

SOLAR SYSTEM HUMAN EXPLORATION AIDED BY LIBRATION-POINT ORBITS, LUNAR GRAVITY ASSISTS, AND “PHASING ORBIT RENDEZVOUS”

David W. Dunham⁽¹⁾, Robert W. Farquhar⁽²⁾, Natan Eismont⁽³⁾, Eugene Chumachenko⁽⁴⁾, Sergey Aksenov⁽⁵⁾, Yulia Fedorenko⁽⁶⁾, Roberto Furfaro⁽⁷⁾, John Kidd, Jr.⁽⁸⁾, and Nathan Mogk⁽⁹⁾,

⁽¹⁾KinetX, Inc., 7913 Kara Ct, Greenbelt, MD 20770, USA, +1-301-526-5590, david.dunham@kinetx.com

⁽²⁾KinetX, Inc., 9007 Parliament Dr, Burke, Virginia 22015, USA, +1-703-303-9824, robert.farquhar@kinetx.com

⁽³⁾Space Research Institute of Russian Academy of Science, 84/32 Profsoyuznaya Str., 117997, Moscow, Russia, +7 916 628 6139, NEismont@iki.rssi.ru

⁽⁴⁾ Moscow Institute of Electronics and Mathematics, Univ. “Higher School of Economics”, per. Trekhsvyatitelskiy B., d. 3, 109028, Moscow, Russia, +7-495-917-07-50, mmkaf@miem.edu.ru

⁽⁵⁾ Moscow Institute of Electronics and Mathematics, Univ. “Higher School of Economics”, per. Trekhsvyatitelskiy B., d. 3, 109028, Moscow, Russia, +7-499-235-20-24, saksenov@hse.ru

⁽⁶⁾ Moscow Institute of Electronics and Mathematics, Univ. “Higher School of Economics”, per. Trekhsvyatitelskiy B., d. 3, 109028, Moscow, Russia, +7-499-235-20-24, ijuwime@gmail.com

⁽⁷⁾ Dept. of Aerospace and Mechanical Engineering, University of Arizona, PO Box 210119, Tucson AZ 85721-0119, USA, +1-520-312-7440, robertof@email.arizona.edu

⁽⁷⁾ Dept. of Aerospace and Mechanical Engineering, University of Arizona, PO Box 210119, Tucson AZ 85721-0119, USA, +1-908-399-0377, jkidd@orex.lpl.arizona.edu

⁽⁷⁾ Dept. of Aerospace and Mechanical Engineering, University of Arizona, PO Box 210119, Tucson AZ 85721-0119, USA, +1-520-312-7440, nmogk@email.arizona.edu

Abstract: A viable program of exploration beyond the Moon will need international collaboration, like for the ISS, and reusable spacecraft. High-energy Earth orbits that can be drastically modified with lunar swingbys and small propulsive maneuvers are used, especially near the collinear Sun-Earth and Earth-Moon libration points. This work builds on ideas developed by the International Academy of Astronautics’ exploration study group. The first human missions beyond low-Earth orbit may go to the vicinity of the translunar Earth-Moon libration point, although a lunar distant retrograde orbit (DRO), as envisioned as the destination for the small asteroid (or asteroidal boulder) returned by the proposed Asteroid Redirect Mission may also be used. This paper will concentrate on the next possible step, the first one into interplanetary space, a one-year return mission to fly by a Near-Earth Object (NEO). Details are presented of a trajectory that leaves a halo orbit about the Earth-Moon L2 libration point, then uses three lunar swingbys and relatively small propulsive maneuvers to fly by the approximately 200m asteroid 1994 XL1, and return to the Earth-Moon L2 halo orbit for a ΔV of only 432 m/s. Next, rendezvous missions to some other NEO’s will be presented. Finally, trajectories to reach Mars, first to Phobos or Deimos, will be outlined. The study uses highly-elliptical Earth orbits (HEOs) whose line of apsides can be rotated using lunar swingbys. The HEO provides a convenient and relatively fast location for rendezvous with crew, or to add propulsion or cargo modules, a technique that we call “Phasing Orbit Rendezvous”.

Keywords: Human space exploration, libration-point orbits, gravity assist trajectories, Near-Earth Asteroids, Mars.

1. Introduction

Human exploration beyond the Moon may be made possible with staging in high-energy orbits (such as in the Earth-Moon L2 region), as explained in large part in reference [1] and including what we now call “phasing orbit rendezvous”, or PhOR, essentially using the techniques that have already been proven with the third International Sun-Earth Explorer (ISEE-3) [2], the WIND double lunar swingby trajectory [3], studies for the proposed Relict-2 mission [4], and the STEREO phasing orbits [5]. The economic realities of the world today make it very unlikely that one nation could accomplish a viable and sustainable program of human Solar System exploration. This will need to be an international program like the International Space Station. There is already an international framework, with the exploration study group of the International Academy of Astronautics (IAA) that largely endorses these ideas. Reference [1] was presented as a paper at an IAA exploration working group meeting that was held during the International Astronautical Congress in 2008; it was an extension of a 2004 IAA exploration study [6]. Our collaborative effort is developing these ideas in more detail, to prove their feasibility with full force-model simulations. There will be an emphasis on NEO missions, and later on formulating optimal strategies for deflecting potentially hazardous objects (PHO’s).

The Apollo program showed us the value of looking at the design goals as a whole and exploiting the benefits of more efficient trajectories whenever possible. Mass and ΔV savings from staging made possible by using optimal combinations of trajectories can be quite significant, even enabling, so trajectory designs are of paramount importance in any architecture plan. Without the concept of Lunar orbit rendezvous, the Apollo program might never have been successful. These lessons are not being ignored in our analyses.

Candidate detailed trajectories, to prove feasibility and assess realistic ΔV requirements, will be developed for an approach to extend human exploration beyond the Earth-Moon environment using “stepping stones” described below. First, a capability will be developed to easily transfer from highly elliptical Earth orbits to other destinations throughout the Earth’s sphere of influence, out to the vicinity of the Sun-Earth L1 and L2 libration points, and to the vicinity of the Moon, and the Earth-Moon libration points. Next, before humans start to venture beyond the Earth-Moon system, we should learn more about NEO’s and their threat to Earth via a series of robotic missions that will survey the NEO population more thoroughly than can be done from Earth, and learn more about these objects. Finally, manned missions can visit NEO’s, test deflection strategies, and travel to Phobos, Deimos, and Mars. Our planned approaches for calculating optimal trajectories for these “stepping stones”, including the first results for a one-year asteroid flyby mission (whose purpose would be more for testing human interplanetary exploration for that duration, and for navigating towards a small body, than for scientific research), are described in the next sections.

2. The Earth's Sphere of Influence

2.1. Earth Orbit to an Earth-Moon L2 Halo Orbit

A return to the vicinity of the Moon, the natural celestial body closest to the Earth, will likely be the first destination for a crewed mission. However, before (or even instead of) landing on the Moon, it is more likely that a halo or Lissajous orbit about one of the collinear Earth-Moon libration points will be the next destination, and these orbits can be used as a transportation node. Rather than enter a libration-point orbit, the first mission might just loop around the libration point and return to the Earth, as described in an earlier paper [7]. A next step might be to rendezvous with an uncrewed vehicle that has been placed in an Earth-Moon L2 (EM-L2) halo orbit; details of 18-day and 31-day missions that could accomplish that, including abort options and the possibility to tele-operate an uncrewed vehicle on the surface of the back side of the Moon, are given in another paper [8].

An EM-L2 halo orbit has advantages for lunar exploration, including continuous visibility from the Earth and communication access to most lunar far-side locations most of the time [8]. A station in an EM-L2 halo orbit would serve as a good transportation node to support shuttle missions to and from lunar orbit and the Moon's surface [9,10], and it would be useful as long as lunar exploration goals are pursued. But EM-L2 might not be the best location for a permanent node for travelling to interplanetary destinations, or for near-Earth purposes such as servicing large space telescopes in Sun-Earth L2 (SE-L2) orbits. An Interplanetary Transfer Vehicle (ITV) might be assembled at a station in an EM-L2 halo orbit, but it could be transferred to other locations for storage between missions. For the first explorations, we are assuming that an ITV would depart from an EM-L2 halo orbit, and might be "parked" there between at least the first 2 or 3 missions beyond the Earth's sphere of influence. Another early destination might be a stable distant retrograde orbit (DRO) about the Moon, where a small asteroid might be placed with NASA's proposed Asteroid Redirect Mission (ARM) [11]. It should be easy to reach such an orbit from an EM-L2 halo orbit, and a lunar DRO might serve as a transportation node almost as good as an EM-L2 halo orbit (and the stability of a lunar DRO has the advantage of not needing small station-keeping maneuvers approximately twice a month that the lunar halo orbit needs), but we have not computed such trajectories yet.

2.2. Earth Orbit to the Sun-Earth L2 Point

Transfers between a highly elliptical Earth orbit and the vicinity of the Sun-Earth L2 (SE-L2) libration point can be accomplished easily via multiple lunar swingbys, a technique utilized by ISEE-3, Wind, and especially the Wilkinson Microwave Anisotropy Probe [12]. The Sun-Earth L2 libration point region is important, already the destination for several current and planned missions [13,14]. A capability to repair the James Webb Space Telescope and other expensive assets planned for that region could be valuable. Fig. 1, which is derived from Fig. 8 of Ref. [1], is a schematic that shows how such a mission might be accomplished with a "Deep Space Shuttle" (DSS) that might be derived from Orion hardware.

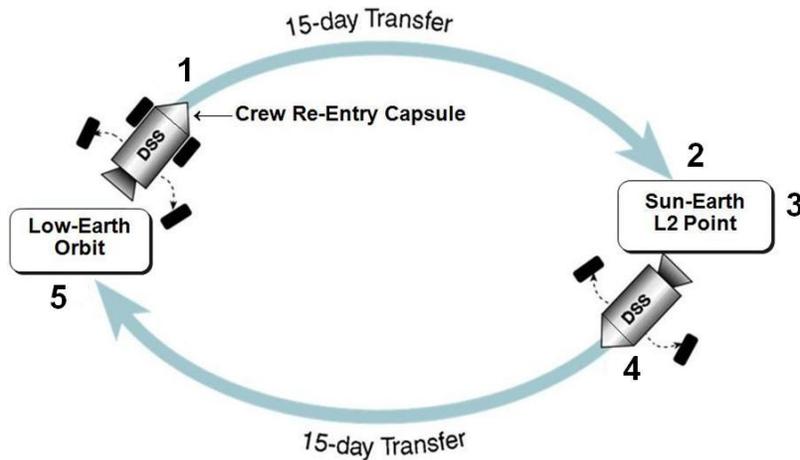


Figure 1. Deep Space Shuttle (DSS) concept for crewed missions to and from the vicinity of the Sun-Earth L2 libration point

The five large numbers in Fig. 1 refer to the events listed below:

- (1) DSS leaves low-Earth orbit ($\Delta V \sim 3230$ m/sec). First set of drop tanks discarded. (Alternative: use expandable high-performance kick stage for injection into SE-L2 transfer orbit.).
- (2) DSS enters SE-L2 orbit ($\Delta V \sim 900$ m/sec).
- (3) DSS services L2 telescope (stay time ~ 5 days).
- (4) DSS exits SE-L2 orbit ($\Delta V \sim 900$ m/sec). Second set of drop tanks discarded.
- (5) Crew returns to Earth in re-entry capsule. DSS returns to low-Earth orbit using multiple aerobraking maneuvers.

An “Interplanetary Transfer Vehicle” (ITV) can be built in an elliptical Earth orbit easily accessible to astronauts, and then transferred with a lunar swingby to the vicinity of the SE-L2 point. But more likely, the ITV will be constructed instead in an orbit about the EM-L2 point. The ITV can be built incrementally, starting with a capacity to support 3 or 4 astronauts for a year (it could be derived from the DSS) for the first NEO flyby mission. Following a NEO rendezvous mission, modules can be added to increase the ITV’s capacity for supporting perhaps 5 or more astronauts for the three years needed for a mission to Mars or its moons. This paper concentrates on the basic orbital concepts; we leave the detailed hardware design to others.

2.2. Earth Orbit to the Sun-Earth L2 Point

Although nominally based in an Earth-Moon L2 halo orbit (or possibly sometimes in an SE-L2 orbit), where the ITV can be uncrewed and robotically controlled most or all of the time, the ITV can use lunar swingbys to travel to reach other locations in Earth-Moon space and beyond with relatively little ΔV expenditure. The lunar gravity assists and small ΔV maneuvers would move the ITV to a highly elliptical Earth orbit (apogee from 50 to 90 Earth radii, or “Re”) that would line up with the departure asymptote of a trajectory to a specified destination. Astronauts would rendezvous with the interplanetary vehicle while it is in the elliptical orbit one or two orbits

before departure, when fuel tanks and other supplies could also be added; we call this “phasing-orbit rendezvous” (PhOR).

Possible destinations within the Earth’s sphere of influence that could be reached by this technique, most of which will be investigated later, include:

- Orbits around the Sun-Earth L1 point and the Earth-Moon L1 points
- Elliptical lunar orbits (periselene altitude ~100 km)
- Double-lunar swingby trajectories

Double-lunar swingby (DLS) trajectories alternately raise (to distances near the Sun-Earth L1 and L2 distance, about 240 Re or 1.5×10^6 km) and lower the elliptical orbit apogee, while advancing the line of apsides at about the same rate that the Earth moves around the Sun. DLS trajectories are useful for studying various regions of the Earth’s extended magnetic field and were used extensively by the ISEE-3, Geotail, and WIND missions.

Transfers between a highly elliptical Earth orbit and the vicinity of the SE-L2 libration point can be accomplished easily via lunar swingbys, a trailing-edge lunar swingby to reach the L2 point, and a leading-edge swingby to decrease the orbital energy to return from near the L2 point. Such trajectories could be used by astronauts to repair space observatories orbiting the L2 libration point, as one practical application. There is a family of solutions to this problem, one of them calculated with patched-conics shown in Fig. 2. Similar DLS trajectories involving multiple lunar swingbys, computed with realistic full-force models, were successfully flown first by ISEE-3, then by the Geotail and WIND missions, among others.

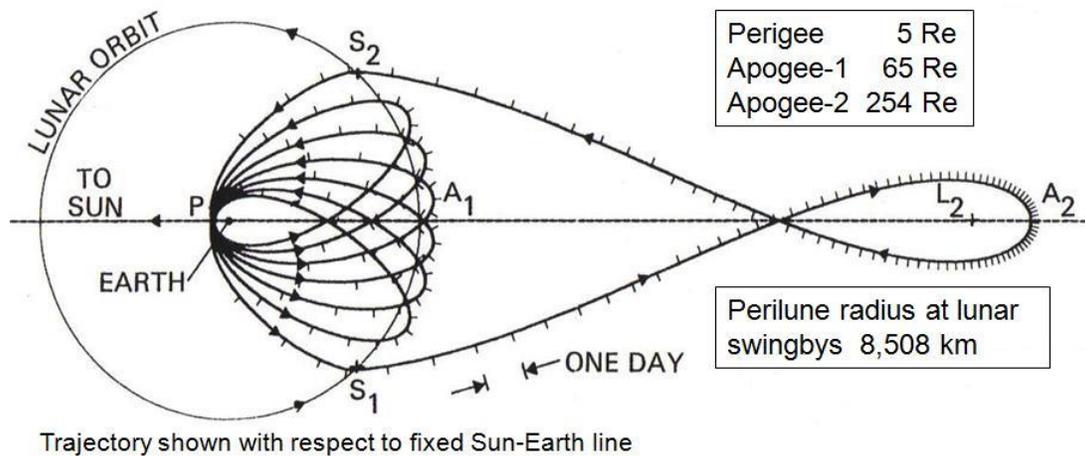


Figure 2. Simplified Version of a Double Lunar-Swingby Trajectory (patched conic calculation)

Large structures can be built up in the elliptical Earth orbit, which with a period of about 12 days would be easily accessible to astronauts, then transferred with the S1 lunar swingby to the vicinity of the Sun-Earth L2 point. Similarly, a large robotic space observatory in an L2 halo orbit could be moved with little ΔV out of the halo orbit to a trajectory similar to that shown in Fig. XVII where an S2 leading-edge lunar swingby would put it in the elliptical Earth orbit

where astronauts would have 2 or 3 months to make repairs before an S1 trailing-edge swingby would return it to L2. In a similar fashion, an ITV could be assembled in the elliptical Earth orbit and transferred to a “storage” or staging orbit near the SE-L2 point for possible future use to more distant destinations described in the next sections; alternately, it could be constructed and/or “stored” in an EM-L2 orbit, and then transferred to an SE-L2 orbit with only small ΔV costs [15,16]. Astronauts could reach the vicinity of SE-L2 relatively quickly, in about 2 weeks, using small vehicles with reasonable ΔV costs¹. Similar transfers to EM-L2 are also easy and need 10 days or less [7,8].

3. Beyond the Earth’s Sphere of Influence

3.1. Interplanetary Transfer Vehicle (ITV)

A general mission scenario using L2 staging is outlined below. The mission sequence begins with the ITV (sans crew) departure from the L2 orbit. Small propulsive maneuvers and lunar gravity-assists are used to target the final perigee ΔV maneuver. Approximately two to three weeks before the Earth-escape maneuver, a “taxi” (it might be the DSS described above, or derived from it, so we just call it the DSS below) is used to transfer the crew from LEO to the ITV in its elliptical Earth orbit. At the proper time in LEO, a disposable propulsive stage, such as a solid rocket engine, inserts the DSS into a HEO whose line of apsides and apogee distance closely matches that of the ITV. The perigee altitudes of the the DSS’s HEO and the ITV will generally differ, so a relatively small ΔV in the high part of the HEO can then match the state vector of the ITV for the rendezvous. This is the basic idea of PhOR. When the crew transfer has been accomplished, the DSS uses multiple aerobraking maneuvers to return to LEO. The ITV with the crew then executes the escape maneuver and proceeds to its interplanetary destination. After completing its mission, the ITV returns the crew to the Earth’s vicinity where the crew returns directly to the Earth’s surface in a re-entry capsule. The ITV then performs a perigee maneuver for a loose (highly elliptical) Earth orbit capture. The period of this orbit could be changed with a small perigee burn so that the ITV crosses the Moon’s orbit when the Moon is there. The resulting lunar swingby could be targeted (using small propulsive maneuvers, near perigee to change the timing, and thus the distance from the Moon, and near apogee to change the out-of-plane motion) to transfer to the desired L2 “parking” orbit (either Sun-Earth or Earth-Moon), or the swingby could start a double-lunar swingby sequence that might allow capture into the desired L2 orbit later with a better geometry. The basic idea of use (and re-use) of the ITV is illustrated in Fig. 3, modified from Fig. 10 of Ref. [1] to show that the “parking” location is EM-L2, which may be likely for at least the first few missions.

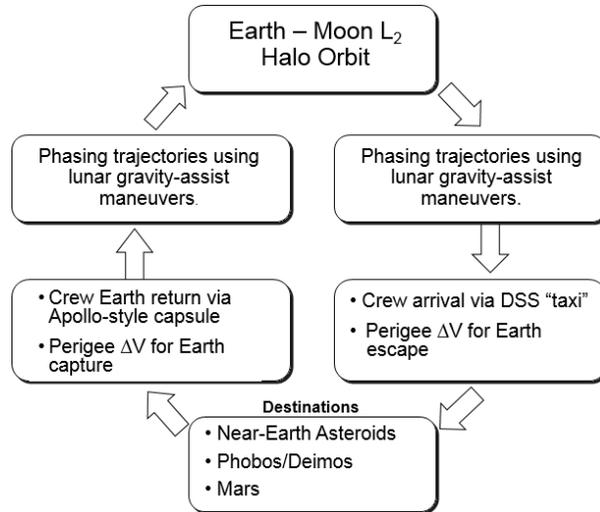


Figure 3. Reusing the ITV. The “crew arrival via DSS taxi” is phasing orbit rendezvous (PhOR) described earlier.

An initial habitat used for operations near the EM-L2 point might be transferred to the SE-L2 region for servicing space telescopes there, or for better positioning for departure to other destinations. A lunar swingby could be used to put the habitat in a highly-elliptical Earth orbit (HEO, probably the “inner loop” orbit of a DLS sequence) where astronauts could easily reach it, to add structures and propellant tanks to transform it into an ITV. Perhaps more likely, a habitat would be left in an EM-L2 halo orbit for lunar exploration, while a separate module assembled there could be moved to HEO (inner loop of a DLS) where it would be built up to form the ITV. The DLS sequence could be used to align the line of apsides to coincide with an outgoing hyperbolic asymptote to reach an interplanetary destination, with the help of a perigee ΔV , sometimes called an Oberth maneuver. Small propulsive maneuvers near perigee and apogee can change the B-plane points for the lunar swingbys of the DLS sequence to align the line of apsides properly, a process quite similar to that used for ISEE-3’s DLS sequence that sent that spacecraft to Comet Giacobini-Zinner in 1985. The DLS technique moves the line of apsides at approximately the same rate as the Earth moves around the Sun, and that is more exactly the case for storage in a Sun-Earth L2 orbit. But with the DLS technique, you can just stop the lunar swingbys as soon as the line of apsides is in nearly the desired direction in inertial space, and adjusting the timing to avoid later lunar encounters will keep it there. A wide variety of DLS orbits with different periods (multiples of a month) and different geometries allow targeting the desired perigee conditions for an interplanetary departure using only small ΔV ’s. The direction of a departure asymptote can be in virtually any direction, and the strategy described above can achieve that. For NEO’s, the departure from Earth could occur on either an inbound (before perihelion) or outbound (after perihelion) leg of the elliptical transfer orbit to the target. For Mars and objects with larger semi-major axes, the departure direction will generally be in the Earth’s leading edge (6 am) direction since the heliocentric velocity needs to be increased, while for Venus and Mercury, the departure would generally be in the trailing (6 pm) direction to decrease the heliocentric velocity.

As early as 1969, it was suggested that an ITV could operate between the Sun-Earth and Sun-Mars collinear libration points, with other vehicles used to transfer crews between Mars and the Sun-Mars collinear points [17]. This additional staging would produce large reductions in the round-trip ΔV requirements for an Earth-Mars ITV, and could be advantageous for human flights to Mars on a regular basis.

3.2 Missions to Near-Earth Asteroids (NEAs)

A first detailed trajectory to an NEA, to begin the process to prove feasibility and assess realistic ΔV requirements, has been developed for this approach to extend human exploration beyond low-Earth orbit (LEO) using the “stepping stones” described above [8,18,19] and elsewhere, described below.

Currently, rather than travel to a NEO beyond the Earth’s sphere of influence, NASA is considering a robotic mission, the ARM mentioned above, to move a small asteroid into a high stable lunar orbit where astronauts could study it [11]. The idea is to develop efficient large low-thrust propulsion systems (to move the heavy asteroid) that could be used later to support human interplanetary exploration, while providing an early opportunity for astronauts to study a small NEO in a relatively short time (a month or less) without the need to venture beyond the Earth’s sphere of influence. Although this is a clever idea that could have applications for planetary defense (by moving small asteroids that pass near the Earth to a different point on the B-plane such that it is targeted to hit a larger threatening asteroid) [20,21], it is controversial and at the moment, it is not certain that funding will be provided to carry out the mission [22,23]. In this paper, we do not comment on the merits of NASA’s ARM, or take one side or the other in the current debate about it, but we do endorse a human mission to travel beyond the Earth’s sphere of influence to reach a NEO, first with a flyby whose goal would be mainly as an engineering and navigation test flight, with science only a secondary goal. Following that, we propose using the ITV for a NEO rendezvous, but first, we discuss a possible flyby mission.

3.2.1 Human Near-Earth Asteroid Flyby

We assume that an ITV is assembled in an EM-L2 halo orbit, and then will be moved from that location to a DLS orbit whose perigee can be aligned with the departure asymptote of a hyperbolic trajectory that flies by a NEO on a one-year return trajectory. A trajectory from the Earth to the EM-L2 halo orbit is shown in Fig. 4, similar to Fig. 7 of Ref. [7]. The halo orbit shown in that figure is the starting point for our ITV to go to an asteroid.

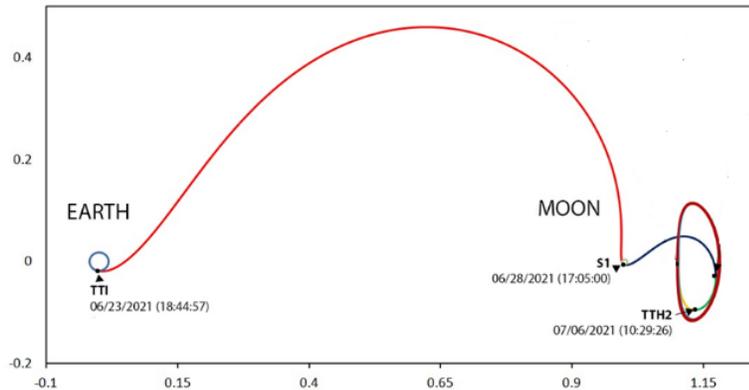


Figure 4. Transfer from the Earth to an EM-L2 halo orbit, 2021 June 23 – July 6, plotted in a rotating lunar orbit plane view with fixed horizontal Earth-Moon line. ΔV maneuvers are listed in the top part of Table 2 on page 11.

Shorter missions exist but are either not as frequent as one-year returns, or they involve very small asteroids that may pose navigation challenges, or they require a larger departure ΔV . We assume that departure from the halo orbit will be later than September 1, 2021, so we considered flyby opportunities with Earth departures between April 1, 2022 and April 1, 2023, to give enough time to orient the departure with a small amount of ΔV . These dates are only for computational purposes, to demonstrate feasibility; they do not represent any real schedule, or any commitment to one. Also, we selected objects that are likely to be 100m or more in diameter. Following previous methodologies [24,25] and using SpaceFlightSolution’s Mission Analysis Environment software [26], we found low-energy one-year-return trajectories to 18 asteroids that met the criteria described above. However, when we tried to calculate a trajectory for one of them using realistic force models with the General Mission Analysis Tool (GMAT) [27], high ΔV costs were encountered at the departure perigee since it was not very close to the lunar orbit plane, in which DLS trajectories stay. So we repeated the selection process described above to include higher flyby speeds and smaller-sized targets, but added a requirement that the departure perigee direction had to be in the average lunar orbit plane; in general, this wasn’t the case, but by considering launch dates other than the one with minimum C3, we could usually find a trajectory whose perigee met the-lunar-orbit-plane requirement. Ten trajectories that met the new requirements are listed in Table 1 in the timeframe of interest. This might not be a comprehensive list, but it includes what we consider to be some of the best opportunities from the currently-known NEO’s, 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 $2.6 \text{ km}^2/\text{sec}^2$ (so with some effort, the departure and/or return might be accomplished from/to a HEO using a lunar swingby) except for 2013 NK4 in 2022 August ($C3 \text{ } 5.2 \text{ km}^2/\text{sec}^2$) and 2006 UC185 in 2023 March ($C3 \text{ } 3.3 \text{ km}^2/\text{sec}^2$). So if the departure perigee velocity direction can be perfectly matched with the HEO perigee, then the ΔV needed to depart from a HEO with perigee and apogee distances of 7000 km and 65 Re, respectively, are shown in the 2nd column. The ΔV to recapture the ITV upon return will be a little less since it can be captured into a HEO with a higher apogee than that of the departure HEO. The 3rd column gives the geocentric inclination of the departure orbit to the ecliptic plane of J2000. 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.

Table I: Selected NEO Flyby Opportunities with 1-year free return departing from 2022 April to 2023 March.

Departure Date	Perigee ΔV m/s	Geo-centric Incl., $^\circ$	Flyby Asteroid	Abs. Mag.	Diameter, meters	Flyby Date	Flyby Speed, km/s
22 Apr. 09.2	205	23.0	2005 XX77	23.9	45 - 101	22 Nov. 29.9	6.62
22 June 05.0	110	8.3	2001 EU29	19.8	292 - 652	22 Sep. 25.7	17.08
22 Aug. 08.9	162	17.5	1994 XL1	20.8	64 - 143	22 Dec. 12.5	14.78
22 Aug. 07.5	331	17.7	2013 NK4	18.8	456-1019	23 Apr. 10.8	15.03
22 Aug. 16.6	180	19.2	2005 WJ56	18.1	651-1456	23 Jan. 18.9	11.35
22 Nov. 06.2	132	18.0	1998 MZ	19.3	361 - 807	23 June 09.1	16.65
22 Nov. 20.9	149	14.5	2007 HA	20.4	225 - 504	23 Apr. 24.7	19.82
23 Jan. 01.8	146	6.2	2004 HS56	22.2	98 - 219	23 May 09.1	16.59
23 Feb. 13.6	157	19.6	2002 JX8	19.8	290 - 648	23 May 24.9	7.15
23 Mar. 25.3	245	24.5	2006 UC185	23.5	54 - 121	23 Oct. 15.3	8.26

We found that the least costly way (lowest ΔV) to move an ITV from an EM-L2 halo orbit to a DLS orbit is to transfer the large vehicle to the vicinity of SE-L2 (or SE-L1), since that can be accomplished for almost no ΔV , although the half-year period of an EM-L2 orbit means that the transfer will take several months [15,16]. Our departure from the EM-L2 halo orbit was only 0.1 m/sec, of the order of typical halo orbit stationkeeping maneuvers²⁸. Due to the long time for this trajectory and a desire to depart the EM-L2 halo orbit in 2021 September, we skipped the first two cases listed in Table 1 and concentrated on the next (3rd) case, to asteroid 1994 XL1. An original version of Table 1 was in chronological order by departure date, but some of the departure dates moved as we more accurately calculated the perigees in the lunar orbit plane; case 4 to 2013 NK4 has a slightly earlier departure than our selected case 3. Although the case 4 asteroid is larger, its departure C3 is atypically large, so we decided to keep case 3 for the GMAT calculations with a realistic force model, which includes point mass gravity of the Earth, the Sun, and the Moon. As we varied the ΔV leaving the halo orbit by very small amounts, we looked at how the trajectory, after passing through an apogee near the SE-L2 point, intersected the Moon's orbit during its fall from apogee. If the apogee of the trajectory was too high (the flight took too much time), then the trajectory would arrive at the lunar orbit at a shallow angle and the lunar swingby distance would be too small to bend the perigee to a low-enough altitude. If the apogee was too low (flight took too little time), then the angle of arrival at the lunar orbit was too steep and any lunar swingby strong enough to achieve an inner loop of a DLS orbit would cause the spacecraft to impact Earth. The six-month loop shown in Fig. 5 turned out to intersect the lunar orbit at the right angle to achieve a useful DLS orbit. There are three lunar swingbys that take place during 2022, on March 19 at distance from the Moon's center of 23,111 km, on April 13 at distance 23,269 km, and on July 16 at distance 10,241 km. The maneuvers to achieve the trajectory are listed in Table 2. The last lunar swingby (with the help of the apogee maneuvers, especially #5) bent the trajectory out of the lunar orbit plane so that it had the right inclination to the ecliptic for the departure orbit to the asteroid.

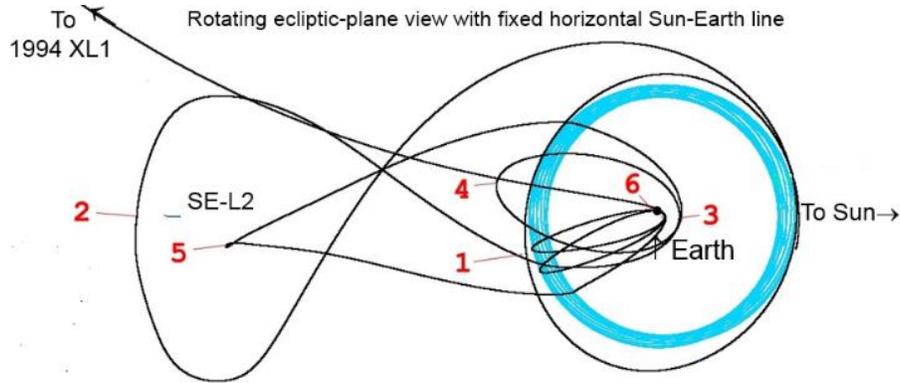


Figure 5 Solar rotating ecliptic-plane view of the trajectory from the EM-L2 halo orbit to 1994 XL1. The red numbers show the locations of ΔV maneuvers listed in Table II. The turquoise circle is the lunar orbit.

Table 2 Maneuvers to, then from an EM-L2 halo orbit to fly by 1994 XL1, and return to the EM-L2 halo orbit..

Event Date, UTC	Event	ΔV , m/s	Notes
2021 June 23, 18:41	Parking Orbit Insertion		KSC launch, Inclination 28.7°
2021 June 23, 18:45	Transfer Traj. Insertion	3130.7	By launch vehicle upper stage
2021 June 28, 17:05	S1 lunar swingby	179.1	Retro maneuver
2021 July 06, 10:29	Halo Orbit Insertion	60.1	Insert into EM-L2 Halo
2021 Sep. 21, 22:49	1 - Depart EM-L2 Halo	0.10	
2022 Jan. 20, 19:51	2 - Apogee near SE-L2	53.1	Targets lunar swingby to start DLS
2022 Mar. 23, 02:20	3 - Perigee	0.16	Adjusts period to minimize ΔV #6
2022 Mar. 31, 22:19	4 - Apogee	9.9	Targets lunar swingby for min. ΔV #5
2022 June 01, 10:14	5 - Apogee	0.9	Targets inclination to asteroid (17.8°)
2022 Aug. 11, 10:56	6 - Perigee, h = 1374 km	180.0	Oberth ΔV to reach 1994 XL1
2022 Dec. 13, 13:26	1994 XL1 flyby	----	No ΔV ; flyby speed 14.74 km/s
2022 Dec. 14, 13:26	ΔV , to target Earth return	9.4	A day after flyby
2023 July 30, 17:02	Perigee ΔV , h = 622 km.	110.3	Oberth ΔV for loose capture
2023 Sep. 19, 20:41	Apogee near SE-L1, ΔV	17.1	To reach EM-L2 halo orbit
2023 Nov. 09, 07:30	Perigee ΔV	25.5	ΔV to reach EM L2 halo orbit
2023 Nov. 29, 21:18	EM-L2 halo insertion ΔV	25.4	Total ΔV 432 m/s

The perigee maneuver #6, the Oberth maneuver to reach 1994 XL1, is a little larger than the value given in Table 1 since the perigee altitude, 1374 km, had to be higher than the 7000 km distance (height 622 km) used in Table I; the previous perigee was lower, distance about 7200 km, and the luni-solar perturbations raised it after that. Fig. 6 shows the trajectory reaching 1994 XL1 and returning to the Earth, while Fig. 7 shows the return trajectory near the Earth, showing how, after the capture maneuver, the trajectory extends out to the SE-L1 point, from which it falls back towards the lunar orbit (with the help of a small maneuver near the apogee in the SE-L1 area), allowing capture back into the EM-L2 halo orbit. Fig. 7 shows the trajectory in the Earth-Moon rotating frame centered on the EM-L2 point (so, due to the eccentricity of the Moon's orbit, the lunar center moves along a short horizontal line), to illustrate the departure from and return back to the halo orbit, as well as the three lunar swingbys of the DLS trajectory that was needed to move the line of apsides to the right direction for the departure to 1994 XL1.

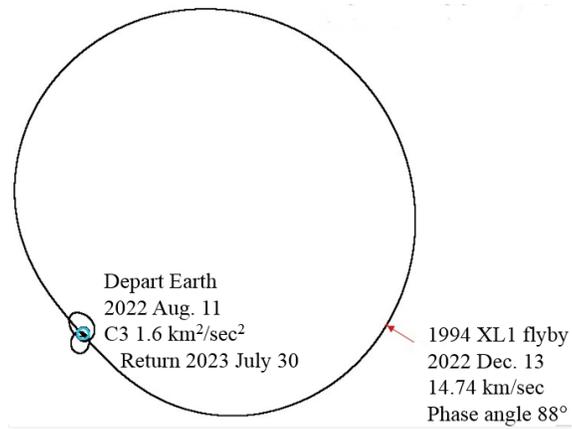


Figure 6: Inertial ecliptic-plane view of the trajectory with flyby of 1994 XL1 and return to Earth. One day after the flyby, a ΔV of 9.4 m/sec returns the ITV to a 622 km perigee height.

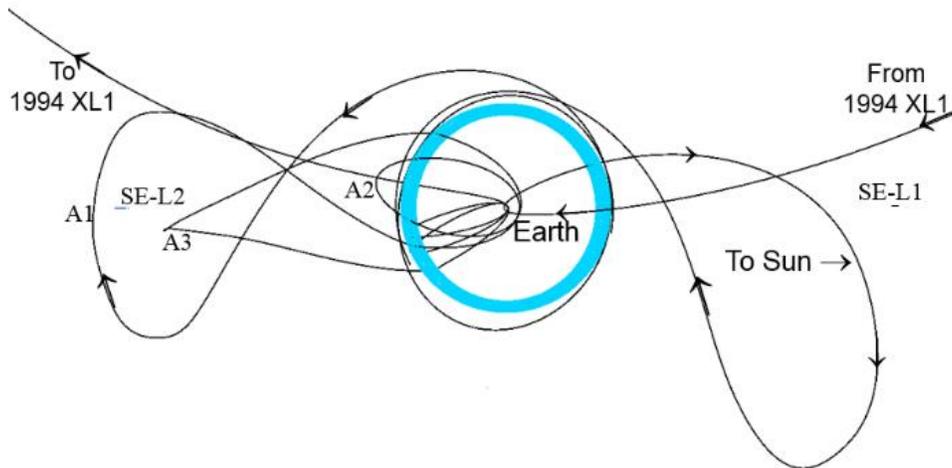


Figure 7. Solar rotating ecliptic-plane view of the trajectory showing both the departure and return parts of the trajectory near the Earth. The turquoise circle is the Moon's orbit.

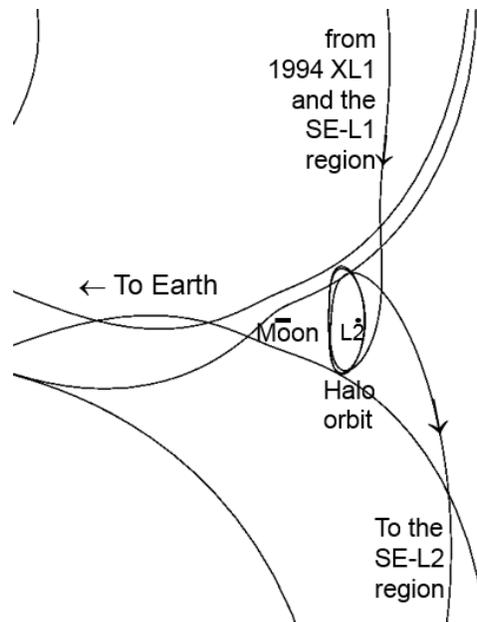


Figure 8 Rotating lunar orbit plane view centered on the EM-L2 libration point and with fixed horizontal Earth-Moon line. The departure from and return to the halo orbit are shown, as well as the three close lunar swingbys described in the text and two more distant passes where the geocentric orbit was changed much less than during the close swingbys..

At the return, the ITV would in practice be targeted to an atmospheric re-entry, with perigee height approximately 100 km. About a day before perigee, the crew would enter the return capsule and detach it from the ITV for their return. The ITV would then robotically perform a ΔV of about 6 m/sec to raise its perigee by about 520 km to a geocentric distance of 7000 km, and then perform the 110 m/sec Oberth maneuver at the perigee for the capture shown in Figure VII. One drawback of this return trajectory is that the return orbit is retrograde, so the surface of the Earth, and its atmosphere, will be moving about 1.4 km/sec faster than if the re-entry were from a direct (prograde) orbit. This will be an important consideration for a higher-speed return such as one from the asteroid rendezvous missions or one from Mars mentioned below, but for a low-energy orbit like the one returning from 1994 XL1, it is actually an advantage, to start gaining experience for the higher-speed returns that will be needed for the higher-energy later interplanetary missions that are our ultimate goal. We note that the robotic Stardust return capsule entered the atmosphere at a significantly higher velocity than is the case with the 1994 XL1 trajectory.

We attempted quicker transfers from the EM-L2 halo orbit to a DLS orbit by decreasing the selenocentric velocity in the halo so that the ITV fell down to a close lunar swingby. But as long as that swingby is unpowered, the perigee of the trajectory remains high, since the orbit's apogee is too low for the solar perturbations to alter it significantly, and a maneuver of over 200 m/sec was needed to lower the perigee enough to start a DLS sequence. Another technique would be to perform a ΔV at the lunar swingby, like the trajectories that return to the Earth shown in References [7] and [8]; although that technique certainly lowers the perigee enough for our purpose, the size of the lunar swingby ΔV (an additional time-critical maneuver) is over 150 m/sec, a cost that we wanted to avoid.

Reference [8] showed that as much as 300 m/sec more ΔV would be needed to leave the halo orbit at non-optimal times (like the abort options shown) so that the lunar swingby would be timed right to achieve the necessary perigee conditions. So although a transfer to a useful DLS might be accomplished in less than two months with this technique (and possibly as little as 10 days), the ΔV costs would be significantly greater than those for the slow route via the SE-L2 region.

For asteroid rendezvous missions, or those to Mars, the perigee Oberth maneuver could be as high as 3 km/sec; that high ΔV would be imparted with a disposable propulsive stage, probably a solid rocket motor, that would be brought to the ITV and attached to it at the same time that the astronauts rendezvous with the ITV in the HEO several days before the departure burn.

3.2.1 Human Near-Earth Asteroid Rendezvous

To understand the performance advantage of basing a reusable ITV at EM-L2 or SE-L2 instead of LEO, it is instructive to compare the ΔV costs of the two staging locations for an example of a rendezvous mission to an NEA. A particularly good opportunity for a piloted rendezvous mission is a 2025 launch to asteroid 1999 AO10, estimated to be between 50 and 100 meters across. The mission profile for this opportunity is illustrated in Fig. 9. An ITV operating from L2 can perform this mission for a ΔV cost of only about 4.9 km/sec which is less than half of the cost for an ITV based in LEO. Of course, this advantage is only for the ITV; the astronauts, and some fuel modules and other consumables, need to be brought out of the Earth's gravity well. The advantage only works for a reusable system for multiple missions, not for a single mission. Still, the ITV is a major component so its reuse from high-energy Earth orbits should result in significant savings. A proper comparison needs to use another metric besides ΔV , such as the total mass lofted to LEO, or the total number of launches with SLS (or some other standard vehicle), or the equivalent cost in some fixed year's dollars. This involves assumptions about the spacecraft hardware that we plan to address later.

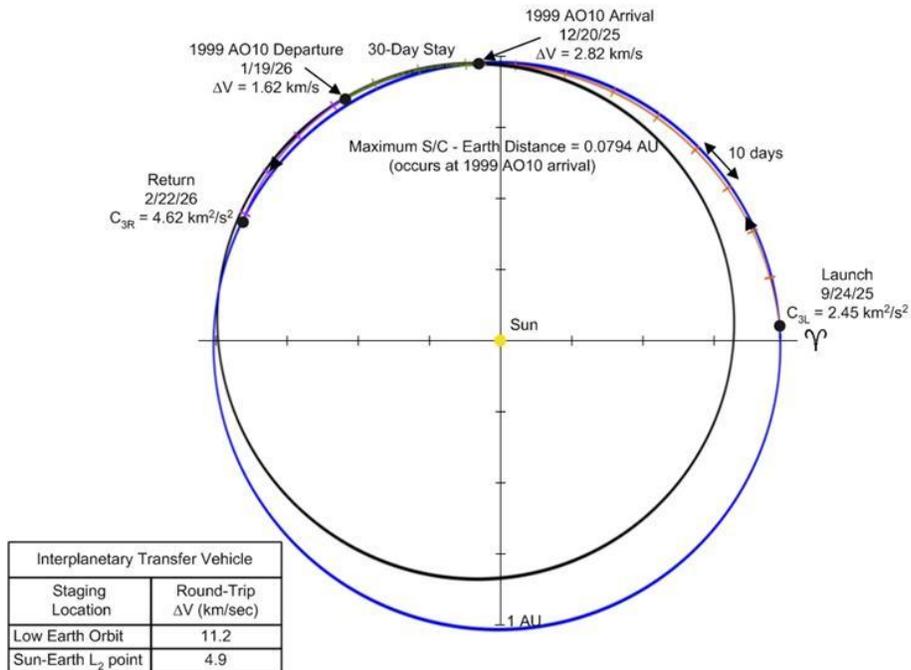


Figure 9. Five-Month Mission to Near-Earth Asteroid 1999 AO10, from Reference [1].

Since our 2008 paper [1], many more accessible (and some potentially hazardous) near-NEAs have been discovered, and good papers about possible human missions have surveyed the possibilities [24,25,29]. Aline Zimmer has used SE-L2 as a staging area for multiple trips to asteroids, finding some interesting low- ΔV solutions by taking advantage of large orbital changes possible near SE-L2 [30,31]. Papers by Joshua Hopkins’ group at Lockheed-Martin show the value for first tests, of missions to the vicinity of the Earth-Moon L_2 libration point, called “Fastnet”. It is part of their “Stepping Stone” approach to extend exploration to NEO’s and the Martian moons that is very similar in concept to that described here [8,18,19].

Additional mission opportunities for near-Earth asteroids are listed in Table 3. For comparison, ΔV requirements are also given for missions to Phobos and Deimos. It is interesting to note that the energy requirements to reach the Martian moons are not very different from what is needed to carry out the asteroid rendezvous missions.

Table 3: Approximate ΔV Costs for ITV Missions to NEAs and Martian Moons using L_2 Staging.

Target	Launch Date	Round-Trip Flight Time	Round-Trip ΔV Cost (km/sec)
1999 AO10	Sept. 24, 2025	5 months	4.9
2000 SG344	April 25, 2028	5 months	4.6
1999 CG9	Aug. 18, 2033	6 months	4.3
2001 FR85	Sept. 24, 2039	6 months	3.0
Phobos	Oct. 20, 2041	31 months	4.9
Deimos	Oct. 20, 2041	31 months	4.3

4. Conclusions

It is time to adopt an architecture for human spaceflight that will generate public enthusiasm by doing things that have never been done before. The pace of such a program must be consistent with budget constraints. Funding for human exploration of space should be based on realistic long-range planning. Plundering budgets allocated to highly successful scientific programs, no matter how expedient, must be avoided.

Attractive features of the IAA plan include the following:

- Creation of a deep space taxi that would pave the way for human missions beyond the Moon's orbit. Early test flights could include circumlunar missions, trips to geosynchronous orbit, and operations from an Earth-Moon L2 halo orbit.
- An emphasis on high-priority science that would be carried out by constructing and maintaining large astronomical observatories near the Sun-Earth L2 libration point.
- Development of a substantial and capable ITV that could be used again and again for human missions to near-Earth asteroids and Mars. The ITV envisioned in the IAA study would possess considerable radiation shielding and living space that would allow astronauts to travel to Mars in relative safety and comfort.
- A realistic possibility that human exploration of Mars will occur before 2050 without adversely affecting scientific programs.

Our project plans to develop the orbital concepts during the next few years to further the IAA study goals, foster collaborations for trajectory calculations and mission design between Russia, the USA, and other space-faring nations, and contribute to ideas for planetary protection.

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