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TRAJECTORY OPTIONS FOR INTERPLANETARY HUMAN EXPLORATION

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We have received a "megagrant" from the Russian Ministry of Education and Science to study orbital options to extend human exploration beyond the Moon's orbit. For a viable program, an international collaboration (as now for the ISS) and reusable spacecraft will be needed. With reusable spacecraft, we will use high-energy Earth orbits that can be drastically modified with lunar swingbys and small propulsive maneuvers in weak stability regions, especially near the collinear Sun-Earth and Earth-Moon libration points. The work will build on ideas developed by the International Academy of Astronautics' exploration study group presented at the 2008 International Astronautical Congress in Glasgow. The first efforts could support backside lunar exploration from an Earth-Moon L2 temporary Lissajous or relatively permanent halo orbit This paper shows that low-post-launch ΔV options exist for missions of about 18d total duration, with slightly higher costs for one-month missions. Next, large space telescopes in Sun-Earth libration-point orbits might be serviced by astronauts. Next, flyby and rendezvous missions to Near-Earth Objects (NEO's) will be designed, with an emphasis on options for defense against potentially hazardous objects. Finally, trajectories to reach Mars, first to Phobos and/or Deimos, will be calculated. The study will use highly-elliptical Earth orbits whose line of apsides can be rotated using lunar swingbys; then a propulsive maneuver, considerably smaller than that needed from a circular low-Earth orbit, can be applied at the right perigee to send the spacecraft on the right departure asymptote to a desired destination. Aerocapture can be used at the return, perhaps helped with a lunar swingby. But the astronauts could separate in an Apollo-style capsule for a direct return. Sun-Earth libration point orbits (most likely L2) and double-lunar swingby orbits, like those flown first by the third International Sun-Earth Explorer, will be used, along with time to change the orbital orientation between missions. There might be waits of several months between missions, when the interplanetary spacecraft could be "parked" in a small-amplitude Lissajous orbit about the Sun-Earth L2 point, similar to that flown by the WMAP mission. During that time, if there wasn't an L2 space telescope needing servicing, the spacecraft could be uncrewed and controlled remotely from the Earth. The first missions might start with an Orion capsule (or something similar) to which modules could be added, including fuel tanks, as needed, for later missions that could include rendezvous with a NEO.

I. INTRODUCTION

Human exploration beyond the Moon may be made possible with staging in high-energy orbits (such as in the Sun-Earth 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 ISEE-3[2], the WIND double lunar swingby trajectory[3], studies for the proposed Relict-2 mission[4], and the STEREO phasing orbits[5]. With the realities of the world today, it is 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. The grant will be used to develop these ideas in much more detail, to prove their feasibility with full force-model simulations. There will be an emphasis on NEO missions, and on formulating optimal strategies for deflecting PHO's.

The Apollo program taught 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 very 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 will not be ignored in these 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 much 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" are described in the next sections.

II. THE EARTH'S SPHERE OF INFLUENCE

II.I Earth Orbit to an Earth-Moon L2 Halo Orbit

No spacecraft have landed on the back side of the Moon, although lunar orbiting survey missions show this area to be of high scientific interest[6]. A useful easily-accessible destination for a "first step" after Apollo would be an Earth-Moon L2 halo orbit. From such an orbit, astronauts and cosmonauts could easily control rovers and sample-return landers on the far side of the Moon.

Lunar L2 halo orbits were studied in detail in the early 1970's as a possible location for communications satellites to support a possible last Apollo mission to the Moon's far side[7-10]. About 1230 m/sec of ΔV is needed to reach the Earth-Moon L2 libration point with a direct transfer, as shown in Figure I.

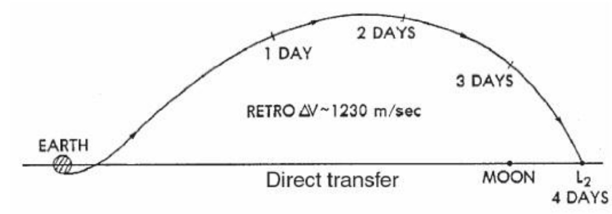

Fig. I: Direct transfer from a low-Earth parking orbit to the Earth-Moon L2 point. This is a rotating lunar orbit plane view with a fixed horizontal Earth-Moon line. From references 8 and 10.

However, it was found that an indirect trajectory with a powered lunar swingby could reach the Earth-Moon L2 point for less than a third of the post-injection ΔV, a significant savings for the spacecraft.

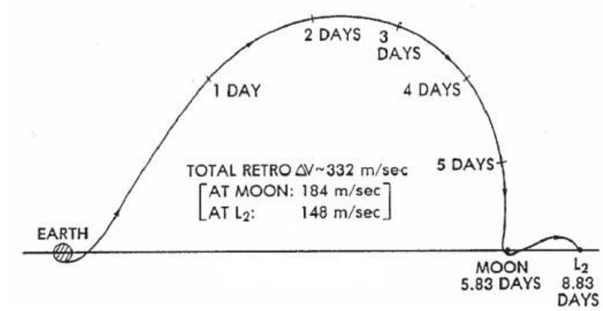

Fig. II: Indirect transfer from a low-Earth parking orbit to the Earth-Moon L2 point. This is a rotating lunar orbit plane view with a fixed horizontal Earth-Moon line. From references 8 and 10.

A first crewed mission might use the trajectory shown in Fig. II, only looping around the L2 point once and then returning quickly to the Earth, on a trajectory that would be a mirror image (about the horizontal "x-axis" line) of Fig. II. Farquhar illustrated such a trajectory as part of his post-Apollo exploration studies[8,10], shown in Fig. III.

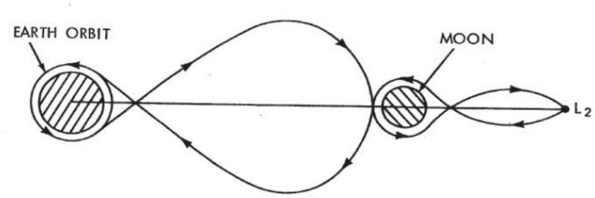

Fig. III: Mission Profile for a Lunar Shuttle System with (Earth-Moon L2) Halo Orbit Staging. From references 8 and 10 (Fig. 3-3).

The trajectories shown in figures I, II, and III were computed with a circular restricted three-body model to the L2 libration point. But the L2 point is hidden behind the Moon as seen from the Earth. The lead author has made a first calculation of an indirect transfer, with an almost due-east launch from Kennedy Space Center (KSC) and powered lunar swingbys, using a realistic force model with solar, terrestrial, and lunar data obtained from the JPL DE405 ephemerides. It is a realization of Fig. III, but with no stopping or ΔV at L2; the result is shown in figure IV.

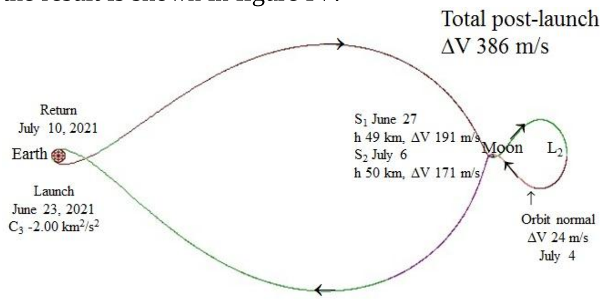

Fig. IV: Indirect transfer from a KSC launch to near the Earth-Moon L2 point and return using a realistic force model. This is a rotating lunar orbit plane view with a fixed horizontal Earth-Moon line.

In figure IV, S1 and S2 refer to the two powered lunar swingbys, with the altitude (h) above the lunar mean surface given. The return to the Earth has a perigee radius of 6478 km (height about 100 km), an approximate value for an atmospheric re-entry. The total flight time is 17 days. Note the symmetry of the outbound (Earth to L2) and inbound (L2 to Earth) trajectories about the horizontal Earth-Moon line (x-axis in the lunar rotating frame; the z-axis is normal to the lunar orbit plane and the in-plane y-axis completes the right-handed system).

The launch date was selected so that the arrival at the Moon would occur before last quarter phase. With this timing, maximum sunlight will be available for backside operations during the first two weeks after arrival near the Earth-Moon L2 point for a longer-stay mission (one to a halo orbit described below that was computed before this “quick” trajectory was calculated using nearly the same Earth-to-Moon transfer), but it is not optimum for this short mission, as can be seen in the solar-rotating view of figure V. It shows that the trajectory passes behind the Earth, where it suffers a 101-minute eclipse starting 86 minutes after the transfer trajectory insertion (TTI) from the parking orbit. Also, the lighting on the lunar far side is poor, with much of the far side in shadow during the trans-lunar phase. If necessary, the eclipse could be shortened by launching at a different time into a more-inclined (to the ecliptic) trajectory, or by launching two or three days later, when the passage through the shadow would be lower and quicker. For the best far-side lighting conditions, a launch 4 or 5 days later would be better, and that would shorten the post-launch eclipse as well.

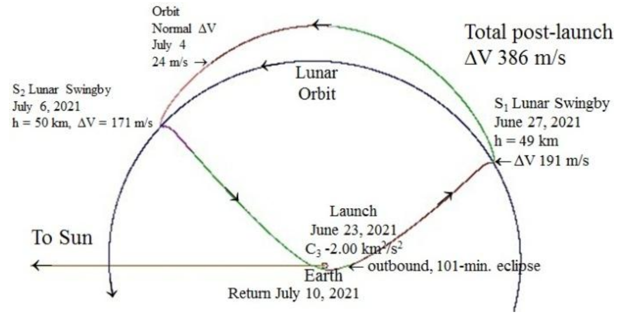

Fig. V: The trajectory shown in Fig. IV in a rotating ecliptic plane view with a fixed horizontal Sun-Earth line.

The timeline of major events for the trajectory are given in Table I below.

2021 Date	UTC	Event
June 23	18:06	Launch
June 23	18:18	POI, h 185 km
June 23	18:19	TTI, C_3 -2.00 km^2/sec^2
June 23	19:46	Start Eclipse
June 23	21:27	End Eclipse
June 27	05:20	Moon, h 49 km, ΔV 191 m/sec
July 4	19:45	Orbit normal ΔV 24 m/sec
July 6	05:46	Moon, h 50 km, ΔV 171 m/sec
July 10	18:34	Earth return, h 100 km

Table I: Major events for the trajectory shown in Fig. IV to Fig. VI. POI is parking orbit insertion and TTI is transfer trajectory insertion.

Figure 6 shows the trajectory near the Moon as seen from the Earth. The trajectory beyond the Moon is actually the start of a Lissajous trajectory, starting at the Moon at S1. But from there, due to the unequal in-plane and out-of-plane frequencies, the pattern starts to expand, making it necessary to add an out-of-plane maneuver, here performed on July 4, to target the trajectory back to the Moon for the S2 swingby.

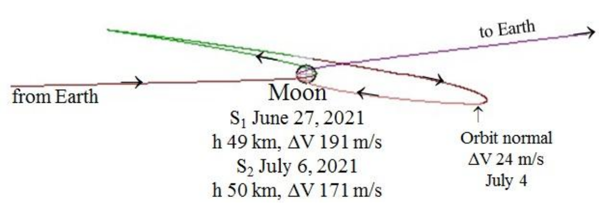

Fig. VI: View from the Earth of the trajectory shown in Fig. IV and V. The horizontal direction passing through the Moon is the lunar orbit plane, while the vertical direction is perpendicular to the lunar orbit plane.

The trajectory shown above was targeted from S1 to S2, with an S2 altitude of 50 km, without any in-plane ΔV between them; there is no other similarly "free" (or low-cost) trajectory that accomplishes this with a flight time different from 17 days. We attempted to compute trajectories that would spend more time behind the Moon by decreasing the S1 breaking burn slightly. But then the in-plane trajectory passed far from the Moon when it fell back, and an in-plane maneuver had to be added to counter-act that, to target a low lunar swingby where a ΔV could be added to send the spacecraft back to the Earth. A view of such a trajectory is shown in figure VII below.

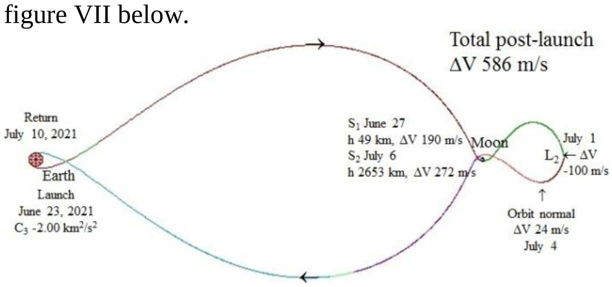

Fig. VII: An attempt to increase the time behind the Moon by decreasing the S1 ΔV by a m/sec failed because a large maneuver was then needed to target the S2 swingby, and that caused a corresponding decrease in the flight time. Rotating Earth-Moon view, in the lunar orbit plane.

Since the added ΔV near aposelene (near L2) sped up the return to S2, the total flight time is actually almost exactly the same as for the trajectory shown in Fig. IV, but the total post-launch ΔV is 200 m/s sec greater. So this approach is not useful for increasing the trans-lunar stay time.

The next approach, shown in figure VIII below, attempted to return to the Moon after completing about half a revolution around L2.

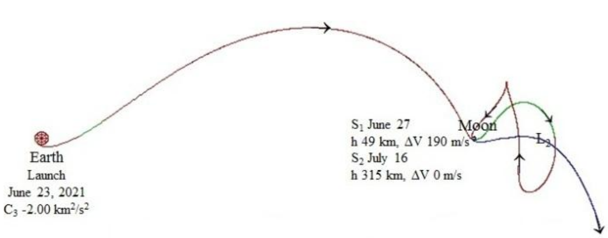

Fig. VIII: An attempt to make a partial orbit about L2 to increase the trans-lunar flight time by several days. Rotating Earth-Moon view, in the lunar orbit plane.

Unfortunately, the return to the Moon in this case is in a very unfavorable direction, sending the spacecraft away from the Earth rather than towards it. Adding a 200 m/sec ΔV at S2 changed the geometry only slightly, just speeding up the escape. Applying similarly-sized ΔV's before S2 also did not help.

In order to return the Moon with approximately the right geometry, and have a longer flight time than the figure IV trajectory, it is necessary to complete nearly a full revolution around L2. An example is shown in figure IX below.

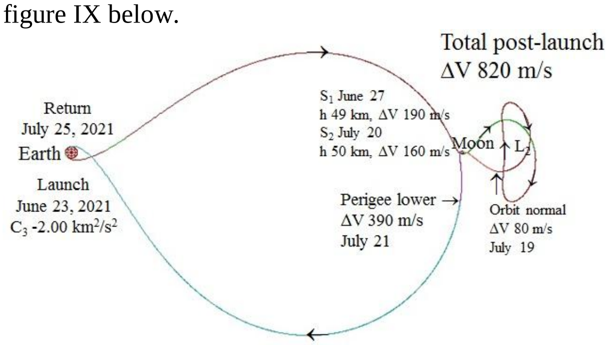

Fig. IX: A trans-lunar trajectory with a total flight time of 32 days, making a complete revolution around L2, in addition to the outbound (Moon to L2) and inbound legs. Rotating Earth-Moon view, in the lunar orbit plane.

The trajectory is asymmetric about the horizontal Earth-Moon (x) axis. As a result, a free return has a very high perigee that can be counteracted only with a large ΔV near apogee, about a day after the S2 lunar swingby, to lower perigee enough. Due to the longer time from S_1 to S_2, a larger orbit normal ΔV, 80 m/s, is needed.

In order to try to make the return more symmetric to eliminate the perigee lower ΔV, in-plane maneuvers were added before S2 as shown by the trajectory depicted in figure X below.

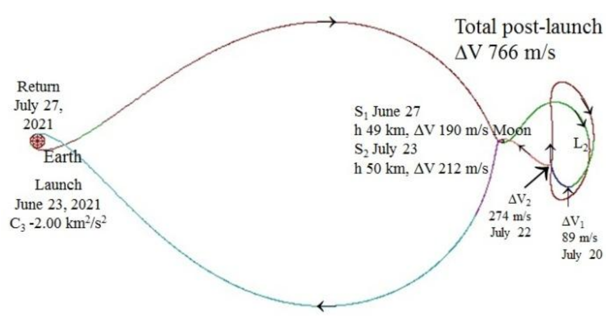

Fig. X: A trans-lunar trajectory with a total flight time of 34 days, making a complete revolution around L2, in addition to the outbound (Moon to L2) and inbound legs. In-plane ΔV's have been added before S2 to eliminate the large ΔV after S2 near apogee of the trajectory shown in Fig. IX. Rotating Earth-Moon view, in the lunar orbit plane.

Like the figure IX trajectory, the orbit normal ΔV, performed on July 20th, needed to be increased to 80 m/sec. This was necessary because the Lissajous pattern had more time (from S1 to S2) to expand away from the

Moon than for the short figure IV trajectory. In addition, an in-plane ΔV of 40 m/sec (for a total 89 m/sec, added vectorially with the out-of-plane component) was added on July 20, but even another large maneuver had to be added two days later (just increasing the July 20th in-plane component didn't work) to send the spacecraft to the S2 lunar swingby with a geometry favorable enough to eliminate the post-S2 near-apogee ΔV. The parts of the trajectory between the Earth and the Moon in figure X are much more symmetric than those shown in figure IX, which is why the apogee ΔV could be eliminated. There is a net savings with the new trajectory, which we will adopt for now as a month-long return trajectory with more information about it given below, but the savings is only 54 m/sec over the figure IX trajectory. Further effort could undoubtedly result in a better optimization of this class of trajectories. A view of the trajectory looking from the Earth towards the Moon is shown below, showing how the Lissajous trajectory expands away from the Moon, but then targeted back to the Moon with two maneuvers, on July 20 and 22.

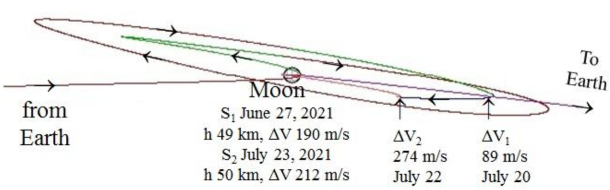

Fig. XI: View from the Earth of the trajectory shown in Fig. X. The horizontal direction passing through the Moon is the lunar orbit plane

A view of the figure X trajectory in the solar rotating frame is shown in figure XII below.

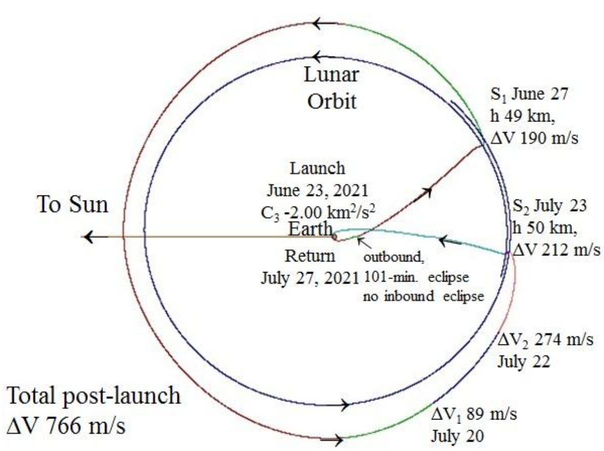

Fig. XII: The trajectory shown in Fig. X in a rotating ecliptic plane view with a fixed horizontal Sun-Earth line.

The trajectory has a full lunar day view of the far side of the Moon while it is near L2. The inbound leg, from S2 to the Earth, has no eclipse because it has a higher inclination to the ecliptic plane than the outbound leg. This trajectory has a further advantage over the shorter figure IV trajectory in that the Earth return is in daylight. Although more optimization is possible, the total post-launch ΔV cost is about twice that of the quick figure IV trajectory, but still well under a km/sec.

A trajectory like that of figure IV was computed with a launch one day later, shown in figure XIII below.

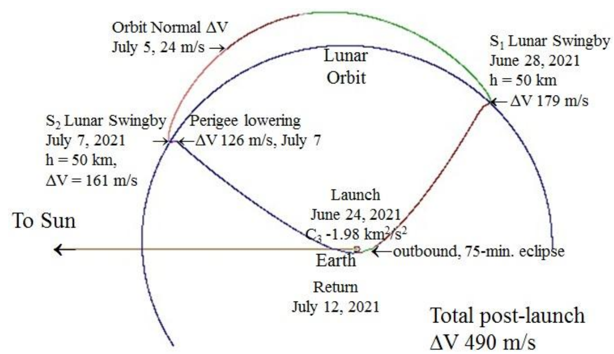

Fig. XIII: A 17-day trajectory like that shown in Fig. V but with launch a day later. Rotating ecliptic plane view with fixed horizontal Sun-Earth line.

With the later launch, the outbound eclipse was decreased to 75 minutes. However, a launch about six days later would provide a better view of the lunar far side, and should also have a significantly shorter outbound eclipse. A view of the trajectory in the Earth-Moon rotating frame is shown below.

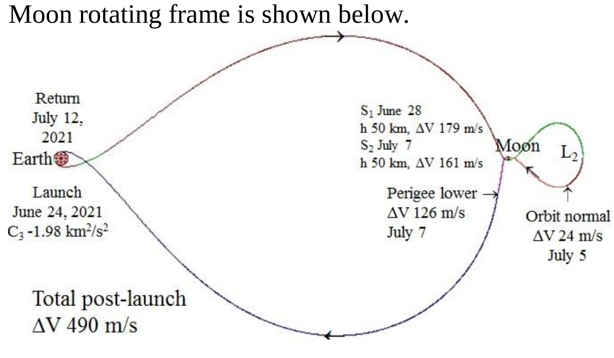

Fig. XIV: The Fig. XII trajectory (launch on June 24) with a rotating Earth-Moon view, in the lunar orbit plane.

Since the flight time to the Moon is 6 hours longer than for the figure IV-V trajectory, the trajectory is asymmetric about the horizontal axis, necessitating a perigee lower ΔV after S2. Further optimization should result in a shorter flight time to the Moon and elimination of the perigee-lowering ΔV. But rather than work on this trajectory more, those with launch about a week later will be computed in the future, for the better lunar far-side lighting geometry that they provide.

Rather than quickly return to the Earth, a future mission might instead rendezvous with a module with additional resources, perhaps a small space station, which is already in a halo orbit about the Earth-Moon L2 libration point. The total post-injection ΔV is 336 m/s, only 4 m/s more than for the figure II trajectory, remarkable agreement for a calculation that is not fully optimized; see figure XV below.

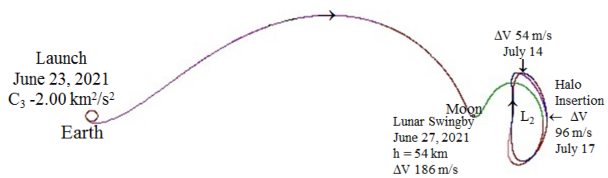

Fig. XV: Indirect transfer from a KSC launch to an Earth-Moon L2 halo orbit using a realistic force model. This is a rotating lunar orbit plane view with a fixed horizontal Earth-Moon line.

As noted before, the date was selected so that the arrival at the Moon would occur before last quarter phase. With this timing, maximum sunlight will be available for backside operations during the first two weeks after arrival near the Earth-Moon L2 point. As noted for the similar outbound legs of trajectories with launch on June 23 shown previously, the solar-rotating view shown in figure XVI shows that the trajectory passes behind the Earth, where it suffers a 101-minute eclipse starting 86 minutes after the transfer trajectory insertion (TTI) from the parking orbit. If necessary, the eclipse could be shortened by launching at a different time into a more-inclined (to the ecliptic) trajectory, or by launching two or three days later, when the passage through the shadow would be lower and quicker.

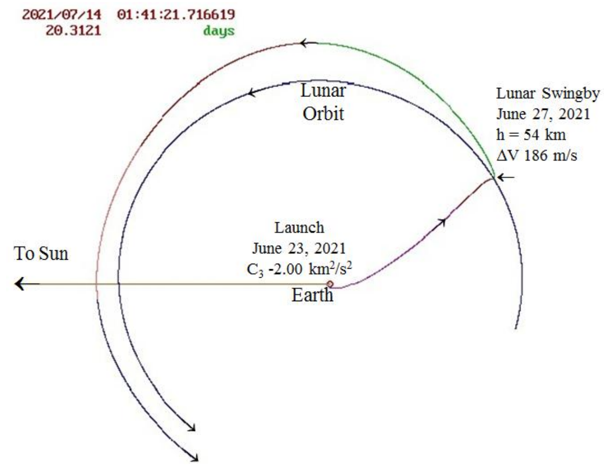

Fig. XVI: The trajectory shown in Fig. XV in a rotating ecliptic plane view with a fixed horizontal Sun-Earth line.

The timeline of major events for the trajectory are given in Table II below.

2021 Date	UTC	Event
June 23	18:12	Launch
June 23	18:23	POI, h 185 km
June 23	18:24	TTI, C_3 -2.00 km^2/sec^2
June 23	19:51	Start Eclipse
June 23	21:32	End Eclipse
June 27	06:44	Moon, h 54 km, ΔV 186 m/sec
July 14	04:19	ΔV 54 m/sec
July 17	18:44	HOI, ΔV 96 m/sec

Table II: Major events for the orbit shown in Fig. XV and Fig. XVI. POI = parking orbit insertion; TTI = transfer trajectory insertion; and HOI = halo orbit insertion.

The trajectory near the Moon is shown in more detail in figures XVII and XVIII on the next page. After the lunar swingby, the trajectory completes more than a full revolution around the L2 Earth-Moon libration point to more closely align with the desired halo orbit in order to decrease the two ΔV's needed to attain the halo orbit. Although the alignment looks good in the XY-plane view of figure XVII, figure XVIII, the view from the Earth (YZ-plane view) shows that the trajectories are not very close. This is because the lunar swingby forces the trajectory to start at the center of the YZ view, at the Moon, so the trajectory after the swingby is a narrow Lissajous path rather different from the halo orbit. Normally, the ΔV cost to insert into a halo orbit is considerably less than exactly into the L2 point, but the change from the narrow Lissajous to the halo increases the cost, approximately compensating each other. But some further decrease in the total post-injection ΔV is possible by varying the inclination of the lunar swingby trajectory, and by varying the times of the two maneuvers after the lunar swingby.

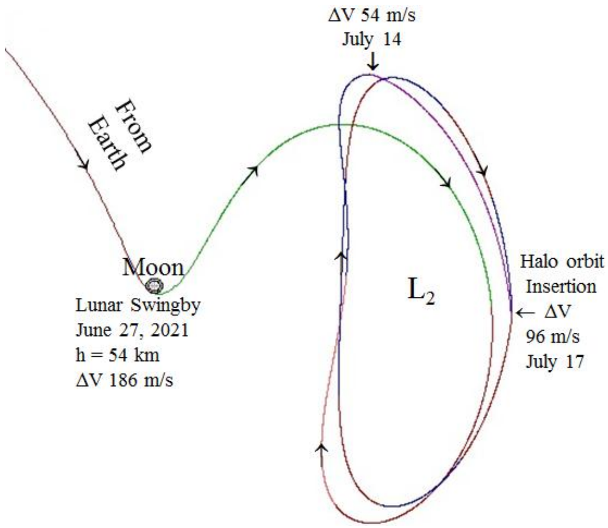

Fig. XVII: An expanded view near the Moon of the trajectory shown in Fig. XV, also in a rotating lunar orbit plane view with a fixed horizontal Earth-Moon line.

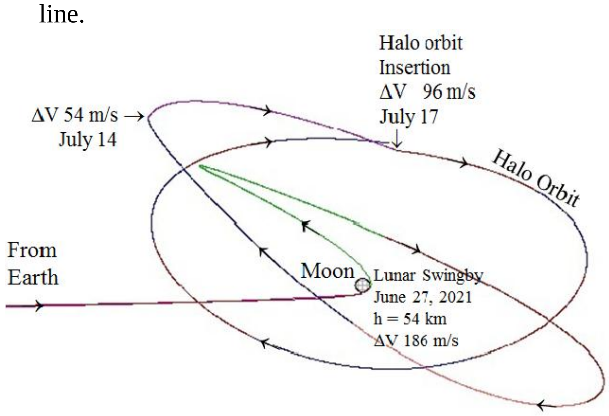

Fig. XVIII: View from the Earth of the trajectory shown in Fig. XV. The horizontal direction passing through the Moon is the lunar orbit plane, while the vertical direction is perpendicular to the lunar orbit plane.

Lunar halo orbits are not easily determined with realistic models due primarily to strong solar perturbations. Due to an error, the "halo" for the trajectory shown in the figures above was computed using the orbital energy balancing technique[11], but for a point that was not very close to a theoretical halo orbit. The trajectory, although close to a periodic halo orbit, is actually a Lissajous trajectory that nevertheless should not pass behind the Moon for a few months, longer than a crewed lunar mission is likely to last. The "halo" orbit used is certainly close enough to a "real" halo orbit to establish the size of ΔV maneuvers reasonably well.

II.II Earth Orbit to the Sun-Earth L2 Point

Transfers between a highly elliptical Earth orbit and the vicinity of the Sun-Earth L2 libration point can be accomplished easily via multiple lunar swingbys, a technique utilized by ISEE-3 and other missions. The Sun-Earth L2 libration point region is important, already the destination for several current and planned missions[4,12]. 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 Sun-Earth L2 point.

II.III Sun-Earth L2 to Other Destinations

Although nominally based in a Sun-Earth L2 halo orbit, where the ITV can be unmanned 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 for 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) 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, which will be investigated in this study, include:

- Orbits around the Sun-Earth L1 point and the Earth-Moon L1 and L2 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 x 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.

Candidate detailed trajectories, to prove feasibility and assess realistic ΔV requirements, will be developed for this approach to extend human exploration beyond low-Earth orbit (LEO) using the "stepping stones" described above, and amplified below.

Transfers between a highly elliptical Earth orbit and the vicinity of the Sun-Earth 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 Figure XIX. 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.

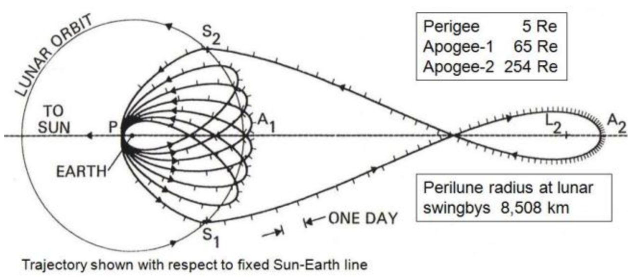

Fig. XIX: Simplified Version of a Double Lunar-Swingby Trajectory (patched conic calculation)

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. VII 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 Sun-Earth L2 point for possible future use to more distant destinations described in the next sections. Astronauts could reach the vicinity of L2 relatively quickly, in about 2 weeks, using small vehicles with reasonable ΔV costs[1].

III. BEYOND THE EARTH'S SPHERE OF INFLUENCE

III.I Interplanetary Transfer Vehicle (ITV)

A general mission scenario using L2 staging is outlined below and described in more detail in Ref. 1. The mission sequence begins with the ITV (sans crew) departure from the L2 orbit. Small propulsive maneuvers (total ΔV less than 50 m/sec) 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" (perhaps a variant of the planned Orion capsule) is used to transfer the crew from LEO to the ITV in its elliptical Earth orbit. When the crew transfer has been accomplished, the "taxi" 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 Earth orbit capture followed by lunar gravity assists and small propulsive maneuvers to return to its L2 base.

A variant of the mission scenario described above would transfer the ITV to one of the Earth-Moon collinear libration points instead of an elliptical Earth orbit for the rendezvous and crew transfer with the DSS. This strategy might simplify some of the LEO departure and rendezvous constraints for the DSS, but would require a substantial increase in the DSS round-trip ΔV cost. The departure ΔV for the ITV would also be increased---an additional 350 m/sec for Earth-Moon L2 and 700 m/sec for Earth-Moon L1. Given time (a few months, all right for unscrewed spacecraft), transfers between the Earth-Moon L2 and Sun-Earth L2 can be accomplished with very little ΔV.

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[13]. 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.

III.II Missions to Near-Earth Asteroids

To understand the performance advantage of basing a reusable ITV at Sun-Earth L2 instead of LEO, it is instructive to compare the ΔV costs of the two staging locations for an example of a mission to a near-Earth asteroid. A particularly good opportunity for an early piloted mission is a 2025 launch to near-Earth asteroid 1999 AO10. The mission profile for this opportunity is illustrated in Figure XX. Notice that an ITV operating from L2 can perform this mission for a ΔV cost of only 4.9 km/sec which is less than half of the cost for an ITV based in LEO.

Since our 2008 paper[1], many more accessible (and some potentially hazardous) near-Earth asteroids (NEAs) have been discovered, and good papers about possible human missions have surveyed the possibilities[14,15]. Aline Zimmer has accomplished further impressive work, also using the Sun-Earth L2 as a staging area for multiple trips to asteroids, finding some interesting low-ΔV solutions by taking advantage of large orbital changes possible in weak stability areas near Sun-Earth L2[16]. The recent NHAT Web tool, connected to the latest updated asteroidal databases, will further enhance our ability to find affordable human missions to asteroids. Papers by Joshua Hopkins' group at Lockheed-Martin show the value for first tests, of missions to the vicinity of the Earth-Moon L2 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[17,18].

IV. 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 megagrant project plans to develop the orbital concepts during the next two 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.

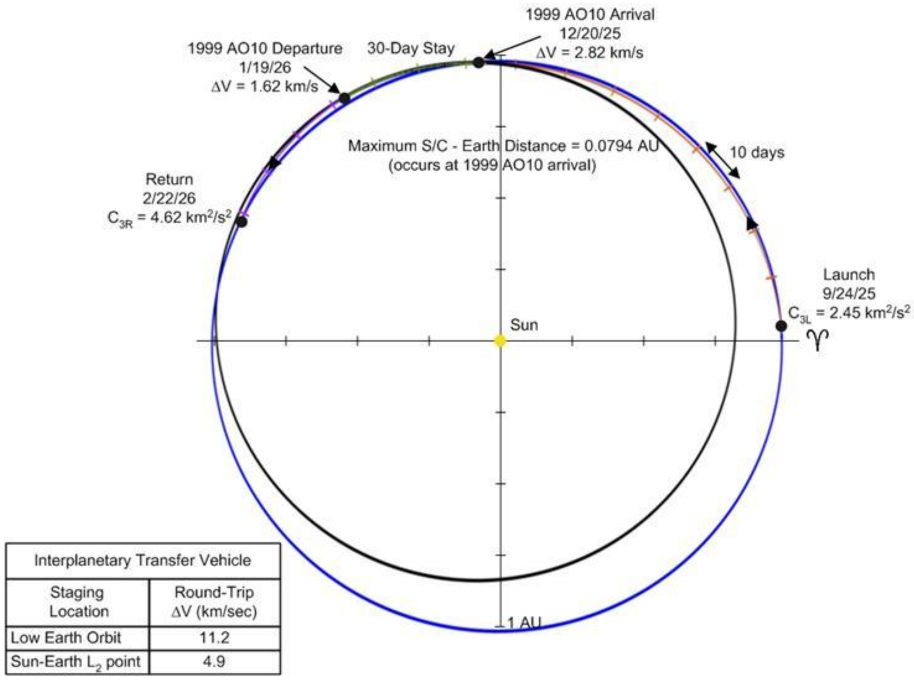

Fig. XX: Five-Month Mission to Near-Earth Asteroid 1999 AO10, from Reference 1.

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