosion and transport risk**
Toxicity	Ranges from non-toxic to highly toxic	Exhaust products highly toxic	**Exhaust products non-toxic (CO_2, H_2O)**
Fabrication Costs	Extremely expensive	Expensive, mostly due to handling difficulties	**Inexpensive**
Complexity/ Reliability	Highly complex, moderate reliability	Simple-to-moderate complexity, high reliability	**Moderate complexity, high reliability**
Operation	Throttleable, restartable, high performance	No restart, throttle capability, high-to-moderate performance	**Potentially, Thottleable, restartable, moderate performance**

Electric Propulsion



Electric propulsion systems comprise the following main components:

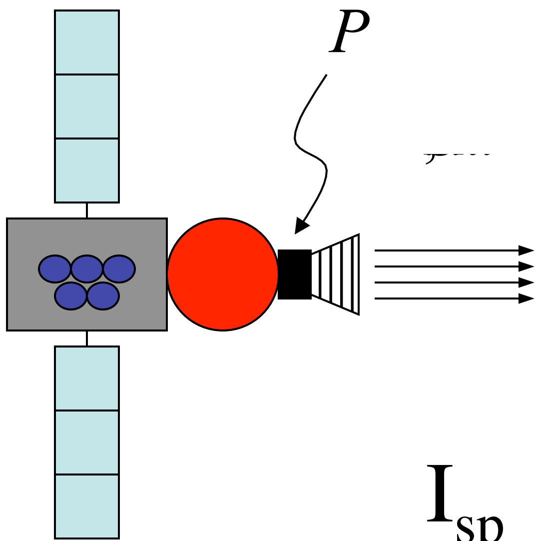


- Storage and feed system that stores and feeds the propellant to the thrusters to generate thrust
- Valves, piping which connects the propellant storage system with the thruster
- Electric control unit to operate electrically the valves and thrusters
- Electric power supply and power processing system

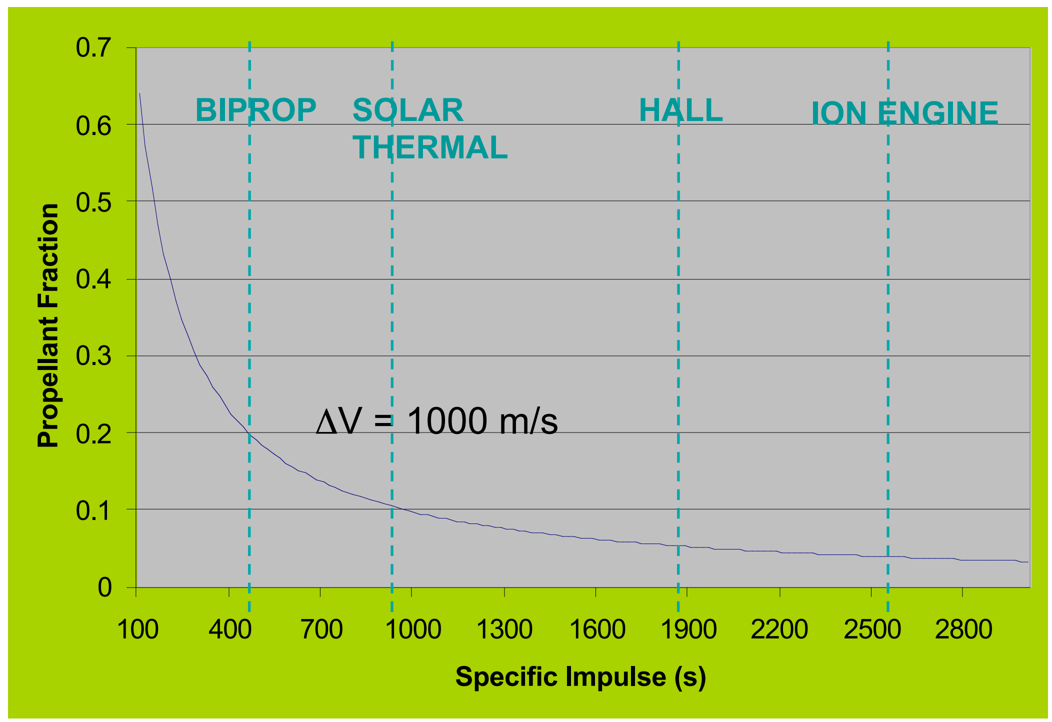
- **Electric Power Generator**

  Energy can be obtained from either sunlight or from a nuclear reactor. In the case of solar electric propulsion, solar photons are converted into electricity by solar cells.

Benefits of Electric Propulsion



	I_{sp} (s)
Chemical	**400**
Solar Thermal	**800**
Nuclear Thermal	**800+**
Electric	**ANY**

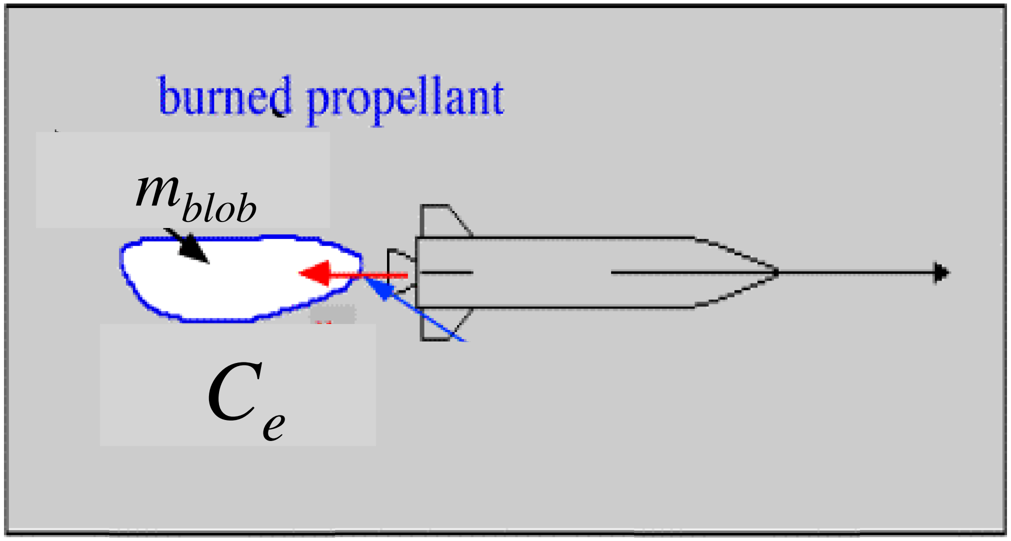


Basic Concept

- Use electric power to accelerate the propellant to produce thrust.
- Because the effective velocity of the exhaust, C_e, is limited only by the speed of light, the I_{sp} can be very high.
  - As I_{sp} increases, the required propellant mass decreases.
  - $$\boxed{I_{sp} = \frac{F}{\left(\dot{m}\right) g_0} = \frac{C_e}{g_0}}$$ $$\boxed{C_e \approx V_e + \frac{p_e A_e - p_\infty A_e}{\dot{m}_{ex}}}$$
- However, unless we have megawatts of electricity available, the total thrust will be small. Accelerations in the range of 0.001g

Power Required (1)

- Mechanical Power of a Rocket Exhaust



Added K.E. of burned Propellant "Blob"

$$K.E. = \frac{1}{2} m_{blob} \cdot C_e^{\,2}$$

$$\rightarrow P_{ex} = \frac{d}{dt}\left(K.E.\right) = \frac{1}{2}\left(\dot{m}_e\right) \cdot C_e^{\,2}$$

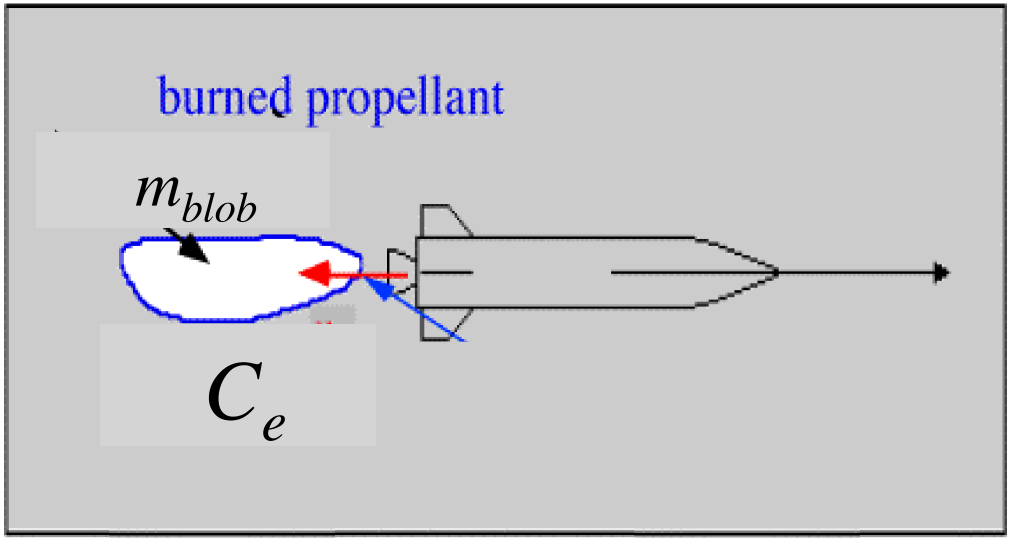
$$C_e \approx V_e + \frac{p_e A_e - p_\infty A_e}{\dot{m}_{ex}} \rightarrow F = \dot{m}_e V_e + \left(p_e A_e - p_\infty A_e\right) = \dot{m}_e C_e$$

- ***Effective Exhaust Velocity***

$$\boxed{\Rightarrow P_{ex} = \frac{1}{2} F \cdot C_e}$$

Power Required (2)

- Mechanical Power of a Rocket Exhaust



$$\Rightarrow P_{ex} = \frac{1}{2} F \cdot C_e$$

$$C_e \approx \frac{F}{\dot{m}} = g_0 I_{sp} \rightarrow \boxed{P_{ex} = \frac{1}{2} F \cdot g_o \cdot I_{sp}}$$

Power Required (3)

- Define Power Plant Efficiency

$$\eta = \frac{P_{out}}{P_{in}} \rightarrow P_{ex} = \eta \cdot P_{in}$$

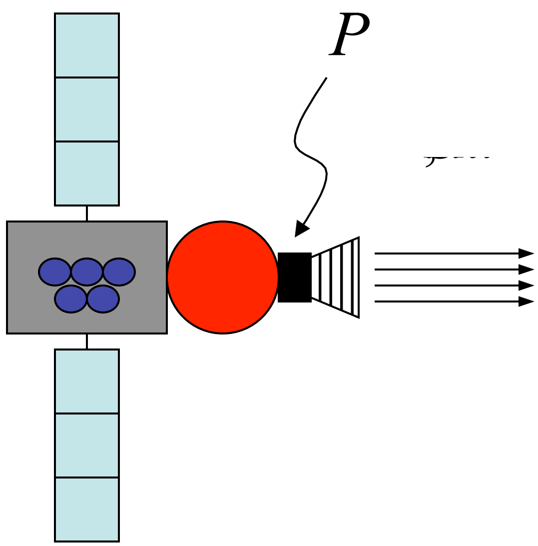
- The electrical power needed is a function of the thrust required, and the I_{sp}.

$$P_{in} = \frac{F \cdot I_{sp} \cdot g_0}{2\eta}$$

(High I_{sp} means low Thrust Unless you have a BIG! Power source.)

- η is an efficiency factor that depends on the specific electric thruster, and varies from 0.3 to 0.95

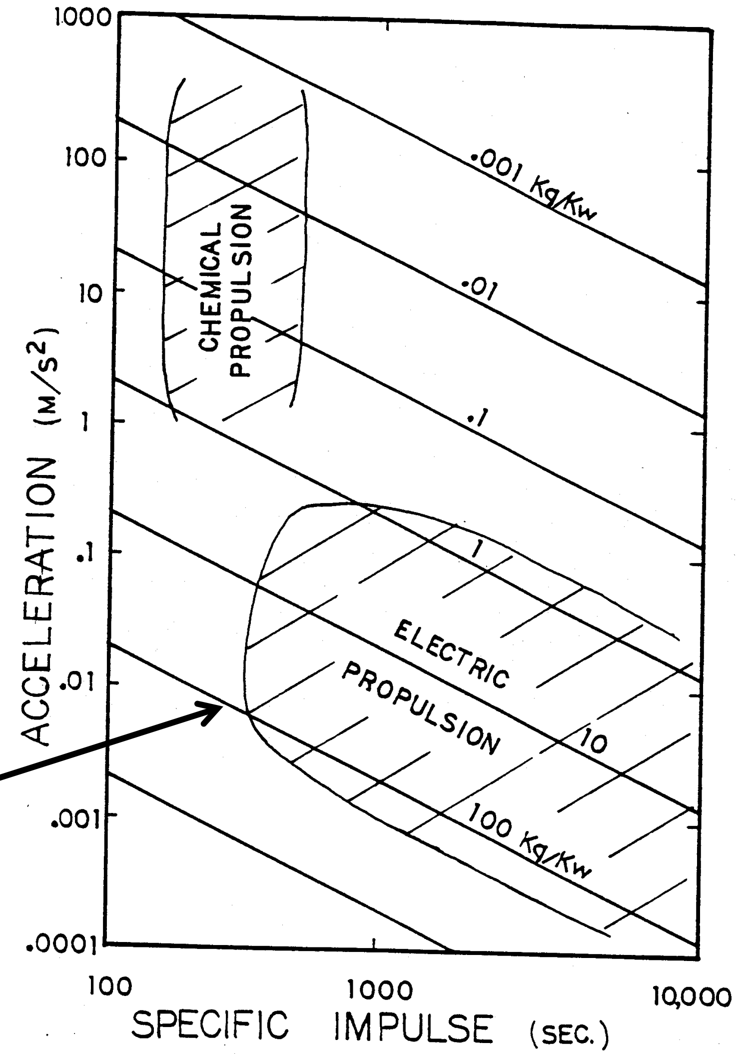
Power Required (cont'd)



• **Specific Mass** (Kg/KW)

How much systems weighs per unit of power delivered

• EP systems tend to have larger specific Masses than chemical rockets due to Higher complexity of systems



Classes of Thrusters

- **Electro-Thermal** - The propellant gas is electrically heated and expanded through a nozzle.
  - **resistojets and arcjets.**
- **Electro-Static** - The propellant is ionized and the resulting ions are accelerated through an electric potential.
  - **Hall effect and Kaufmann type thrusters.**
- **Electro-Magnetic** - Both electric and magnetic body forces are used to accelerate ions.
  - **Magnetoplasmadynamic thruster, or MPD**

Three Classes of Electro-propulsion

Electrothermal	Electrostatic	Electromagnetic
• Gas heated via resistance element or arc and expanded through nozzle • Resistojets • Arcjets	• Ions electrostatically accelerated • Hall effect (HET) • Ion • Field emission	• Plasma accelerated via interaction of current and magnetic field • Pulsed plasma (PPTs) • Magnetoplasmadynamic (MPD) • Pulsed inductive (PIT)
Power Range; 0.4–2 kW	1–50 kW	50 kW–1 MW
Specific Impulse, I_{sp}; 300–800 sec	1,000–3,000 sec	2,000–5,000 sec

Survey of Electrical Thrusters (Examples of typical performance values)

(Data listed are indicative only)

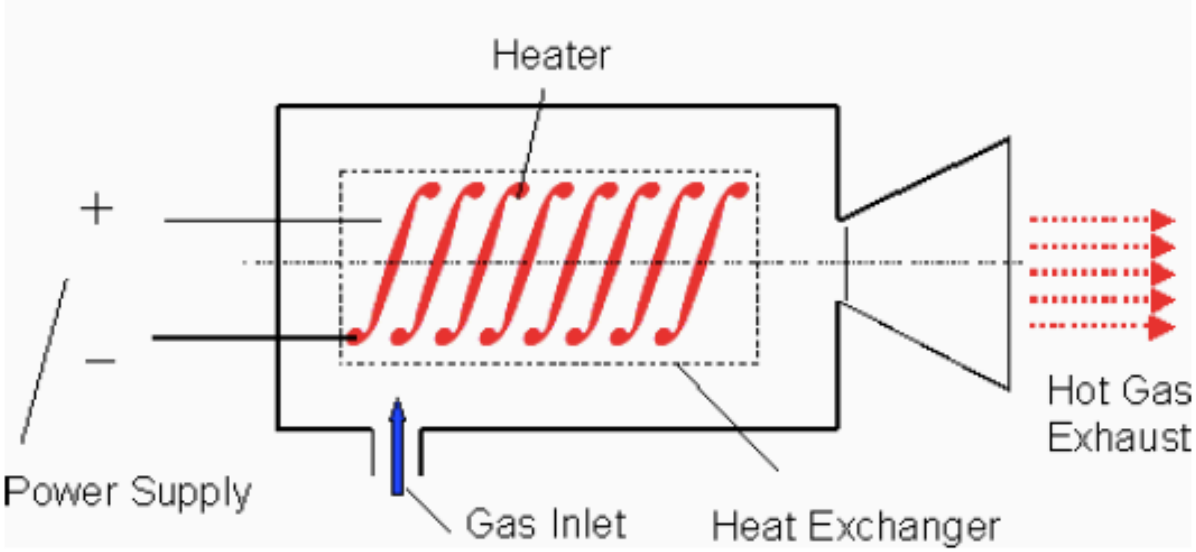
Type of Propulsion System	Thrust (N)	Power Consumption (W)	Exhaust Velocity (m/s)	Propellant (formula)	Potential Application
Resistojet	0.2	345	1 500	NH_3;CH_4	Orbit-Control (Biowast)
Hydrazine-Resistojet (PACT)	0.3	300	3 000	N_2H_4	Orbit-Control (N/S)
Arc-Jet (Hydrazine)	0.2	1 800	5 000	N_2H_4	Orbit-Control (N/S)
MPD (Teflon)	0.015	600	30 000	Teflon	Orbit-Control (N/S)
RIT10 (Ion.-Engine)	0.01	390	30 700	Xe	Orbit-Control (N/S)
RIT35 (Ion.-Engine)	0.271	7 540	31 400	Hg; Xe	Interplanetary Missions
UK-10 (Kaufman)	0.011	600	$\geq$30 000	Xe	Orbit-Control (N/S)
UK-25 (Kaufman)	0.196	6 000	$\geq$30 000	Xe	Interplanetary Missions
SPT100 (Ion.-Engine)	0.08	1 350	16 000	Xe	Orbit-Control (N/S)
Hughes 8 cm (Kaufman)	$4.5 \cdot 10^{-3}$	175	25 500	Hg	Orbit-Control
Field-Emission	$\approx 10^{-5}$ - $2 \cdot 10^{-3}$	$\approx$ 60 - 300	$\approx$ 60 000 – 100 000	Cs	Orbit and Attitude Control
					NB: (N/S) $\equiv$ North/South station keeping for Geostionary Orbits

Electrothermal Systems

- Electrothermal Systems, where the propellant (gas) is heated by passing over an electric heated solid surface (resistojet) or by passing it through an arc discharge (arcjet).

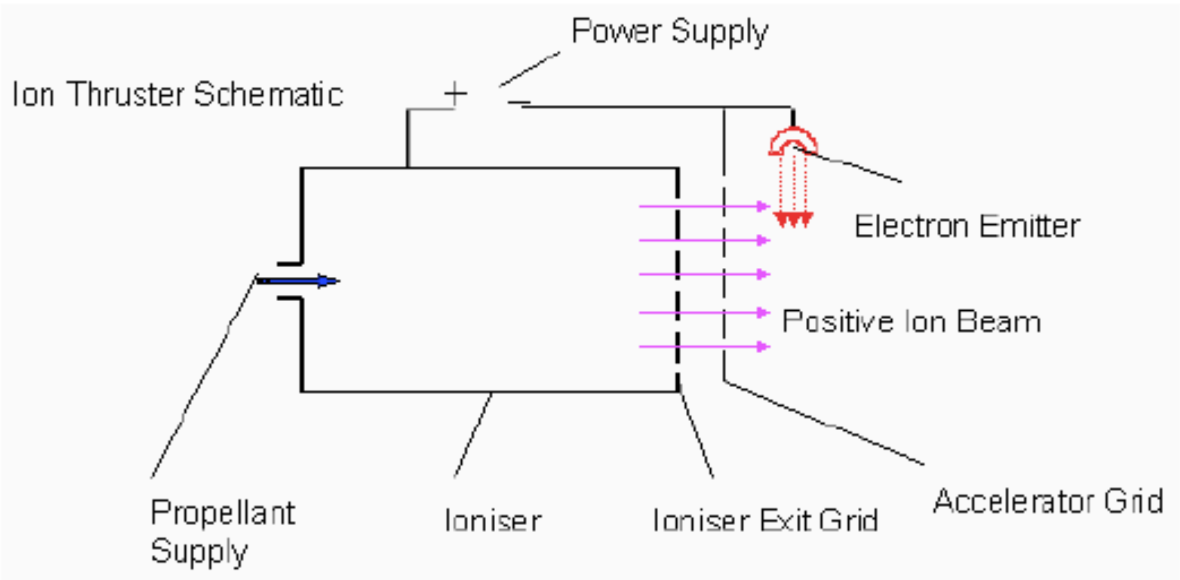
- The heated gas is then accelerated by a gas-dynamic expansion in a nozzle. Typical applications of this principle are the monopropellant hydrazine operated Power Augmented Catalytic Thruster (*PACT*) and Hydrazine-Arcjet.

Resistojet Thruster Schematic



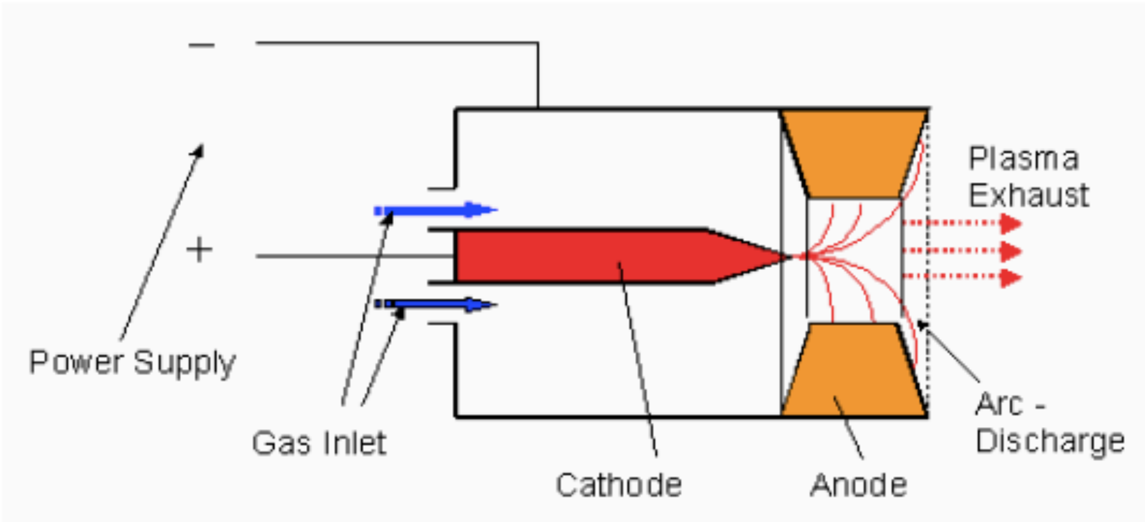
Electrostatic Systems

- Electrostatic Systems, where usually a high molecular propellant, such as Xenon gas, is ionized (ion thruster) by e.g. electron bombardment (Kaufman), in a high frequency electromagnetic field (*radio-frequency*) or by extracting ions from the surface of a liquid metal (cesium) under the effect of a strong electrostatic field (field emission).
- The ions are then accelerated to high velocity (30 to 60 km/s) by a strong electric field. Electrons are injected into the ion beam from an electron emitter in order to keep it electrically neutral, thus preventing an electric charge build-up of the spacecraft.
- In addition to the above described category of ion thrusters, the Stationary Plasma Thruster (SPT) which belongs to the category of 'Hall-effect Thrusters', uses an applied magnetic field to control electrons in a quasi-neutral plasma discharge.



Electromagnetic Systems

- Electromagnetic Systems, where a gas is heated in an arc discharge to such a high temperature, that it is converted to neutral plasma (plasma thruster).
- The plasma is then expelled at high velocity by the interaction of the discharge current with the magnetic field (Lorentz force). A typical application of this principle is the Magneto-Plasma-Dynamic (MPD) type of thruster.



Types of Electro Thermal Thrusters

- **Resistojet -** Use electric heaters to increase the temperature and thus the thrust generated by N_2H_4 monoprop systems
- **Arcjet -** Same, but using an electric arc instead of a resistance heater.
- **Pulsed Plasma -** A high voltage arc creates a plasma that ablates the surface of a Teflon fuel bar, the ablated particles are accelerated by the plasma toward the nozzle where they provide the mass for the rocket effect.

Types of Electro-Static Thrusters

- **Ion thrusters -** xenon gas is ionized by an electric arc, and then accelerated electrically toward the nozzle.
- **Hall effect thrusters -** includes a static magnetic field which improves the acceleration of the xenon

Characteristics

Concept	Characteristics					
	Specific Impulse, (sec)	Input Power, (kW)	Thrust/ Power, (mN/kW)	Specific Mass, (kg/kW)	Propellant	Supplier
Resistojet	296	0.5	743	1.6	N_2H_4	Primex
	299	0.9	905	1	N_2H_4	Primex, TRW
Arcjet	480	0.85	135	3.5	NH_3	IRS/ITT
	502	1.8	138	3.1	N_2H_4	Primex
	>580	2.17	113	2.5	N_2H_4	Primex
	800	26*	—	—	NH_3	TRW, Primex, CTA
Pulsed Plasma Thruster (PPT)	847	< 0.03†	20.8	195	Teflon	JHU/APL
	1,200	< 0.02†	16.1	85	Teflon	Primex, TSNIIMASH, NASA
Hall Effect Thruster (HET)	1,600	1.5	55	7	Xenon	IST, Loral, Fakel
	1,638	1.4*	—	—	Xenon	TSNIIMASH, NASA
	2,042	4.5	54.3	6	Xenon	SPI, KeRC
Ion Thruster (IT)	2,585	0.5	35.6	23.6	Xenon	HAC
	2,906	0.74	37.3	22	Xenon	MELCO, Toshiba
	3,250	0.6	30	25	Xenon	MMS
	3,280	2.5	41	9.1	Xenon	HAC, NASA
	3,400	0.6	25.6	23.7	Xenon	DASA

Typical Parameters of Small Thrusters

Thruster	I_{sp} (s)	η	Thrust
Solid Rocket Motor	185	90+%	100+ N
Chemical Bipropellant	315	95+%	>2 N
Arcjets	~500 - 700s >1000 s (H_2)	~30%	0.1-1 N
Pulsed Plasma Thruster	200-1500	~15%	2 μN – 4.5 mN
Colloid Thruster	450-1350	~50%	20 μN
Hall Thruster	1500-3000	~50-60%	1.8–500 mN
Ion Thrusters	1700-3900 s	~65%	1-100 mN
Field Emission Thruster	6000-9000 s	~90%	40 μN – 1.4 mN

Electric Propulsion Thrusters

Comparison

THRUSTER COMPARISON: Typical Electrical vs. Chemical Figures

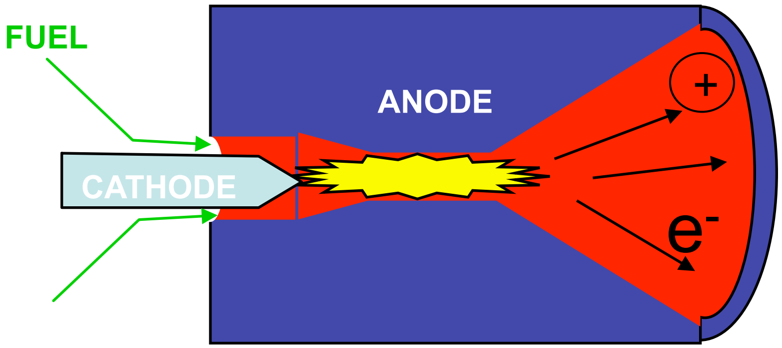
(Data listed are indicative only)

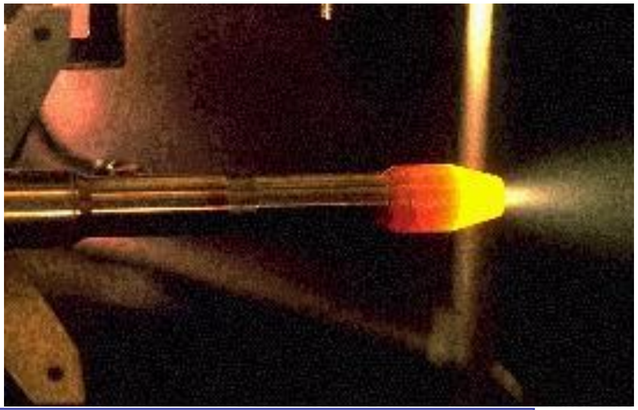
Type of Thruster	Spec. Impulse (Ns/Kg)	Thrust F (N)	DC Power Required (W)
Electrical (Ion thruster)	≈ 30 000	10^{-3} – 0.2	400 – 800
Chemical (Bi-Propellant)	≈ 3 000	5 – 500	4 – 8 (short term)
Order of magnitude of the ratio ION/Chemical	10^{1}	10^{-4}	10^{2}

Electrothermal Arcjets

•Principle: High translational energies through efficient heating

•Propellant: N_2/H_2, N_2H_4, H_2

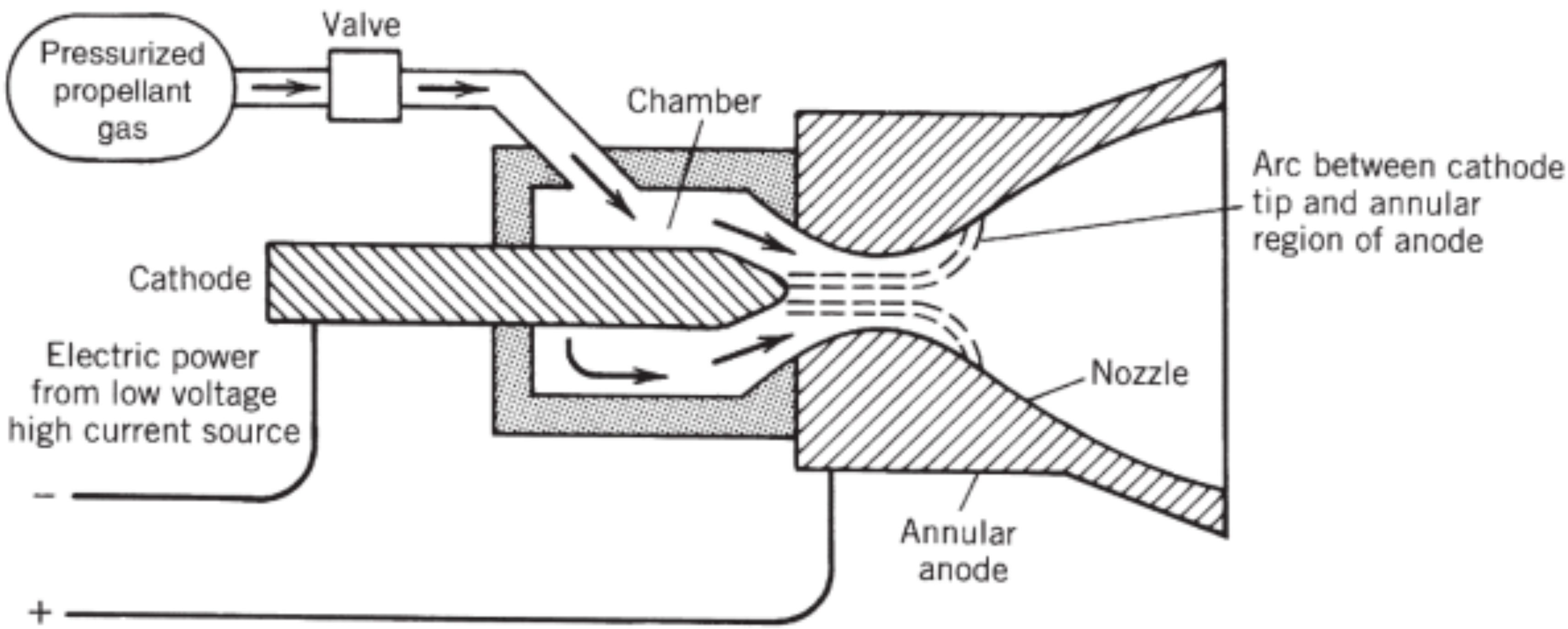




ESEX 30kW ammonia arcjet
Air Force flight experiment
Flew successfully in 1999

Isp = 500-1200 sec
η = 20-30%
Thrust = 0.1-1 N

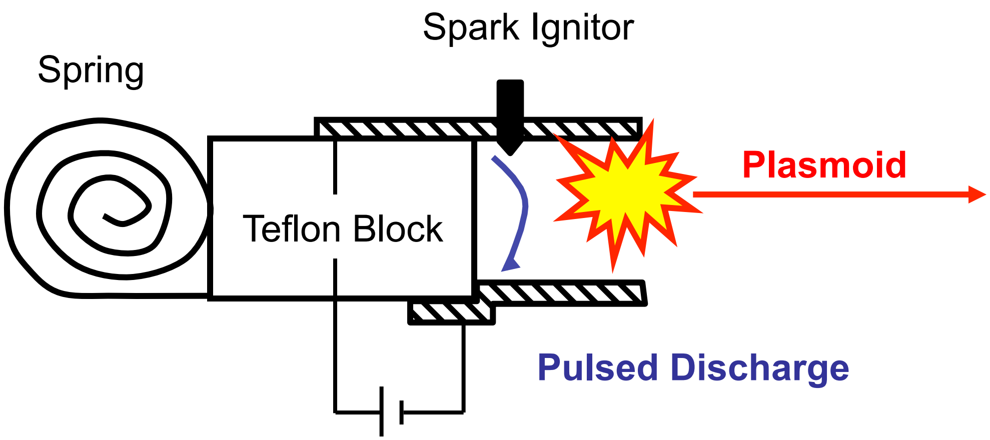
Electrothermal Arcjets



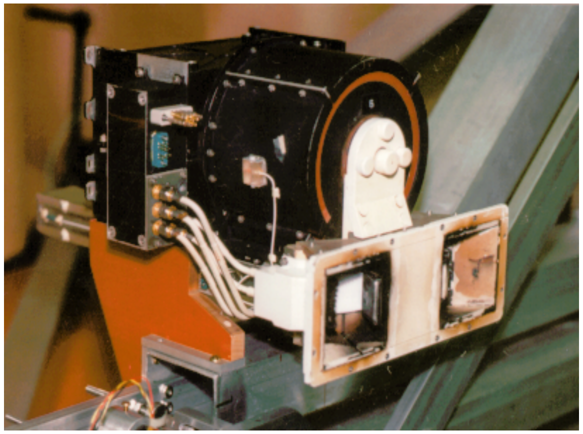
Simplified schematic diagram of arc-heating electric rocket propulsion system. The arc plasma temperature is very high (perhaps 15,000 K) and the anode, cathode, and chamber will get hot (1000 K) due to heat transfer.

Pulsed Plasma Thruster (PPT)

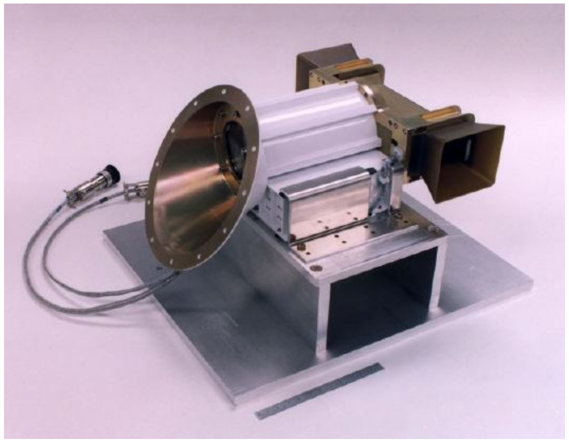
- •Principle: Plasma Surface Ablation + Acceleration
- •Propellant: Teflon, Teflon derivatives



Isp = 200 -1500 sec
η = ~15%
Thrust = 2 uN - 4 mN

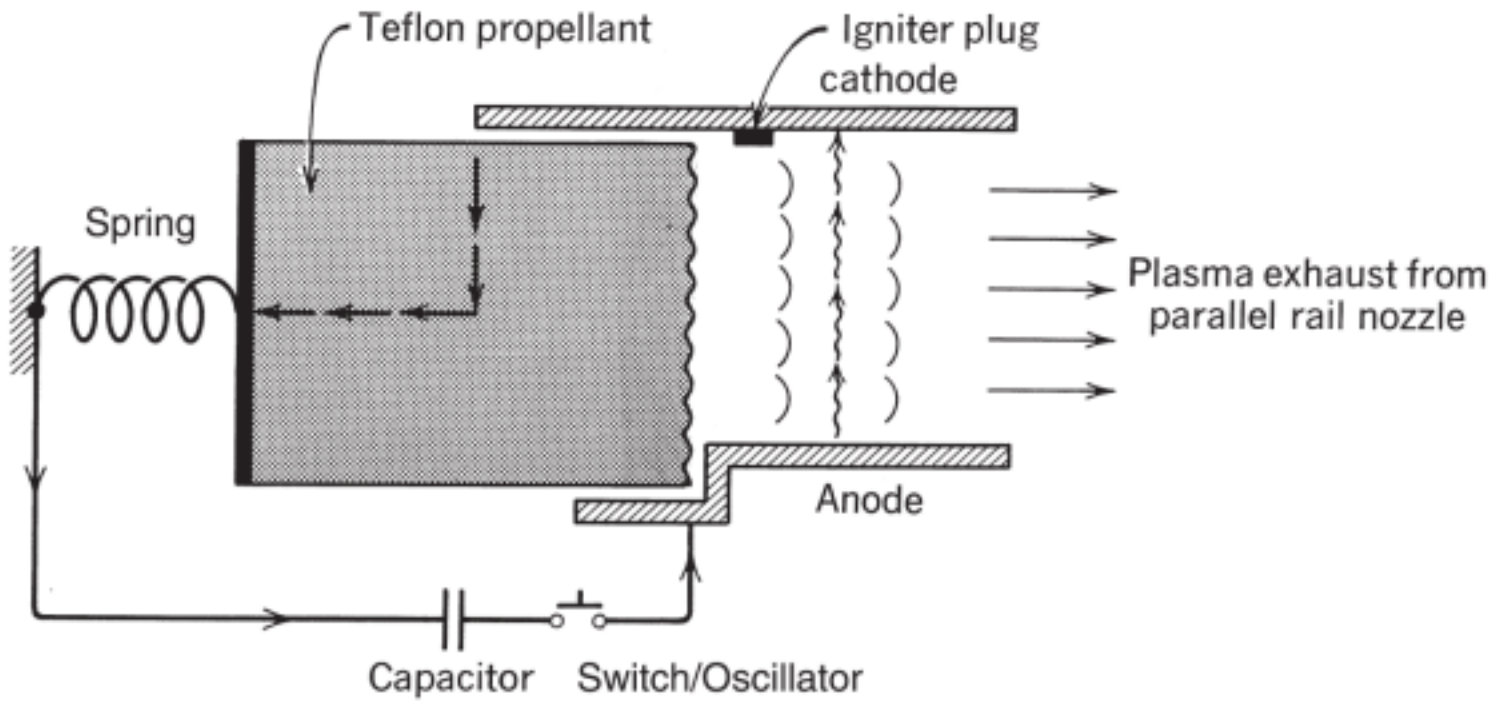


LES 8/9



Primex EO-1

Pulsed Plasma Thruster (PPT)



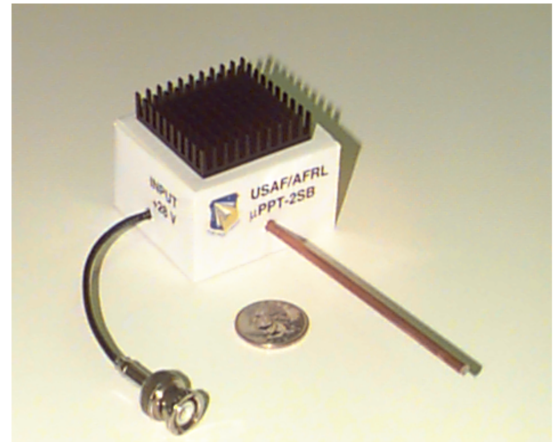
The high current in the plasma arc induces a magnetic field. The action of the current and the magnetic field causes the plasma to be accelerated at right angles to both the magnetic field and the current, namely in the direction of the rails. Each time the arc is created a small amount of solid propellant (Teflon) is vaporized and converted to a small plasma cloud, which (when ejected) gives a small pulse of thrust. Actual units can operate with many pulses per second.

UtahState UNIVERSITY

Mechanical & Aerospace Engineering

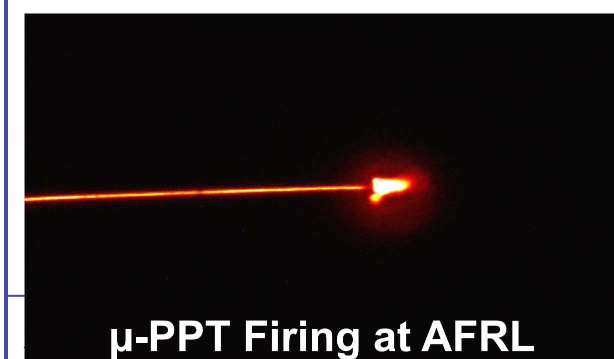
Micro-PPTs

- Key development issues:
  - Thruster life as propellant recedes
  - Minimize operational voltage
  - Low mass power supplies and switching mechanisms
  - Quantify effluents
- Flight demo on TechSat 21



AFRL Patented Designs

COTS Capacitors

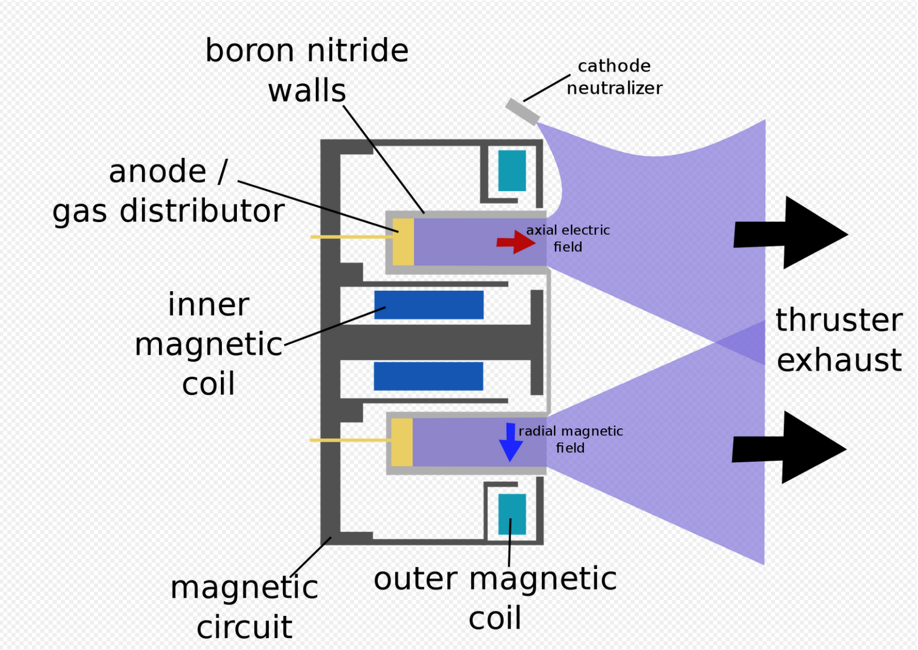


µ-PPT Firing at AFRL

COTS PPUs

Hall Effect Thrusters

- **Hall effect thrusters** accelerate ions by means of an electric potential between a cylindrical anode and a negatively charged plasma that forms the cathode.
- Bulk of the propellant (typically xenon) is introduced near the anode, where it ionizes and flows toward the cathode.
- Ions accelerate towards and through it, picking up electrons as they leave to neutralize the beam and leave the thruster at high velocity.



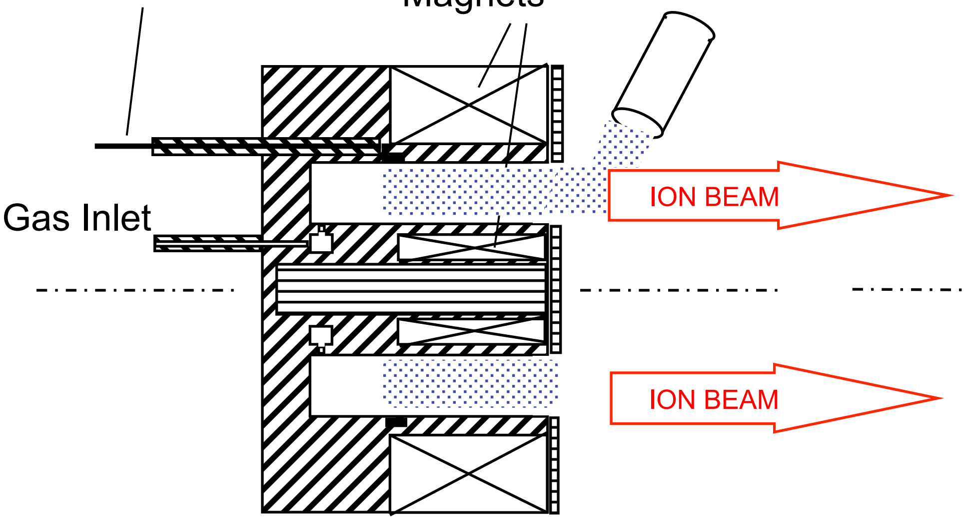
Hall Thruster (2)

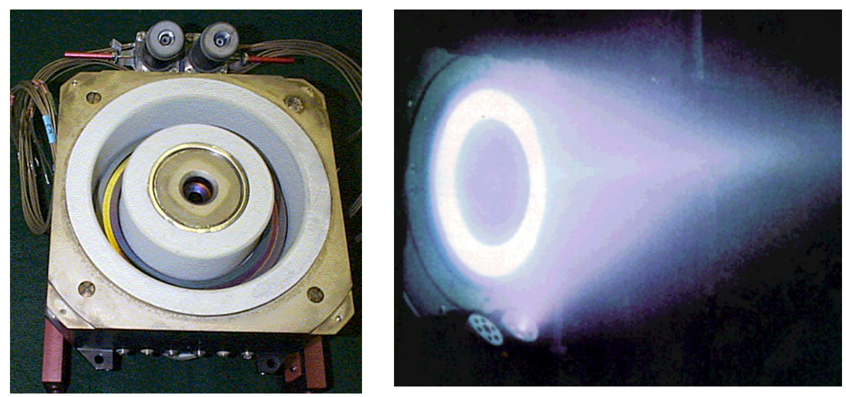
•Principle:
Electromagnetic Acceleration of Ions

•Propellant: Xe, Kr

Isp = 1000-3000 sec
η = 30-60%
Thrust = 5-400 mN
Power = 50W - 4.5 kW

1. Electrons emitted from the cathode travel toward the anode.
2. Electrons are impeded in the discharge channel by a strong radial magnetic field, causing a strong axial electric field to concentrate in this region.
4. This electric field heats the electrons, which subsequently ionize gaseous propellant (xenon) emitted near the anode.
6. The ionized gas accelerates axially through the electric field in the discharge channel, exiting the device at high speed, thus producing thrust.





SPT-140 DM3

Ion Engines

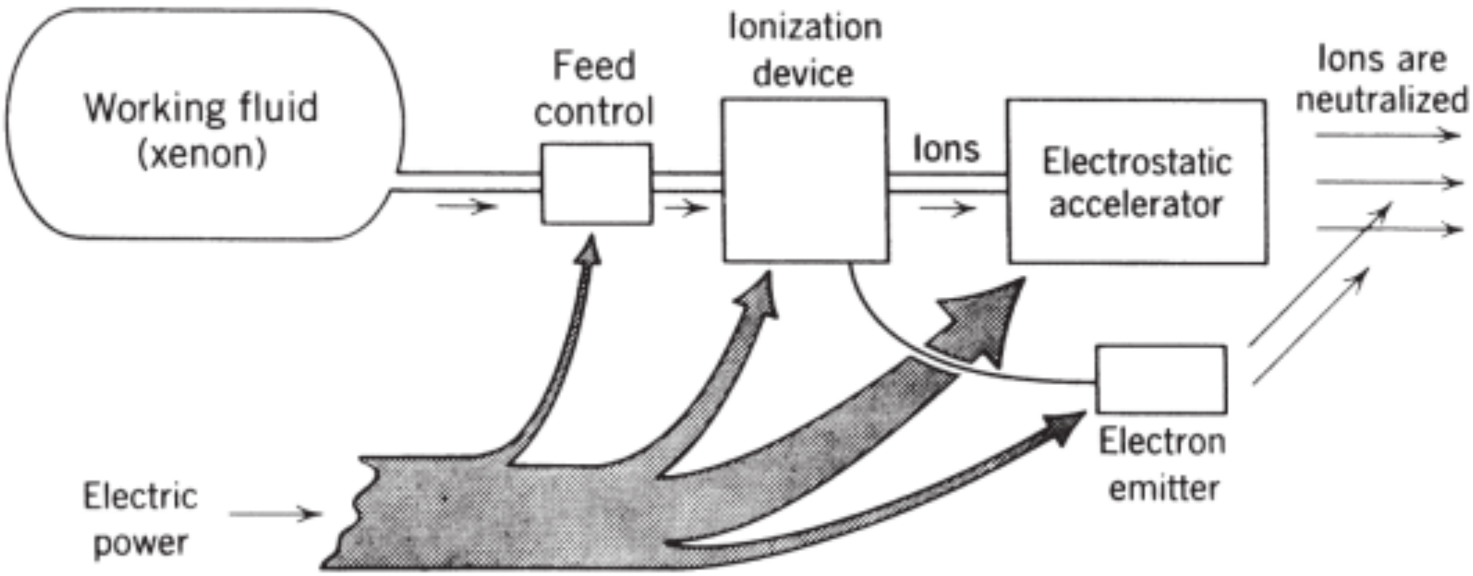
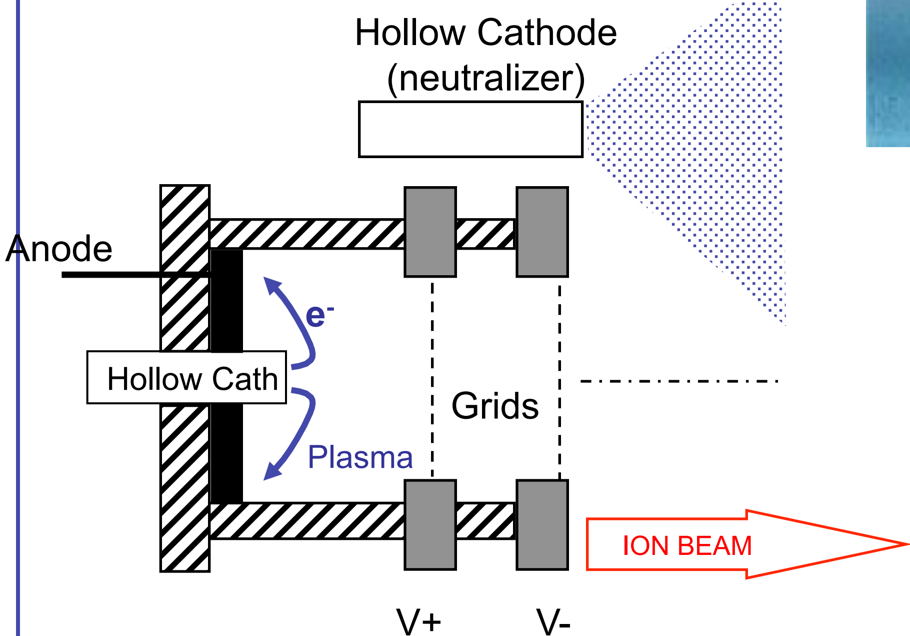
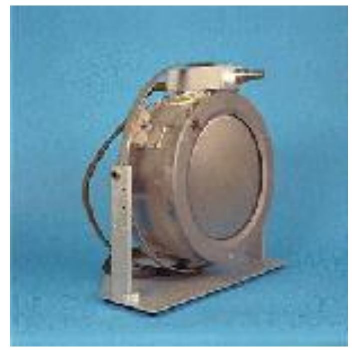


FIGURE 1–9. Simplified schematic diagram of a typical ion rocket, showing the approximate distribution of the electric power.

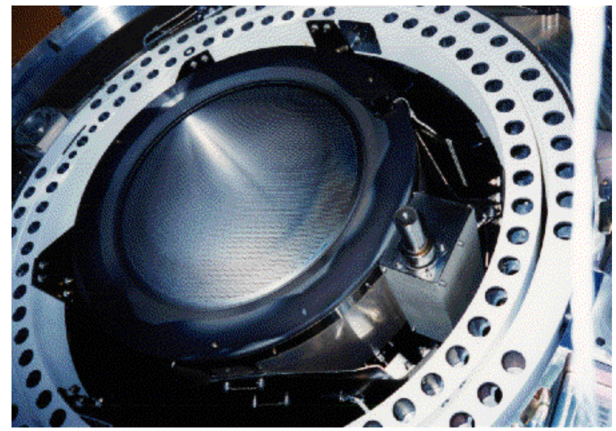
Ion Engines (2)

- Principle:
  Electrostatic Acceleration of Ions
- Propellant: Xe, Kr

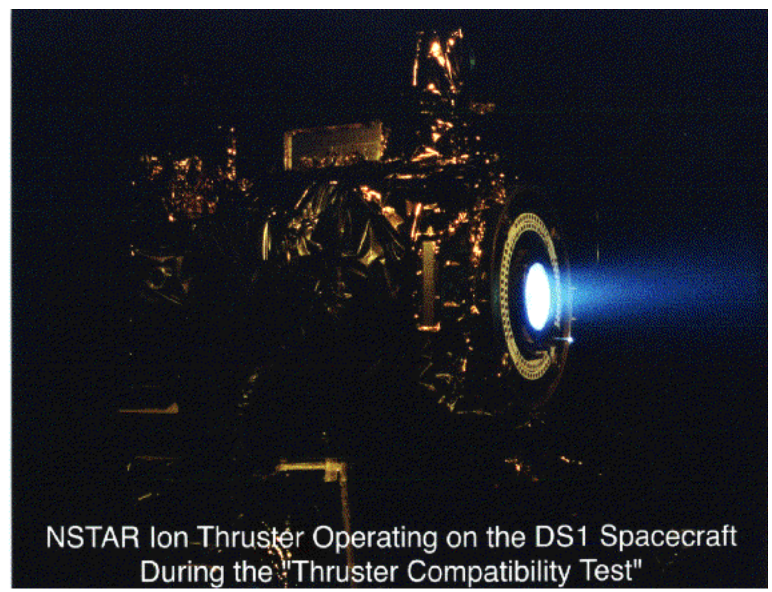




NASA's NSTAR
30cm Ion Engine



Isp = 1500-4000 sec
η = ~65%
Thrust = 1-100 mN



Ion Drive Engines

- Ion thruster is highest TRL of electrically powered spacecraft propulsion.
- Dawn Spacecraft ion drive would require two days to accelerate a car to highway speed.
- Ion Drive thrust levels are limited by the space charge created by ions, and limited available power source.
- Ion thrusters create small thrust levels compared to conventional chemical rockets, but achieve high specific impulse, or propellant mass efficiency, by accelerating the exhaust to high speed.
- Power imparted to the exhaust increases with the square of exhaust velocity while thrust increase is linear.
- Given the practical weight of suitable power sources, the acceleration from an ion thruster is frequently less than one thousandth of standard gravity.
- However, since they operate as electric (or electrostatic) motors, they convert a greater fraction of input power into kinetic exhaust power. Chemical rockets operate as heat engines, and Carnot's theorem limits the exhaust velocity.

Deep Space 1

- First operational demonstration of Ion Drive System.

Launch Date: October 24, 1998

Destinations: asteroid 9969 Braille, comet Borrelly

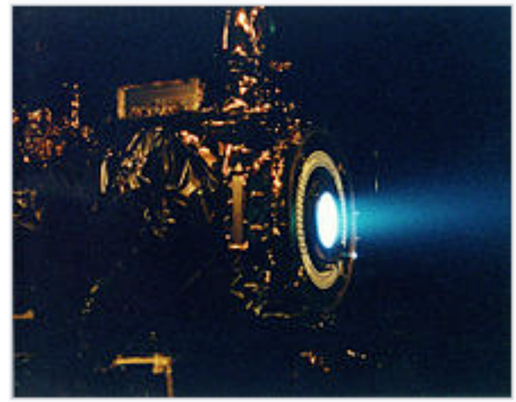
Mission Events

July 29, 1999: Having completed its technology testing within the first couple months after launch, Deep Space 1 makes a bonus flyby of the asteroid 9969 Braille, flying within about 17 miles (27 kilometers) of the object.

November 1999: While embarking on a new journey to comet Borrelly, the spacecraft's star tracker used for determining its orientation in the zero gravity of space fails, nearly ending Deep Space 1's extended mission.

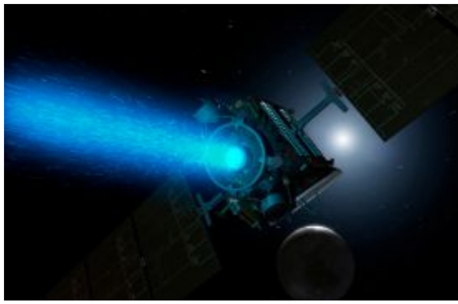
June 2000: Engineers develop a new way to operate the Deep Space 1 spacecraft after the potentially mission-ending failure of its star tracker. Software is radioed to the probe using the camera on board to serve as a replacement navigational tool. The operation marks one of the most successful robotic space rescues in the history of space exploration.

September 2001: Deep Space 1 approaches comet Borrelly, using all of its advanced science instruments to collect important data on the comet's environment and its icy, rocky nucleus. Despite the challenges faced by the spacecraft, it's able to snap the best up-close pictures of a comet to-date.

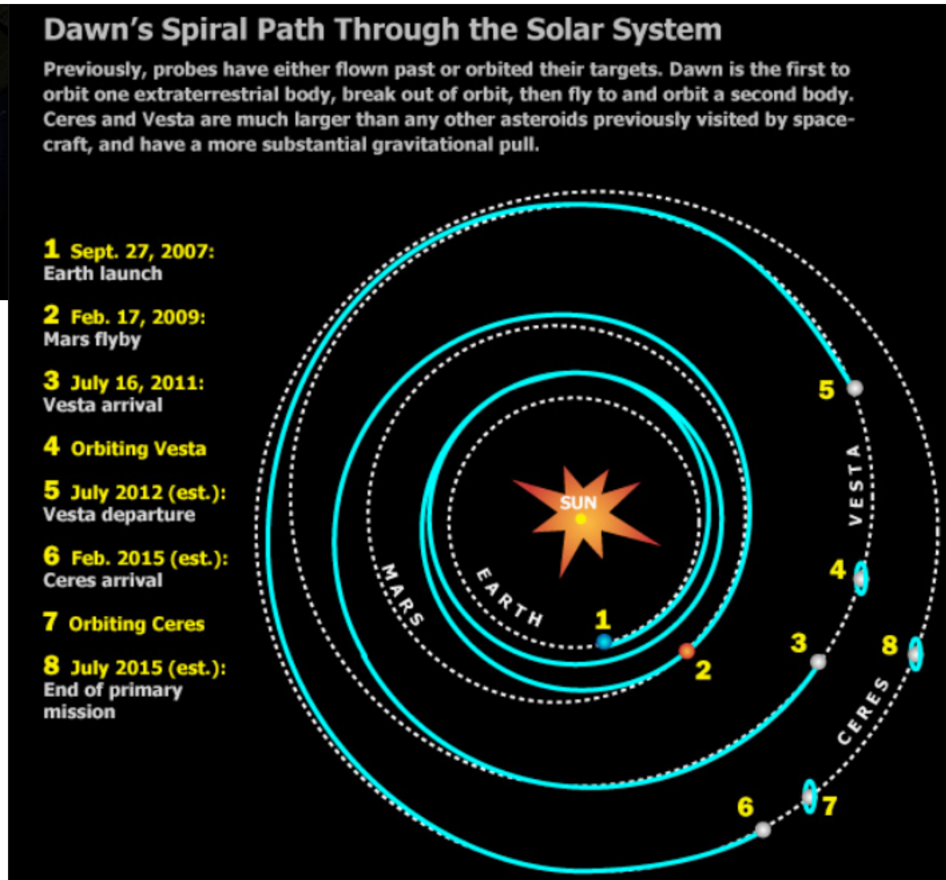


NASA's 2.3 kW NSTAR ion thruster for the Deep Space 1 spacecraft during a hot fire test at the Jet Propulsion Laboratory

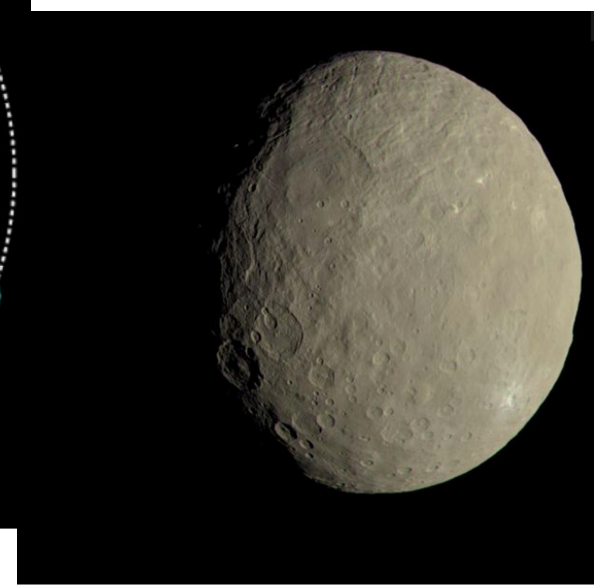
Dawn Spacecraft, Ceres Rendezvous and Orbit



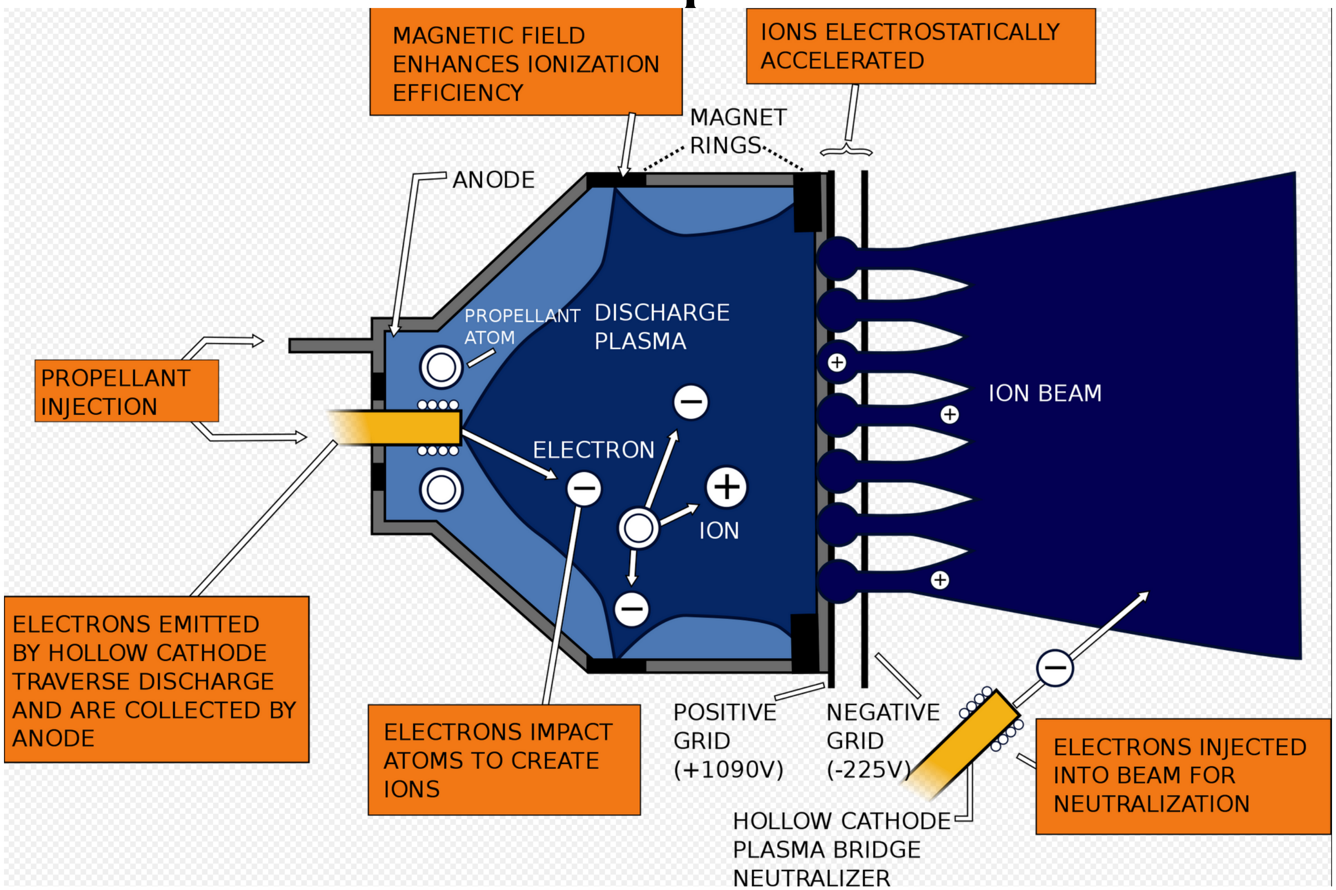
The Deep Space 1 spacecraft, powered by an ion thruster, changed velocity by 4.3 km/s while consuming less than 74 kilograms of xenon. The *Dawn* spacecraft broke the record, with a velocity change of 10 km/s.



- Mission Objectives Achievable ONLY using Ion Drive System



Dawn Spacecraft Ion Drive



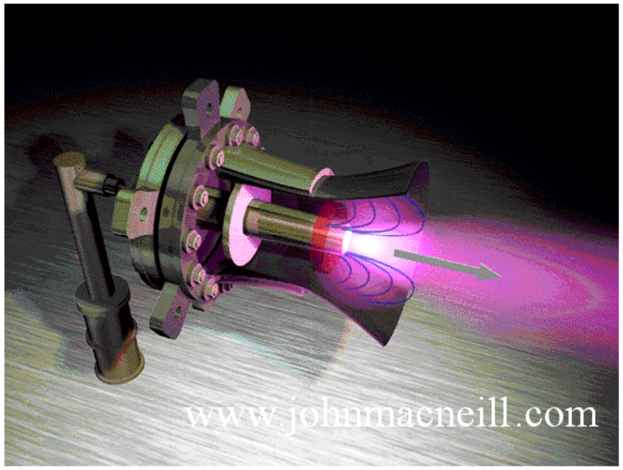
- Dawn ion propulsion system consists of three ion thrusters and is based on the Deep Space 1 spacecraft ion drive, using xenon which is ionized and accelerated by electrodes.
- Xenon ion engines have a maximum thrust at 2.6 kW input power of 92 mN and a specific impulse of 3200 to 1900 s.
- The 30-cm diameter thrusters are two-axis gimbal mounted at the base of the spacecraft.
- Xenon tank held 425 kg of propellant at launch.

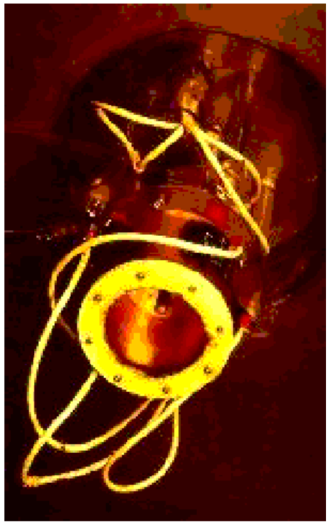
Worked EP Example

• Continuous Thrust GTO

Magnetoplasmadynamic (MPD) Thruster

Isp = ~4500 sec
η = 30%
Thrust = ~1 N (Steady)
~10 N (Pulsed)





lets look at the extreme case (cause I don't want to wait all day for my code to run)

Orbital Initial Conditions

- **Initial Orbit** $\{t_0,\ a_0,\ e_0, \nu_0\}$

$$\begin{bmatrix} r_0 \\ \nu_0 \end{bmatrix} = \begin{bmatrix} \left[\dfrac{a_0\left[1 - e_0^2\right]}{1 + e_0 \cos(\nu_0)} \right] \\ \nu_0 \end{bmatrix}$$

- **Initial Velocity**

$$\begin{bmatrix} V_r \\ V_\nu \end{bmatrix}_0 = r_0\ \omega_0 \begin{bmatrix} \dfrac{\left[\ e_0 \sin(\nu_0)\right]}{\left[1 + e_0 \cos(\nu_0)\right]} \\ 1 \end{bmatrix}$$

Orbital Initial Conditions

- **Initial Angular Velocity**

$$\omega_0 = \frac{\sqrt{\mu}}{\left[a_0\left[1 - e_0^2\right]\right]^{3/2}}\left[1 + e_0 \, \mathrm{Cos}\left(\nu_0\right)\right]^2$$

- $M_0 \equiv$ Initial Mass

• Continuous Thrust GTO

Thrust (Newtons)

10.00000

Isp (seconds)

2500.0

• MPD Thruster

• **Initial Spacecraft Mass**

1000kg

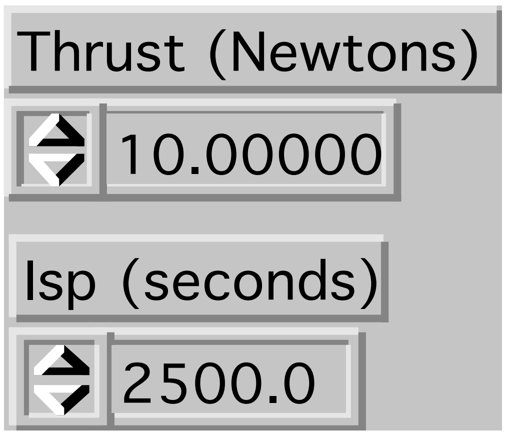
• **Initial Orbit**

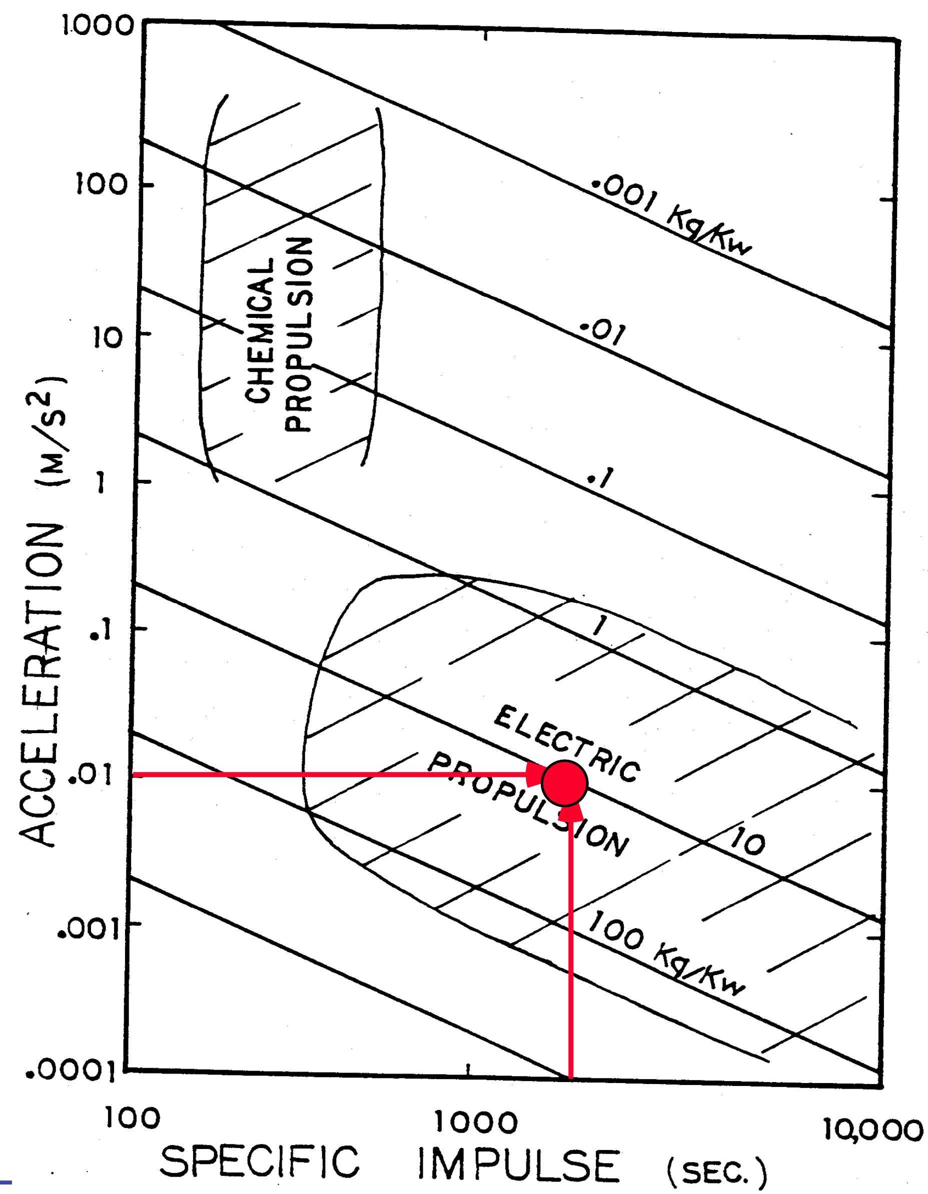
6571 km, e=0.0

• **Initial Orbit Velocity**

7.7885 km/sec

Worked Example (cont'd)





- **Initial Spacecraft Mass**

  1000kg

Worked Example (cont'd)

• Continuous Thrust GTO

Thrust (Newtons)

10.00000

Isp (seconds)

2500.0

• MPD Thruster

Accumulated burn time (sec.)

418000.00 116 hrs

• **Terminate Thrust when**

a(1+e) **Instantaneous**

= 42164.2 km (Geo radius)

• **Final Orbit**

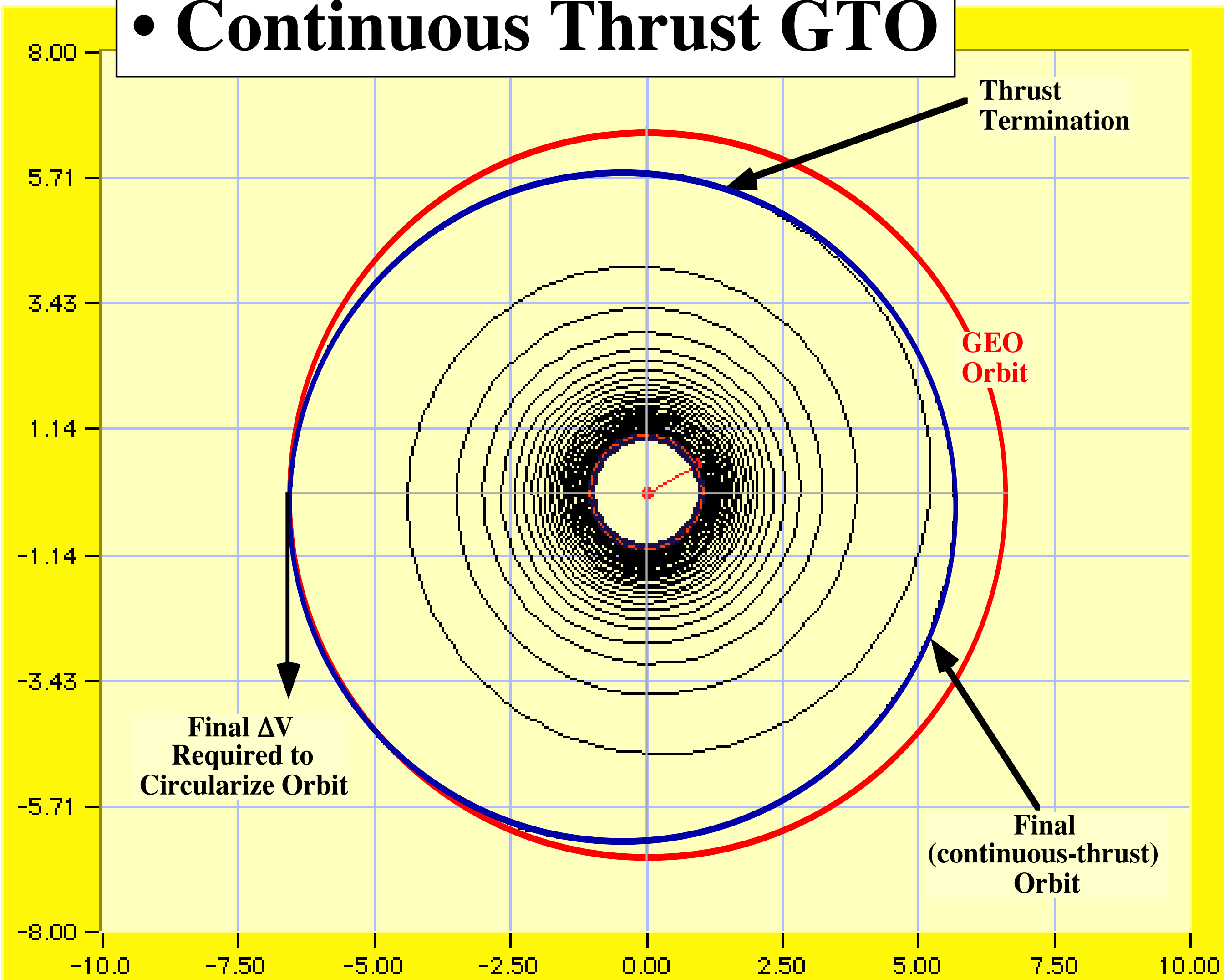
a = 38830 km

• **Final Orbit**

e = 0.08584

Worked Example (cont'd)

• **Continuous Thrust GTO**



Worked Example (cont'd)

• Continuous Thrust GTO

Thrust (Newtons)
10.00000

Isp (seconds)
2500.0

• MPD Thruster

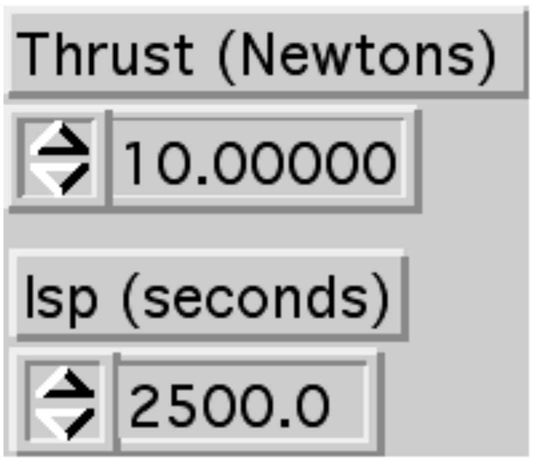
• **Propellant Required to Reach Final GTO (elliptical)**

$$M_{initial} = 1000 \text{ kg}$$
$$M_{final} = 829.5 \text{ kg}$$

$$P_{ropellant}\, M_{ass} = 170.5 \text{ kg}$$

Worked Example (cont'd)

• Continuous Thrust GTO



• MPD Thruster

• ΔV required to circularize final orbit

$$V_{GEO} = \sqrt{\frac{\mu}{r}} =$$

$$\sqrt{\frac{3.986 \times 10^5 \ \frac{\text{km}^3}{\text{sec}^2}}{42164.2 \text{ km}}} = 3.0746 \ \frac{\text{km}}{\text{sec}}$$

Worked Example (cont'd)

• Continuous Thrust GTO

Thrust (Newtons)

10.00000

Isp (seconds)

2500.0

• MPD Thruster

• ΔV required to circularize final orbit

$$V_{GTO}^{(apogee)} = \sqrt{\frac{2\mu}{r} - \frac{\mu}{a}} =$$

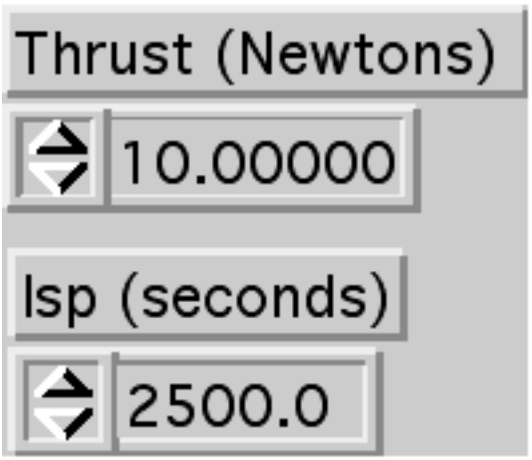
$$\sqrt{\frac{2 \times 3.986 \times 10^5 \frac{km^3}{sec^2}}{42164.2\ km} - \frac{3.986 \times 10^5 \frac{km^3}{sec^2}}{38830\ km}} = 2.9397\ \frac{km}{sec}$$

$$\Delta V = 3.0746\ \frac{km}{sec} - 2.9397\ \frac{km}{sec} = 0.135\ \frac{km}{sec}$$

Worked Example (cont'd)

- **Likely Need Conventional Propulsion for Final Burn**

  - **I_{sp} 270 sec**

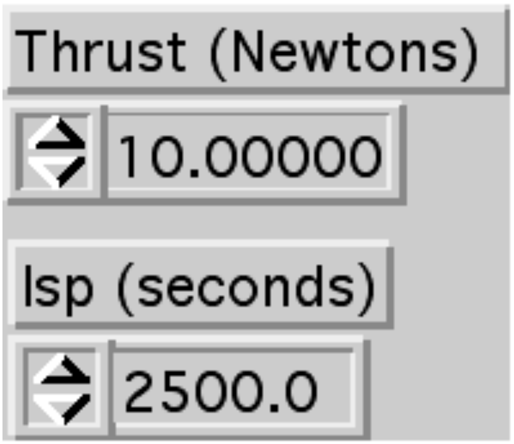


- MPD Thruster

$$P_{mf} = e^{\frac{\Delta V}{g_0 I_{sp}}} - 1 =$$

$$e^{\left[\frac{135.0 \frac{m}{sec}}{9.806 \frac{m}{sec^2} 270 \text{ sec}}\right]} - 1 = .0523$$

Worked Example (cont'd)



- **MPD Thruster**

- **Conventional Propulsion Final Burn**

$$P_{mf} + 1 = \frac{M_{propellant}}{M_{final}} + \frac{M_{final}}{M_{final}} =$$

$$\frac{M_{propellant} + M_{final}}{M_{final}} \Rightarrow 1.0523 = \frac{829.5 \text{ kg}}{M_{final}}$$

$$\Downarrow$$

$$M_{final} = \frac{829.5 \text{ kg}}{1.0523} = 788.2 \text{ kg} \Rightarrow$$

$$M_{propellant} = 829.5 \text{ kg} - 788.2 \text{ kg} = 41.2 \text{ kg}$$

Worked Example (cont'd)

- **Total Propellant Mass Fraction for GEO Transfer**

Thrust (Newtons)
10.00000
Isp (seconds)
2500.0

- Continuous Thrust

$$P_{mf} = \frac{M_{propellant}}{M_{final}} =$$

- MPD Thruster

$$\frac{41.2\ \mathrm{kg} + 170.5\ \mathrm{kg}}{788.2\ \mathrm{kg}} = 0.26858\ wow!$$

Compare to Hohmann transfer using Conventional Propulsion

- I_{sp} 270 sec

"delta Vee" data

DV Orbit 1 (KM/sec)	2.45536
DV Orbit 2 (KM/sec)	1.47723
DV Total (KM/sec)	3.93259

- WHAT IS PROPELLANT FRACTION?

$$P_{mf} = e^{\frac{\Delta V}{g_0 I_{sp}}} - 1 =$$

$$e^{\frac{3932.59 \frac{m}{sec}}{9.806 \frac{m}{sec^2} 270\ sec}} - 1 = 3.4164$$

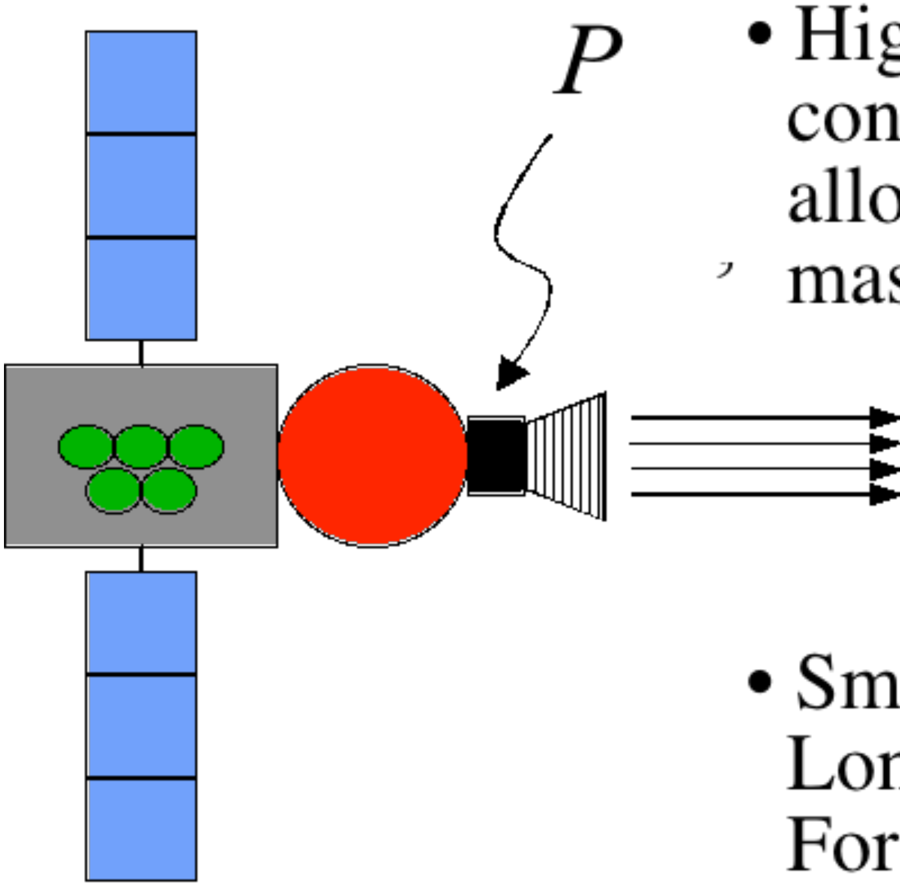
- Final Mass 788.2 kg

requires ... 2692.8 of propellant!

Versus 211.7 kg for EP

EP, in the Right Circumstances

Big Advantages

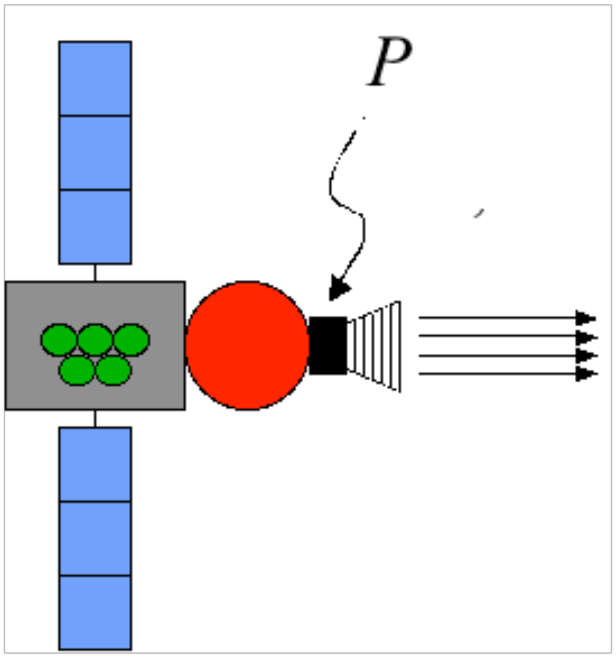


- High Isp
  continuous thust
  allows small propellant
  mass fractions

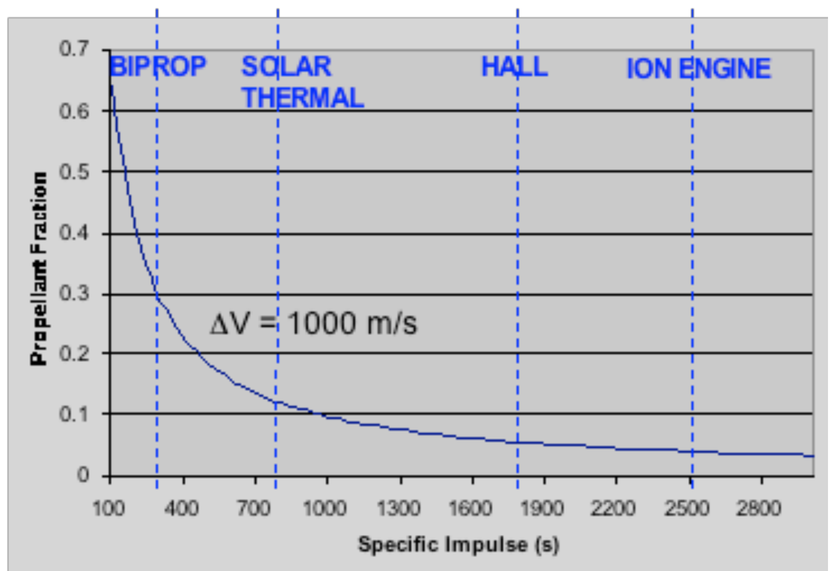
- Small Thrust Requires
  Long Operating Life
  For Engine

EP, in the Right Circumstances

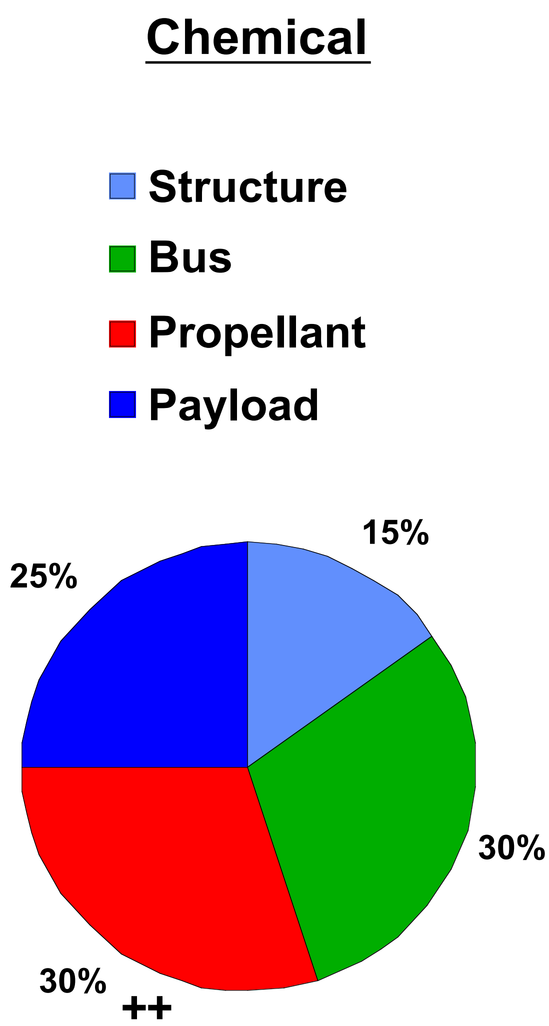
Benefits of Electric Propulsion

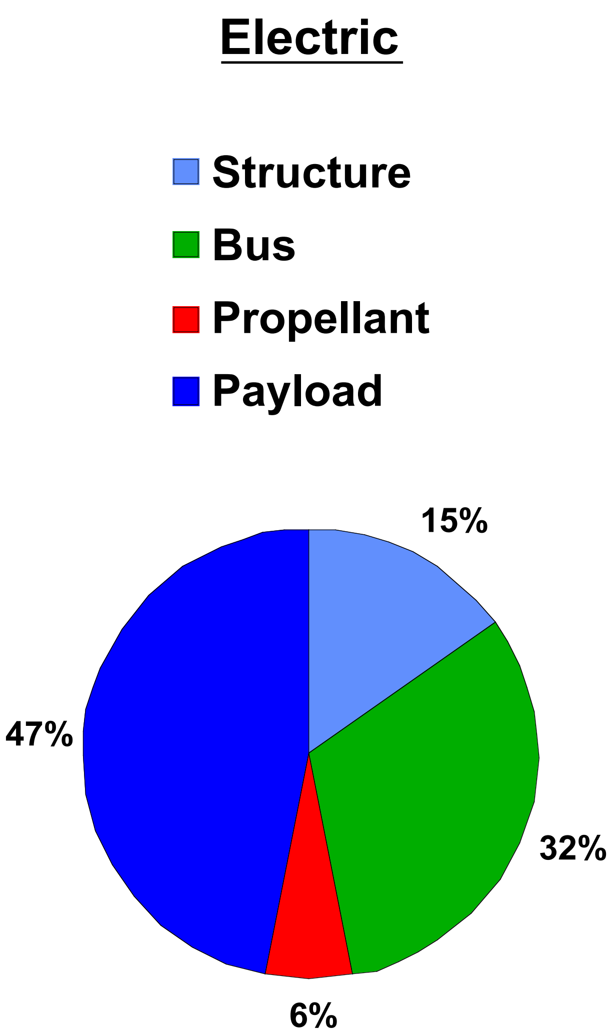


Chemical	**400**
Solar Thermal	**800**
Nuclear Thermal	**800+**
Electric	**ANY**



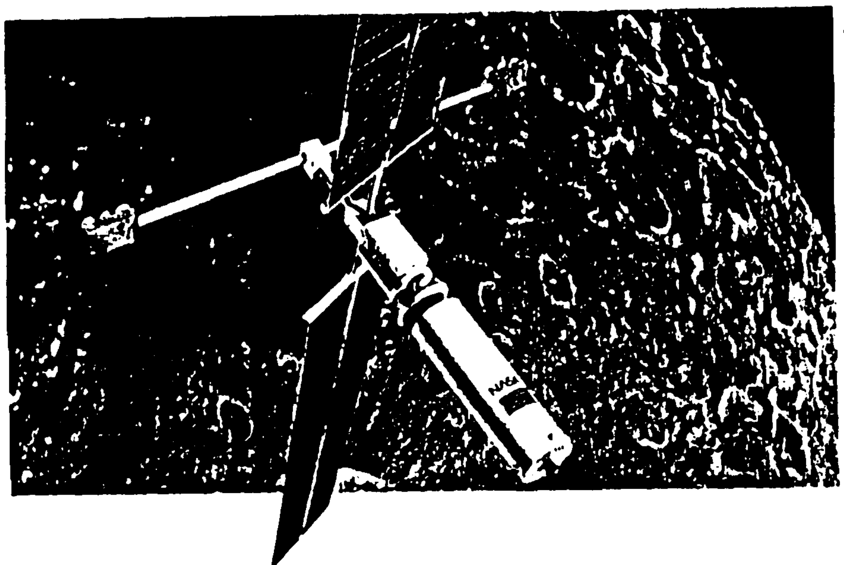
Benefits of Electric Propulsion





How about EP driven Ferry to the Moon?

ION-PROPELLED LUNAR FERRY



- 77,000 POUND SPACECRAFT
- POWERED BY 300 KILOWATT SOLAR ARRAYS
- 60% PAYLOAD MASS FRACTION!
- LOW ALTITUDE EARTH ORBIT TO LUNAR ORBIT

MISSION CHARACTERISTICS

- 4 SOLAR-ELECTRIC CARGO CARRIERS
- OPERATING BETWEEN THE EARTH AND THE MOON
- 44,000 POUND DELIVERED TO THE MOON EVERY 100 DAYS

PROPULSION SYSTEM

- SOLAR ARRAYS 40 FT x 200 FT
- XENON PROPELLANT
- ROUND-TRIP TRAVEL TIME = 100 DAYS
- ION ENGINE SPECIFIC IMPULSE = 4,500 SEC

a Continual round Trips between earth and Moon

Other Forms of Propulsion

You're going to bolt a JATO bottle to a car and jump the Snake River???

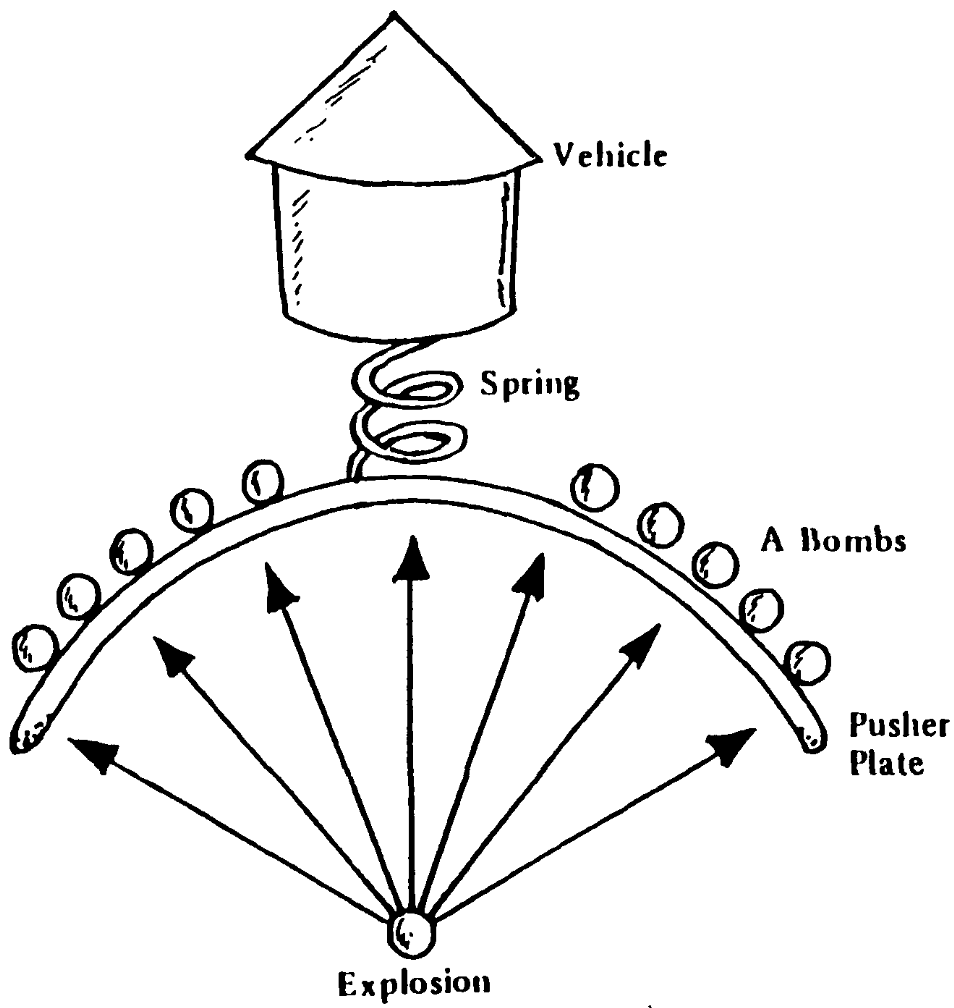
- This guy wins a Darwin Award!

Project Orion

- Pulse rocket, propelled by multiple nuclear explosions (Ganswindt's dynamite propelled craft?)
- Thirty one-kiloton explosions per second, exploding 150 to 1000 ft behind the ship.
- ARPA and NASA both seriously studied it.
- A 2,500,000 lb spaceship would be assembled in orbit (eight Saturn V flights) and with 2,000 atomic bombs, would carry 20 people to Mars on a 250 day round trip.
- I_{sp} between 1,850 and 2,550
- Ran out of money in 1965

- Another Darwin Award!

Project Orion (Ouch!)

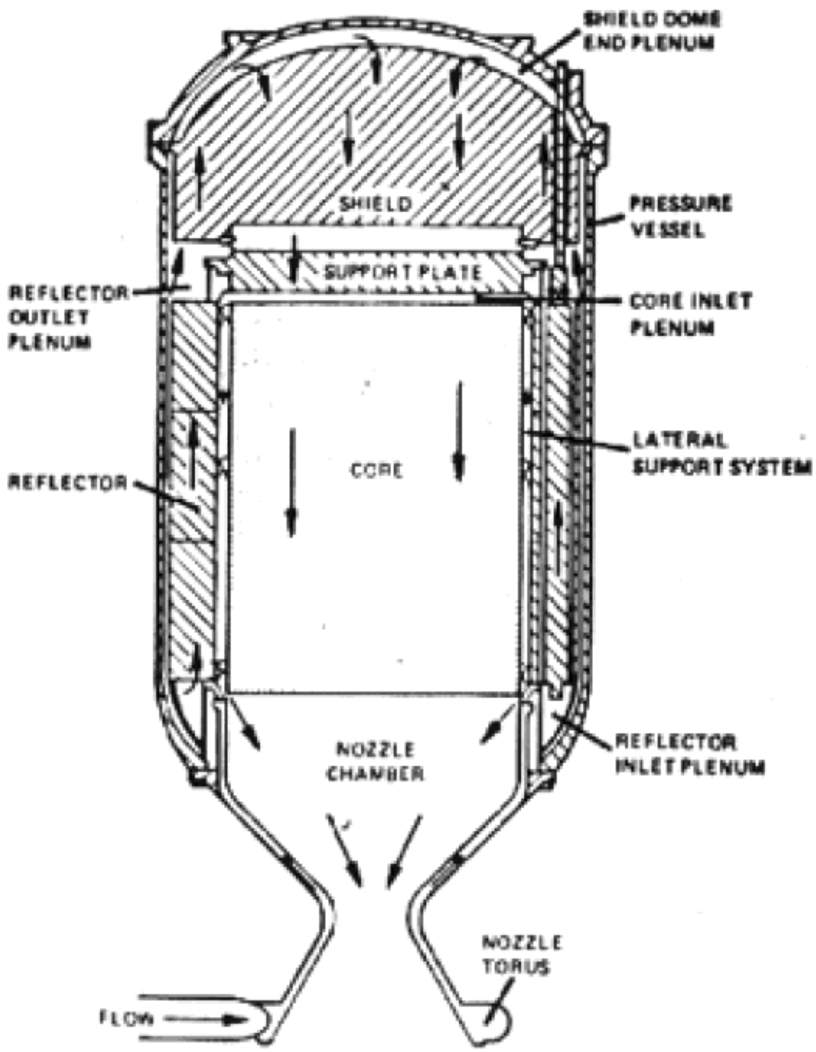


More Realistic Nuclear Propulsion

(When we glow in the dark, we'll be there)

- Goddard and Tsiolkovsky both suggested atomic rockets
- The Air Force and the AEC started serious development in 1955
- Theoretically, the temperatures generated by a reactor can heat propellant and produce exhaust velocities of twice those of chemical rockets. I_{sp} in the range of 1,000.

Nuclear Propulsion



Nuclear Engine for Rocket Vehicle Application (NERVA)

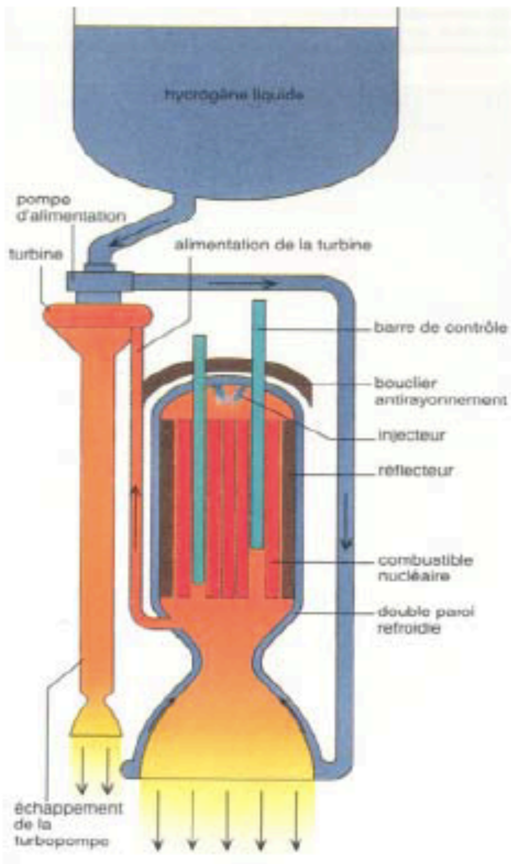
- High Energy Density

May enable High I_{sp} And moderate thrust

- .. Or High Thrust and Moderate I_{sp}

Nuclear Propulsion (2)

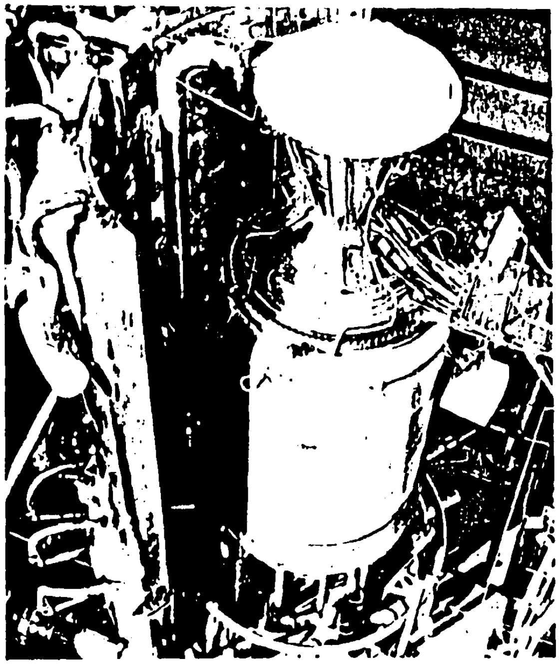
Nuclear-Thermal Propulsion

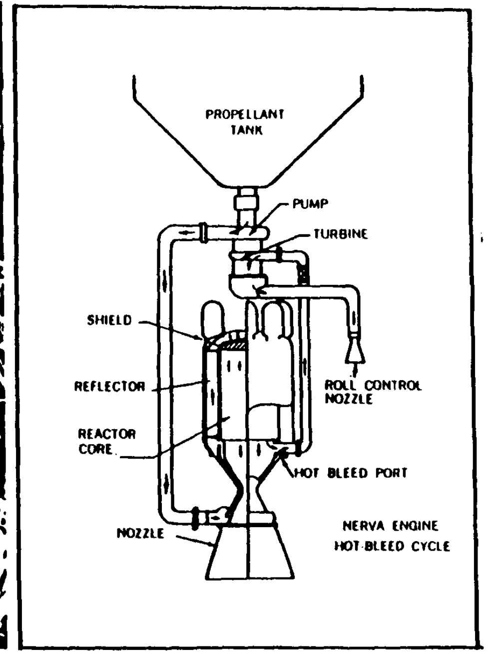


(Courtesy of SNECMA)

- **Concept: There are two main different categories of nuclear technology for space power and propulsion:**
  - **radioisotope thermoelectric generators (RTG) and close-cycle (e.g. Sterling technology) for nuclear electric power, NEP, to power electric propulsion**
  - **open-cycle nuclear thermal reactors, NTR, which heat e.g. liquid hydrogen propellant directly to produce rocket thrust**
- **NEP: Flight heritage of RTG's with power level < 10 kWe while future NEP's aim at 10 kWe to MWe's for electric propulsion: v_e = 20 000 m/s to 100 000 m/s (FEEP)**
- **NTR: liquid hydrogen propellant absorbs heat from the core of a fission reactor, before expanding through a nozzle: v_e = 8000 m/s to 9000 m/s, F = 20kN to 70 kN**
- **Extensive research performed into nuclear-thermal rockets in U.S. in 1960 as part of the NERVA program.**
- **Status: Environmental and political concern about save ground test and launch of fueled reactor has reduced research in NEP and NTR technology.**

Nuclear Engine for Rocket Vehicle Application (NERVA)

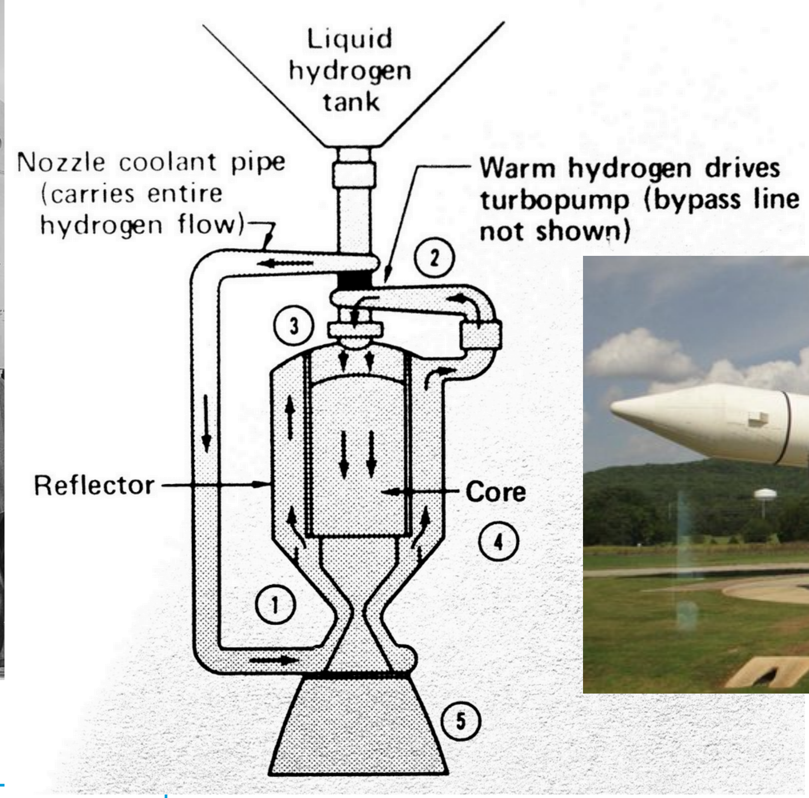


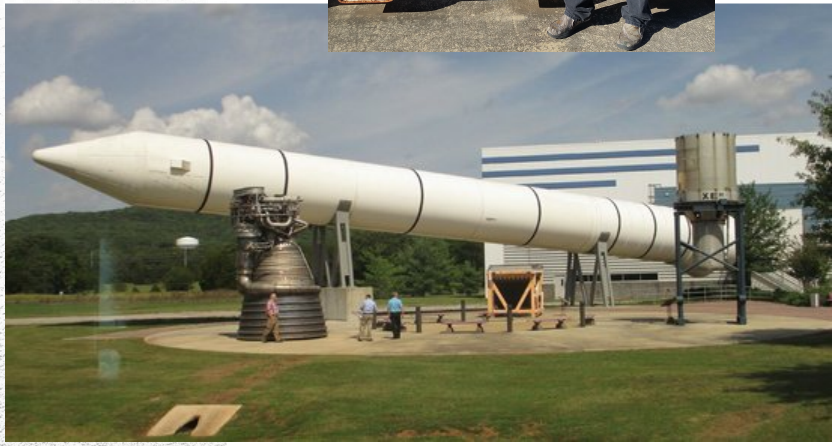


EARLY NUCLEAR-POWERED ROCKETS

- NERVA = EARLY VERSION = 75,000 POUNDS OF THRUST
- ULTIMATE TARGET THRUST = 200,000 TO 250,000 POUNDS
- SPECIFIC IMPULSE = 700 SECONDS
- TESTED AT JACKASS FLATS, NEVADA, LATE 1960's

NERVA Rocket





"Rocket Garden" Bldg 4205 MSFC

Example Nuclear Transfer Problem Specification

- Transfer from LEO (200 km altitude) to GEO, both orbits equatorial

- Use Nuclear Rocket, Ionized H Fuel, $u_{exit} \sim 10^5$ m/sec

- Required Transfer time $\leq$ 100 hours

- Dry Vehicle Mass, 50,000 kg

Compute Radius of Geo Orbit

- Kepler's Third law

$$\boxed{T = \frac{2\,\pi\, a^{3/2}}{\sqrt{\mu}}}$$

$$\boxed{T_{geo} = 23 \text{ hrs } 56 \text{ min } 4.1 \text{ sec } = 86164.1 \text{ sec}}$$

$$a_{geo} = \left[\frac{\sqrt{\mu}\, T_{geo}}{2\,\pi}\right]^{2/3} =$$

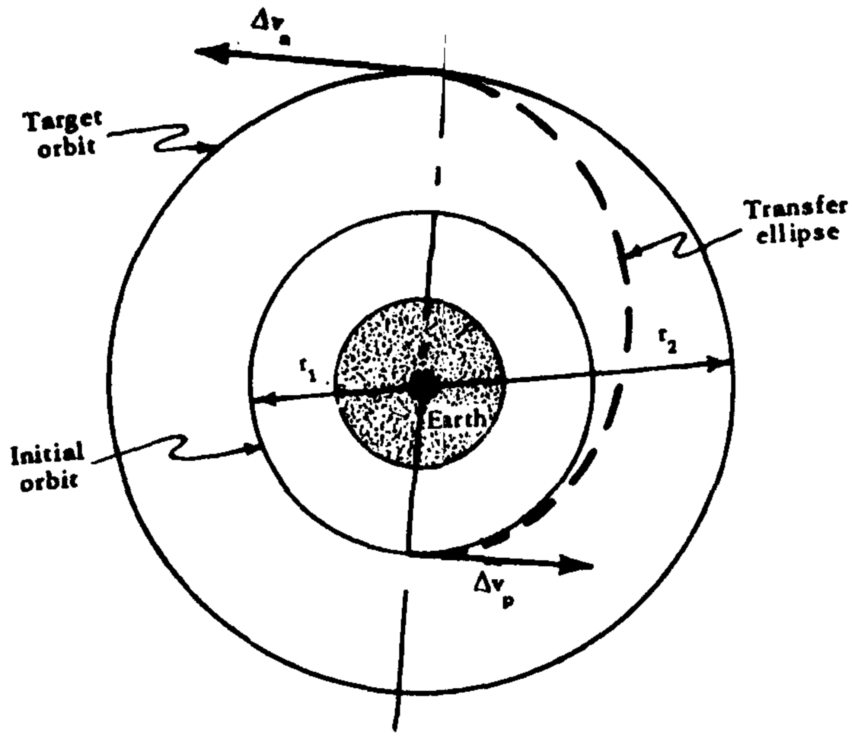
$$\left[\frac{\sqrt{3.986 \times 10^5 \frac{km^3}{sec^2}} \times 86164.1 \text{ sec sec}}{2\,\pi}\right]^{2/3} = \boxed{42164.2 \text{ km}}$$

Hohmann Transfer from LEO to GEO

$$R_{LEO} = 6378 + 200 = 6578_{km}$$

$$R_{GEO} = 42164.2_{km}$$

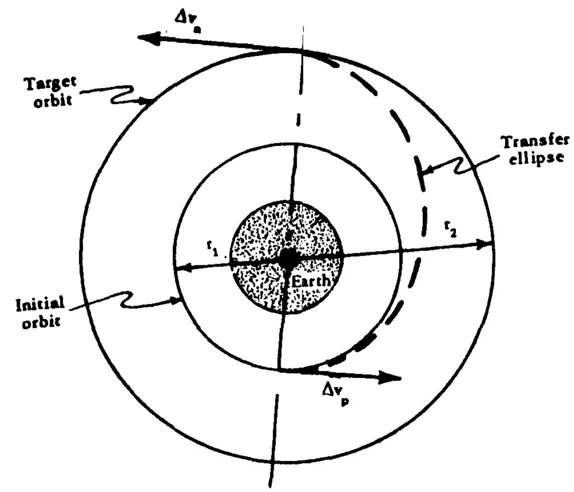
$$a_{transfer} = \frac{42164.2_{km} + 6578_{km}}{2} = 24351.1_{km}$$



Hohmann Transfer from LEO to GEO (cont'd)

- Earth Departure

(Impulsive Burn, Infinite Thrust)



$$\Delta V_1 = \sqrt{\frac{2\mu}{R_{LEO}} - \frac{\mu}{a_{transfer}}} - \sqrt{\frac{\mu}{R_{LEO}}} =$$

$$\left(2\frac{3.9860044\cdot 10^5}{6578} - \frac{3.9860044\cdot 10^5}{\frac{42164.2+6578}{2}}\right)^{0.5} - \left(\frac{3.9860044\cdot 10^5}{6578}\right)^{0.5}$$

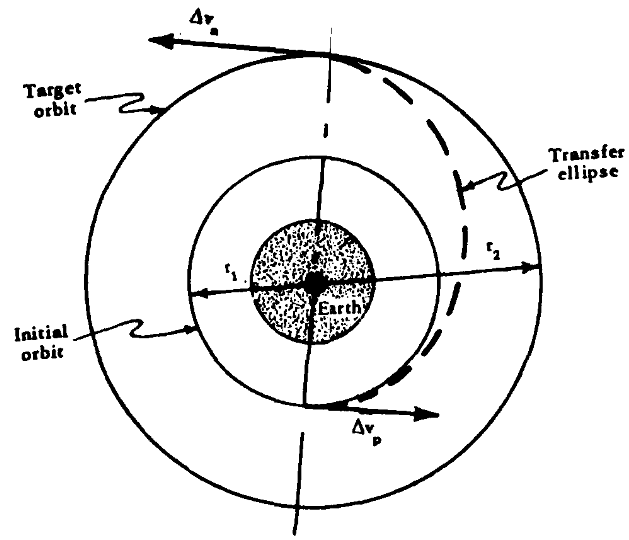
$= 2.45463$ km/sec

Hohmann Transfer from LEO to GEO (cont'd)

- GEO Arrival

(Impulsive Burn, infinite thrust)

$$\Delta V_2 = \sqrt{\frac{\mu}{R_{GEO}}} - \sqrt{\frac{2\mu}{R_{GEO}} - \frac{\mu}{a_{transfer}}} =$$

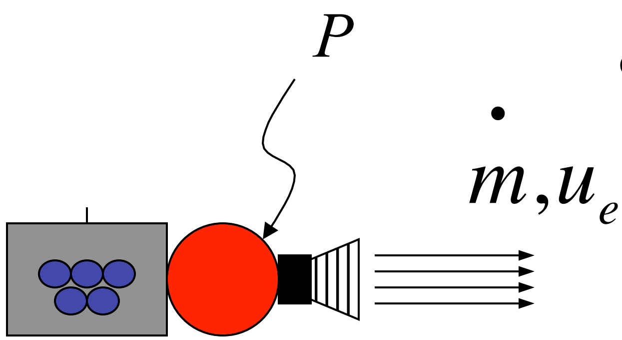


$$\left(\left(\frac{3.9860044 \cdot 10^5}{42164.2}\right)^{0.5}\right) - \left(\left(2\frac{3.9860044 \cdot 10^5}{42164.2} - \frac{3.9860044 \cdot 10^5}{\frac{42164.2 + 6578}{2}}\right)\right)^{0.5}$$

$= 1.47729$ km/sec

- Total Delta V *(2 Impulsive Burns) = 3.9319 km/sec*

Nuclear Rocket Thrust Modeling



- ***Exit Pressure thrust negligible***

$$F = g_0 \dot{m} I_{sp} \rightarrow F = \dot{m} u_e \rightarrow I_{sp} = \frac{u_e}{g_0}$$

$$\boxed{u_e = 1 \times 10^5{}_{m/\sec} \quad \rightarrow I_{sp} \sim 10{,}200_{\sec}}$$

Notional Engine

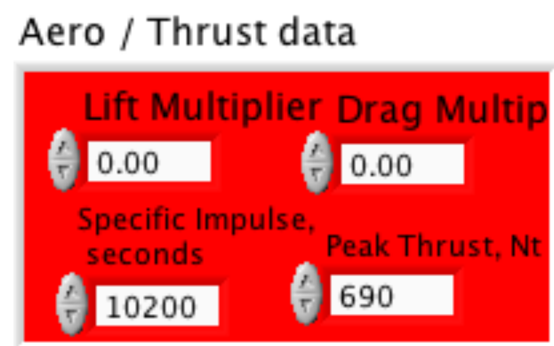
- **Impulsive Burn Propellant requirements**

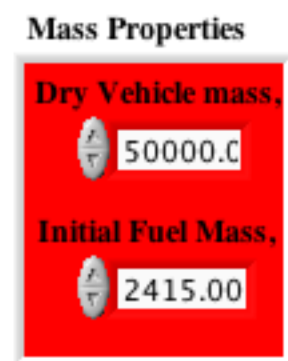
$$M_{prop} = M_{dry}\left(e^{\frac{\Delta V}{g_0 I_{sp}}} - 1\right) = 50000\left(\exp\left(\frac{3.93191 \cdot 1000}{9.8066 \cdot 10200}\right) - 1\right)$$

$$= 2004.558 \text{ kg}$$

Finite Thrust Transfer

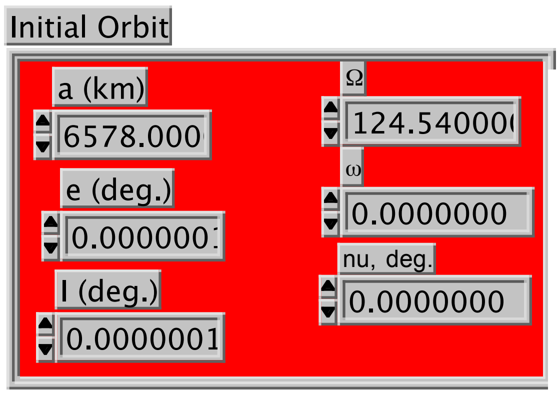
- Initial Orbit Radius 6578 km, Final Orbit radius 42164.2 km
- $I_{sp} \equiv$ 10,200 sec
- Transfer time $\equiv$ 100 hours (360,000 sec)
- Target Longitude $\equiv$ 32 deg east
- Dry Vehicle mass $\equiv$ 50,000 kg
- Assumed initial fuel mass, 2415 kg
- Constant Thrust, 690 Nt

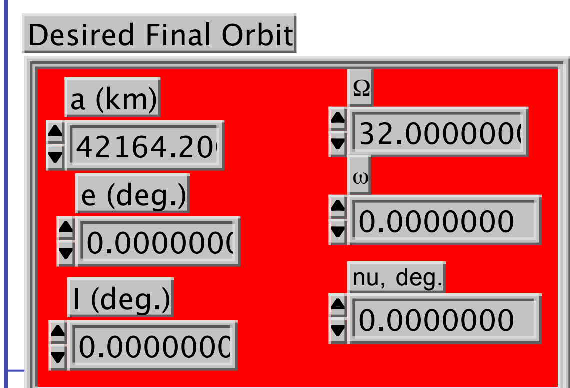


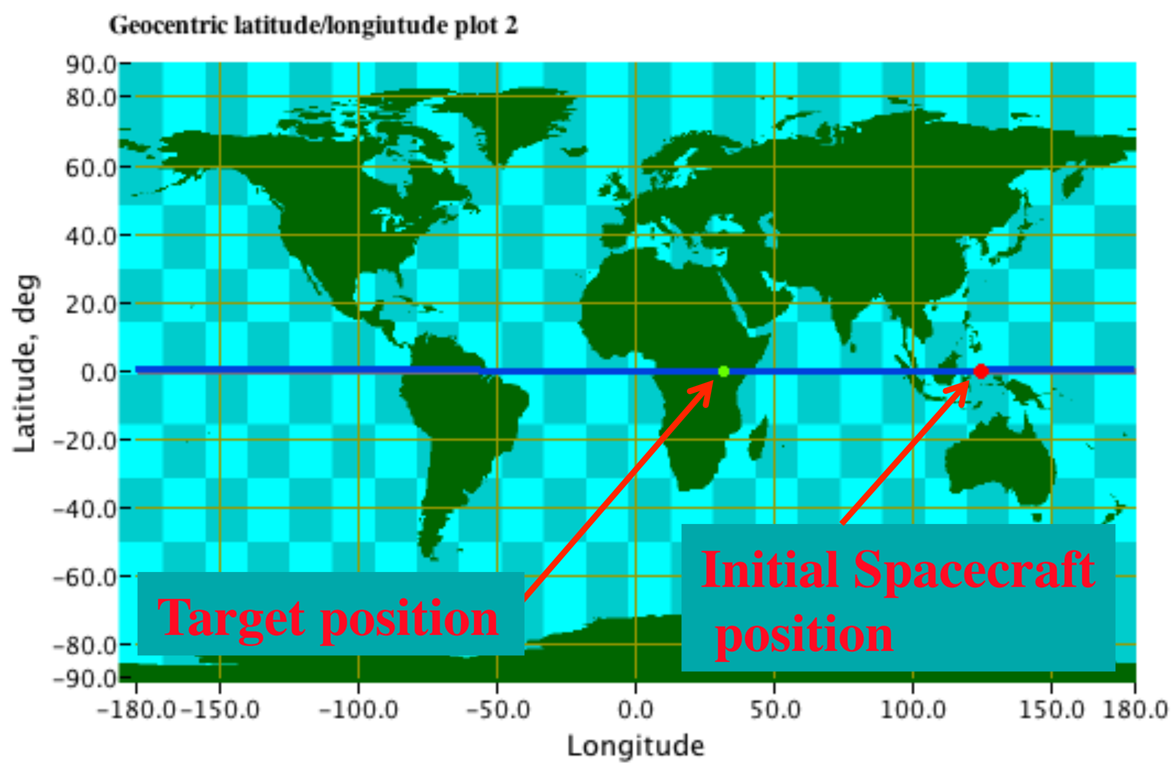


Initial & Final Orbit, Keplerian Elements

Simulation Starts at 0z Greenwich Mean Sidereal Time







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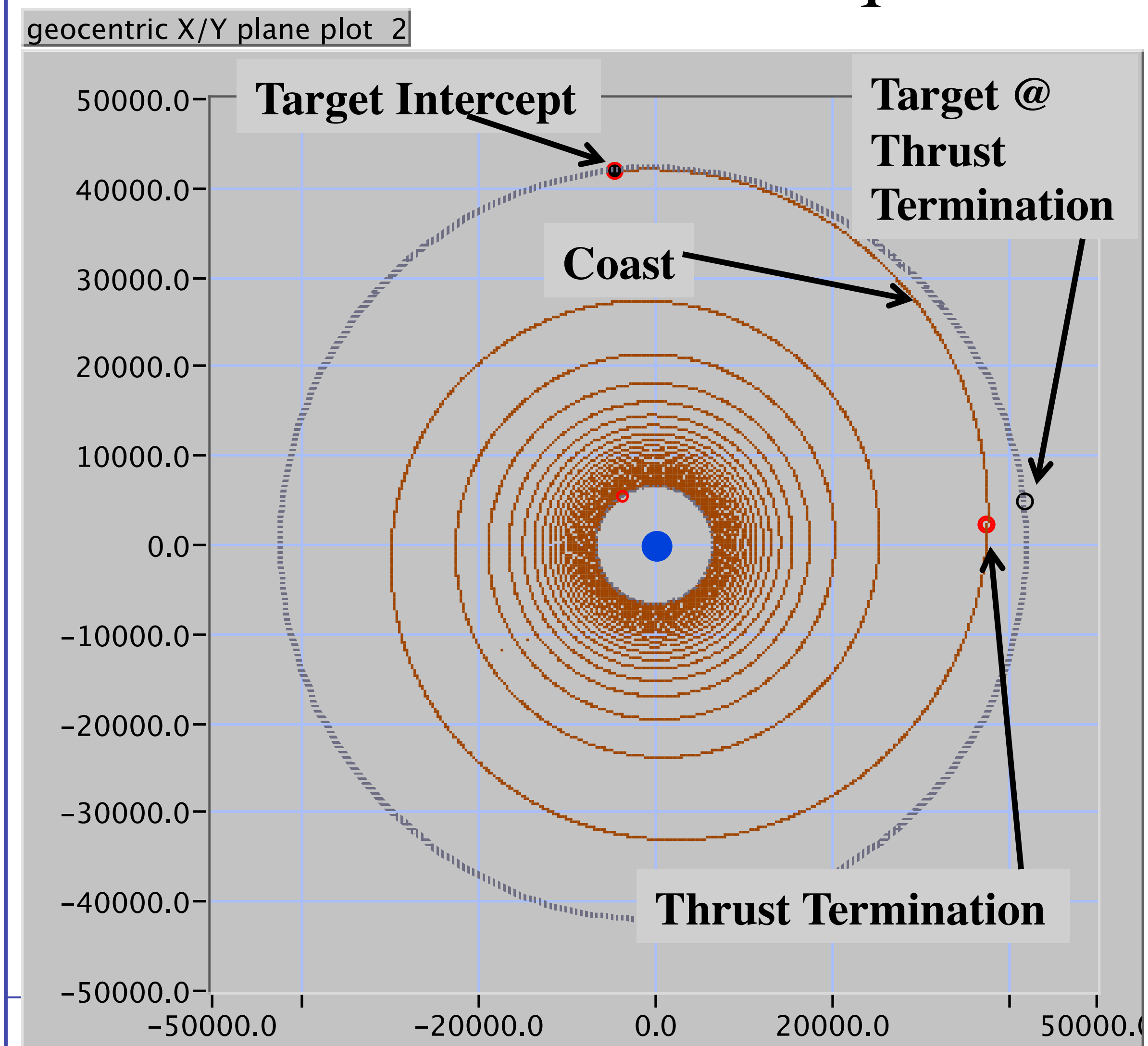
Spiral Transfer Plot, Looking Down on Equatorial plane



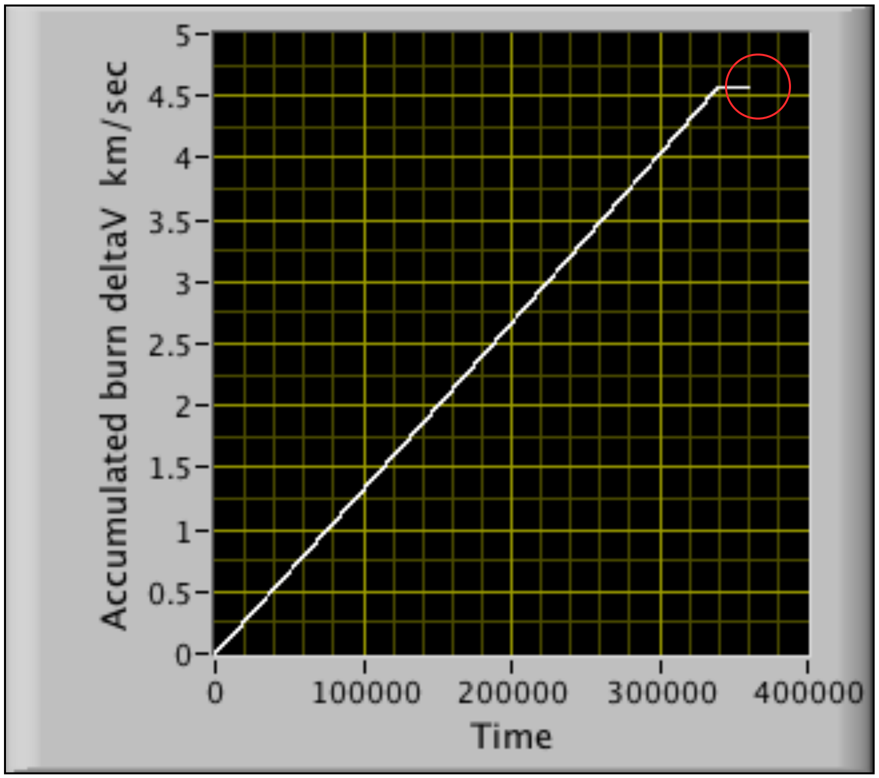
Intercept Time, sec	Thrust Termination Time, sec
360050	338550

Propogated State Vector out

Vr (km/sec)	0.01968
Vv (km/sec)	2.92449
Orbit Pseudo-Inclination (deg)	0.00
Geocentric Radius (km)	42168.671
True Anomaly from Initial Orbit	9691.7698
Current Vehicle mass (kg)	50079.982

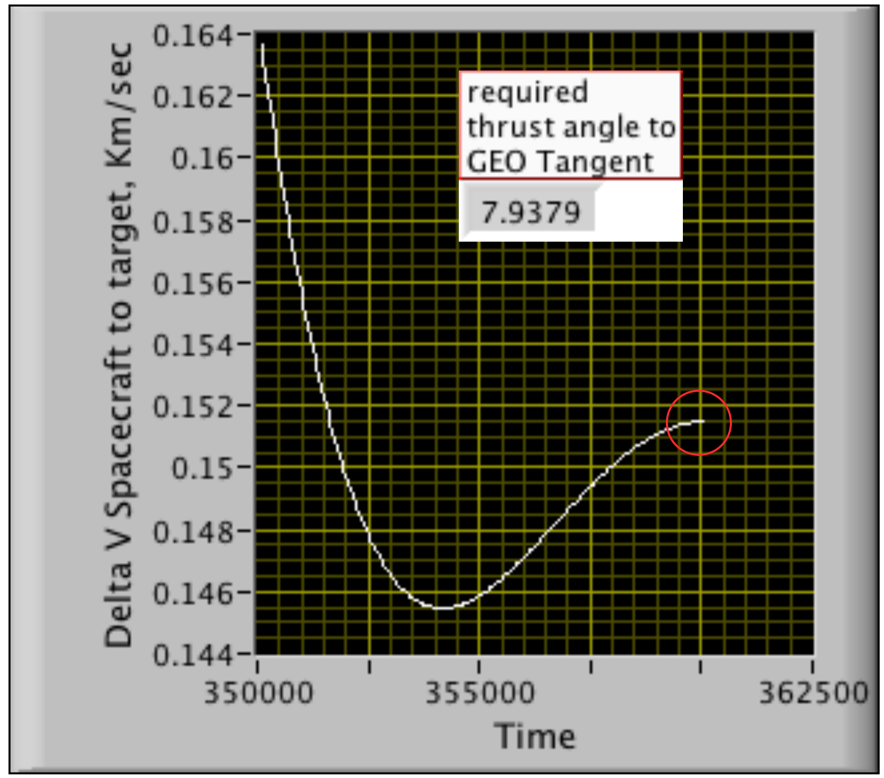
Delta V/Propellant Budget

- **Accumulated Delta V During Constant Thrust Segment of Transfer**



Consumed propellant: 2335.02 kg

- **Residual Delta V at Target Intercept**



Consumed propellant: 75.8 kg

Delta V/Propellant Budget (cont'd)

- **Total Required Delta V: 4.7012 km/sec**
- **Total Consumed Propellant: 2410.82 kg**

$$\Delta V(t) = \left(g_0 I_{sp}\right)\cdot Ln\left[\frac{M_{initial}}{M_{final}}\right] = \left(g_0 I_{sp}\right)\cdot Ln\left[\frac{M_{initial}}{M_{initial} - \int_0^t \left(\dot{m}\right)\cdot d\tau}\right]$$

$$\rightarrow \dot{\Delta V}_{(t)} = \frac{\left(g_0 I_{sp}\right)\cdot\left(\dot{m}\right)}{M_{initial} - \left(\dot{m}\right)\cdot t} = \frac{F_{thrust}(t)}{M_{initial} - \frac{F_{thrust}(t)}{g_0 I_{sp}}}$$

Power Budget

$$P_{in} = \frac{\dot{m}u^2}{2\eta} = \frac{Fu}{2\eta}$$

$$Assume: \eta = 0.7 \rightarrow P_{in} = \frac{690_{Nt} \times 10{,}200_{sec} \times 9.8066_{m/sec}}{2 \times 0.7} =$$

$$P_{in} = \frac{F I_{sp} g}{2\eta}$$

$\boxed{49.299\ \text{Mw}}$

Ouch!

$$\frac{P_{in}}{M} = \frac{F}{M}\frac{I_{sp} g}{2\eta} = \frac{a I_{sp} g}{2\eta}$$

$$a = \frac{P_{in}}{M}\frac{2\eta}{I_{sp} g}$$

Total Energy Consumed:

$$E = \frac{690_{Nt} \times 10{,}200_{sec} \times 9.8066_{m/sec}}{2 \times 0.7} \times 338550_{sec} = 16.69 \times 10^6\, MJ$$

Power Density

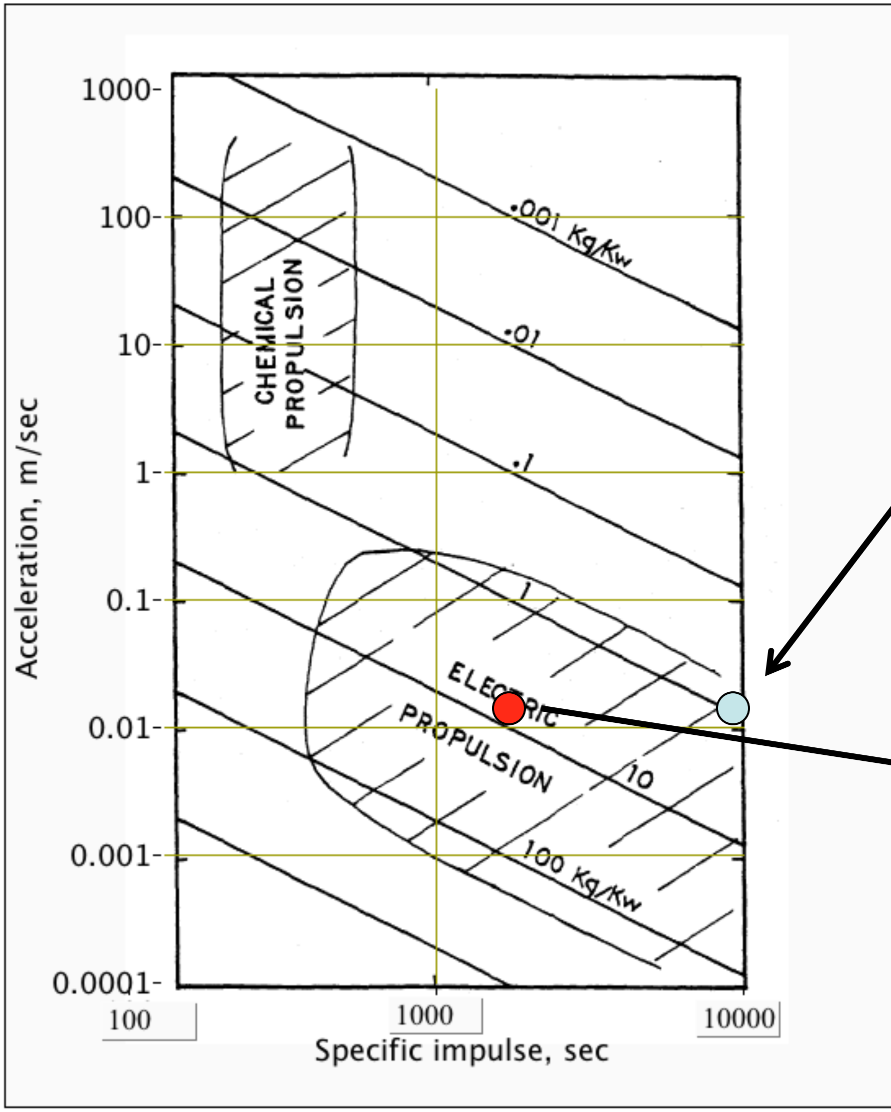
- Required Mean Acceleration

$$a = \frac{F}{M} \approx \frac{690_{Nt}}{\left(50{,}000 + \frac{2411}{2}\right)_{kg}} = 0.013475_{m/\sec^2}$$

- Calculate Specific mass

$$\frac{M}{P_{in}} = \frac{2\eta}{a I_{sp} g_0} = \frac{2 \times 0.7}{0.013475_{m/\sec^2} \times 10{,}200_{\sec} \times 9.8066_{m/\sec^2}} \times 1000 = 1.0387_{\frac{kg}{Kw}}$$

Power Density (cont'd)

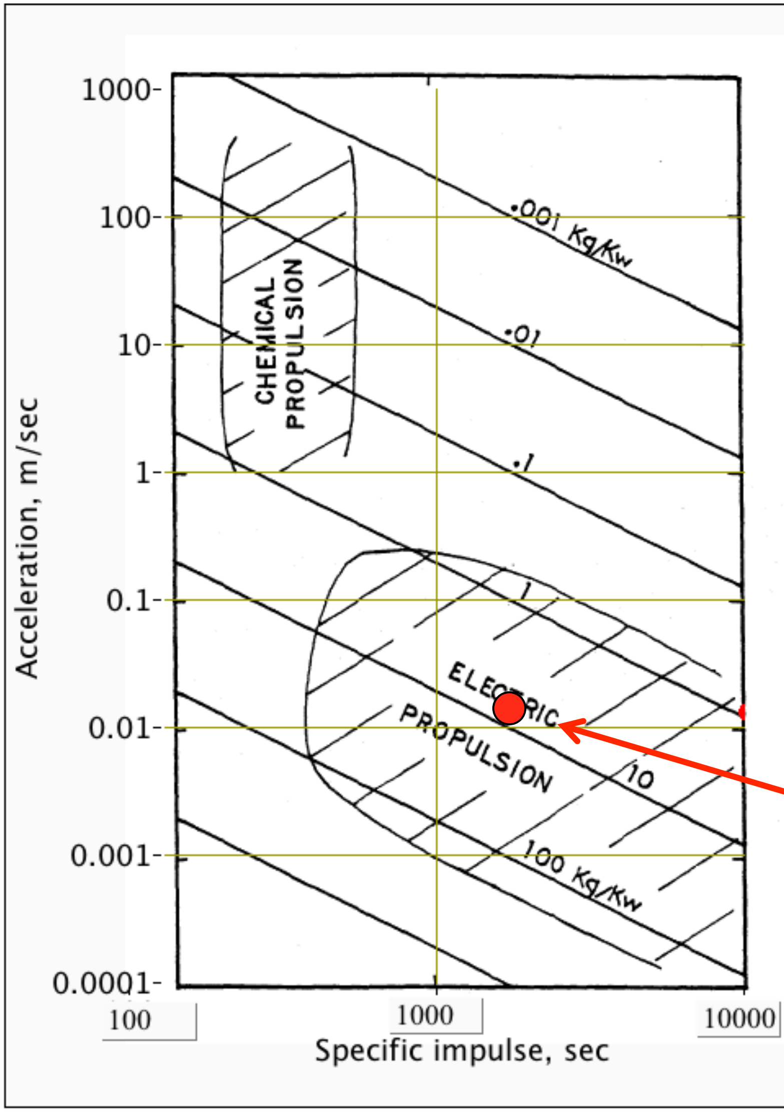


- Required Power density likely not feasible at this point for 100 hour mission

I_{sp} is Unrealistically high

Specific Mass of 7 *Kg/Kw* is more realistic

Power Density (cont'd)



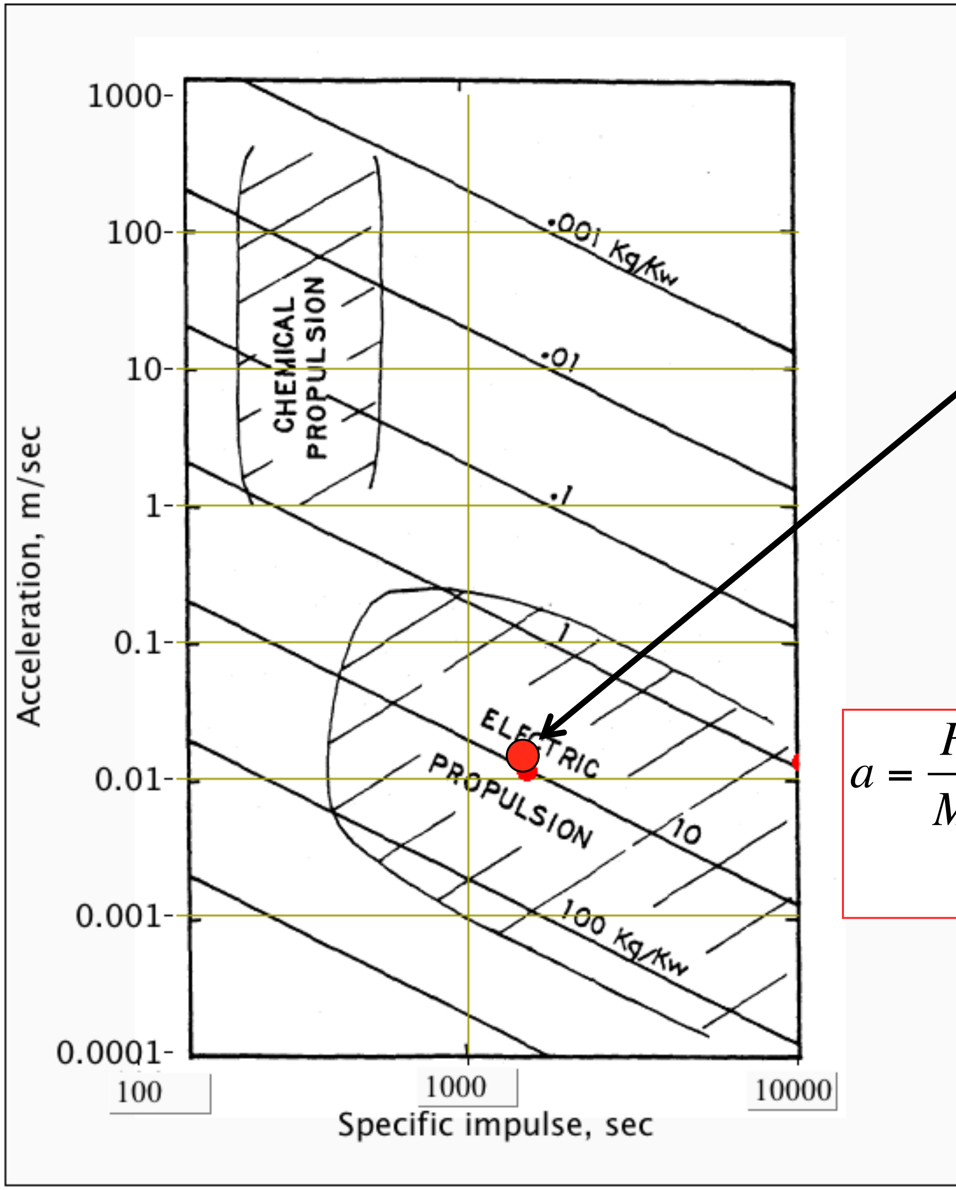
$$\frac{M}{P_{in}} = \frac{2\eta}{a I_{sp} g_0} = \frac{2\eta}{\frac{F}{M} I_{sp} g_0} \rightarrow I_{sp} = \frac{M}{F} \cdot \frac{2\eta}{g_0} \cdot \frac{P_{in}}{M}$$

$$I_{sp} = \frac{\left(50{,}000 + \frac{2411}{2}\right)_{kg}}{690_{Nt}} \cdot \frac{2 \cdot 0.7}{9.8066_{m/\sec^2}} \cdot \frac{1}{7\,\frac{kg}{KW}} \cdot \frac{1000W}{1KW} =$$

~ 1500 sec

Specific Mass of 7 *Kg/Kw* is more realistic

Power Density (cont'd)



- To get power density of 10kg/Kw with same thrust (690 Nt) … Isp ~ 1500 sec

$$M_{prop} = M_{dry}\left(e^{\frac{\Delta V}{g_0 I_{sp}}} - 1\right) =$$

$$50000\left(\exp\left(\frac{(4.5587+0.055)\,1000}{9.8066\cdot 1500}\right) - 1\right)$$

18420.3 kg

$$a = \frac{F}{M} \approx \frac{690_{Nt}}{\left(50{,}000 + \frac{18420.3}{2}\right)_{kg}} = 0.011653_{m/\sec^2}$$

- Required Power

$$\frac{690\cdot 1500\cdot 9.8066}{2\cdot 0.7\cdot 10^6} = 7.25\ \text{Mw}$$

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Power Density (cont'd)

$$u_e = I_{sp} \times g_0 = 1500_{sec} \times 9.8066_{m/\sec^2} = 14709.9_{m/\sec}$$

- Required Power

$$\frac{690 \cdot 1500 \cdot 9.8066}{2 \cdot 0.7 \cdot 10^6} = 7.25 \text{ Mw}$$

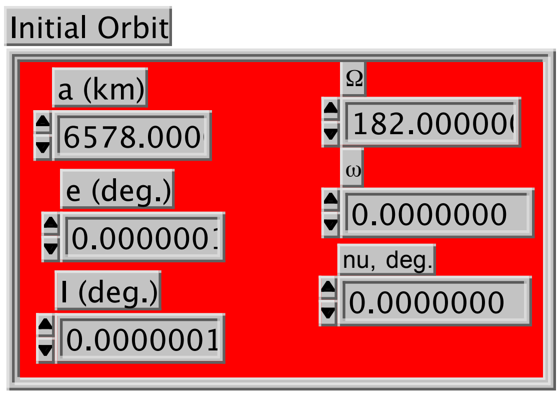
Factor of 7 less power consumed

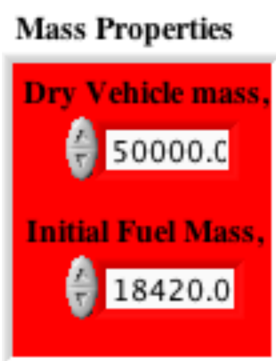
- Energy consumed

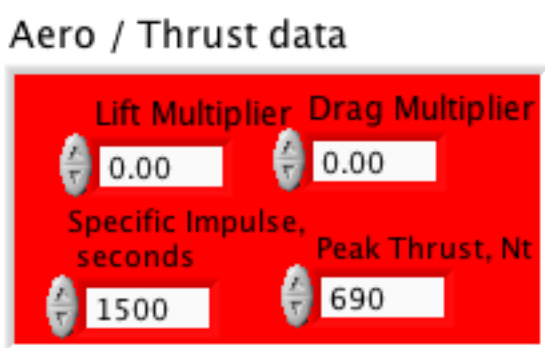
$$E = 7.25\, Mw \times 389{,}000_{\sec} = 2.8203 \times 10^6{}_{MJ}$$

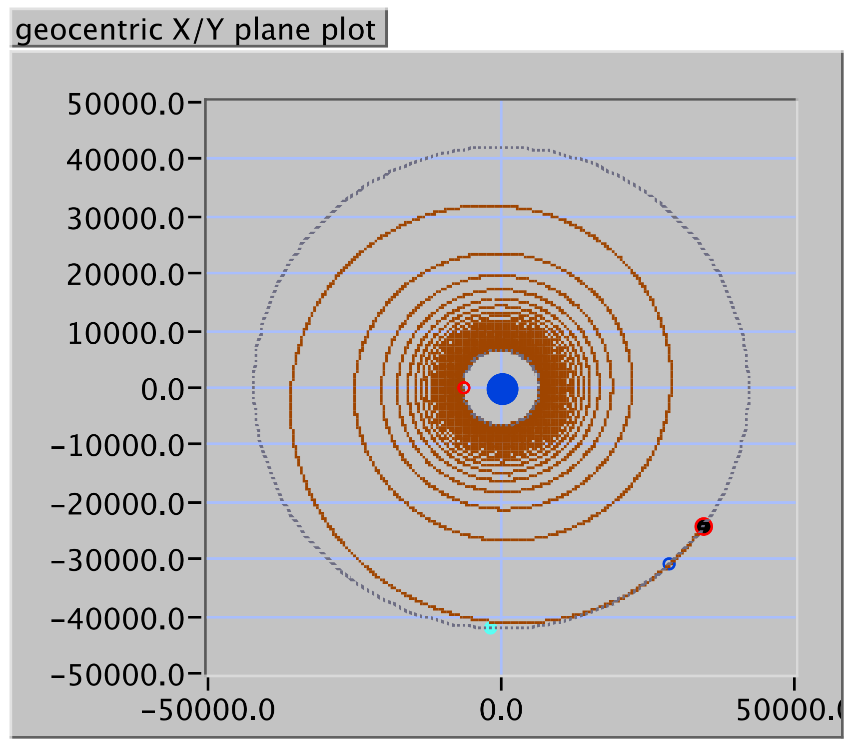
Mission profile 2: Spiral Transfer Plot

- Mission Duration: 115.19 hours @ 690 Nt Thrust
- Burn Duration: 389,000 sec
- Requires 664% more propellant, but 85% less input power

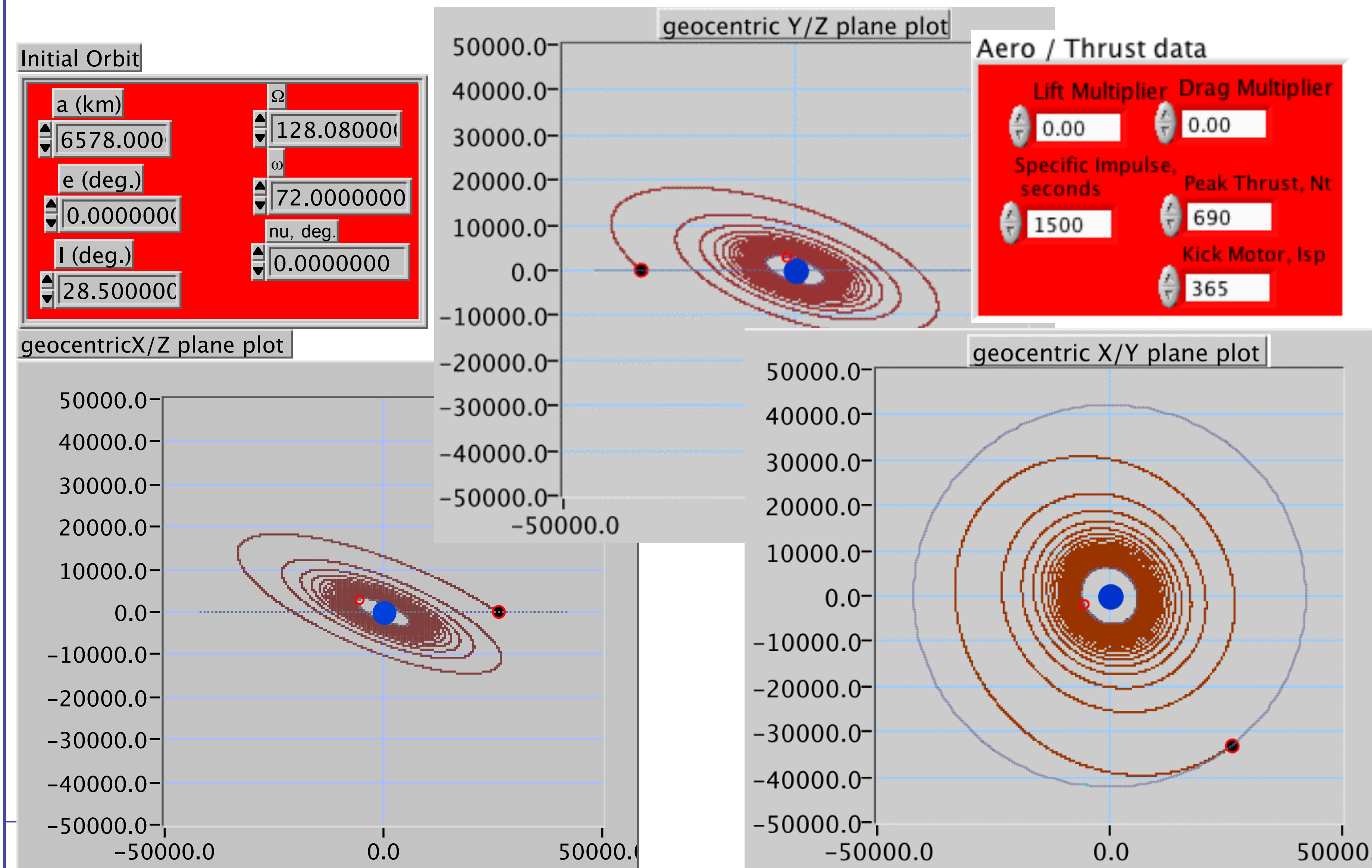




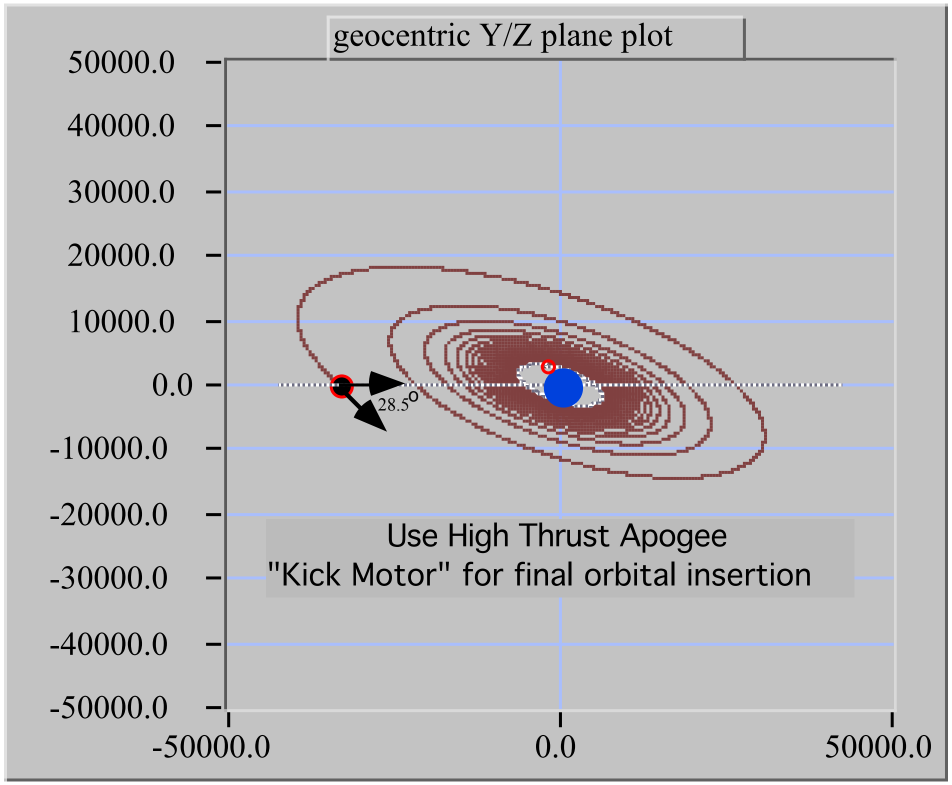




Mission Profile 3:
Transfer from 28.5° initial LEO orbit to GEO final orbit



Final Orbit Insertion



- Required Delta V for plane change is 1.484 km/sec
- Assume Kick Motor

$$I_{sp} = 350 \text{ sec}$$

- 8:1 propellant-to-dry mass ratio for kick motor
- Delivered payload mass fraction ~ 45.34% of initial mass

Mission Delta V / Mass Budget

Mass budget cluster

Dry Mass, kg: 33088.1	Propellant consumed during transfer: 18202.3	DV Budget for mission (Km/sec): 6.0440
Propellant Load, kg: 35201.9	DV budget for Transfer km/sec: 4.55999	Residual Propellant (kg): 0.137546
Total Original Mass, kg: 68290	DV Needed for (km/sec) final orbit insertion: 1.48402	Final mass at target orbit insertion, kg: 33088.2
Mass of vehicle (kg) at Target Rendezvous: 50087.7	Propellant Needed for final orbit insertion (kg): 16999.5	Delivered payload mass, kg: 30963.3

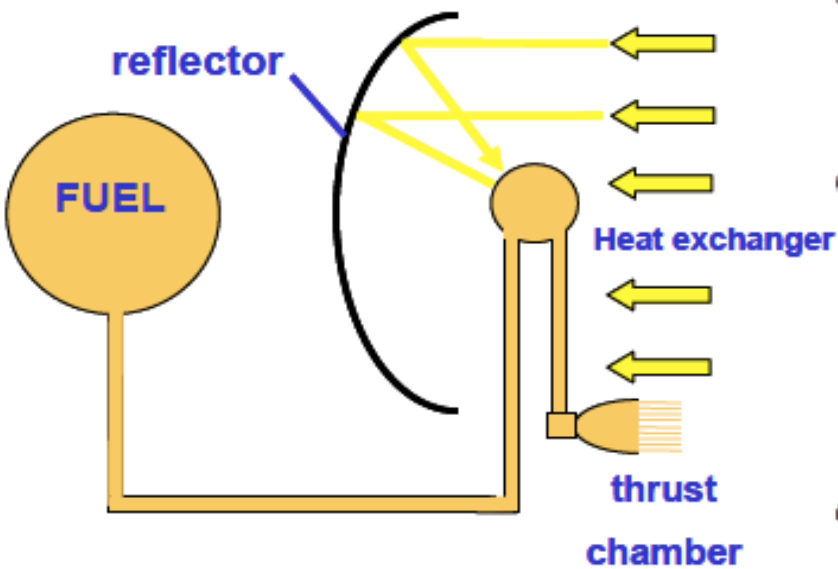
Other Concepts

- Futuristic .. To say the least

Solar Thermal Rocket

Solar- thermal rockets using solar energy to heat a propellant via a concentrator to high temperature

Schematic of Solar-Thermal Propulsion



Possible application for future orbit transfer vehicles

- Concept: Solar energy is concentrated on a heat exchanger by using mirrors or lenses.
- E.g. liquid hydrogen propellant is passed through a heat exchanger, reaching very high temperatures up to 2500 K, before expanding through a nozzle.
- By this, solar-thermal rockets make use of the limitless power of the sun to produce relatively high thrust, *F*, with high exhaust velocities, v_e:
  - F = 5 to 10 N continuous for 70 kW (solar power)
  - $v_e \approx 8000$ m/s
- Basic engineering problems limit thrust levels due to limit in heat transfer from heat exchanger to propellant.
- In addition, the deployment and steering of large mirrors to collect and focus the solar energy presents an operational challenge.
- Status: Several concepts for solar-thermal propulsion systems have been proposed, however, so far none have been tested.

Solar Thermal Rocket

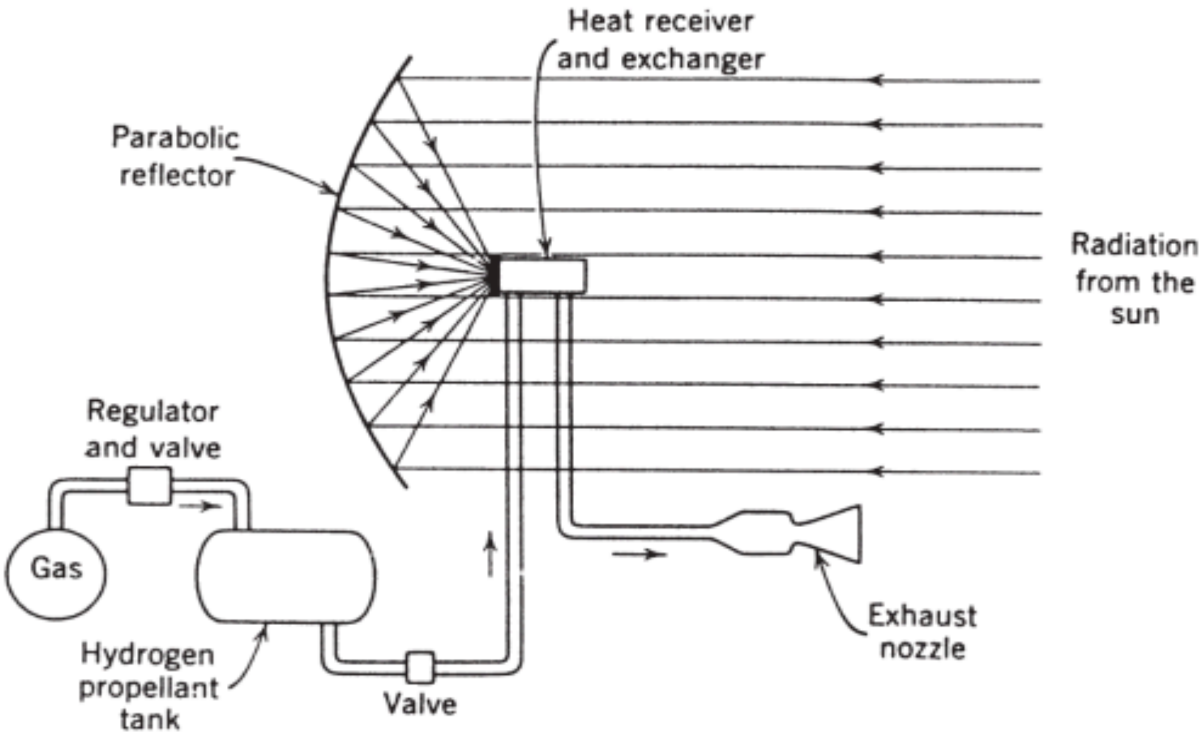


FIGURE 1–11. Simplified schematic diagram of a solar thermal rocket concept.

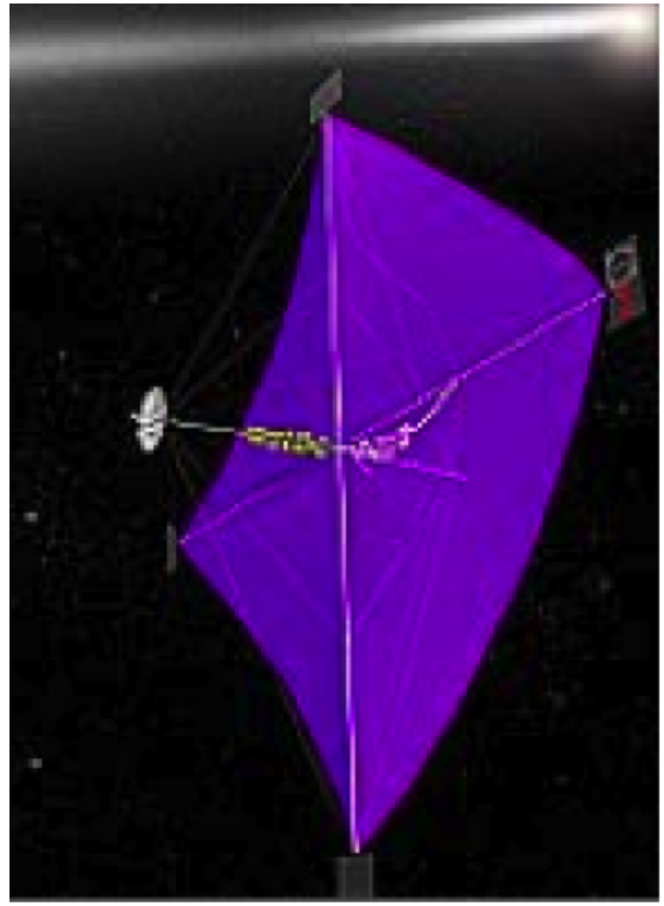
Performance can be two to three times higher than that of a chemical rocket and thrust levels in most studies are low (1 to 10 N). Since large lightweight optical elements cannot withstand drag forces without deformation, the optical systems are deployed outside the atmosphere. Contamination is negigible, but storage or refueling of liquid hydrogen is a challenge.

Fusion Engine

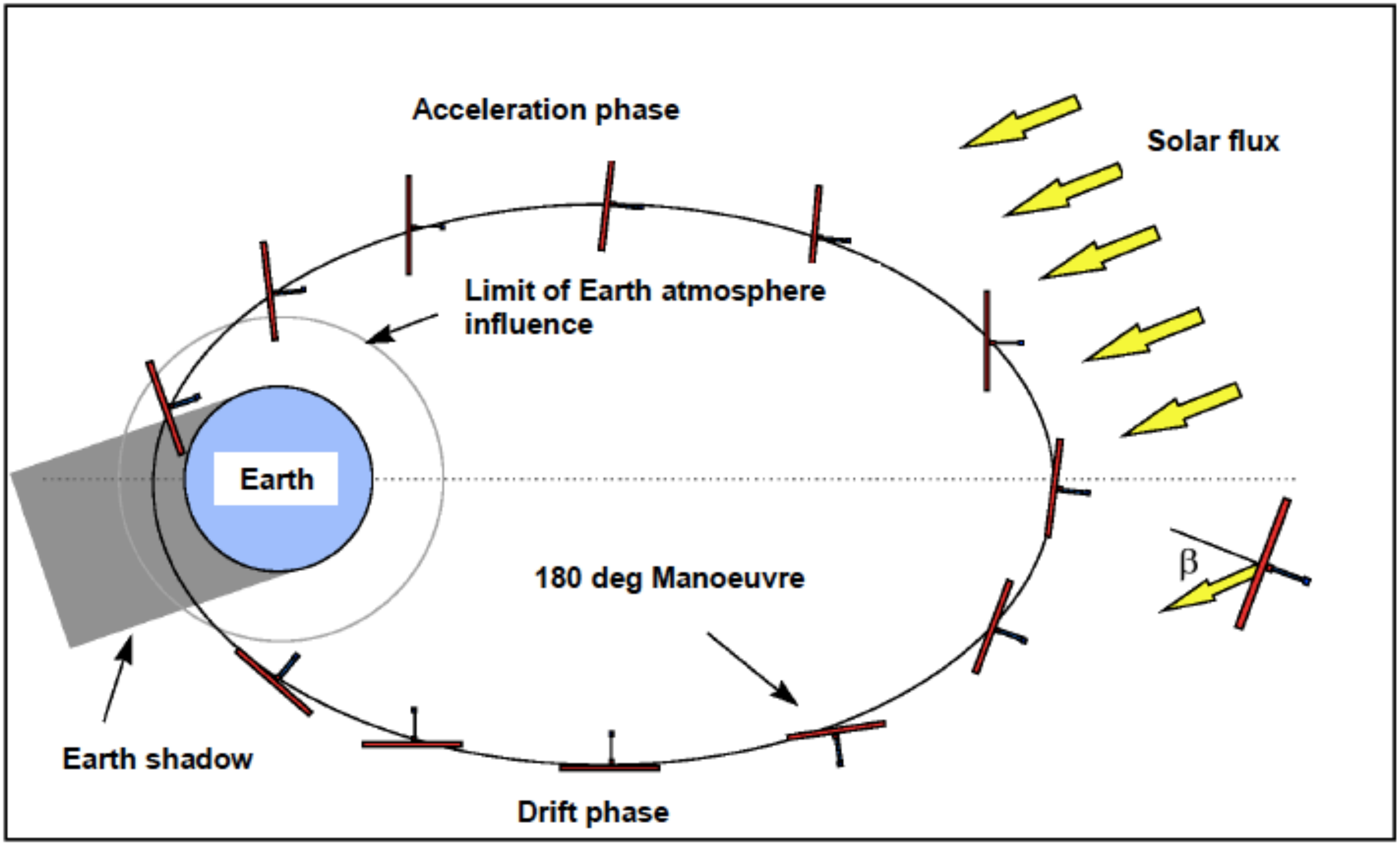
- I_{sp} as high as 4.1 million seconds!
- An unmanned trip to Barnard's Star (six light-years away)
- Powered by a fusion pulse engine
- 1.2 billion pound spacecraft, 1.1 million pounds of payload
- The "only practical problem" is that the engine requires helium-3, which must be obtained by mining the atmosphere of Jupiter.

Solar Sails

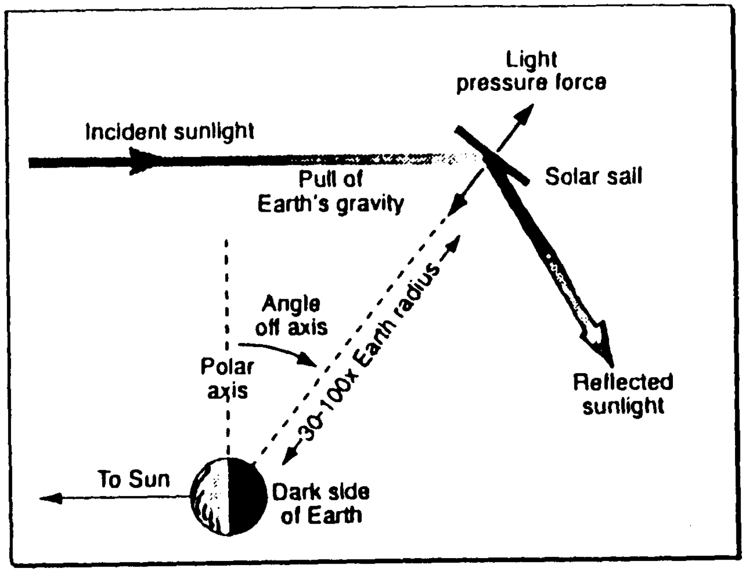
- A very large, very light, very reflective surface can be used to sail the solar wind.
- Currently used for attitude control on existing satellites
  - you either use it or compensate for it.
- Very low thrust, but infinite I_{sp}
- Not usable beyond Jupiter
- cannot be deployed below 600 miles (air drag)



Schematic of Solar Sail “Tacking” to Gain Thrust



Space Tethers: Pole Sat



Riding on a sunbeam: for a craft with a "solar sail" the pull of the Earth's gravity can be counterbalanced by the force of sunlight

THE POLESAT CONCEPT

- EARTH'S GRAVITY COUNTERBALANCED BY SOLAR RADIATION PRESURE
- DEVELOPED BY ROBERT FORWARD HUGHES RESEARCH LABS (PATENT PENDING)

THE POLESAT GEOMETRY

- TYPICAL "INCLINATION" ANGLES = 30° TO 40°
  - ANGLE MUST EXCEED 23.5°
  - 90° ANGLE IS POSSIBLE (POLESAT OVER THE POLE)
- TYPICAL ALTITUDES (AREA/WEIGHT DEPENDENT) = 30 TO 100 EARTH RADII
  - SIGNAL TRAVEL TIMES (TWO WAY) = 1.3 TO 4.2 SECONDS
  - OVER-THE-POLE POLESAT = 270 EARTH RADII (TWO-WAY SIGNAL TRAVEL TIME = 11.5 SECONDS)

SOURCES: <u>THE NEW SCIENTIST</u>, MARCH 9, 1991, PAGE 23;
THE AIAA'S JOURNAL OF SPACECRAFTS AND ROCKETS, MAY/JUNE 1991

Exotic Propulsion Methods: Antimatter and Photon Propulsion

"I have learned to use the word 'impossible' with the greatest caution."

Wernher von Braun

- **Exotic Propulsion Systems are those "far out" ideas still under study. They will be required for the ultimate dream of space exploration to travel to other star systems, as depicted in TV shows like 'Star trek'. Two examples of such exotic propulsion systems are outlined below.**
- **Antimatter Propulsion: Matter- antimatter annihilation offers the highest possible physical energy density of any known reaction substance. Since matter and antimatter annihilate each other completely, it is an incredibly compact way of storing energy. E.g. a round trip to Mars with a 100-ton payload might require only 30 gram of antimatter. However, sufficient production and storage of antimatter (with potential complex and high storage system mass) is still very much in the future.**
- **Photon Propulsion: The generation of usable thrust by ejection of photons is still very hypothetic. The generation of photons by e.g. laser technology and their subsequent decay in space, involves the mass-energy transfer expressed by Einstein's equation, $E = mc^2$. Consequently, very large quantities of energy will be required even for nominal levels of thrust. Possibly, matter-antimatter annihilation can be harnessed for photon propulsion in the future.**

Laser Power

- Laser on the ground, vaporizing material from the lower edge of the rocket to produce thrust.
- May provide a payload mass fraction of 0.15
- Small Payloads, but continuous operation could lift 64,000 lbs per year
- theoretically $45 per pound

Laser Power (cont'd)

Chemical Mass Drivers

- Jules Verne, Earth to the Moon
- 900 ft tunnel, filled with 200 ft of gun powder
- A slight problem with g forces, but the chickens loved it - at last they could fly.

HARP

(High-Altitude Research Project)

- Gerald Bull tried to demonstrate achieving orbit with a rocket fired from a 16 inch Naval Gun
- 100 pound payload to orbit for $50,000 ($500/lb)
- 2,000 G's

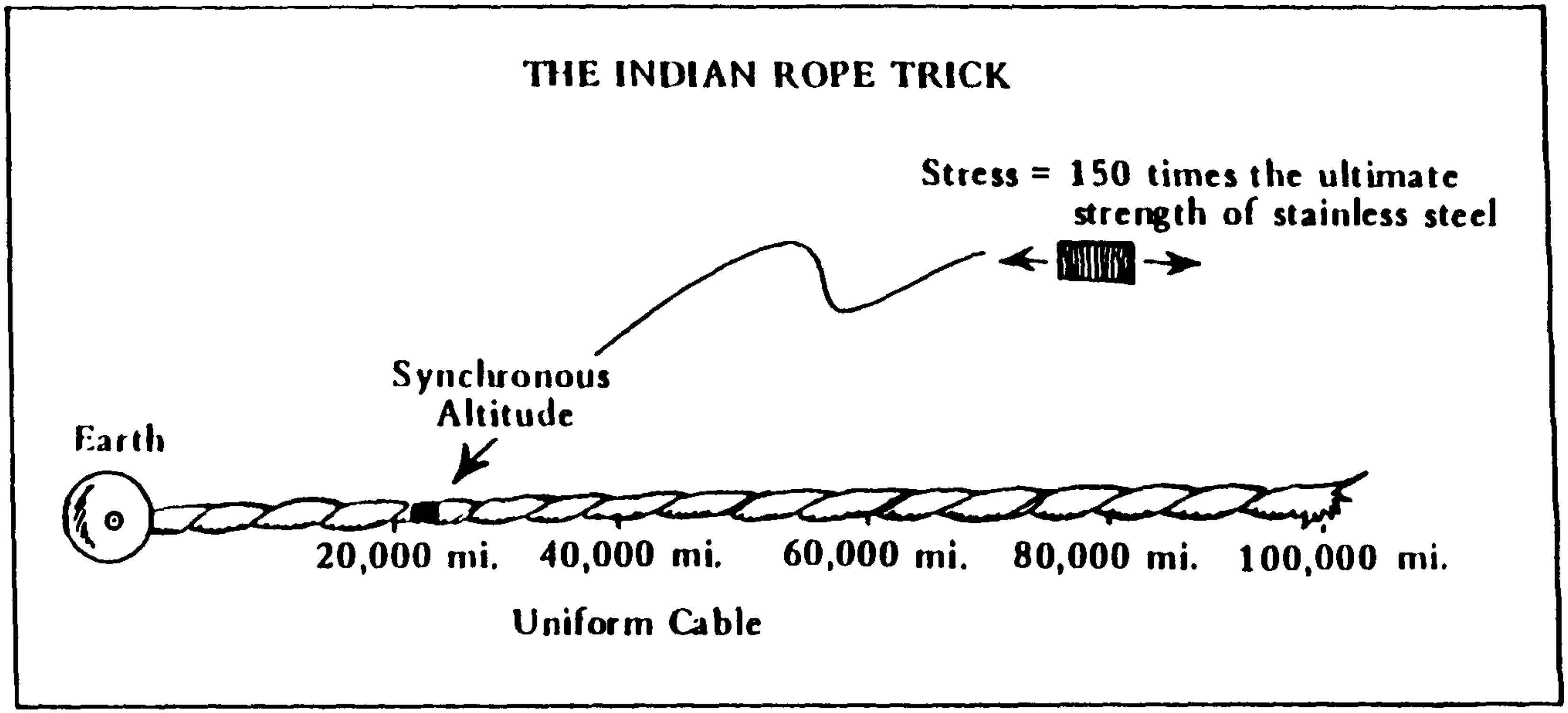
ElectroMagnetic Catapults

- Rail guns or Mass drivers
- linear electric motors
- Moon based system to make it cheaper to mine the moon
  - 20,000 tons of material to build it
  - 20,000,000 tons off of the moon at $0.50/pound
- 1000 G?
- Similar concepts for Earth based systems

Anti-matter Propellants

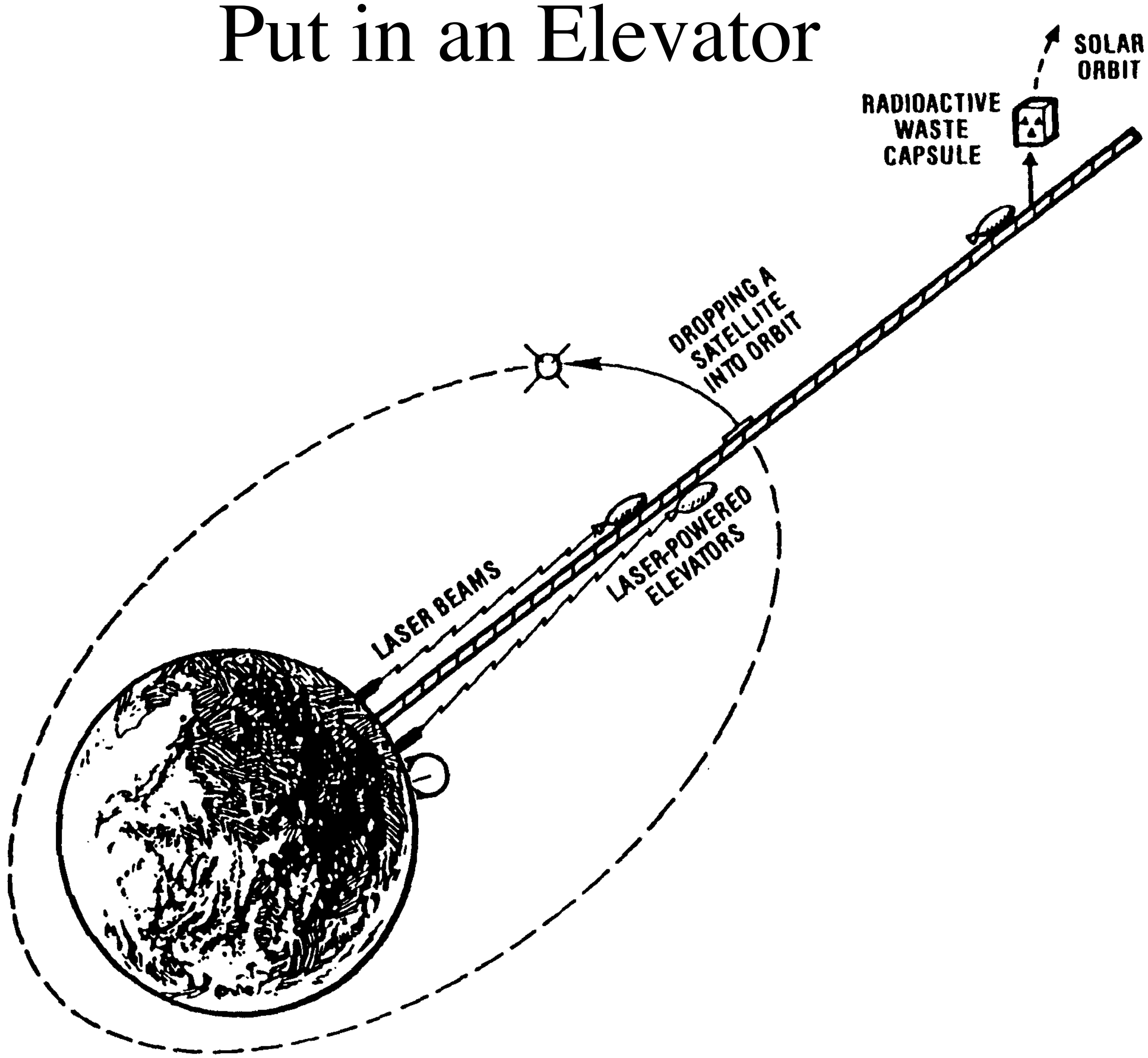
- Production, cooling, and storage of the anti-matter may be a problem.
- A 10-ton spacecraft could go to Barnard's star with 25 KG of anti-matter
- It has actually been produced in linear accelerators (pico-grams) - just some engineering details to work out.

Indian Rope Trick



If it could be gotten into position, a 100,000-mile flexible cable would hang suspended above the equator. It could be used as a skyhook to hoist satellites into earth orbit or to fling instrumented probes to the distant corners of space.

Put in an Elevator



Generate energy, when you aren't launching