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LOW-THRUST ROUNDTRIP TRAJECTORIES TO MARS WITH ONE-SYNODIC-PERIOD REPEAT TIME*

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We investigate the use of nuclear-electric propulsion for human missions to Mars. Employing a patched-conic solution as our initial guess, low-thrust trajectories are designed via a continuation method. We determine the propulsion requirements for a reusable human transportation system, in which a single vehicle makes a round trip between the Earth and Mars within a synodic period, thus allowing a mission at every launch opportunity. We find that the acceleration required for a typical mission exceeds current technology by an order of magnitude. However, technology improvements in the next decade could make such a mission feasible.

INTRODUCTION

More than a thousand human-to-Mars scenarios have been studied in the past fifty years.[1] Among the recent proposals, the most elaborate architectural study is NASA's Design Reference Mission[2,3] (DRM), which is based (in part) on the Mars Direct concept envisioned by Robert Zubrin.[4] In Mars-mission planning, nuclear-based propulsion is a double-edged sword that creates technological advantages and political disadvantages. In Mars Direct, for example, chemical propulsion is favored over nuclear-thermal propulsion (NTP) as a baseline method for trans-Mars injection (TMI), despite significant mass savings of NTP that essentially reduce the injected mass to low Earth orbit by half. Zubrin argues that a lack of political support for the development of NTP could potentially delay the first human landing on the Red Planet.[4] In 2003, however, NASA has established the Prometheus Program to develop nuclear space propulsion. The program's near-term objective is the demonstration of hundred-kilowatt-class nuclear electric propulsion (NEP) for robotic missions, such as the Jupiter's Icy Moons Orbiter (JIMO).[5,6] If NEP can be scaled up to megawatt levels, the technology will bring unique advantages in human exploration of Mars.[7-18]

Nuclear Electric Propulsion

For a given ΔV, electric propulsion requires much less propellant than chemical propulsion. However, this mass saving is partially offset by the greater dry mass of

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electric-propulsion. In the case of the JIMO spacecraft, for instance, about a half of the spacecraft mass at Earth departure is inert mass, including the nuclear reactor, radiators, power-processing units, structure, and thrusters.[19] In long-duration missions (e.g. to the outer planets) electric-propulsion could achieve shorter time of flight (TOF) than chemical propulsion. On the other hand, electric propulsion requires significantly longer TOF for orbital transfer from LEO to Earth escape. Electric propulsion does not necessarily reduce TOF for Mars missions, but the technology is often considered for cargo flights, where the TOF constraint may be relaxed.[3,11,17]

One of the disadvantages of electric propulsion is its low level of thrust. Even with multiple ion engines, the JIMO spacecraft produces thrust on the order of 2 newtons.[19] Human missions require much greater payloads than robotic missions, so the thrust must be increased to meet the acceleration requirement; Brown and Coates, for example, calculate that a human mission to Mars would require 600 ion thrusters.[7] The number of thrusters, and mass thereof, can be significantly reduced if the thrusters have higher thrust density. For instance, a single magnetoplasmadynamic (MPD) thruster produces thrust on the order of tens of newtons, as opposed to hundreds of milli-newtons for an ion thruster. Because of its high capability, the MPD thruster is long considered as a leading candidate in an NEP vehicle for human Mars missions. Unlike a space-rated ion thruster, however, MPD thrusters are still in the research phase in laboratories. (In 1996 Japanese Space Flyer Unit successfully flight-tested a quasi-steady MPD thruster.[20]) Nevertheless, the MPD thruster is the only electric propulsion device which promises the capability of operating at megawatt power levels.[21]

Both solar and nuclear electric propulsion have been proposed for human missions to Mars. However, because solar power decreases proportionally to the inverse-square distance from the Sun, Mars missions using solar electric propulsion (SEP) tend to have large solar arrays. For example, the SEP vehicle proposed by (the Russian Space Corporation) Energia has solar arrays spanning 700 m and a total mass of up to 600 metric tons (mt) in Earth orbit.[22] (Energia opted for SEP over NEP due in part to the environmental concerns.)[23] We note that in human missions, electric propulsion (NEP or SEP) eliminates the need for a separate power source for the crew's life-support systems.

Reusable Spacecraft

The concept of a reusable space vehicle continues to appear in Mars mission proposals. For instance, the DRM considers solar electric propulsion as a reusable space elevator, which transfers most of the payload from the low-Earth orbit (LEO) to high Earth eccentric orbit (HEEO).[3] Because this process of raising orbital energy takes many months, the crew is placed into the HEEO (directly and quickly) via chemical launch vehicle. The trans-Mars injection can then be achieved by a small chemical burn. The SEP vehicle left in HEEO spirals down to LEO to be refurbished for the subsequent mission.

Another system component typically considered for reuse is a crew cabin. For example, a spacecraft may be captured into a parking orbit at both Earth and Mars without landing on the surfaces.[24-26] Such a vehicle may shuttle back and forth between the two planets, whereas the crew transfers between the interplanetary vehicle and

planetary surfaces via small "taxis." Because the expensive interplanetary vehicle is repeatedly used and only a small taxi is launched from and landed on the planetary surfaces, such a mission scheme is potentially cost effective.

The principle of not accelerating (or decelerating) the bulk of the transported mass is pushed to its extreme, when the interplanetary vehicle is not captured into a parking orbit. Such trajectories are referred to as cyclers.[27-35] Sometimes the term "full cycler" is used for an interplanetary vehicle performing flybys at both planets to distinguish it from the "semi-cycler,"[36] where an interplanetary vehicle is captured into the parking orbit either at Mars or at Earth, while performing flybys at the other planet.

A classic example of a cycler is the Aldrin cycler.[30] The Aldrin cycler has a repeat time equal to the synodic period of Earth and Mars (2 1/7 yrs). The gravity assist at Earth rotates the orientation of the orbit, so that the interplanetary transfer vehicle continues to encounter both planets in every synodic opportunity. The Aldrin cycler has one transit leg (i.e. from the Earth to Mars or vice versa) that has a much shorter time of flight than the other. So, having two cycler vehicles allows short transits in both directions (inbound and outbound) for all mission opportunities. The drawback of the Aldrin cycler, however, is its high V_∞ with respect to the planets, especially with Mars. Nock[27] calculates that the wet mass of "small" Mars taxi for the Aldrin cycler can be as big as 240 mt in Mars orbit, and a separate vehicle is required from the Martian surface to Mars orbit. It turns out that the estimated overall cost (in dollars) would be less, if the interplanetary vehicles are captured at both planets, as the case for the "stopover cycler."[26,27]

Since the 1960's, researchers have looked for ballistic or nearly ballistic cyclers (including semi-cyclers) with low flyby V_∞. Because each interplanetary vehicle represents a significant cost in development and maintenance, the number of vehicles should be as small as possible. However, although several new cyclers with low V_∞ have been discovered, all such cases require three, four, or more vehicles to ensure a mission in every synodic opportunity. Thus, in a search for ballistic or nearly ballistic cyclers, reductions in flyby V_∞ have always increased the number of interplanetary vehicles.

Low-thrust trajectories may change everything. Instead of increasing the number of vehicles, the propellant expenditure might be reduced by employing the high specific impulse of NEP.

MISSION DESIGN

Mission Architecture

The most significant technology development assumed in our mission architecture is the two reusable NEP vehicles with a jet power capability of 7.5 MW each; one orbits the Earth and the other shuttles between Earth and Mars (Fig. 1). These vehicles never enter the atmosphere at either planet, and are continuously reused in a series of missions. Refurbishment and maintenance for both vehicles occur every two years. An Earth-spiral tug vehicle spends approximately 200 days to transport payload from LEO to the interplanetary transfer vehicle temporarily captured in HEEO. The crew then rendezvous with the interplanetary vehicle departing for Mars. Approaching the Red Planet with zero V_∞ (i.e. $V_\infty = 0$ at arrival), the crew taxi enters the atmosphere to land on the surface of

Mars. Meanwhile, the interplanetary transfer vehicle spirals about Mars inward to reach a loose parking orbit. After a few weeks (or only days if a separate habitat is unavailable) of surface exploration, the crew is launched from the Martian surface via Mars taxi to rendezvous with the transfer vehicle in the parking orbit. After the return flight of one year (on average) the transfer vehicle arrives at Earth with zero V_∞, when the Earth-return capsule enters the Earth atmosphere to return to the surface. The interplanetary transfer vehicle then gradually decelerates inward to reach HEEO to be refurbished for the next mission. The crew's total trip time is two years on average.

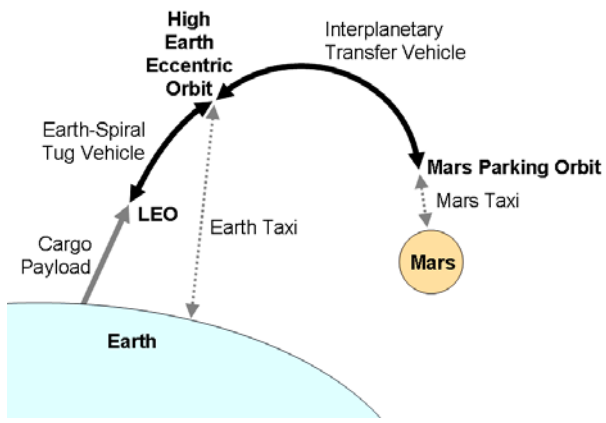


Figure 1 Schematic of Mission Sequence

Planetary stopovers allow for exploration at Mars and refurbishment at Earth. However, because each round trip is completed within a synodic period, more time at each planet means less time on the interplanetary transfers, and more thrust (and thus more power and dry mass) is required. We use a minimum stay time of 30 days at each planet; the one-way transfer time is thus 361 days on average. (During such long durations in interplanetary space, artificial gravity and radiation shielding may be required. The DRM vehicle does not employ artificial gravity, since its one-way TOF is limited to 6 months.)

We now discuss two types of masses in our mission architecture: the "synodic payloads" that are launched to LEO every mission and the reusable hardware that are never brought down to the ground.

Synodic Payload

The initial mass at Earth orbit is determined, in part, by the crew size. The choice of crew size is both technical and philosophical, involving a trade between costs, risks, and scientific returns. In principle, Mars missions can be designed for any number of astronauts ranging from one to an arbitrarily large number. (Wernher von Braun's *Das Marsprojekt* has a crew of 70, where 37,000 mt of mass in LEO is delivered by 46 shuttles making 950 flights.[37]) In our analysis, we choose a crew size of four (all to be

landed on the Martian surface), as a reasonable starting point for a conceivable mission in a 10-20 year time frame.

Figure 2 shows the logistics of payload transportation via the interplanetary transfer vehicle and the Earth-spiral tug vehicle. (The hardware in Fig. 2 does not necessarily represent actual configurations). Our interplanetary vehicle transports a total of about 85 mt of payload to Mars. Of which, 35.3 mt is a Mars "taxi," consisting of the small (6-mt) habitable capsule, two ascent stages, and one landing stage. The Mars taxi is fully fueled; it does not employ in-situ propellant production. We assume that this 6-mt capsule has a payload capability for a few days of surface activities for 4 astronauts. For extended surface activities, however, a larger habitation module and surface rovers are required, but they are not included in our mass calculations. After a few weeks on the Mars surface, the crew rendezvous with the interplanetary vehicle in Mars parking orbit. The Mars capsule is then discarded, along with the waste produced during the first half of the journey. The crew spends 30 days in the vicinity of Mars.

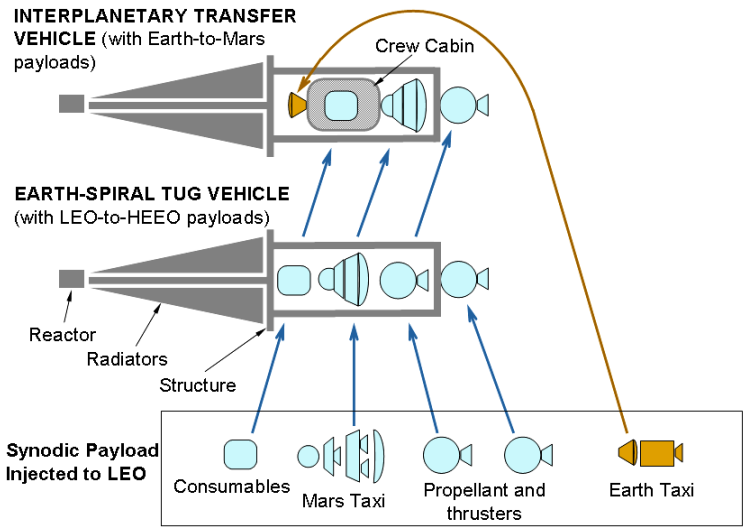


Figure 2 Delivery of Synodic Payload from LEO to Transfer Vehicle via Tug Vehicle

After Mars departure, the payload of the interplanetary vehicle is down to about 50 mt. Of which, 28 mt is budgeted for the crew cabin. Consumable mass for each mission is estimated based on a guideline of 5 kg per day per person. Our Earth return capsule is 6 mt, comparable to the Apollo command module. Of the payload on the interplanetary vehicle, only the crew cabin may be reused. During the Earth stopovers, a new Mars taxi, consumables, Earth-return capsule, propellant, and a new set of thrusters are supplied.

Our crew makes a short (a few days) transit via an Earth taxi and rendezvous with the interplanetary vehicle in HEEO. The 6-mt crew capsule remains attached to the interplanetary vehicle for the 2-yr round trip and serves as a crew-return capsule upon an Earth arrival. (As mentioned earlier, synodic payload refers to the sum of all payloads delivered to LEO in every mission year.)

Reusable Hardware

Because trajectory design is coupled with the hardware capabilities, we need to estimate the propulsion hardware mass for a given power. By propulsion hardware, we mean the reusable portion of vehicles, including the nuclear reactor, shielding, radiators, the structure, and the power-processing unit, but excluding the crew cabin (Fig. 2). Figure 3 shows our hypothetical model for the vehicle hardware mass as a function of electric power. As we see in Fig. 3, we estimate that the hardware mass increases linearly for the range of power considered. The hardware-mass-to-power ratio decreases as the power increases, so our model favors vehicles with higher power and greater mass. Our model predicts similar hardware masses described in Ref. 38, in which Falck and Borowski analyze missions that employ a multi-megawatt NEP vehicle. Falck and Borowski predict this technology will be available in 15-20 years.

Tankage mass is estimated to be 15% of the (hydrogen) propellant. All MPD thrusters are replaced every two years; we allocate 4 mt per vehicle (a total of 8 mt) of thrusters to refurbish the electric propulsion system.

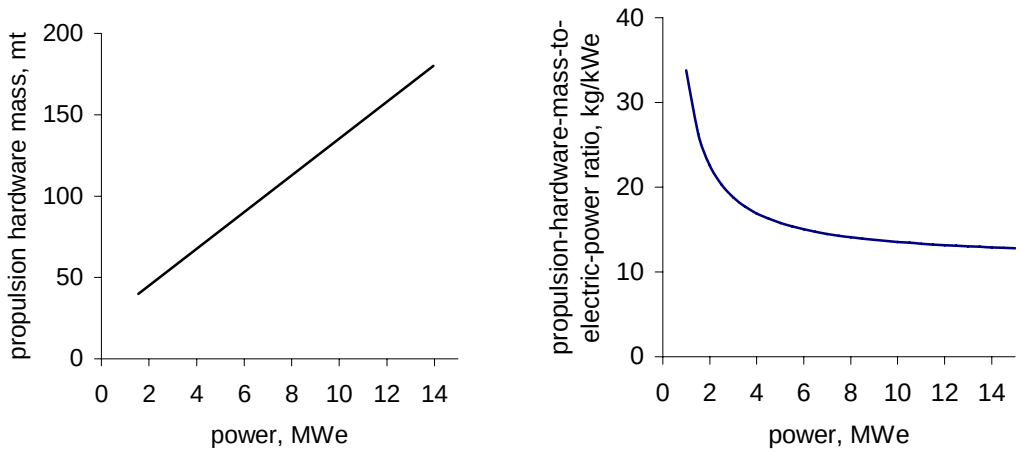


Figure 3 Hypothetical Model for Propulsion Hardware Mass.

NUMERICAL RESULTS

Interplanetary Trajectory

The synodic period of the two planets is approximately 2 1/7 years, and both planets return to their original positions after approximately 15 years. Thus, design of 7 round trips in any 15-yr-time-period (e.g. 2009–2024 in our case) gives an estimate of trajectory characteristics and hardware requirements of the mission architecture.

Using conic trajectories as our initial guesses, we design low-thrust trajectories with the Gravity-Assist Low-thrust Optimization Program (GALLOP),[39-41] in which low-thrust trajectories are modeled as a series of small impulsive ΔV maneuvers.[42] The current version of GALLOP has three objective functions, where users may choose to maximize

the final mass, minimize the initial mass, or minimize the time of flight. However, our software, in its present form, is not set up to minimize the total propellant re-supplied over seven missions. Instead, we independently minimize the mass at Earth departure for each roundtrip, then concatenate the solutions to span seven missions.

Figure 4 shows a representative trajectory plot of a single round trip. In this mission, the Earth departure date is February 23, 2022, and the crew spends 347 days on the way to Mars. After 30 days in the vicinity of Mars, the crew begins a 410-day return trip, in which the trajectory makes one complete revolution about the sun. All departure and arrival V_∞ are constrained to zero with respect to the planets. The trajectory plot in Fig. 4 has undesirably low perihelion values during the return flight. Solar radiation is hazardous to the astronauts, and trajectories with such a low perihelion may require more radiation shielding.

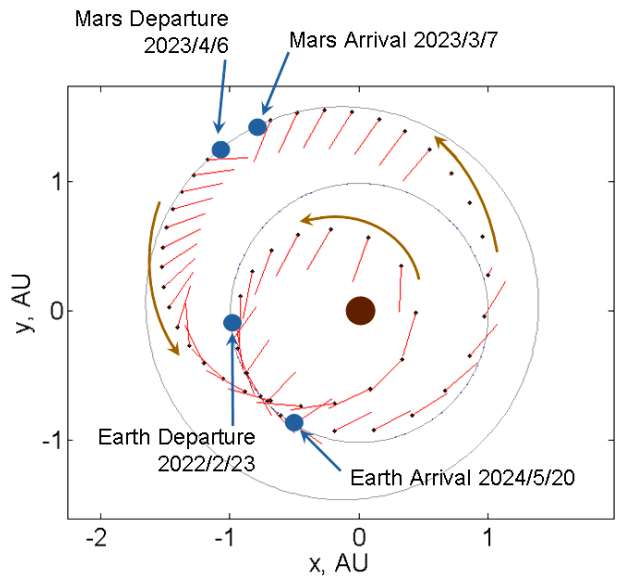


Figure 4 Trajectory Plot of an Earth-to-Earth Round Trip for an Earth Launch in 2022. The Low-Thrust Trajectory is Modeled as a Series of Impulsive ΔV Maneuvers Indicated by the Lines at Each Segment.

Table 1 shows the summary of trajectories for the 15-year period starting with the Earth departure on June 9, 2009 (E1). While all Mars encounter dates were allowed to move freely, Earth departure and arrival dates are frozen. So while our solutions patched together are feasible (i.e. flyable trajectory), further improvement is possible by solving the entire 15 years together.

In our concatenated solution in Table 1, the one-way TOF ranges between 279 and 436 days, with an average being 361 days. The mission with the Earth launch in 2009 (E1) has the longest roundtrip duration (from Earth departure to Earth arrival) of 849 days, whereas the mission with the Earth departure in 2018 (E9) has the shortest mission duration of 710 days, including the 30-day Mars stopovers. We see in Table 1 that for a given round trip, the Earth-to-Mars leg always takes less time than the Mars-to-Earth leg.

The lower bound for the planetary stopover was set to 30 days, and optimal solutions are always on the TOF bounds.

TABLE 1
SUMMARY OF TRAJECTORIES: PRELIMINARY RESULT

stop-over	arrival[a] date, yyyy-mm-dd	planet	inter-planetary TOF, days	propellant expen-diture, t	arrival mass t	net mass gain at planet[b], t	stay time,[c] days	departure mass, t	departure[d] date, yyyy-mm-dd
E-1	—	Earth	—	—	—	—	—	346.6	2009-06-09
M-2	2010-06-21	Mars	382.2	33.0	313.6	-47.1	30.0	266.5	2010-07-21
E-3	2011-09-30	Earth	436.4	52.1	214.3	128.8	30.0	343.1	2011-10-30
M-4	2012-09-03	Mars	308.4	34.2	309.0	-45.7	30.0	263.3	2012-10-03
E-5	2013-11-07	Earth	399.9	49.9	213.4	108.1	30.0	321.5	2013-12-07
M-6	2014-09-30	Mars	296.9	24.0	297.5	-45.4	30.0	252.1	2014-10-30
E-7	2015-12-09	Earth	405.4	41.3	210.7	108.1	30.0	318.8	2016-01-08
M-8	2016-10-14	Mars	279.4	24.1	294.7	-45.1	30.0	249.6	2016-11-13
E-9	2017-12-18	Earth	400.4	39.3	210.3	128.4	30.0	338.6	2018-01-17
M-10	2018-11-05	Mars	292.6	34.9	303.7	-45.3	30.0	258.4	2018-12-05
E-11	2020-01-06	Earth	396.3	45.5	212.9	130.3	30.0	343.2	2020-02-05
M-12	2020-12-17	Mars	316.3	36.7	306.4	-45.8	30.0	260.6	2021-01-16
E-13	2022-02-23	Earth	402.8	47.1	213.5	137.6	30.0	351.1	2022-03-25
M-14	2023-03-07	Mars	347.0	37.0	314.0	-46.4	30.0	267.6	2023-04-06
E-15	2024-05-20	Earth	409.6	53.1	214.5	[132.1]	[30.0]	—	—

a. Arriving with V_∞ = 0 km/s with respect to a planet, entering a planetocentric (inward) spiral.
b. The net mass gain is departure mass minus arrival mass.
c. The lower bound of the stay time is 30 days. (The concatenated optimization found solutions at the TOF bounds.)
d. Departing with V_∞ = 0 km/s with respect to a planet, exiting a planetocentric (outward) spiral.

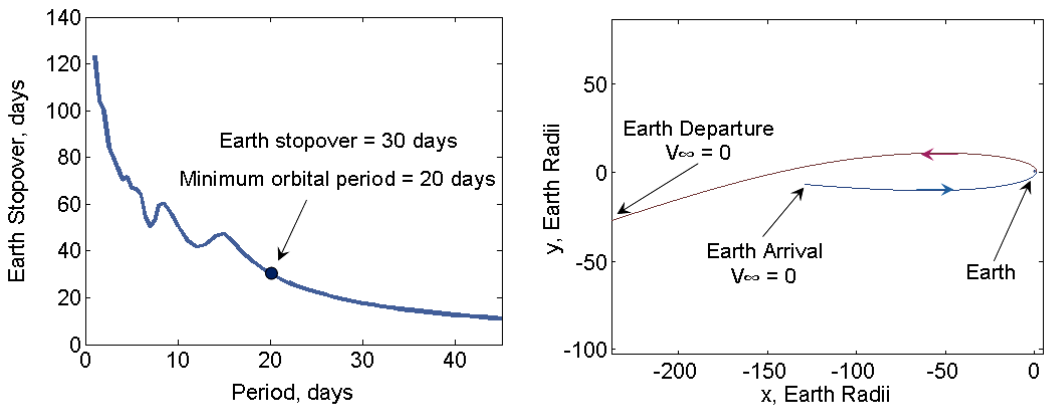
Propellant expenditure also varies from one transfer to another. The mission with the Earth departure in 2022 (E13) has the highest propellant expenditure of 97.9 mt and the mission with the Earth departure in 2016 (E7) has the lowest propellant expenditure of 71.2 mt. (Here, we include the 3.9 mt of propellant used for the 30-day stopovers at Mars and Earth.)

A net mass gain is the departure mass minus the arrival mass, where the planetary departure and arrival is defined as when the interplanetary vehicle begins or ends an interplanetary transfer with zero V_∞ with respect to a planet. At Mars stopovers, the net mass gain is negative. This mass loss at Mars includes the 35-mt Mars taxi, propellant used during Mars capture and escape, and the discarded waste produced during the Earth-Mars transfer. The variation in the mass loss at Mars comes from the waste produced by the crew: 5 kg per day per person multiplied by four astronauts and by the number of days from Earth to Mars. At Earth stopovers, the mass gain is much greater than the mass loss. The mass gain at Earth includes the payload for the subsequent mission: the Mars taxi, consumables, the Earth-return capsule, propulsion refurbishments (i.e. propellant, tank, and thrusters). The mass loss includes payloads from the previous mission: the Earth-return capsule, used consumables, the propellant used during Earth capture and escape, the empty propellant tank, and used thrusters.

We see in Table 1 that the Earth arrival mass ranges between 210 and 215 mt; the variation coming from the empty tangake mass assumed to be 15% of the propellant used for a given mission. We did not stage propellant tanks, but staging may be desirable considering the magnitude of the tankage mass for human missions. If the vehicle hardware mass for a given power is smaller than our model, a similar trajectory with a lower mass requirement can be obtained. Our thrust-to-weight ratio is baselined to the technologies available in a fifteen to twenty year time frame.[38]

Planetary Captures and Escapes for the Interplanetary Vehicle

In this section, we estimate the TOF requirements for the planetocentric trajectories for the transfer vehicle. As in the heliocentric trajectories, our analysis is based on the two-body model. For the planetocentric trajectories, our estimate is probably accurate only to one significant figure, as the perturbation from the sun is significant in a loose parking orbits considered for our mission.



(a) Stopover Duration vs. Orbital Period

(b) Trajectory plot for 30-day Stopover for E13 Earth Capture and Escape (the Most Difficult Opportunity)

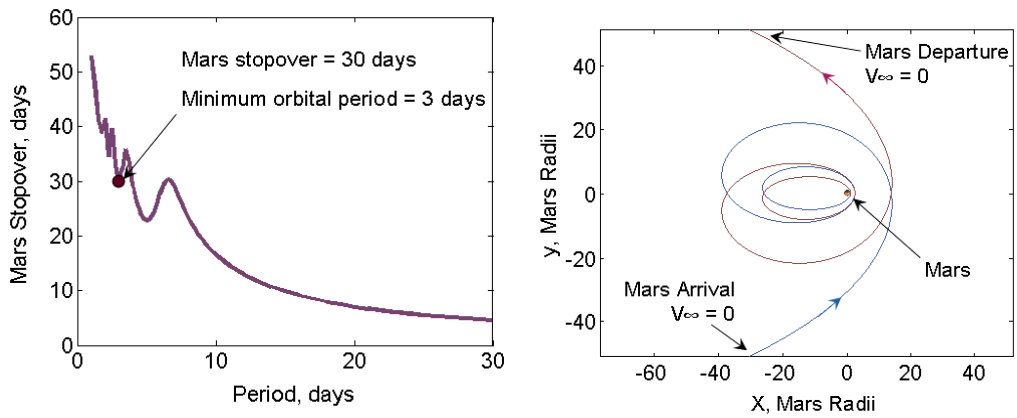
Figure 5 Earth Capture and Escape

Figure 5 (a) shows the Earth stopover duration plotted against the minimum geocentric orbital period achieved by our interplanetary vehicle. During the 30-day stopover, the interplanetary transfer vehicle gradually reduces its orbital energy to momentarily reach a "parking orbit" to be refurbished and then gradually increases orbital energy again to reach $V_\infty = 0$. We see in Fig. 5 (a) that the shorter the stopover, the larger the minimum orbital period that can be achieved. For example, if an Earth parking orbit has the period of 1 day, the required stopover is at least 120 days. On the other hand, if the Earth parking orbit can be very loose with the period of 45 days, then the required stopover is only 11 days.

In our mission, the Earth stopover duration is 30 days at minimum. From the plot in Fig. 5 (a), we see that the minimum orbital period we can achieve is 20 days. During the Earth stopover for E13 in 2022, for example, the interplanetary transfer vehicle spends 7.5 days to reduce its orbital energy to the parking orbit and only 22.5 days to get from the parking orbit to an Earth escape. The transfer vehicle spends 3 times more time to raise the orbital energy than to reduce it, because the vehicle mass is 1.6 times heavier after the E13 refurbishment.

As the trajectory plot in Fig. 5 (b) shows, the interplanetary transfer vehicle does not have the time to make revolutions about the Earth. (Although if the vehicle stops thrusting when it reaches its perigee, it would orbit in the 20-day eccentric orbit with a perigee of 300 km.)

The cargo payload is delivered to the interplanetary vehicle at perigee. (Our assumption of continuous thrust during the 30-day stopover may pose a problem for thruster replacement.) We see in Fig. 5 (b) that the trajectory to achieve an Earth escape is much longer than the one that reduces orbital energy, because it takes longer time to accelerate the heavier interplanetary vehicle after refurbishment (i.e. 22.5 days as opposed to 7.5 days).



(a) Stopover Duration vs. Orbital Period

(b) Trajectory plot for 30-day Stopover for M14 Mars Capture and Escape (the Most Difficult Opportunity)

Figure 6 Mars Capture and Escape

We note that the Earth-return capsule will separate from the interplanetary vehicle right after an Earth arrival to limit the time spent in the radiation belt. Likewise, our crew rendezvous with the interplanetary vehicle just prior to an Earth escape. The trajectories for the Earth taxi are not shown in the figure.

Figure 6 shows similar plots for Mars capture and escape. However, we see that the required stay time is much shorter to achieve a desired orbital period at Mars than at Earth. Within the same stopover duration of 30 days, the period of the Mars parking orbit is as low as 3 days. Because the gravity of Mars is much smaller than that of Earth, it takes less time to spiral into a parking orbit at Mars than at Earth. (For example, the

powerful spacecraft described in Ref. 18 takes 30 days from LEO to an Earth escape, and only 7 days to spiral into a low-Mars orbit after the planetary arrival.) Figure 6 (b) shows that 30 days at Mars is enough to make more than 4 revolutions about the planet.

At Mars arrival, when the interplanetary transfer vehicle begins its inward spiral, the crew taxi separates from the interplanetary transfer vehicle and lands on the Martian surface. The crew will rendezvous with the interplanetary transfer vehicle after two weeks of surface stopover when the transfer vehicle reaches 300-km periapsis, corresponding to the lowest periapsis in Fig. 6 (b). (Alternatively, we may choose to extend the surface stay time during 30 days in the vicinity of Mars and to keep the period of Mars parking orbit high. Our Mars taxi has the ΔV capability for a Mars-escape, so all ranges of orbital period can be used from an energy point of view.) After the Mars taxi delivers the crew to the interplanetary vehicle, the capsule is jettisoned.

Because the mass loss while spiraling at Mars (i.e. 3.9 mt of propellant) is less than 2% of the total transfer vehicle mass, the times for inward and outward spirals are approximately the same (i.e. 15 days each).

Earth-Spiral Tug Vehicle

The Earth-spiral tug vehicle and the interplanetary transfer vehicle rendezvous with each other at the periapsis (at 300 km) of the HEEO parking orbit. Within two years, the interplanetary transfer vehicle makes a round trip between HEEO and Mars parking orbit, whereas the Earth-spiral tug vehicle spirals down to LEO and back up to HEEO.

The tug vehicle is assumed to have the same hardware capability as the interplanetary vehicle, so transporting payloads from LEO to HEEO every two years is relatively easy. We estimate that it takes approximately 200 days to transfer synodic (cargo) payload from LEO to HEEO, and approximately 80 days from HEEO to LEO when the spacecraft mass is much smaller. (These TOFs are rough estimates, as the spiral trajectory used for this calculation does not have the same periapsis value as the transfer vehicle.)

Synodic Payloads

The synodic payload injected to LEO includes consumables, Mars taxi, and the thruster replacements for both the interplanetary transfer vehicle and the Earth-spiral tug vehicle. Consumable mass and wet propellant tank mass for the interplanetary vehicle differs from one mission to another. Figure 9 shows synodic payload injected to LEO for all seven opportunities. We see that the mission in 2022 requires the largest synodic payload mass of 233 mt, a delivered mass to LEO equivalent of two Saturn V vehicles (each delivering 120 mt to LEO). In practice, the Earth taxi is launched separately, so that the crew rendezvous directly with the already-refurbished interplanetary vehicle just prior to an Earth departure. The easy mission years, such as 2014 and 2016 are good opportunities to deliver extra cargo, such as a large surface habitation module and rovers for extended surface exploration.

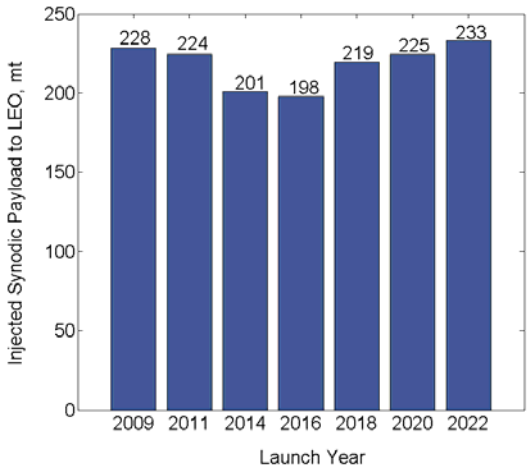


Figure 7 Synodic Payload Injected to LEO Every Mission Opportunity

TABLE 2
VEHICLE AND PAYLOAD CHARACTERISTICS

NEP system characteristics	
Electric power, MWe	11.6
Specific impulse, s	10204.0
Thrust, N	150.0
Efficiency, %	64.5
Mass Breakdowns (mt)	
Interplanetary Transfer Vehicle	
Hardware	153.8
Propellant	71.2–97.9
Propellant tank	10.7–14.7
Thruster	4.0
Crew cabin	28.0
Consumables	14.2–15.7
Earth-Spiral Tug Vehicle	
Hardware	153.8
Propellant	37.9–40.5
Propellant tank	5.7–6.1
Thruster	4.0
Mars Taxi (35.3 mt)	
Capsule	6.0
1st ascent stage	18.1
2nd ascent stage	2.8
Landing stage	3.2
Heat shield/parachute	4.6
Earth Taxi (14.7 mt)	
Capsule	6.0
Upper stage	8.7

Table 2 shows the summary of hardware characteristics. Both the interplanetary transfer vehicle and the Earth-spiral tug vehicle have common NEP engines: the thrust is 150 N and the exhaust velocity is 100 km/s (for an I_{sp} = 10,200 s). For a typical efficiency for a multi-megawatt NEP vehicle with MPD thrusters (of 64.5%),[38] this equates to an electric power of 11.6 MWe. This combination of thrust and exhaust velocity was obtained using a method described by Zola in Ref. 43. Using Zola's heuristic method, we minimized the total synodic payload mass (i.e. the sum of all synodic payload injected to LEO shown in Fig. 2).

CONCLUSIONS

Ballistic and nearly ballistic cycler systems with low ΔV and thus low propellant mass discovered to date require three or more interplanetary vehicles to ensure missions in every opportunity. In this paper, we presented a system, which allows the crew to come home within two years, with only one interplanetary vehicle, providing potential cost savings in both infrastructure and operation.

Our architecture is not a cycler in a traditional sense—the interplanetary vehicle flies in low-thrust trajectories and is captured at Earth and Mars. For this reason, we achieve a low propellant expenditure for the interplanetary vehicle via nuclear electric propulsion and for the taxis via low ΔV requirements at both planets. Consequently, the total mass requirement for our mission is relatively low compared to other scenarios with reusable or one-time-only interplanetary vehicles, as the injected mass to LEO is 200-240 mt per mission. (NASA's Design Reference Mission has a LEO mass on the order of 420 mt.) Our architecture calls for a development of two identical NEP vehicles: the interplanetary transfer vehicle and the Earth-spiral tug vehicle. The nuclear electric propulsion engine assumed in our study is based on technologies available in 15-20 years. If such NEP vehicles can be developed, human Mars missions can be achieved without nuclear thermal propulsion, aerocapture of heavy vehicles, or in-situ propellant production, all common in many mission proposals.

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