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NUCLEAR ELECTRIC PROPULSION

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ABSTRACT

We investigate the feasibility of using nuclear electric propulsion (NEP) for slow "freighter" ships traveling from a 500 km low Earth orbit (LEO) to the Moon's orbit about the Earth, and on to Mars. NEP is also shown to be feasible for transporting people to Mars on long conjunction-class missions lasting about nine months one way, and on short "sprint" missions lasting four months one way. Generally, we have not attempted to optimize ion exhaust velocities, but rather we have chosen suitable parameters to demonstrate NEP feasibility. Various combinations of missions are compared with chemical and nuclear thermal propulsion (NTR) systems. Typically, NEP and NTR can accomplish the same lifting task with similar mass in LEO. When compared to chemical propulsion, NEP was found to accomplish the same missions with 40% less mass in LEO. These findings are sufficiently encouraging as to merit further studies with optimum systems.

INTRODUCTION

Space propulsion systems can be placed into two broad categories: (1) "impulse" rockets, which produce large accelerations for short periods of time, typically several minutes, and (2) "low-thrust" rockets, which produce small accelerations for long periods of time, typically several months. All of today's operational rockets are of the impulse type. Usable low-thrust engines have been developed in the laboratory.

We address here a specific low-thrust rocket by assuming the engines to be 30 cm diameter mercury ion thrusters with characteristics that exist in the laboratory today. A specific thruster power of 125 w/kg is assumed (see Table I). The thrusters are powered by a nuclear reactor

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NOTE TO THE READER: As the Manned Mars Mission Workshop approached, the authors were asked to investigate the feasibility of using nuclear electric propulsion in a manned Mars program. The present paper constitutes our preliminary findings as of June, 1985. Because low-thrust propulsion showed such promise with this first investigation, more careful studies involving numerical integration techniques were subsequently undertaken and the findings were published as two Los Alamos reports (Refs 8,9). The conclusions have not changed significantly.

TABLE 1

PROJECTED NUCLEAR REACTOR CHARACTERISTICS

	REF 1	REF 2	REF 3	TAKEN HERE
ELECTRIC POWER (Mw_e)	8.5	1	10	3
MASS (metric tons)	26	4	75	20
SPECIFIC POWER (w/kg)	327	250	133	125

PROJECTED ION THRUSTER CHARACTERISTICS

	Ar MPD	Xe ION	Hg ION	CURRENT Hg ION
SPECIFIC IMPULSE (s)	5,000	5,000	4,250	3,000
THRUST PER ENGINE (n)	14.7	13.4	0.63	0.132
DIAMETER (cm)	3	30	30	30
SPECIFIC POWER (w/kg)	300,000	7,500	1,900	125
SYSTEM EFFICIENCY	0.5	0.78	0.7	0.7

TABLE 2

FOUR MONTH "SPRINT" TO MARS WITH NEP

	MASS
MISSION MODULE (3 people)	
(tons)	28
(k lbs.)	62
CONSUMABLES	
(tons)	6
(k lbs.)	13
STRUCTURE (k = 0.05)	
(tons)	2
(k lbs.)	4
REACTOR (3Mw_e, 8 kg/kw)	
(tons)	24
(k lbs.)	53
ENGINES	
(tons)	24
(k lbs.)	53
PROPELLANT	
(tons)	34
(k lbs.)	75
TOTAL MASS IN EARTH-MOON ORBIT	
(tons)	115
(k lbs.)	253

supplying 3 megawatts of electrical power. In addition, we have conservatively assumed a specific power of 125 w/kg to describe the power source reactor, shielding, and electrical conversion system. (Ref. 1-4) Low-thrust propulsion relying on nuclear reactors for electrical energy - which is then used to accelerate ions - is referred to as nuclear electric propulsion (NEP).

Specific impulse, I_{sp}, which relates directly to exhaust velocity, c, is used to characterize rocket engines. Ideally, the specific impulse is given by

$$I_{sp} = c/g_o$$

were g_o is the gravitational acceleration at the Earth's surface. Here we take $g_o = 9.8\ m/s^2$ and for our purposes, we characterize chemical, nuclear thermal, and nuclear electric propulsion systems by I_{sp} = 460 sec, 850 sec, and 3,000 sec, respectively.

The purpose of this paper is to examine the feasibility of using nuclear electric propulsion for slow "freighter" ships traveling from a 500 km low Earth orbit (LEO) to the Moon's orbit about the Earth, and on to Mars. We also show that NEP is feasible for transporting people to Mars on long conjunction-class missions, lasting about 9 months one way, and on short "sprint" missions, lasting 4 months one way. Various combinations of missions are compared with chemical and nuclear thermal propulsion systems.

Our study shows that NEP matches with Nuclear thermal performance about evenly. However, when we compared NEP with chemically fueled impulse rockets, we found NEP could accomplish the same missions with 40% less mass. We arrive at these factors by comparing the amount of mass that must be delivered initially from the surface of the Earth to low Earth orbit. When other criteria are used, such as obtaining reusable ships, low-thrust rockets become even more attractive. In short, we believe the best rocket propulsion system for most situations is a hybrid system combining the best features of impulse rockets and low thrust rockets.

WHY CONSIDER LOW THRUST ROCKETS

In its simplest form, the fundamental rocket equation relates M_p, the mass of propellant required to change the rocket velocity by delta v, with the constant propellant exhaust velocity, c. The equation may be written

$$M_p/M_i = [1 - e^{-(\Delta v/c)}]$$

where M_i is the initial rocket mass. Since the exhaust velocity of ion engines is extremely high, less propellant is required than for a purely chemical rocket. This illustrates just one of the advantages of a low thrust propulsion system.

Another advantage of low-thrust propulsion is illustrated in Figure 1. Here an NEP rocket is slowly spiraling out from low Earth orbit. (It should be mentioned that this process is not drawn to scale, i.e. there would be many more turns of the spiral at low altitudes.) For small accelerations ($a/g_o << 1$), the ship velocity will be nearly equal to the velocity, V_c, required for a circular orbit at each point along the trajectory. This means that $V(r) \sim V_c$. When the ship reaches the moon's orbit, for example, it can have nearly zero hyperbolic velocity relative to the Moon. The same can be true of a ship traveling to Mars, where little or no braking maneuvers are required. This gives NEP the advantage that a ship can either choose to spiral slowly into Mars orbit, or be captured into a highly elliptical orbit with a small (chemical) delta v of, say, 200 m/S applied at periapsis.

NEP ORBITAL CALCULATIONS

The calculations for this paper, except for the last section, follow those of Jones (Ref. 5), where the initial mass and trip time are parameterized in terms of specific impulse, power, thruster efficiency, tankage fraction, specific reactor power, specific thruster power, delta v, and payload mass. In this work, we have taken thruster efficiency to be 0.7, the tankage fraction to be 0.05, the specific reactor power to be 125 w/kg, and the specific thruster to be 125 w/kg. Specific impulses ranged from 3,000 sec to 10,000 sec, and the power ranged from 3 Mw_e to 30 Mw_e. The delta v used for LEO to Moon's orbit was 6.93 km/s and included a 28.5^o orbital plane change. The delta v used for the Earth to Mars mission was 5.82 km/s and included a 1.85^o orbital plane change. The payload mass was either adjusted to make the trip time about one year, or was fixed to compare NEP with some mission using chemical propulsion. In addition, a factor of 0.05 times the reported payload mass was subtracted from the calculated payload mass to account for the

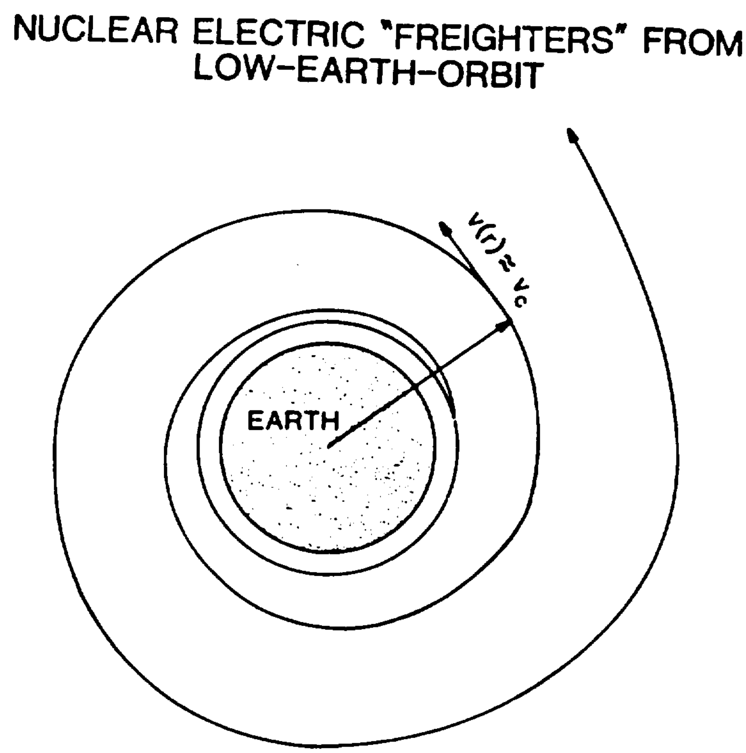

Figure 1. The low thrust-spiral of nuclear electric propulsion (NEP) rocket leaving low Earth orbit (LEO).

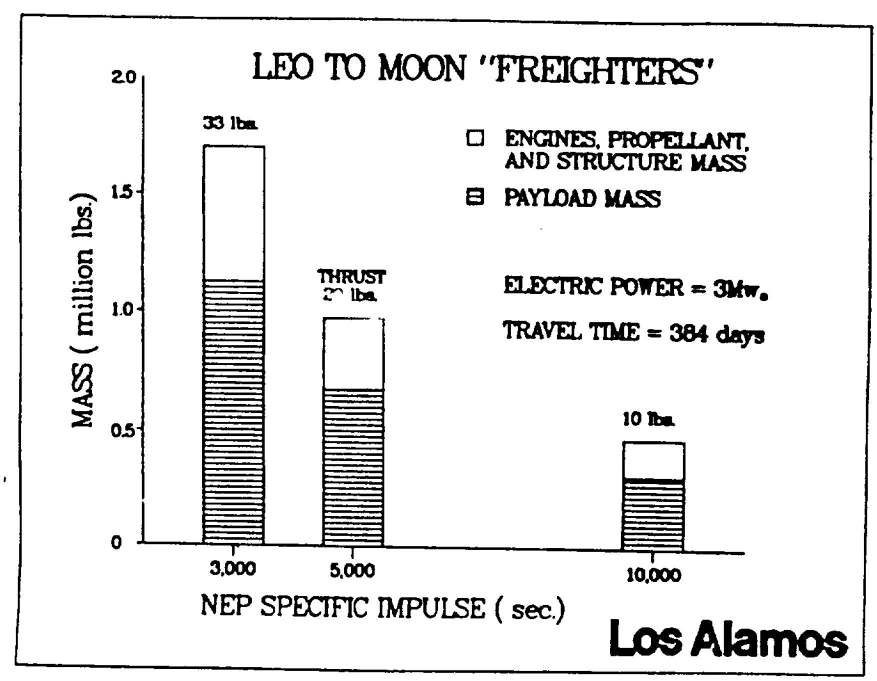

Figure 2. Payload capabilities of NEP freighters going from LEO to the Moon's orbit.

payload structure mass. The equations reported by Jones are valid for $a/g_0 \ll 1$ and a tangential thrust, provided the polar coordinate angle of the trajectory is small (See Ref. 6). Initially, the rocket must increase its velocity by accelerating away from its host planet to develop enough centrifugal acceleration to increase its radius vector. Subsequently, as the radius vector increases, the ship's velocity decreases, and it falls behind its host planet. This initial process is not addressed in our calculations. Based on Irving's report (Ref. 7), we have verified that our calculations are valid for the long-duration missions to Mars reported here, but not for times much smaller than 9 months.

We used the result of Irving's work to derive our 4 month sprint mission to Mars. Irving formulates low thrust propulsion in terms of a fundamental integral

$$\gamma^2 = \frac{\alpha}{2} \int_0^T a^2(t)\, dt$$

here α is one divided by the specific power and the thrust acceleration, a(t), varies with time. Irving then shows how to optimize reactor mass, payload mass, and propellant mass one γ^2 is known.

For the last section of this study, we used $\alpha = 8\text{kg/kw}_\alpha$ and a 3 Mw_e power supply to address a 4-month one-way mission to Mars. The remainder of the ship components were optimized accordingly.

NEP FREIGHTERS

We began our study by noticing that months are usually required for NEP to lift a large payload from payload from LEO to the Moon's orbit. Consequently, we focused first on unmanned freighters where long transfer times are not critical. By extending the transfer time to a year, freighters make use of the large mass carrying capability of NEP. Figure 2 shows the payload mass which can be delivered to the Moon's orbit about the Earth from LEO for three specific impulses. Notice that when trip time and electrical power are held constant, the payload decreases as the specific impulse increases. More detailed information is given in Appendix A, Table A1.

Once the freighter is in the Moon's orbit, gravitational assists from the Moon can be used to direct the ship's velocity toward Mars, as illustrated in Figure 3. We now concern ourselves with the cargo we wish to take to Mars. Figure 4 shows the payload mass that can be transported

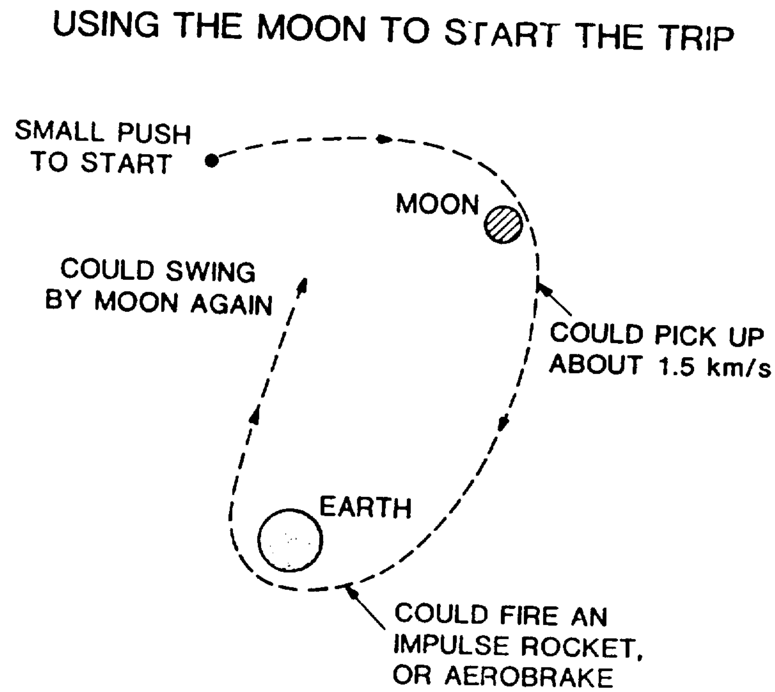

Figure 3. Gravitational assists from the Moon can start an NEP rocket to Mars.

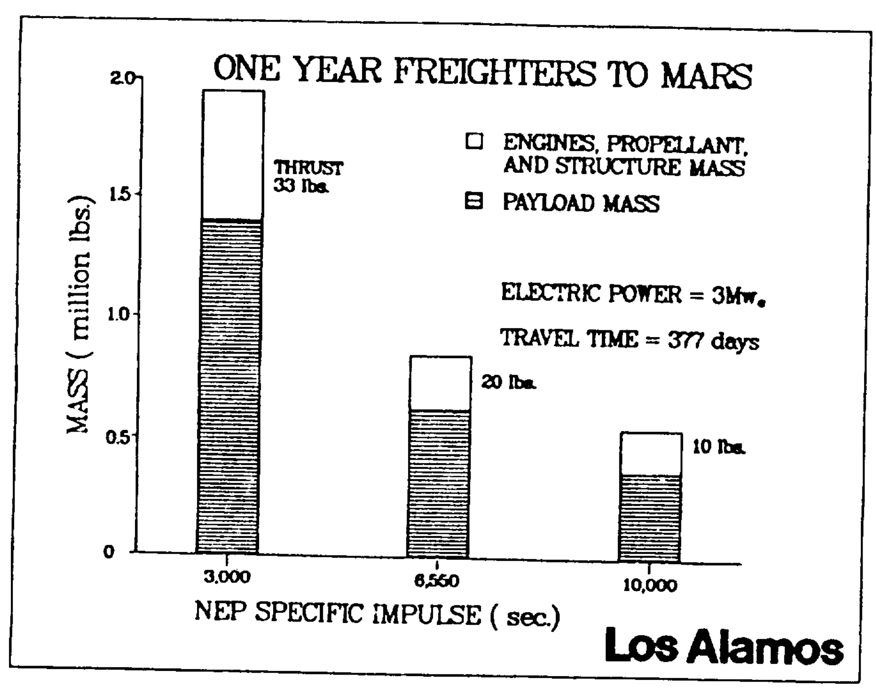

Figure 4. Payload capabilities of NEP freighters going from the Earth's orbit around the Sun to Mars' orbit around the Sun.

for three specific impulses. Notice that the same inverse relationship holds between payload and specific impulse as in traveling from LEO to the Moon's orbit. However, more importantly, for the same reactor power and approximate trip time, more payload can be taken from the Earth-Moon system to Mars than from LEO to the Moon's orbit (see Figure 2). In short, it is cheaper to take cargo to the Moon from Mars than from LEO. This fact is extremely interesting if a lunar base already exists. A further analysis is provided in Table A2.

HYBRID NEP VERSUS IMPULSE ROCKETS

We now address the issue of sending a manned mission to Mars using NEP. To make such a comparison with impulse rockets, we have first identified a "hybrid" rocket combining NEP and chemical propulsion. We consider the 1999 opposition class mission with Mars and Earth aerobraking as described by the Marshall Space Flight Center for a chemical rocket. In the hybrid rocket, we have kept the mass of all the chemical rocket components the same, except for the first stage, which we replaced with an NEP system in LEO. The NEP freighter is used to lift the chemical rocket to lunar orbit. At that point, the crew joins the ship. From there, the Moon is used for gravitational assist, as stated earlier, and the chemical engine is fired at perigee. Otherwise, the I_{sp} = 460 (chemical) and I_{sp} = 3,000 (NEP) systems shown in Figure 5 are the same. As another comparison, I_{sp} = 850 (nuclear thermal reactor, NTR) delivering the same payload to Mars and back to Earth is shown in Figure 5. Again, more detail is given in Appendix A, Table A3.

Another mission scenario involves a conjunction-class trajectory. In Figure 6, NEP is compared with NTR and chemical rockets for conjunction class missions. The NEP system here is a different type of hybrid rocket. Four 15,000 lb. thrust chemical engines with storable propellant and I_{sp} = 345 sec are contained within the NEP system. These chemical engines are used so that small velocity changes of about 200 m/s can be made quickly for escaping from and braking into Earth and Mars orbits. Table A4 gives more specific information about this mission.

FOUR MONTH "SPRINT" TO MARS WITH NEP

Lastly, we consider getting a fast manned mission to Mars from the Moon's orbit about the Earth, using NEP - a "sprint" mission in effect. Table 2 shows an initial rocket mass of 252,000 lbs. that delivers a

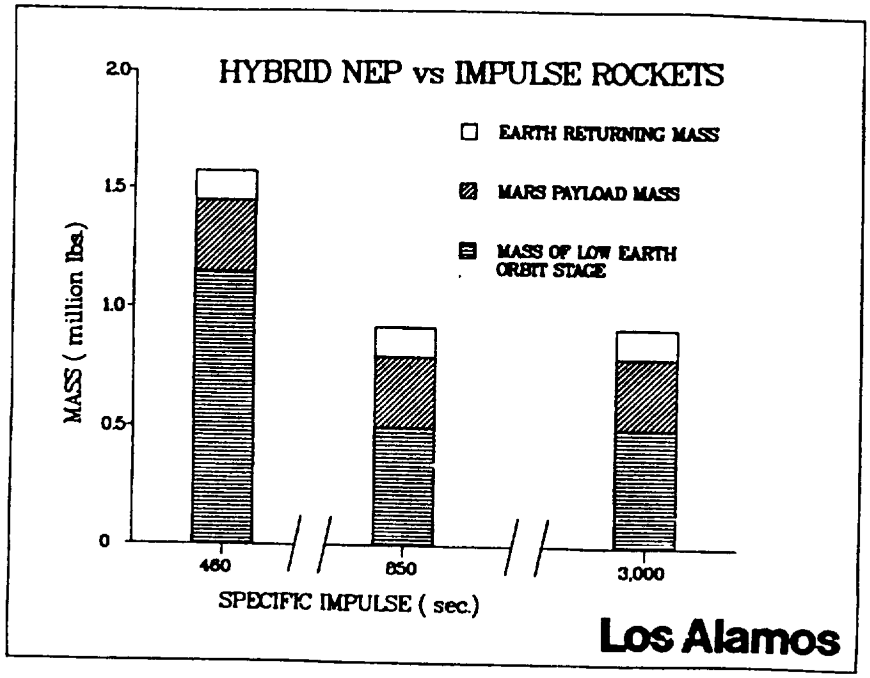

Figure 5. A 1999 opposition-class mission from Earth to Mars. Specific impulses of 460, 850, and 3,000 represent chemical propulsion, nuclear thermal rocket propulsion, and hybrid NEP rockets, respectively. The second and third stages of all three rocket systems are kept the same.

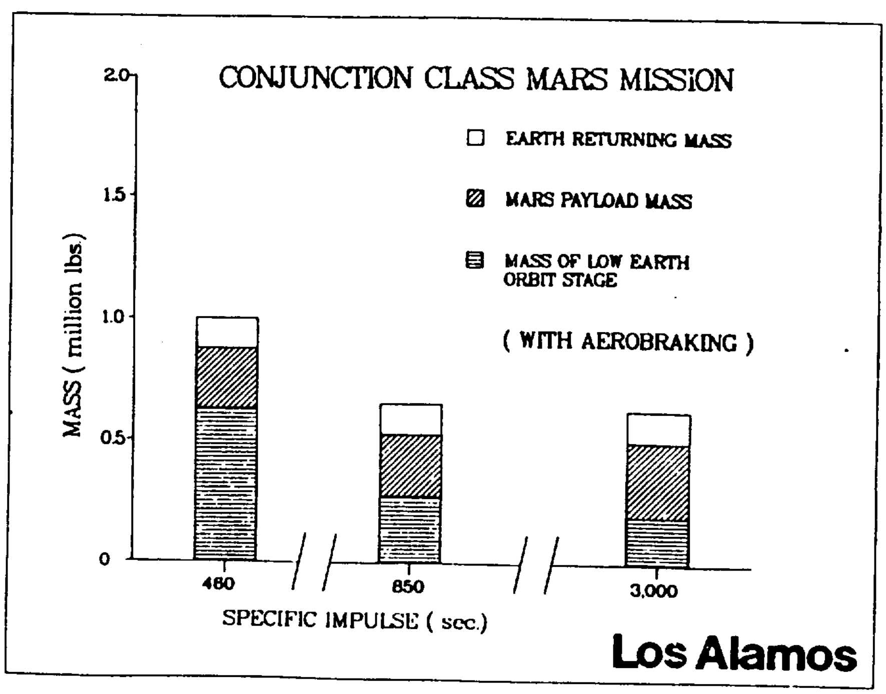

Figure 6. A typical conjunction-class mission from Earth to Mars. See the figure caption for figure 5.

three person crew to Mars in four months. These numbers are taken from Reference 7, as stated earlier. Reference 7 uses a variable thrust rather than the constant NEP thrust assumed in all other calculations for this study. However, this establishes the feasibility of a four-month "sprint" mission to Mars, which would be very difficult with chemical propulsion.

ACKNOWLEDGEMENTS

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APPENDIX - TABLE 1

LOW EARTH ORBIT (LEO) TO MOON "FREIGHTERS"

	CASE 1	CASE 2	CASE 3	CASE 4
SPECIFIC IMPULSE (sec.)	3,000	5,000	10,000	5,000
ELECTRIC POWER (Mw_e)	3	3	3	30
TRAVEL TIME (days)	384	383	386	383
THRUST (n)	143	86	43	867
(lbs.)	32	19	10	192
MASS IN LEO (tons)	768	439	214	4,388
(k lbs.)	1,690	966	471	9,654
PROPELLANT MASS (tons)	161	58	15	579
(k lbs.)	354	128	33	1,274
ENGINES, STRUCTURE (tons)	48	48	48	480
(k lbs.)	106	106	106	1,056
PAYLOAD MASS (tons)	514	311	142	3,105
(k lbs.)	1,131	684	312	6,831

APPENDIX - TABLE 2

ONE YEAR FREIGHTERS TO MARS

	I_{sp}=3,000 sec.	I_{sp}=6,550 sec.	I_{sp}=10,000 sec.
TRAVEL TIME (days)	377	375	378
THRUST (n)	143	65	43
(lbs.)	32	15	10
MASS LEAVING E/M (tons)	881	381	247
(k lbs.)	1,938	838	543
PROPELLANT MASS (tons)	158	33	14
(k lbs.)	348	73	32
MARS PAYLOAD (tons)	633	283	175
(k lbs.)	1,393	623	385

APPENDIX - TABLE 3

HYBRID NEP vs IMPULSE ROCKETS FOR MARS MISSIONS

AEROBRAKING (PROPULSIVE BRAKING)

	CHEMICAL I_{sp} = 460 sec.	NTR I_{sp} = 850 sec.	CHEMICAL + NEP I_{sp} = 3,000 sec.
EOI (tons) (k lbs.)	60 (150) 133 (329)	60 (115) 133 (254)	60 (150) 133 (329)
TEI (tons) (k lbs.)	101 (246) 222 (541)	94 (151) 207 (332)	101 (246) 222 (541)
MOI (tons) (k lbs.)	187 (632) 412 (1,390)	179 (329) 394 (723)	187 (632) 412 (1,390)
TMI (tons) (k lbs.)	715 (2,121) 1,574 (4,667)	421 (757) 926 (1,666)	281 (900) 618 (1,979)
LEO (tons) (k lbs.)	715 (2,121) 1,574 (4,667)	421 (757) 926 (1,666)	422 (1,216) 929 (2,675)

APPENDIX - TABLE 4

CONJUNCTION-CLASS MARS MISSION WITH NEP

	CHEMICAL I_{sp} = 460 sec.	NTR I_{sp} = 850 sec.	CHEMICAL + NEP I_{sp} = 5,660 sec.
EOI (tons) (k lbs.)	60 133	60 133	112 247
TEI (tons) (k lbs.)	85 187	84 185	130 286
MOI (tons) (k lbs.)	170 375	169 371	201 442
TMI (tons) (k lbs.)	453 996	300 660	287 631