MORPH THE GATEWAY INTO AN EARTH-MARS CYCLER? TRAJECTORIES TO/FROM MARS

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ABSTRACT

Some of the history is presented that has resulted in NASA's plans for a Lunar Gateway (or just Gateway), and how constructing it to become a Deep Space Transport (DST) could be less costly than the current plan. Also discussed is how a DST can reach a variety of interesting destinations from an Earth-Moon halo orbit. Robert Farquhar promoted a lunar halo orbit station in 1971 and 2004. Others expanded upon his ideas to create what is now known as the Lunar Gateway. However, if the goal is only exploration of the lunar surface, a "Moon-direct" approach is more efficient. For human missions to Mars and near-Earth objects, a lunar halo orbit is a good high-energy perch for a reusable DST between missions. The Gateway might be changed to a DST; building only one maneuverable habitat instead of two provides large savings. A technique we call Phasing Orbit Rendezvous (PhOR) is proposed for exploration by the DST, to transfer astronauts and supplies to it just before departure from Earth to an interplanetary destination.

INTRODUCTION

NASA published its Integrated Program Plan (IPP) for post-Apollo lunar exploration in 1969.[1] The IPP envisioned a reusable winged shuttle to reach low-Earth orbit (LEO), a LEO space station, a lunar-orbit space station (LOSS) in a 60 nautical mile polar orbit, a propellant storage depot located near the LOSS, and a reusable chemical lunar space tug (LST) that would operate between the LOSS and the lunar surface. Robert Farquhar believed that the performance of the system could be significantly improved by locating the lunar staging node in a halo orbit around the Earth-Moon L2 (EM-L2) libration point instead of the low lunar polar orbit. In 1971, Farquhar completed a quantitative study of a lunar shuttle system that used a halo-orbit space station (HOSS) to replace the LOSS that showed the advantages of the HOSS.[2] Without maneuvers, the LOSS would impact the Moon in 4 months, which Farquhar sarcastically noted would be a "real LOSS". With that, NASA changed the name to Orbiting Lunar Station (OLS).[3] Farquhar again promoted staging at libration points when NASA began its Space Exploration Initiative in 1989, but that program ended in 1993 when President George H. W. Bush left office.

Libration Point Orbits as Staging Nodes for Human Interplanetary Exploration

Farquhar describes the basics of libration points in the planar restricted three-body problem, showing the seven libration points in and near the Earth-Moon that could be useful for exploration of the Moon and the Earth-Moon environment[4]. He was also the first to realize the value of staging at libration points for interplanetary exploration. Figure 1 sketches his early idea for a reusable "cycler" between the Sun-Earth L1 libration point and the Sun-Mars L2 point.[5]

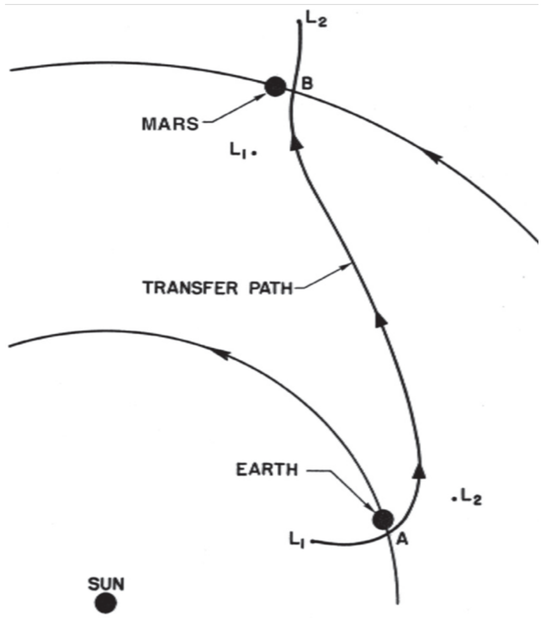


Figure 1. Robert Farquhar's concept of an interplanetary libration-point cycler, 1969.

In the late 1990s, several of Farquhar's libration-point staging concepts were adopted by a small internal NASA planning team known as the NASA Exploration Team (NExT).[6,7] However, the NExT team focused their studies on staging at the Earth-Moon L1 (EM-L1) point instead of the L2 point as Farquhar had recommended. NExT was part of the Decadal Planning Team (DPT) that developed the Gateway concept and gave it the name, within the context of a detailed Moon-Mars architecture[8]. Reference 8 gives links to several DPT reports that will be described in a formal history, to be published soon[9], including its significant influence on the Vision for Space Exploration (VSE) that was announced by President George W. Bush on January 14, 2004.

At virtually the same time that the NExT studies began, Wes Huntress, former NASA Associate Administrator for Science, initiated an International Academy of Astronautics (IAA) "Cosmic Study" entitled, "The Next Steps in Exploring Deep Space". Huntress invited Farquhar to participate in the study in the role of Mission Design Lead. The first phase of the Cosmic Study was completed in 2004 under the leadership of Wes Huntress.[10] The second phase began in 2006 under Farquhar's leadership.

Because Farquhar had played a key role in the IAA Cosmic Study, he was adamantly opposed to the Constellation Program, and the NASA Administrator's proposal for a Moon base. In Farquhar's view, the Moon was not a stepping stone to Mars, but was instead, a stumbling block[11]. In 2006, Farquhar decided to mount a campaign against the VSE (including the Gateway) and NASA's Constellation Program. Instead of merely criticizing NASA's strategy, Farquhar argued that the plan described in the IAA study offered a more sensible alternative. Farquhar's efforts were muted at the time by his employer, the Johns Hopkins Applied Physics Laboratory.[12] The VSE was replaced by a new space policy, that ultimately resulted in the Asteroid Redirect Mission (ARM) studies under the Obama administration, starting in June 2010. Under ARM, any lunar outpost would be a temporary one built near the returned asteroid boulder in a distant retrograde orbit about the Moon.

The IAA Cosmic Study and Russian Megagrant Work, 2011-2014

In 2011, Dunham at KinetX and Natan Eismont, at the Space Research Institute (IKI) in Moscow, submitted a proposal, "An International Program of Tasks for Human Exploration of the Solar System and for Planetary Defense", to the Russian Ministry of Education and Science for what they called a megagrant. The proposal won and work began in November 2011 at the Moscow Institute of Electronics and Mathematics (MIEM). Dunham and Farquhar developed a program of tasks that meshed with the IAA Cosmic Study. A series of papers were published, describing how spacecraft could be staged from an outpost in a small-amplitude EM-L2 halo orbit; the outpost was called the International Exploration Station (IES). A few of the more important papers developed the ideas.[13,14,15] The papers noted that the basic ideas of using halo orbits, "Weak Stability Boundaries (WSB)", and lunar swingbys, although chaotic, involved long-enough time scales that they could be accurately flown, and flight-proven first by the ISEE-3/ICE mission in 1978-1983 (Farquhar was the flight director of that mission), and subsequently by several other missions. The IAA Cosmic Study team collaborated with Lockheed-Martin's early studies for Orion trajectories.[16] Some of the work performed under the Russian megagrant is presented below, with some changes and additions. In April 2015, Farquhar gave a presentation about this work, and his ideas for the future, at NASA Headquarters. Robert Farquhar died in October 2015, when NASA was still pursuing ARM (the Asteroid Redirect Mission) and not considering any libration-point trajectories for human exploration. ARM was cancelled in 2017.

THE NRHO AND THE GATEWAY – IS IT THE BEST WAY FORWARD?

Following the cancellation of ARM, NASA turned its attention to human lunar exploration, proposing the construction of a lunar Gateway in an orbit that can be reached by Orion, as launched with an SLS, as well as an as-yet-unspecified crewed vehicle to take astronauts from the Gateway to the lunar surface and back. The idea of the Gateway was inherited from the NExT studies, and ultimately from Farquhar's earlier publications, as described above. NASA has redirected the large solar electric propulsion (SEP) system, developed for ARM, to be used instead for other exploration goals. This has led to a new emphasis on low-thrust trajectories. We had followed Lockheed-Martin's lead a few years ago to use a relatively small-amplitude northern EM-L2 halo orbit for staging at the Moon[16]. But now, nearly everyone advocates using a large-amplitude EM-L2 halo orbit, called a Nearly Rectilinear Halo Orbit, or NRHO (from its appearance in a sideways Earth-Moon rotating system).[17,18] We have calculated most of our trajectories using only the smaller northern EM-L2 halo orbit. Some comparisons can be made, given below, between our trajectories and similar ones computed mainly by others using an NRHO[18].

Especially with the acceleration of the Artemis program, to land a woman and a man on the Moon by 2024, some have questioned the need for the Gateway, and NASA is looking at ways to scale it down, especially initially, to meet the 2024 deadline. If our goal is only to go to the Moon as quickly as we can, then no, the Gateway should not be built. Arthur C. Clarke recognized this as early as 1961, when he commented: “A Moon-bound spaceship stopping at the L1 station to pick up a passenger and some cargo would waste time and a lot of ΔV”. In his novel, Clarke was referring to a space station at the EM-L1 point for supporting lunar surface operations.[19] Robert Zubrin has been more vocal in his opposition to the Gateway and using SEP systems especially for going to Mars; he proposes “Moon-Direct”[20] and “Mars-Direct”[21] approaches instead. An analysis of “Moon-Direct” by Ryan Whitley’s NASA team found that Zubrin’s numbers are too optimistic. The Dragon mass is unrealistically low for operating in cislunar space, also considering the assumed propellant mass fraction and reusability issues, so the numbers must all be higher for realistic systems. New technologies are needed for the lunar excursion vehicle that is assumed to reach the high percentage of the lunar surface, and the time and cost of reaching an acceptable TRL for that is not considered. There are also questions of orbital rendezvous that would be needed at LEO. Nevertheless, it’s clear that Moon Direct would still have lower total cost, but not by quite as large a margin as Zubrin claims. Zubrin notes that using only a SEP system adds many months to spiral away from the Earth, to leave the Earth’s gravitational sphere of influence to go to Mars. Any Earth-Mars architecture will benefit greatly if propellant can be manufactured on Mars, as SpaceX plans to do with their methane-based propulsion system; the xenon needed for SEP systems can’t be manufactured on Mars. Zubrin concludes that a goals-driven architecture is needed, not as he calls it, a vendor-driven one[22]. He notes that neither SpaceX nor Blue Origin have a Gateway in their plans. At the end of Zubrin’s talk at the International Space Development Conference in June 2019, someone asked him about the utility of Aldrin cyclers. Zubrin said they might be useful later on, comparing the cyclers to the transcontinental railway, which was built well after San Francisco was settled, not before it. The same might be said for the “cycler” system that we describe below, that stages from an EM-L2 halo orbit, and highly elliptical orbits at Mars, so they don’t need high ΔV hyperbolic rendezvous as required by the Aldrin cyclers.

In any case, although a Gateway doesn’t make much sense for quick trips to the lunar surface, there is some advantage of having some infrastructure in a high-energy orbit, from which departures to a variety of interplanetary destinations, not just the Moon, would be possible. Plans for the Gateway may be too far advanced to stop it at this point. However, NASA’s current plan envisions not only a Gateway (Farquhar would have called it the IES) but also a rather similar Deep Space Transport (DST; Farquhar called it the Interplanetary Transportation Vehicle, or ITV) for interplanetary trips, especially to Mars. NASA’s plan calls for both to use large SEP propulsion systems. They are so similar that there is a real question, why build both? Many savings could result if only the DST were built; in the early years, the DST could serve the purposes for which the Gateway is currently planned. In fact, this approach was advocated by NExT in 2000[23] and recently, at the 2019 Humans to Mars Summit, in a panel discussion about using the Gateway for exploration, Neeraj Gupta, Sierra Nevada Corp., said that Gateway elements should be modular so they can also be used for the DST.[24] Plans for the Gateway have not been finalized and the National Space Council’s Users’ Advisory Group (UAG) has asked for ideas from others.[25] So if a Gateway is built at all, the authors recommend that it be built in a way that can easily be changed to the DST. Further, we recommend that a hybrid propulsion system, with chemical propulsion modules that can be refueled with methane and oxidizer at Mars, be added, just large enough to do the sizeable burns around the periapsis of the Moon, the Earth, and Mars. That would eliminate the long spiral trajectories needed with a pure SEP system to escape the planets. But if chemical propulsion modules are added, the SEP system, and the large solar arrays that it needs, might not be necessary as the maneuvers away from the periapsis are all relatively small. However, workers at NASA Langley Research Center have designed trajectories for Mars exploration that are similar in many principles to ours, but use a hybrid propulsion system and a lunar swingby for the final departure from (and capture back to) Earth, then using SEP between the Earth and Mars, to make up for the lower departure V-infinity from and to the Earth, that the lunar swingby can provide, relative to the higher V-infinities needed to reach Mars[26]. They optimized their trajectory and spacecraft so that refueling at Mars is not necessary, a very important property.

Below, we give our view of how relatively low-ΔV "cycler" trajectories from an EM-L2 halo orbit, that could be flown with a DST, might be used, but only considering impulsive ΔV's. Similar trajectories should be possible from (and to) an NRHO, but probably with a little higher ΔV. First, we will show how an LST (lunar lander) could reach any point on the Moon from an EM-L2 halo orbit. Next, we discuss the advantages of PhOR, with some detailed calculations using a trajectory to an NEA. Most of our trajectories were computed with the General Mission Analysis Tool (GMAT).[27] Other software used by SpaceFightSolutions is documented in the last two sections. We will present trajectories that could rendezvous with the NEA 2000 SG344 in 2029, including a comparison of high-thrust and hybrid (SEP, with some high-thrust maneuvers). We will present similar calculations to Mars for the 2033 opportunity.

LUNAR AND CIS-LUNAR EXPLORATION

Our previous papers showed how rapid and relatively low ΔV transfers could be made, from Earth to an EM-L2 halo orbit, and back to Earth, using powered lunar swingbys.[15,16] Adding a Mid-Course Correction (MCC) ΔV between the lunar swingby and the halo orbit usually reduced the overall cost.[14] A comparison of this "powered lunar swingby" technique with some other popular techniques is shown in Table 1.

Table 1. Comparison of LEO to EM-L2 Halo Transfers

Transfer Type	Flight Time, days	TTI* ΔV, m/sec	Total (post TTI) ΔV, m/sec
Direct	5	3150	1230
Powered Lunar Swingby	10	3129	308
Ballistic via WSB	140	3212	15
WSB with un-powered lunar flyby	173	3152	26

* TTI = Transfer Trajectory Insertion

Examining Table 1, all options have very similar TTI costs from LEO. It is clear that only the first two options are suitable for crewed missions, with a significant advantage in post-TTI ΔV for the 2nd option. The last two options, using a Weak Stability Boundary (WSB) transfer, are best for cargo missions where flight times can be large.

The most important for human missions are the transfers with a powered lunar swingby. The costs in Table 1 are one-way, but as shown in[15], a mirror-image trajectory with similar ΔV cost exists for the return trip, for a similar post-TTI cost and use of a capsule with heat shield for returning the astronauts to the Earth. Thus, the post TTI cost for a mission to and from the EM-L2 halo orbit is about 600 m/sec. Fig. 3 of[18] shows a similar trajectory that uses an NRHO rather than the small EM-L2 orbit, but the ΔV to either enter or depart the NRHO is about 100 m/sec more than from our small EM-L2 halo; they show the total post-TTI cost as 791 m/sec. That's well within the capability of Orion, but a smaller vehicle might be used if our smaller halo orbit is used, since the post-TTI cost would be nearly 200 m/sec less.

Starting in 2015, we began noting that a small outpost in an EM-L2 halo orbit could be used for lunar exploration goals, as well as serving as a staging area for human exploration beyond the Earth. We proposed that it be called the International Exploration Station (IES). We showed that any part of the lunar surface, even near-side locations, could be reached with similar ΔVs, as shown in Figure 2 (showing the similar trajectories from the EM-L2 halo orbit), Figure 3 (trajectories to the lunar destinations), and Table 2. Table 2 gives the ΔVs from the EM-L2 halo orbit to the lunar surface. The trajectories to the far side are direct, except of course for the MCC, that lowers the costs for all trajectories. But to reach most near-side locations, it's necessary to enter a low lunar orbit (we selected a circular one with mean altitude 50 km), then drop down from it to the landing. The velocity angle relative to the lunar zenith at the landing was restricted to 30°.

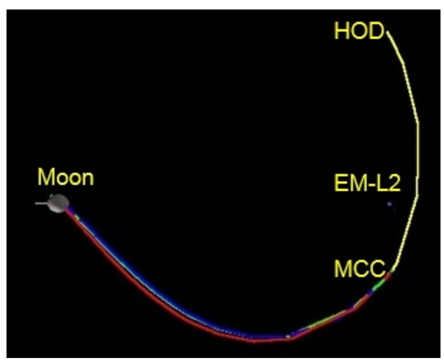


Figure 2. Trajectories from HOD to the Moon. Rotating Lunar Orbit View, fixed horizontal Earth-Moon line.

Primarily for accurate navigation, others have advocated that a low lunar orbit should always be entered, regardless of the surface destination, and a few revolutions completed in the orbit before performing a drop-down maneuver and landing.[28] In that case, the total ΔV costs for all three lunar targets would be similar to that for Rainer γ. We have computed similar trajectories to low lunar orbit, and then the surface, from an NRHO; the total cost was similar, 2532 m/s, or 55 m/s less than from the small EM-L2 halo to Rainer γ.

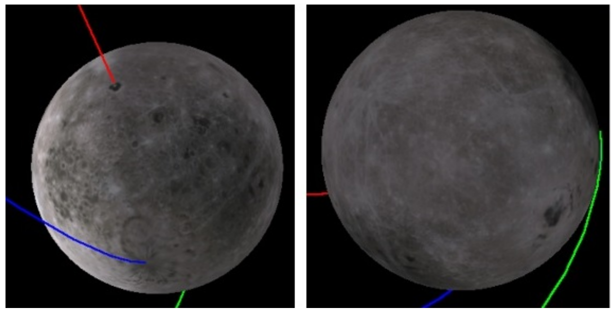


Figure 3. Trajectories near the Moon. Red to Tsiolkovsky, Blue to S. Pole, Green to Rainer γ.

Table 2. ΔV costs in m/s from HOD (18 m/s for all) to three lunar destinations

Lunar target	MCC	LOI	Drop	Landing	Total
Tsiolkovsky	113	0	0	2341	2472
South Pole	119	0	0	2342	2477
Rainer γ	114	663	1600	192	2587

Our work used a 7000-km Z-amplitude EM-L2 halo orbit, like those favored in[15] and[16]. But others also had an idea for an IES, which is now commonly referred to as just the Gateway, but earlier was called the Lunar Orbital Platform-Gateway and still earlier, the Deep Space Gateway. The orbit envisioned for these is a large-amplitude EM-L2 orbit, specifically a NRHO. It is expected to have a large SEP propulsion system such that it can be moved to other lunar orbits that are most suitable for different exploration goals. The reasons for selecting an NRHO for the Gateway are given in[29]. An NHRO with its aposelene high above the southern polar regions could provide communication between the Earth and Shackleton Crater at the South Pole over most of its orbit. But for communications, if a "Lunar Village" or other significant infrastructure is constructed on the Moon, it would be better to have three communications satellites about 120° apart in a relatively large-amplitude EM-L2 halo orbit, but with smaller amplitude than a NRHO. Such a system, along with Earth ground stations or GEO satellites, could provide continuous communication with all parts of the Moon, and of cis-lunar space. Then the Gateway orbit could be optimized solely for its current exploration goal.

PHASING ORBIT RENDEZVOUS

For the discussion on Phasing Orbit Rendezvous (PhOR), we start with a trajectory to the NEA 1994 XL1 that makes a close approach to Earth in 2022 that we calculated in 2013.[13] Figure 4 shows the whole heliocentric trajectory, in an inertial ecliptic-plane view. There will certainly not be a crewed mission following this year-long orbit and it is only a fast flyby. The asteroid, (480808) 1994 XL1, is of some interest because it has one of the smallest semi-major axes of any asteroid, less than that of Venus and with a perihelion just inside Mercury's orbit. Also, the post-injection ΔV is quite small, perhaps making it suitable for a smallsat or even large cubesat mission, but time is short. The trajectory is of more interest to us for the multiple possibilities of rendezvous from a LEO, during the last HEO orbits before the departure to the object. The total ΔV from the halo orbit and back to it, only 432 m/sec, is itemized in Table 3.

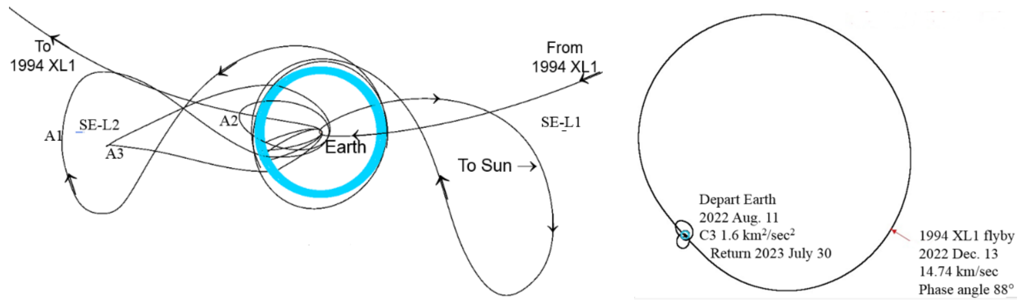


Figure 4. Left, solar rotating ecliptic-plane view showing both the departure and return (from and back to the EM-L2 halo orbit) near the Earth. The turquoise circle is the Moon's orbit, for scale. The right side is an inertial view, zoomed out to show the whole trajectory.

Table 3. ΔV's for the Figure 4 Trajectory to 1994 XL1, from the EM-L2 halo orbit and back.

ΔV Date	ΔV, m/sec	ΔV location	ΔV Date	ΔV, m/sec	ΔV location
2021 Sep. 21	0.1	HOD	2023 Dec. 14	9.4	1d after XL1
2022 Jan. 20	53	A1	2023 Jul. 30	110	P6 capture Δ/v
2022 Mar. 23	0.2	P1	2023 Sep. 19	17	A6
2022 Mar. 31	9.9	A2	2023 Nov. 09	25.5	P6
2022 Jun. 01	0.9	A3	2023 Nov. 29	25	HOI
2022 Aug. 11	180	P5 (to XL1)			

This trajectory uses slow trajectories to and from the lunar halo orbit, but during those long periods, the ITV (or DST) can be operated robotically. If a faster trajectory is desired, a powered lunar swingby several days after departure from the halo orbit could be used, but that adds almost 300 m/sec to the ΔV cost [for the outbound leg; an Apollo-style (or Orion) return capsule might be used for the direct return leg, for astronaut return].

We'll assume that the ITV is operated robotically until shortly before the Oberth (perigee) maneuver on 2022 Aug. 11 that sends the ITV away from the Earth. The three HEO orbits before the departure provide several weeks for that. as shown in some detail in Figure 5 and Table 4. The period of the ITV phasing orbit is 12 days. Opportunities for a Crew Transfer Vehicle (CTV) to rendezvous with the ITV with just one orbit occur on dates near the ITV perigees on 2022 July 19, July 31, and Aug. 11. The distance of the unpowered S3 lunar swingby is 10,289 km.

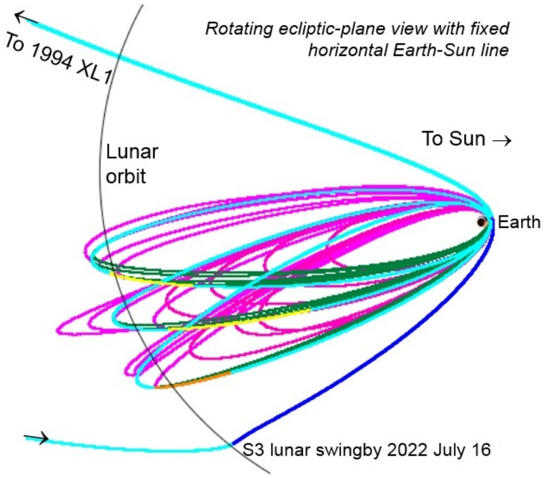


Figure 5. Multiple opportunities for rendezvous with the ITV (Transportation Hab) before the departure from Earth on 2022 Aug. 11. See text for explanation of the color-coded trajectories.

The light blue trajectory is that of the ITV, but dark blue from the S3 lunar swingby to the first phasing orbit perigee on July 19, and yellow or orange during the times when the CTV is staying with the ITV (for 2 days) for some CTV trajectories. The CTV trajectories are in pink outbound and dark green for its Earth return. The ITV last phasing orbit perigee on Aug. 11 has the 180 m/s Oberth ΔV to 1994 XL1.

Table 4. Phasing orbit rendezvous ΔV's for the trajectories shown in Figure 5.

CTV Departs Earth	C3, km2/s2	TTI ΔV, m/s	Apogee ΔV, m/s	Perigee ΔV, m/s	CTV-ITV Rendez-vous	Rendez-vous ΔV, m/s	CTV Departs ITV	Depar-ture ΔV, m/s	Total post-TTI ΔV, m/s.
July 17	-1.779	3144			July 24	340	July 26	34	374
July 18	-1.825	3142			July 24	206	July 26	34	240
July 19	-1.831	3142			July 24	44	July 26	34	78
July 20	-1.724	3147			July 24	246	July 26	34	280
July 21	-1.913	3138			July 26	297	July 28	83	380
July 22	-1.846	3141			July 26	548	July 28	83	631
July 22	-2.199	3125	20	20	Aug. 03	75	Aug. 05	23	138
July 24	-2.653	3104	20	40	Aug. 03	34	Aug. 05	23	117
July 25	-2.921	3092	42	60	Aug. 03	96	Aug. 05	23	221
July 27	-1.595	3153			Aug. 07	226	Aug. 09	119	345
July 29	-1.730	3146			Aug. 07	136	Aug. 09	119	255
July 31	-1.835	3142			Aug. 05	92	Aug. 07	51	143
Aug. 02	-2.067	3131			Aug. 08	261	Aug. 10	179	440

The CTV launch date, C3, and TTI ΔV from LEO are shown in the first 3 columns of Table 4. Apogee and perigee ΔV's are needed for some of the CTV orbits. The CTV rendezvous and departure dates and ΔV's are given in the next 4 columns, while the total post-TTI ΔV by the CTV is in the last column. The values in maroon are considered too high, but there are still 11 CTV launch dates where the total post-TTI ΔV is 380 m/s or less. In all cases, the astronauts have 2 days docked with the ITV, for transferring themselves and supplies to it. Following the CTV's departure, the CTV is operated robotically for a controlled atmospheric re-entry, so the CTV can be refurbished and re-used. The CTV capabilities are much less than Orion's, so the CTV can be smaller and launched with a rocket less powerful than an SLS. Primarily for this reason, we believe that PhOR is better than trying to rendezvous with the ITV (DST) at the NRHO for an interplanetary journey.

Qu et al.[26] also use PhOR for their clever trajectories from a lunar NRHO to Mars. They did not use the PhOR term, but described using lunar-distance highly elliptical orbits, or LDHEO's, that could be reached easily by astronauts during the almost 3 months that their spacecraft was in the LDHEO, designed with an inclination of 28° and low perigee altitude to facilitate the rendezvous. That is exactly the PhOR that we described in 2012[30], although the basic idea (called just "phasing" but not PhOR) was described at least as early as 2008[31]. But rather than use an Oberth maneuver to leave the LDHEO, Qu et al. instead used a lunar swingby to enter a higher 28-day orbit that sets up a final lunar swingby, providing an effective Earth departure C3 of about 2.0 in the right direction to go to Mars. Since a higher C3 is needed to reach Mars ballistically, their hybrid-propulsion spacecraft uses SEP to further increase the heliocentric orbital energy to reach Mars. After his presentation, Qu was asked about radiation dosage in the LDHEO's; he said that was not considered. However, their spacecraft, and any DST capable of being crewed, would be large, not like a mass-starved scientific mission with delicate sensors, and the crew (and any delicate sensors onboard) would need to have a safe place, in case of a large solar flare during the interplanetary cruise. The CTV that brings the crew to the ITV (DST) would need to protect the crew from passage through the radiation belts as well as Orion does. So we also did not consider the Van Allen belts for PhOR.

OTHER NEO FLYBY OPPORTUNITIES

Since the trajectory shown in the previous section is too soon to practically implement, we present Table 5, similar to Table 1 of[13], but with departures in 2026 rather than 2022-2023. The departure dates given in Table 5 are the dates of the last perigee of the phasing orbits when the Oberth maneuver is performed to go to the asteroid, so the actual departure from the halo orbit would generally be 4 to 6 weeks earlier. These dates are only for computational purposes, to demonstrate feasibility; they do not represent any real schedule, or any commitment to one. They show the rather frequent low-C3 opportunities, and these are expected to increase significantly as new surveys such as those by the Large Synoptic Survey Telescope become operational. We selected objects that are 150m or more in diameter, and have arranged the table in order of increasing C3, which is equivalent to ordering by the total ΔV, which in this case, is just the sum of the two Oberth maneuvers, the first being for departure to the asteroid and the second being for capturing the ITV back into a HEO with perigee geocentric distance 7000 km and apogee 65 Earth radii, a little beyond the Moon's orbit. About 500 m/s more ΔV would be needed for the powered lunar swingbys, and the halo orbit departure and return, but if the astronauts could rendezvous using a CTV during the phasing orbits before and after the Earth departure and return, respectively, then the extra cost could be much less since the ITV, without crew, could be transferred from and to the EM-L2 halo orbit using slow transfers, like those described in the previous section. Following previous methodologies[32,33] and using SpaceFlightSolution's Mission Analysis Environment software[34] , we found low-energy one-year-return trajectories to 12 asteroids that met the criteria described above. However, there is an additional constraint on our trajectories; the perigees of the departure HEOs must be close to the lunar orbit plane, in order for the ITV to depart from the EM-L2 halo orbit and use a powered lunar swingby for the approximately one month duration to the departure perigee. Even if the trajectory from the EM-L2 halo orbit takes 6 or more months via a WSB slow transfer passing near one of the Sun-Earth libration points, that trajectory needs to use an unpowered leading edge lunar swingby to enter into the phasing orbits that are needed for the astronauts to rendezvous with the ITV via the CTV. So although the WSB transfer can pass a relatively large distance from the lunar orbit plane, the phasing orbits will still be close to the lunar orbit plane, due to the lunar swingby that is needed to set up the phasing orbits. This lunar orbit plane constraint eliminated 4 of the 12 trajectories, leaving 8 trajectories in Table 5. This might not be a comprehensive list, but it includes what we consider to be some of the best opportunities from the NEO's discovered before 2016, including those with flyby velocities under 20 km/sec. The dates are given in the form YY Mmm DD.D where YY is the year - 2000; Mmm is the month, or its 3-letter abbreviation; and DD.D is the day of the month to the nearest tenth of a day. The C3's are all less than 2.6 km^2/sec^2 (so with some effort, the departure and/or return might be accomplished from/to a HEO using a lunar swingby) except for the last 3 cases. The return date is not given since in every case, it is exactly a year after the departure date. The re-entry speeds for the crewed capsule upon return are all in the 11.0 – 12.0 km/sec range. These opportunities are not just for crewed missions. They might be used by cubesats deployed from the Gateway, and could be returned there, refurbished, and flown to a new target.

Table 5: Selected NEO Flyby Opportunities with 1-year free return departing Earth during 2026.

2026 Departure Date	Departure C3, km^2/s^2	Total ΔV, m/s	Flyby Asteroid	Abs. Mag.	Diameter, meters	Flyby Date	Flyby Speed, km/s
Feb. 26.1	0.43	218	1997 NC1	18.0	668-1493	26 Jul. 2.8	8.69
Jul. 19.2	0.66	240	2008 LV16	20.3	232 - 518	26 Nov. 15.2	14.86
Jul. 24.5	0.90	262	2002 VE68	20.5	221 – 472	26 Nov. 19.5	7.70
Jun. 22.4	1.02	273	2014 BW2	19.9	278 - 622	27 Feb. 26.6	12.61
Mar. 27.0	1.48	316	2010 TH19	20.5	211 - 472	26 Dec. 3.3	6.33
Oct. 16.3	2.48	409	1998 CS1	17.7	767 - 1714	27 Jul. 24.2	17.77
Jul. 2.4	3.23	478	2009 EO2	19.6	320 - 715	26 Dec. 3.5	12.57
Mar. 12.4	3.80	531	Anteros	15.8	2000	26 May 29.0	6.50

TRAJECTORIES TO 2000 SG344

A half-year crewed mission to rendezvous with a NEA could serve as a good demonstration mission before making the much longer journeys to Mars. The small (estimated size 37m) Aten asteroid 2000 SG344 provides a good opportunity for this.

GMAT Trajectories

Figure 6 shows a trajectory from the EM-L2 halo orbit to this asteroid, and return to Earth, with a total impulsive ΔV of 1881 m/sec. This does not include capture back into the EM-L2 halo orbit; that should cost about 150 m/sec more. Figure 6 shows only one and a half phasing orbits before the Oberth maneuver departure. This should provide at least a few days of low-cost CTV rendezvous possibilities, but the halo orbit departure might be better a month earlier, which would give the same departure geometry needed with two or three more phasing orbits, providing many more PhOR opportunities. Table 6 gives the dates, sizes, and locations of the maneuvers. In the table, 2000 SG344 has been abbreviated to just "SG344" and "rend." Means rendezvous (with the asteroid). At the Earth return, the ITV performs the 142 m/s maneuver that captures it into a HEO with apogee a little beyond the lunar orbit. Then during 2 or more of these HEO phasing orbits, a CTV can rendezvous with the ITV and return the astronauts to Earth. After that, the ITV can target a lunar swingby to send it, uncrewed, to a slow WSB transfer to return to the EM-L2 halo orbit several months later. We looked at another option, where the astronauts return to Earth in a

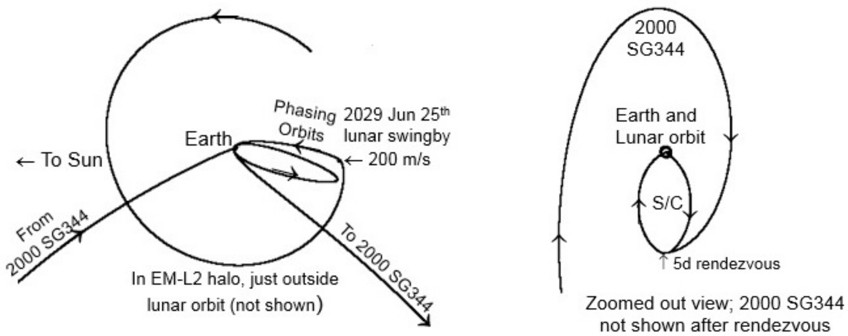


Figure 6. The trajectory to 2000 SG344 is shown in a solar rotating ecliptic-plane projection with a fixed horizontal Sun-Earth line. The left side shows the trajectory near the Earth while the right side is a zoomed out view that also shows part of the path of 2000 SG344 in this frame.

Table 6. ΔV's for the Figure 6 Trajectory to 2000 SG344, from the EM-L2 halo orbit and back Earth.

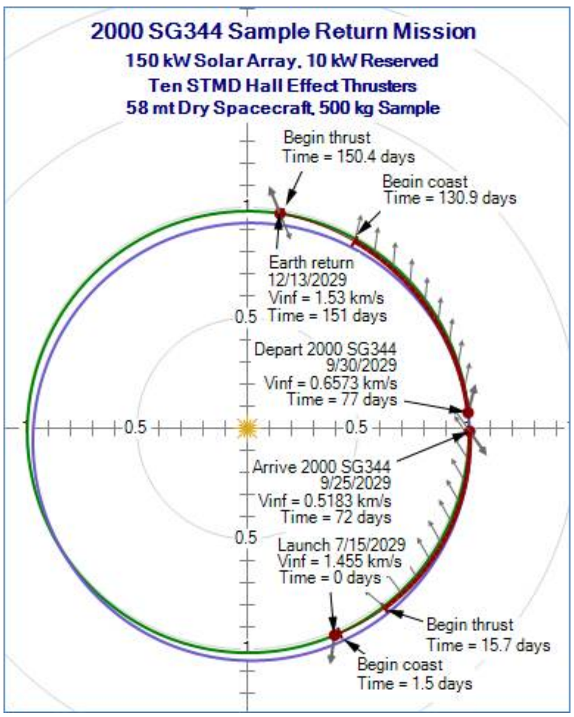
ΔV Date	ΔV, m/sec	ΔV location
2029 Jun. 17	8	HOD
2029 Jun. 18	55	MCC
2029 Jun. 25	200	S1
2029 Jul. 11	163	P2 (to SG344)
2029 Sep. 25	561	SG344 rend.
2029 Sep. 30	760	Depart SG344
2029 Dec. 25	142	Earth return

capsule with a heat shield for atmospheric re-entry into the Pacific Ocean. In that case, the ITV and capsule are targeted to the Pacific Ocean, but then the capsule is separated several days before. Right after the separation, the ITV performs an MCC to a perigee distance of 7000 km so that it does not enter the atmosphere. In that case, a smaller perigee ΔV could capture the (now uncrewed) ITV into a looser orbit for a direct slow WSB transfer to the halo orbit. But the overall ΔV savings are small due to the MCC. It would be more efficient to carry the extra fuel needed to perform the 142 m/s capture maneuver (about 50 m/s more than needed for the WSB capture) rather than carry the heavy heat shield and other return capsule hardware all the way to the asteroid and back to the Earth. Longer stay times at the asteroid are possible. If the stay time is 30d rather than 5d, and keeping a similar total duration of 180d, the total ΔV would increase by about 300 m/s, to about 2050 m/s.

MAnE and MAnE-EP Trajectories

Optimized trajectories to rendezvous with 2000 SG344 have also been computed with SpaceFlightSolution's Mission Analysis Environment (MAnE) and MAnE-Electric Propulsion

(MANE-EP)[34] as presented below. These programs model the major planets as zero sphere-of-influence point masses so the Moon and other natural satellites are not included. But in the interplanetary realm, the approximation is useful, providing good starting conditions for GMAT and other high-fidelity solfware. And the optimization is robust, providing early good estimates of propellant and spacecraft masses. The trajectories computed are Ballistic (high thrust only) and Hybrid (including high thrust and low thrust propulsion). We note, before going into the comparison of these two trajectories, there is an opportunity in 2027 to go to 2000 SG344 with a total ΔV of only 1090 m/s, from departing the 65 Re HEO on Mar. 15 to returning to a similar HEO on 2028 June 5. The trajectory was not used due to the flight time of 449d, almost 3 times the duration of our mission, and there seems to be little chance that a crewed mission to 2000 SG344 could be ready to depart in early 2027. But that opportunity might be good for a robotic mission; the stay time at the asteroid would be 60d. A hybrid trajectory computed with MAnE-EP is shown in Figure 7 and detailed in Table 7 below.



	Ballistic	Hybrid
Earth launch date	7/15/2029	7/15/2029
Initial mass in orbit (kg)	105,378.06	99,039.97
Launch excess speed (km/s)	1.380350	1.454522
Launch Δv (m/s)	177.871	187.636
Launch HT propellant (kg)	5,806.76	5,748.27
Outbound LT propellant (kg)	N/A	1371.85
Outbound flight time (days)	72	72
2000 SG344 arrival date	9/25/2029	9/25/2029
Arrival Δv, vinf (m/s)	697.666	518.3
Arrival HT propellant (kg)	19,848.56	13,994.23
2000 SG344 arrival mass (kg)	79,722.75	77925.63
2000 SG344 stay time (days)	5	5
Sample mass (kg)	500	500
2000 SG344 depart mass (kg)	80,222.75	78,425.63
2000 SG344 depart date	9/30/2029	9/30/2029
Depart Δv, vinf (m/s)	941.881	657.292
Depart HT propellant (kg)	20,800.66	14,820.29
Return LT propellant (kg)	N/A	1,293.59
Return flight time (days)	74	74
Earth return date	12/13/2029	12/13/2029
Return excess speed (km/s)	1.390208	1.53
Return Δv (km/s)	179.139	198.089
Return HT propellant (kg)	3297.09	3,811.74
Return mass in orbit (kg)	56,125	58,500
Mission duration (days)	151	151

Figure 7 on the left. The hybrid trajectory detailed in Table 7 (on the right) is portrayed in a heliocentric ecliptic-plane projection. 2000 SG344's orbit is blue, the Earth's is green, and the spacecraft's is red.

The assumptions for the above trajectory are:

- Array power at 1 AU = 150 kW with 10 kW reserved for non-propulsion purposes. The power drops off with distance from the Sun, as $1/r^2$ where r is the heliocentric distance in Astronomical Units.
- Array performance varies as the inverse square of distance
- Propulsion system consists of ten Hall effect thrusters, each with a maximum PPU input power of 13.254 kW operating at an Isp of 2290.18 sec, efficiency of 58.037%, and 90% duty cycle.

- Dry spacecraft mass = 58,000 kg (excludes high- and low-thrust propellant)
- Sample mass = 500 kg
- High-thrust Isp = 320 sec, velocity losses ignored
- Earth departure and return orbit = 7,000 x 414,579 km
- Ephemeris of Earth taken from the JPL DE430 file and a JPL spice kernel (.bsp) file for 2000 SG344

The problem posed is to minimize the mass in Earth orbit that delivers a final mass on return to Earth orbit equal to the dry spacecraft mass plus the sample mass (58,500 kg). To compare to an all-ballistic mission, the spacecraft dry mass is reduced by an estimate of the low-thrust power and propulsion system masses. For this purpose, we assume each of the ten thruster/PPU units weigh 50 kg and the array is presumed to have a specific mass of 12.5 kg/kW. These assumptions lead to a SEP power and propulsion system mass of 2,375 kg and a mass delivered back at Earth of 56,125 kg for the all-ballistic mission.

The data in Table 7 indicate that the hybrid mission offers a net reduction of initial mass in Earth orbit of about 6 mt. It is interesting to note that the excess speeds at Earth actually increase over those of the ballistic mission, but this is more than made up by the substantial reductions in HT propellant at asteroid arrival and departure (about 6 mt at each). Figure 7 shows the trajectory profile of the Pacific Ocean splashdown, while the ITV flys by at a distance of 7000 km to perform the 80.2 m/sec maneuver that captures it into a trajectory that goes to the WSB, setting up an EM-L2 halo orbit return in 2030 April. The Earth orbit used is a HEO, very similar to those used for PhOR with the earlier-presented trajectories. So if the ITV departs from, and returns to, an EM-L2 halo orbit, there would be an additional time of about 2 months (more if a WSB transfer were used) and an additional ΔV of perhaps 400 m/s (but only about 120 m/s .or less using a WSB transfer, which is fine if the ITV is uncrewed at that time).

TRAJECTORIES TO MARS

2033 provides one of the best opportunities to reach Mars during the 2030's. It might be used for the first manned mission to the red planet, or to one of its small moons.

GMAT Trajectory to/from Phobos

The left side of Figure 8 shows the departure from an EM-L2 halo orbit near the Earth, including 3 complete phasing orbits before the departure Oberth maneuver.

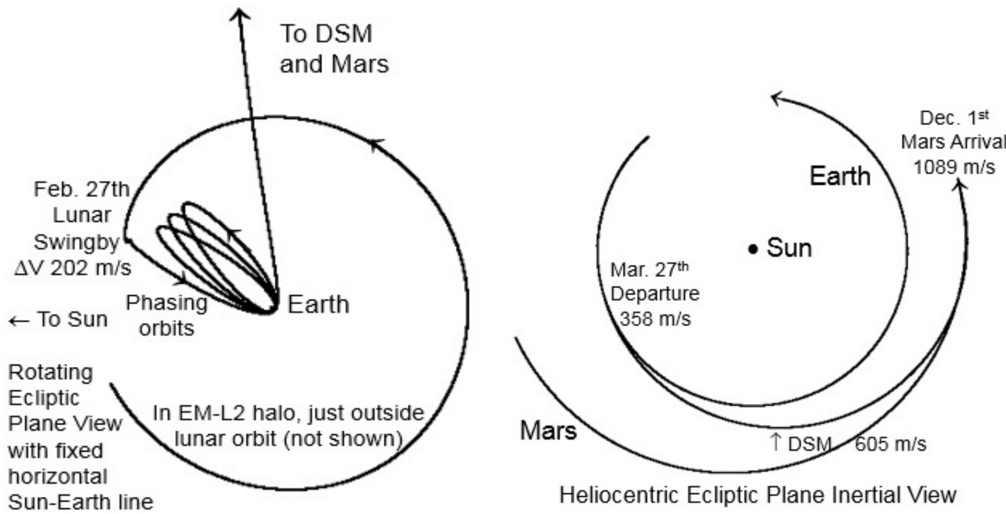


Figure 8. The trajectory to Mars near the Earth is shown in a solar rotating ecliptic-plane projection with a fixed horizontal Sun-Earth line on the left. To the right is an inertial ecliptic-plane projection zoomed out to show the whole heliocentric trajectory to Mars.

The right side of Figure 8 shows the heliocentric trajectory to Mars, including a Deep Space Maneuver almost 4 months after departure from Earth. Figure 9 shows the arrival trajectory near Mars, reaching Phobos. The Mars capture ΔV is at a periapse altitude of 300 km. The apoapse distance is 48 Mars radii. The 77 m/s apoapse maneuver raises periapse to the radius of Phobos' orbit.

Figure 9 shows the arrival trajectory near Mars, reaching Phobos. The Mars capture ΔV is at a periapse altitude of 300 km. The apoapse distance is 48 Mars radii. The 77 m/s apoapse maneuver raises periapse to the radius of Phobos' orbit.

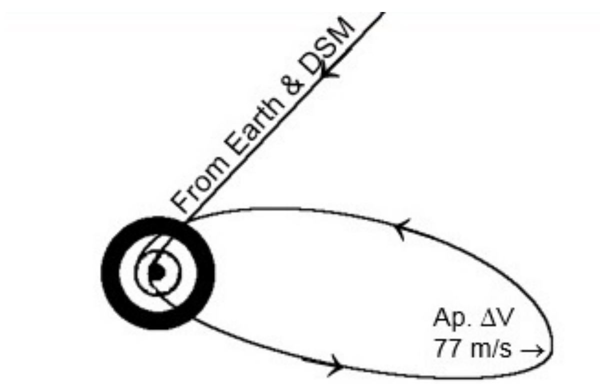


Figure 9. This shows the Mars arrival and Phobos rendezvous (the orbit of the outer satellite Deimos is also shown). This view and that of Figure 10 is an inertial Mars equatorial plane view.

The left side of Figure 10 shows the departure first from Phobos, and then from Mars.

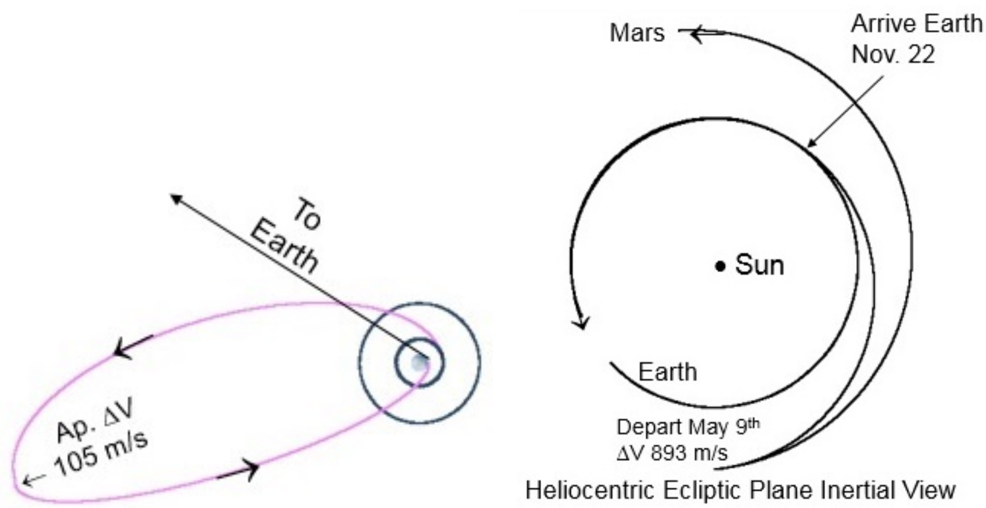


Figure 10. Both views are inertial. The left side shows the departure from the Mars system in 2035 July in the Mars equatorial plane. The right side is a zoomed-out view showing the whole return trajectory in an ecliptic-plane projection.

Figure 11 shows the arrival trajectory near Earth.

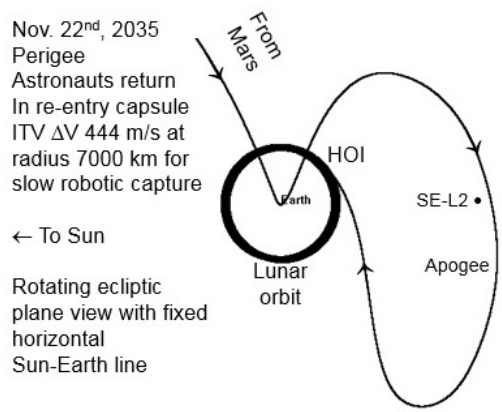


Figure 11. The return trajectory near the Earth.

Table 8. ΔV's for the Trajectory to Phobos shown in Figures 8-11, from the EM-L2 halo orbit and back.

ΔV Date	ΔV, m/sec	ΔV location
2033 Feb. 18	9	HOD
2033 Feb. 20	41	MCC
2033 Feb. 27	202	S1
2033 Mar. 04	13	P1
2033 Mar. 23	4	A3
2033 Mar. 27	358	P4, to Mars
2033 Jul. 19	605	DSM
2033 Dec. 01	1089	Mars Capture
2033 Dec. 05	76	Apoapse
2033 Dec 10	25	TA 290°
2034 Mar. 04	824	Phobos Rend.
2035 May 01	818	Depart Phobos
2035 May 01	1	TA 70°
2035 May 05	105	Apoapse
2035 May 09	893	Mars Per.
2035 Nov. 22	444	Earth return
2036 Feb. 17	45	Apogee
2036 Mar. 31	25	HOI

The 2035 May 09th maneuver is an Oberth maneuver at Mars periapse at a height of 300 km. The total ΔV is 5577 m/s, from the EM-L2 to Phobos and back to the EM-L2 halo. If a separate pre-positioned Mars vehicle could rendezvous with the ITV in the highly-eccentric Mars orbit, to carry the astronauts to and from Phobos, then over 1600 m/s of ΔV would be transferred from the ITV total, reducing it to a little under 4000 m/s, to that Mars vehicle. At the Earth return, the astronauts get into a capsule that separates from the ITV a day or two before arrival, for a splashdown in the Pacific Ocean. The ITV passes Earth at 7000 km distance, performing the 444 m/s maneuver robotically that captures the spacecraft into a loosely captured orbit. The information about the last two maneuvers is approximate.

MAnE and MAnE-EP Trajectories

Optimized trajectories to reach Mars, also during the 2033 opportunity, have been computed with SpaceFlightSolution's MAnE and MANE-EP programs as presented below. See some additional remarks at the start of the MAnE/MAnE-EP subsection for 2000 SG344 above.

Table 9 (Right). Key properties of the trajectory to and from Mars optimized two ways.

	Ballistic	Hybrid
Earth launch date	3/16/2033	3/27/2033
Initial mass in orbit (kg)	167,860.88	157,607.7
Launch excess speed (km/s)	2.36894	2.454872
Launch Δv (m/s)	348.74	367.491
Launch HT propellant (kg)	17,655.10	17,416.98
Outbound DSM date	7/11/2033	N/A
Outbound DSM Δv (m/s)	651.99	N/A
Outbound DSM propellant (kg)	28,178.94	N/A
Outbound LT propellant (kg)	N/A	5,185.30
Outbound flight time (days)	252.348	250.4
Mars arrival date	11/23/2033	12/3/2033
Arrival excess speed (km/s)	3.26330	3.456958
Arrival Δv (m/s)	1,055.48	1,449.119
Arrival HT propellant (kg)	34,853.24	49,930.12
Mars arrival mass (kg)	87,173.60	85,075.32
Mars stay time (days)	532.31	522.73
Sample mass (kg)	500	500
Mars depart mass (kg)	87,673.60	85,575.32
Mars depart date	5/10/2035	5/10/2035
Depart excess speed (km/s)	2.960997	2.447938
Depart Δv (m/s)	891.403	640.315
Depart HT propellant (kg)	21,679.50	15,794.91
Return LT propellant (kg)	N/A	3,621.15
Return flight time (days)	198	198
Earth return date	11/24/2035	11/24/2035
Return excess speed (km/s)	3.020993	2.535802
Return Δv (km/s)	508.326	386.109
Return HT propellant (kg)	9,869.09	7,659.25
Return mass in orbit (kg)	56,125	58,500
Mission duration (days)	982.66	971.13

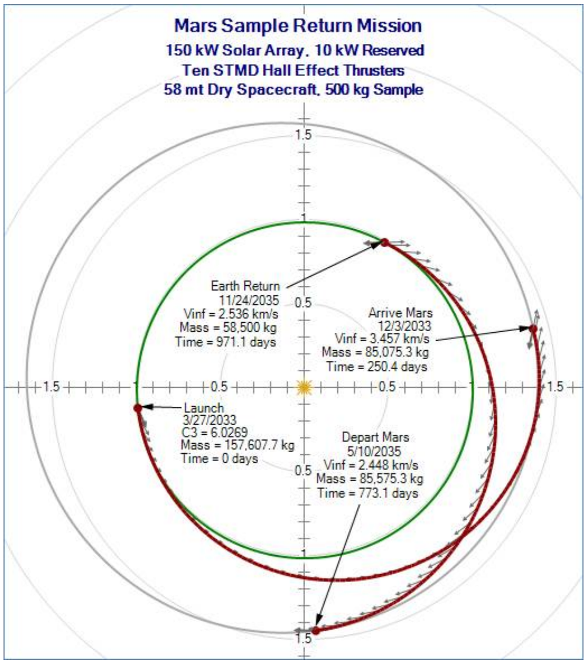


Figure 12 (Left). The hybrid trajectory detailed in Table 9 is portrayed in a heliocentric ecliptic-plane projection. Mars' orbit is blue, the Earth's is green, and the spacecraft's is red.

The propulsion system performance and spacecraft mass assumptions for the Mars mission are identical to those listed above for the mission to 2000 SG344. Additionally, the Mars capture orbit dimensions specified are 3,696km (300 km altitude) x 163,017 km (48 Mars radii). The Earth departure date for both the ballistic and hybrid missions were chosen such that perigee of the Earth escape hyperbola lies within the plane of the lunar orbit, a prerequisite for linking with the trajectory from the EM-L2 halo orbit. The departure and arrival dates for the return leg were optimized for the ballistic mission and used unchanged for the hybrid mission.

These data indicate that hybrid propulsion permits a reduction of about 10 mt of initial mass in Earth orbit. Although the hyperbolic excess speeds are higher for the hybrid mission at both departure and arrival of the outbound leg, the replacement of the DSM of the ballistic trajectory with the higher efficiency Hall thrusters results in a net reduction of over 8 mt of propellant consumed during the leg. It is also interesting to note that although the mission launch date for both propulsion modes were chosen such that the escape perigee lies in the lunar orbit plane, the dates nevertheless differ by eleven days. This is because the perigee locations change as a consequence of the different hyperbolic excess velocities of the two cases. The later launch date for the hybrid mission results in a nearly equal adjustment (10 days) in the arrival date for the leg. The hybrid mission would likely have a greater performance advantage over the pure ballistic mission with a larger SEP system; 252 kW and larger (at 1 AU) have been assumed by others[35].

RESULTS, DISCISSION, AND CONCLUSIONS

As noted previously, the ITV (or DST) can stay with the IES (or Gateway) near the Moon between missions to asteroids, comets, or Mars and its moons. The near-Moon orbit is an EM-L2 low-amplitude halo orbit in our system, but others have shown that NRHO's or other lunar orbits might be used. If there are extensive lunar exploration efforts, a system of three comsats spaced around a large-amplitude halo orbit should be used to provide 24/7 coverage of all of the Moon and cis-lunar space, freeing the IES and ITV to use the lunar orbit best suited for their goals. The reusability cycle is illustrated in Figure 13.

After leaving the lunar orbit, the ITV can perform a powered lunar swingby, like those shown in Figures 6 and 8 above; that provides the fastest transfer from the EM-L2 orbit to the departure trajectory. But if there is enough time, an uncrewed lower ΔV slow transfer, like that shown in Figure 4, is usually preferred. While in the HEO phasing orbits before the Earth departure, astronauts can use a CTV and rendezvous with the ITV, as described in the section on PhOR. Figure 13 is set up for Martian destinations, but Mars could be replaced with an NEA that simplifies what happens at the destination, only a simple rendezvous and departure.

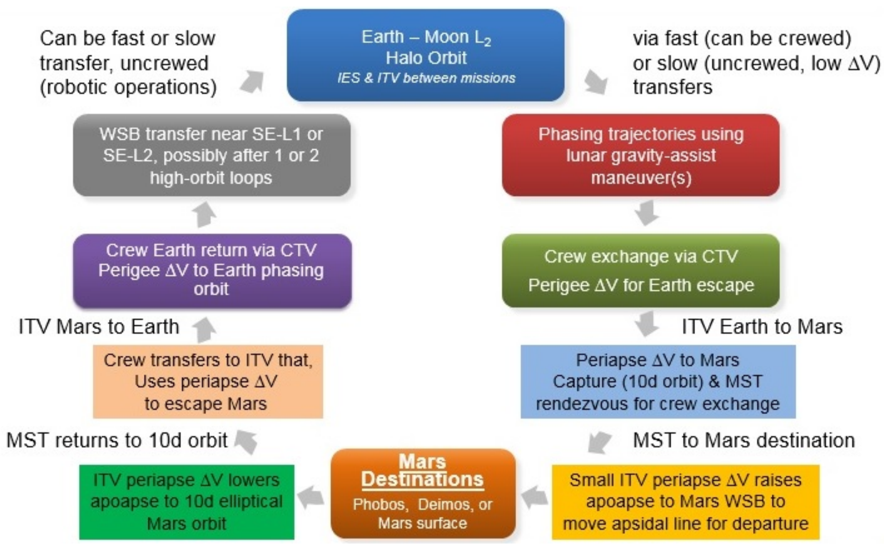


Figure 13. A Reusable Low-ΔV Cycler System for Human Deep Space Exploration.

At Mars, our GMAT trajectory simply rendezvoused with, and then left, Phobos, incurring one-way ΔV costs of around 900 m/sec. In the Mars MAnE/MAnE-EP section, the rendezvous highly elliptical orbit has an apoapse of 48 Mars radii, which has a period of 8.4 days. In Figure 13, we recommend a period of 10 days, which has an apoapse distance of 54 Mars radii, which is roughly equivalent. For a reusable system to work, there needs to be pre-positioned assets at Mars, especially a Mars Space Tug (MST) to rendezvous with the ITV, to transfer the astronauts to different Martian destinations. This will work best when fuel can be produced on Mars; the easiest is likely the methane and oxygen production envisioned by Elon Musk. We don't think it will be practical to manufacture xenon on Mars, so the Martian infrastructure systems would best be at least mainly high-thrust systems that can use Martian fuel.

After the rendezvous with the MST, the uncrewed ITV can raise its apoapse to the Sun-Mars WSB region about one million km from the planet. There, solar perturbations can be used to advantage, to move the apoapse point to the direction that will be needed for the departure trajectory, then the apoapse can be lowered to the ten-day orbit to keep it in that direction. The lead author used this technique successfully for moving the line of apsides for the proposed but unfunded Aladdin Phobos-Deimos sample return mission[36]. After the MST rendezvous with the ITV to transfer the astronauts to the ITV, the MST can do similar maneuvers, to raise its apoapse to the WSB to move its line of apsides to the direction needed for rendezvous with the ITV at the next Martian exploration opportunity. The basic idea of using a 10-day staging orbit in this way was published by Merrill et al. in 2015[35]. But another way of moving the line of apsides, by using Mars' oblateness perturbations and an "apotwist" maneuver near apoapse, was used in the recent work at Langley, which also used a 5-sol rather than 10-day highly elliptical orbit at Mars[26].

In conclusion, we believe that a robust program of human exploration beyond LEO can benefit from a reusable architecture using techniques such as phasing orbit rendezvous and other concepts detailed above. This constitutes a reusable cycler system that does not need the time-critical high ΔV's needed by hyperbolic rendezvous cyclers[37,38]. For our cycler to be viable, refueling at Mars is probably necessary. This refers to our Mars (actually, Phobos) trajectory computed with GMAT, where we only calculated the trajectory, without regard to spacecraft wet and dry mass, or propellant used. Mars refueling is not needed with the "cycler" using a SEP/chemical hybrid propulsion system described by Qu et al[26]. Since Qu et al calculated a different opportunity, with very different techniques, than we did for our simpler hybrid propulsion trajectory, with some unknown assumptions, it's not possible to compare the two in a meaningful way. If for any reason, SEP is not available or not wanted, then trajectories similar to the ballistic impulsive ΔV ones we describe should be considered. Much work remains, to link the zero sphere-of-influence trajectories to realistic cislunar staging areas, and more work is needed to select cislunar areas that are optimum for a wide range of exploration goals; we acknowledge that most others have already settled on the NRHO as the cislunar staging area[29] as it appears the best for accessing the ice in craters around the lunar South Pole. In any case, a viable program will need cooperation between the major aerospace government and commercial agencies.

Many of the ideas above were discussed and archived in a Future In-Space Operations telecon[39] where Dunham had more time to discuss them than he had at the conference at USC.

ACKNOWLEDGMENTS

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