In-Space Transportation Propulsion Architecture Assessment Final Report August 15, 2000

Prepared for NASA MSFC

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1.0 Introduction and Background

Almost all space propulsion development and application has been chemical.

Aerobraking has been used at Venus and Mars, and for entry at Jupiter.

One electric propulsion mission has been flown (DS-1) and electric propulsion is in general use by commercial communications satellites for stationkeeping.

Gravity assist has been widely used for high-energy missions (Voyager, Galileo, Cassini, etc.). It has served as a substitute for high-energy propulsion but is limited in energy gain, and adds mission complexity as well as launch opportunity restrictions. It has very limited value for round trip missions such as humans to Mars and return.

High-energy space propulsion has been researched for many years, and some major developments, such as nuclear thermal propulsion (NTP), undertaken. With the exception of solar electric propulsion at a scale of a few kilowatts, high-energy space propulsion has never been used on a mission.

Most mission studies have adopted TRL 6 technology because most have looked for a near-term start. The current activity is technology planning aimed at broadening the options available to mission planners.

Many of the illustrations used in this report came from various NASA sources; their use is gratefully acknowledged.

2.0 Study Objectives

<u>Disclaimer:</u> The study reported herein was consultative. This report serves as an input to NASA deliberations which are intended to develop technology recommendations and plans for the agency. Results, conclusions and recommendations of this report do not represent official results, recommendations or plans on the part of NASA.

Objectives of the study were:

1. To review the state-of-the-art of in-space transportation propulsion;
2. To develop internally consistent assessment and planning data re advanced technology;
3. To determine the benefits, advantages and drawbacks of comparative systems;
4. To develop roadmaps for technology advancement (to TRL 6) for each system evaluated;
5. To draw overall conclusions and make recommendations for technology advancement initiatives.

3.0 Overview of Needs

This study investigated possible future missions, not approved or firm-planning missions. The missions considered are, or could be, in the category of "Long-Term Plans" in NASA's strategic plan. Firm requirements do not exist. Needs may be identified from basic technical characteristics of such missions.

The missions considered need high energy, large payloads, long duration, or combinations thereof and are consequently beyond the capabilities of current or near-term in-space propulsion technology. It is often the case, in devising these missions, that high energy is needed to reduce mission duration, which would otherwise be so long as to be impractical.

Figure 3-1 is a delta V-Time chart showing needs for delta Vs up to and even exceeding 100 km/s which indicates Isps as high as 10,000 to 20,000 seconds. For "close-in" missions such as to the Moon and the Earth-related libration points, delta Vs are modest and times reasonable. For distant destinations there is a clear tradeoff wherein more delta V buys shorter duration. For low-thrust systems one must be careful, in evaluating this tradeoff, whether or not a particular low-thrust system is capable of delivering the required delta V in the duration of interest. For example, an electric propulsion system with mass-to-power ratio (α) 10 kg/kWe delivers, with typical efficiency, about 0.06 kW jet power per kg of propulsion system and about 0.03 kW jet power (or less) for a complete vehicle. At an Isp of 5000 seconds, this amount of power is capable of about 40 km/sec in a year. Thus, even with continuous thrusting (no stopover time) it is not capable of a typical 1-year Mars round trip.

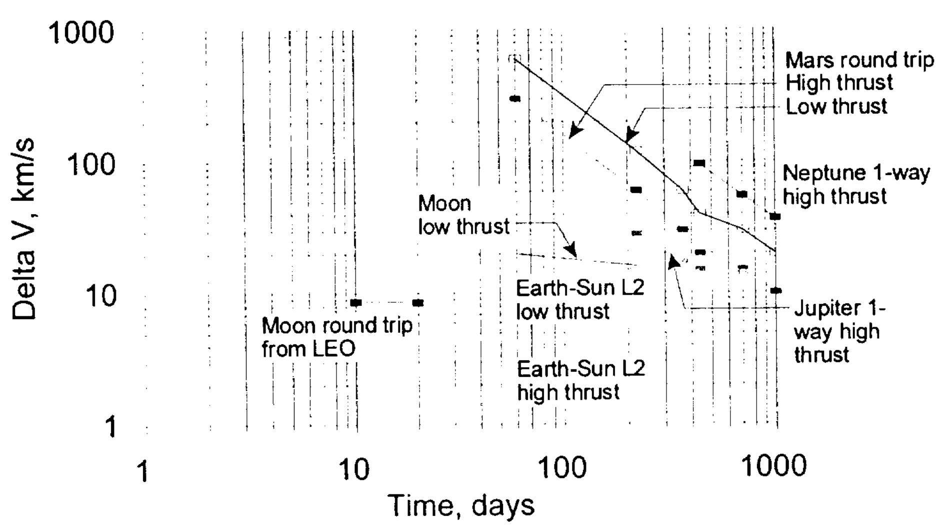


Figure 3-1: Time and Delta V versus Destination

We should avoid the inclination to insist that one propulsion technology satisfy all in-space transportation needs just as we cannot

expect one terrestrial transportation technology to satisfy all terrestrial transportation needs. We should seek solutions less diverse than terrestrial transportation because the total market for in-space transportation is not great enough to support an infrastructure of many diverse technologies.

4.0 Systems and Concepts Reviewed

The intent of this review was to include all technologies applicable to in-space transportation architectures: (1) not in current use, i.e. requiring technology advancement; (2) for which a sufficient scientific understanding exists to estimate performance and mass characteristics. For example, concepts involving alteration of space-time are excluded. Also, we excluded certain highly specialized concepts such as electromagnetic catapults on the lunar surface because they tend to be application-specific, not suitable for general in-space transportation architectures.

All space propulsion technologies can be represented in the 3-dimensional binary matrix of figure 4-1. This is useful because locations on the matrix have inherent characteristics; for example energy density limitations of chemical energy sources, or the inevitable mass of conversion machinery for converting heat energy to electric energy. Concepts reviewed in this report are named in the Figure.

	Chemical		Nuclear	
	Direct	Indirect	Direct	Indirect
Onboard	Conventional Rocket	No Logical Candidates	NTP GCNTP EPP Fusion	NEP
Off-board	Laser heated propulsion, power beamer on Earth	Laser electric propulsion, power beamer on Earth	Solar Thermal Propulsion Solar & magnetic sails	SEP

Figure 4-1: Categories of In-Space Propulsion

This allows us to make certain generalizations about in-space propulsion:

(1) Only nuclear energy and off-board energy are capable of delivering enough energy per unit mass to a space vehicle to achieve challenging missions.

(2) Off-board energy has limitations: if directed, range and pointing limits; if broadcast (i.e. solar), range (from the Sun) limits. These limits exclude certain challenging missions, such as missions to the outer planets which perform maneuvers such as orbit capture at the destination.

(3) Conversion systems, which convert energy from the source form, e.g. nuclear heat, to a propulsive form, i.e. kinetic energy of a jet, as in electric propulsion systems, are limited in power per unit mass such that high delta V, short duration missions, are excluded.

(4) Therefore, there is a tendency to prefer onboard nuclear energy sources with direct conversion from nuclear energy to jet kinetic energy.

The position of fusion propulsion in the figure recognizes that while conversion from reaction energy to jet energy is direct, there is likely to be significant circulating power in the propulsion system to "keep the fusion fire lit". This indicates we may prefer fusion systems with high Q (large ratio of energy released to energy input required to sustain the reaction), and this in turn suggests that mainly inertial confinement systems may prove more desirable than mainly magnetic confinement systems.

5.0 Results

5.1 Low-Cost Chemical Propellant Delivery to Orbit

This concept was investigated by MSFC personnel and received little consideration under the present study. The status is included for completeness.

The essence of the concept was that if propellant could be delivered to low Earth orbit at low cost, large initial masses would not preclude affordable missions. The specific mission concept employed gun-launch to deliver water to orbit; the water was to be electrolyzed to hydrogen and oxygen for use in conventional rocket stages.

The case examined to test the idea was an opposition Mars round trip in a difficult year. The mission ideal delta V is on the order of 30 km/sec. Chemical rocket jet velocity is about 4.5 km/sec. The rocket equation yields a mass ratio for a single stage of almost 800; practical mass ratios are less than 10. If we divide the 30 km/sec into 6 segments of 5 km/sec each, the mass ratio per segment is about 3, and the ratio payload/gross mass for each segment is about 4.5 for practical system masses. The payload/gross mass ratio for a six-stage system is in the range 5000 to 10,000.

The result was that hardware masses for the mission case examined were prohibitive even if propellant in orbit is without cost. The power levels required to produce the hydrogen and oxygen were excessive.

This result does not show that low-cost propellant in orbit is a bad idea, nor that propellant production in space by electrolysis is a bad idea. What it does show is that low-cost propellant delivery to orbit is not a practical substitute for a high-energy propulsion system for highly demanding missions.

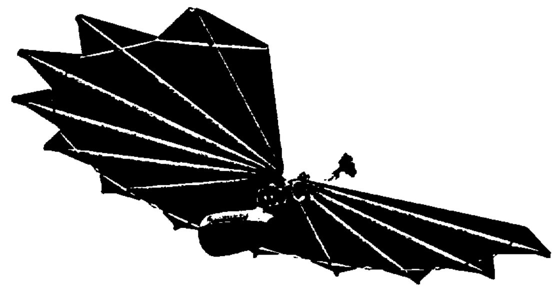
5.2 Electric Propulsion

5.2.1 Characteristics

Electric propulsion uses an electric power source to drive a thruster which converts electric power to kinetic energy of a flowing stream of propellant. Because the conversion process is, in general, not highly efficient, the energy source must be nuclear or off-board. Solar electric propulsion at low power (few kWe) is in operational use for stationkeeping of communications satellites. One operational NASA mission, Deep Space 1, has occurred and others are planned. Nuclear electric propulsion has been studied in some depth for years and a few studies have considered electric propulsion powered by offboard laser sources using photovoltaic conversion on the spacecraft.

Main features and characteristics of electric propulsion are noted in Figure 5.2-1. Electric propulsion is a power limited system, i.e. its main limiting factor is the net power-to-mass ratio available from the powerplant (or conversion system, e.g. solar array) and thrust delivery system. Specific impulse is not limited except by the speed of light. For any particular mission and trip time, a power-limited system will have mass and cost optima for specific impulse. Short trip times at high delta V may not be feasible because the system simply cannot deliver enough delta V in the time available due to its power limitations.

- Main Features
  - Nuclear or solar electric multi-megawatt generator
  - Electric propulsion system
    - Plasma, MHD or ion
  - Propellant storage & delivery system
- Operating characteristics
  - Long thrusting periods
  - Typically > 50% of transfer time
  - Adapts to various mission profiles
  - Low accelerations
  - Low propellant consumption
    - Isp 2500 - 10,000 seconds
  - Power-limited
    - Higher Isp translates to less thrust
    - Roughly inversely proportional
    - There is an optimum Isp for any given transfer time



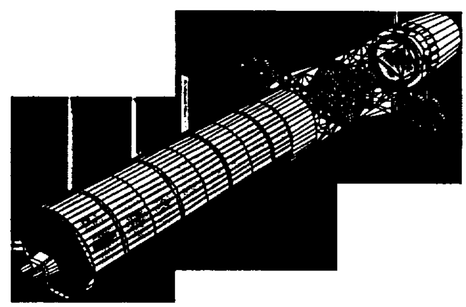


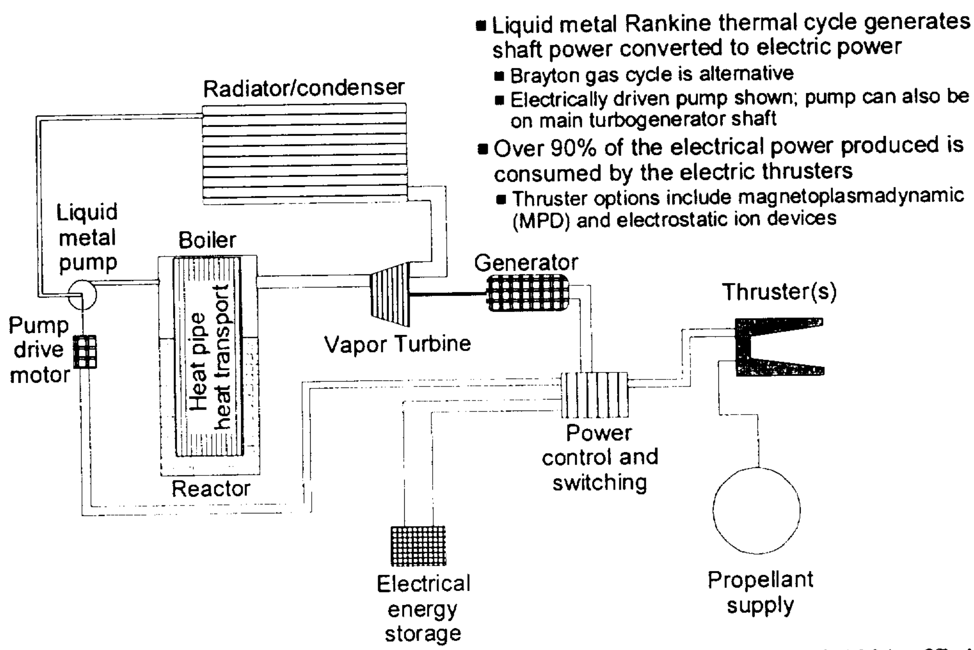
Figure 5.2-1: Characteristics of Electric Propulsion

5.2.2 Principles of Operation

In the simplest representation, an electric propulsion system is an electric powerplant and a propellant supply connected to a thruster which uses electric power to accelerate the propellant to form the propulsive jet. Figures 5.2-2 and 5.2-3 show schematic diagrams for nuclear electric propulsion systems using Rankine and Brayton power generation cycles. In the case of a solar electric system, the nuclear generating plant would be replaced by solar arrays.

The nuclear options use a fission reactor to generate heat, which is converted to electrical power by a thermodynamic cycle (to produce shaft power) and an electrical generator. The Rankine system is a liquid-vapor phase system like a steam powerplant. To reach temperatures needed for efficient heat rejection in space, a liquid metal (usually sodium) is commonly proposed. The liquid metal is pumped to high pressure, boiled to vapor in a heat exchanger heated by the reactor, expanded through a turbine, cooled and reliquefied by a radiator, and returned to the pump. Rankine cycles optimize (for minimum mass) at about 20% efficiency due to mass trends for the radiator.

The Brayton system is a closed-cycle gas turbine powerplant. Operation is similar to the Rankine cycle except that a gas compressor replaces the liquid pump, and an additional heat exchanger (the recuperator) is used. The gas compressor is usually on the shaft with the turbine, since total shaft power in this cycle is high. The recuperator allows the cycle to reject heat at a higher average temperature, significantly improving efficiency. Optimization is more complex, involving mass trends for the turbomachinery,



radiator and recuperator. The cycle usually optimizes between 30% and 40% efficiency for large (multi-megawatt) systems, and less than 30% for small systems.

Figure 5.2-2: Nuclear Electric Rankine Cycle System

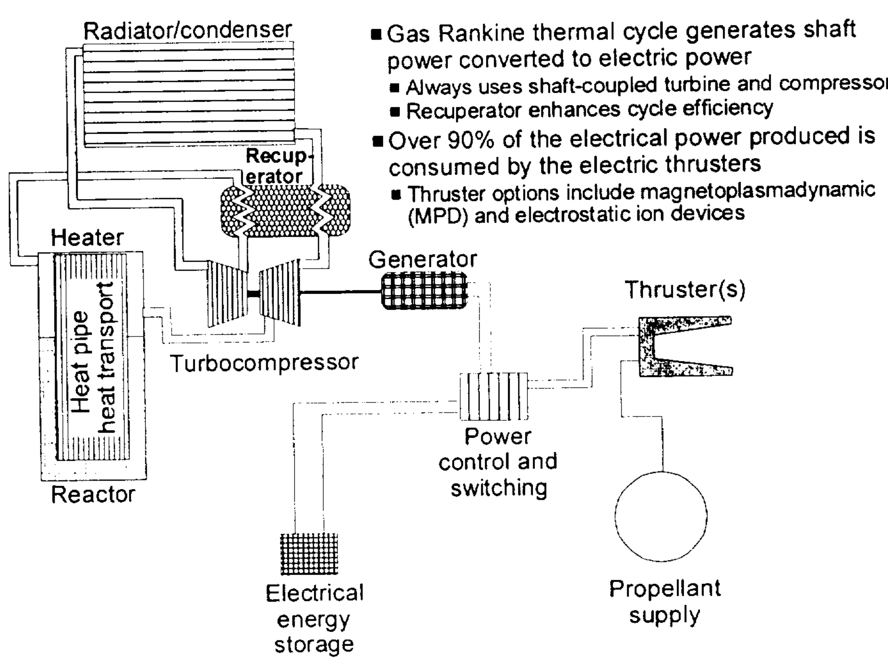


Figure 5.2-3: Nuclear Electric Brayton Cycle System

Thrusters use electrostatic or electromagnetic forces to accelerate the propellant, which must be ionized prior to or during acceleration. Thermal arcjets have been used on communications satellites but are limited to specific impulse below the requirements of missions considered here. Electrostatic engines have been tested for many years, flew on DS-1, and are now in commercial use for large communications satellites. The flight experience is with Isps between 3000 and 4000 seconds at power levels of 2 to 5 kWe. Hall effect thrusters are also commercially available at the few-kW power levels with Isps in the range 1500 to 2000 seconds. Electrostatic devices exhibit efficiencies between 60 and 70%. Several types of electromagnetic force (**E** X **B**) devices have been tried; these use the vector force produced by an electric current in a magnetic field to accelerate a neutral plasma. Figure 5.2-4 shows an experimental device in operation. Electromagnetic devices are better suited to high power, and experimental ones have exhibited efficiencies in the range 50 to 60%.

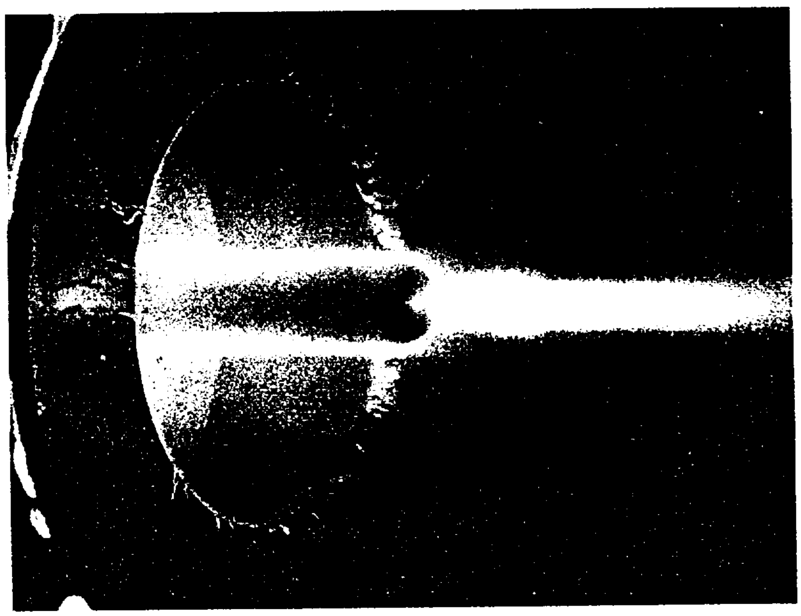


Figure 5.2-4: Electromagnetic Thruster in Operation

5.2.3 Technology and Development Challenges

The main challenges are summarized in Figures 5.2-5 and 5.2-6. This family of technologies is presently in use, and the main challenges are scaling up to high power, developing nuclear power sources, and improving efficiency and mass characteristics.

5.2.4 Benefits

Principal benefits are summarized in Figure 5.2-7.

5.2.5 Representative Mission Applications

Figure 5.2-8 illustrates a low-delta-V mission for transfer from Earth escape to the Earth-Sun L2 point. Two transfer times are shown. The delta Vs do not include reaching Earth escape velocity. One of the limitations of electric propulsion is that departures from deep gravity wells are slow and require more delta V than for high-thrust systems. High-thrust escape from Earth orbit requires about 3.2 km/sec. Low thrust, in the limit of infinitely low thrust, requires about 7.5 km/sec; numerical integration results for typical low thrust systems indicates between 5 and 6 km/sec. Low-thrust system architectures for human missions include a (typically chemical propulsion) means of delivering/retrieving the crew to/from the space vehicle after/before the spiral so that crew onboard time does not include spiral time.

Figure 5.2-9 shows interplanetary trajectories for human round trips for "easy" and "difficult" Mars years. The "easy" year mission could be achieved with a power-to-mass ratio (α) about 10 kg/kWe while the "difficult" year mission needs about 5 kg/kWe.

Figure 5.2-10 shows a mission profile diagram for these missions, illustrating the use of chemical propulsion crew vehicles at Earth and Mars.

• High efficiency with high power devices	• Essential for good system performance
• Thruster lifetime	• Arc devices
• Lightweight efficient power processing – Special characteristics may be required • Arc stabilization • Pulse generation with efficient energy storage/management – Thermal control	• Major weight driver, part of the efficiency chain – Necessary to be compatible with thrusters • Can be efficiency issue • Required for pulsed thrusters; must store energy between pulses – Keep power elex cool
• High power test facilities – Megawatt-class – Need good vacuum and ability to measure thrust accurately	• Necessary for R&D; qual – To handle high-power thrusters – Thrust measurement has been a problem in the past.

Figure 5.2-5: Main Challenges for Electric Propulsion Systems

5.2.6 Evolution Paths

Electric propulsion evolutionary trends are summarized in Figure 5.2-11. Top-level technology performance parameters are indicated for each mission application. This illustrates how growth in technology capability results in growth in mission capability.

5.2.7 Risk Management

Risk management considerations are shown in Figure 5.2-12. These can be significantly different for nuclear and solar electric systems. Solar electric issues are focused on scale-up and cost factors while nuclear electric issues are focused on costs of nuclear electric power source development and nuclear safety.

- Reactor Fuel Form
  - Life at temperature
  - Stable reactivity
  - Materials compatibility
- Heat Transfer Means
  - In-core vs heat pipes
  - Materials compatibility
- Cycle selection
  - Weight and efficiency
  - Cost
  - Testability
  - Materials compatibility
- Systems testing
  - Facilities availability
- High-power solar
  - Lightweight deployable arrays
  - Distributed power processing

- Development time
  - Swelling, materials stability
  - Need reactor control over life
  - Perennial problem
- Input to reactor design
  - First-order trade study
  - Issue for heat pipes, Rankine cycle
- Input to system design
  - Rankine probably favored
  - Brayton probably favored
  - Brayton favored
  - Brayton favored
- Must demo system life
  - Nuclear testing
- No experience
  - Acres & acres
  - Can't distribute megawatts at LVdc

Figure 5.2-6: Challenges for Electric Propulsion Power Generation

- Mission flexibility and adaptability
  - For example, no requirement for parking orbit alignment at Mars; prefer high-altitude parking orbits
- Fully reusable in-space transportation system
- Reduced resupply mass for next mission
- Can be designed to be highly redundant
- Opportunity for substantial long-term technology growth

Figure 5.2-7: Benefits for Electric Propulsion

5.2.8 Satisfaction of Needs

Ability of high-power electric propulsion to satisfy HEDS needs is summarized in Figure 5.2-13. Electric propulsion is a highly flexible technology, but it has power-to-mass ratio limitations that prevent it from being a complete solution to HEDS needs. Nuclear-electric options may have the growth potential to reach beyond the inner planets for human exploration. Nuclear-electric options are in fact very potent for robotic exploration missions to the outer planets. A Neptune orbiter mission, for example, might require a mission delta V about 60 km/sec to reach Neptune and enter an orbit in a reasonable time. A nuclear electric system with Isp 10,000 seconds and power-to-mass ratio 15 kg/kWe can deliver this delta V in something like 3 years, which is far faster than chemical/gravity-assist options, which are limited to much lower mission delta V.

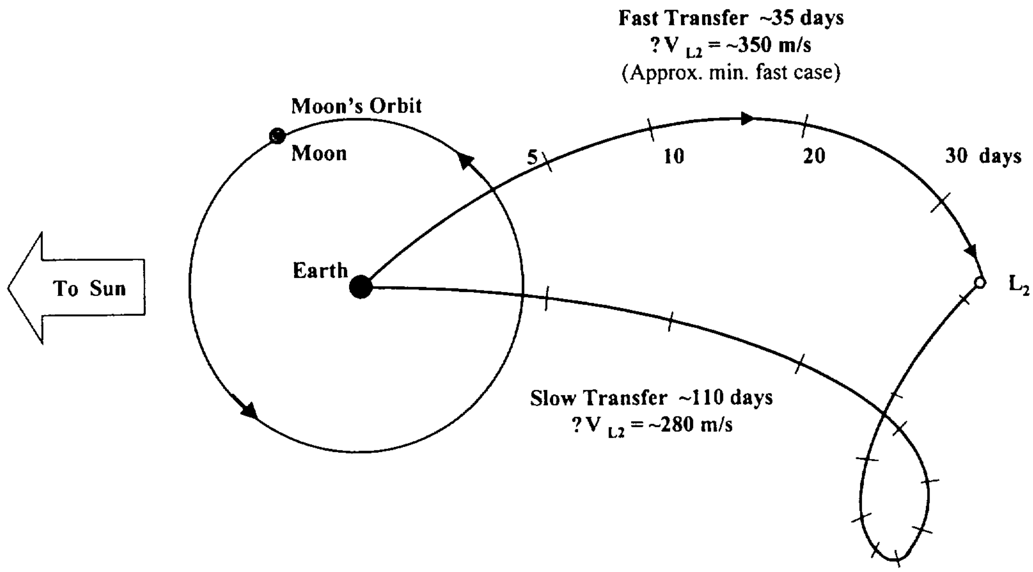
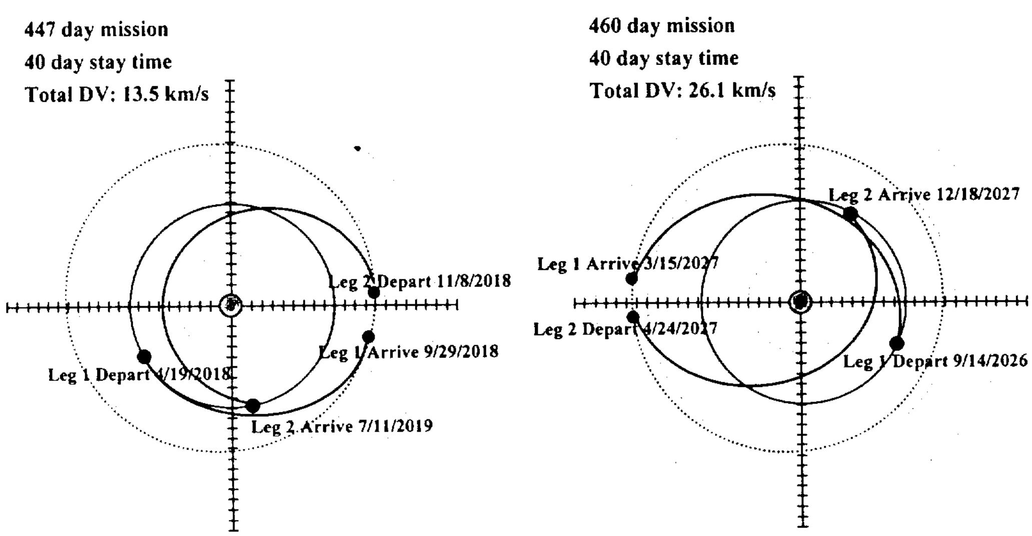


Figure 5.2-8: Mission Profile Maps for Transfer to Earth-Sun L2



"Easy" year (2018) "Hard" year (2026)

Figure 5.2-9: Mission Profile Maps for Mars Round Trips

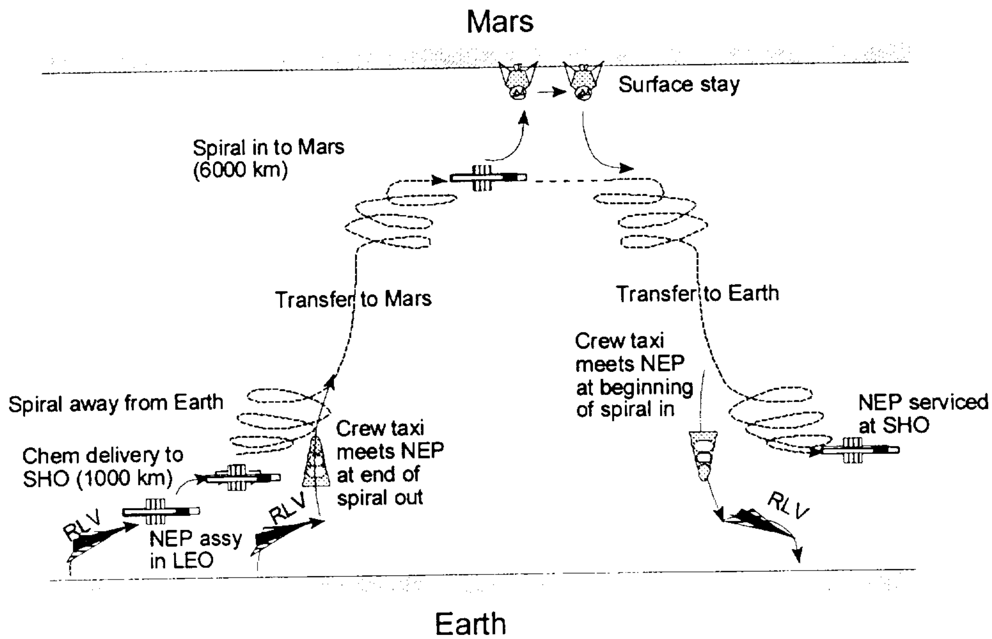


Figure 5.2-10: Mission Diagram for Electric Propulsion to Mars

Mission Class	Isp	Source	Power Level MWe	Alpha kg/kWe	Time Frame
Robotic: Comets, Asteroids, Inner Planets	3000	Solar or Nuclear	0.01	25 - 50	Now
Robotic: Outer Planets, Kuiper Belt	4000	Nuclear, solar if α ~ 10	0.1	10 - 25	2010 - 2015
Robotic: ISM, Oort Cloud	5000 - 10000	Nuclear	0.2	15	2015
Humans to Mars	3000 – 4000	Solar or Nuclear	5 - 20	5 - 10	2020
Humans to Jupiter, or Mars fast trip difficult year	10000+	Nuclear	10's +	<5	2030+

Figure 5.2-11: Electric Propulsion Technology Trends

- Performance (power-to-weight, efficiency); projected vs know-we-can-do; approach to bridge the gap
  - "Lost art" risk; availability of expertise for nuclear space power
- In-space testing and development needed for large-scale SEP
- Cost ... Technology advance, full scale development, unit cost, facilities and operations cost
  - Sources of estimates; uncertainty issues
- Nuclear safety for nuclear electric systems
  - R&D and testing
  - Operational ... Public and crew safety
  - Operational ... environmental impact, e.g. neutron capture in atmosphere -> carbon-14
  - End-of-life disposal

Figure 5.2-12: Risk Management Considerations, Electric Propulsion

- Single in-space transportation technology serves many projected mission needs
  - Probably same vehicle(s) can serve all needs; relatively little tailoring required
  - Reusable; don't need new major appropriation for hardware for every mission
- Power-limited nature precludes high delta V, short duration missions such as 1-year Mars round trip
- Nuclear electric may have growth potential to advanced human missions (humans to Europa?)
  - Depends on duration limitations and power-to-mass ratio achieved.

Figure 5.2-13: Satisfaction of HEDS Needs by Electric Propulsion

5.3 Nuclear Thermal Propulsion (NTP)

The primary subject of this section is solid-core fission reactor nuclear thermal propulsion, a demonstrated technology. Secondary attention is given to gas-core fission reactor nuclear thermal propulsion, which is a speculative technology.

5.3.1 Characteristics

A nuclear fission reactor is used to generate heat, which is directly transferred to hydrogen propellant. The reactor is made of very high temperature materials to enable as high a propellant temperature as possible. Reactors have been tested to about 2700K with hydrogen flow, corresponding to a specific impulse about 900 seconds. Fissionable fuel materials have been tested to about 3000K which corresponds to specific impulse about 950 seconds. Reactor operating times up to one hour have been demonstrated.

Very high power is attained by such reactors. Test reactors reached 5000 megawatts thermal, corresponding to a thrust level roughly 250,000 lb. Current planning deals with power levels less than 1000 megawatts (easier to test) and correspondingly lower thrust levels. Characteristics are summarized in Figure 5.3-1.

- Main Features
  - Nuclear reactor heat source and heat exchanger
  - Liquid hydrogen propellant
  - Isp 900 - 1000 sec (gas-core higher)
  - High thrust
    - Engine thrust/weight depends on thrust level, typically 3 - 10
- Operating Characteristics
  - Thrusting periods ~ 1 hour
  - Restartable, multiple burns
    - Total life ~ 10 hours
  - Not radiological hazard until first power-up in space
  - Deep space end-of-life disposal typically assumed
  - May be capable of producing electric power ("dual mode") during mission coast periods

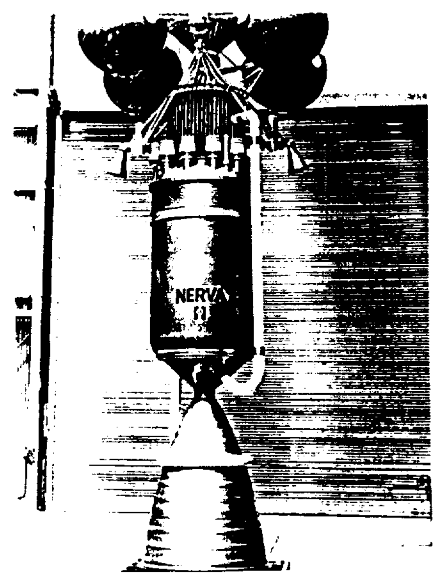
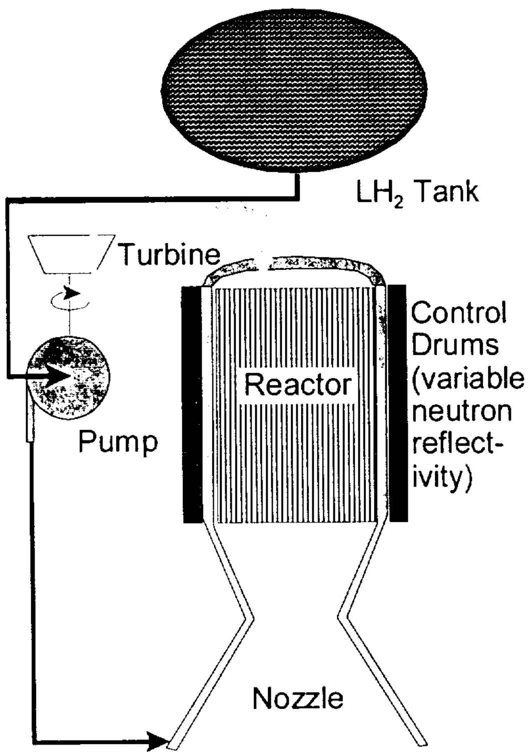


Figure 5.3-1: Main Characteristics of Nuclear Thermal Propulsion

5.3.2 Principles of Operation

The nuclear thermal rocket is a simple cycle, depicted in Figure 5.3-2. The power cycle that drives the hydrogen pump is an expander cycle, in which the hydrogen flow from the pump absorbs heat by cooling the reactor thrust chamber and then expands through a turbine to drive the pump.

In the case of the gas-core fission rocket, the solid structure of the reactor is replaced by a high-temperature, high-pressure uranium/hydrogen plasma circulating in a vortex flow, surrounded by hydrogen flow through the chamber exiting through the nozzle. It is hypothesized that such a configuration can provide good separation between the vortex flow and the through flow such that the fissionable material is retained in the reactor while heating the through flow to very high temperatures. Attainable specific impulse is variously estimated from about 1500 seconds to several thousand seconds. Feasibility of vortex containment remains to be experimentally verified.



- Power Cycle
  - Liquid hydrogen expander cycle
    - Hydrogen pumped through cooling jacket, and reactor support structure cooling channels (not shown)
    - Gains energy by removing heat
    - Expands through turbine to drive pump
      - Typical pump discharge pressure 2000 psi
      - Typical turbine exit pressure 1000 psi
- Thrust Cycle
  - Heating and Expansion
    - Hydrogen heated in reactor flow channels to ~ 2700K
    - Expands through high area ratio nozzle to velocity > 9000 m/sec (Isp 900 to 1000 sec).
    - Reactor power and hydrogen flow are controlled to maintain near-constant hydrogen exit temperature
    - Hydrogen flow is metered after main thrust shutdown to remove residual fission heat from the reactor

Figure 5.3-2: Prinicples of Operation for Nuclear Thermal Propulsion

5.3.3 Technology and Development Challenges

The challenges are summarized in Figure 5.3-3. Since this (NTP, not gas core) is a largely developed technology, the challenges relate to re-establishing the scientific and developmental infrastructure, and dealing with increases in environmental and safety sensitivities since the active development program ceased in 1972.

For the gas-core option, the main technical challenge is to develop and demonstrate a reactor/thrust chamber configuration with satisfactory fissionable containment, neutronics, and heat transfer characteristics.

5.3.4 Benefits

Benefits are summarized in Figure 5.3-4. Nuclear thermal propulsion mission analyses have focused on human Mars missions because planning for these missions was the primary focus of NASA's mission planning during the period of experimental NTP development activity. NTP is the only propulsion technology presently at a high technology readiness level that has enough Isp and thrust to address all Mars opposition round trip mission opportunities.

A number of mission analyses have also addressed use of NTP for lunar missions but the case here is not so compelling; chemical propulsion, especially with use of lunar-derived propellants, is competitive. NTP offers excellent performance for high-energy robotic missions. However, because of reactor critical mass requirements, NTP solutions tend to be large and expensive for these applications.

• Re-establishing nuclear rocket development infrastructure	• No active program since 1972; key technical personnel retired
• Fuel form selection, development and qualification	• Nerva-type vs Russian twisted-ribbon vs particle bed
• Nuclear rocket test facilities for development and qual	• Need fission product containment
• Long-term hydrogen cryogenics in space	• Mars and asteroid missions ~ 1 year

Figure 5.3-3: Challenges for Nuclear Thermal Propulsion

- Extensive technical heritage
  - Technology was far advanced by Rover program
- Enough Isp to address all Mars opportunities
- High thrust
  - Simplifies development and qualification
  - Simplifies mission design
- Potential to serve as electrical power generator during coast periods

Figure 5.3-4: Benefits for Nuclear Thermal Propulsion

5.3.5 Representative Mission Applications

The most representative application is a human round-trip mission to Mars on an opposition profile. NTP has the performance to accomplish these missions on a 1-year duration for "easy" to "moderate" Mars opportunities, and on 15-month duration for "difficult" opportunities. Trajectory path graphs were not conveniently available, but are similar to Figure 5.2-9.

Figure 5.3-5 is a diagram for a representative "difficult" opportunity profile. To reduce initial mass in Earth orbit, a cargo mission is used to pre-place the Earth return propulsion plus the Mars lander/ascent vehicle in Mars orbit. (A 500-km circular orbit is used at Mars since apsidal and nodal alignments for 1-sol elliptic orbits usually cannot be worked out for opposition missions.) The crew mission transfers to Mars on a trajectory which can be altered by gravity assist at Mars to achieve an abort return to Earth, should such a decision be made before Mars capture. (Once capture is initiated, an abort is no longer possible, but the return vehicle could be used for return to Earth without a Mars landing.) In Mars orbit, the crew vehicle performs rendezvous with the cargo vehicle. The crew employs the lander/ascent vehicle for surface exploration. After return to Mars orbit, the transfer habitat and Earth entry module are mated to the return vehicle and the crew returns to Earth. The Earth entry module is used for crew landing on Earth. The interplanetary propulsion systems and the transfer habitat are expended.

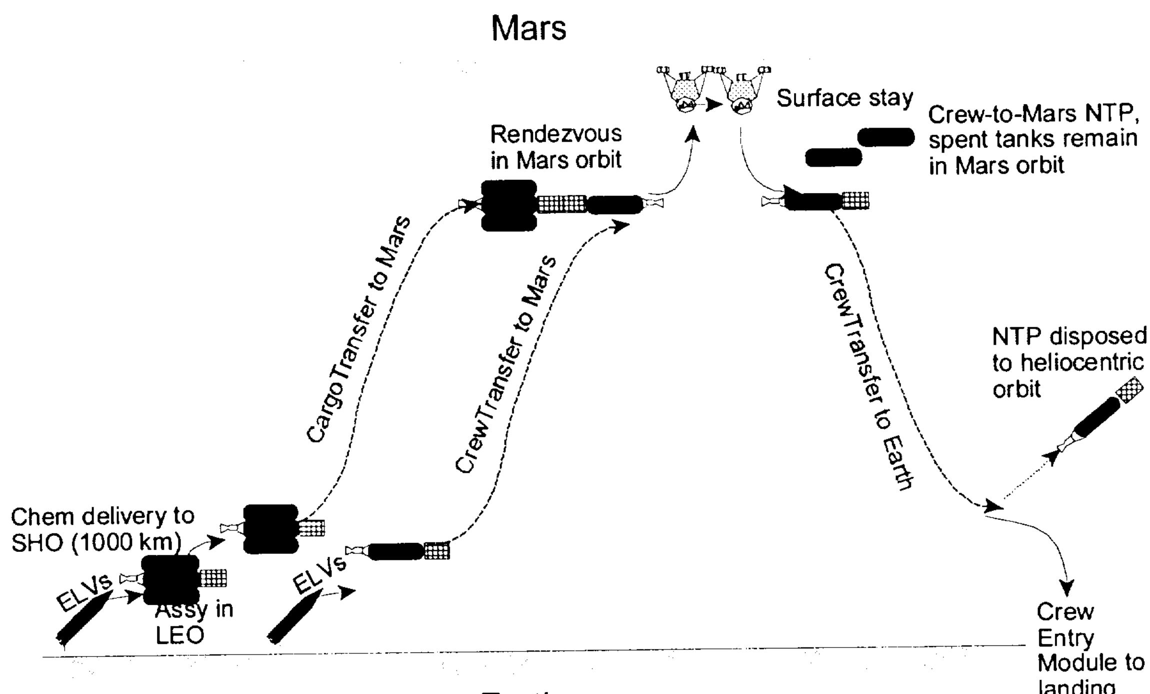


Figure 5.3-5: Mission Profile Diagram, Split Mission

5.3.6 Evolution Paths

Since the technology was matured by extensive full-scale ground test in the 1960s and early 1970s, no evolution path was prepared. While a reasonable effort in fuel form development should precede full-scale development of an NTP engine, the technology has been demonstrated to about TRL-5. A product-improvement evolution could occur, improving thrust-to-weight, engine life, and specific impulse.

The gas-core concept represents an evolution potential, but should not be represented as an evolution path until basic feasibility questions are answered.

5.3.7 Risk Management

Risk management for NTP solid-core rockets is concerned mainly with environmental and safety issues, as indicated in Figure 5.3-6. There is not a feasibility issue. Secondarily, early emphasis on fuel form demonstration will contribute to an orderly development program.

For the gas-core concept, there are feasibility issues, mainly concerning containment of the fissionable energy source, and these need experimental resolution as a first step in technology advancement.

- Exhaust-containment test facility eases environmental concerns
  - Places premium on lower thrust range ~ 15K and engine clustering as needed
  - No radioactivity release
- Design missions for safe spent reactor disposal
- Initiate fuel form development program early
  - Build knowledgeable team
  - Long tent pole technical area
- For gas-core, give priority to containment feasibility analysis and test before significant development commitment

Figure 5.3-6: Risk Management for Nuclear Thermal Propulsion

5.3.8 Satisfaction of Needs

Ability of this technology to satisfy HEDS needs is summarized in Figure 5.3-7. Together with electric propulsion, this technology could represent a complete solution for exploration missions until such time as human missions to the outer planets or more distant asteroids enter into the planning base.

- Capable of all missions in current planning base
  - Lunar
  - Libration points
  - Near-Earth asteroids
  - Mars
- Not enough Isp for human outer planet missions, except possibly gas-core option
- Too large and expensive for robotic missions
  - Electric propulsion suffices for these missions
    - (Usually) no return requirement
    - Relaxed time constraints

Figure 5.3-7: Nuclear Thermal Propulsion Satisfaction of HEDS Needs

5.4 Tethers

5.4.1 Characteristics

The main subject of this review was momentum-exchange tethers since electrodynamic tethers are useful only in a moderately strong planetary magnetic field.

Momentum exchange tethers use gravity gradients ("hanging tethers") or rotation and relative motion to exchange momentum with a payload. In this review we considered only rotating tethers. Characteristics of these are displayed in Figure 5.4-1.

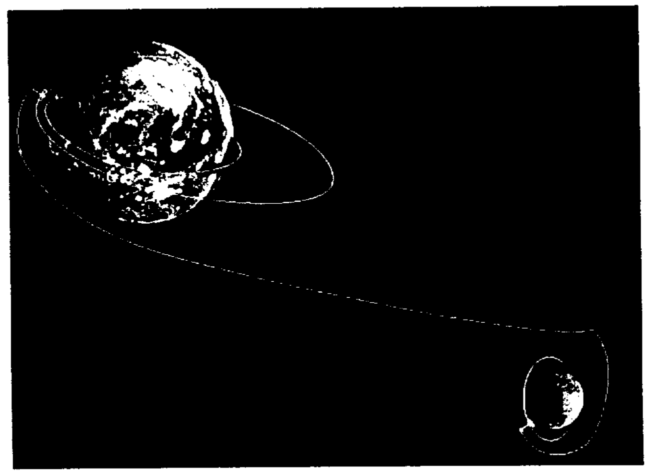
Electrodynamic (ED) tethers are electrical conductors, and the passage of the conductor (at orbital velocity) through a planetary magnetic field leads to **E X B** energy transfer. The transfer can be either from orbital energy to electric power in the tether, which can then be used to drive a load on the spacecraft, or if power from a source on the spacecraft is delivered to the tether, this power can be transferred to orbital energy and/or momentum, thus altering the orbit. Normally, this latter feature would be used to raise an orbit, as in space station reboost, or accomplish a plane change, as in an out-of-plane rendezvous application. As a propulsion device, the electrodynamic tether may be viewed as a form of electric propulsion without propellant.

5.4.2 Principles of Operation

Figures 5.4-2 and 5.4-3 show the principle of operation of a typical rotating tether in Earth orbit. Such a tether is typically 200 km in length so that the tip relative velocity (e.g. 1.5 km/sec shown) represents a slow rotation rate such as one revolution in 15 minutes. The tip speed is limited by strength of the tether material; 1.5 to 2 km/sec is typical for state-of-the-art materials such as Spectra® 2000. As shown in the figure, the tether rotation is such that tip speed subtracts from mean orbital velocity at the lower altitude and adds to it at the higher altitude. Thus, capture at the low point and release at the high point accomplishes the momentum transfer to the payload. The momentum transferred comes from the orbit angular momentum of the tether facility. Accordingly, the facility needs a central mass great enough that transfer of momentum to the payload does not deorbit the tether. Tether facility momentum can be made up by a returning payload, electric propulsion, or using the tether also as an electrodynamic tether to reboost the orbit.

Figure 5.4-4 illustrates the operating principle for the electrodynamic tether.

- Main Features
  - Transfer momentum to/from space vehicle without propellant
  - Permanent/long-life facility in orbit at Earth, Moon or Mars
  - Capable of launch assist by capturing suborbital payload from launch vehicle
- Operating characteristics
  - Large base mass with rotating tether ~100's km length
  - Maneuvering tether tip to assist space-craft capture



(Illustration courtesy Tethers Unlimited, Inc.)

Illustration shows a system of two tethers, in Earth orbit and in lunar orbit, for transfer of space vehicles from Earth orbit to the lunar surface and return. Propellant needed only for course correction and lunar liftoff/touchdown

Figure 5.4-1: Main Characteristics of Tether Systems

5.4.3 Technology and Development Challenges

Challenges are summarized in Figure 5.4-5. The tether material most commonly assumed is in production. Several experimental flights have demonstrated the basic operations of tethers. The physics is well known such that calculated performance and simulations should lead (and have led) to accurate predictions of actual performance. Therefore, the challenges are mainly in practical implementation of the technology.

5.4.4 Benefits

The major benefits of tethers, as noted in Figure 5.4-6, are that they don't use propellant and represent (for most applications) permanent or long-life facilities. Therefore, the use cost is small, although investment cost may be large. Rotating tether delta V capabilities are limited to values approximately twice the tip speed, i.e. 3 to 4 km/sec. These characteristics mean that tethers are best suited to repetitive, routine applications requiring modest delta Vs.

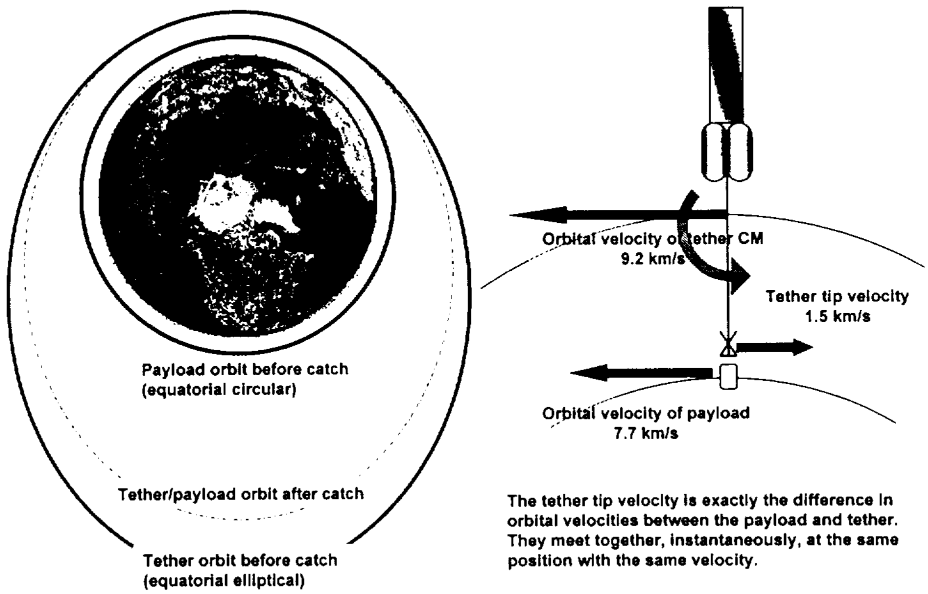


Figure 5.4-2: Momentum Tether Principle .. Payload Catch

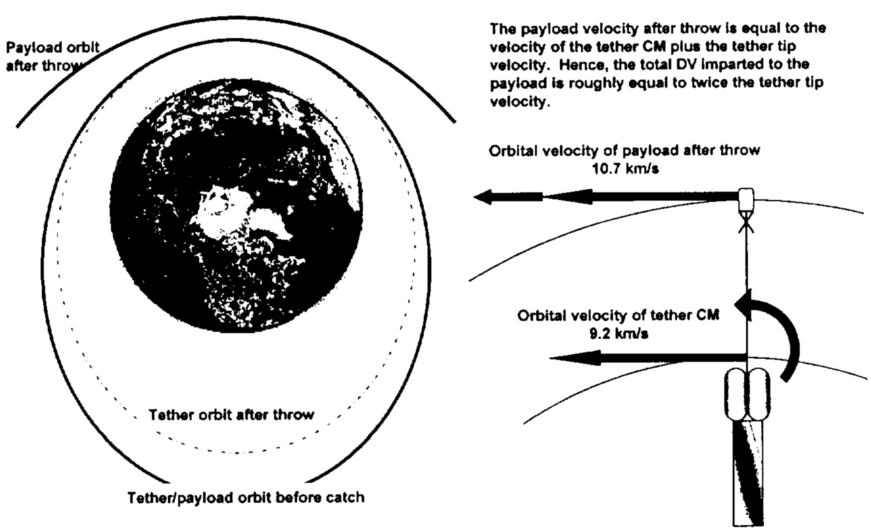


Figure 5.4-3: Momentum Tether Principle .. Payload Release

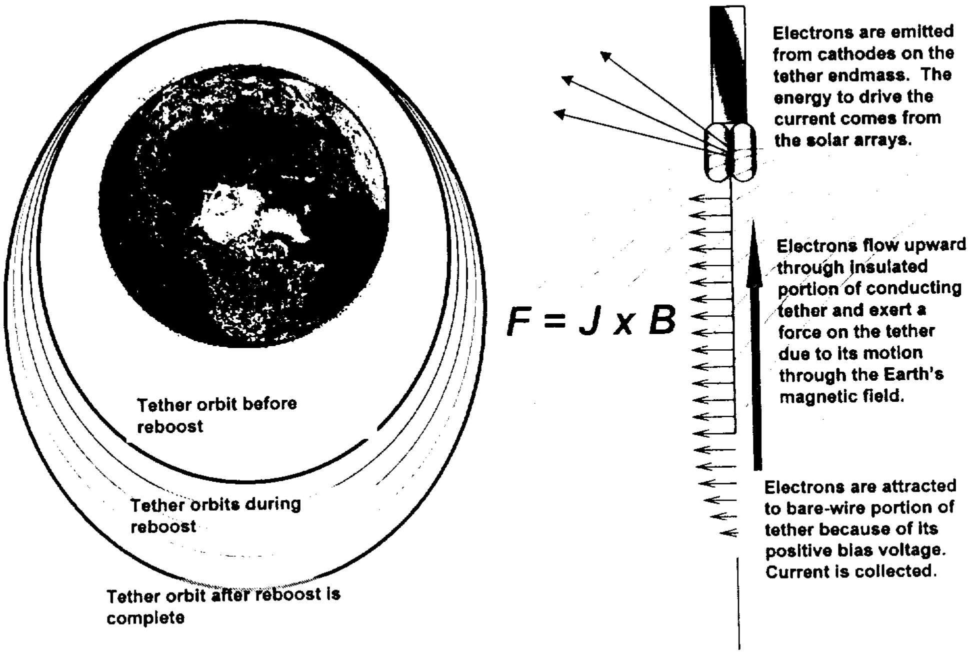


Figure 5.4-4: Electrodynamic Tether Operating Principle

- Long life in space with high-performance tether materials, suitably protected
- Determination of orbital debris generation potential
  - Tether is large cross-section
  - Able to evade known objects
- Fabrication/packaging/ deployment of large multi-strand tether assemblies
- In-space testing to develop/ validate design approaches and criteria
  - Tether operations
  - Tether performance in EM and rotating modes
- Development of operations plans

- Ensure materials and their protection are suitable for semi-permanent tether facility
- Collisions may generate debris even when tether survives
  - Larger than ISS
  - May be unwieldy due to length
- Must demonstrate ability to build large multi-strand tether, package for launch and deploy
- Development and demonstration objectives require in-space testing
  - Deploy and establish rotation
  - EM characteristics; momentum transfer
- Develop design requirements & technology goals

Figure 5.4-5: Challenges for Tether Systems

5.4.5 Representative Mission Applications

Three applications are most representative of advantageous use of tethers: (1) space station or other low Earth orbit facility reboost; (2) delivery of satellites from low

Earth orbit to geosynchronous transfer orbit (GTO); and (3) delivery and return of payloads to/from low Earth orbit to the Moon, as illustrated in Figure 5.4-7. For such applications, tethers may, if their technical challenges can be resolved, be so cost-effective that no alternative propulsion technology can compete.

The lunar transfer application includes a rotating tether in lunar orbit, which permits payloads to be "lowered" to the surface or picked up from the surface with relatively little delta V. The tether tip speed is about equal to the orbital velocity so that orbital velocity is nearly cancelled at the tether low point. This system, used for two-way traffic, could operate with very little propellant expenditure all the way from low Earth orbit to the lunar surface and back.

- Impart major propulsive impulse without propellant consumption
- Permanent facility (stays in Earth orbit), not expended on each mission

Figure 5.4-6: Benefits of Tethers

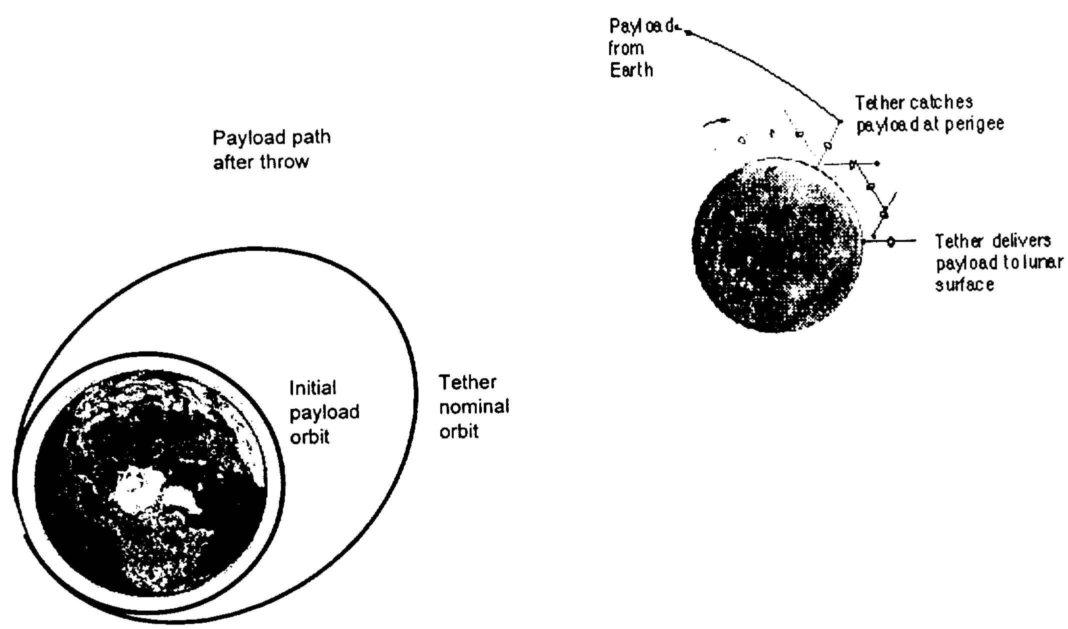


Figure 5.4-7: Mission Pictorial for Earth-Moon Tether Transportation

5.4.6 Evolution Paths

Near-term applications include satellite deorbit. The technology is ready for this today. LEO satellite deorbit is a simple application for a small ED tether, and more economic than conventional propulsion. Mid-term applications include routine transfer of satellite payloads from LEO to geosynchronous transfer orbit (GTO), and space station reboost. An interesting possibility is the use of Jupiter's very strong magnetic field for an ED tether. Orbital capture from an interplanetary trajectory without propellant might, for example, be possible.

Far-term applications include transfer of cargo and human space vehicles between Earth and the Moon, and Earth and Mars.

Figure 5.4-8 provides additional technical information on these mission applications. Figure 5.4-9 summarizes tether technology trends.

Tether Mission	Data Source	System Mass	Payld Mass	Sys/Pld	Improve-m't factor	Payback # uses
LEO Deorbit (ED)	TU website	25 kg	1500 kg	0.04	~5	<1
LEO-GEO transfer and S/C escape assist	Interpolated	23 t.	2500	9	~3	~3
ISS Reboost (ED)	SWAG	5 t.?	400 t.	0.01	>10	< 1 year
Nuc elec ED @ Jupiter	Unk.					
LEO-Lunar payload	TU website	26 t.	2500 kg	10.5	~3	~3
LEO-Lunar human/ cargo S/C	TU website	160 – 320 t.	15t. – 30 t.	10.5	~3	~3
LEO-Lunar human/ cargo S/C with Lunavator	TU website	57.7 + 92	5.5 t.	LEO-TLI 10.5 Lunavator 16.7	~15	2 – 10 depends how lunavator emplaced
LEO-Mars via HEO	TU website	900+	100+	9	~3	~3
LEO-Mars a la MERRITT	TU website	450	30	15	~10	2 – 10 depends how Mars MERRITT emplaced

Note: MERRITT system places tethers in inclined orbits at Earth and Mars; issue of rotating tether nutations (wobble) due to planet oblateness needs to be resolved

Figure 5.4-8: Technical Information for Typical Tether Missions

5.4.7 Risk Management

Risk management areas are summarized in Figure 5.4-10. These are mainly development risk management items, since the basic physics is well understood and key in-space demonstrations have already occurred.

Mission Class	Type	Approx Mass	Sys Mass/ Pld Mass	Sys Mass/ Propellant Saved	Time Frame
LEO Satellite Deorbit	ED	25 kg	0.04	0.2	Now
LEO-GEO and escape assist	Rotating dynamic	23 t. for 2500 kg payload	9	3	2010
ISS Reboost	ED	5 t.	0.01	<0.1	2005 - 2010
Humans to Moon & ret.	Rotating dynamic	150 t.	27	15	2020
Humans to Mars	Rotating Dynamic	500 – 1000 t.	10's +	10 - 15	2030+

Figure 5.4-9: Tether Technology Trends

- Nature of tether operation needs space-based testing for systems demonstrations
  - Ground-based testing can evaluate materials and components, system elements
- Develop incremental space test program
  - Emphasize low cost, high R&D value
    - Subscale where practical
    - Validation of analytical models
    - Reusable test hardware where possible
- Incremental development
  - Initial operational prototypes for robotic missions and satellite delivery
  - Human mission systems after designs validated

Figure 5.4-10: Risk Management Approach for Tether Technology

5.4.8 Satisfaction of Needs

Tether systems offer great cost reduction leverage in the realms of their practical operational feasibility, as noted in Figure 5.4-11. A reboost ED tether could, for example, essentially eliminate the problem of reboost propellant supply for a low Earth orbit space station. Similarly, a tether infrastructure for lunar transportation could make recurring operating cost for delivery and return of payloads to/from the Moon essentially equal to that for low Earth orbit. For conventional rocket propulsion the ratio (lunar surface delivery cost)/(low Earth orbit cost) is roughly 10.

- Significant to major reduction in launch requirements to perform certain HEDS missions as well as related exploration applications
  - Cost reduction potential could be enabling even if other technologies are mission-capable
  - Example ... lunar base
- Increases reusability and permanence of in-space transportation infrastructure

Figure 5.4-11: Tether Systems Satisfaction of HEDS Needs

5.5 Solar Light and Magnetic Sails

5.5.1 Characteristics

Characteristics of solar sails are summarized in Figure 5.5-1. Solar lightsails use the momentum of light coming from the sun and solar magnetic sails (magsails) use the momentum of the particle flux (solar wind) coming from the sun. Also proposed have been sails using artificial sources, either laser or microwave.

The momentum flux of light is P/c where P is the power in watts/m^2 and c is the speed of light in m/sec. For sunlight at Earth's distance P is about 1350 watts/m^2. Since c is 3×10^8 m/s the flux (i.e. force) is small, 4.5×10^{-6} N/m^2. Very large sails are needed to obtain a useful amount of force. For artificial sources, very high powers are required. A source with a gigawatt of light or microwave output generates a force of about 3 N when the light is absorbed by a sail. If the light is specularly reflected, twice this force is produced.

5.5.2 Principles of Operation

Principles of operation are shown in Figure 5.5-2. A lightsail is set at an angle to the incoming light, much as a sailboat sail is set at an angle, in order to generate a net force that can boost the orbit. (Force normal to the orbit path does not add energy to the orbit.)

A magnetic sail uses a large actual or virtual dipole field to deflect solar wind particles, causing momentum transfer to the sail. Since the solar wind momentum flux is far less than sunlight, a very large area field is required. Since the source of the field need not fill the capture aperture, the magsail may be lighter than the lightsail for equivalent force.

- Features
  - No propellant required
    - Momentum of sunlight or solar wind intercepted by large-area sail
  - Very large area intercept
  - Capable of orbit boost as well as deboost by changing angles of reflection
- Characteristics
  - Ultra-light large area structures
  - Reflective mirror surface for light sails
  - Very large area magnetic loop for magnetic sails
  - Typical force 5 N/km2
  - While there is no physical limit on acceleration, practical lower limits on structural mass cause sails to be very low acceleration devices

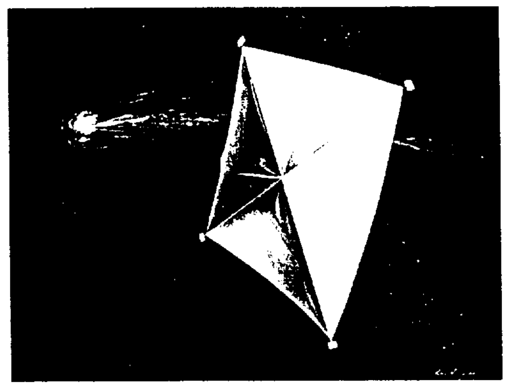


Figure 5.5-1: Characteristics of Solar Sails

- Sail set at angle to sunlight
  - Net force increases orbital momentum
  - Optimal angle about 35
  - Reversing sail angle deboosts orbit
  - Sail angle can be set by attitude control or adjustable flap(s)
- Magnetic sails create large dipole which deflects solar wind, generating force
  - Physical or virtual current loop
  - Solar wind momentum per unit area << light momentum, so very large field area required, but field strength can be low.
- Like solar electric power, efficacy drops ~ 1/r2

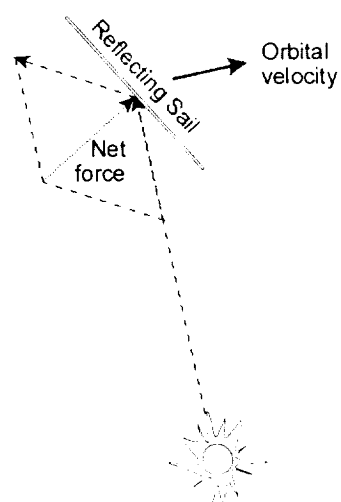


Figure 5.5-2: Solar Sails Principle of Operation

5.5.3 Technology and Development Challenges

Technology and development challenges are summarized in Figure 5.5-3. The main issue is attaining the very low mass per unit area required for an effective sail.

• Large area light weight structures	• Need << $1 g/m^2$ ($1 g/m^2$ = 0.0005 g sail alone)
• Highly reflecting lightweight sail material (light sail)	• Sail material needs to be >90% reflective and < $1 g/m^2$ i.e. <1 mm thick
• Deployment mechanisms/ approach	• Must deploy square km of this ultralight stuff

Figure 5.5-3: Solar Sails Technology Challenges

5.5.4 Benefits

Benefits are summarized in Figure 5.5-4. Main benefits are that sails require no propellant and are thus not delta-V limited, and that since sails do not involve major power conversion equipment, they should be cheaper to develop than options that do.

- No propellant required; directly use momentum from solar emanations
  - No delta V limitations
- Probably low development and unit cost compared to systems with active power generation and/or conversion
- No environmental issues

Figure 5.5-4: Benefits for Solar Sails

5.5.5 Representative Mission Applications

Sails are best suited to missions requiring high delta V which can be applied in the inner solar system (where solar fluxes are high) and for which mission duration is not a particular constraint, as noted in Figure 5.5-5. Note that for solar sails (or for that matter, solar electric) to reach high delta V, the acceleration attained by the sail must be greater than the local solar acceleration of gravity divided by about 6. At Earth's distance the solar gravity acceleration is $m/r^2 = 1.33 \times 10^{11}/(150 \times 10^6)^2 = 6 \times 10^{-6}$ km/sec^2 = 0.006 m/sec^2 = 0.0006g, and the acceleration required is about 10^{-4}g (or better).

- If a sail can achieve 0.0002 g, it can reach solar system escape and is therefore a valid scheme for sending robotic missions to the outer planets (on one-way trips without maneuver capability at destination)
  - 0.0002 g = roughly 60 km/s per year; a typical acceleration period would be ½ year for 30 km/s.
- For human missions, sails are most likely to see cargo mission applications, for delivery of cargo, or the cargo segment of a split mission, to Mars
  - Typical (slow) cargo delivery would be 100 t. in 1 year requiring 10 km/s. Average acceleration is 0.0003 m/s^2 which requires 32N thrust, more or less 6 km2 light sail.
  - Fast crew round trip requires ~ 40 - 60 km/s in 1 year, an average acceleration 0.0015 m/s^2 which may exceed capability of light sail.

Figure 5.5-5: Mission Applications for Solar Sails

5.5.6 Evolution Paths

Evolution paths are described in Figure 5.5-6. There are no physical principles to be proved for the lightsail; technology development requires reaching the needed very low mass per unit area and developing means for deployment and control in space. There is not really a physical principle to prove for the magsail either, but questions of practical implementation should be addressed by suitable space experiments. Unlike some of the technologies, which don't work on a small scale, the sails can start small and be scaled up.

5.5.7 Risk Management

A risk management approach is described in Figure 5.5-7.

5.5.8 Satisfaction of Needs

Satisfaction of HEDS needs is summarized in Figure 5.5-8. While sails cannot satisfy all needs, their expected comparatively low development cost and lack of difficult risk issues make sails a strong candidate for development to serve the needs they are well suited for.

- Experimental sails few hundred sq m in area, few kg mass, minimal instrumentation (decade of the 00s)
- Robotic mission sails thousands m^2 to 1 km^2, payloads 10s to 100s kg (decade of the 10s)
- Human mission cargo sails 5 - 10 km^2 (decade of the 20s)

Figure 5.5-6: Potential Evolution for Solar Sails

- Initial experiments could be low cost, secondary payloads
  - Demonstrate low mass/area structures and reflectors
  - Demonstrate sail deployment, control, navigation
- Developmental larger sail is appropriate for Discovery-class mission
  - First mission(s) should have main objective to demonstrate sail deployment and operation, a la DS-1
  - Later missions could focus on science objectives
  - Combining sails with Jupiter gravity assist gives window to farthest reaches of solar system, but probably relatively slow trips
- Growth from 1 km^2 to 10 km^2 appears routine scale-up
- There appear to be few if any environmental or safety risks
- In-space testing required to prove the technology
  - Ground tests appear limited to basic materials and structures

Figure 5.5-7: Risk Management for Solar Sails

- Because solar sails appear to be a low-entry-cost technology, they are useful whether or not they fill all needs
  - Suitable for some robotic missions
  - Suitable for some cargo missions supporting human missions
- Rapid human trips, or human trips to outer planets, not in the cards
- If the basic lightweight structures and sail materials technology can be developed, sails can probably be ready before the mission applications are ready.

Figure 5.5-8: Solar Sails Satisfaction of HEDS Needs

5.6 External Pulse Plasma Propulsion

This concept uses nuclear detonations to generate expanding plasmas which impact the space vehicle, imparting momentum in the process.

5.6.1 Characteristics

The concept is a derivative of the "Orion" concept on which several million dollars for research was spent in the 1960s. Main features are shown in Figure 5.6-1. The nuclear pulse units are relatively "small", delivering from 1 KT to 10 KT energy per pulse. An elaborate shock absorption mechanism must be integrated into the "pusher plate" system, but tests and analyses conducted during the Orion research indicated this system was feasible and practical.

This is a high thrust, high Isp system. It has few limitations on mission capability except that Isp is probably limited to about 5000 seconds by the requirement to not destroy the pusher plate.

Nuclear pulses in the magnetosphere will charge the van Allen belts with fission fragments and other ions and electrons, and create electromagnetic pulses which can damage or destroy satellites in Earth orbit. Therefore, the system must be delivered to a safe altitude by non-nuclear means.

- Main Features
  - Nuclear detonations generate expanding plasmas which impact vehicle, transferring momentum
  - Nuclear charges carried onboard
  - Vehicle includes an elaborate impact absorption system
- Operating characteristics
  - High thrust, high power, high Isp
  - Accelerations ~ 0.5 g
  - Adapts to various mission profiles
  - Modest propellant consumption
    - Isp 2500 - 5,000 seconds
  - Limited to use far from Earth due to contamination and EMP issues
    - Start altitude beyond GEO

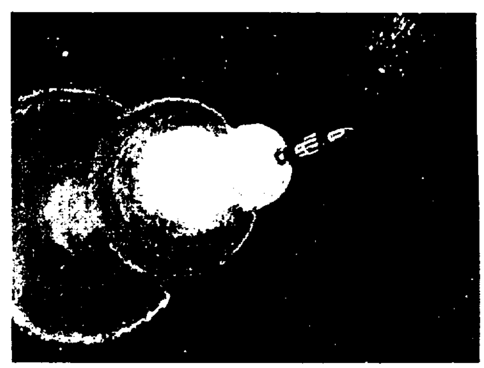


Figure 5.6-1: Characteristics of External Pulse Plasma Propulsion

5.6.2 Principles of Operation

The principle of operation is basically simple, and is illustrated in Figure 5.6-2. Pulse units are armed shortly before use to prevent inadvertent release of nuclear energy.

The pulse unit magazine may need to include neutron absorption materials to prevent the aggregation of pulse units from becoming a thermal-neutron critical mass.

- Nuclear detonations in space generate hot plasma, not blast waves or electromagnetic pulse (EMP)
- EPP uses "small" pulses
- Expanding plasma impacts pusher plate, transferring momentum
- Pulse rate ~ 1 per second; 2-stage shock absorber attenuates momentum impulses to near-constant acceleration
- Vehicle carries thousands of pulse units for typical mission delta V

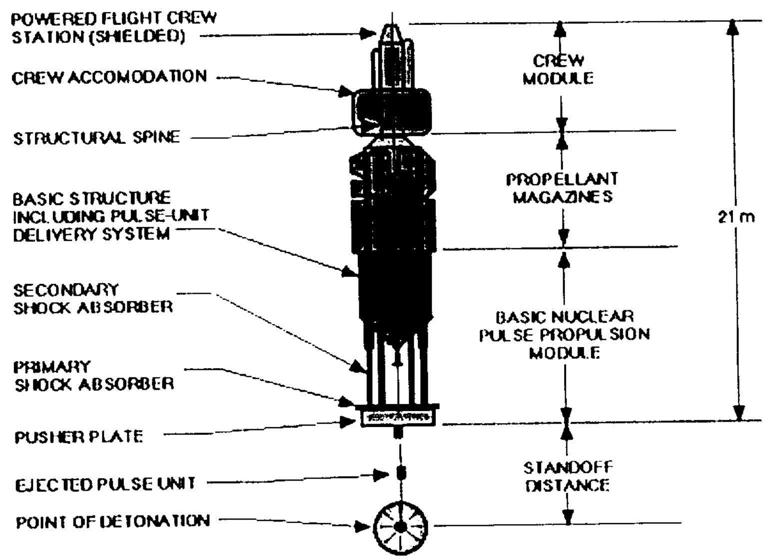


Figure 5.6-2: External Pulse Plasma Principle of Operation

5.6.3 Technology and Development Challenges

Challenges are summarized in Figure 5.6-3. The technology of nuclear detonations is of course well known. These challenges are mainly of an implementation nature. One challenge of importance is to obtain efficient nuclear energy release in small devices. The design of these units is not affected by typical military design constraints, so innovations may be possible.

5.6.4 Benefits

This is a high-thrust, high Isp system, as indicated in Figure 5.6-4. Its power-to-mass approaches that of a chemical rocket at a few hundred kW/kg of "engine" (the SSME is about 1250 kW/kg). The nuclear thermal rocket is about equivalent, but limited to Isp about 900. The most optimistic projections for a fusion system are about 100 kW/kg, and 10 kW/kg may be more likely. Fusion, of course, may reach much higher Isp.

Vehicle neutron activation can be minimized relative to a nuclear thermal rocket or nuclear electric rocket, both of which carry their fission product inventory on board.

• Integrated design of pulse units, pusher plate and shock absorbers	• Essential to reach desired performance levels and durability of system
• Pusher plate thermal/erosion protection	• Impacting plasma must be very hot to get desired Isp
• Delivery of large pusher plate to Earth orbit	• Pusher plate designs tend to 10m - 15m diameter
• Vehicle neutron activation	• Materials choices
• Verification testing compliant with nuclear test treaties	• Verification testing requires nuclear pulses in vacuum
• Operations compliant with nuclear treaties	• Avoid designation as potential weapons devices (achieving efficient operation with smaller pulses would be very helpful)

Figure 5.6-3: Challenges for External Pulse Plasma Propulsion

- High thrust and high Isp, potentially highest Isp of any proposed high-thrust technology
- Minimal vehicle neutron activation, low radiation hazard to crew and proximity operations
- Reusable system may be possible
- Potential disposal means for critical nuclear material

Figure 5.6-4: Benefits of External Pulse Plasma Propulsion

5.6.5 Representative Mission Applications

We show a human Mars mission as representative in Figure 5.6-5, for comparison with other systems. The external pulse system could also perform human missions to at least the nearer of the outer planets, if its upper Isp range of 5000 seconds is reached.

The mission profile is simple, complicated only by the need to use non-nuclear means to reach a safe starting altitude. In the mission diagram, the vehicle is shown returning to low Earth orbit but it may be preferable to leave the vehicle parked in a high orbit and ferry pulse units, payload and personnel for the next mission to the vehicle at its safe parking orbit.

5.6.6 Evolution Paths

The evolution path described in Figure 5.6-6 involves mainly developmental research, since scientific feasibility issues have been resolved. Most of the development activity should be relatively inexpensive and not controversial, but verification testing of the pulse units as well as the system involves significant treaty issues re the CTBT and the outer space treaties. In addition, at least the final systems verification tests presumably need to be done in space far from Earth. Costs will probably be significant.

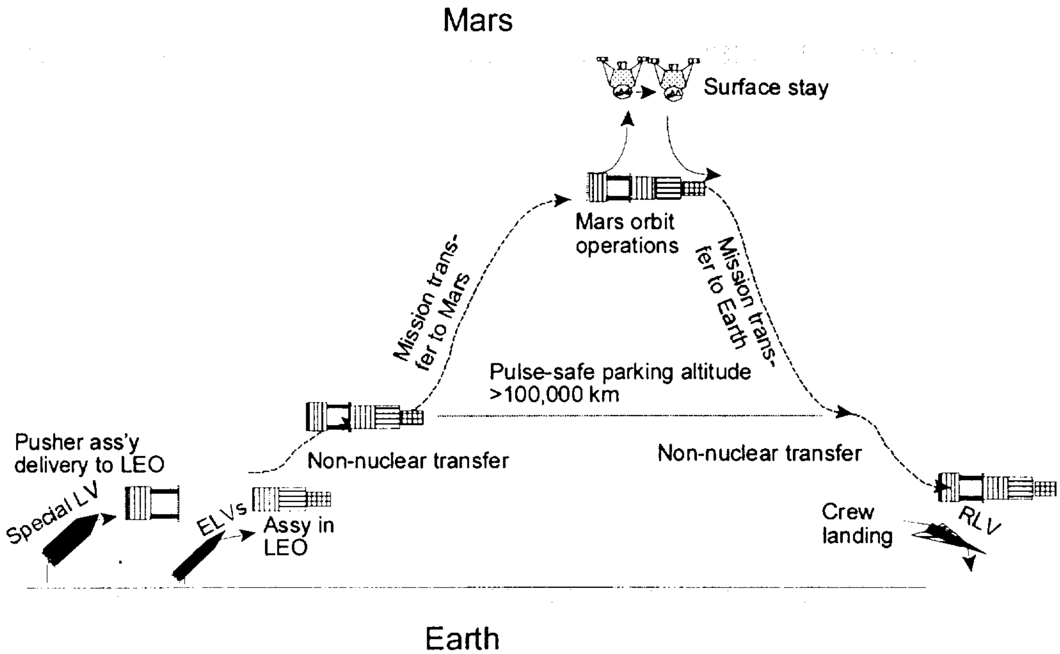


Figure 5.6-5: Mission Diagram for External Pulse Plasma Propulsion

- Design/analysis codes plus very-low-energy- release critical assembly tests should be able to accurately predict pulse unit performance
- Probably can devise a non-nuclear plasma accelerator to test pusher plate interactions and protection
- Pusher plate/shock absorber assembly reactions amenable to analysis
  - Non-nuclear explosive layer on pusher plate probably can give good fidelity full-scale test
- But eventually there must be nuclear tests in space
  - This will require treaty consideration

Figure 5.6-6: Evolution Paths for External Pulse Plasma Propulsion

If use of non-nuclear (i.e. chemical explosive) pulses is chosen as a means to perform the non-nuclear orbit transfers near Earth, another technology thread must be added to develop the non-nuclear pulse units and assess their performance. This may be an important area, since momentum transfer from unconfined pulses to a vehicle is not likely to be very efficient. (It doesn't need to be for nuclear pulses to achieve impressive Isp, but chemical sources are by comparison extremely energy-limited.)

Mission application evolution for this system, as indicated in Figure 5.6-7, will be different than for some of the other systems, since this system leverages a nuclear technology upon which very large sums have been spent over a long period of time.

- Like the mythical Phoenix, this concept probably emerges from the development program fully grown.
- Further evolution would be in the nature of product improvement

Figure 5.6-7: Mission Evolution for External Pulse Plasma

5.6.7 Risk Management

Risk management issues are described in Figure 5.6-8. These issues are for the most part self-explanatory. The security issues will take time to work out satisfactorily, and serious research on this concept cannot proceed until this is done.

- Performance (Thrust-to-weight, Isp, durability)
  - How to reach high confidence in performance attainment without very expensive nuclear testing
- At least some in-space testing and development needed for EPP
- Cost ... Technology advance, full scale development, unit cost, facilities and operations cost
  - Sources of estimates; uncertainty issues
- Nuclear safety
  - R&D and testing
  - Operational ... Public and crew safety
  - End-of-life disposal: Is it required, or can vehicle activation be kept so low that it's not a problem?
- Security and classification
  - Proliferation; protection of classified nuclear data
  - Access to relevant classified data

Figure 5.6-8: Risk Management Issues for External Pulse Plasma

5.6.8 Satisfaction of Needs

As indicated in Figure 5.6-9, the external pulse technology can lead to very potent in-space transportation propulsion. Since it does not render "small" very well, a companion technology is needed to support ambitious robotic missions.

- This in-space transportation technology could serve many projected mission needs
  - Probably same vehicle(s) can serve relevant needs; relatively little tailoring required
  - Best suited for large-scale high delta V missions
    - Not clear that a "small" version can work
- Develop 100 kWe-class solar & nuclear electric propulsion to serve high-performance robotic missions

Figure 5.6-9: External Pulse Plasma Satisfaction of HEDS Needs

5.7 Fusion and Other Advanced Concepts

In this section, we concentrate mainly on fusion, since it is much farther advanced than such concepts as anti-matter.

5.7.1 Characteristics

Characteristics of fusion propulsion are described in Figure 5.7-1. Fusion is characterized by moderate thrust (more than electric propulsion and perhaps a lot more) and very high Isp. Even the minimum projection for fusion acceleration, 0.001 g, represents about 300 km/sec per year, so in the context of year-long (or so) missions, a fusion system can go anywhere in the solar system.

- Main Features
  - Controlled fusion reactions generate hot plasma which forms propulsive jet
  - Magnetic (continuous) & inertial (pulsed) confinement options
  - Propellants D-He^3 or D-T
- Operating characteristics
  - Moderate thrust, high power, high Isp
  - Accelerations ~ 0.001 to 0.1 g
  - Adapts to almost all mission profiles
  - Very low propellant consumption
    - Isp 10,000 - 100,000 seconds
  - Very low contamination compared to fission systems
    - Operation in Earth's magneto-sphere may be precluded due to proton belt charging

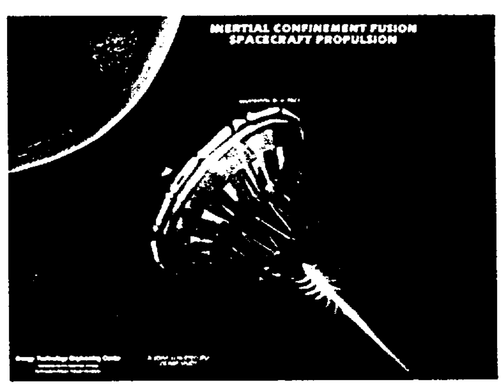


Figure 5.7-1: Characteristics of Fusion Propulsion

5.7.2 Principles of Operation

The purely fusion systems come in two flavors, magnetic confinement (Figure 5.7-2) and inertial confinement (Figure 5.7-3). Magnetic confinement tends to be steady state (effective burns will last at least a few seconds) while inertial confinement provides pulsed energy release.

- Magnetic "bottle" confines ionized plasma, heated to fusion reaction temperature
- D-T or D-He3 propellant fed into reaction zone, ionized and heated
- Plasma leaks from one end of magnetic containment, forming propulsive jet
- Hydrogen propellant used to dilute plasma to decrease temperature, decrease Isp to desired value, and increase thrust
- Thermal power extracted from waste heat, used to generate electricity to operate system

(Cylindrical Geometry)

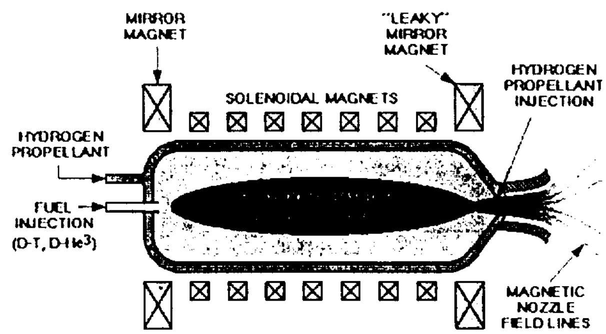


Figure 5.7-2: Principle of Operation, Magnetic Confinement Fusion

- Target pellet of fusion fuel, physically or magnetically contained, compressed to fusion temperature by multiple laser or particle beams
- Energy release per pulse ~ 0.1 T vs ~1000 T for fission pulsed plasma
- Expanding plasma reacts against magnetic nozzle, transferring momentum
- Pulse rate ~ 10 - 100/second
- Electric power extracted from pulse interaction with magnetic nozzle to operate laser or particle beam compression system

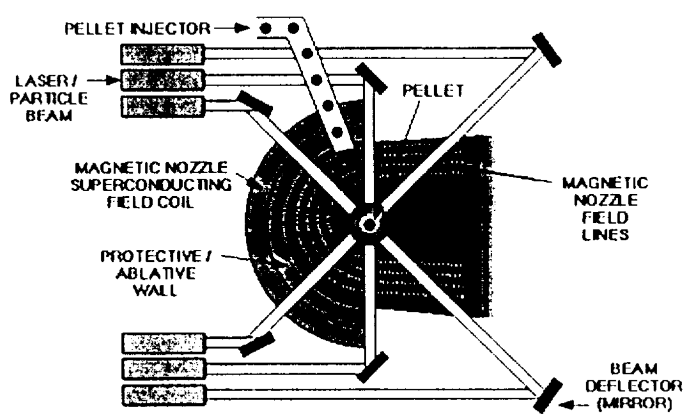


Figure 5.7-3: Principle of Operation, Inertial Confinement Fusion

Magnetic confinement is projected to have lower Q (ratio of energy released to energy input required to sustain the reaction) while higher Q is expected for inertial systems. Therefore, magnetic systems are expected to have a higher circulating power fraction, i.e. the power recirculated by the system to "keep the fire lit" compared to power in the propulsive jet. This leads to projections that inertial confinement machines will be less massive and exhibit higher thrust-to-mass ratios, although that is not necessarily true of contemporary experiments.

Anti-matter does not present confinement issues, it presents containment issues. Anti-matter reacts with normal matter on contact (the ultimate hypergolic) releasing most of the mass of the anti-matter and the (equal mass of) reacting normal matter as energy. Most of the energy can be directed (formed into a jet) by magnetic fields, but some of the particles are neutral and cannot be collimated. Because of its extreme reactivity, anti-matter containment/storage schemes must rely on magnetic or electromagnetic forces for containment, preventing contact with normal matter.

Several schemes propose using anti-matter to catalyze fission or fusion (or both) reactions in variants of the fusion inertial confinement concept. Anti-matter-induced fission, of e.g. uranium, generates about a dozen neutrons instead of the usual two, permitting much smaller critical mass. Anti-matter can catalyze fusion by means of muons (a decay product from the annihilation reaction) attaching to hydrogen isotope atoms in place of the electron (muonic atoms). The muon wave function is so close to the nucleus that nuclear fusion reactions occur at room temperature.

5.7.3 Technology and Development Challenges

For fusion, confinement sufficient to reach significant net energy release is the main issue, as indicated in Figure 5.7-4. Since there are about 15 proposed fusion propulsion concepts, a key challenge is to obtain a valid comparative evaluation in order to select a few for focused technology development. Proof-of-principle experiments are probably required in each case; some have been performed or are under way.

Space propulsion is a very different application than terrestrial power production. Most ongoing fusion research is directed towards the latter. Much effort has been spent on the Tokamak configuration. A Tokamak is a very poor candidate for space propulsion since, apparently, only very large ones will achieve adequate containment and these are massive machines. Space propulsion does not need to close the loop of fuel use and fuel production, as does terrestrial energy. Space propulsion must produce a jet, which may be a blessing or a curse, depending on the confinement configuration (it's a curse for a Tokamak). Space propulsion need not attain 10¢/kWh electricity generation. For these and other reasons space propulsion fusion research cannot rely entirely on the terrestrial energy program.

In the case of anti-matter involvement in the reactions, physics experiments are needed to accurately characterize the reactions (some have been done). Containment research for anti-matter itself is needed. Anti-proton traps have been built; these need to be improved. An actual antimatter atom (anti-hydrogen) has never been observed; CERN researchers have announced they plan to accomplish this by (2000) year-end. Production of antimatter presently operates at an efficiency about 10^{-8}. This is so bad that we can't afford the electric bill to produce useful quantities (for missions; enough can be readily made for experiments). Some have speculated that efficiencies as good as 10^{-4} might be achieved. At that efficiency, enough antimatter to power a piloted Mars vehicle would cost more or less $100 billion. If the anti-matter were used as a catalyst rather than directly, the cost might become almost reasonable.

• Successful confinement configuration/technology (magnetic or inertial); demonstrate system net energy balance	• Necessary to generate net power (thrust) from system; solution expected to be different from terrestrial powerplant fusion
• Selection of most promising options; focusing research to reach "critical mass"	• Large number of options dilutes research; should not winnow until further feasibility research
• Integration of technology elements to create practical propulsion system	• Expect to need an extensive pre-development program to produce working prototype
• Development of test facilities for high-power engine development	• Full-scale engines expected to be multi-gigawatt; test facilities probably major developments
• Resolution of possible environmental issues	• Expected to be minor but need early start

Figure 5.7-4: Challenges for Fusion Propulsion

5.7.4 Benefits

Benefits are summarized in Figure 5.7-5. Fusion is the closest of all the technologies evaluated to a universally "good" propulsion system, but it also has its limitations, mainly that it probably will not work in a robotic spacecraft size. Further, it is the least understood of the systems with probable exception of the gas-core fission rocket.

5.7.5 Representative Mission Applications

In Figure 5.7-6 we have also shown application to a Mars mission. The profile is very similar to that for the gas-core fission rocket or the fission pulse system. Environmental concerns are much smaller, but "charging" of the van Allen belts is still an issue. EMP issues are not expected since the pulses are ~ 1000 times smaller than those for fission pulse. The fusion system has the potential to achieve faster Mars trips than any other system evaluated, e.g. 30 days, and can open the window so that transfers at almost any time are possible. Those at very unfavorable times would probably be several months duration.

Anti-matter systems characteristics are poorly known. Presumably, high specific impulse at moderate thrust is possible. There are significant heat rejection issues.

- Moderate thrust and high Isp, potentially highest usable Isp of any proposed technology
- Very low radiation contamination compared to other nuclear options
- No launch safety issues associated with propellants
- Reusable system may be possible
- Feasibility criteria less stringent than terrestrial energy
  - Not required to produce net energy in global sense
  - Relatively high energy costs (re terrestrial energy) are still very economic in view of performance potential
  - Reactor lifetime short compared to terrestrial energy

Figure 5.7-5: Fusion Propulsion Benefits

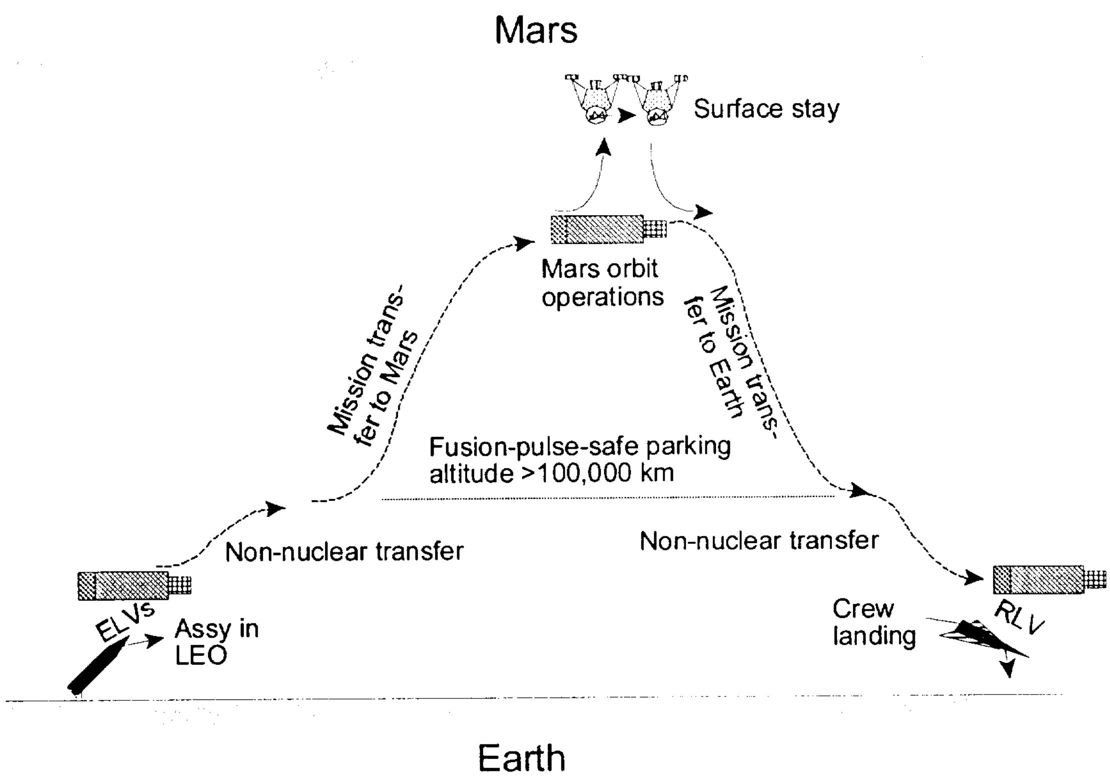


Figure 5.7-6: Mission Profile Diagram for Fusion Propulsion

5.7.6 Evolution Paths

Evolution paths are discussed in Figure 5.7-7. Of the 15 or so options mentioned in current literature, it is essential to advance 3 or more to a state of demonstration which indicates feasibility by proof-of-principle experiments. Fusion technology is difficult and it is important not to narrow prematurely. Mission application considerations are described in Figure 5.7-8.

- Experimental and analysis program which demonstrates by lab experiments ~ 3 options showing attainability of net energy and adequate performance (T/W & Isp)
- Subscale prototype program to develop/demon-strate integrated systems without breaking the bank on test facility cost
- Select system for full-scale development
- Full-scale R&D ground tests, flyable prototype
- In-space test
  - May need to be > GEO altitude to avoid charging van Allen belts

Figure 5.7-7: Evolution Paths for Fusion Propulsion

- It's too early to make discrete predictions
- Expect early working systems to have a ~ 1 kg/kWj at Isp few thousand seconds
- Mature technology promises a ~ 0.01 at Isp > 50,000 seconds

Figure 5.7-8: Mission Application Considerations, Fusion Propulsion

5.7.7 Risk Management

Figure 5.7-9 summarizes risk management considerations. The main factors are to maintain multiple paths to success until one or more is matured to a point of high confidence, to evaluate and consider factors such as development and verification test facility concepts and requirements, and to continually evaluate environmental issues in the context of development testing as well as systems operations.

5.7.8 Satisfaction of Needs

Satisfaction of needs is summarized in Figure 5.7-10. As one would expect, fusion and anti-matter are the most potent of the systems evaluated. They are also the most uncertain, and probably the farthest from realization. However, the payoff is very high, and depending on the timing for future high-performance space missions, these concepts may fit well into a program and architecture development.

- Confinement (Net energy fusion reaction)
  - Self-sustaining reaction (engineering break-even) yet to be demonstrated
    - Magnetic Q ~ 0.25; inertial ~ 1; need ~3, nice to have >10
- Develop research strategy which sustains "critical mass"; incrementally narrows
  - Enough funding to maintain momentum and cadre of expertise, facilities
  - Don't discard options which may bear fruit
  - Incrementally down-select as experimental research identifies discriminators
- Cost ... Technology advance, full scale development, unit cost, facilities and operations cost
  - Sources of estimates; uncertainty issues
- Nuclear safety
  - Much less challenging than fission systems, but maintain vigilance

Figure 5.7-9: Risk Management Considerations for Fusion

- "If it works", this in-space transportation technology could reach any destination in the solar system with human missions
  - Probably same vehicle(s) can serve relevant needs; relatively little tailoring required
  - May be best suited for large-scale high delta V missions
    - Not clear that a "small" version can work; depends on evolution of containment technology
- Develop 100 kWe-class solar electric propulsion to serve high-performance robotic missions
- If fusion works really, really well, approaches performance needed for interstellar probes (but not interstellar human missions)

Figure 5.7-10: Fusion Propulsion Satisfaction of HEDS Needs

6.0 **Conclusions**

Conclusions are summarized in Figure 6-1. We made no attempt to rank the technologies and believe it is not appropriate at this time. It is essential to recognize that selection of technologies and architectures depends on overarching program goals. As goals change, which they frequently have over the years, best-suited technologies change.

A good example is seen in the goals for humans to Mars. In the early years of NASA studies of humans to Mars, the goal of early, fast trips led to baselining of the nuclear thermal rocket for space propulsion. At times, humans to Mars goals have looked towards in-depth exploration missions which need extensive surface time and a lot of surface exploration equipment for human crews. When these goals have been paramount, less ambitious propulsion technology has been selected, such as cryogenic/aerobraking or derivatives of "Mars Direct" (a variation on cryogenic/aerobraking).

In this study, we did not evaluate cryogenic/aerobraking or other propulsion technologies ready for full-scale development. We also did not evaluate in-situ propellant since there is little factual information available.

It is important to note that a human return to the Moon is practical without any of these advanced propulsion technologies (although several of them would enhance lunar operations), but that without technology advancements for in-space propulsion, the future of human exploration of space beyond the Moon is on hold. It is often argued that conventional chemical propulsion can take humans to Mars and this may be true, but chemical propulsion for this application is clearly marginal.

Recent research reported in news media has suggested that human missions to Mars will almost certainly need artificial gravity unless transfers are of quite short duration, such as less than 90 days. This leads to architectures involving large artificial g spacecraft powered by electric propulsion, or high-Isp moderate-to-high thrust systems such as nuclear thermal rocket, gas-core, external plasma pulse, or fusion.

- There is no single solution that satisfies all needs
- Selection of solutions depends on program goals
  - For example, desire for early fast trips to Mars leads to nuclear thermal propulsion (this is the reason it was selected originally in the 1960s).
  - Large-scale exploration or settlement, e.g. of Mars, needs low cost reusable in-space transportation architecture such as high-power solar electric or nuclear fusion.
    - Both low cost to LEO and low cost in-space transportation are needed to obtain affordable costs for large-scale programs.
- Some technologies are actual competitors while others fill complementary roles or niches
  - Gas-core nuclear, high-power nuclear electric, external pulse plasma and fusion compete for high power, moderate-to-high thrust, high Isp
  - High-power solar electric and solar sails compete for low thrust, high Isp, economical operations where fast trips are not required
  - Tethers fill a unique niche for very economic transportation in the case of repetitive missions with modest delta V.
  - Low-power (< 1 megawatt) nuclear electric fills a unique niche for outer planets to Oort cloud robotics
  - Nuclear thermal rocket fills a unique niche for early fast human trips to Mars
- Technology effort should approach these technologies in a successive narrowing manner

Figure 6-1: Conclusions of the Study

7.0 Recommendations

1. Since preferred architectures are dependent on program goals (which change), on costs (which are poorly known at present) and on technological progress (hard to predict), the technology advancement strategy needs to be broad enough to encompass these variables.
2. A successive narrowing-down approach is recommended. It is important to develop and carry out proof-of-principle demonstrations and to continually evaluate risks and costs.
3. This entire area of research is vital to the future of human space flight, as well as many other advanced missions and applications. Accordingly, it should be funded at a level comparable to funding of advanced Earth-to-orbit transportation.

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Objectives of the study were:

1. To review the state-of-the-art of in-space transportation propulsion;
2. To develop internally consistent assessment and planning data re advanced technology;
3. To determine the benefits, advantages and drawbacks of comparative systems;
4. To develop roadmaps for technology advancement (to TRL 6) for each system evaluated;
5. To draw overall conclusions and make recommendations for technology advancement initiatives.

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