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COMPARATIVE ASSESSMENT OF HUMAN MISSIONS TO MARS*

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There is no definite answer to how we shall go to Mars. In this paper, we develop a propellant metric to assess the cost of Earth-Mars transportation systems, and consider how various propulsion choices, trajectory designs, and staging points affect these systems. Results are provided for both short-term (exploration) and long-term (settlement) scenarios. We find that low-thrust propulsion provides significant propellant savings with relatively little development cost, while nuclear thermal launch vehicles (which have a low technology readiness level) provide the greatest propellant savings. As vehicle mass increases (to achieve higher levels of safety and comfort for the astronauts), the propellant-optimal system shifts from Semi-Direct, to Stop-Over, to Cycler scenarios – a tantalizing suggestion for the evolution of the human Mars transportation system.

INTRODUCTION

In the future there will be a sustained human presence on Mars. To reach this goal, a reliable and affordable transportation system must be developed. There are many Mars mission proposals,[1-12] and several comprehensive analyses and comparisons of human missions to Mars.[13-20] Still, there is no definite answer to how we shall go to Mars, though NASA has published a Design Reference Mission (DRM) as a benchmark.[7,8]

An efficient means of determining the cost of a mission is required to compare a wide variety of scenarios. The benefit of near-term technologies (e.g. nuclear propulsion, or in-situ resource utilization) can then be assessed over the spectrum of transportation scenarios. Such a broad analysis provides direction for the establishment and evolution of Mars exploration.

MISSION COST

We choose propellant mass as the key factor in our cost assessment. We believe that propellant mass provides a reliable indication of the long-term costs to sustain human transport between Earth and Mars, though we recognize that hardware development costs may be more significant when establishing the human transportation system. Cost models[21] (in U.S. dollars) are probably unreliable for such large-scale missions that are

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forecast decades in the future. Some of these cost models depend on the dry mass of the system (i.e. excluding propellant and payload), but the dry mass is better suited to estimate start-up costs rather than recurring costs. The initial mass in low-Earth orbit (IMLEO) is another effective cost indicator, but it does not always reflect the entire transportation system (mass transfer to *and* from Mars). For example, if propellant that is produced on Mars is sent to LEO (say to refuel an upper stage), the IMLEO can remain fixed, but the propellant mass decreases because it is often easier to send mass to LEO from Mars than from Earth's surface. Also, we wish to compare different launch vehicle propulsion systems, but the IMLEO calculation does not include launch vehicle propellant mass. In spite of these exceptions, the propellant-optimal and IMLEO-optimal systems are often the same, as a large percentage of the IMLEO is propellant mass. Of the many cost models that are available to us, we choose propellant mass because it provides reliable insight to the mass requirements at all stages of the Mars transportation system.

Specifically, our cost function is the total propellant mass (from Earth launch to Earth landing) divided by the Mars payload mass. By normalizing the propellant cost by the payload mass we create a cost function that is independent of the size of the mission. For example, a mission that lands 10 mt on Mars will require ten times the propellant mass as a similar mission that lands only 1 mt. However, both missions have the same cost value, because their cost-to-return ratios are the same (they get the same "bang for the buck"). Currently, we do not include development costs (e.g. establishing a Mars base, building a reusable interplanetary transport vehicle, or technological developments) in our model, but we do consider these costs when comparing different missions. One mission occurs every Earth-Mars synodic period, which provides a cost rate (propellant cost per payload mass per synodic period). Consequently, our model reflects the recurring costs to *sustain* routine transportation between Earth and Mars for many years.

DESIGN SPACE

To study various scenarios, we define a quantity called the support factor, f:

$$f = \frac{m_{support} + m_{return}}{m_{return}} \tag{1}$$

where m_{return} is the mass of the payload that returns to Earth and $m_{support}$ is mass that may (in principle) be left in space. Propulsion system mass (propellants, engines, tanks, etc.) is not included in the support factor calculation. Examples of return payload mass include astronauts, scientific equipment, and the ascent/descent cabin; examples of support mass are radiation shielding, the life-support system (in transit), artificial gravity, and interplanetary transfer vehicle structure. Larger values of f generally correspond to safer and more comfortable trips for the astronauts.

To clarify the meaning of the support factor, let us suppose a mission requires 5 mt of astronauts and an ascent/descent cabin and 20 mt for an Earth Return Vehicle (ERV). (These mass values are similar to those provided in NASA's DRM.[8]) If a Mars habitat is sent with the crew, then the ERV is (technically) not needed on the surface and may be left in space. In this case:

$$m_{return} = 5 \text{ mt}, \ m_{support} = 20 \text{ mt, and } f = (20 + 5)/5 = 5$$

We find that a support factor of 5 corresponds to moderate missions (e.g. NASA's DRM[7]), while lower values of f are more Spartan (e.g. Zubrin's Mars Direct[4,6]) and higher f values are more luxurious (e.g. the cycling space hotels of Aldrin[10] and Nock[9]).

If there was no habitat on Mars, then the ERV would have to double as surface living quarters and could not be left in space. Here,

$$m_{return} = 5 \text{ mt} + 20 \text{ mt}, \ m_{support} = 0 \text{ mt, and } f = (0 + 25)/25 = 1$$

This latter case is rare, as almost all proposed Mars missions either leave a habitat on the surface or assume that habitable structure already exists on the surface from previous missions. Also, some mission designs require the ERV to land on Mars even when an extra habitat is present. The support factor for these missions is still not unity (even if no mass is actually left in space) because the ERV *could* have been left in space as support mass.

The support factor helps distinguish between the two main vehicles in a human Mars transportation system: 1) the Transfer Vehicle (e.g. ERV or Cycler) and 2) the Taxi (launch vehicle and upper stage). The Transfer Vehicle transfers the payload and support mass between Earth and Mars, while the Taxi ferries the payload mass between the Transfer Vehicle and the surface of Earth or Mars. There is often significant propellant savings when the Transfer Vehicle (support mass) is left in space (e.g. in a parking orbit or in interplanetary space) between missions. Thus, the support factor is a crucial parameter in comparing Mars transportation systems.

Another important variable in human missions to Mars is the payload return ratio, μ_{return}:

$$\mu_{return} = m_{return} / m_{outgoing} \qquad (2)$$

where m_{return} is the mass of the payload that returns to Earth, and $m_{outgoing}$ is the mass of the outgoing payload that lands on Mars. For example, if 10 mt of payload lands on Mars, but only 5 mt returns to Earth, the payload return ratio is simply,

$$\mu_{return} = 5/10 = 0.5$$

In most cases the return payload is only the astronauts and a descent cabin, while the outgoing payload may include astronauts, consumables, habitats, a power production plant, or other cargo.

Table 1 provides a list of the transportation architectures covered in our analysis. We characterize these scenarios by the placement of the Transfer Vehicle at planetary encounters. For example, Direct missions launch the payload with the support mass at both Earth and Mars. However, since the support mass is not needed at the surface (by definition), it could have been placed in a parking orbit (Stop-Over) or left on a heliocentric trajectory (Cycler). We do not examine missions that incorporate a surface launch at Mars with a parking orbit or flyby at Earth, nor do we examine missions that launch the support mass at Earth with a Mars flyby, because these scenarios are not as prevalent in the literature as the others.

Table 1
SUMMARY OF EARTH-MARS TRANSPORTATION SCENARIOS

System	Ref.	Earth Encounter	Mars Encounter
Direct	4	Surface	Surface
Semi-Direct	7	Surface	Parking Orbit
Stop-Over	11	Parking Orbit	Parking Orbit
Semi-Cycler	10	Flyby	Parking Orbit
Reverse Semi-Cycler[a]	20	Parking Orbit	Flyby
Cycler	12	Flyby	Flyby

[a] Mars free return with parking orbit at Earth.

The benefit of near-term technologies is also of key interest in Mars mission design. The various technologies that we consider along with their approximate "technology readiness levels" (TRL) are presented in Table 2. Higher TRL values correspond to lower development costs, and vice versa. In this paper, we do not include lunar propellant production either at the Moon or Phobos. We also forgo analysis of libration-point stations, which may be attractive (from a propellant standpoint) if there is lunar propellant production. However, we expect that the propellant cost of missions with libration-point stations will be comparable to Stop-Over missions as the energy requirements of the libration points and a high-energy elliptic parking orbit are similar. Travel to and from the libration-points also requires a trade between the ΔV and time-of-flight. An analysis of libration-point stations and lunar propellant production are beyond the scope of the present paper, but may be addressed in a future work.

Table 2
CURRENT AND NEAR-TERM TECHNOLOGIES

Technology	Approximate Readiness Level
Chemical Propulsion	9
Parking Orbit Rendezvous (Earth)	9
Low-Thrust (SEP)	8
Parking Orbit Rendezvous (Mars)	8
Refuel in Orbit (Earth)	8
Refuel in Orbit (Mars)	7
Hyperbolic Rendezvous (Earth)	7
Hyperbolic Rendezvous (Mars)	6
Nuclear Thermal Rocket (NTR)	6
Aerocapture	6
In-Situ Propellant Production (ISPP)	5
NTR (Launch Vehicle)	4
Mars Water Excavation	3

MODELING ASSUMPTIONS

1.) The support factor, f, for both the exploration and settlement phases is 5. This value for the support factor is estimated from mass values found in Refs. 6-9. We also consider a range of f (from 1 to 10), since the support factor drives the cost metric and may vary for different missions.
2.) During the early phases of Mars exploration only one-tenth of the Earth-Mars payload mass will return to Earth (μ_{return} = 10%).[6,8] For example, only the astronauts (5 mt with ascent/descent cabin) will return to Earth, while the Mars payload may include a habitat (20 mt), cargo (20 mt), consumables (5 mt), as well as the astronauts (5 mt). This estimate gives a payload return ratio of

$$\mu_{return} = 5/(20+20+5+5) = 10\%$$

However, in the settlement phase, half of the Earth-Mars payload will return to Earth (μ_{return} = 50%).[6,8] In this scenario, the astronauts (5 mt) return to Earth, but only the astronauts (5 mt) and consumables (5 mt) are sent to Mars. A range of μ_{return} (from 0% to 100%) is also examined.
3.) Consumables for the interplanetary transfers (Earth-Mars and Mars-Earth) have a mass equal to the return payload mass.[6,8] All consumables come from Earth and consumables for the return trip are left on the Transfer Vehicle at Mars (i.e. they are not landed, then re-launched).
4.) The Mars surface payload is divided into two categories 1) time-sensitive (e.g. astronauts and ascent/descent cabin) and 2) time-insensitive (e.g. Mars habitat and consumables). The time-insensitive payload is sent to Mars on a low-energy (Hohmann) transfer, while the time-sensitive payload requires injection to a higher V_∞. The outgoing time-sensitive payload mass is equal to the return payload mass.
5.) For simplicity, we assume the V_∞ at Mars is 90% of the Earth V_∞. Nominally, the V_∞ is 4 km/s (at Earth) and 3.6 km/s (at Mars) for all scenarios except for Cyclers, which have a V_∞ of 5 km/s (at Earth) and 4.5 km/s (at Mars). (The higher V_∞ ensures that all scenarios have similar transfer times). Since the V_∞ depends on the actual trajectory, we also vary the Earth V_∞ from 3 km/s to 8 km/s to assess its effect on relative mission cost. The Cycler V_∞ is always 25% greater than the V_∞ of the other scenarios. The low-energy (Hohmann) transfer V_∞ are 2.95 km/s and 2.65 km/s at Earth and Mars, respectively.
6.) Cyclers and Reverse Semi-Cyclers use four Transfer Vehicles to ensure a transport opportunity each synodic period. The other systems only require two Transfer Vehicles.
7.) Replacement portions of the Transfer Vehicle are launched at each Earth-to-Mars leg, to account for maintenance or renovation of the vehicles over an extended period of time. We assume that each Transfer Vehicle is completely renewed every twenty synodic periods (about 43 years).
8.) When a Transfer Vehicle enters a high-energy parking orbit (HPO) about a planet, the periapsis is 300 km above the planet's surface and the period is seven days. All maneuvers to enter or depart the parking orbit are assumed to occur tangentially at periapsis. The orientation and precession (due to planetary oblateness, lunar, and

solar perturbations) of the parking orbit are not directly computed, but we do estimate the cost of reorienting the orbit (0.2 km/s at Mars and 0.3 km/s at Earth).[22]

9.) Earth and Mars launch vehicles are modeled as two-stage rockets that ascend from the surface to a low (300 km) circular orbit (LCO). We note that the altitude of the LCO and the periapsis of the HPO are the same. The launch vehicle ΔV calculation does not directly include gravity, drag, or steering losses nor the velocity due to planetary rotation, but we do add a 5% margin to the ΔV value [see Eq. (3)].

10.) Mars launch vehicles and upper stages are recovered at Mars via aerobrake for reuse.

11.) If the taxi must rendezvous with the Transfer Vehicle in a parking orbit, then there is a single upper stage; if the taxi must reach the Transfer Vehicle on a hyperbolic trajectory, then there are two stages. The second stage may be used as an abort option in the event of rendezvous failure.

12.) Due to timing and orbit-phasing concerns, the taxis will not go into an HPO prior to hyperbolic rendezvous, because this maneuver is time critical Also, low-thrust propulsion will not be used for hyperbolic rendezvous because of the same timing concerns.

13.) Transfer Vehicles retain their engines and jettison used propellant tanks. There is only one type of engine on the Transfer Vehicle (e.g. it does not use NTR at Earth and methane/oxygen at Mars).

14.) All propellant tanks come from Earth. Propellant tanks have an inert mass fraction of 5%.

15.) The Transfer Vehicle is refueled at the most economical location (Earth, Mars, or both) for the given propulsion system.

16.) Methane/oxygen propellant can be produced at Mars (using ISPP). This propellant will be made from hydrogen sent from Earth on a low-energy transfer. One kilogram of this hydrogen is combined with the carbon dioxide in the Martian atmosphere to yield 16 kilograms of propellant.[6] We also examine scenarios where water is plentiful and readily available at Mars; in this case hydrogen and oxygen may be produced via electrolysis for propellants.

17.) Ten percent of hydrogen mass is lost due to boil-off during transfer from Earth to Mars (and vice versa).

18.) Propellant from Earth (Mars) may be sent to a Mars (Earth) parking orbit on a low-energy trajectory via a "Tanker." This propellant may then be used by an upper stage or Transfer Vehicle after refueling in orbit.

19.) Once the payload is in a parking orbit at planetary arrival (either via aerocapture or a propulsive maneuver) it will aerobrake down to the surface.

20.) The aeroshell is twenty percent of the decelerated mass in the case of aerocapture and fifteen percent for aerobraking.

21.) The trip from LCO to HPO can take a few months and include several weeks in radiation belts with a low-thrust upper stage, thus precluding the transfer of humans to HPO via low-thrust propulsion. We assume that the time-sensitive outgoing payload and all the return payload mass must use high-thrust rockets to reach HPO.

22.) In Direct and Semi-Direct missions, we expend no propellant to save the Transfer Vehicle at Earth return. In these cases a new Transfer Vehicle would be launched for the next mission.

Propulsion system characteristics are provided in Table 2. The I_{sp} indicates engine performance; inert mass fractions (μ_{inert}) are used to size the inert propulsion mass (engines, tanks, etc.) for a given maneuver.

Table 3: Propulsion System Parameters

Propulsion System	I_{sp} (seconds)	Inert mass fraction
Chemical (H_2/O_2)	450	0.1
Chemical (CH_4/O_2)	380	0.1
Nuclear (NTR)	900	0.3
Low-Thrust (SEP/NEP)	5,000	0.4

PROPELLANT COST MODEL

The following fundamental equations allow us to estimate the amount of propellant that is required to sustain a transportation system between Earth and Mars. Equation (3) provides the ΔV to launch from the surface to LCO.

$$\Delta V_{launch} = 1.05\sqrt{GM\left(\frac{2}{r_{surf}} - \frac{1}{r_{LCO}}\right)} \tag{3}$$

The ΔV required from the upper stage to reach the HPO from LCO is found from Eq. (4),

$$\Delta V_{US} = \sqrt{GM\left(\frac{2}{r_{LCO}} - \frac{1}{a_{HPO}}\right)} - \sqrt{\frac{GM}{r_{LCO}}} \tag{4}$$

or from Eq. (5) if low-thrust is used.

$$\Delta V_{US} = \sqrt{\frac{GM}{a_{HPO}}} - \sqrt{\frac{GM}{r_{LCO}}} \tag{5}$$

If the upper stage achieves hyperbolic rendezvous with the Transfer Vehicle, Eq. (6) provides the ΔV from LCO to V_∞.

$$\Delta V_{esc} = \sqrt{\frac{2GM}{r_{LCO}} + V_\infty^2} - \sqrt{\frac{GM}{r_{LCO}}} \tag{6}$$

The ΔV required from the Transfer Vehicle to reach V_∞ from HPO is calculated via Eq. (7),

$$\Delta V_{esc} = \sqrt{\frac{2GM}{r_{LCO}} + V_\infty^2} - \sqrt{GM\left(\frac{2}{r_{LCO}} - \frac{1}{a_{HPO}}\right)} \tag{7}$$

or Eq. (8) with low-thrust.

$$\Delta V_{esc} = V_\infty + \sqrt{\frac{GM}{a_{HPO}}} \tag{8}$$

We note that Eqs. (5) and (8) do not model instantaneous burns, but instead provide useful correlations to determine propellant mass with low-thrust propulsion.[21, 23, 24]

The rocket equation is used to determine mass fractions for a single stage

$$\mu_{stage} = \exp\left(\frac{\Delta V}{n g I_{sp}}\right) \tag{9}$$

We derive expressions for the initial mass and propellant mass from Eq. (9):

$$\mu_0 = \frac{m_0}{m_{pay}} = \left[\frac{\mu_{stage}\left(1-\mu_{inert}\right)}{1-\mu_{inert}\mu_{stage}}\right]^n \tag{10}$$

$$\mu_p = \frac{m_p}{m_{pay}} = \left(\mu_0 - 1\right)\left(1-\mu_{inert}\right) \tag{11}$$

We note that equation (11) provides the propellant mass as a ratio of (propulsion) payload mass. Thus by stacking the mission payload, aeroshells, inert mass, and propellants, we calculate the propellant mass at each stage, then sum these propellant masses to calculate the total propellant cost. Since each propellant value is a ratio (m_p/m_{pay}), the total propellant cost can be expressed in terms of outgoing payload mass ($m_p/m_{outgoing}$), which is our cost function.

RESULTS

Exploration and Settlement Phases

Propellant costs for a variety of missions are provided in Tables 4 (exploration phase) and 5 (settlement phase). The first two columns of these tables denote the propulsion system for the launch vehicle, upper stage, and transfer vehicle at Earth and Mars, respectively. The third column provides additional propulsion system information. For example, row 7 (of both tables) describes a system that uses H_2/O_2 for the launch vehicle, upper stage and Transfer Vehicle at Earth (HHH in column 1), and CH_4/O_2 for the Mars launch vehicle and upper stage with H_2/O_2 propulsion for the Transfer Vehicle (MMH in column 2). The methane and oxygen for the Mars vehicles are created via in-situ propellant production as indicated (by the I) in column 3.

We note that while the propellant cost is independent of the outgoing payload mass, the actual propellant mass calculation does depend on this payload. For example, the cost values for a Direct mission in row 1 are 149 in Table 4 and 618 in Table 5. This means that 149 mt and 618 mt of propellant are required for every 1 mt of payload landed on Mars during the exploration and settlement phases, respectively. If the outgoing payload is 50 mt in the exploration phase, but only 10 mt in the settlement phase, the actual propellant mass requirements are 7,450 mt (exploration phase) and 6,180 mt (settlement phase). So, even though the propellant cost values are significantly higher in the settlement phase than in the exploration phase, the actual propellant mass may be less in the settlement phase if the outgoing payload mass is lower.

Table 4
PROPELLANT COST (TOTAL PROPELLANT MASS / PAYLOAD): EXPLORATION PHASE (μ_{return} = 10%, f = 5, Earth V_∞ = 4 km/s)

	Propulsion System[a]			Trajectory Architecture					
	Earth	Mars		Direct	Semi-Direct	Stop-Over	Semi-Cycler	Reverse S-C	Cycler
1	HHH	HHH		149	71.3	69.6	63.1	60.8	62.3
2	HLH	HLH		64.7	38.5	36.9	43.3	32.1	38.2
3	HLL	HLL		56.0	33.2	28.7	32.6	30.6	36.1
4	HHH	HHH	T_M	110	69.5	68.9	61.9	60.7	56.8
5	HHH	HHH	A	124	61.0	54.3	51.1	52.3	53.6
6	HNN	HNN		79.8	42.6	37.1	35.5	35.5	36.2
7	HHM	MMM	I	79.6	71.1	66.8	59.0	57.0	54.9
8	HLL	HLL	A	64.4	59.2	50.3	46.4	46.8	45.1
9	HLL	MML	I	32.0	28.3	23.8	27.8	23.0	26.9
10	HNN	NNN		57.6	39.3	33.9	32.2	31.5	31.7
11	HHH	HHH	AT_M	93.0	60.4	53.6	50.7	46.9	46.3
12	NNN	NNN		21.1	14.1	12.3	11.6	11.3	11.4
13	HNN	HNN	A	73.0	39.9	32.9	32.0	33.1	33.6
14	HHH	HHH	W	54.6	52.3	39.5	39.0	38.3	39.8
15	HHM	MMM	AI	64.4	59.2	50.3	46.4	46.8	45.1
16	HNN	MMN	I	43.1	37.2	31.7	30.1	28.7	28.5
17	HHH	HHH	WT_E	37.7	36.0	29.9	31.7	28.1	29.8
18	HLL	HLL	W	28.1	27.8	23.3	27.2	22.1	25.7
19	HLL	MLL	AI	30.2	27.1	21.3	25.1	21.3	25.8
20	HHM	MMM	AIT_E	53.2	49.4	41.6	38.2	38.6	36.7
21	HNN	MMN	AI	38.8	34.7	27.7	26.8	26.4	26.2
22	HNN	HNN	W	35.2	33.5	25.4	25.5	25.4	26.2
23	HNN	HNN	WT_E	27.5	26.1	20.8	21.1	20.4	21.0
24	NNN	NNN	AWT_E	10.6	10.0	7.74	7.64	7.74	7.78
25	NLN	NLN	AWT_E	8.03	7.5	5.89	6.16	5.81	6.28

[a] A = Aerocapture, H = H_2, I = ISPP, L = Low-Thrust, M = CH_4, N = NTR, T_E = Tanker to Earth, T_M = Tanker to Mars, W = Mars Water.

Tables 4 and 5 are organized so that mission complexity increases from top to bottom and from left to right. Technology complexity increases from row 1, which represents missions that require no technology development, to row 25 where the missions require all of the technologies found in Table 2. The trajectory complexity increases from Direct missions, which require no parking orbits or gravity assists at Earth or Mars, to Cycler missions, which incorporate flybys at both Earth and Mars. A NASA DRM-type mission[8] is found in row 21 under "Semi-Direct" in Table 4. Zubrin's Mars Direct[4] is similar to the mission found in row 15 (or row 21 with the NTR option) under "Direct" in Table 4. We note that as trajectory and technology complexity increases, propellant cost tends to decrease.

Table 5
PROPELLANT COST (TOTAL PROPELLANT MASS / PAYLOAD):
SETTLEMENT PHASE (μ_{return} = 50%, f = 5, Earth V_∞ = 4 km/s)

	Propulsion System[a]			Trajectory Architecture					
	Earth	Mars		Direct	Semi-Direct	Stop-Over	Semi-Cycler	Reverse S-C	Cycler
1	HHH	HHH		618	232	224	191	180	187
2	HLH	HLH		261	130	122	154	98.5	129
3	HLL	HLL		216	102	79.8	99.5	89.5	117
4	HHH	HHH	T_M	425	223	220	185	179	159
5	HHH	HHH	A	513	198	165	149	155	161
6	HNN	HNN		317	130	103	94.9	95.4	98.5
7	HHM	MMM	I	243	201	179	140	130	120
8	HLL	HLL	A	194	168	123	103	106	97.0
9	HLL	MLL	I	96.3	77.9	55.5	75.1	51.2	70.8
10	HNN	NNN		206	114	86.9	78.6	75	75.9
11	HHH	HHH	AT_M	358	195	161	147	128	125
12	NNN	NNN		76.2	41.4	32.2	29.0	27.5	27.7
13	HNN	HNN	A	289	124	89.1	84.2	89.6	92.3
14	HHH	HHH	W	149	137	73.0	70.5	66.9	74.6
15	HHM	MMM	AI	194	168	123	103	106	97.0
16	HNN	MNN	I	133	104	76.3	68.0	61.1	60.3
17	HHH	HHH	WT_E	108	99.4	66.0	62.4	59.6	65.2
18	HLL	HLL	W	76.7	75.2	52.8	72.4	46.6	65.1
19	HLL	MLL	AI	91.9	76.7	47.6	66.5	47.4	69.9
20	HHM	MMM	AIT_E	162	142	103	86.4	88.7	78.9
21	HNN	MMN	AI	119	98.0	63.0	58.2	56.4	55.2
22	HNN	HNN	W	93.5	85.4	44.9	45.3	44.6	48.7
23	HNN	HNN	WT_E	74.8	67.9	41.5	43.2	39.7	42.5
24	NNN	NNN	AWT_E	29.9	27.3	15.8	15.3	15.8	16.0
25	NLN	NLN	AWT_E	23.6	20.9	12.9	14.3	12.5	14.9

[a] A = Aerocapture, H = H_2, I = ISPP, L = Low-Thrust, M = CH_4, N = NTR, T_E = Tanker to Earth, T_M = Tanker to Mars, W = Mars Water.

Propulsion System

Of the propulsion system technologies we examined, low-thrust (rows 2 and 3) provides the greatest reduction in propellant mass with the least amount of technology development. Even though our estimate of 5,000 s for the I_{sp} may be optimistic, a more conservative value of 3,000 s results in the same conclusions. (In the latter case the propellant mass savings is less than that of NTR upper stages, though development costs are still lower.) We note that the combination of low-thrust with ISPP is particularly effective (rows 9 and 19). The most significant reduction in propellant cost comes from the development of NTR launch vehicles (especially row 12), although developing a

NTR with enough thrust to lift a significant payload off the Earth may be quite a challenge.

We note that using propellants produced from water on Mars provides more propellant savings than the combination of aerocapture and ISPP, though these two options (water on Mars vs. aerocapture and ISPP) require about the same amount of technology development. If H_2 and O_2 can be produced from water on Mars, then sent to Earth orbit via a tanker, there is potential for a significant reduction in propellant mass (see row 17). This scenario depends on the availability of Martian water, however. Martian water excavation can also provide significant savings in payload requirements if this water is used for washing, food preparation, and drinking (that is, if it doesn't taste funny).

Sending ISPP methane and oxygen to Earth orbit from Mars requires less propellant mass than simply using terrestrial propellants at Earth (row 20 vs. row 15). However, the cost of an additional vehicle (an Earth Tanker) may offset the lower propellant cost. Also, sending propellant to Mars orbit (via a Mars tanker) instead of the surface does not reduce propellant cost significantly (with the exception of direct missions), and the added complexity probably outweighs the propellant savings.

The efficacy of Earth tankers hinges on the assumption that Mars launch vehicles and upper stages are reused at Mars. If instead these vehicles always had to come from Earth, there would be no propellant reduction from sending Martian produced propellants to Earth orbit. The additional propellant cost of sending (even empty) rockets to Mars exceeds any potential benefit.

The greatest propellant savings, independent of TRL, is given in Table 6. Here we note that NTR launch vehicles rank first, but their TRL is only 4. The second largest reduction in propellant cost comes from low-thrust (during the exploration phase) or from Mars water excavation (during the settlement phase). Aerocapture and ISPP also tie in propellant savings as aerocapture is more effective during the exploration phase while ISPP is better suited to the settlement phase. Of the propulsion technologies examined, propellant tankers (when used alone) offer the smallest reduction in propellant cost.

Now let us consider ranking the propulsion technologies while taking TRL into account. It seems clear that low-thrust propulsion, with its high TRL, provides the best combination of mass savings and technological development of any propulsion technology. In addition to low-thrust, NTR upper stages should be developed and used early on. Also the development of upper stage NTR's would logically support the subsequent development of NTR launch vehicles. It is still unclear if water will be readily accessible on Mars; thus, Mars water excavation has a relatively low ranking in Table 7. We note that determining the best combination of propellant cost and TRL is a highly subjective process, and different weightings of these two parameters can produce disparate results.

Table 6
GREATEST SAVINGS INDEPENDENT OF TRL

Rank	Propulsion Technology	Approx. TRL
1	NTR launch vehicles	4
2	Low-thrust[a]	8
2	Mars water excavation[b]	3
4	NTR upper stages	6
5	Aerocapture[a]	6
5	ISPP[b]	5
7	Refuel in orbit via a tanker	7

[a] Most effective during exploration phase (μ_{return} = 10%).
[b] Most effective during settlement phase (μ_{return} = 50%).

Table 7
GREATEST SAVINGS BALANCED AGAINST TRL

Rank	Propulsion Technology	Approx. TRL
1	Low-thrust	8
2	NTR upper stages	6
3	NTR launch vehicles	4
4	Aerocapture	6
5	ISPP	5
6	Mars water excavation	3
7	Refuel in orbit via a tanker	7

We note that in-situ resource utilization (ISPP or Mars water excavation) is most effective when the mass requirements at Mars are large (e.g. Direct missions and as μ_{return} increases). Also, the combination of Mars water excavation with propellant tankers (item six plus item seven in Table 7) provides significant propellant cost savings and can be more effective than low-thrust or NTR upper stages. So it is important to consider the cumulative impact of various combinations of the technologies listed in Tables 6 and 7.

Trajectory Architecture

Of the system architectures we examined, Cyclers, Semi-Cyclers, and Reverse Semi-Cyclers often require about the same propellant mass. Stop-Overs cost slightly more on average. Semi-Direct cost is often substantially higher than the Stop-Over cost. Finally, Direct scenarios always require the most propellant mass for a given mission. Cyclers have a noticeable mass penalty due to the higher V_∞ requirement. If a Cycler trajectory is found that has similar V_∞ *and* time-of-flight as other architectures, then this trajectory would almost always require the least propellant. (However, this low propellant cost must ultimately be balanced against the risk associated with hyperbolic rendezvous at Earth and Mars).

We note that Cyclers and Semi-Cyclers do not benefit as much as the other architectures from low-thrust propulsion. This is because we assume that low-thrust propulsion cannot meet the critical timing requirements of hyperbolic rendezvous at Earth. On the other hand, Stop-Over systems benefit greatly from low-thrust technology, especially if low-thrust is used to escape the HPO (see rows 3 and 8). Aerocapture is also well suited to Stop-Overs, as it reduces the orbit capture requirements at Earth and Mars. Indeed, the Stop-Over architecture seems to provide the best balance between propellant savings and trajectory complexity. However, if the hyperbolic rendezvous at Mars can be perfected, then Reverse Semi-Cyclers often require the least propellant. We note that the lowest propellant mass (and IMLEO) system is a Reverse Semi-Cycler with a combination of all the propulsive technologies (row 21). This mission would use NTR launch vehicles, low-thrust upper stages, NTR Transfer Vehicles, Aerocapture, parking orbits at Earth, flybys at Mars, and the hydrogen for the NTR upper stages would come from water excavated on Mars. We estimate that this Reverse Semi-Cycler system requires the greatest development cost and the lowest recurring costs.

Cost as Analytic Function of f and μ_{return}

In addition to examining human transportation to Mars at a single design point, we also explore how the propellant-optimal system varies with the support factor, payload return ratio, and V_∞. Figure 1 shows how the propellant mass increases linearly with the support factor and payload return ratio. This linear dependence suggests that simple, analytic expressions for propellant mass are available. Propellant cost equations for different architectures are provided in Table 8.

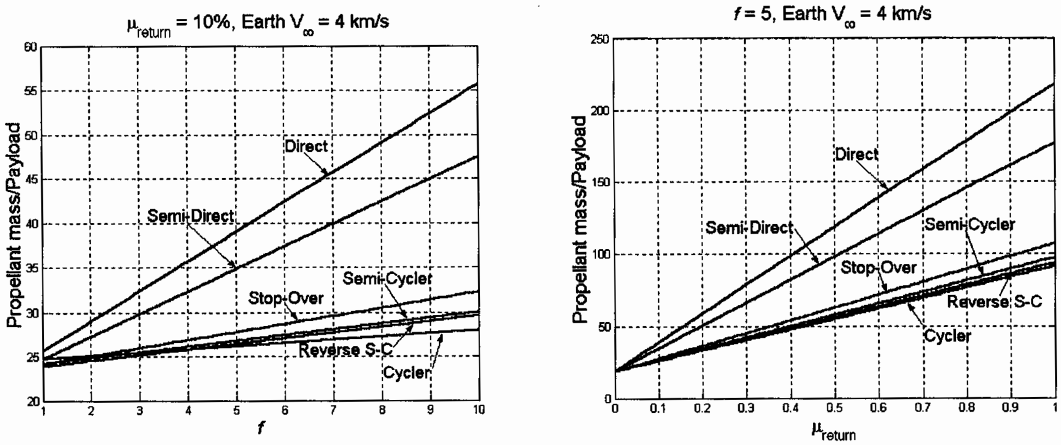

Figure 1 Propellant Cost as a Function of μ_{return} and f. Data Corresponds to Row 21 of Tables 4 and 5.

Table 8

ANALYTIC PROPELLANT COST EQUATIONS CORRESPONDING TO ROW 21 OF TABLES 4 AND 5

Architecture	Cost Equation
Direct	$18.9 + 33.8\mu_{return} + 33.1\mu_{return}f$
Semi-Direct	$18.9 + 33.1\mu_{return} + 25.0\mu_{return}f$
Stop-Over	$18.9 + 43.5\mu_{return} + 8.94\mu_{return}f$
Semi-Cycler	$18.9 + 46.6\mu_{return} + 6.40\mu_{return}f$
Reverse Semi-Cycler	$18.9 + 43.4\mu_{return} + 6.29\mu_{return}f$
Cycler	$18.9 + 54.7\mu_{return} + 3.57\mu_{return}f$

We note that when μ_{return} goes to zero, all six cost values approach the same value (18.9 in this example). This value is the propellant cost to send 1 mt to Mars on a low-energy trajectory and bring nothing back to Earth. Since there is no mass transfer from Mars to Earth ($\mu_{return} = 0$), no Transfer Vehicles are required and the architectures become indistinguishable (i.e. they all mimic Direct missions). While the direct linear dependence on propellant mass is a property of our particular cost model, this behavior may also be present in higher fidelity mass equations. Consequently, trade studies over a wide variety of payload requirements can be performed using analytic correlations that agree closely (if not exactly) with a more detailed model.

Propellant-Optimal Systems

Figures 2-4 show how the optimal (lowest propellant mass) mission scenario changes as a function of support factor (f) and trajectory (V_∞). Because the payload mass values are directly related to the return payload mass, the rank order of the system architectures does not vary with the payload return ratio (μ_{return}). For example, in Figure 2, the propellant optimal system at (V_∞ = 4 km/s, f = 5) is the Reverse Semi-Cycler, regardless of the payload return ratio (i.e. for μ_{return} = 0% to 100%). As the support factor decreases along the $V_\infty = 4$ line in Figure 2, the optimal system shifts from Reverse Semi-Cycler to Semi-Cycler to Semi-Direct, and as f increases, the propellant-optimal system shifts to Cyclers. We find that as the support factor increases, and the Transfer Vehicle becomes safer and more comfortable, the optimal system almost always shifts to Cyclers. A few notable exceptions are missions that employ low-thrust Transfer Vehicles (rows 3 and 8 of Tables 4 and 5), in which Stop-Overs require the least propellant mass regardless of f, μ_{return}, or V_∞. Also, when all of the propulsive technologies are used (row 25) the propellant-optimal system is always the Reverse Semi-Cycler.

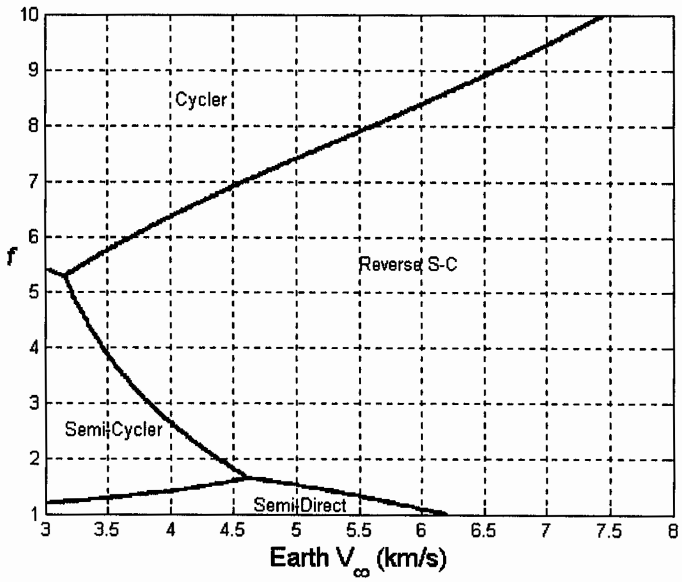

Figure 2: Optimal Scenarios Corresponding to Row 1 of Tables 4 and 5 where results are independent of μ_{return}.

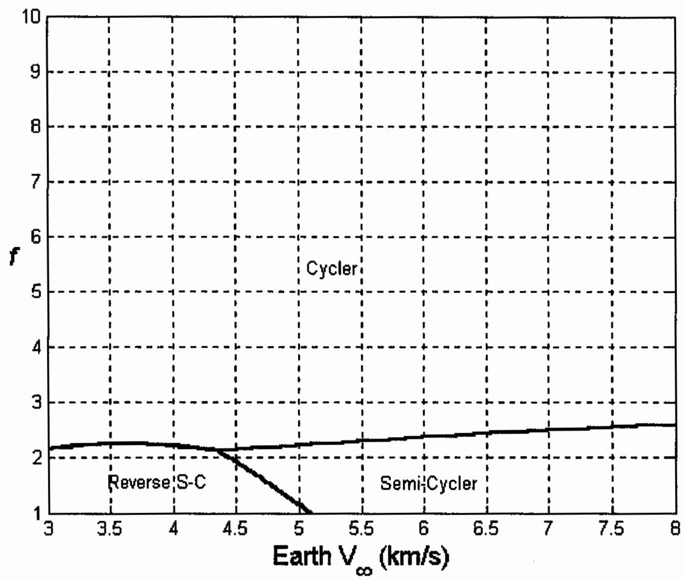

Figure 3: Optimal Scenarios Corresponding to Row 15 of Tables 4 and 5 where results are independent of μ_{return}.

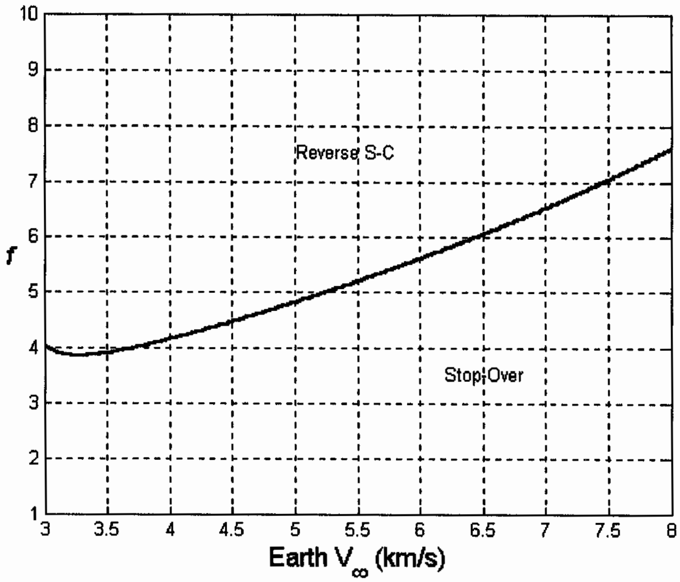

Figure 4: Optimal Scenarios Corresponding to Row 19 of Tables 4 and 5 where results are independent of μ_{return}.

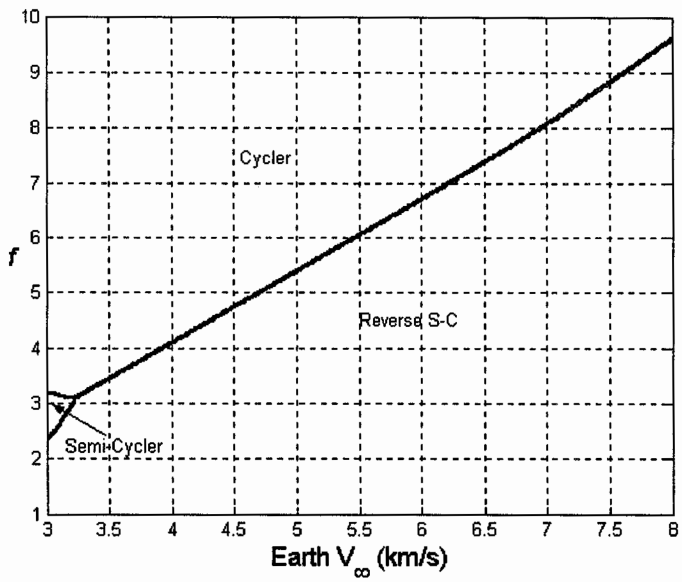

Figure 5: Optimal Scenarios Corresponding to Row 21 of Tables 4 and 5 where results are independent of μ_{return}.

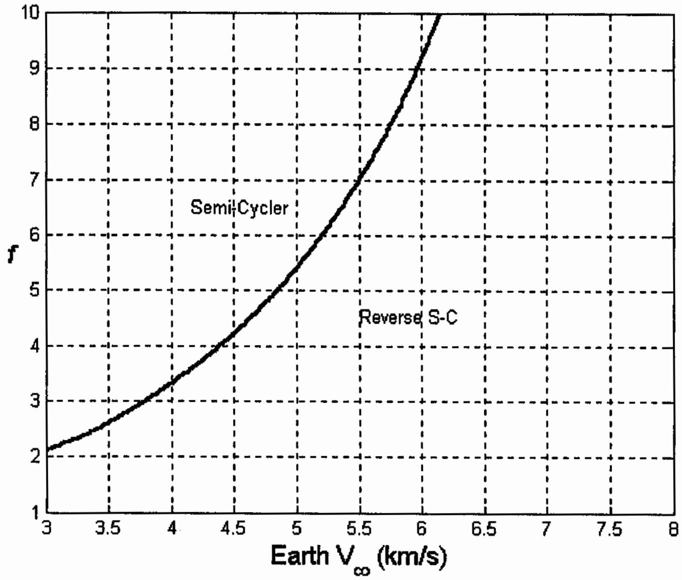

Figure 6: Optimal Scenarios Corresponding to Row 24 of Tables 4 and 5 where results are independent of μ_{return}.

Our analysis shows that small support factors generally suggest direct missions, while large *f* values imply cycling concepts. From these results, there is a trend which indicates that as the support factor grows from unity to high values, the optimal system shifts from Semi-Direct to Stop-Over to Cycler. This is a practical scenario for the evolution of the human Mars transportation system.

CONCLUSIONS

Landing a crew on Mars and returning them home safely would be a tremendous achievement, and the inaugural mission need not be done in the most efficient manner. However as trips to Mars become more routine, propellant efficient systems may lead to significant cost savings. Many insights into human missions to Mars are gained by examining the full range of the design space. We find that low-thrust propulsion systems provide significant savings for all mission scenarios. The largest reduction in propellant mass is achieved with the development of NTR launch vehicles. Also, the availability of water on Mars can dramatically reduce propellant cost if Martian produced propellant is sent to Earth orbit via a tanker.

While the first missions to Mars will most likely employ Direct or Semi-Direct architectures, these missions can evolve into Stop-Over scenarios by placing the Transfer Vehicle into a parking orbit upon Earth return. Cycling scenarios may then lead to propellant-optimal missions with the advancement of hyperbolic rendezvous. This mode of evolution for Mars transportation results in safe, comfortable and efficient missions for human beings to travel to and explore this new world

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NOTATION

CH_4	=	methane
f	=	support factor, $(m_{return} + m_{support}) / m_{return}$
g	=	standard acceleration due to gravity at Earth's surface, 9.80665 m/s^2
H_2	=	liquid hydrogen
HPO	=	high-energy (elliptical) parking orbit
I_{sp}	=	specific impulse, seconds
ISPP	=	in-situ propellant production (methane/oxygen)
LCO	=	low circular orbit
m	=	mass, mt (metric tons)
n	=	number of rocket stages
NTR	=	nuclear thermal rocket
NEP	=	nuclear electric propulsion
O_2	=	liquid oxygen
SEP	=	solar electric propulsion
TRL	=	technology readiness level
V_∞	=	hyperbolic excess speed, km/s
ΔV	=	instantaneous change in velocity, km/s
μ	=	mass fraction

Subscripts

0	=	initial
as	=	aeroshell
f	=	final
inert	=	inert propulsion system mass, $\mu_{inert} = m_{inert}/(m_{inert} + m_p)$
outgoing	=	payload mass that lands on Mars
p	=	propellant
pay	=	propulsion payload
peri	=	periapsis
return	=	payload mass that returns to Earth
stage	=	stage of a rocket
support	=	mass that may (in principle) be left in space
surf	=	surface of a planet

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