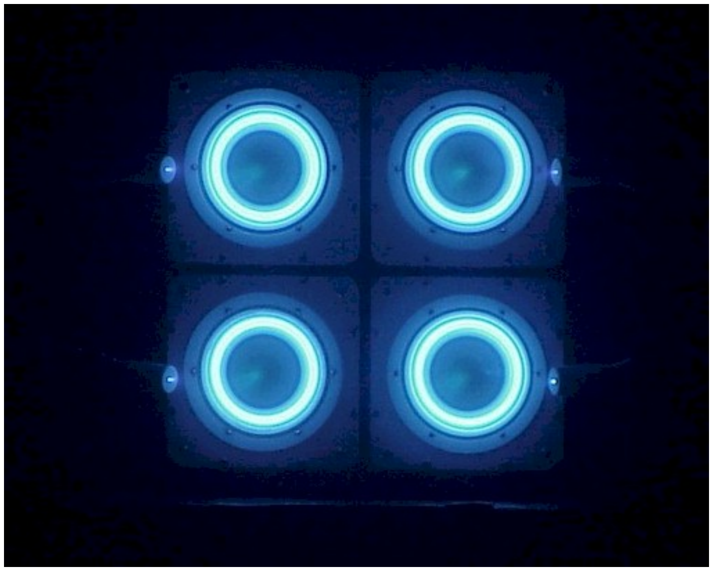
The Application of Next Generation Electric Propulsion Systems to Mars Exploration



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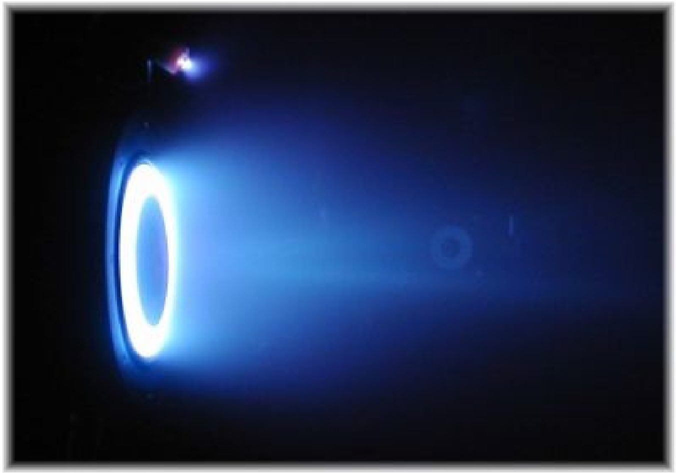
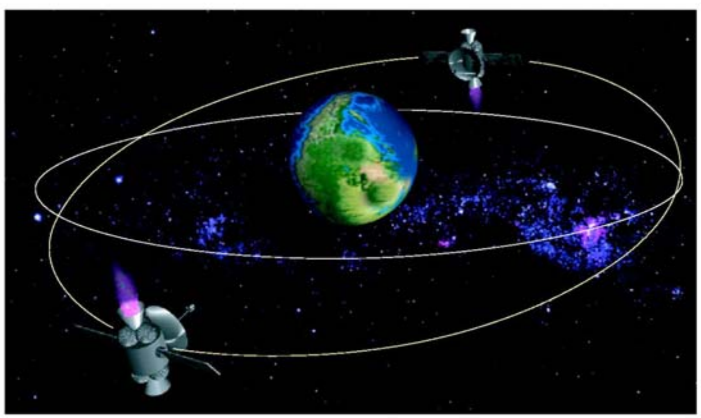
Overview of EP Systems

- *Electrothermal*
  - propellant heated by electrical process & expanded through a nozzle
  - resistojets (hydrazine, $I_{sp} \approx 300$ sec, 300 mN, $\varepsilon \approx 80\%$, 750 W)
  - arcjets (hydrazine, $I_{sp} \approx 600$ sec, 250 mN, $\varepsilon \approx 40\%$, 1.5k W)
- *Electrostatic*
  - propellant accelerated by electrostatic forces to ionise particles
  - ion thruster (xenon, $I_{sp} \approx 4000$ sec, 100 mN, $\varepsilon \approx 65\%$, 0.75 - 27 kW)
  - FEEP (cesium, $I_{sp} \approx 10000$ sec, <5 mN, $\varepsilon \approx 95\%$, ≈ 500W)
- *Electromagnetic*
  - propellant accelerated by combined action of electric & magnetic fields
  - MPD/Li-LFA (lithium, $I_{sp} \approx 4000$ sec, 12.5 N, $\varepsilon \approx 48\%$, 200 kW)
  - Hall thruster (xenon, $I_{sp} \approx 1800$ sec, 80 mN, $\varepsilon \approx 50\%$, 200 kW)

Station Keeping & Orbit Transfers

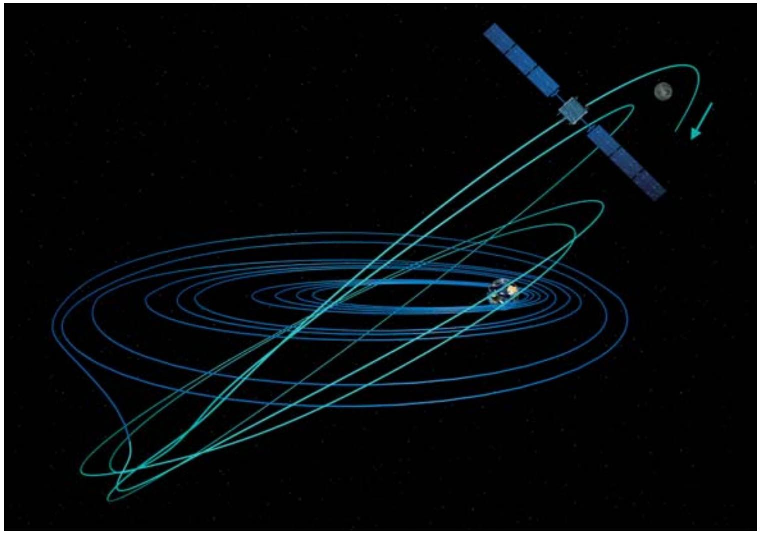


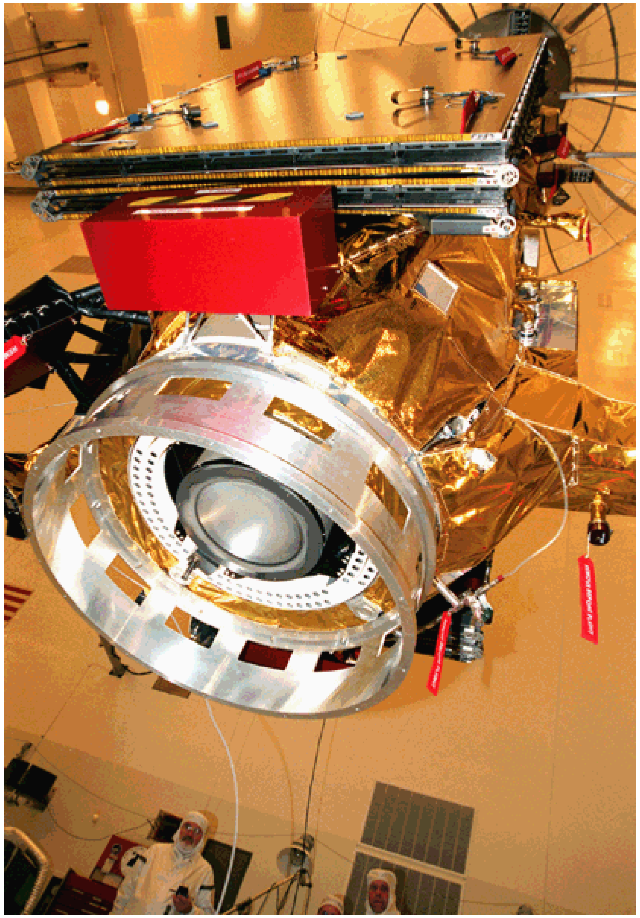


- Used on Russian satellites since 1970s
- > 150 EP systems have flown
- Applications
  - Attitude control
  - Station keeping
  - Drag reduction
  - Orbit changing functions
    - Gentle spiral trajectories
- Advantages
  - High precision thrust
  - Low propellent consumption
  - Long life

Interplanetary Travel

- Deep Space 1 (Braille asteroid -1998)
  - ion thruster, 12kg Xe, 1800 hr thrust
- Hayabusa (Itokawa - 2003)
  - 4 ion thrusters, 22kg Xe, >26,000 hrs
- SMART-1 (Moon - 2003)
  - Hall thruster, 12kg Xe, 0.07 N, 1.5 year trip





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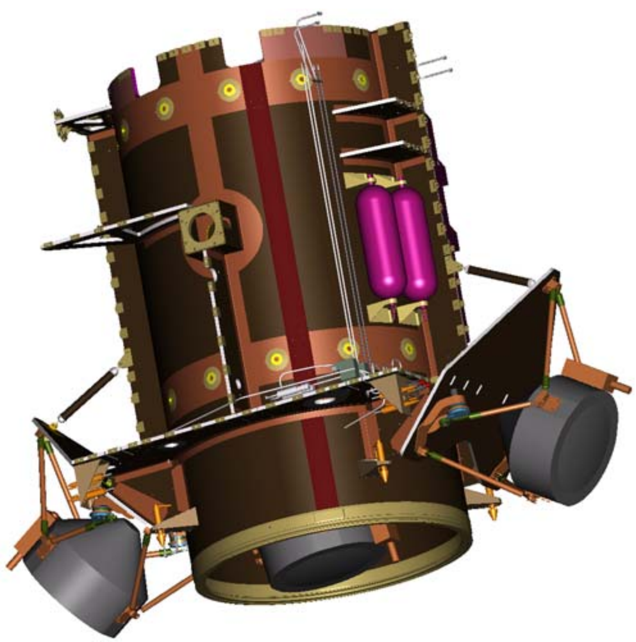
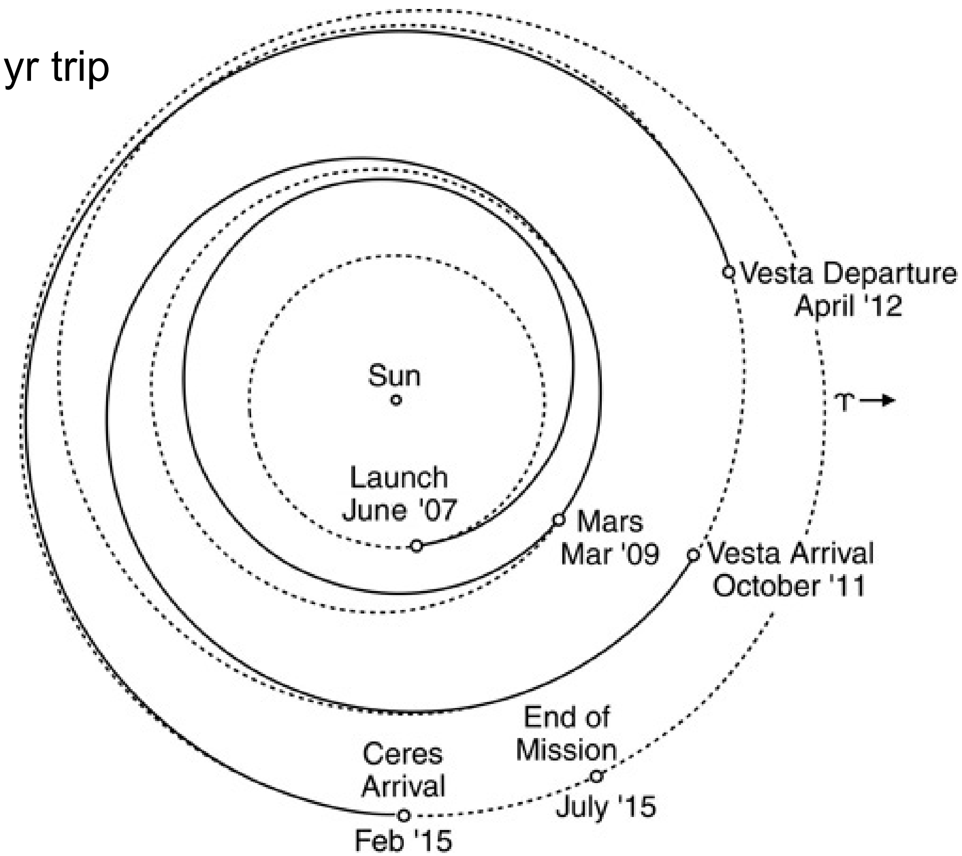
Interplanetary Travel

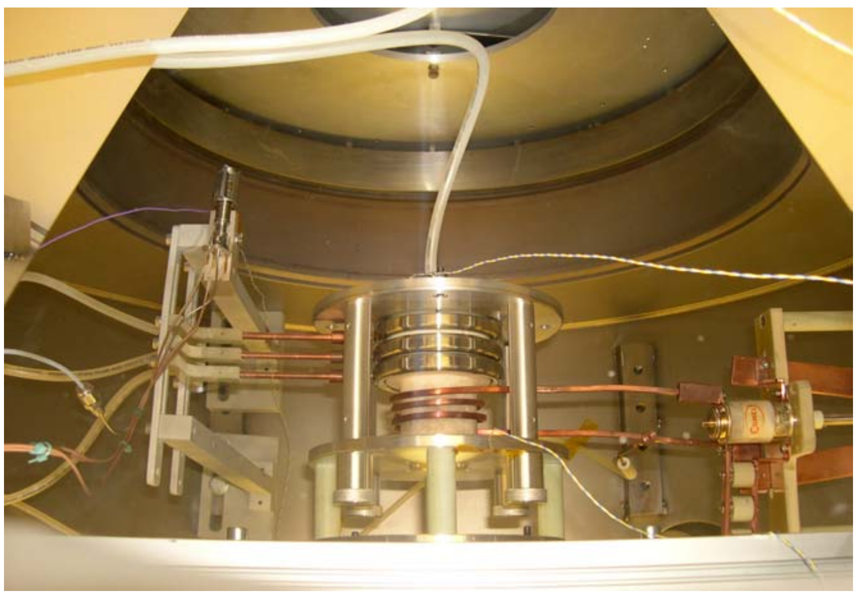
- DAWN (Ceres & Vesta - 2007)
  - 4 ion thrusters, 250kg Xe, 90mN, 6 yr trip
- BepiColombo (Mercury - 2013)
  - ion thruster, 0.24 N, 6 yr trip

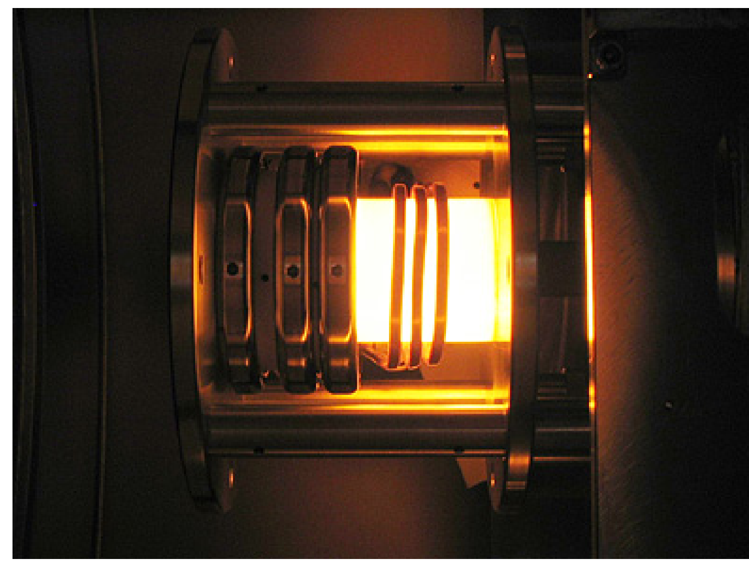




Dual-Stage 4 Grid (DS4G) Ion Thruster

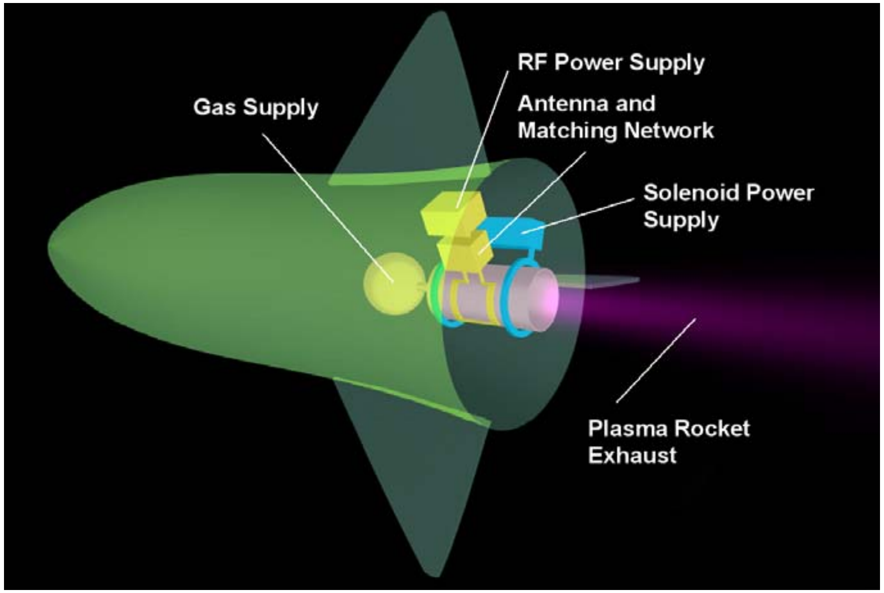
- ESA test campaigns 2005 & 2006
  - 43 aperture grid
  - Max 30kV beam potential
  - Thrust = 2.7 mN
  - Isp = 14000 sec
  - Total Power = 300 W
  - Mass ultilisation efficiency = 96%
  - Total efficiency = 63%
  - Thrust density = 0.86 nN/cm^2
  - Beam divergence = 4-6º





Helicon Double Layer Thruster (HDLT)

- Magnetoplasmadynamic system
- Simple design – no moving parts
- Low thrust but scalable (~1-10mN)
- Propellants – H_2, O_2, Xe, Ar
- No electrodes or neutraliser needed
  - Long operating life
- High exhaust velocity > 10 km/sec
- $V_{exhaust}$ ~ 15 km/sec with O_2
- Beam divergence ~ 2°
- Scalable in size and power



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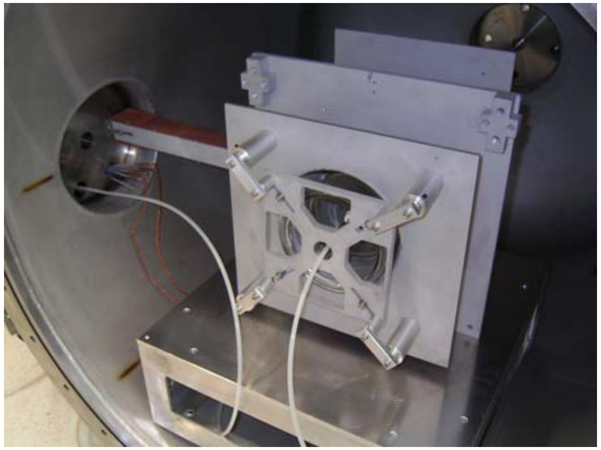

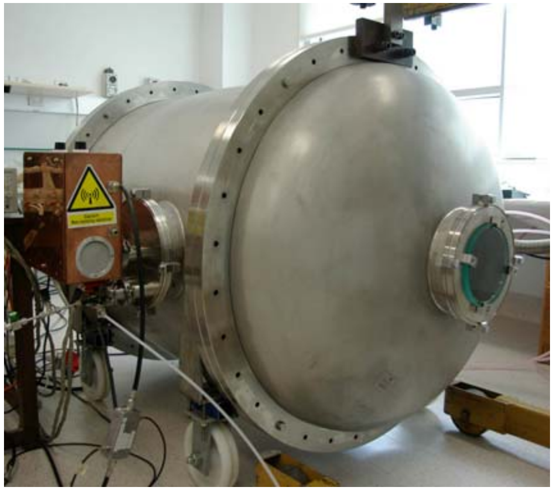

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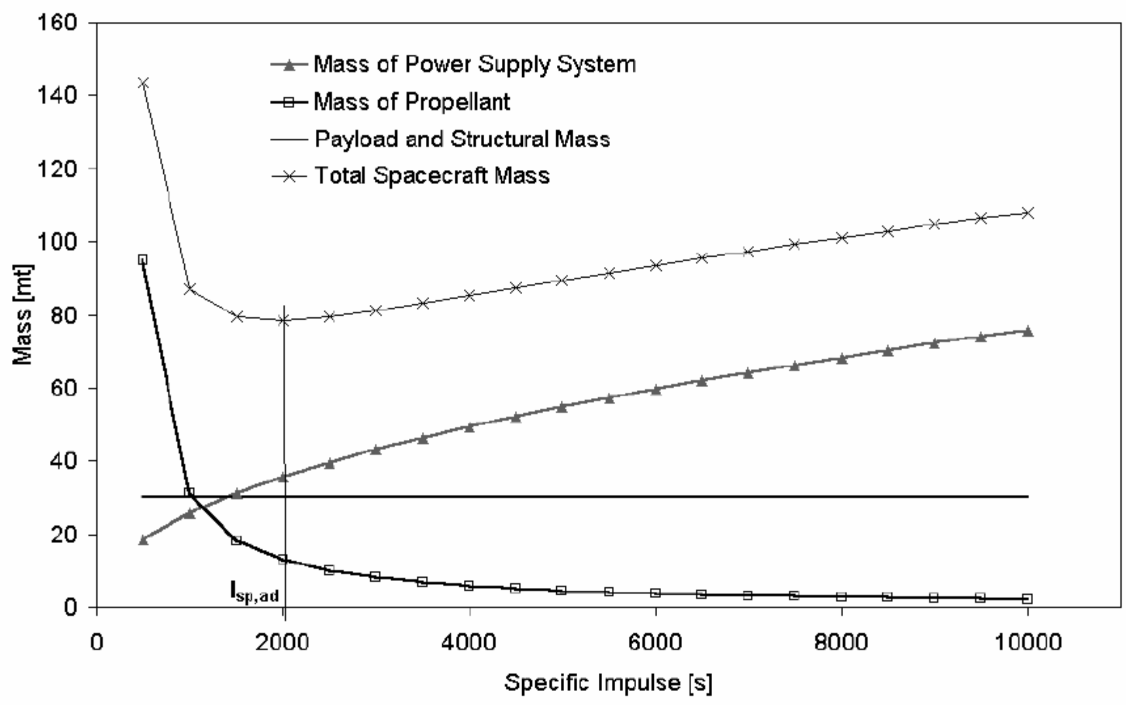
Propulsion System Parameters

$$P_e = \frac{F_{\max} \cdot I_{sp}}{2 \cdot \eta_{th}}$$

where F_{max} is maximum thrust level, I_{sp} is specific impulse, η_{th} is thruster efficiency and P_e is electrical input power



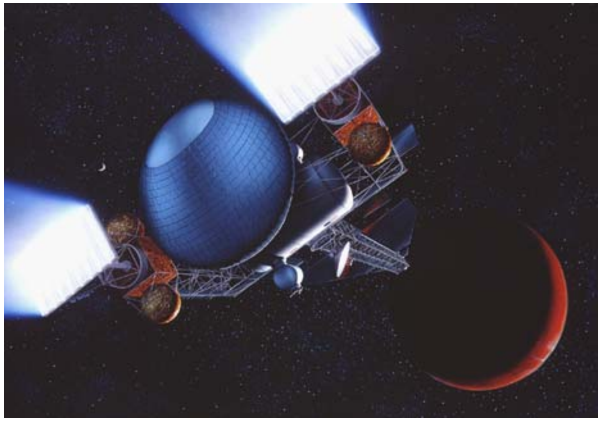
For Mars missions optimum I_{sp} ~ 3000 sec and thrust ~ 100N

From *Schmidt, T. D. & Auweter-Kurtz, M., 2005, 'Adequate electric propulsion system parameters for piloted Mars missions', IEPC-2005 Proceedings.*

Mars Mission Requirements

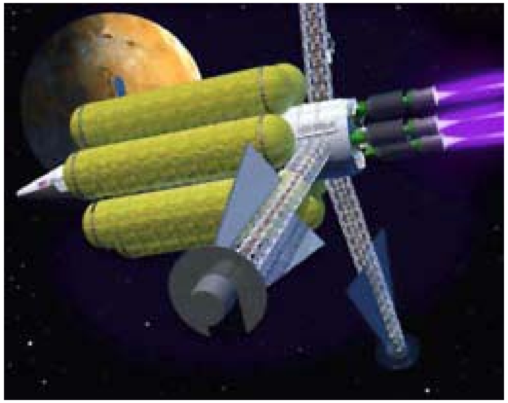
Crewed Missions

- Fast transit times
- Numerous abort options
- Power for life support
- High system redundancy
- Reasonable acceleration
  - avoid van Allen belt radiation



Cargo Missions

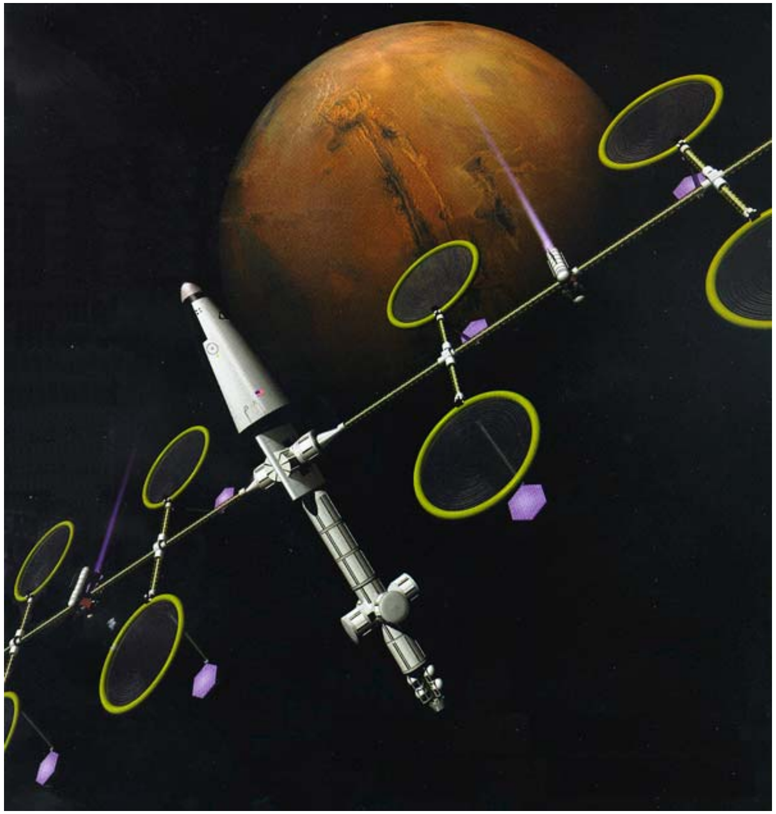
- High payload capacity
  - reduce propellent
- Transit time non critical
- Nominal power requirements
- Autonomous & limited complexity
- Reusable system



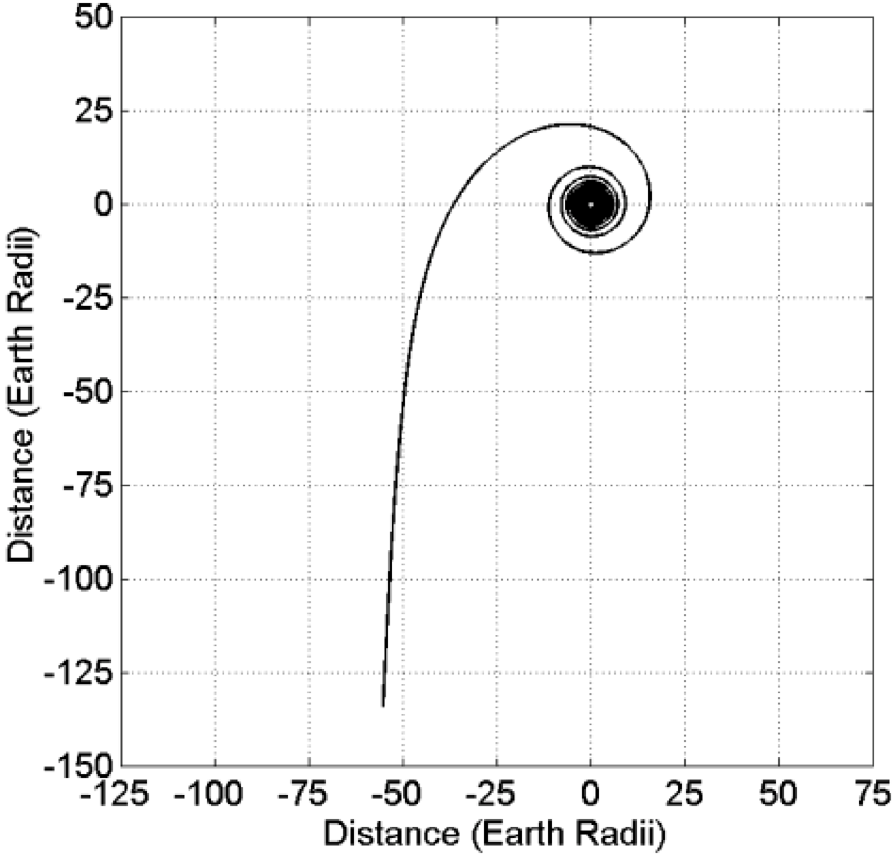
Mission Scenarios

Traditional Direct NEP or SEP

- Large vehicle constructed on orbit
  - separate nuclear reactor from crew
  - sufficient solar panel array
- Long spiral from LEO
- Continuous operation
  - long acceleration & deceleration
- Land & leave vehicle in Mars orbit
- Spiral (yet shorter) from Mars orbit



Mission Scenarios



Hybrid SEP/Chemical

- Cargo vehicle departs first
- 30-85 day spiral from LEO
- Crew follow in smaller vehicle
- Dock & proceed with either EP or chemical system
- Chemical system assists with Mars Orbit Insertion (MOI)

OR

- Cargo vehicle uses EP
- Crew vehicle use chemical

Mars Mission Propulsion Options

Propulsion Option	Description	Advantages	Disadvantages
Chemical	Conventional cryogenic rocket engines, usually one stage per major maneuver (TMI, MOI & TEI). Insulated tanks with vapor-cooled shields to reduce boil off. Start T/W 0.1 to 0.25. I_{sp} ~ 460s.	• Mature technology • High thrust, short burn times • Ballistic interplanetary transfers facilitate implementing artificial gravity	• Low performance leads to high IMLEO except for conjunction profile with long transfer times • Cryogenic with H_2, low density, needs leak control • Expendable system
Chemical & Aerocapture	As above except aerocapture used for MOI. Large aeroshell needed requiring either intact launch or in-space assembly. Lander may capture separately to simplify configuration	• Reduces IMLEO by replacing one major maneuver with aerocapture	• Performance still marginal for 'hard year' opportunities • Aerocapture risk: TPS/thermal, GN&C • Expendable system
NTR	Nuclear thermal rocket engine, H2 propellant, I_{sp} ~ 900s. Insulated tanks as above; start T/W <= 0.1 to reduce nuclear engine size.	• Known technology • Twice I_{sp} of chemical reduces IMLEO & sensitivity to opportunity • High thrust, short burn times • Ballistic interplanetary transfers facilitate implementing artificial gravity	• Nuclear costs & risks • Engine test protocols not resolved (containing radioactive products) • Cryogenic with H_2, low density, needs leak control (worse for H_2) • Expendable system

From *Griffin, B, et al, 2004, 'A Comparison of Transportation Systems for Human Missions to Mars', Proceedings of Joint Propulsion Conference 2004, AIAA 2004-3834.*

Mars Mission Propulsion Options

Propulsion Option	Description	Advantages	Disadvantages
SEP	Large (multi-MW) solar electric propulsion system, performs all major maneuvers. I_{sp} typically 3000s; MPD or comparable thrusters.	• Known technology with increasing flight experience • High I_{sp} reduces IMLEO & sensitivity • No hydrogen propellant • **Reusable system**	• Large size may require more space assembly • High power EP systems not mature (TRL 2-3) • Achievable power-to-mass ratios may exclude some opposition class profiles
NEP	Large (multi-MW) nuclear electric propulsion system, probably Brayton or liquid metal Rankine power generation, performs all major maneuvers. I_{sp} typically 3000s; MPD or comparable thrusters.	• Known technology (no space experience & few experimental prototypes) • High I_{sp} reduces IMLEO & sensitivity • No hydrogen propellant • Potentially reusable system	• Nuclear costs & risks • Large size may require more space assembly • High power EP systems not mature (TRL 2-3) • Achievable power-to-mass ratios may exclude some opposition class profiles
SEP & Chemical	Large SEP ‘tug’ system ~ 1 MW delivers chemical interplanetary vehicle to highly elliptical Earth orbit (perhaps in parts for assembly). Chemical propulsion system departs from this orbit & proceeds as chemical option.	• **Placement in elliptic orbit reduces chemical Δv ~ 3 km/s, reducing IMLEO & sensitivity to opportunity** • Same as chemical • SEP ‘tug’ is reusable	• Costs & mission complexity added by use of SEP ‘tug’ • Cryogenic with H_2, low density, needs leak control • Chemical component is expendable system

From *Griffin, B, et al, 2004, ‘A Comparison of Transportation Systems for Human Missions to Mars’, Proceedings of Joint Propulsion Conference 2004, AIAA 2004-3834.*

Conclusions

- Electric propulsion is a maturing technology
- Australia is contributing to next generation EP system development
- The benefits of EP for Mars exploration warrant further research particularly in light of performance of next generation systems
  - desirable I_{sp} & thrust range
  - reduce propellant requirement & therefore reduce IMLEO
- Straight SEP or NEP systems are plausible (yet may not be practical)
- Hybrid approach may be the most appropriate - more studies needed
- Irrespective, lets get to Mars anyway!! :o)

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