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ELECTRIC PROPULSION FOR MANNED MARS EXPLORATION

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ABSTRACT

Advanced high-power electric propulsion systems can significantly enhance piloted Mars missions. An increase in the science payload delivered to Mars and the reduction of the total Earth-departure mass are the major system-level benefits of electric propulsion. Other potential benefits are the return of the cargo vehicle to Earth orbit and the availability of high power in Mars orbit for high-power science and communications.

Parametric analyses for sizing the cargo mission vehicle for Mars exploration missions are presented. The nuclear-electric propulsion system thruster size, power level, mass, propellant type and payload mass capability are considered in these system-level trade studies. Descriptions of the propulsion system selection issues for both ion and MPD thruster technologies are also discussed.

On a manned Mars mission, the total launch mass for an unmanned cargo vehicle in Low Earth Orbit (LEO) can be reduced by up to 50 percent over the baseline oxygen/hydrogen propulsion system. Because the cargo vehicle is sent to Mars prior to the manned mission, the trip time for the vehicle is not a critical factor. By taking advantage of the high specific impulse (I_{sp}) of an ion or a Magneto-Plasma-Dynamic (MPD) thruster system, the total LEO mass is reduced from 590,000 kg for the oxygen/hydrogen propulsion system to 309,000 kg for the MPD system and 295,000 kg for the ion system. To provide these mass savings, the ion propulsion system must operate at a 6000-lb_f-s/lb_m I_{sp} and a power level of 4 MW. The MPD I_{sp} must be 5000 lb_f-s/lb_m and must have the same 4-MW power level.

Many factors must be analyzed in the design of a electric propulsion Mars cargo vehicle. The propellant selection, the number of thrusters, the power level and the specific impulse are among the most important of the parameters. To fully address the electric propulsion system design, trade studies for the differing ion and MPD propulsion system configurations (thruster power levels, number of thrusters, propellants and power systems) must be conducted.

NOMENCLATURE

Ar	Argon	NASA	National Aeronautics and Space Administration
C_3	Injection Energy (km^2/s^2)	NH_3	Ammonia
H_2	Hydrogen	O_2/H_2	Oxygen/Hydrogen
I_{sp}	Specific Impulse	PPU	Power Processing Unit
Kr	Krypton	TVS/VCS	Thermodynamic Vent System/Vapor Cooled Shield
LEO	Low Earth Orbit	Xe	Xenon
LMO	Low Mars Orbit	Greek Symbols	
MPD	Magneto-Plasma-Dynamic	ΔV	Velocity Change (km/s)

INTRODUCTION

In this paper, a series of sensitivity studies for an electric propulsion Mars cargo vehicle are presented. The cargo vehicle uses a megawatt-class space nuclear power reactor and a high-power ion or MPD electric propulsion system. Both the trip time and the initial mass in LEO are used as the primary figures of merit for the cargo vehicle.

These types of sensitivity analyses can allow the selection of the "best" propulsion operating points: the best I_{sp}, power level and thruster technology. The best operating point may be the minimum mass for the cargo vehicle in LEO. It may also be the power level where the LEO mass and the trip time have been selected to fulfill other specific mission requirements (deliver added payload, etc.). Because

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electric propulsion can deliver a wide range of I_{sp}, each different mission's system performance level can be tailored to that mission. Selecting the best or optimum operating points will allow a significant mass savings during the life cycle of the propulsion systems in the Mars transportation system.

PILOTED MARS EXPLORATION

The exploration of Mars has been considered since the early 1950's (Ref. 1). In a recent series of studies, the NASA Office of Exploration has proposed several new mission scenarios for the exploration of Mars. In each mission, the departure dates for the vehicles are between the year 2000 to 2020. Two type of missions are being contemplated. The first includes the exploration of Phobos, the innermost moon of Mars. No astronauts would land on Mars itself. Instead, unmanned rovers on the Martian surface would be remotely operated from the orbital spacecraft. The second mission type would be similar to the Phobos mission but it would send astronauts to Mars' surface.

For the sprint-class missions described in this paper, the Mars mission is accomplished using two vehicles. One vehicle delivers cargo (called the cargo vehicle) and one sends the crew to Mars (called the piloted vehicle). Because of the potential adverse effects of long-term zero gravity and solar radiation, it is considered important to reduce the total piloted vehicle trip time to between one and two years. Because the mission is piloted, a large payload for Mars landers and life support systems is needed. A fast transfer to Mars and back with large 70,000- to 250,000-kg payloads is very demanding for state-of-the-art propulsion systems.

By "splitting" the payload between the cargo vehicle and piloted vehicle, the total mass in LEO for the complete mission is significantly reduced. The larger payload mass is placed on the cargo vehicle and sent to Mars orbit well before (up to one year before) the piloted vehicle. The cargo mission trajectory is a low-energy trajectory. The required C_3 is 8 to 16 km^2/s^2. It requires much less propellant than a mission using a fast Mars transfer. A fast transfer can require a C_3 of over 100 km^2/s^2 (Refs. 2 and 3). In the mission scenario discussed here, the payload for the cargo vehicle is 180,000 kg and a 71,500-kg payload is used for the piloted vehicle. The cargo vehicle C_3 at Earth departure is 8.8 to 15.5 km^2/s^2 and the piloted vehicle C_3 is 105 km^2/s^2. Large masses are LEO is required for the mission. For a 2003 mission, a minimum of 1,330,000 kg in LEO is needed (Ref. 2). Transferring large payloads to Mars orbit and returning them to Earth will require significant propulsion systems. Figures 1 and 2 depict the Mars vehicles' masses. The piloted vehicle mass in LEO is 740,000 kg. The cargo vehicle mass is 590,000 kg. The vehicles' propulsion systems used in the most current studies are based on the oxygen/hydrogen (O_2/H_2) engine technology with an I_{sp} of 480 $lb_f\text{-}s/lb_m$.

WHY ADVANCED ELECTRIC PROPULSION?

Advanced electric propulsion has several potential advantages over chemical propulsion systems. Advanced propulsion can reduce the vehicle mass, increase the delivered payload, allow the return of the cargo vehicle to Earth orbit and provide high power levels for active science experiments and communications.

The potentially most-significant advantage of advanced electric propulsion for the Mars cargo vehicle is the reduction of its initial mass in LEO. Figure 1 contrasts the mass of the chemical propulsion option with that of a nuclear-electric cargo vehicle. The power level of the electric propulsion vehicle is 4 MW. The specific impulse of the chemical propulsion system is 480 $lb_f\text{-}s/lb_m$, while the electric MPD vehicle has a 5000-$lb_f\text{-}s/lb_m$ I_{sp}. The LEO mass of the MPD electric propulsion vehicle is 309,000 kg. This is a 48-percent savings over the baseline system. Using ion propulsion at 6000 $lb_f\text{-}s/lb_m$, the mass is reduced to 295,000 kg, a 50-percent mass reduction.

Instead of reducing the LEO mass, an increased payload can be delivered to Mars. If the initial mass in LEO is a fixed mass of 590,000 kg, the electric propulsion system can deliver 385,000 kg or 114-percent more payload than the 180,000-kg chemical propulsion payload. This example uses a 5000-$lb_f\text{-}s/lb_m$ MPD I_{sp} and a 4-MW power level. The trip time for this case is 1220 days.

By adding additional propellant to the cargo vehicle, it can be returned to LEO for reuse. To return it to Earth orbit, 40,000 kg of propellant would have to be added to the cargo vehicle. Again, this example uses a 5000-$lb_f\text{-}s/lb_m$ MPD I_{sp} and a 4-MW power level. For the ion vehicle at the same power level and I_{sp}, 47,000 kg of propellant would have to be added. Successive missions would then not have to lift an additional cargo vehicle to LEO. This may reduce the total program cost by reducing the total number of launches for the Mars initiatives.

Added power for active science and communications once the vehicle has entered orbit can also provide significant benefits. Once the electric cargo vehicle has reached Mars orbit, the reactor would still be able to deliver 4 MW for other systems. These systems include high-power telecommunications for the crew. A complement of very-high-power science instruments can also be used in orbit to identify potential landing sites, locate subsurface water or perform atmospheric studies.

There are two advanced electric propulsion systems for the Mars missions that can provide the most benefit. For the Mars mission, the technologies of ion and MPD propulsion are the only ones which can

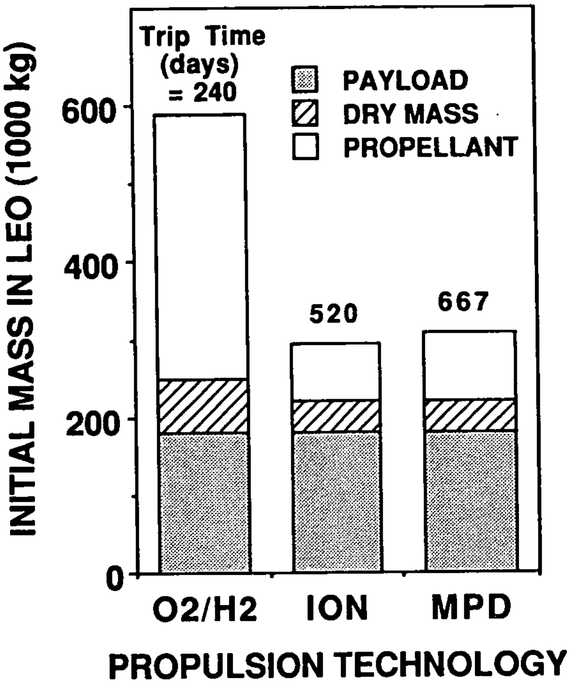

Figure 1. Cargo Vehicle Mass Comparison:
O_2/H_2, Ion and MPD
4-MW Electric Propulsion Power Level

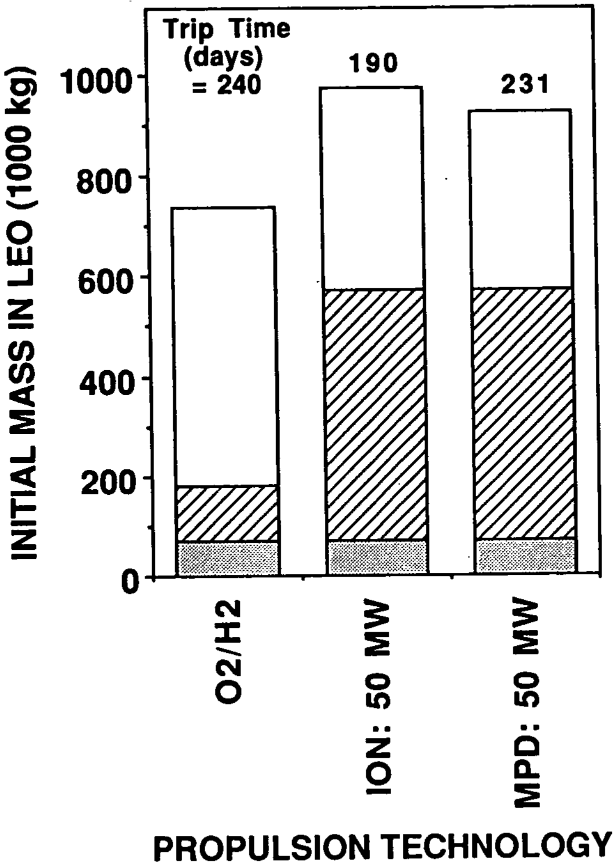

Figure 2. Piloted Vehicle Mass Comparison:
O_2/H_2, Ion and MPD
50-MW Electric Propulsion Power Level

provide significant mass savings over the baseline chemical propulsion system. The rationale for their selection will be discussed in this paper.

MISSION DESIGN AND ANALYSIS

Mission Description and Payloads: In the cargo vehicle mission, a 180,000-kg payload is delivered to Low Mars Orbit (LMO) from LEO. The initial LEO has a 500-km altitude and the LMO has a 1000-km altitude. No payload is returned to Earth and the cargo vehicle remains in LMO. The payload of the cargo vehicle is composed of a significant science payload and the propellant for the Earth return portion of the piloted vehicle mission. A 82,500-kg O_2/H_2 propellant load is transferred to the piloted vehicle after it arrives in Mars orbit. The mission scenario and the payload masses for the cargo vehicle were derived from Ref. 2.

Chemical Propulsion Mission Analysis: The ΔVs for the high-thrust transfer are 3.800 km/s for the Earth departure and 250 m/s for the aerobraking circularization firing into Mars' orbit. The Earth departure corresponds to a injection energy (C_3) of 15.5 km^2/s^2. The injection ΔV data were derived from Ref. 4. After the Earth departure stage has places the payload onto the Mars trajectory, it returns to LEO using aerobraking. The additional ΔV required for this is 1000 m/s. Because the Mars vehicle is on an interplanetary trajectory when it separates from the Mars payload, a 750 m/s ΔV is needed to return the transfer vehicle to the gravitational influence of the Earth. The remaining 250 m/s comprises 50 m/s for small trajectory correction maneuvers of the stage and 200 m/s for the circularization firing after the aerobraking maneuver. The one-way trip time for the chemical propulsion cargo vehicle is 240 days.

Electric Propulsion Mission Analysis: The baseline cargo mission between Earth and Mars is a one-way transfer between a 500-km LEO and a 1000-km LMO. The ΔV is computed using the VARITOP program (Ref. 5). Table I provides the mass ratios for the cargo missions with a 5000-lb_f-s/lb_m I_{sp}. The launch date for the electric propulsion cargo vehicle is July 18, 2005. The trip times for the cargo vehicle are dependent upon the power level and the thruster technology used. In the succeeding sections of this paper, the variation of the cargo vehicle trip time with its I_{sp} and power level is detailed.

Table I
Electric Propulsion Performance Data
I_{sp} = 5000 lb_f-s/lb_m
Launch Date: July 18, 2005
Earth to Mars Transfer

MO/PJ	MF/PJ	TES	TEND	TCS	TFL	TP
50.	32.30	83.1	180.9	21.4	285.4	246.3
60.	39.66	100.6	197.8	26.8	325.2	283.0
70.	47.08	118.3	213.2	32.3	363.8	318.9
80.	54.53	136.0	227.6	37.9	401.4	354.4
90.	61.99	153.8	241.0	43.5	438.3	389.7
100.	69.47	171.7	253.8	49.1	474.6	424.8
110.	76.94	189.6	266.1	54.8	510.5	460.0
120.	84.40	207.5	278.0	60.5	546.0	495.3
130.	91.86	225.5	289.6	66.2	581.3	530.6
140.	99.31	243.6	301.0	72.0	616.5	566.1
150.	106.75	261.6	312.2	77.7	651.6	601.8
160.	114.17	279.7	323.5	83.5	686.7	637.7
170.	121.58	297.9	334.8	89.3	721.9	673.7
180.	128.98	316.0	346.3	95.0	757.3	709.8
190.	136.37	334.2	358.0	100.8	793.0	746.2
200.	143.74	352.4	370.1	106.6	829.1	782.7
210.	151.10	370.7	382.7	112.4	865.7	819.5
220.	158.45	388.9	396.0	118.1	903.0	856.4
230.	165.77	407.2	410.0	123.9	941.1	893.7
240.	173.08	425.5	424.9	129.7	980.1	931.2
250.	180.36	443.8	441.0	135.4	1020.2	969.0
260.	187.61	462.1	458.4	141.2	1061.7	1007.2
270.	194.83	480.5	477.2	146.9	1104.5	1045.9
280.	202.01	498.8	497.4	152.6	1148.8	1085.2
290.	209.14	517.2	518.9	158.3	1194.4	1125.0
300.	216.23	535.6	541.5	163.9	1241.0	1165.6

Definitions:

MO/PJ	Initial Mass/Jet Power (kg/kW)
MF/PJ	Final Mass/Jet Power (kg/kW)
TES	Escape spiral time, days
TEND	Heliocentric transfer time, days
TCS	Capture spiral time, days
TFL	Total transfer time, days
TP	Total propulsion time, days

Because of the long transfer times required for the relatively low-power electric cargo vehicle, an electric propulsion system is not proposed for the piloted vehicle of the Mars Sprint mission. Fast piloted missions using electric propulsion are possible but the initial mass in LEO would be substantially higher than the chemical propulsion baseline mission. This is because the power level required for the fast piloted mission is very high. A high power level requires a large mass for the nuclear reactor and the propulsion system.

Figure 2 compares the mass of the piloted Mars vehicle using a nuclear-electric propulsion. A 925,000-kg mass is needed in LEO using an MPD propulsion system while the ion system has a 973,000-kg LEO mass. This is in comparison to the 740,000-kg mass for the O_2/H_2 propulsion option. To attain the same flight time as the chemical propulsion vehicle, a 50- MW power level is needed. This vehicle uses an I_{sp} of 5000 lb_f-s/lb_m. As with the O_2/H_2 propulsion piloted vehicle, the electric propulsion-based mission would have its electric propulsion propellant delivered to Mars by the cargo vehicle. Once the propellant transfer is complete, the piloted vehicle would return to Earth. Because the nuclear-electric vehicle mass is greater than the chemical propulsion baseline, it is not a likely candidate for the piloted Mars vehicle.

Because the 50-MW piloted vehicle uses considerably more propellant than the chemical propulsion piloted vehicle, the cargo vehicle for this mission is also significantly more massive. In the chemical propulsion case, the return propellant mass is 82,500 kg. Using electric propulsion, the 50-MW piloted vehicle requires up to 400,000 kg of propellant for a fast return. Again, based on the initial mass in LEO, electric propulsion is not attractive for fast piloted sprint missions to Mars.

PROPULSION SYSTEM DESCRIPTIONS

O_2/H_2 PROPULSION

The state-of-the-art propulsion system baselined for the piloted Mars missions is O_2/H_2 propulsion (Ref. 6). The engine technology is currently under development under the direction of the NASA-Lewis Research Center (Refs. 7 and 8). This system also uses long-term cryogenic propellant storage with O_2 at a temperatare of 90 K and H_2 at a 20-K temperature. Long-term storage is required to maintain the O_2 and H_2 as liquid cryogens during the transfer to Mars and during the storage period before the arrival of the piloted Mars vehicle.

The O_2/H_2 Mars mission vehicles have a 480-lb_f-s/lb_m I_{sp}. The Mars mission thrust level is 400,000 to 500,000 lb_f. The thrust level selection is based on minimizing the gravity losses incurred during the Earth departure firing. A thrust to weight ratio of the Mars transfer vehicle is 0.3. This thrust level reduces the total gravity loss at Earth departure to less than 100 m/s. For the O_2/H_2 Mars vehicles, the dry masses from Ref. 2 are used.

Table II
Propulsion System Technology

Propulsion Technology	I_{sp} (lb_f-s/lb_m)	System Efficiency
O_2/H_2	480	n/a*
Arcjet	1500	49
Ion	2000-10000	60-85
MPD	2000-10000	50

* not applicable

NUCLEAR-ELECTRIC PROPULSION

The I_{sp}s of the chemical and electric propulsion systems are shown in Table II. A range of I_{sp} for the ion, MPD and arcjet propulsion systems is provided in the table (Refs. 9, 10 and 11). Because these thruster technologies allow a wide range of I_{sp}, they can be designed to provide the "best" I_{sp} for widely-varying mission requirements (lowest LEO initial mass, etc.). The propellants that are used for the electric transfer vehicles are hydrogen (H_2) for the arcjet, xenon (Xe) for the ion and ammonia (NH_3) or argon (Ar) for the MPD propulsion systems.

In both ion and MPD propulsion systems, the "best" I_{sp} is 5000 to 6000 lb_f-s/lb_m. This I_{sp} selection will be discussed later in the paper. For this I_{sp} range, the total system efficiency for the ion propulsion system is 70 percent. In this range of I_{sp} for the MPD systems, the system efficiency is 50 percent. These efficiencies for MPD and the ion propulsion include the thruster efficiency and the efficiency of the propulsion power processors.

The masses of the power and propulsion system (the entire transfer vehicle) are described by an overall system specific mass. This specific mass is the ratio of the total vehicle dry mass to the electrical power level of the transfer vehicle. The range of specific masses is 0 to 30 kg/kW.

Based on the design studies of high-power electric propulsion (Refs. 12 and 13) and the current studies of advanced space nuclear power (Refs. 14 and 15), the best estimate of the system mass for a nuclear-electric propulsion vehicle is 10 to 15 kg/kW. This mass would include 5 kg/kW for the electric propulsion system: thrusters, propellant tankage and feed system and the power processing units for the thrusters. Table III provides a mass summary for an ion propulsion system without tankage. At an I_{sp} of 6000 lb_f-s/lb_m, the total system mass for the thrusters and power processors is 10,108 kg (or 2.5 kg/kW at a 4-MW power level).

An added mass of 10,638 kg would be required to contain the 100,000-kg propellant load for the electric propulsion cargo vehicles. Included in this mass are the propellant tank, the residual propellant, the thermal control system and feed system. This tankage stores xenon as a liquid cryogen at a temperature of 165 K. The tankage is aluminum with a 30-psia storage pressure. The tank wall is sized to a maximum operating pressure of 150 psia. This allows the tank to withstand the high hydrostatic pressure in the tank during the launch from Earth. Because of the high density of the xenon propellant, the hydrostatic pressure creates a significant pressure at the bottom of the tank during the launch from Earth. The tank is designed to accommodate the higher maximum operating pressure. A coupled Thermodynamic Vent System/Vapor-Cooled Shield (TVS/VCS) surrounds the tank and intercepts heat leaks into it. The

Table III
Ion Propulsion System Mass Summary

INPUT PARAMETERS:		
Specific Impulse (lb_f-s/lb_m)	=	6000
Input Power (kW)	=	4000
Engine Diameter (cm)	=	100
RESULTS:		
Number of Engines*	=	24
Number of Engines (no redundancy)	=	19
Propulsion System Mass (kg)**	=	10108.0
Propulsion System Specific Mass (kg/kW)	=	2.53
Propulsion System Efficiency	=	0.73
PPU Efficiency	=	0.90
Engine Efficiency	=	0.81
Engine Mass (kg)	=	20.0
Gimbal Mass (kg)	=	6.8
PPU Mass (kg)	=	220.0
PPU Specific Mass (kg/kW)	=	1.0
Thermal Control Mass per PPU (kg)	=	84.2
Interface Module Mass (kg)	=	646.0
Thrust Module Structure Mass per Engine (kg)	=	8.3
Engine Input Power (kW)	=	214.0
Beam Current (A)	=	70.5
Beam Voltage (V)	=	2890.0
Engine Thrust (N)	=	5.95
Discharge Current (A)	=	350.0
Total Voltage (V)	=	3210.0
Voltage Ratio	=	0.9

* The number of engines has been increased 25-percent for redundancy.
**Includes a 15-percent mass contingency

additional system masses include auxiliary power, structure, command and data, communications and the other remaining subsystems for the cargo vehicle.

The remaining 5 to 10 kg/kW of the specific mass includes the reactor power system. This specific mass is significantly lower than the 30 kg/kW (at a 100 kW power level) that is possible with an SP-100-class reactor. To have a reactor with this low a specific mass, it must be one which operates at a power level higher than one megawatt. It must also use a dynamic conversion cycle. Both the Brayton Cycle and the Stirling Cycle are possible candidates (Refs. 14 and 15).

A 0-kg/kW specific mass is regarded only as the limiting case for this analysis. It is not presented as a suggested option for a Mars mission. It does, however, represent the case where the entire propulsion system mass is considered as part of the payload mass. This option is not viable in some cases. One such case is when the estimated mass of the electric propulsion system would be greater than the payload mass.

PROPULSION SENSITIVITY ANALYSIS

To determine the best operating points for the propulsion system, a series of LEO initial mass and trip time sensitivity studies were conducted. The initial mass in LEO is the mass of the vehicle which departs Earth at the beginning of its flight. This includes the payload, the interplanetary cargo transfer vehicle and its propellant. The cargo vehicle includes the power, propulsion and other vehicle subsystems to deliver the payload and itself to Mars. The cargo mission trip time is defined as the trip time from a 500-km altitude LEO to a 1000-km LMO. This trip time includes the LEO escape time, the heliocentric transfer time and the LMO capture spiral time.

The electric propulsion cargo vehicle in Figure 1 uses a 4-MW power level with an overall power-and-propulsion specific mass of 10 kg/kW. These ion and MPD system designs were selected based on the results of the sensitivity analyses that are described below.

The selection of the power level and the I_{sp} for the cargo vehicle is conducted parametrically to find the "best" operating points. In this case, the best I_{sp} is that which produces the lowest initial mass in LEO. The trip time delivered by this "best" I_{sp} should also be within the design limitation of the Mars cargo mission. As currently described (Ref. 6), the cargo mission has a relaxed trip time constraint (no fixed trip time as with the piloted mission). A "best" trip time of less than two years was selected. These preliminary studies of the mass and trip time of the cargo vehicle show the range of operating design points which will either fulfill the mission requirements or provide insights into how the mission requirements might be changed to take advantage of some unique system-level benefit of an electric propulsion technology.

For the cargo vehicle analyses, the results presented here are for the specific mission design and payload requirements of the Mars cargo vehicle with a 180,000-kg payload. If the mission requirements for the cargo vehicle change, additional analysis is required to identify the best I_{sp} and propulsion system design.

ELECTRIC PROPULSION CARGO VEHICLE: DESIGN SENSITIVITY EXAMPLE

All of the design parameters of the electric propulsion cargo vehicle are very highly interrelated. Any changes in the mass of the power system or the propulsion system can have significant effects on the vehicle trip time and initial mass. In this study, the cargo vehicle initial mass in LEO and the trip time are determined as a function of the power-and-propulsion specific mass, the I_{sp} and the propulsion system's power level.

Specific Impulse

In Figure 3, the effect of I_{sp} on the cargo vehicle mass is shown. In the sensitivity example, an MPD propulsion system is used. The specific mass of the power-and-propulsion system is 10 kg/kW. The power level is 4 MW. For an I_{sp} below 2000 lb_f-s/lb_m, the mass of the MPD cargo vehicle is comparable to the chemical propulsion option. To provide the greatest mass savings the I_{sp} for the vehicle should be 5000 lb_f-s/lb_m or higher.

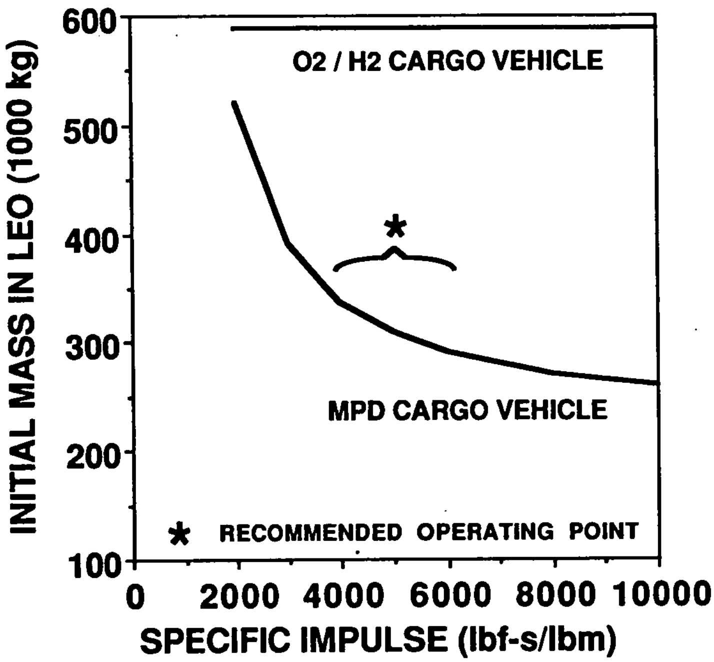

Figure 3. MPD Initial Mass in LEO vs. I_{sp}
4-MW Electric Propulsion Power Level

It is clear from this analysis that an electric propulsion system with an I_{sp} below 2000 lb_f-s/lb_m has no mass advantage over chemical propulsion for this Mars cargo mission. Thus, arcjet propulsion systems, such as a H_2 arcjet with a 1500-lb_f-s/lb_m I_{sp} or a NH_3 arcjet with an I_{sp} of 1000 lb_f-s/lb_m, are not contenders for this mission.

Trip Time

The other important parameter is the cargo mission trip time. In Figure 4, the trip time variation with I_{sp} is given. The system uses a 4-MW power level. An MPD system with a 10-kg/kW power-and-propulsion specific mass is used. At an I_{sp} that is greater than 9000 lb_f-s/lb_m, the one-way trip time becomes very long: greater than 1000 days. To minimize the trip time, a 5000-lb_f-s/lb_m I_{sp} is recommended.

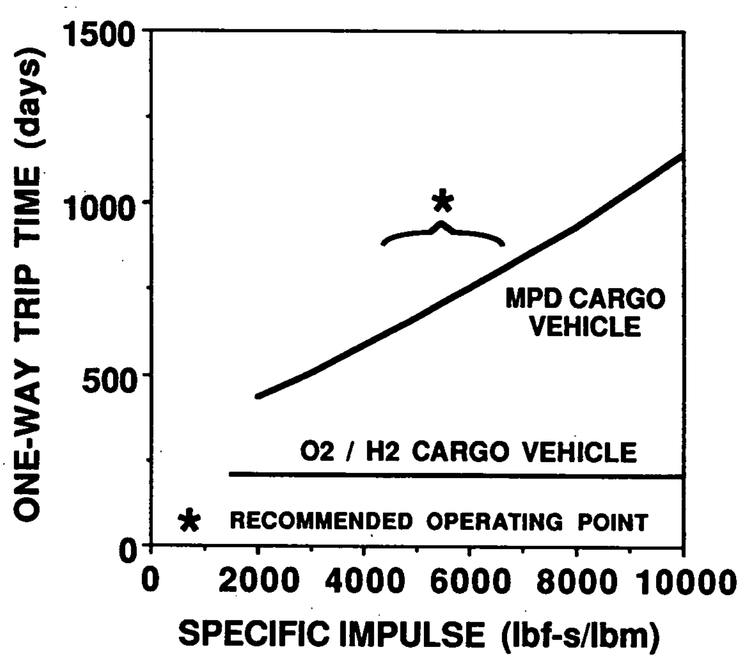

Figure 4. MPD Trip Time vs. I_{sp}
4-MW Electric Propulsion Power Level

Using ion propulsion, the trip time delivered for the same power level, I_{sp} and specific mass is 520 days. The Xe-Ion propulsion system electrical efficiency is 70 percent. This higher efficiency allows a significantly shorter trip time than the MPD cargo vehicle. The LEO mass of this system is 314,000 kg. An ion system using a 6000-lb_f-s/lb_m and a 4-MW power level reduces the mass to 295,000 kg and the trip time only increases to 580 days.

Power Level

The power level selection is also based on the mass savings over the chemical propulsion baseline system mass. In Figure 5, the power level of the cargo vehicle is shown from 1 to 30 MW. Again, as with the

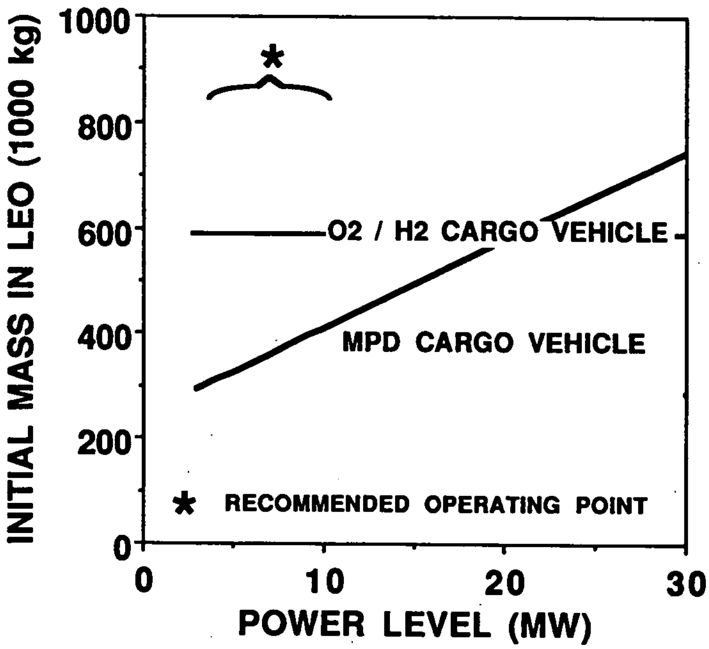

Figure 5. MPD Initial Mass in LEO vs. Power Level
5000-lb_f-s/lb_m Electric Propulsion I_{sp}

previous example, an MPD propulsion system is used. The specific mass of the power-and-propulsion system is 10 kg/kW. If the power level is above 10 MW, the cargo vehicle mass is not reduced significantly over the mass of the O_2/H_2 system. At a power level of 20 MW, both the electric propulsion system and the chemical propulsion system have a similar mass. Operating the electric cargo vehicle at a power level of 4 to 10 MW is recommended. A 4-MW power level can provide the greatest mass savings with a trip time of 667 days.

Based on the trip time and LEO mass analyses, the recommended I_{sp} is 5000 to 6000 lb_f-s/lb_m. The power level of the cargo vehicle is between 4 and 10 MW, with 4 MW giving the greatest mass savings over O_2/H_2 propulsion. These recommendations are applicable to both the ion and MPD systems. All of the analyses presented here use a 10-kg/kW power-and- propulsion specific mass. If the specific mass is higher than this value, the initial mass in LEO and the trip time for the same I_{sp} and power level would increase. This is because of the higher power system mass. A more massive power system requires more propellant for the cargo vehicle transfer.

Vehicle Specific Mass

One of the most-important influences on the cargo vehicle performance is the power-and-propulsion system specific mass. Figure 6 depicts the payload mass capability of the cargo vehicle for a wide range of specific masses. In this analysis, the initial mass of the cargo vehicle in LEO is fixed at 300,000 kg. This mass constraint was chosen because it is a significant reduction over the O_2/H_2 baseline LEO mass. The required mission payload is 180,000 kg. For a specific mass of 10 kg/kW, the I_{sp} of the cargo vehicle must be between 5000 and 6000 lb_f-s/lb_m if it is to deliver the required payload.

For future missions, the power-and-propulsion specific mass could be significantly higher than the 10 to 15 kg/kW specific mass predicted for the cargo vehicle using advanced megawatt-class nuclear reactors and ion or MPD propulsion system. In using the higher mass reactor, a small propellant load must be used to maintain the same mass savings. A judicious increase in the propulsion system I_{sp} can provide the same total initial mass in LEO. It is clear from Figure 6 that for a specific mass of greater than 30 kg/kW, the I_{sp} of the cargo vehicle must be above 10,000 lb_f-s/lb_m to deliver the needed payload with the same mass savings as the 10-kg/kW specific-mass case. With trade studies similar to those discussed in this paper, a propulsion designer can identify the "best" I_{sp} and power level for this more-massive power and propulsion systems.

Figure 7 compares the trip time and the LEO mass for an MPD propulsion system (5000 lb_f-s/lb_m) with a 10- and a 30-kg/kW power-and-propulsion system specific mass. The trip time is increased from 667 days to 860 days and the mass increase is from 309,000 to 417,000 kg. Using ion propulsion at 5000 lb_f-s/lb_m, the LEO mass increases from 314,000 kg to 421,000 kg and the trip time is increased from 520 days to

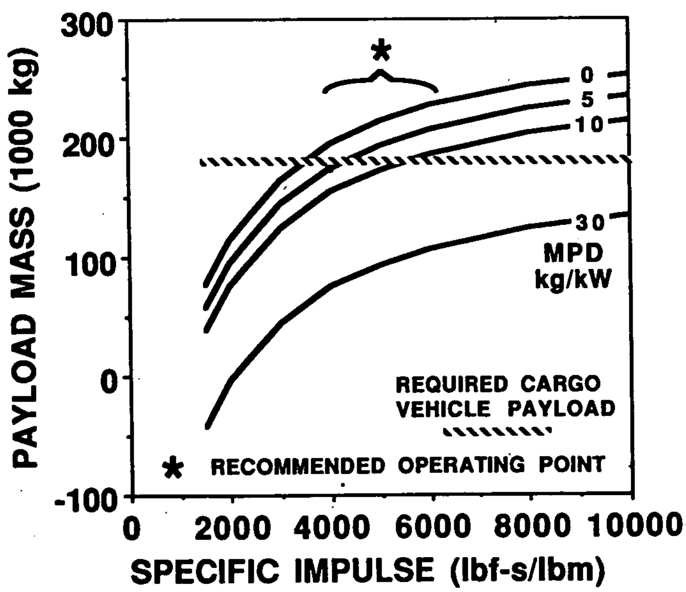

Figure 6. MPD Payload Mass vs. I_{sp}
4-MW Electric Propulsion Power Level

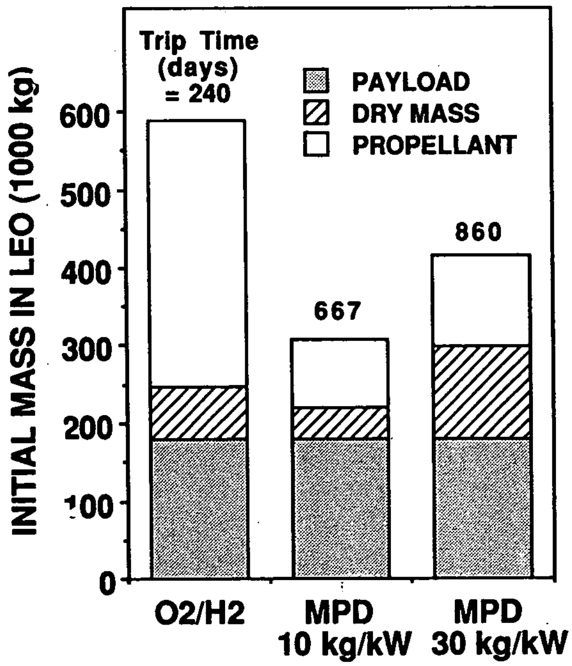

Figure 7. Initial Mass in LEO: O_2/H_2 and MPD, 10 and 30 kg/kW Power-and-Propulsion Specific Mass,
4-MW Electric Propulsion Power Level

653 days. As was described in a previous section, in order to deliver the same mass savings as the 10-kg/kW system, the I_{sp} of the MPD system would have to be greater than 10,000 lb_f-s/lb_m.

ADDITIONAL ISSUES FOR SYSTEM DESIGN

Propulsion Technology Assessment: Ion propulsion and MPD propulsion both have advantages. Ion propulsion with its higher efficiency can reduce the trip time over MPD propulsion. The MPD however is potentially a simpler device than the ion thruster. The MPD may therefore be a more reliable propulsion system than the ion system. Both the propulsion trip time and reliability are important issues in the future selection of long-lifetime propulsion systems.

The current state of the art with MPD and ion thrusters is presented in Figure 8. The required operating power levels and I_{sp} for the Mars mission is also shown. With current ion and MPD systems, there is not sufficient power-handling capability per thruster to make either propulsion system practical for the Mars mission. Because the thrusters cannot process large amount of power, each propulsion system requires many thrusters to process the total power level for the cargo vehicle. If the current ion thruster at 20 kW per thruster is selected, a 4-MW cargo vehicle needs at least 200 thrusters. Integrating so many thruster on a single vehicle results in a prohibitively high propulsion system mass and an extremely complex overall system. By increasing the power level per thruster, the complexity and the mass of the propulsion system is reduced.

The limitations with ion thrusters for power handling capability may be 500 kW per thruster. MPD thrusters can potentially process many MW of power per thruster. Because of the high power levels required for the Mars cargo mission, many ion and MPD thrusters will be needed. Figure 9 contrasts the number of thrusters for a range of different power levels. In this analysis, the ion thruster lifetime is 15,000 hr and the power level for each thruster is 214 kW. Their I_{sp} is 6000 lb_f-s/lb_m. For each MPD thruster, the lifetime is 2000 hr and each MPD processes one-half of the total power level. A 25-percent contingency on the number of thrusters is also added. For example, for a 4-MW power level, each MPD processes 2 MW. Because the MPD firing time is 617 days during the 667- day flight time, each MPD cargo vehicle must have a total of 19 thrusters to fulfill the total mission propulsion lifetime requirement. The ion system requires 24 thrusters.

As the power level increases, the number of MPD thrusters is significantly lower than that needed for ion propulsion. At a power level above 6 MW, the number of thrusters for the ion propulsion system is over 36, while the MPD system only requires 15. Thus, for very high power systems, the MPD propulsion system may have an advantage of greater simplicity with fewer thrusters over ion propulsion.

There is an unusual structure to the ion propulsion curve at power levels below 4 MW. This behavior is caused by the very long firing time required for the ion system. At power levels below 4 MW, the total ion propulsion firing time is greater than the 15,000-hr ion thruster lifetime. If the propulsion system must fire longer than 15,000 hr, additional sets of ion thrusters must be provided. For example, at a

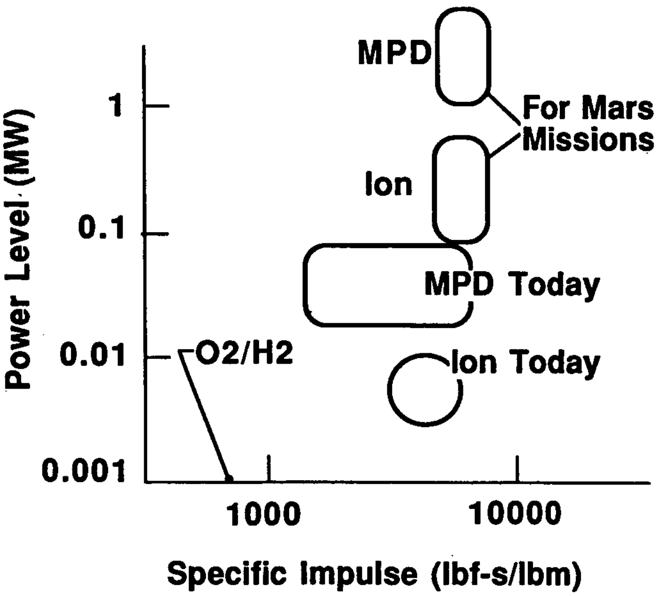

Figure 8. Ion and MPD Engine Performance Needed for Mars Missions

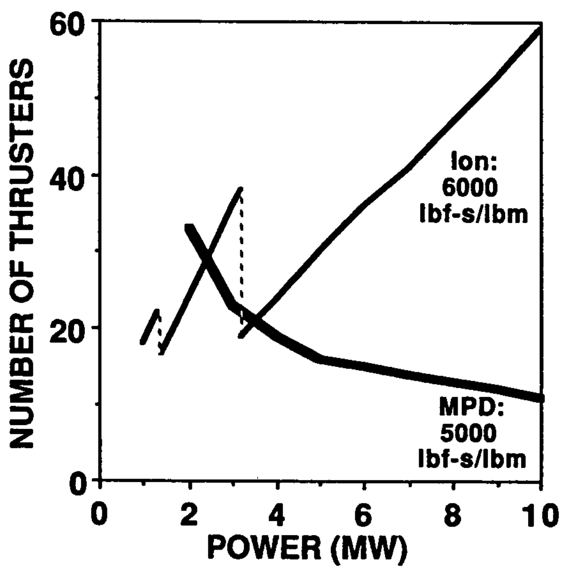

Figure 9. Number of Thrusters vs. Power Level

2-MW power level, the total ion propulsion firing time is 21,700 hr. With this long lifetime, a total of 24 ion thruster are needed. If the total firing time were only 15,000 hr, the number of thrusters needed to process the 2-MW power level would only be 12. Thus the longer lifetime would double the number of thrusters. At a 1-MW power level, the lifetime required of the propulsion system would triple the number of thrusters required from 6 to 18. This increased thruster life at lower power levels is an important issue to consider in the propulsion system design. This issue suggests that the power level for the cargo vehicle should be 4 to 5 MW.

Propellant Selection: Another important aspect of an electric propulsion cargo vehicle is the propellant selection. The cost, availability and the thruster erosion and electrical efficiency and their effects on the mission must be analyzed. While this discussion primarily considers ion propulsion, similar decisions must be made for MPD propulsion once more experimental data on MPD erosion and thruster efficiency are available.

Propellant Costs: Table IV lists the costs from Refs. 16 and 17. The cost of each propellant is based on the largest quantity purchase per cylinder from Refs. 16 and 17. Xenon is the most-expensive propellant. The krypton costs are 23 percent of the Xenon costs ($1.75/liter or $510.7/kg). Argon propellants are only 0.4 percent of the xenon cost ($0.03/liter or $18.3/kg). Based on cost alone, argon is the best ion propulsion propellant.

Table IV
Electric Propulsion Propellant Costs

Propellant	Molecular Weight	$/liter (1985$)	Electrical Efficiency (Thruster Only)	Boiling Point (K) (14.7 psia)
Ion:				
Helium	4.00	0.04	n/a*	4.22
Neon	20.18	0.15	n/a	27.05
Argon	39.95	0.03	0.50-0.60	87.45
Krypton	83.80	1.75	0.70-0.72	119.75
Xenon	131.30	7.60	0.72-0.75	165.05
MPD:				
Ammonia	17.03	0.60/kg	0.60**	239.80
Argon	39.95	0.03	0.60**	87.45

* not available
** projected efficiency

Propulsion System Electrical Efficiency: In Ref. 18, the thruster electrical efficiencies for several different ion thruster propellants were determined. Figure 10 shows the thruster efficiency for several ion propulsion propellants. Argon is not the best performer: only 50- to 60- percent thruster efficiency. Xenon and krypton have a higher thruster efficiency range: 67 to 75 percent. Based on thruster efficiency, krypton and xenon are excellent choices for ion propulsion.

Synthesis: Which Propellant Should Be Chosen? If the issues of propellant cost and propellant effects on thruster efficiency are considered, these two parameters lead the analysts to conclude different propellants are best for different reasons. There is not a clear choice between propellants for electric propulsion systems. Because there is no clear choice, an analyst must review several criteria in any choice of an electric propulsion system propellant:

1) How does the propellant choice affect the propulsion system mass and the total spacecraft mass and complexity? A refrigerator may be required for cryogenic storage or a high-pressure high-mass propellant tank may be needed for a propellant stored at supercritical pressures and temperatures.

2) What I_{sp} is needed for the mission?

3) Is a high-efficiency thruster required (to deliver a short trip time)? The trip time is a function of the power level and the spacecraft mass.

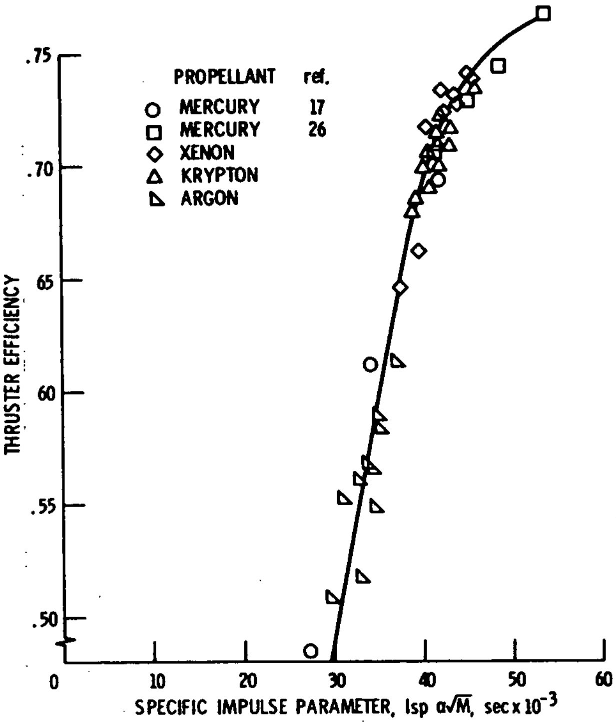

Figure 10. Thruster Efficiency Comparison

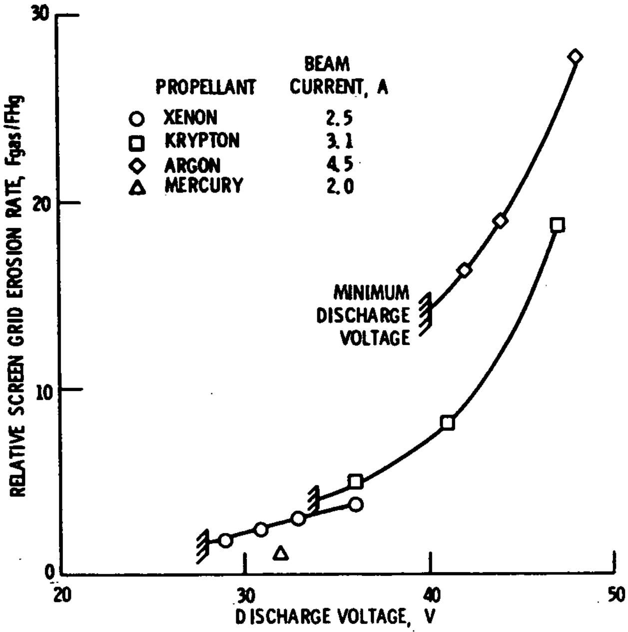

Figure 11. Thruster Erosion Rates

4) What is the thruster erosion rate for each propellant? Are the existing data on thruster erosion rates applicable? The ion thruster erosion data in Reference 18 are for low-power (less than 10 kW) ion thrusters. Additional data may be needed to verify these thruster erosion rates at high power levels.

5) Is the propellant cost a major program cost?

6) Is the propellant available in large quantities? Large quantities may be anywhere from 800 to 1200 kg for a unmanned planetary mission (Refs. 19 and 20) to 100,000 kg for a Manned Mars Cargo Mission (Ref. 12) or 10,000 kg for a lunar base mission (Ref. 13).

Each mission scenario may require a different propellant selection. For the ion propulsion Mars cargo missions, the selection may favor krypton. Xenon is four times as expensive as krypton. Also, the propellant availability for 100,000 kg of propellant per mission may force the use of krypton. Krypton can deliver an efficiency similar to xenon (Ref. 18). If the slightly-reduced efficiency over xenon is acceptable, then, again, krypton may be the best propellant.

Argon propellant is substantially less costly than either xenon or krypton. It suffers, however, because it produces a higher ion thruster erosion rate than xenon or krypton. Figure 11 depicts the ion thruster erosion rates (Ref. 18). Also, and perhaps more importantly, the ion thruster efficiency produced using argon is only 50 to 60 percent. For a fixed power level, this lower efficiency will increase the trip time for lunar and Mars missions. Table V compares the trip time for several Mars cargo missions. For the lower-efficiency propulsion systems, the trip time can be substantially longer. Thus, argon is not the first choice amongst ion propulsion propellants.

An MPD thruster using argon or any other propellant may have a 55- to 60-percent thruster efficiency. For this thruster, only the I_{sp} delivered by each propellant may differ. In this case, cost and I_{sp} may be the important selection factors. Ammonia and argon propellant costs are comparable. They are also significantly less costly than xenon or krypton ion propulsion propellants.

An unmanned planetary mission (Refs. 19 and 20) may not require a large mass of propellant. If only 1200 kg are needed, the xenon propellant cost is only 1200 kg x \$1415.5/kg = \$1.7M. Also, the unmanned planetary mission would require a 2- to 3-yr firing time. A Mars cargo vehicle could also require an equally-long lifetime. Erosion on such a long flight may be an overriding propellant selection factor. If the propellant cost is a small fraction of the program cost, xenon may be the best propellant choice.

After analyzing all of the factors, krypton may be the best ion propulsion propellant. It delivers a similar efficiency to xenon and is only 23 percent as costly. The most-attractive MPD propellant

Table V
Mars Trip Time Comparison:
Ion and MPD Propulsion

	Propellant (M_p, kg)	Electrical Efficiency (Complete Propulsion System)	Trip Time (days)
Ion:			
	Xenon (94,115)	0.70	518.34
	Krypton (92,751)	0.65	547.03
	Argon (90,038)	0.55	619.76
MPD:			
	Argon (88,740)	0.50	667.00
	Ammonia (88,740)	0.50	667.00

Derived from Reference 4 Design:
Power Level = 4 MW
OTV Dry Mass = 40,000 kg
Payload Mass: 180000 kg to Mars Orbit, One-Way Cargo Mission
I_{sp} = 5000 lb_f-s/lb_m

selection is not as easily determined. Additional experimental results on the thruster lifetime and efficiency are needed before a choice can be made.

Aerobraking and Electric Propulsion: When using nuclear-electric propulsion for the Mars cargo mission, the use of aerobraking to place the vehicle in orbit at Mars is an option. This option has a significant impact on the vehicle design, the mission trip time and the payload carrying capability of the cargo mission.

Because aerobraking may be used on the interplanetary approach to Mars or to Earth, there is the benefit of a trip time reduction. By eliminating the low-thrust spiral into the Mars' orbit, a savings of 50 to 100 days of trip time is possible. This savings must also be weighed against the added mass required for the aerobrake and the propulsion system required for the payload to circularize its orbit once the aerobraking maneuver is complete. The option that will deliver the most payload mass into LMO is the low-thrust spiral capture mode. Using aerobraking, the mass of the aerobrake is typically 15 to 20 percent of the spacecraft mass that enters the atmosphere prior to the aerobraking maneuver. This mass penalty is significant and in many cases the electric propulsion propellant mass to place the payload into Mars' orbit is smaller than the mass of the aerobrake. There are trip time reductions enabled by combining aerobraking with electric propulsion but the system implications must be contemplated prior to selecting this hybrid system.

Upon approaching Mars, the payload that is performing the aerobrake maneuver must typically be very compact to fit behind the aerodynamic shell. Because of the long flexible structures that are proposed for electric propulsion systems, these systems do not immediately lend themselves to being protected by an aeroshell. This implies that the electric propulsion cargo vehicle will not be involved in the aerobraking maneuver. If electric propulsion were to be considered with aerobraking, it likely that the cargo vehicle might be an expendable vehicle which is not placed into Mars' orbit. An alternative would be to allow the vehicle to spiral into Mars' orbit and only the cargo would aerobrake into Mars. This option would provide a MW-power source in orbit for mission communications support or high-power science, such as surface radar mapping.

CONCLUSIONS

Both ion propulsion and MPD propulsion offer significant mass reductions over the baseline O_2/H_2 propulsion system. The selection of the "best" operating points for these two propulsion systems is required to allow the maximum benefit to be derived from them. In using electric propulsion for the Mars cargo vehicle, many parameters must be considered. Only through judicious selection of the thruster and power system technology, power level, I_{sp} and propellant type will the "best" system design be determined.

To provide a significant mass reduction for the Mars cargo vehicles, the MPD system should have an I_{sp} of 5000 lb_f-s/lb_m. The best power-and- propulsion system operating point is 4 MW. This power level reduces the one-way transfer time to less than 667 days for the MPD. An ion system using a 6000-lb_f-s/lb_m and a 4-MW power level will reduce the trip time to 580 days. These performance levels and power levels allow the up to a 50-percent LEO mass savings over the baseline O_2/H_2 propulsion system and deliver the payload to Mars with an acceptable trip time.

To minimize the mass of the cargo vehicle, a power level no greater than 4 to 10 MW is needed. At power levels higher than 20 MW, the mass of the cargo vehicle is comparable to or higher than the chemical propulsion baseline option. The lower power levels allow a very significant reduction in the LEO initial mass. This reduction in launch mass can significantly reduce the launch costs for a piloted Mars mission.

The power-and-propulsion system specific mass that will be available for the Mars cargo vehicle is dependent on several factors. The power level per thruster and the technology readiness of megawatt-class reactors are very important in reducing the cargo vehicle mass. If the advanced lightweight cargo vehicles (10 kg/kW) are not available, a different thruster I_{sp} will be required. If the power-and-propulsion specific mass is greater than 30 kg/kW, an I_{sp} of over 10,000 lb_f-s/lb_m is needed. With this high I_{sp} the electric propulsion cargo vehicle can deliver the same mass savings as a 5000-lb_f-s/lb_m I_{sp} MPD cargo vehicle with a 10- kg/kW specific mass.

The thruster technology selected for the cargo mission is also dependent upon the complexity and the number of engines in the thruster array. At the power levels analyzed in this study, the number of MPD thrusters can be fewer than those required by an ion propulsion system. This reduction in the number of thrusters occurs at a power level above 10 MW. If the power level is below 4 MW, the ion system requires fewer thrusters. Specifically, at a 4- to 10-MW power level, the number of thrusters can be similar.

There a several other factors that must be considered for the selection of electric propulsion: cost, thruster erosion rate and propulsion efficiency. These factors may not be applicable to each mission type. Each of these factors should be reviewed in any electric propulsion system selection. Thruster erosion and efficiency are dependent upon the propellant type. For Mars cargo missions, krypton may be the best choice for ion propulsion. It has almost the same thruster efficiency as xenon and is only 23 percent of the cost of xenon. Argon has a low efficiency. This lower efficiency significantly increases the trip time for lunar and Mars missions. Also, there is a higher ion-thruster erosion rate with argon than with xenon or krypton.

Electric propulsion is not a likely candidate for the piloted vehicle of the first Mars missions. In later years, after the establishment of a Mars base, the Mars transfer missions would no longer require the fast sprint-class missions. Electric propulsion transfer vehicles could enable a greater cost savings in this scenario. Piloted vehicles using electric propulsion, with power levels similar to the cargo transfer vehicles (4 to 10 MW), could be used for the more-routine resupply and crew transfer missions once a significant infrastructure has been established around Mars.

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