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PRACTICAL METHODOLOGIES FOR LOW DELTA-V PENALTY, ON-TIME DEPARTURES TO ARBITRARY INTERPLANETARY DESTINATIONS FROM A MEDIUM-INCLINATION LOW-EARTH ORBIT DEPOT

Michel Loucks,[*] Jonathan Goff[†] and John Carrico[‡]

The authors present a 3-burn injection method that enables manned and robotic spacecraft to depart for interplanetary destinations from a Low-Earth Orbit propellant depot with only minor ΔV penalties. In this paper, the authors discuss related injection methodologies; illustrate the underlying concept behind this three-burn injection method; discuss implications of using this method, including potential mission safety benefits; and present some details on estimates of the worst-case ΔV penalty for performing this sort of departure maneuver, compared with a traditional one-burn departure from a LEO parking orbit.

INTRODUCTION: PROPELLANT DEPOTS

Propellant depots have long been identified as an enabling in-space technology. It has been suggested that the emergence of depots, and other infrastructure-building technologies, will be a key sign that space utilization for "human purposes" has become routine.

> "If we are serious about this, then our objective must be more than a disconnected series of missions, each conducted at huge expense and risk, and none building a lasting infrastructure to reduce the expense and risk of future operations. If we are serious, we will build capability, not just on the ground but in space. And our objective must be to make the use of space for human purposes a routine function."[1]

In addition to the logistical advantages, there are potentially economic benefits of orbital propellant depots. These benefits include providing demand for launch services that can help drive down launch costs by creating a robust and competitive marketplace, providing launch demand sufficient to encourage the development of commercial Reusable Launch Vehicles, enabling the reuse of in-space assets, and greatly increasing the utility of future ISRU efforts[2]. In addition, propellant depots can create economic demand in space for materials that could be mined from the Moon and asteroids. Depots have the potential to not

[*] CEO and Astrogator, Space Exploration Engineering Corp., Friday Harbor, Washington 98250, USA. E-mail: loucks@see.com.
[†] President and CEO, Altius Space Machines, Broomfield, Colorado 80020, USA. E-mail: jongoff@altius-space.com.
[‡] CTO and Astrogator, Space Exploration Engineering Corp., Ellicott City, Maryland, USA. E-mail: astrogatorjohn@see.com

only greatly reduce the difficulty of travel beyond LEO, but also to help promote the development of much more affordable and reliable means for accessing LEO in the first place. Indeed, NASA has recently started the Restore-L technology demonstration mission which is an important step towards making refueling of spacecraft routine[3].

Conceptually, there are two general places where propellant depots could be located. The first we call "Low-Orbit Depots" which are located at the first practical stopping point on the way out of a gravity well, particularly for deep gravity wells such as near Earth, Mars, Venus, Jupiter, Saturn, etc. The second set of locations are "High-orbit Depots," which are at or near the last stopping point on the way out of the gravity well. The can be temporary for highly elliptical departure orbits or permanent depots, such as at Lagrange point locations. Also included in this are locations near shallow gravity wells, such as near asteroids.

Table 1 provides a proposed taxonomy for a range of in-space propellant transfer facilities. On one extreme end of the spectrum is Distributed Launch, where two or more upper stages rendezvous and dock together, enabling propellant to be transferred from one or more single-use "tanker" stages to a stage that will perform the departure burn for its attached payload[4]. On the other extreme of the spectrum are traditional Stationary Depots, and more sophisticated "Way Stations" that combine propellant-depot functionality with other space station functionality such as crew quarters, cargo transfer facilities, maintenance and repair bays, and even vehicle assembly facilities. In the middle of the spectrum is a novel class of depot we term "roving depots", which are depots that are capable of large ΔV maneuvers, aerocapture, and long-term storage of propellant. This allows them to be temporarily placed at a desired orbit in support of a mission or campaign of missions, fueled by one or more tankers over a period, and then after providing propellant to the desired missions, returned to a more permanent depot location for restocking and reuse. One example of such roving depot operations would be placing a roving depot in a highly-elliptical Earth orbit that is optimally aligned with a future departure asymptote for a large manned space mission. The roving depot could then be topped off over the course of months, and after fueling the mission, could brake back into LEO to be reused.

Table 1: A Proposed Taxonomy for Propellant Depots

	Distributed Launch	In-space Reuse Tankers	Roving Depots	Stationary Depots	Way Stations
In-Space Refillable	No	Yes	Yes	Yes	Yes
Propellant Capacity	Light (35-70mT)	Light (35-70mT)	Medium (70-120mT)	Medium/Heavy (120-200+mT)	Heavy (120+ mT)
Storage Duration	Short (Days/Weeks)	Short (Days/Weeks)	Medium (Months)	Medium/Long (Months/Years)	Medium/Long (Months/Years)
Large ΔV Propulsion	Limited (Single Use)	Yes (Multi-Mission)	Yes (Multi-Mission)	No	No
Aerocapture	No	Yes	Yes	No	No
Additional Services	No	No	No or Limited	No or Limited	Hab, Maint, Cargo Xfer, Assy

For the foreseeable future, Earth will be the focus of humanity which means that Low Earth Orbit (LEO) will most likely remain one of the key transportation nodes in the Solar System. Some examples of potential future markets for LEO depots include:

- Persistent Geosynchronous (GEO) and Medium Earth Orbit (MEO) platform assembly and delivery of upgrades and maintenance/repair hardware and materials
- Distributed launch for lower-cost MEO and GEO deliveries
- Lunar and cislunar orbital space settlements
- Lunar In-Situ Resource Utilization (ISRU).
- Space solar power satellites
- Cislunar tourism
- Deep-space departure for science, planetary defense, and commercial missions

In addition to Earth being the likely center of gravity for humanity for the foreseeable future, there are several other reason why LEO depots are likely important. For instance, for most of these potential markets, in-space reusable vehicles will need to be refueled, and the Earth is the most likely near-term location for such activity. For missions involving people, many will involve people leaving from or returning to Earth, both of which also would suggest LEO as a logical transportation node location for such missions. Also, fully-reusable launch vehicles are important for transportation affordability, but lose payload capacity rapidly with altitude. Additionally, the ability to use Earth's atmosphere to enable aerocapture/aerobraking can enable much cheaper return of ISRU propellants from the Moon (and NEOs) to LEO. Finally, LEO depots can work well for lunar development, since the opposing motion of LEO orbit nodal regression and lunar orbital motion provides frequent departure opportunities exist every ~7-9 days.[5]

PROPELLANT DEPOT ARCHITECTURE ASSUMPTIONS

For our discussions here, we will assume that one of the following hardware architectures is in place

1. A propellant depot exists in LEO, and our satellite is launch empty to that depot. After refueling, the satellite performs all of the subsequent burns

2. A propellant depot exists in LEO, and satellites are services from this depot by a "Tug" spacecraft. The "Tug" takes our spacecraft to the large elliptical phasing orbit, releases the spacecraft, and comes back to the depot (perhaps via aerobraking). The spacecraft then performs any plane change and escape maneuvers necessary to achieve the outgoing asymptote conditions.

PROPELLANT DEPOT CHALLENGES FOR DEEP-SPACE MISSIONS

While LEO depots can work well for lunar missions, deep-space missions require hitting a desired departure asymptote during a tight launch window, usually only days to weeks[6].

Coplanar injection opportunities from typical LEO depot orbits only occur twice per nodal regression period, which is usually 50 to 70. Thus, the odds of hitting an arbitrary unmanaged departure asymptote/window with a coplanar departure from a LEO depot are very low (20-50%).[7]

Previously suggested solutions to this challenge include:

1. Distributed Launch: A single-use "depot" can be launched into an orbital plane to support a single mission. This can work, but misses out on many of the benefits of using a reusable LEO depot, including long-duration, low-boiloff propellant storage
2. Multiple LEO depots: This plan would have several (2-5) depots in evenly spaced orbit planes that can hit all destinations with coplanar departures. While this would work well, it requires a high departure tempo to justify the costs of launching and maintain multiple depots. We hope mankind will need this someday, but that is a long-term proposition.
3. Depot Orbit Phasing: This technique would use maneuvers to increase or decrease the semimajor axis to change the nodal precession rate.
    a. The precession rate is targeted so that the depot orbital plane is co-planer with the outgoing asymptote on the desired departure date and time.
    b. To be perfectly co-planer at departure, the launch and rendezvous windows are very tight and near instantaneous.
    c. Phasing for a new mission can take a long time, and must be considered for delivery logistics.
    d. Radiation exposure and atmospheric drag limit how much the orbit can be raised or lowered.
    e. This works best if depots are used infrequently (1, 2-year departure cadence).

The optimal inclinations for LEO propellant depots from a launch and supply logistics standpoint are probably between approximately 28.5 degrees and 51.6 degrees, which means that in many cases the departure declination for a NEO departure will not be available given the LEO depot's orbital inclination.

We describe a low ΔV penalty three-burn departure methodology that holds the potential to solve these problems, enabling a LEO depot located in a medium-inclination orbit to place an interplanetary departure stack into the right interplanetary trajectory, at the right time, that is relatively insensitive to the orientation of the depot plane and the departure trajectory at the time of departure.

GEOMETRY OF INTERPLANETARY DEPARTURES

A simple, direct deep-space transfer trajectory from Earth to Mars is shown in **Figure 1**. The Sun is in the center, and the yellow transfer trajectory is essentially a Holman transfer—half of an elliptical orbit—from Earth's orbit (drawn in cyan) to Mars' orbit (drawn in red).

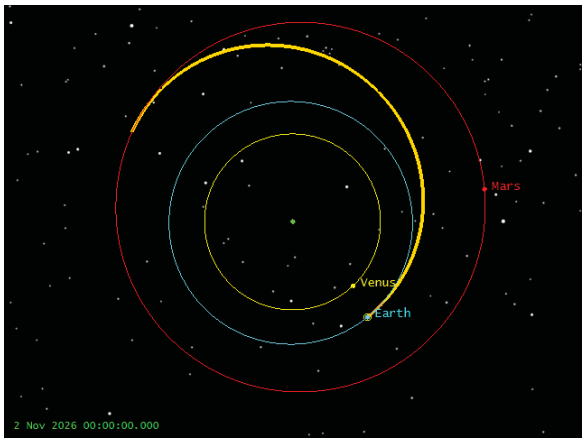

Figure 1: Representative Mars transfer trajectory in heliocentric coordinates

The same trajectory, of course, is a hyperbola when observed in a geocentric inertial coordinate system. The inclination of the hyperbola at the Earth is irrelevant because Mars is so far away, and no matter what the original Earth inclination is, the trajectories will approach the outgoing hyperbolic asymptote.

The family of possible hyperbolic departures from Earth is shown in **Figure 2,** which is the classic figure from Bate, Mueller, and White (BMW)[8]. (This figure inspired the visualizations in this paper.) Several hyperbolic trajectories are shown, all with the same outgoing hyperbolic asymptote. The perigee points of each hyperbola form a circle, which is the locus of all the possible perigees. Because this is the point where the maneuver from LEO to the interplanetary transfer happens, it is called the "locus of injection points".

The locus of injection points can be thought of as lying on a cone, with the vertex centered at the center of the earth on the asymptote. The half-angle of the cone is then the same as the value of the angle between the eccentricity vector and the outgoing asymptote vector. This angle is the supplement of the angle η as described in BMW[9], which is calculated as $\eta = \arccos(-1/e)$.

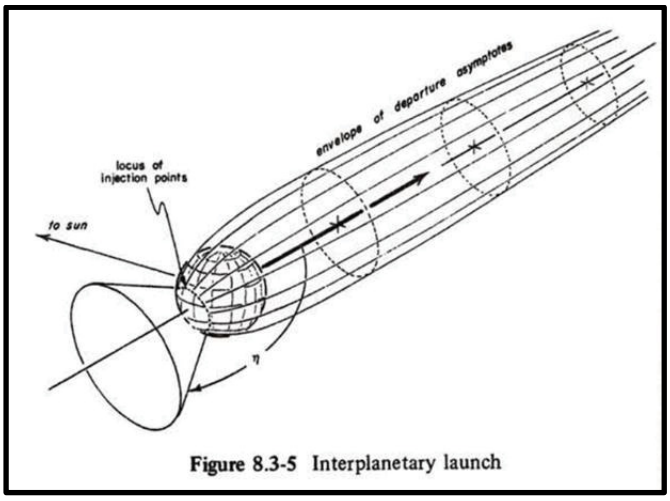

Figure 2: Departure Asymptote Geometry[8]

3-BURN DEPARTURE METHODOLOGY

The 3-burn departure idea was derived from a concept for enabling deep-space departures from an equatorial MXER tether[10]. This concept is also like the trans-Earth Injection concepts analyzed for the NASA's Exploration Systems Architecture to enable an "anytime Earth return"[11] from the Moon. In this concept, the spacecraft first enters a highly elliptical phasing orbit with a periapsis on the locus of injection points timed to return to periapsis at the desired departure date. Prior to this periapsis, at apoapsis, the spacecraft performs a plane-change maneuver to rotate the plane of the orbit to contain the outgoing asymptote.

Figure 3 through **Figure 6** through show a representative 3-burn departure from Earth to NEO 2007 XB23 from an ISS-type orbit. The annotations show the sequence of events.

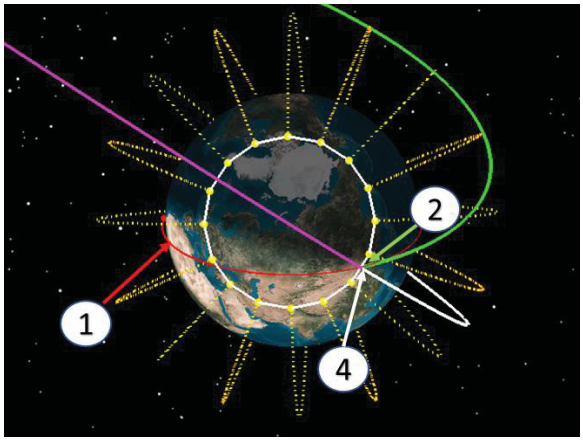

Figure 3: Initial depot orbit and the locus of injection points

Numeral (1) points to the initial depot orbit, in red; with an ISS-like inclination of 51 degrees. Numeral (2) points to the transfer phasing orbit, in green. The apogee raise maneuver occurs between the white and the green orbits as the spacecraft crosses the locus of injection points for the outgoing asymptote. Numeral (3) is near the apogee, where the plane-change maneuver occurs.

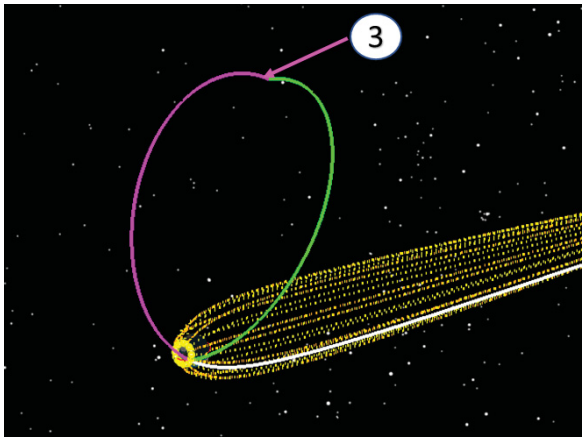

Figure 4: Phasing Orbit Plane Change and Insertion

The plane change is targeted to align the departure plane to contain the departure asymptote. (Note that this maneuver is not only a plane change, but also targets the radius of perigee to be maintained at a desired altitude, 300 km for this example.) After the plane-change maneuver, the trajectory is drawn in magenta. Numeral (4) shows the return perigee where the hyperbolic injection maneuver occurs, putting the spacecraft on the outbound deep-space transfer trajectory, drawn in cyan. This final maneuver is targeted to match the outgoing C3 of the transfer trajectory and to place the spacecraft fully on the targeted outgoing asymptote.

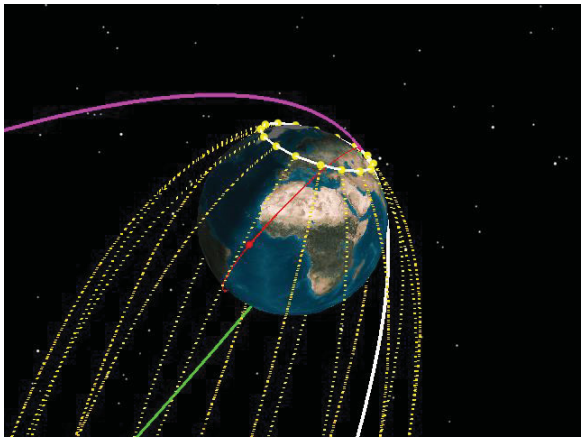

Figure 5: Phasing orbit return (magenta) and outgoing departure (white)

The following steps are the generic method followed to calculate the 3-burn trajectories in this analysis.

1. Calculate a direct transfer trajectory from Earth to the other body, similar to solving Lambert's problem[12]. The initial state uses a LEO orbit with an inclination equal to the outgoing declination, which is used as a baseline to compare subsequent ΔV values.

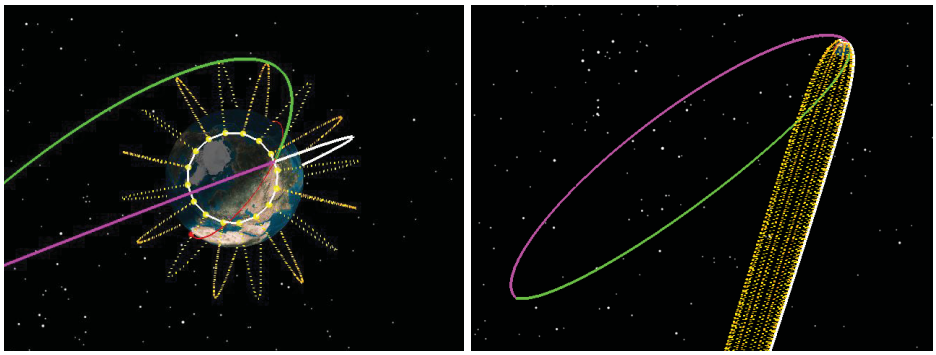

Figure 6: Additional views of a high declination 3-burn departure to Asteroid 2007 XB23

2. Calculate the angle between the eccentricity vector and the hyperbolic asymptote. This is the supplement of hyperbolic angle, η, described above.
3. Calculate the outgoing asymptote vector from the direct transfer trajectory.
4. Select a depot RAAN that intersects with the locus of injection points (given the assumption of a 400,000-km apogee used in this case). Adjust the depot RAAN to minimize the angle between the orbital plane and the anti-asymptote vector (this yields the minimum plane change ΔV)
5. Propagate the initial LEO orbit until the spacecraft is at an angular distance from the outgoing asymptote equal to the supplement of η, and is moving away from the asymptote. This is the same as stopping on the outgoing crossing of the locus of injection points. Doing this will also place the final departure maneuver on locus of injection points.
6. Target a maneuver at the locus of injection points to raise apogee. The amount to raise apogee depends on several factors, including hardware mass fractions, timing, propulsion storage, and operational considerations. For this paper, a fixed apogee of 400,000 km was used for all scenarios to simplify analysis, but this is an arbitrary value. The actual value used could be varied to accommodate other mission constraints.
7. Target a plane-change maneuver at apogee to achieve the desired perigee altitude AND rotate the orbit plane to contain the outgoing asymptote.
8. Propagate the orbit to perigee, which will be on the locus of injection points.
9. Target a maneuver at perigee to achieve the outgoing hyperbolic C3 from the direct transfer trajectory.

Figure 7 shows 2 three-burn depot transfers. On the left, the parking orbit inclination is less than that of the outgoing asymptote. On the right, the parking orbit inclination (51 deg) is greater than the declination of the outgoing asymptote. In both cases, the minimum ΔV solution occurs when the angle between the parking orbit and the asymptote vector is small. Both figures show the extremes, where the orbit is tangent to the locus of injection points. On the left, the "best" solution only crosses the locus of points, and requires a plane change maneuver at apogee. On the right, the best solution devolves to a 2-burn solution, and doesn't require a plane change maneuver at all. In this scenario, the parking orbit intersects the asymptote prior to the apogee raise maneuver.

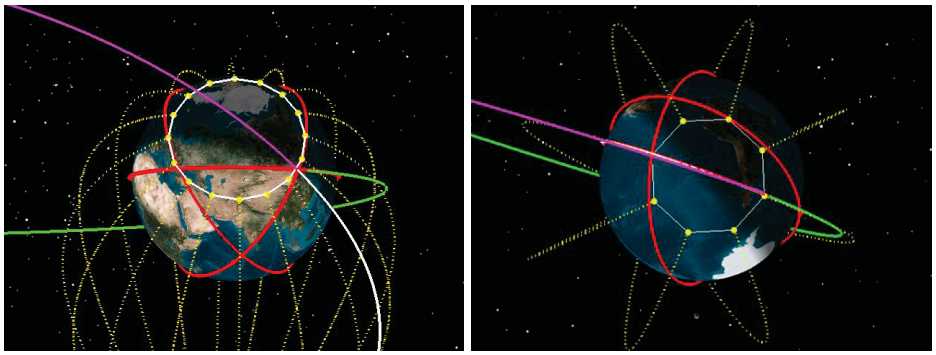

Figure 7: Depot orbit RAAN range: Minimum (crossing) and Extremes (tangent)

Figure 8 and Figure 9 shows the total injection ΔV for the 3-burn strategy (with the inclination less than the outgoing declination) as a function of the depot orbit Right Ascension of the Ascending Node (RAAN). The minimum ΔV occurs when the angle between the orbit plane and asymptote vector is the smallest. The maximums ΔV occur at the extremes, when the orbit plane is tangent to locus of injection points. Outside of this range of RAAN values, the orbit doesn't intersect the locus of points, and a 3-burn departure is not available without additional maneuvers. Figure 9 shows two minima. In both cases, the penalty for being off-nominal is minimal (> 200 m/sec).

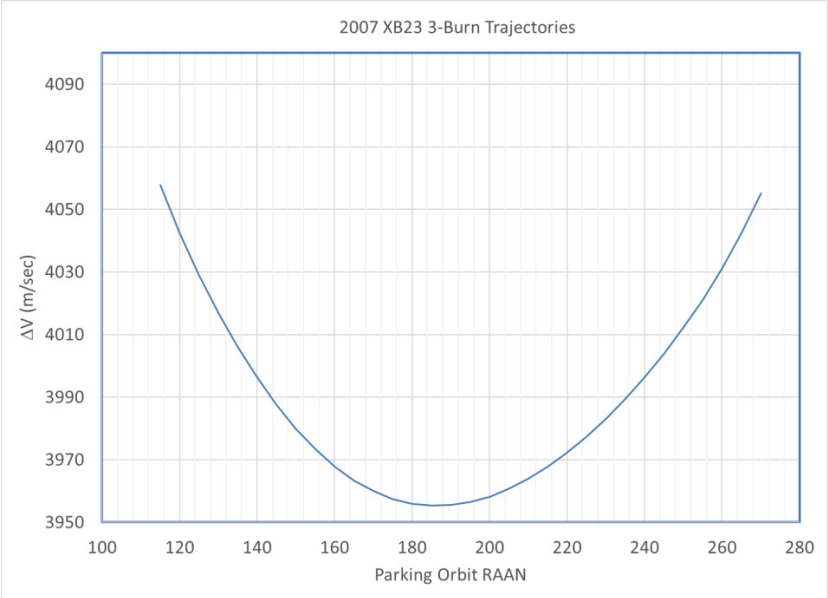

Figure 8: ΔV as a function RAAN (Inc > Dec)

Figure 9: ΔV as a function RAAN (Inc < Dec)

Additional Considerations and Features

There are several other considerations that could impact mission planning:

1. The 3-burn departure enables hitting almost any departure asymptote and departure date from a single stationary LEO depot.
2. This technique enables hitting declinations higher than the depot orbit inclination.
3. The losses compared to a direct injection can be very modest, which are primarily due to the plane change ΔV at apogee.
4. The phasing & plane-change trajectory should easily account for lunar and solar perturbations when the apogee is high.
5. This technique does increase mission duration by days to weeks, which could be a consideration for consumables or other mission constraints.
6. This adds two extra Van Allen Belt passes and two extra burns, which requires either a longer-lived upper stage, or requires the spacecraft to perform the extra burns.
7. The phasing orbit provides opportunity for checking out spacecraft before committing to deep-space (with abort opportunity if needed).
8. The spacecraft may not be able to deploy some structures until after periapsis, depending on what is needed to perform the maneuvers.
9. If the C3 is low, the angle of the locus cone will be small. If this occurs with a very high declination, it's possible that the depot-orbit will not ever intersect with locus of injection points, and this method won't work. This case would occur if the declination was greater than the sum of the inclination and the angle between the outgoing hyperbolic asymptote

and the eccentricity vector. These are expected to be rare cases, but they are physically possible.

Phasing the Depot Orbit Plane or Apogee Modification

In some cases, the depot orbit will not intersect with the locus of injection points at the time of departure. To ensure that the orbit plane will cross this locus at the desired time of departure, small changes to the depot orbit can be made months in advance, slightly changing the orbital nodal precession rate. This method is used to move the ISS orbit to the desired plane to support the ISS/Soyuz quick rendezvous technique[13]. (The ISS orbit is adjusted up to six months before the rendezvous, similar to the adjustment needed for the 3-burn rendezvous leaving from a similar LEO orbit.) This is like the Depot-Orbit-Phasing technique described above, but does not have the restriction that the depot orbit plane is perfectly co-planer with the departure asymptote. Therefore, the launch and rendezvous widow constraints are relaxed, lowering operational and schedule risks, therefore improving the likelihood of success.

If the depot is not able to perform the modest phasing suggested previously, this departure methodology still works, but the apogee of the initial phasing ellipse will need to be modified compared to the fixed 400,000km value used for this analysis. To minimize ΔV penalties, the stage would perform an apogee-raising maneuver when the depot orbital plane reaches the closest point to the anti-asymptote vector. Raising the apogee will freeze the RAAN by making the RAAN rate zero. The spacecraft would target an elliptical orbit with a period that was equal to some integer fraction of the time between that point and the desired departure burn. This would guarantee that the spacecraft and the departure stage were back at the perigee at the time of final departure burn. This may require longer time in the elliptical phasing orbit and potentially more Van Allen Belt passes, but otherwise still allows for a low ΔV departure even without adjusting the depot's RAAN.

EXAMPLE TRANSFERS

Our good friend and colleague at LMCO, Josh Hopkins[14], suggested several asteroids that would be interesting test cases. These included 2000 SG344 (roughly equatorial declinations), 1999 AO10 (low ΔV, short transfers, and higher declinations) and 2007 XB23 (challenging high-declination case) are below. We added 2001 QJ142 and 2008 EV5, as the transfers to these asteroids also involved higher declinations) and Mars, for comparison.

If the declination of the outgoing transfer asymptote is less than the depot orbit inclination (51.5 degrees for these cases) the 3-burn transfer devolves to a 2-burn solution (if the departure RAAN is flexible), with no need for a plane-change. If an off-asymptote RAAN is used, then the 3-burn strategy is required.

The advantage of the depot, though, is that a tug or expendable stage can be used to raise apogee, and the spacecraft only needs to perform the final deep-space injection maneuver at perigee.

Some of the asteroids, however, had declinations greater than the 51-degree inclination, and so utilize the 3-burn sequence.

Table 2 gives the ΔV values for all the transfers presented in this paper. Note that the "plane-change" maneuver values for the 2-burn transfer types are not perfectly zero. This maneuver, in addition to causing any needed plane change, also adjusts the perigee altitude height so that the solar and lunar gravitational effects don't cause a re-entry. (The altitude used for this study was 300 km.)

Table 2: ΔV values for deep-space transfers

	Transfer Year	Type	ΔV to Apo	Plane Change ΔV	Departure ΔV	Total ΔV
Object			m/sec	m/sec	m/sec	m/sec
1999 AO10	2024	2-Burn	3125.06	8.30	137.47	3270.83
2000 SG344	2024	2-Burn	3122.09	1.88	135.81	3259.78
Mars	2026	2-Burn	3124.05	10.69	502.83	3637.57
2001 QJ142	2020	3-Burn	3124.32	96.22	289.60	3510.15
2007 XB23	2025	3-Burn	3152.58	99.38	703.31	3955.28
2008 EV5	2020	3-Burn	3123.68	110.03	816.56	4050.27

Two-Burn Transfers

Figure 10 and **Figure 11** show a 2-burn departure to the asteroid 1999 AO10, which for this opportunity trajectory in the year 2024 requires a declination of -37.7 degrees and a departure C3 of 1.0264 km^2/sec^2. This low C3 yields a small locus of injection points. The right side of Figure 10 shows the baseline direct transfer from a LEO depot orbit inclined at 37.7 degrees. In **Figure 11** the depot trajectory is inclined at 51 degrees, and the phasing orbit can be seen: it is green from the apogee raise maneuver to apogee, and magenta back to perigee. Then the hyperbolic transfer trajectory is shown in white.

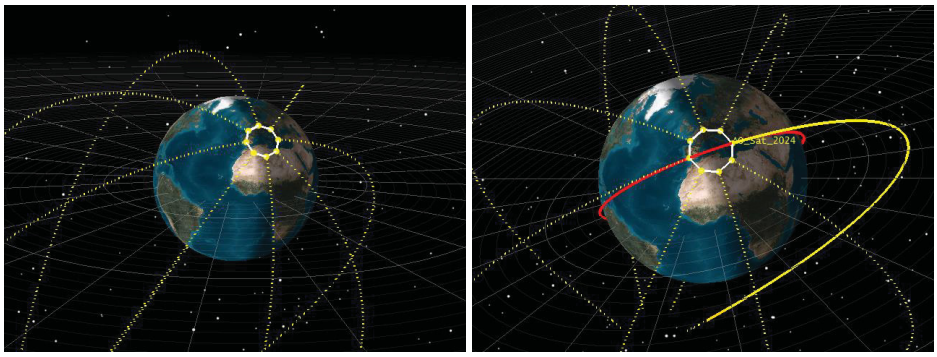

Figure 10: 1999 AO10 Locus of Injection Points and Direct Departure Trajectory from ideal inclination depot orbit

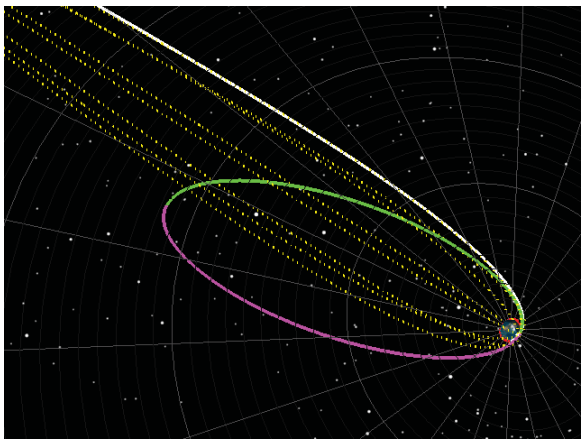

Figure 11: 1999 AO10 2-Burn Departure from 51 degrees depot orbit

Figure 12 and **Figure 13** show a 2-burn departure to the asteroid 2000 SG344, which for this opportunity trajectory in the year 2024 requires a declination of 15.05 degrees and a departure C3 of 0.9535 km^2/sec^2. This is also a low C3 which yields a small locus of injection points. The right side of the first figure shows the baseline direct transfer from a LEO depot orbit inclined at the ideal inclination equal to the declination (15.05) degrees. In **Figure 12** the depot trajectory is inclined at 51 degrees, and the phasing orbit can be seen: it is green from the apogee raise maneuver to apogee, and magenta back to perigee. Then the hyperbolic transfer trajectory is shown in white.

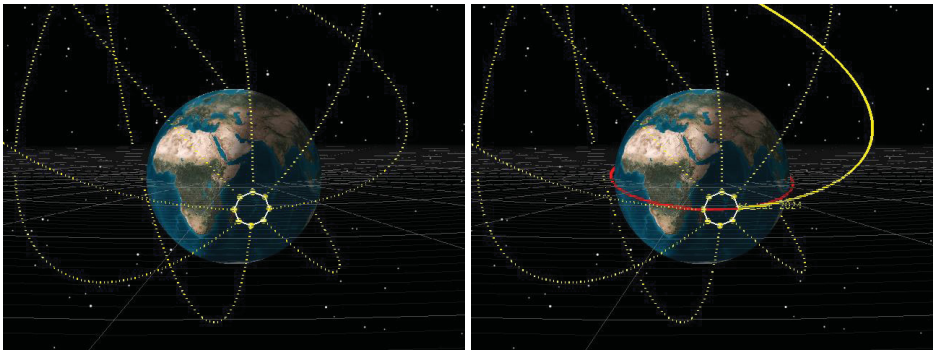

Figure 12: 2000 SG344 Trajectory Locus of Injection Points and Direct Departure Trajectory with ideal inclination depot orbit

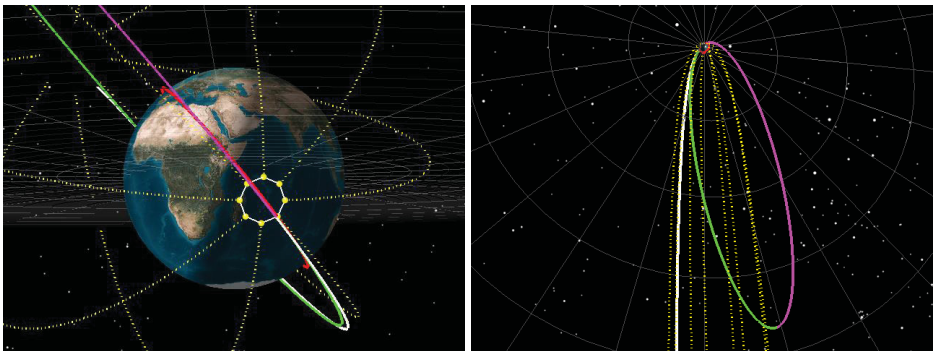

Figure 13: 2000 SG344 Trajectory 2-Burn Departure from 51 degrees inclination depot orbit

Figure 14 and **Figure 15** show a 2-burn sequence to Mars in the year 2026. For this trajectory, the declination is 29.54 degrees and the departure C3 is 9.1759 km^2/sec^2.

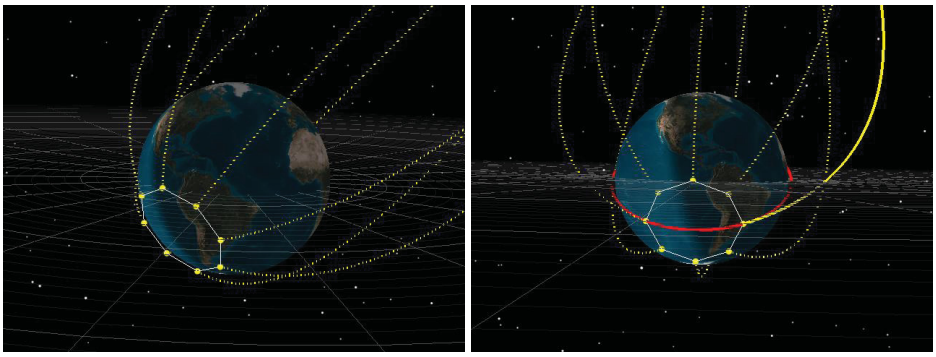

Figure 14: Mars Locus of Injection Points and Direct Departure Trajectory with ideal inclination

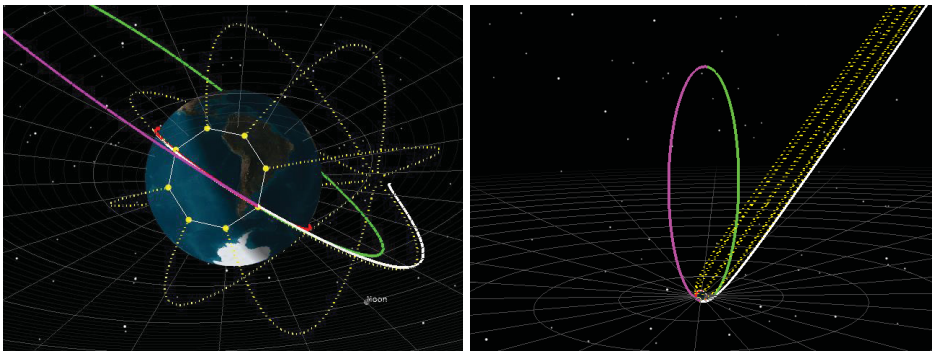

Figure 15: 2026 Mars 2-Burn Departure from 51 degrees inclination depot orbit

Three-Burn Solutions

Figure 16 and **Figure 17** show transfer trajectories to 2007 XB23 leaving Earth in the year 2025. The declination is -71.93 degrees, which is greater than the inclination, thus requiring the plane-change maneuver at apogee. The departure C3 energy is 13.76 km^2/sec^2.

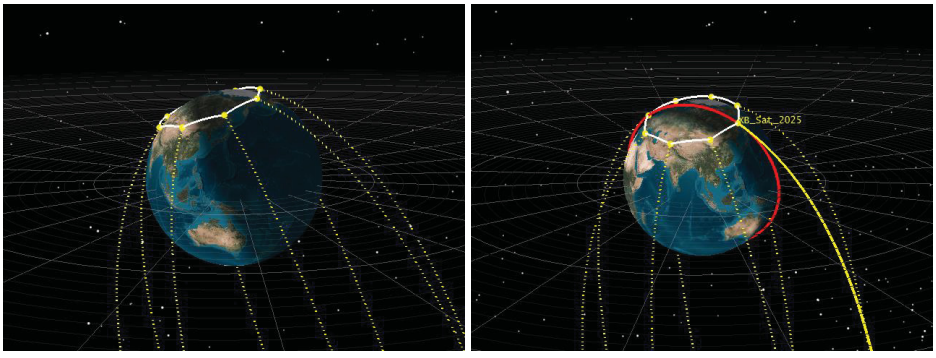

Figure 16: 2007 XB23 Trajectory Locus of Injection Points and Direct Departure Trajectory from ideal inclination depot orbit

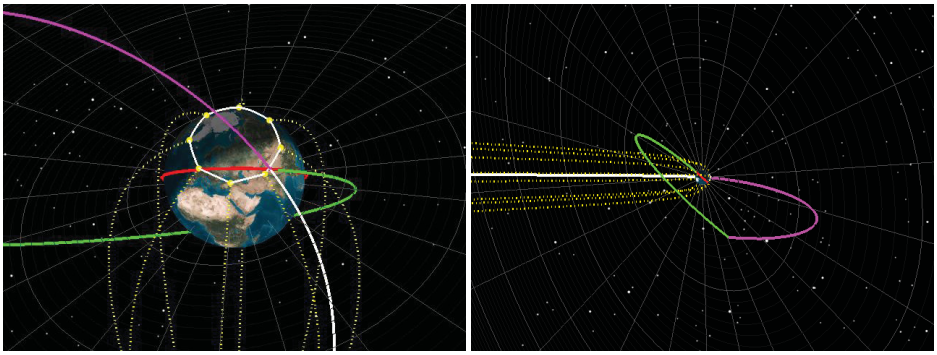

Figure 17: 2007 XB23 2025 Trajectory 3-Burn Departure from 51 degree inclination depot orbit

Figure 18 and **Figure 19** show the transfer trajectories to 2008 EV5 leaving Earth in the year 2020. The declination is -74.45 degrees, which, again, is greater than the inclination, thus requiring the plane-change maneuver at apogee. The results of the plane-change maneuver can be clearly seen on the right half of Figure 19 as the trajectory makes a sharp turn at apogee as it goes from green to magenta. The departure C3 energy is 16.391 km^2/sec^2. This high C3 yields a large locus of injection points, which can be seen in the figures.

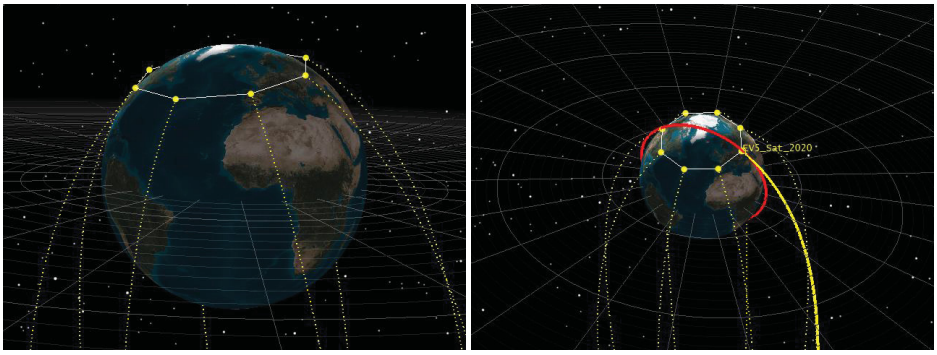

Figure 18: 2008 EV5 Trajectory Locus of Injection Points and Direct Departure Trajectory from an ideal inclination depot orbit

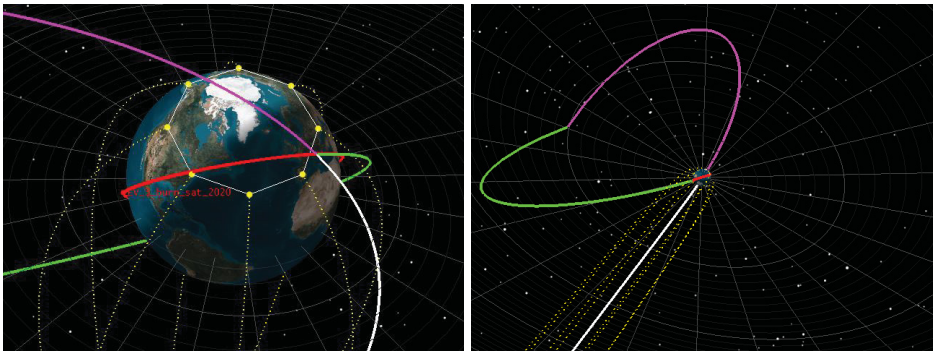

Figure 19: 2008 EV5 3-Burn Departure

Figure 20 and **Figure 21** show the transfer trajectories to 2001 QJ142 leaving Earth in the year 2020. The declination is 64.2 degrees, which, again, is greater than the inclination, thus requiring the plane-change maneuver at apogee. The results of the plane-change maneuver can be clearly seen on the right half of Figure 21 as the trajectory makes a sharp turn at apogee as it goes from green to magenta. The departure C3 energy is 4.36 km^2/sec^2.

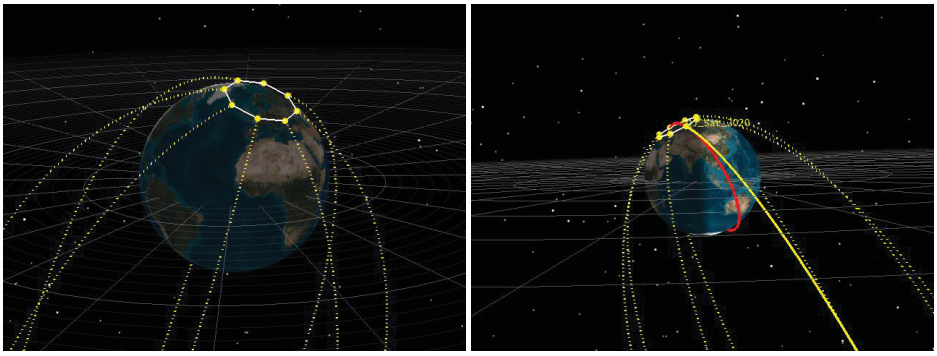

Figure 20: 2001 QJ142 Trajectory Locus of Injection Points and Direct Departure Trajectory from an ideal inclined orbit

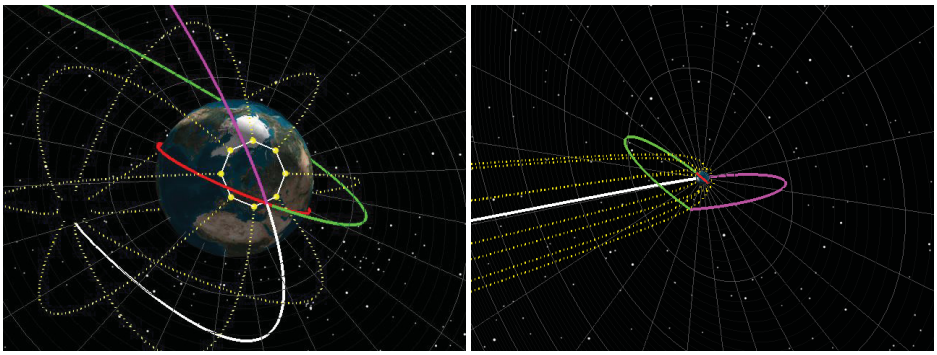

Figure 21: 2001 QJ142 3-Burn Departure Trajectory from a 51 degree inclination depot orbit

CONCLUSION

The 3-Burn departure is described, and examples using real asteroids were given. The incremental ΔV is small compared to the total, and can be compared with other methods. The 3-burn technique enables departures to deep-space destinations, and can take advantage of a depot in LEO.

For situations when the outgoing asymptote is less than the inclination of the depot orbit, the 3-burn case turns into a 2-burn case. However, because of solar and lunar perturbations, there is a small maneuver at apogee to control perigee altitude, so there really are 3 maneuvers needed for all of these cases.

Ephemerides are available from the authors upon request in a variety of formats. Additional information may be found on the Astrogator's Guild blog at: www.astrogatorsguild.com which is managed by authors Carrico and Loucks.

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