AAS 05-269

MARS EXPLORATION VIA EARTH-MARS SEMI-CYCLERS*

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We present an architecture for the human exploration of Mars. This architecture is characterized by the use of parking orbits at Earth and gravity assists at Mars. An interplanetary transfer vehicle cycles from Earth orbit to Mars flyby and back, eliminating the need to launch transfer vehicles from the surfaces of Earth and Mars. Necessary developments for an Earth-Mars semi-cycler mission (beyond traditional architectures) include reusable transfer vehicles and rendezvous during planetary flyby. When compared to scenarios similar to NASA's Design Reference Mission, the Earth-Mars semi-cycler requires 10%–35% less injected mass to low-Earth orbit once in operation.

INTRODUCTION

The allure of people traveling to Mars has been the inspiration for numerous mission proposals.[1–32] While many Mars exploration plans emphasize the benefits of advanced propulsion concepts (e.g. nuclear propulsion, aerocapture, or in-situ propellant production), a change in system architecture can also significantly reduce the mass that must be launched from the Earth's surface. We differentiate Mars exploration architectures by the placement of the interplanetary transfer vehicle at Earth or Mars. For example, NASA's Design Reference Mission[21, 22] places the transfer vehicle into a parking orbit at Mars arrival (which we call a semi-direct architecture). Other ideas include parking orbits at both Earth and Mars (stop-over),[23, 24] flybys at both Earth and Mars (cycler),[25–29] a flyby at Earth and a parking orbit at Mars (Mars-Earth semi-cycler),[30, 31] and a flyby of Mars with limited stay time (FLEM).[32] The Earth-Mars semi-cycler architecture specifies a parking orbit at Earth and a flyby of Mars with relatively short interplanetary transfers and a long exploration time at Mars. (The inspiration for this architecture derives from the Mars-Earth semi-cycler and FLEM concepts.) The key mass savings for Earth-Mars semi-cycler missions arise from eliminating the need to launch the transfer vehicle from Earth's surface and the need to inject the transfer vehicle from Mars orbit to return to the Earth; only moderate ΔV is required during the interplanetary trajectories.

An Earth-Mars semi-cycler mission begins by launching the crew to high Earth orbit in a taxi vehicle. The taxi then rendezvous with the transfer vehicle, which was left in high Earth orbit at the conclusion the preceding mission. Once the crew is cleared for departure, the taxi/transfer vehicle combination injects onto the semi-cycler trajectory. Time-insensitive Mars payload (e.g. cargo and consumables) is launched directly to Mars on a minimum-energy trajectory. At Mars arrival, the taxi (including the crew) detaches from the transfer vehicle and lands on the surface via aero-assisted direct entry. While

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the crew lands, the empty transfer vehicle receives a gravity assist from Mars and remains in interplanetary space until it picks up another crew at Mars before returning to Earth. After a 550-day mission at Mars, the crew departs the surface in the taxi to rendezvous with a transfer vehicle as it swings by Mars (i.e. the rendezvous occurs on a hyperbolic trajectory). At Earth arrival the crew again separates from the transfer vehicle and descends to the surface in a capsule (which is all that is left of the taxi). The transfer vehicle brakes into high Earth orbit to await refurbishment before the next departure opportunity. Fig. 1 contains a schematic of a typical mission.

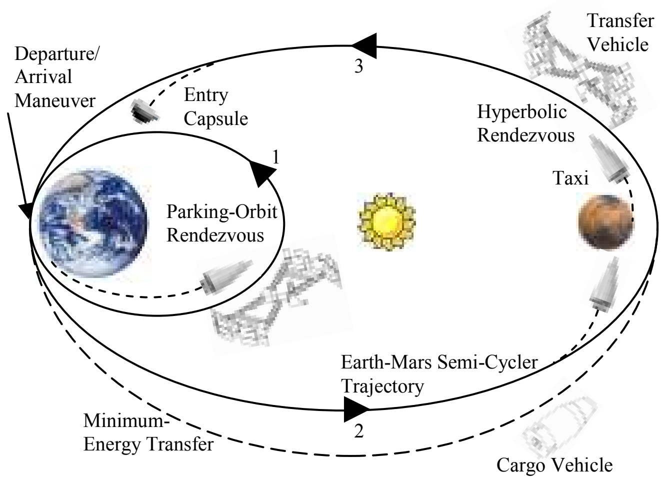

Fig. 1 Schematic of an Earth-Mars semi-cycler mission.

Thus, there are three types of vehicles in an Earth-Mars semi-cycler mission: 1) the taxi, which ferries the crew from the Earth's surface to the transfer vehicle in Earth orbit, lands the crew at Mars, ferries the crew from the surface of Mars to the transfer vehicle during Mars flyby, and finally lands the crew on Earth; 2) the transfer vehicle, which houses and protects the crew in between Earth and Mars (i.e. an interplanetary habitat), and 3) the cargo vehicle, which transports cargo (habitat, power plant, etc.) and consumables (food, air, water) on a low energy trajectory to the surface of Mars.

EARTH-MARS SEMI-CYCLER TRAJECTORIES

We require trajectories that depart Earth orbit, flyby Mars twice, then arrive back at Earth (thus the sequence is Earth-Mars-Mars-Earth) for an Earth-Mars semi-cycler architecture. We have identified four types of trajectories that provide this sequence with moderate ΔV. These four trajectory types can be classified by the ratio of spacecraft revolutions to Earth revolutions about the sun. For example, the first trajectory type (in Fig. 2) makes about 5 revolutions about the sun in the time that Earth makes 7 orbits (i.e. 7 years), thus the ratio is 7:5. (This nomenclature conveniently provides the period of the spacecraft orbit as approximately $7/5 = 1.4$ years.) The second trajectory (in Fig. 3) begins with a nearly 3:2 Earth:spacecraft resonance (and a short Earth-Mars leg), then an Earth gravity assist places the spacecraft on a 1:1 resonant transfer followed by another 3:2 resonance trajectory (with a short Mars-Earth leg). The body sequence is thus Earth-Mars-Earth-Earth-Mars-Earth, and the ratio sequence is 3:2-1:1-3:2. The third trajectory (in Fig. 4) makes about 4 revolutions about the sun in 5 years (a 5:4 ratio). Finally, the fourth trajectory type (in Fig. 5) has a 2:1 ratio with Earth, followed by a half-year Earth-

Earth inclined transfer, and concludes with another 2:1 resonant transfer, making the ratio sequence 2:1-0.5:0.5-2:1. Because the first two trajectories take about seven years (or 3.3 synodic periods) from Earth launch to Earth arrival, the spacecraft will be unavailable during the next three launch opportunities. As a result four vehicles are required to provide short Earth-Mars and Mars-Earth transfers every synodic period. Trajectories three and four have a total flight time of about 4.8 years (or 2.2 synodic periods) and thus require three vehicles to provide short transfers each synodic period.

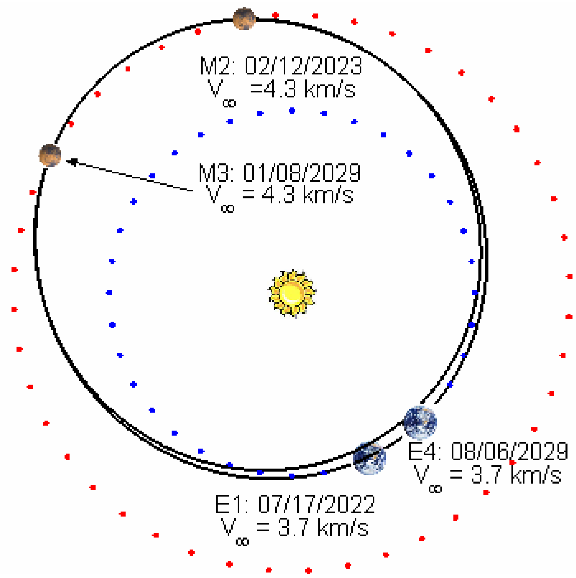

Fig. 2 Four-vehicle trajectory based on a 7:5 Earth:spacecraft resonance.

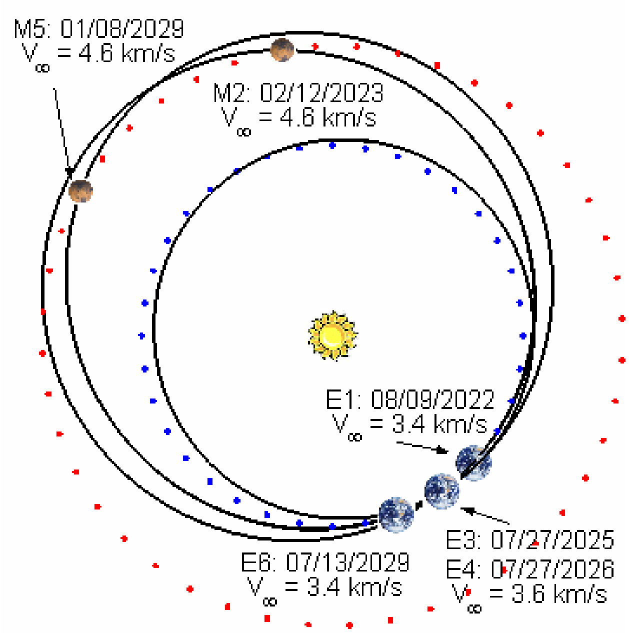

Fig. 3 Four-vehicle trajectory based on a 3:2-1:1-3:2 resonance sequence with Earth.

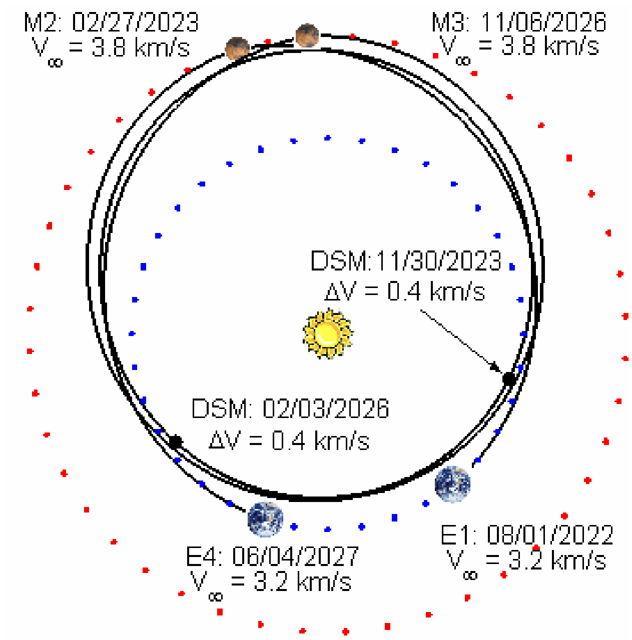

Fig. 4 Three-vehicle trajectory based on a 5:4 resonance with Earth.

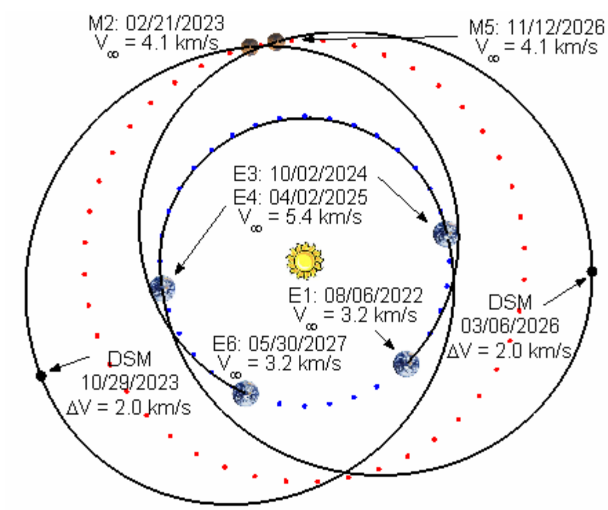

Fig. 5 Three-vehicle trajectory based on a 2:1-0.5:0.5-2:1 resonance sequence with Earth.

To characterize the V_∞ and ΔV requirements of each trajectory type, we minimize the sum of the Earth and Mars transfer V_∞ and deep space maneuver (DSM) ΔV in a circular coplanar solar-system model. (We use a circular coplanar model for Fig. 2–Fig. 5 because the trajectories will repeat exactly each synodic period.) When we optimize these trajectories in a more accurate solar system model (e.g. with integrated ephemeredes for Earth and Mars) it turns out that a combination of the four-vehicle trajectories (Fig. 2 and Fig. 3) require significantly less ΔV than the three-vehicle ones. We thus choose a four-vehicle architecture above a three-vehicle one in an attempt to reduce the injected mass to low-Earth orbit (IMLEO). We construct trajectories so that the time of flight (TOF) on the Earth-Mars and Mars-Earth legs is constrained to 180 days or less (in Table 1) and to 240 days or less (in Table 2). Itineraries spanning seven missions are provided because the trajectories approximately repeat in inertial space every seven synodic periods (and are therefore representative of the total solution into the far future). Table 3 contains a timeline that demonstrates how the four transfer vehicles operate in concert to complete seven Mars exploration missions. (The transfer TOF in Table 3 are all 180 days or less.)

Table 1
ITINERARIES WITH TRANSFER TOF $\leq$ 180 DAYS

Launch year	Earth launch	Mars arrival	Earth flyby or DSM	Earth flyby or DSM	Earth flyby or DSM	Mars launch	Earth arrival
2009	11/06/2009	05/05/2010	10/06/2010		06/20/2015	02/16/2016	08/14/2016
	4.59[a]	6.10	0.59[b]		0.41[b]	3.94	4.20
2011	12/20/2011	06/17/2012	03/13/2014	11/04/2014	11/05/2015	05/02/2018	10/09/2018
	4.81	5.80	0.31[b]	3.84	3.84	3.36	3.50
2014	01/18/2014	07/17/2014	08/01/2015			06/27/2020	12/24/2020
	3.72	5.81	0.54[b]			3.60	3.07
2016	03/07/2016	09/03/2016				08/13/2022	02/09/2023
	3.31	4.43				4.24	4.31
2018	05/07/2018	10/09/2018				09/20/2024	03/19/2025
	3.05	4.31				4.77	5.60
2020	07/17/2020	01/13/2021	01/15/2023			10/27/2026	04/25/2027
	3.67	3.22	0.38[b]			5.09	6.07
2022	09/09/2022	03/08/2023	06/05/2025	09/04/2025	05/14/2026	12/09/2028	06/07/2029
	4.43	4.64	10.10	1.10[b]	8.39	5.16	5.39

[a] All values except for DSMs are V_∞ in km/s.
[b] DSM, km/s.

Table 2
ITINERARIES WITH TRANSFER TOF $\leq$ 240 DAYS

Launch year	Earth launch	Mars arrival	Earth flyby or DSM	Earth flyby or DSM	Mars launch	Earth arrival
2009	11/05/2009	07/03/2010			01/21/2016	08/22/2016
	4.29[a]	3.18			3.04	3.99
2011	11/24/2011	07/21/2012	05/31/2013		04/19/2018	10/20/2018
	3.18	3.89	0.19[b]		2.85	3.27
2014	01/02/2014	08/03/2014	08/07/2015		07/04/2020	12/28/2020
	3.72	5.81	0.23[b]		3.60	3.07
2016	02/29/2016	09/09/2016			08/12/2022	03/01/2023
	3.15	4.24			4.24	3.22
2018	04/29/2018	10/17/2018			09/07/2024	05/05/2025
	2.97	3.85			3.97	2.81
2020	07/20/2020	01/18/2021			09/10/2026	05/08/2027
	3.66	3.08			3.08	4.76
2022	09/22/2022	05/20/2023	07/09/2025	07/09/2026	10/31/2028	06/24/2029
	4.79	2.52	5.52	5.52	3.43	4.44

[a] All values except for DSMs are V_∞ in km/s.
[b] DSM, km/s.

Table 3
TIMELINE FOR SEVEN TRIPS TO MARS FOR FOUR TRANSFER VEHICLES

Event	Date	V_∞ or DSM ΔV, km/s			
		TV 1	TV 2	TV 3	TV 4
Earth launch 1	11/06/2009	4.59			
Mars arrival 1	05/05/2010	6.10			
Earth flyby	06/14/2010				10.85
DSM	09/06/2010				1.16
DSM	10/06/2010	0.59			
Earth flyby	05/25/2011				8.86
Mars launch 1	11/23/2011			5.85	
Earth launch 2	12/20/2011		4.81		
Earth arrival 1	05/21/2012			4.97	
Mars arrival 2	06/17/2012		5.80		
Mars launch 2	12/26/2013				4.92
Earth launch 3	01/18/2014			3.72	
DSM	03/13/2014		0.31		
Earth arrival 2	06/24/2014				5.12
Mars arrival 3	07/17/2014			5.81	
Earth flyby	11/04/2014		3.84		
DSM	06/20/2015	0.41			
DSM	08/01/2015			0.54	
Earth flyby	11/05/2015		3.84		
Mars launch 3	02/16/2016	3.94			
Earth launch 4	03/07/2016				3.31
Earth arrival 3	08/14/2016	4.20			
Mars arrival 4	09/03/2016				4.43
Mars launch 4	05/02/2018		3.36		
Earth launch 5	05/07/2018	3.05			
Earth arrival 4	10/09/2018		3.50		
Mars arrival 5	10/09/2018	4.31			
Mars launch 5	06/27/2020			3.60	
Earth launch 6	07/17/2020		3.67		
Earth arrival 5	12/24/2020			3.07	
Mars arrival 6	01/13/2021		3.22		
Mars launch 6	08/13/2022				4.24
Earth launch 7	09/09/2022			4.43	
DSM	01/15/2023		0.38		
Earth arrival 6	02/09/2023				4.31
Mars arrival 7	03/08/2023			4.64	
Mars launch 7	09/20/2024	4.77			
Earth arrival 7	03/19/2025	5.60			

MISSION ASSUMPTIONS

The main advantage of an Earth-Mars semi-cycler mission over a more traditional mission is a reduction in the injected mass to low-Earth orbit (IMLEO). We note that IMLEO is often strongly correlated to the dollar-cost of a given mission.[33, 34] Therefore, we assess the potential benefit of the Earth-Mars semi-cycler architecture by comparing its IMLEO to the IMLEO of a semi-direct mission (with an Earth launch and Mars parking orbit for the transfer vehicle). This IMLEO comparison is made for missions that rely solely on chemical propulsion [liquid hydrogen and liquid oxygen (LH2/LOX)], as well as for missions that incorporate nuclear thermal rocket (NTR) Earth upper stages, aerocapture, and in-situ propellant production (ISPP) at Mars (which are the key propulsion technologies in NASA's Design Reference Mission[21, 22]). We also vary the cargo-mass from 40 metric tons for infrastructure development (such as habitats, power plants, etc.) to 0 mt for settlement scenarios (already provided with habitats and power-plants from previous missions). The Mars-surface consumables are transported on the cargo vehicle but they are not included as part of the cargo mass. The rest of the surface payload (crew and taxi) travel to Mars with the transfer vehicle. Finally, we calculate the IMLEO for missions where the transfer TOF between Earth and Mars is restricted to below 180 days in addition to missions where the TOF is as long as 240 days. The other mission assumptions are as follows.

1.) There are four crew members.
2.) The taxi ascent/descent capsule is 5 mt (including the crew and excluding the aerobrake).
3.) The transfer vehicle (TV) has a mass of 20 mt.
4.) During the first four Earth-Mars semi-cycler missions a new transfer vehicle is launched from Earth's surface. The follow-on missions do not require a transfer vehicle launch because the transfer vehicles from the first four missions have returned to Earth orbit. (We note that an Earth-return vehicle from the first two missions could be captured into Earth orbit for use as a transfer vehicle in the third and fourth missions, but the additional mass to achieve orbit insertion often increases the IMLEO.)
5.) For the first three Earth-Mars semi-cycler missions, an extra transfer vehicle is sent to Mars orbit (as in a semi-direct mission) to return the crew to Earth. This vehicle is necessary because the transfer vehicle from the first launch does not reach Mars again until the end of the fourth mission.
6.) For Earth-Mars semi-cycler missions, each transfer vehicle is completely renewed every 15 missions. To account for this, 27% of the transfer vehicle mass (5.33 mt) is launched from Earth for refurbishment after each mission.
7.) A new propulsion system is launched and attached to the transfer vehicle before each Earth-Mars semi-cycler mission. (That is, the transfer-vehicle propulsion system is modular.)
8.) The Mars ascent taxi is sent with the crew to Mars. This eliminates the need to launch two taxis (one Earth ascent and one Mars ascent) from Earth.
9.) All Mars payload except for the crew and taxi is sent to Mars on a minimum-energy transfer.

10.) The consumables requirement is 5 kg/person/day. For ISPP missions, we assume that only 40% (2 kg/person/day) of the Mars-stay consumables mass must come from Earth; the rest is derived from the atmosphere by combining H_2 from Earth with Martian CO_2 to produce water and oxygen.
11.) The aerobrake is 15% of the entry mass. Aerobrakes are not reused.
12.) 500 m/s of ΔV is provided to soften the landing on Mars.
13.) Nuclear thermal rockets have an I_{sp} of 900 s and an inert mass fraction (μ_{inert}) of 30%.
14.) Liquid hydrogen/liquid oxygen rockets have an I_{sp} of 450 s and an inert mass fraction of 10%.
15.) Liquid methane/liquid oxygen rockets have an I_{sp} of 380 s and an inert mass fraction of 10%.
16.) Liquid hydrogen/liquid oxygen boiloff losses are 10% from Earth launch to Mars launch.[35]
17.) Liquid hydrogen boiloff losses are 10% from Earth launch to Mars arrival.[35]
18.) A cryocooler is required to store liquid hydrogen or liquid oxygen for longer than two synodic periods. The effective cryocooler inert mass fraction is 5%.[36]
19.) For in-situ propellant production, 18 mt of methane and oxygen are produced for every 1 mt of hydrogen landed on Mars.[18]
20.) The high-energy parking orbits (HPO) at either Earth or Mars have a periapsis altitude of 300 km and a period of one day. We also designate these orbits as HEO and HMO at Earth and Mars, respectively.
21.) The altitude for low-circular orbits (LCO) is 300 km.
22.) The parking orbit reorientation ΔV (to achieve proper departure alignment) is 300 m/s at Earth and 200 m/s at Mars.
23.) The hyperbolic rendezvous ΔV at Mars is 200 m/s.[37]
24.) The trajectory V_∞ and ΔV requirements are calculated from the data presented in Ref. 38. Earth-Mars semi-cycler trajectories are used for Earth-Mars semi-cycler missions and direct trajectories are employed in semi-direct missions.

IMLEO CALCULATION

The following fundamental equations allow us to estimate the IMLEO for a round-trip mission to Mars. The Mars launch vehicle is a two-stage rocket that ascends from the surface to a low-circular orbit. We do not include drag, steering, or gravity losses nor the velocity due to planetary rotation in the launch ΔV; instead we add a 5% ΔV cost.

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

The ΔV required to reach the HPO from the LCO by an upper stage is

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

Finally, the ΔV to achieve a given V_∞ from the HPO is

$$\Delta V_{esc} = \sqrt{\frac{2GM}{r_{LCO}} + V_\infty^2} - \sqrt{GM\left(\frac{2}{r_{LCO}} - \frac{1}{a_{HPO}}\right)} \tag{3}$$

We note that the ΔV to reach V_∞ from the surface may be calculated as the sum of Eq. (1) through Eq. (3).

The rocket equation[39] is used to determine mass fractions for a single stage

$$\mu_{stage} = \frac{m_0}{m_f} = \exp\left(\frac{\Delta V}{ngI_{sp}}\right) \tag{4}$$

The ratio of initial mass to the payload mass for a given ΔV is thus

$$\frac{m_0}{m_{pl}} = \left[\frac{\mu_{stage}\left(1-\mu_{inert}\right)}{1-\mu_{inert}\mu_{stage}}\right]^n \tag{5}$$

By stacking the mission payload, aeroshells, and propulsion stages, we can calculate the mass in low Earth orbit.

The IMLEO for seven missions (for launch years 2009 to 2022) are provided for Earth-Mars semi-cycler (EMSC) and semi-direct missions in Table 4–Table 7. Because the trajectories (nearly) repeat every seven synodic periods, a seven-mission cycle represents the range of IMLEO values. For each combination of TOF, propulsion system, and Mars payload mass in Table 4–Table 7, we provide two columns of Earth-Mars semi-cycler IMLEO: 1) the initial EMSC, which accounts for launching four cycling transfer vehicles and three return transfer vehicles, and 2) the repeat EMSC, where we assume that the four transfer vehicles have been previously launched. We note that the initial EMSC is a one-time investment, while the repeat EMSC characterizes recurring IMLEO costs. Finally, we provide example IMLEO mass breakdowns in Table 8–Table 11 to examine individual mission components and to further compare Earth-Mars semi-cyclers with semi-direct architectures.

Table 4
IMLEO (in mt) FOR TOF $\leq$ 180 DAYS WITH LH2/LOX PROPULSION

	40 mt of Cargo[a]			No Cargo[a]		
Launch Year	Initial EMSC	Repeat EMSC	Semi-Direct	Initial EMSC	Repeat EMSC	Semi-Direct
2009	597	516[b]	576[c]	463	381	441
2011	553	477	551	421	344	418
2014	535	398	476	401	266	344
2016	386	356	412	255	225	281
2018	355	355	394	224	224	264
2020	410	410	435	270	270	296
2022	491	491	505	350	350	365
Total	3,327	3,002	3,349	2,385	2,060	2,408

[a] Cargo includes habitat, power plant, etc., but does not include consumables, crew, or taxi.
[b] Mass breakdown found in Table 8.
[c] Mass breakdown found in Table 9.

Table 5
IMLEO (in mt) FOR TOF $\leq$ 180 DAYS WITH NTR, AEROCAPTURE, AND ISPP

	40 mt of Cargo			No Cargo		
Launch Year	Initial EMSC	Repeat EMSC	Semi-Direct	Initial EMSC	Repeat EMSC	Semi-Direct
2009	311	197[a]	253[b]	214	99	156
2011	295	191	237	198	94	140
2014	277	186	219	181	90	123
2016	197	173	213	101	78	117
2018	175	175	224	79	79	128
2020	188	188	246	89	89	147
2022	200	200	259	100	100	159
Total	1,644	1,311	1,652	963	630	971

[a] Mass breakdown found in Table 10.
[b] Mass breakdown found in Table 11.

Table 6
IMLEO (in mt) FOR TOF $\leq$ 240 DAYS WITH LH2/LOX PROPULSION

	40 mt of Cargo			No Cargo		
Launch Year	Initial EMSC	Repeat EMSC	Semi-Direct	Initial EMSC	Repeat EMSC	Semi-Direct
2009	501	384	416	366	249	281
2011	482	368	400	349	235	267
2014	484	354	407	351	222	275
2016	370	340	397	239	209	266
2018	354	354	390	223	223	259
2020	390	390	422	251	251	283
2022	409	409	438	269	269	298
Total	2,990	2,599	2,871	2,048	1,657	1,929

Table 7
IMLEO (in mt) FOR TOF $\leq$ 240 DAYS WITH NTR, AEROCAPTURE, AND ISPP

	40 mt of Cargo			No Cargo		
Launch Year	Initial EMSC	Repeat EMSC	Semi-Direct	Initial EMSC	Repeat EMSC	Semi-Direct
2009	283	182	228	186	85	130
2011	272	178	217	176	81	120
2014	271	176	213	174	80	116
2016	196	173	212	101	77	116
2018	177	177	225	81	81	129
2020	186	186	243	86	86	143
2022	191	191	242	91	91	142
Total	1,576	1,262	1,579	895	581	898

Table 8
EMSC IMLEO BREAKDOWN
WITH LH2/LOX PROPULSION
(2009 LAUNCH YEAR IN Table 4)

Element	Mass (mt)
Cargo	40.0
Surface consumables	10.9
Cargo landing propulsion	6.9
Cargo aerobrake	8.7
Cargo LEO-to-Mars propellant	94.3
Cargo LEO-to-Mars inert mass	10.5
TV HEO-capture propellant	16.6
TV HEO-capture inert mass	1.8
TV DSM propellant	15.6
TV DSM inert mass	1.7
In-space consumables	7.2
TV refurbishments	5.3
Crew, capsule, aerobrake	5.8
Mars-taxi propellant	40.2
Mars-taxi inert mass	4.5
Mars-taxi landing propulsion	6.8
Mars-taxi aerobrake	8.6
HEO-to-Mars propellant	47.3
HEO-to-Mars inert mass	5.3
LEO-to-HEO propellant	160.3
LEO-to-HEO inert mass	17.8
Total	516

Table 9
SEMI-DIRECT IMLEO BREAKDOWN
WITH LH2/LOX PROPULSION
(2009 LAUNCH YEAR IN Table 4)

Element	Mass (mt)
Cargo	40.0
Surface consumables	10.9
Cargo landing propulsion	6.9
Cargo aerobrake	8.7
Cargo LEO-to-Mars propellant	94.3
Cargo LEO-to-Mars inert mass	10.5
Transfer vehicle	20.0
TV HMO-to-Earth propellant	17.0
TV HMO-to-Earth inert mass	1.9
TV HMO-capture propellant	40.0
TV HMO-capture inert mass	4.4
In-space consumables	7.2
Crew, capsule, aerobrake	5.8
Mars-taxi propellant	18.1
Mars-taxi inert mass	2.0
Mars-taxi landing propulsion	3.5
Mars-taxi aerobrake	4.4
LEO-to-Mars propellant	252.3
LEO-to-Mars inert mass	28.0
Total	576

Table 10
EMSC IMLEO BREAKDOWN
WITH NTR, AEROCAPTURE, AND ISPP
(2009 LAUNCH YEAR IN Table 5)

Element	Mass (mt)
Cargo	40.0
Surface consumables	4.3
Cargo landing propulsion	6.0
Cargo aerobrake	7.5
Cargo LEO-to-Mars propellant	35.0
Cargo LEO-to-Mars inert mass	15.0
TV Earth aerobrake	3.0
TV DSM propellant	2.7
TV DSM inert mass	1.2
In-space consumables	7.2
TV refurbishments	5.3
Crew, capsule, aerobrake	5.8
Mars-taxi propellant feedstock	3.3
Mars-taxi inert mass	5.3
Mars-taxi landing propulsion	1.9
Mars-taxi aerobrake	2.4
HEO-to-Mars propellant	11.7
HEO-to-Mars inert mass	5.0
LEO-HEO propellant	23.9
LEO-HEO inert mass	10.2
Total	197

Table 11
SEMI-DIRECT IMLEO BREAKDOWN
WITH NTR, AEROCAPTURE, AND ISPP
(2009 LAUNCH YEAR IN Table 5)

Element	Mass (mt)
Cargo	40.0
Surface consumables	4.3
Cargo landing propulsion	6.0
Cargo aerobrake	7.5
Cargo LEO-to-Mars propellant	35.0
Cargo LEO-to-Mars inert mass	15.0
Transfer vehicle	20.0
TV HMO-to-Earth propellant	22.2
TV HMO-to-Earth inert mass	2.5
TV Mars aerobrake	7.2
In-space consumables	7.2
Crew, capsule, aerobrake	5.8
Mars-taxi propellant feedstock	1.7
Mars-taxi inert mass	2.7
Mars-taxi landing propulsion	1.4
Mars-taxi aerobrake	1.7
LEO-to-Mars propellant	51.2
LEO-to-Mars inert mass	22.0
Total	253

ARCHITECTURE COMPARISON

From Table 4–Table 7 we find that Earth-Mars semi-cycler and semi-direct missions require about the same average IMLEO during the first seven missions. The first three Earth-Mars semi-cycler missions require substantially higher IMLEO because two transfer vehicles (one semi-cycler vehicle and one Earth-return vehicle) are launched from Earth. During the fourth mission, the fourth (and final) semi-cycler transfer vehicle departs Earth without an accompanying Earth-return vehicle, which lowers the IMLEO considerably. (The first semi-cycler vehicle acts as the Earth-return vehicle on the fourth mission.) After the fourth mission, the Earth-Mars semi-cycler architecture is established and no further transfer-vehicle launches are required. Semi-direct missions have a more consistent IMLEO during these first seven missions as a single transfer vehicle is launched from Earth during each mission. We note that Earth-Mars semi-cyclers require, at most, seven transfer vehicles (with upkeep), while semi-direct missions require the construction of a new vehicle for every mission (and thus an indefinite number of transfer vehicles). After the third mission to Mars, the Earth-Mars semi-cycler consistently requires less IMLEO than semi-direct architectures.

The highest IMLEO missions (180-day TOF with LH2/LOX propulsion in Table 4) also result in the largest absolute savings in IMLEO (50 mt per mission) between Earth-Mars semi-cycler and semi-direct missions. Missions with NTR, aerocapture, and ISPP technology result in similar IMLEO-savings of 48 mt and 45 mt per mission for 180-day TOF and 240-day TOF, respectively. Of the examined missions, LH2/LOX propulsion with a TOF of 240 days results in the lowest absolute IMLEO-savings of 39 mt per mission. Considering a proposed capability of 80 mt to LEO for a next-generation launch vehicle,[22] the Earth-Mars semi-cycler architecture eliminates multiple Earth-to-orbit launches during a seven-mission cycle. (Here we note for comparison that the shuttle capacity is around 30 mt, while that of the Saturn V was approximately 120 mt to LEO.)

The relative mass difference (between Earth-Mars semi-cycler and semi-direct missions) is lowest (9.5% in Table 6) or large cargo missions that employ only LH2/LOX propulsion. When the IMLEO dedicated to cargo delivery is large compared to the IMLEO for crew transport, the architecture differences become less pronounced as cargo missions are generally independent of the architecture selection (e.g. the cargo elements in Table 8 and Table 9 are the same). Thus, the key architecture differences lie in how the crew gets to Mars and back.

A significant portion of the mass dedicated to crew transportation is the Mars taxi (or Mars launch/ascent vehicle). The taxi in an Earth-Mars semi-cycler mission ferries the crew from the surface of Mars to escape, whereas a semi-direct taxi only achieves a high-energy parking orbit about Mars before rendezvous. As a result, in-situ propellant production lowers the taxi mass more for Earth-Mars semi-cyclers than semi-direct architectures because more propellant must be created at Mars. In fact, the largest savings in IMLEO from semi-direct to Earth-Mars semi-cycler architectures (35% in Table 5 and Table 7) occurs for missions with ISPP (as well as NTR, aerocapture, and no cargo).

The details of the taxi-mass savings are found in Table 10 where the taxi feedstock and inert mass for Earth-Mars semi-cyclers combine to 8.6 mt. The mass required to transport the crew from Mars to escape in a semi-direct scenario is 29.1 mt in Table 11. [This mass includes 4.4 mt for the capsule propulsion system to HMO (Mars-taxi in Table 11) and 24.7 mt for crew, capsule, and transfer-vehicle propulsion from HMO to Earth (TV HMO-to-Earth in Table 11).] Thus we see that eliminating the transfer-vehicle departure from Mars orbit eliminates much of the mass sent to Mars. We note that in Table 11 the propellant for the transfer vehicle does not come from ISPP (i.e. it all comes from Earth). This option is more efficient than using ISPP to escape orbit because of the additional mass to launch the transfer vehicle propellant off of the surface. (The NASA Design Reference Mission also employs terrestrial propellants to depart Mars orbit.[21]) Earth-Mars semi-cyclers often benefit from a smaller surface-to-escape mass because only the crew and capsule depart Mars, whereas the transfer vehicle (in addition to the crew and capsule) also departs from Mars orbit in a semi-direct mission.

For LH2/LOX propulsion missions, the reduction in IMLEO for Earth-Mars semi-cyclers derives mainly by removing the Mars-orbit insertion maneuver in the semi-direct mission. We note that capturing the transfer vehicle at Earth requires less ΔV than capturing into a loose orbit at Mars because of the stronger gravity at Earth. From Table 9 the mass for HMO-capture is 44.4 mt, which places the transfer vehicle and HMO-to-Earth propulsion system in Mars orbit. The only maneuvers that the Earth-Mars semi-cycler transfer vehicle needs to accomplish are HEO-insertion (18.4 mt in Table 8) and the DSM (17.3 mt). Thus, almost 9 mt of propulsion system mass is eliminated at Mars arrival. This mass-saving is multiplied by the reduction in propellant required to transport the propulsion systems out of LEO. Moreover, only 5.3 mt of transfer-vehicle refurbishments is transported from LEO to HEO for Earth-Mars semi-cyclers, while a complete 20 mt transfer vehicle makes the trip in a semi-direct mission. The reduction in transfer-vehicle and propellant mass is significant for current and near-term propulsion systems.

Additional mass savings are possible by extending the time of flight. For example, an increase in TOF from 180 days to 240 days reduces the IMLEO by an average of 58 mt (20%) per Earth-Mars semi-cycler mission with LH2/LOX. The IMLEO is only reduced by 7.0 mt (8%) with NTR, aerocapture, and ISPP with the same increase in TOF. The percent IMLEO savings between semi-cycler and Earth-Mars semi-cycler missions does not vary significantly as a function of TOF as the ΔV for both missions decrease at about the same rate as the TOF increases.

While a reduction in IMLEO is the primary benefit of the Earth-Mars semi-cycler architecture, the two main disadvantages are hyperbolic rendezvous at Mars and the continuous upkeep of the transfer vehicle. Rendezvous on a hyperbolic trajectory and rendezvous in an elliptical orbit (as in a semi-direct mission) comprise three similar steps: 1) depart a low circular orbit to closely match the path of the (target) transfer vehicle, 2) determine where the taxi is in relation to the transfer vehicle, and 3) guide the taxi toward the transfer vehicle for safe docking. The chance of failure during any stage is about the same for hyperbolic and elliptical rendezvous because similar hardware is required for each. The key difference is that the taxi *must* dock with the transfer vehicle during hyperbolic rendezvous because it has already left Mars for Earth. During elliptic rendezvous the crew could abort to the surface of Mars because the taxi is still trapped in a parking orbit. Extra propellant should be included on the taxi to correct thrusting or navigational errors during hyperbolic rendezvous, but determination of an adequate safety margin requires a more detailed analysis.

The problem with reusing a transfer vehicle (or any piece of hardware) is that eventually something is going to break. To mitigate the effects of fatigue we replace more than a quarter of the transfer vehicle each time it departs Earth. However, this renovation must occur in Earth orbit (with an allotted time of about 600 days), and in-orbit refurbishment is more demanding than Earth-based construction (though we are developing techniques by building and maintaining the International Space Station). Another drawback of continually operating a transfer vehicle is that extra safety checks are required to ensure that the older and critical parts continue to function. An expendable transfer vehicle (used in a semi-direct architecture) may not require as much inspection because it will never spend more than three years in space. Finally, the transfer vehicle in an Earth-Mars semi-cycler mission will be empty for up to six years in-between Mars flybys. Should an unforeseen problem occur, no one is on board to fix it and automated systems may not be sufficient. Moreover, when the crew is ready to return to Earth, they enter an empty house and some spring cleaning may be required to make it livable. Alternatively, the transfer vehicle is only unoccupied for about 550 days during a semi-direct mission, and it is never more than a day away from the crew while they are on the surface of Mars.

CONCLUSIONS

There are myriad proposals for how people could travel between Earth and Mars. We present a Mars exploration architecture (the Earth-Mars semi-cycler) with reusable transfer vehicles that depart Earth orbit, fly by Mars twice, then return to Earth. There are at least four trajectory types that enable this type of mission with moderate ΔV. Of these trajectories, we recommend the four-vehicle versions to minimize the IMLEO over several launch opportunities. To evaluate the performance of Earth-Mars semi-cyclers we calculate the IMLEO for current and near-term propulsion technologies, large and small Mars payloads, and moderate to long TOF for seven consecutive missions. If the same crew and vehicles are used in a semi-direct architecture (i.e. with a Mars parking orbit) then at least 10% extra IMLEO is required for LH2/LOX propulsion systems and up to 50% additional IMLEO is required with NTR, aerocapture, and ISPP technologies. The reduced IMLEO achieved with the Earth-Mars semi-cycler (compared to the semi-direct architecture) lowers the number of launches from Earth. Of course, these savings do not accrue until after the seventh mission because of an initial investment to launch the four transfer vehicles off the Earth. Compared to other mission proposals, the Earth-Mars semi-cycler ranks as an ambitious, yet efficient system for the sustained exploration of Mars.

ACKNOWLEDGEMENTS

We thank Jonah Skoog for creating the vehicle images in Fig. 1. The first author's work has been sponsored by a National Science Foundation Graduate Research Fellowship and a National Defense Science and Engineering Graduate Fellowship.

NOTATION

DSM = deep space maneuver
g = standard acceleration due to gravity at Earth's surface, 9.80665 m/s^2
HEO = high Earth orbit
HMO = high Mars orbit
HPO = high-energy (elliptical) parking orbit
I_{sp} = specific impulse, s
ISPP = in-situ propellant production (methane/oxygen)
LCO = low circular orbit
LH2 = liquid hydrogen
LOX = liquid oxygen

n	=	number of rocket stages
NTR	=	nuclear thermal rocket
surf	=	surface of a planet
V_∞	=	hyperbolic excess speed, km/s
ΔV	=	instantaneous change in velocity, km/s
μ_{inert}	=	inert mass fraction, $m_{inert}/(m_{inert} + m_{propellant})$

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