# Practicalities and Difficulties of a Mission to 'Oumuamua

LEAD FEATURE

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Our i4is colleague Adam Hibberd has driven the computational thinking behind our proposals for missions to interstellar objects (ISOs) such as the still-mysterious 1I/'Oumuamua. Here he considers some broader issues in reaching and investigating this asteroid-like object - or is it just a very weird comet? Astronomers remain divided and some have even suggested that this, our first observed, ISO, will remain a mystery forever. The i4is Project Lyra team have shown that this need not be so. But what are the challenges for an intercept mission?

So we have discovered this very weird object, how do we get to it? I have researched the general theoretical feasibility of trajectories to 'Oumuamua using some software I developed called OITS (Optimum Interplanetary Trajectory Software (Ref.1) and everything is looking fine as far as that is concerned, see Hibberd et al, Hein et al (Ref. 2 & 3) – it seems doable.

However what about the practicalities? What are the main details to address and are there any limitations which could render the mission infeasible and deny us our scientific prize? The following is a peek at some of these - with a focus on trajectory considerations - which are not addressed by the Project Lyra papers (spoiler alert – generally speaking it's all looking rather rosy).

Let us recount the story of the spacecraft's (s/c) journey to 'Oumuamua:

1) First there is a launch from Earth.

2) There then follows a long drawn-out 3 year near-elliptical orbit (actually two hemiellipses stuck together) which results in the s/c returning to its home planet, Earth. As we shall see, this phase is just the quiet before the storm. A Deep Space Manoeuvre (DSM) at the farthest point from the sun on this ellipse (aphelion) slows the s/c down and alters the return encounter kinematics with respect to the Earth. You may ask, what is the possible benefit in such a journey if the s/c merely ends up where it started off? Well to mission designers it is known as a '$V_\infty$ Leveraging Manoeuvre' and amplifies the s/c's kinetic energy relative to the Earth, and reduces the velocity increment, $\Delta V$* (and so rocket fuel) required to get to Jupiter.

3) Arriving at Jupiter is where the action starts. With a reverse gravitational assist (GA), the s/c is dramatically slowed by the pull of Jupiter's immense gravity to such an extent as to be almost brought to a halt relative to the sun. The purpose of this reverse-GA is to allow the s/c to start falling towards the sun in a free-fall, gradually accelerating more and more as the sun is approached.

4) A very close approach to the sun then ensues and so it is a matter of 'hang on to your hats' (or maybe your parasols) over this hazardous segment of the trajectory, where the awesome mass and might of the sun in its proximity exert their terrifying influences. Its huge mass is the very reason this hazardous approach is necessary, because in a gravitational well, the maximum benefit of a $\Delta V$ kick occurs at the point of closest approach (perihelion). This is the Oberth effect and this whole slingshot manoeuvre is known as an Oberth manoeuvre. The power of the sun, although of course fundamental to life on Earth, is of no help to us whatsoever, as unshielded this would create huge thermal problems for the s/c. We therefore need a heat shield.

5) There is then a long coast before we arrive at 'Oumuamua 22 years after the launch from Earth and at a distance from the sun of around 200 AU, outside the heliopause and into the pristine Interstellar Medium.

* $\Delta V$ here is in normal font. This is the standard convention for a scalar quantity. But velocity has both magnitude and direction so change in velocity is shown in bold font $\mathbf{\Delta V}$. The magnitude of $\mathbf{\Delta V}$ is $|\mathbf{\Delta V}| = \Delta V$ in normal font (ie not **bold**) since it is a scalar, a value with only magnitude, not direction.

For this article, two of the ‘Oumuamua mission trajectory options A & B are summarised in Tables 1 & 2 respectively. We shall concentrate on Trajectory A, refer to Figure 1. Trajectory B is the same sequence of encounters as A but with a lower Solar Oberth of 4 Solar Radii.

Table 1: E-DSM-E-J-6SR-1I Trajectory A **ΔV** Analysis

| Encounter Number | Planet | Date | Arrival Speed (km/s) | Departure Speed (km/s) | ΔV at encounter (km/s) | Cumulative ΔV (km/s) | Periapsis Altitude (km) |
|---|---|---|---|---|---|---|---|
| 1 | Earth | 2030 JUN 09 18:53:14 | 0 | 7106.2 | 7.1062 | 7.1062 | N/A |
| 2 | DSM | 2031 NOV 25 11:39:06 | 11.5701 | 11.0052 | 0.6593 | 7.7656 | N/A |
| 3 | Earth | 2033 APR 17 17:26:14 | 11.9715 | 12.4288 | 0.3395 | 8.1051 | 200.0 |
| 4 | Jupiter | 2034 JUL 12 11:54:56 | 14.6104 | 14.4661 | 0.0674 | 8.174 | 262748.9 |
| 5 | 6 Solar Radii | 2036 FEB 24 04:32:46 | 251.2642 | 258.338 | 7.171 | 15.3434 | N/A |
| 6 | ‘Oumuamua | 2052 JUL 29 04:32:46 | 30.6801 | 30.6801 | 0 | 15.3434 | N/A |

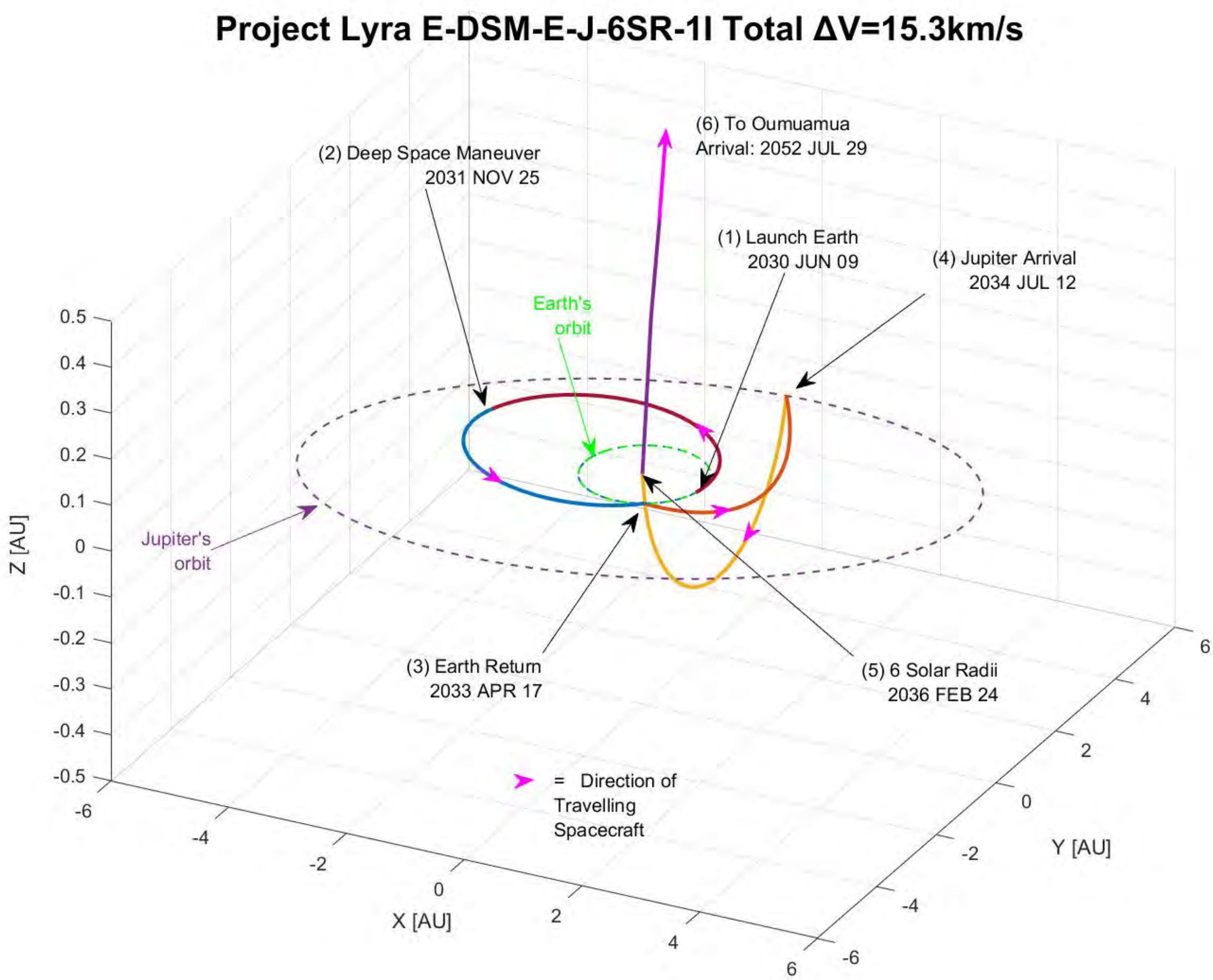


Figure 1: Trajectory A (see text) to ‘Oumuamua with launch in 2030.

Table 2: E-DSM-E-J-4SR-1I Trajectory B $\Delta \mathbf{V}$ Analysis

| **Encounter Number** | **Planet** | **Date** | **Arrival Speed (km/s)** | **Departure Speed (km/s)** | **ΔV at encounter (km/s)** | **Cumulative ΔV (km/s)** | **Periapsis Altitude (km)** |
|---|---|---|---|---|---|---|---|
| 1 | Earth | 2030 APR 11 03:44:11 | 0 | 7.0054 | 7.0054 | 7.0054 | N/A |
| 2 | DSM | 2031 OCT 29 00:02:21 | 11.4996 | 11.1093 | 0.500 | 7.5054 | N/A |
| 3 | Earth | 2033 MAY 21 18:33:07 | 10.7061 | 11.8454 | 0.8149 | 8.3203 | 200 |
| 4 | Jupiter | 2034 JUL 12 16:06:40 | 15.1624 | 19.2138 | 1.2396 | 9.5599 | 17204.3 |
| 5 | 4 Solar Radii | 2035 JUL 17 15:18:59 | 308.294 | 313.6442 | 5.388 | 14.9479 | N/A |
| 6 | 'Oumuamua | 2051 DEC 20 15:18:59 | 29.6896 | 29.6896 | 0 | 14.9479 | N/A |

So let us negotiate each of the challenges (1) to (5) above one at a time, assessing their feasibility on the way.

## 1) Launch Earth

When addressing the issue of launch, the necessary conditions must be referenced to the Earth, so rather than a heliocentric reference frame, we need to relate everything to a geocentric reference frame. We know that first of all, the s/c must embark on an interplanetary trajectory and this necessitates departing Earth's gravitational sphere of influence (SOI). Therefore an Earth escape orbit (also known as a hyperbolic orbit) is required, and the transition between this Earth-referenced escape orbit into the sun-referenced interplanetary elliptical orbit can be assumed to be at this departure point. In addition, the speed relative to Earth on leaving its SOI can be assumed equivalent to the 'hyperbolic excess', the theoretical speed at infinity of the escape orbit. Generally speaking the lower the magnitude of this hyperbolic excess, the greater the payload mass capability of a particular launch vehicle. As we are necessarily trying to maximise this mass, we are equivalently attempting to minimise this hyperbolic excess.

To this end, let us denote the velocity on departing the SOI as $\mathbf{\Delta V}$, the magnitude of which is the hyperbolic excess speed $\Delta V$. From Table 1 we see that the hyperbolic excess required at Earth is 7.106 km/s. As a fundamental requirement then, we need a powerful enough launcher to achieve an escape orbit with this value of hyperbolic excess. The payload capability of a particular launch vehicle to an escape mission is usually provided in its user guide and is generally not given as a function of hyperbolic excess, but usually the square of this, designated 'C3', or the 'Characteristic Energy'. We shall assume that the launcher utilised for the 'Oumuamua mission is either the mighty still-not-yet-operational NASA Space Launch System (SLS), or the significantly-less-mighty (at least for escape missions) SpaceX Falcon Heavy, both of which can achieve the C3 value of $7.106^2 = 50.5\ km^2/s^2$. Note as well that the payload mass for the SLS would be significantly higher than for the Falcon Heavy.

The escape asymptote – the direction the s/c should ultimately head off in - is in the direction of $\mathbf{\Delta V}$. For an escape mission, a particular launcher will need to achieve an orbit with a launcher-dependent perigee distance $R_p$ (closest approach to Earth) greater than the Earth's radius. The distance $R_p$ and excess speed $\Delta V$ combine together to define the angle, $\eta$, between the escape asymptote and the injection point (which can usually be assumed to be at $R_p$).

Theoretically, ignoring launch vehicle considerations, the plane and therefore orbital inclination of the escape orbit can be any value as is illustrated in Figure 2. This results in a theoretical locus of points which an unspecified hypothetical launcher could target for orbital insertion.

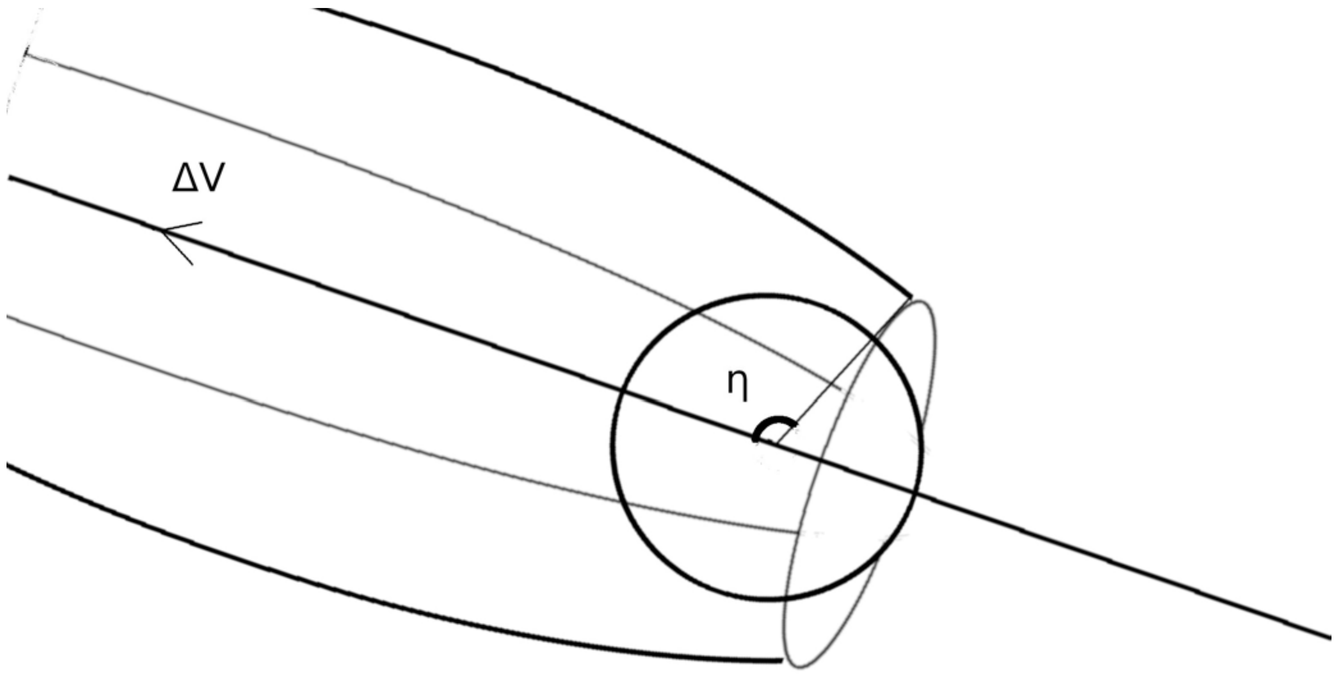


Figure 2: Possible Escape Orbits Around Earth, All Resulting in the Same Required Escape Asymptote.

Figure 3 shows a 3D image of the Earth in a reference frame with origin at the centre of the Earth and looking from a vantage point perpendicular to the meridian of the launch site. The geocentric launch site latitude φ is provided.

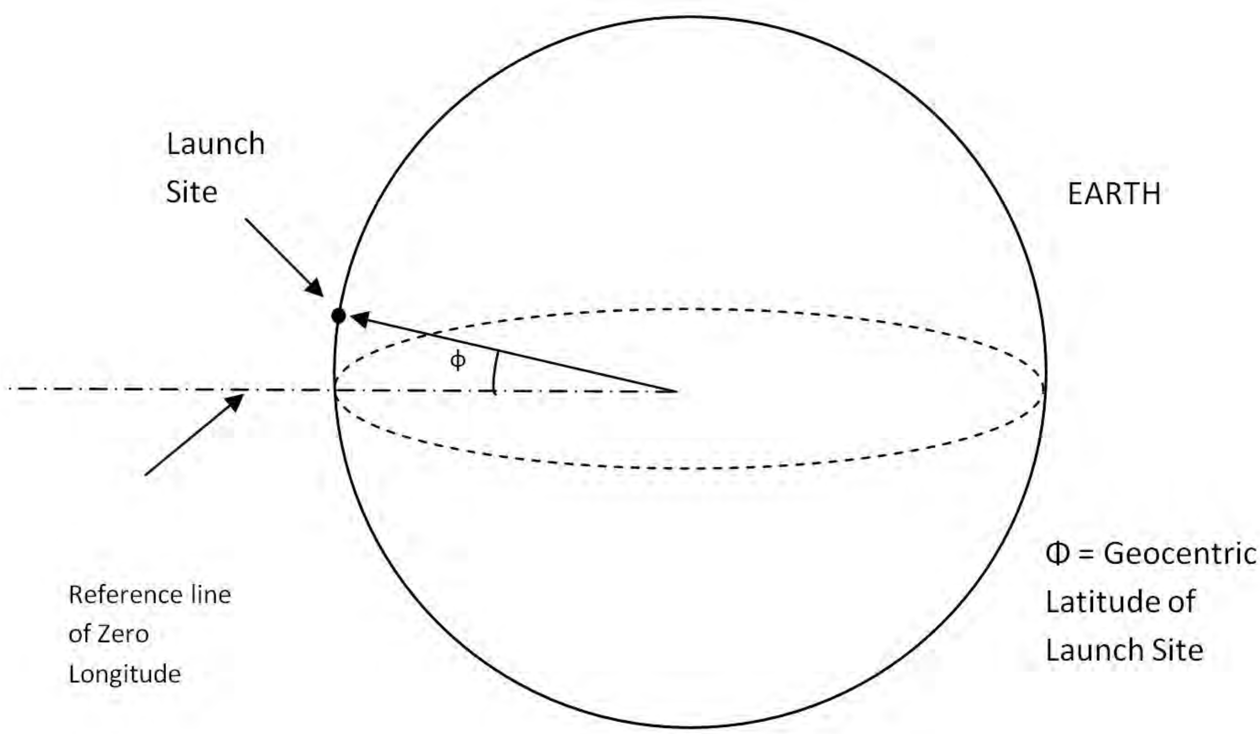


Figure 3: The Earth in 3D - Looking from an Angle Perpendicular to the Launch Site Meridian.

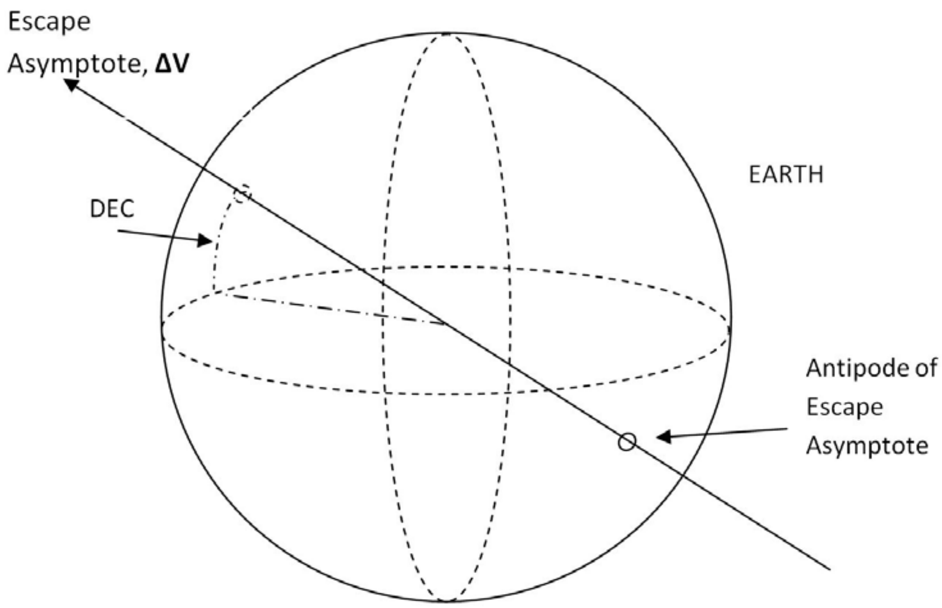


Figure 4: The Escape Asymptote Direction and its Declination.

Figure 4 shows a required escape asymptote with respect to the Earth and the direction of this can be completely specified by two parameters - Right Ascension (RA) and Declination (DEC), the latter is provided in the diagram and to all intents and purposes is equivalent to the latitude of the escape asymptote. The antipode of the escape asymptote is provided in this figure as well.

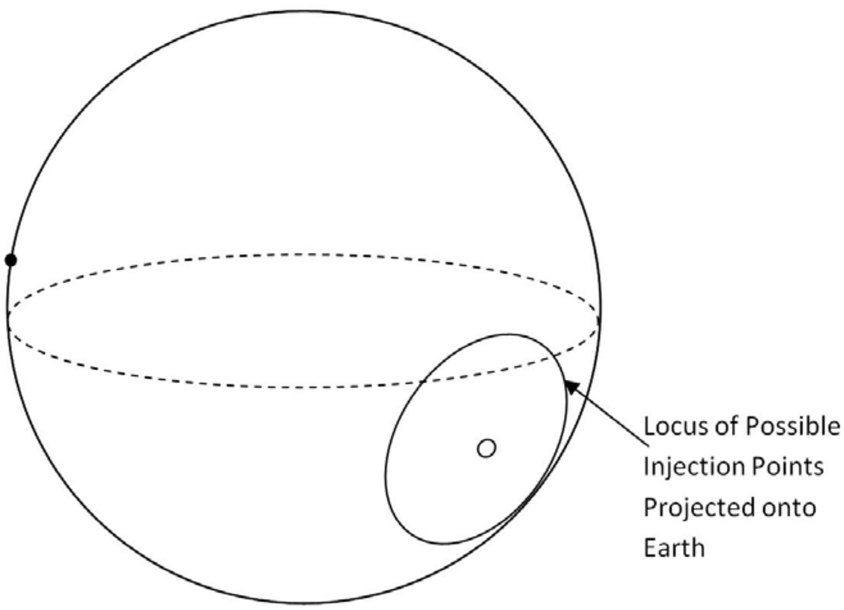


Figure 5: Locus of Possible Orbital Insertion Points Which would Allow Acquisition of the Required Escape Asymptote.

Figure 5 shows the locus of orbital insertion points projected onto the Earth assuming the inclination of the target escape orbit can be free. Ideally the orbit achieved by a particular launcher must pass through the antipode of the escape asymptote. Therefore, in order to avoid wasting precious fuel in changing the plane of the launcher's trajectory, the ground track of the launcher ascent trajectory (ie the projection of the launcher's trajectory onto the Earth) must also pass through this point. This constrains the possible inclinations of escape orbits viable for a particular launch vehicle. Furthermore, ideally we wish the inclination of the target orbit to equate to the latitude of the launch site because then the thrust direction at launch (the thrust azimuth) is to the east which reaps maximum benefit from Earth's rotational velocity.

The arc between launch site and injection is provided in Figure 6, and we'll denote it here as ψ. Figure 7 shows the ground track of the launcher's trajectory in 2D.

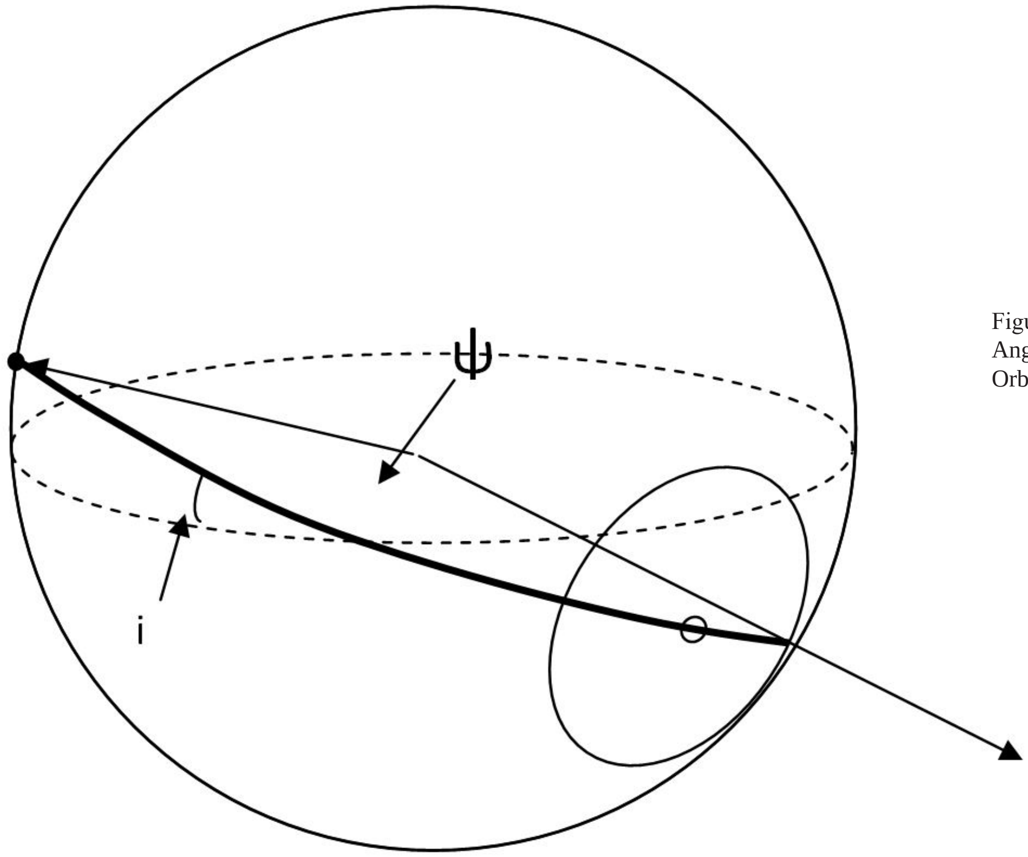


Figure 6: Definition of ψ, the Angle Between Launch and Orbital Insertion.

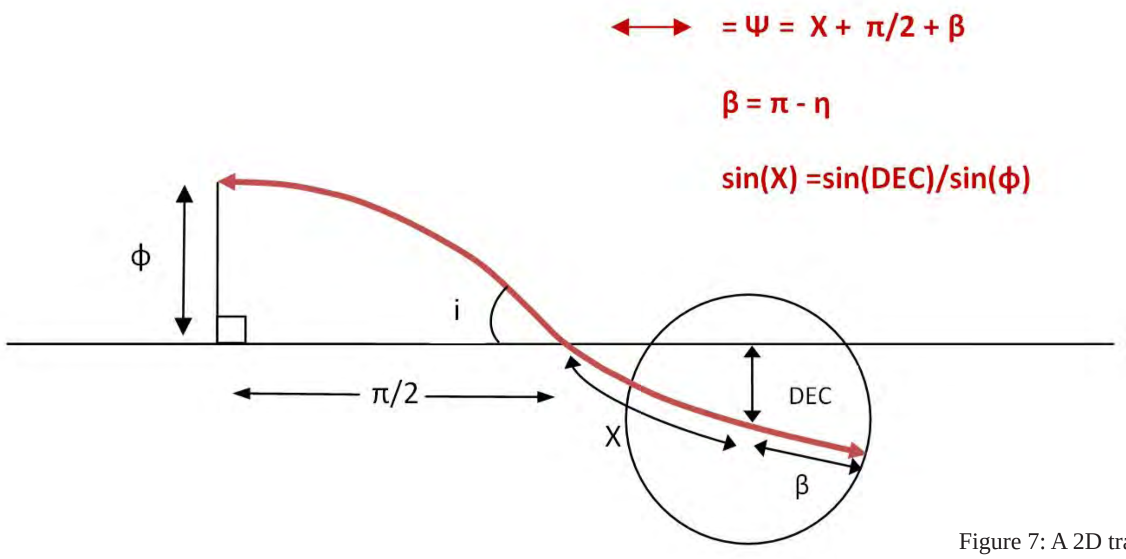


Figure 7: A 2D transformation of Figure 6, Demonstrating the Relationships Between Relevant Parameters (Using Spherical Trigonometry).

We have enough information now to do some number crunching – using the equations provided in Figure 7 - for both of the two selected vehicle options SLS and Falcon Heavy. We shall assume in both cases that $R_p$ has an altitude of 185 km (though the results don't change much for a wide range of possible perigees) and also in both cases that the geocentric launch latitude is about 28.6°. It can be calculated that η = 112.1° for the escape asymptote. From Table 3 (next page) we have DEC= -5.319°. This gives a value of ψ = 146.7°.

Table 3 Detailed Data on Each of the Encounters in Turn (Trajectory A)

| | Units | Earth | DSM | Earth | Jupiter | 6 Solar Radii | ‘Oumuamua |
|---|---|---|---|---|---|---|---|
| **Distance from Sun** | AU | 1.015 | 3.2 | 1.004 | 4.959 | 0.028 | 199.509 |
| **Longitude** | degrees | -101.433 | 72.526 | -152.446 | 0.973 | 180 | 8.637 |
| **Latitude** | degrees | 0.004 | 0.014 | 0.002 | -1.285 | 4.566 | 23.231 |
| **True Anomaly** | degrees | 37.696 | 180 | 36.455 | 66.225 | 180 | 145.186 |
| **Angle between departure and arrival velocities** | degrees | N/A | 1.726 | 33.651 | 79.882 | 0.265 | N/A |
| **Angle between arrival velocity and periapsis** | degrees | N/A | N/A | 107.287 | 129.771 | N/A | N/A |
| **Angle between departure velocity and periapsis** | degrees | N/A | N/A | 106.364 | 130.112 | N/A | N/A |
| **Periapsis Radius** | km | N/A | N/A | 6578 | 334241 | N/A | N/A |
| **Impact Parameter** | km | N/A | N/A | 8727 | 299948 | N/A | N/A |
| **Miss Distance** | km | N/A | N/A | 8937 | 713054 | N/A | N/A |
| **Arrival Velocity Right Ascension (RA)** | degrees | N/A | 161.073 | 0.678 | 340.112 | 285.856 | 358.321 |
| **Arrival Velocity Declination (DEC)** | degrees | N/A | 7.823 | 0.211 | -7.913 | 82.807 | 24.475 |
| **Departure RA** | degrees | 345.339 | 162.747 | 333.81 | 257.417 | 287.86 | 358.321 |
| **Departure DEC** | degrees | -5.319 | 7.347 | -20.829 | -27.862 | 82.727 | 24.475 |

Thus we have discovered that for both launchers, the injection into the required escape orbit must occur at a significant distance downrange – in other words at a significant angular displacement - from the launch site. To be clear, this is feasible but obviously would require a significant coast arc in the 2nd Stages of both the Falcon Heavy and the SLS. Fortuitously both 2nd stages are indeed restartable.

But what is the direction of this hyperbolic excess, **ΔV**? What defines it? Well actually this requires stepping back a bit and looking at the situation from a heliocentric standpoint. In order to understand the situation we are presented with at launch, we need to look into the precise heliocentric orbit – ie the orbit around the sun - we are trying to escape into.

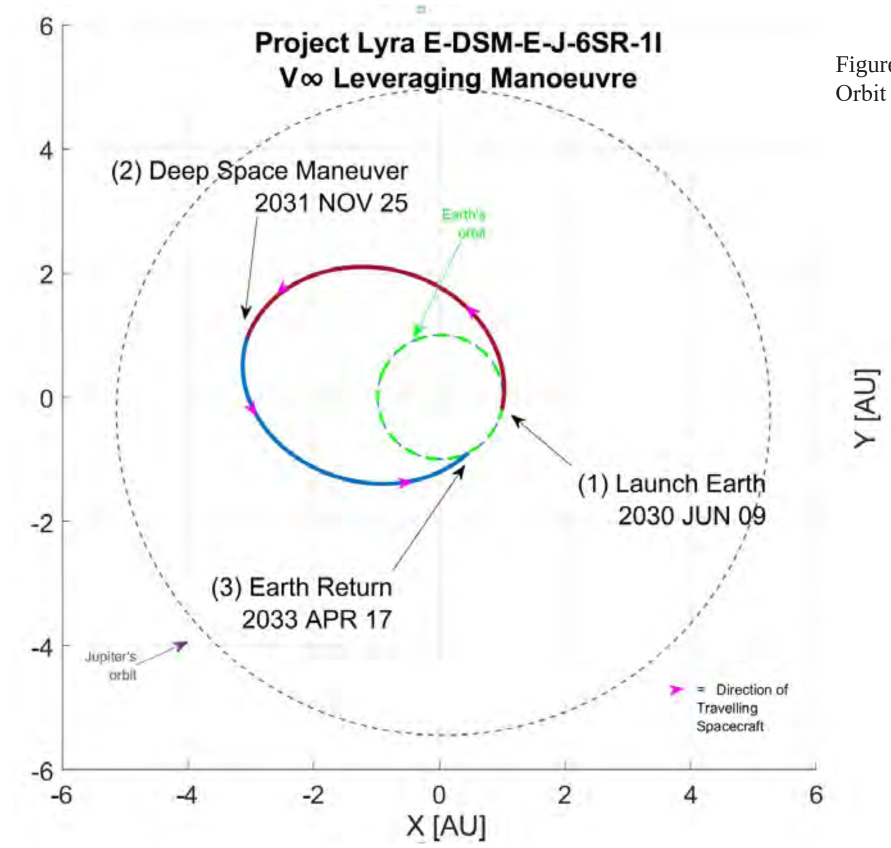


Figure 8: The Required Heliocentric Near-Elliptical Orbit - The V∞ Leveraging Manoeuvre.

Figure 8 shows the target heliocentric ellipse (actually two hemiellipses stuck together at a DSM) required for the V∞ Leveraging Manoeuvre. Figure 9 is the same as Figure 8 but shows the required heliocentric velocity vector (in pink) needed at Earth in order to follow this heliocentric ellipse. This pink velocity vector - let's call it **VDEP** - is relative to a sun reference frame. Observe that this direction is just about in line with Earth's orbital velocity, ie tangential to Earth's orbit. This makes sense because then we can best exploit Earth's own motion (**VE**) which has a considerable magnitude of around 30 km/s relative to the sun in this direction.

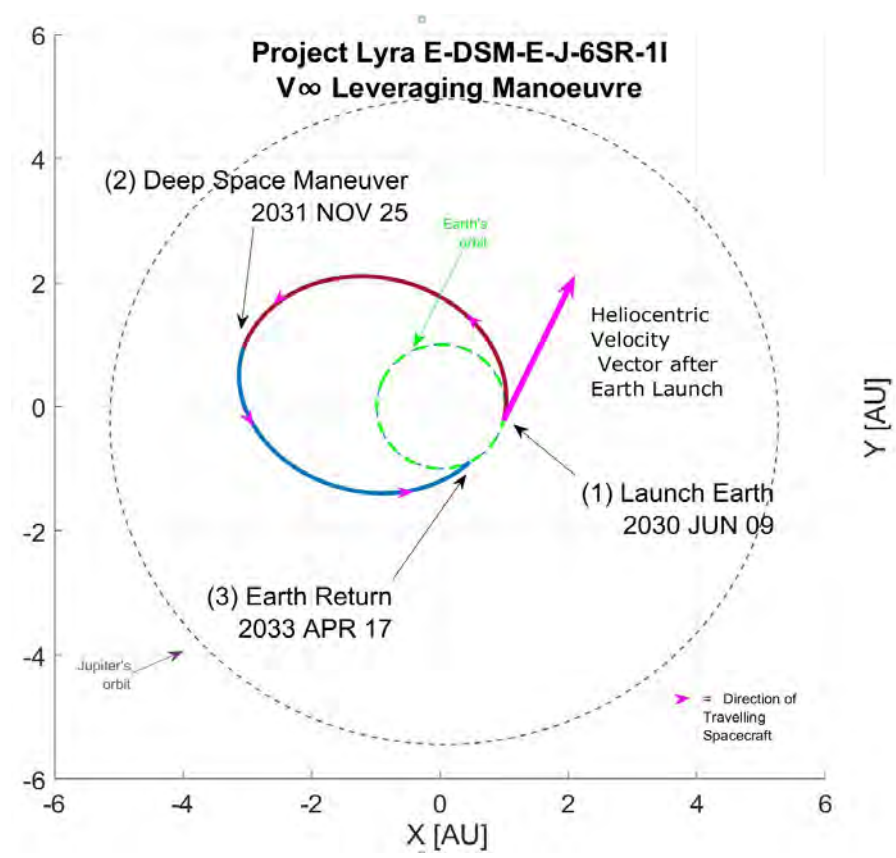


Figure 9: The Pink Arrow Shows the Required Heliocentric Velocity, VDEP, Needed at Earth to Achieve the V∞ Leveraging.

The desired **ΔV** at Earth in the heliocentric reference frame is the change in velocity relative to Earth needed to acquire a heliocentric velocity **VDEP**. It follows quite straightforwardly that:

**ΔV = VDEP – VE**

**VDEP** and **VE** are virtually aligned with each other and so **ΔV** will be aligned with them also. Because the Earth is orbiting the sun, the direction of **VE** is changing, by almost 1°/day, Thus the direction of heliocentric velocity we wish to acquire, **VDEP**, will also change with launch date in order to maintain co-linearity (and optimality) with **VE**. This means that due to Earth's orbital motion around the sun, RA will recede to the East with each day of launch delay.

So we have the direction of **ΔV** in the sun-centred reference frame (along Earth's direction of motion), we now need to transform this into an Earth-centred frame. It turns out this direction is lined up to the Earth's sun/shadow terminator. Figure 10 illustrates what is happening. As Earth spins on its axis, the location on Earth which points in this direction changes, but the time associated with this point will approximately be 6 am apparent solar time.

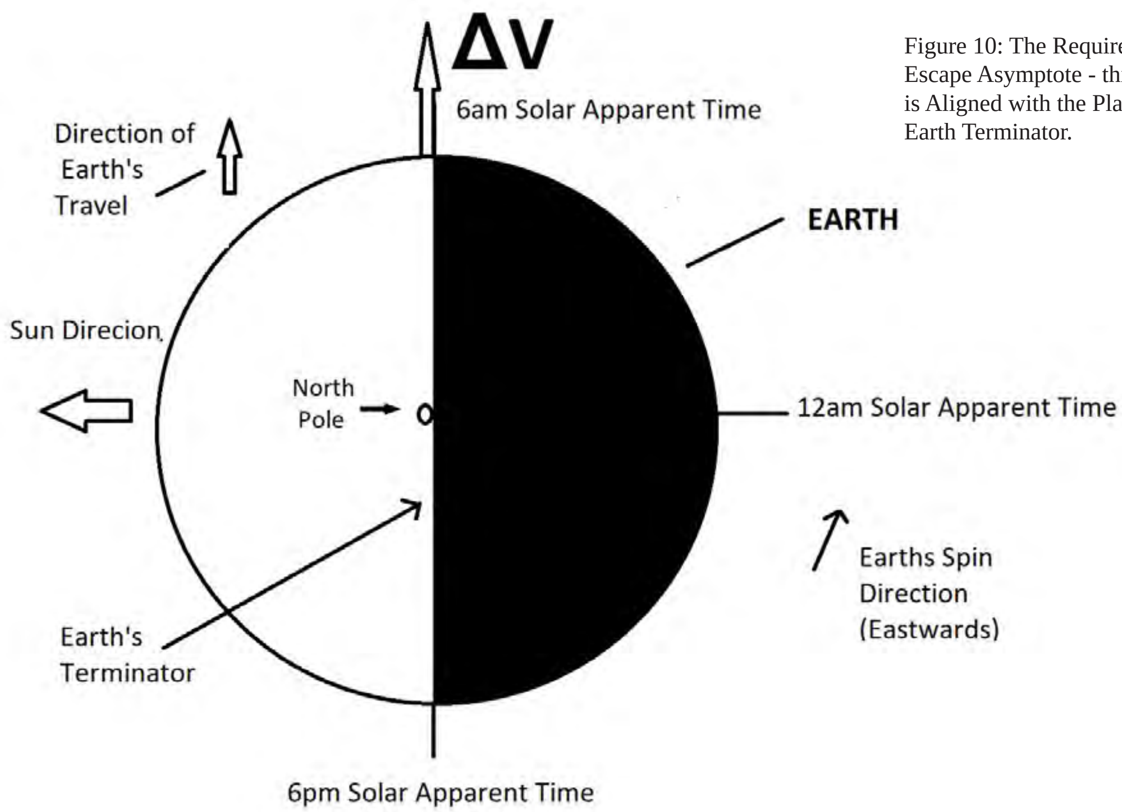


Figure 10: The Required Earth Escape Asymptote - this Direction is Aligned with the Plane of the Earth Terminator.

## 2) V∞ Leveraging Manoeuvre

Figure 8 gives a clear view of the V∞ Leveraging Manoeuvre from above the solar system projected onto the ecliptic – it is a near-elliptical heliocentric orbit with a return to Earth after approximately 3 years of uneventful travel. There is also a Deep Space Manoeuvre (DSM) in the middle of this (after 1 ½ years) where the s/c slows down with respect to the sun (another ΔV is applied, though from now onwards we shall define **ΔV** as an impulsive change in velocity, not the hyperbolic excess).

The natural question is what is the purpose of this manoeuvre? Well as I see it, there are three key advantages to this leveraging:

1) It reduces the total mission ΔV, by reducing the ΔV needed to get to Jupiter.

2) It extends the launch window compared to a direct mission to Jupiter.

3) It reduces the C3 at launch.

Figure 11 provides the evidence in plot form for the claims (1) & (2) above. It shows the total ΔV for a mission to 'Oumuamua with the V∞ Leveraging (the blue line) and without the Leveraging (the red line). The horizontal x-axis is the launch date compared to the theoretically optimal launch date – this optimal launch date being 9th June 2030 for the trajectory with leveraging, and 8th May 2033 without. Clearly the former scenario has a significantly lower mission ΔV of 15.3 km/s (at its optimal value) compared to 18.3 km/s for the latter (that's point (1) demonstrated). Furthermore the former scenario has a much more gradual rise in ΔV either side of the optimal launch date (which demonstrates point (2)) – particularly prior to this date. For (3) above, the C3 at the respective optimal launch dates are 50.5 $km^2/s^2$ with leveraging as compared to 120 $km^2/s^2$ for the direct to Jupiter trajectory. These are three significant performance enhancements and allow an appreciable improvement in the payload mass we can eventually get to 'Oumuamua. The theory behind how this manoeuvre actually works is rather beyond the remit of this article, but is explained in detail by Sims and Longuski Ref 6.

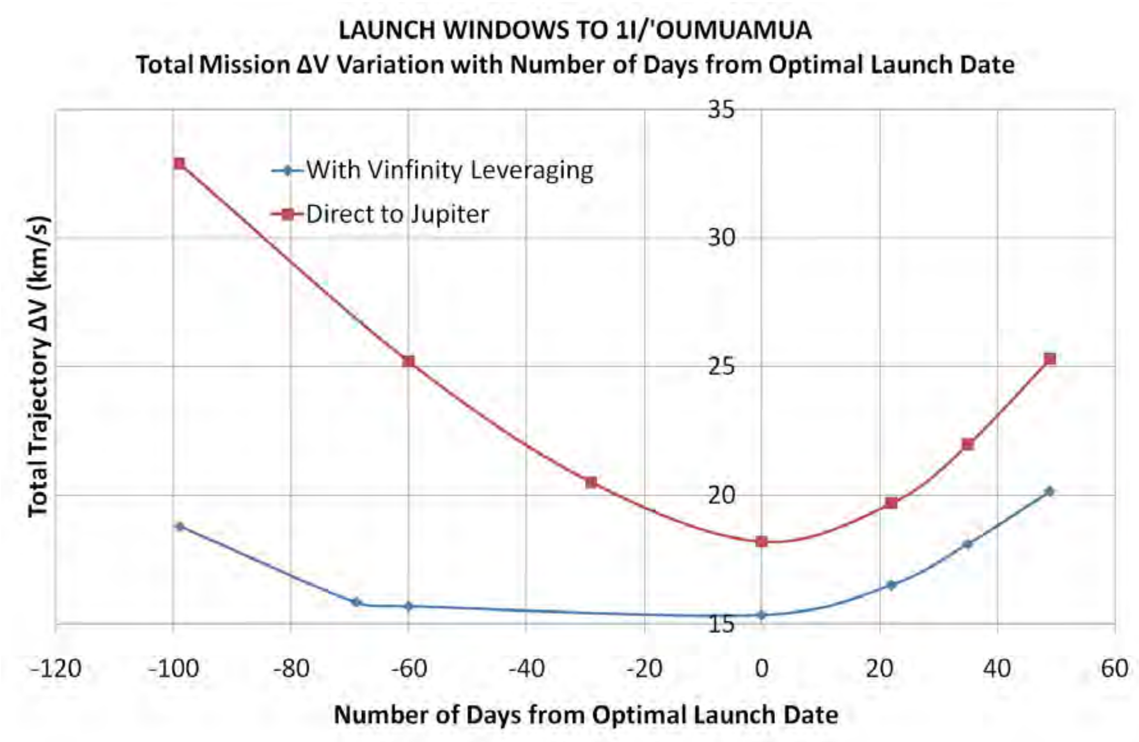


Figure 11: Plot of ΔV Dependency on Launch Date Showing the Clear Improvements of V∞ Leveraging.

It has to be said this is not an interesting phase of the overall journey. However note that the near-ellipse followed by the s/c is very close to the ecliptic plane. As a result of this low inclination, the spacecraft is obstructed by the sun for two separate windows of time, firstly on the out-bound leg of the journey (to aphelion) and then again on the in-bound leg (back towards perihelion - the Earth). The window opening and closing times for these two communication outages are from 2031 6th May 18:06 to 7th May 14:36 and from 2032 30th June 17:12 to 1st July 14:22 respectively.

A note on the aphelion distance of the DSM. The placement of this DSM is modelled in my software by an 'Intermediate Point' which allows the user to specify a distance from the sun, so then the heliocentric longitude and latitude can be optimized by the NOMAD optimizer (see Ref 1). In researching missions to 'Oumuamua, I set the value of this heliocentric distance at 3.2 AU. You may ask what was the reasoning behind this? Well the clue is we need the overall ellipse to have a time period of around 3 years. Kepler's third law relates the time period of an orbit to its semi-major axis (the semi-major axis is defined as half the distance between the perihelion and the aphelion points). The perihelion is taken as Earth's distance, which by definition is 1.0 AU, and so it is quite straight forward to calculate the required aphelion distance for the Intermediate Point from all this.

This ellipse will take the s/c right though the heart of the asteroid belt, no doubt affording plenty of opportunity for pictures.

Let us move onto the return encounter with Earth, where the s/c will come extremely close to the Earth, at a perigee altitude of 200 km. On the face of it this would seem a perilous approach, and what's more the speed of approach from infinity relative to Earth, VA, (the arrival hyperbolic excess) is 12.0 km/s. This is an alarmingly high speed for such a close approach to the planet - but how do we quantify the degree of danger?

Well Figures 12 & 13 show the definition of two important parameters for this encounter. Given a Perigee $R_p$ a speed of approach, VA, and the mass of Earth, there is a straight forward equation which provides what we shall call here the 'miss distance', or $R_{MISS}$ – defined as the nearest the s/c would have come to the Earth, assuming Earth had no gravitational pull. The second parameter to consider is the impact parameter, $R_{IMP}$, and is the minimum value of 'miss distance' needed to avoid collision with Earth. Clearly, we wish $R_{MISS} > R_{IMP}$.

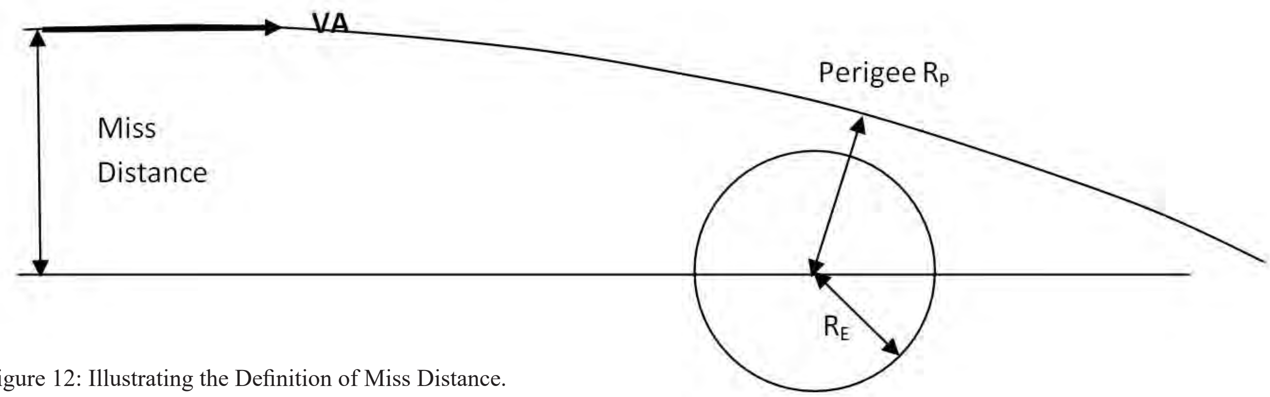


Figure 12: Illustrating the Definition of Miss Distance.

For the trajectory we are analysing here, we have:

$R_P$ = 6,578 km, $R_{MISS}$ = 8,937 km, $R_{IMP}$ =8,727 km

Thus there is indeed a narrow corridor – of around 210 km - which the s/c must travel through in order to avoid an impact with the Earth. Now at the time of writing this article, the Perseverance lander to Mars had arrived at Mars and missed its planned target in the Jezero crater by a mere 5 m. I think we can assume therefore, with all the navigational apparatus available in close proximity to the Earth (which are not available at Mars), that this is an entirely feasible fly-by of Earth. In addition there is a thrust of the on-board engines at perigee with $\Delta V$ of magnitude 0.3 km/s making this a powered fly-by of Earth. A small adjustment it would seem, but this would have been entirely unnecessary had there been sufficient scope for an even closer approach.

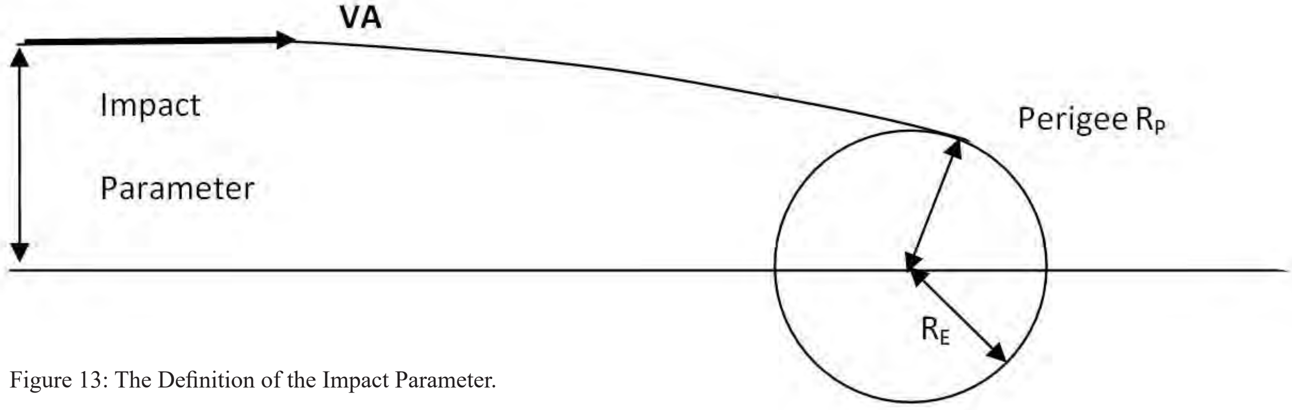


Figure 13: The Definition of the Impact Parameter.

## 3) Jupiter Encounter

What now of the encounter with Jupiter? This is a gravitational assist which doesn't accelerate the spacecraft, but quite the opposite, slowing it down dramatically from a heliocentric velocity of 15.7 km/s to one of 3.5 km/s.

How does it do this? Look at Figures 14-17. The key is that the s/c arrives at Jupiter's SOI ahead of Jupiter in its orbit around the sun. Observe that the component of the arrival heliocentric velocity perpendicular to the line to the sun is less than Jupiter's tangential velocity of 13.7 km/s, hence Jupiter catches up with the s/c and in effect ploughs into it.

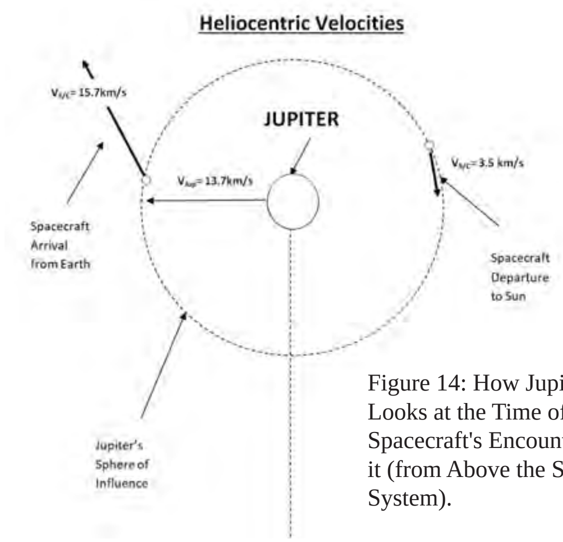


Figure 14: How Jupiter Looks at the Time of the Spacecraft's Encounter with it (from Above the Solar System).

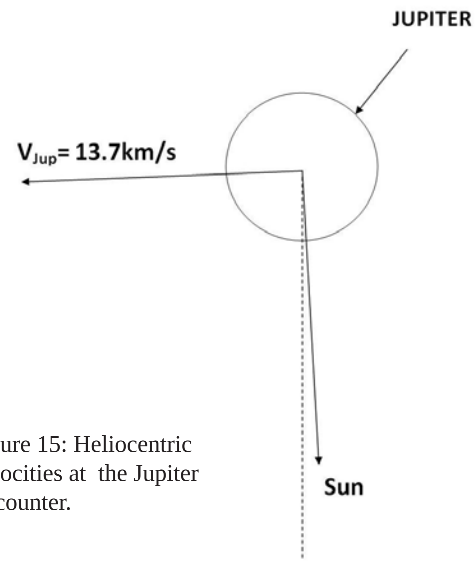


Figure 15: Heliocentric Velocities at the Jupiter Encounter.

If we subtract Jupiter's heliocentric velocity from the arrival velocity and departure velocity, we get everything relative to Jupiter, so we get a look at how Jupiter sees the encounter. As far as Jupiter is concerned, its gravitational pull is bending the approach velocity to eventually spit the s/c out backwards - in a direction almost tangential, but opposite, to Jupiter's motion.

By so doing, when Jupiter's heliocentric velocity is added back on in order to return to the sun-referenced frame, the s/c has actually slowed down quite dramatically.

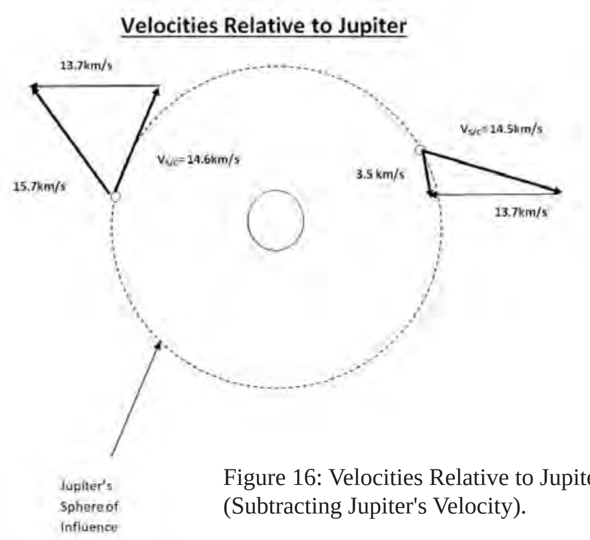


Figure 16: Velocities Relative to Jupiter (Subtracting Jupiter's Velocity).

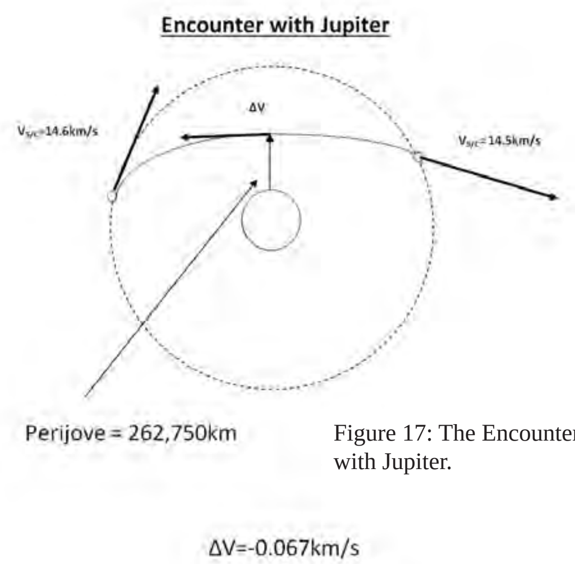


Figure 17: The Encounter with Jupiter.

The perijove altitude turns out at 262,750 km and one Jupiter radius is around 71,490 km, so the s/c will come within 4 Jupiter radii of Jupiter. We can perform the same analysis with Jupiter's encounter as we did with the Earth encounter in the previous section, we get the following:

$R_P$= 334.241 km, $R_{MISS}$= 713.054 km, $R_{IMP}$=299.948 km

Thus the arrival corridor to avoid a Jupiter collision is a fairly large value of 413,000 km.

### 4) Solar Oberth

Having encountered Jupiter, the s/c leaves Jupiter's SOI with a helpful nudge towards the sun of 3.5 km/s. As the sun is approached, so the acceleration of the s/c increases and is in fact proportional to the inverse square of the sun distance (of course by Newton's law of gravity). This free-fall phase of the trajectory lasts altogether just over 1½ years from leaving Jupiter, ample opportunity to develop a huge heliocentric speed higher than 250 km/s. At the time of writing, the Parker Solar Probe has yet to reach its maximum speed, which apparently will be around 200 km/s, and so the 'Oumuamua s/c would smash this record by a considerable amount.

Since the s/c will come within close proximity of the sun (a distance of 5 solar radii from the sun's surface), there is an inevitable question as to the possibility of the s/c being at some point obstructed by the sun, or alternatively transiting in front of the sun with respect to observers on Earth (and therefore the Deep Space Network), both of which would affect the communications link. Such an eventuality during this crucial period of the mission (after all, the success of the entire mission hinges on success here) would be extremely problematic and could compromise the mission. The news is good, however, serendipitously the Earth and the s/c are favourably aligned during the Solar Oberth, and no obscurations or transits arise. This is shown in Figure 18 which shows a view of the s/c, as seen by an observer in the northern hemisphere of the Earth, from slightly before until slightly after the Solar Oberth, the horizontal and vertical axes are in units of solar radius (696,342 km).

What is certain is that things will get hot, with a solar flux at perihelion of 1850 times that of the solar flux incident on Earth. The latter solar flux is 1.37 kW/m$^2$ (on the order of a tenth of the heat flux of your average

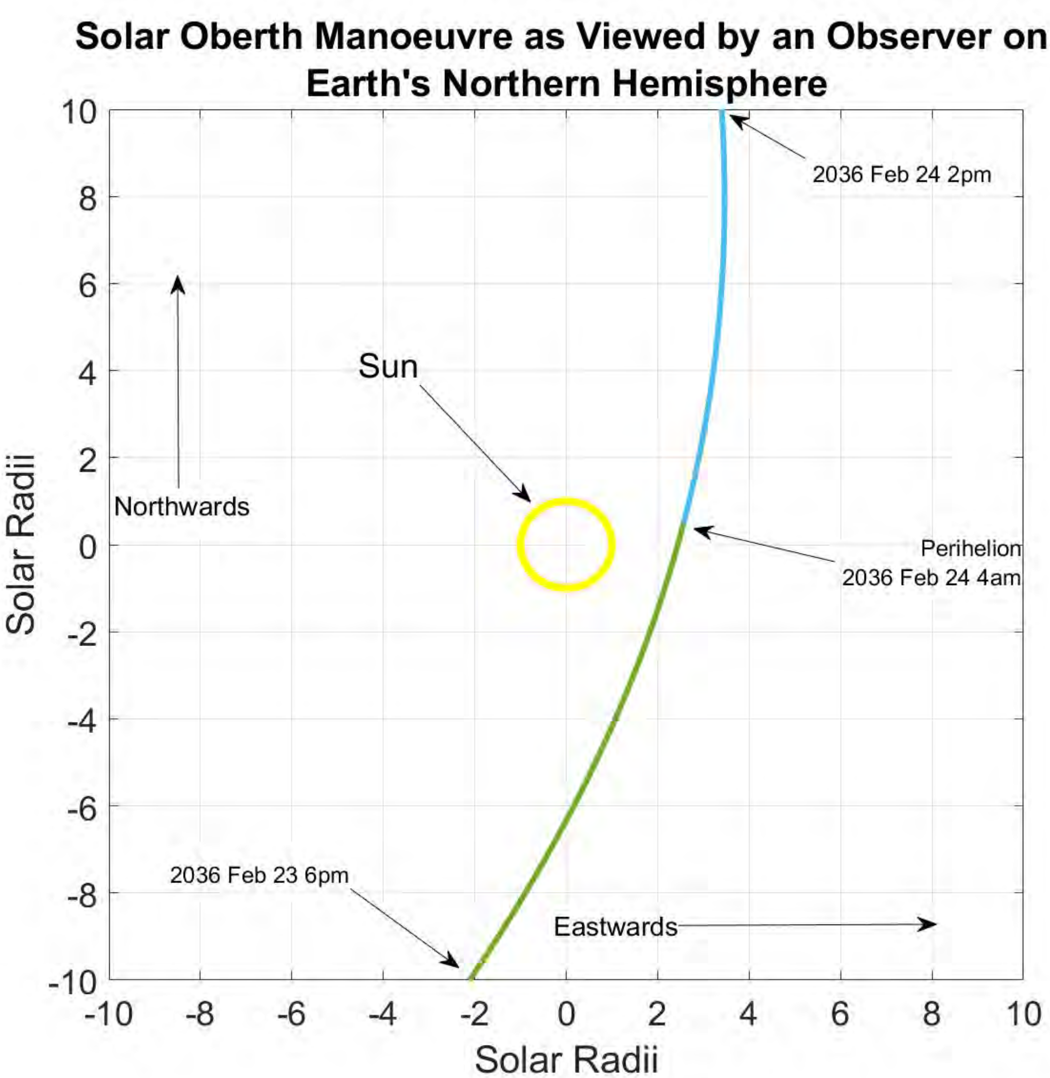


Figure 18: The Solar Oberth as viewed from Earth.

toaster) giving a value at the Solar Oberth of 2.534 MW/m². This enormous level of solar flux necessitates a heat shield to protect the precious payload. A study into Solar Oberth manoeuvres with Solar Thermal Propulsion – STP - (but aiming for the interstellar medium rather than specifically 'Oumuamua) has been conducted and is provided in Sauder et al *System Engineering a Solar Thermal Propulsion Mission Concept for Rapid Interstellar Medium Access* (2020), Ref 7. For our purposes, the long stretch to 'Oumuamua will indeed take us into the pristine interstellar medium, and we also need to get there fast. There is some mention in the Sauder et al study of a heat shield which is composed mainly of the same carbon-carbon composite material utilised for the Solar Parker Probe. However it also mentions there is an issue with this in that this material alone would quickly heat up to high temperatures when close and exposed to the sun. The solution is to coat this C-C composite with a thin layer of something highly reflective like Barium Fluoride, which also has an almost ideal low absorption profile in the most intense part of the solar spectrum.

A ΔV kick of around 7.2 km/s is required at the Solar Oberth and the original Project Lyra paper went into some detail as to how a similar kick could be delivered using, not STP (which has a relatively low Technical Readiness Level) but by the tried and trusted solid propellant rocket. That paper however only dealt with the 2021 mission which is now practically speaking infeasible due to time constraints.

For Trajectory A, which is considered here (and explored in the second Project Lyra paper), there is an issue in that one single stage solid rocket would not be able to deliver such a kick, so we ideally need two stages (or even three). There are plenty of candidates for powerful solid rockets, including the Thiokol STAR 48B, 63F and 75, in order of increasing ΔV kick. For our Solar Oberth burn we could try some combination of these for our two stage solid rocket boost, or even better, why not calculate the optimal ratio of stage masses and construct two bespoke solid rockets with this ratio of masses especially for the mission? I have done some analysis into all of this and Table 4 and Figure 19 are the results. Referring to Table 4, the liquid propellant motor present is there to apply all the ΔVs on the s/c from encounters 2 to 4 in Table 1. Note it is assumed the popular combination of hydrazine and nitrogen tetroxide is used, but this is likely to be phased out in the future, due to health and environmental concerns, and there are plenty of alternatives. Column 7, the theoretical minimum exhaust velocity needed for the liquid propellant motor is a calculation of the Specific Impulse which would result if a perfect liquid rocket stage were used to apply all the ΔVs for the encounters 2 to 4, assuming all the spare capacity in the column 6 could be used as fuel.

Table 4: Payload Masses achievable to 'Oumuamua Using 2 Stages for the Solar Oberth and a NASA SLS.

| Payload to 'Oumuamua (including heat shield mass) | Traj | 1st Rocket Motor (Location) | 2nd Rocket Motor (Location) | Minimum SLS Config. Needed | Spare Capacity for Liquid Propellant Rocket Motor (LPRM) | Minimum Exhaust Velocity (Ve) Required for LPRM | Possible Liquid Propellant Oxidiser+Fuel (N2O4+MMH Has Ve = 3.347km/s) |
|---|---|---|---|---|---|---|---|
| 200kg | A | STAR 48Bs (SO) | STAR 48Bs (SO) | Block 1 | 5.5mt | 1.252km/s | N2O4+MMH |
| 427kg | A | STAR 63F (SO) | STAR 63F (SO) | Block 1B | 8.4mt | 1.591km/s | N2O4+MMH |
| 541kg | A | STAR 75 (SO) | STAR 48B (SO) | Block 1B / 2 | 7.2mt / 12.2mt | 1.958km/s / 1.323km/s | N2O4+MMH/ N2O4+MMH |
| 639kg | A | STAR 75 (SO) | STAR 75 (SO) | Block 2 | 6.2mt | 3.184km/s | N2O4+MMH |
| 640kg | A | STAR 75 (SO) | STAR 63F (SO) | Block 1B / 2 | 4.7mt / 9.7mt | 3.305km/s / 1.826km/s | N2O4+MMH?/ N2O4+MMH |
| 711kg | B | STAR 63F extended (PJ) | STAR 75 (SO) | Block 2 | 9.1mt | 2.581km/s | N2O4+MMH |
| 296kg (3 Stages: 378kg) | A | Liquid Propellant Used | Liquid Propellant Used | Block 1 | N/A | N/A | N2O4+MMH |
| 532kg (3 Stages: 681kg) | A | Liquid Propellant Used | Liquid Propellant Used | Block 1B | N/A | N/A | N2O4+MMH |
| 680kg (3 Stages: 870kg) | A | Liquid Propellant Used | Liquid Propellant Used | Block 2 | N/A | N/A | N2O4+MMH |

SO = Solar Oberth PJ = Perijove

For Table 4 and Figure 19, the Block 1 data assumed is based on an older design spec for the SLS. Since that time the power of Block 1 has evolved and become much closer to what was then known as the Block 1B variant. The vertical lines show the total payload capacity for each of the SLS versions Block 1, Block 1B and Block 2. The stars indicate what mass payload can be delivered to ‘Oumuamua (the vertical axis) against the total mass combined of the payload and solid booster stages 1 & 2 just prior to the Solar Oberth. The available mass for the liquid propellant stage (for encounters 2 to 4) is therefore the horizontal difference between a star and the vertical bar of the SLS version of interest. The diagonal line represents the capability for an optimal ratio of Solar Oberth stage masses. Masses of 711 kg are achievable to ‘Oumuamua by a Block 1B and the old Block 1 could easily have delivered masses of 200 kg.

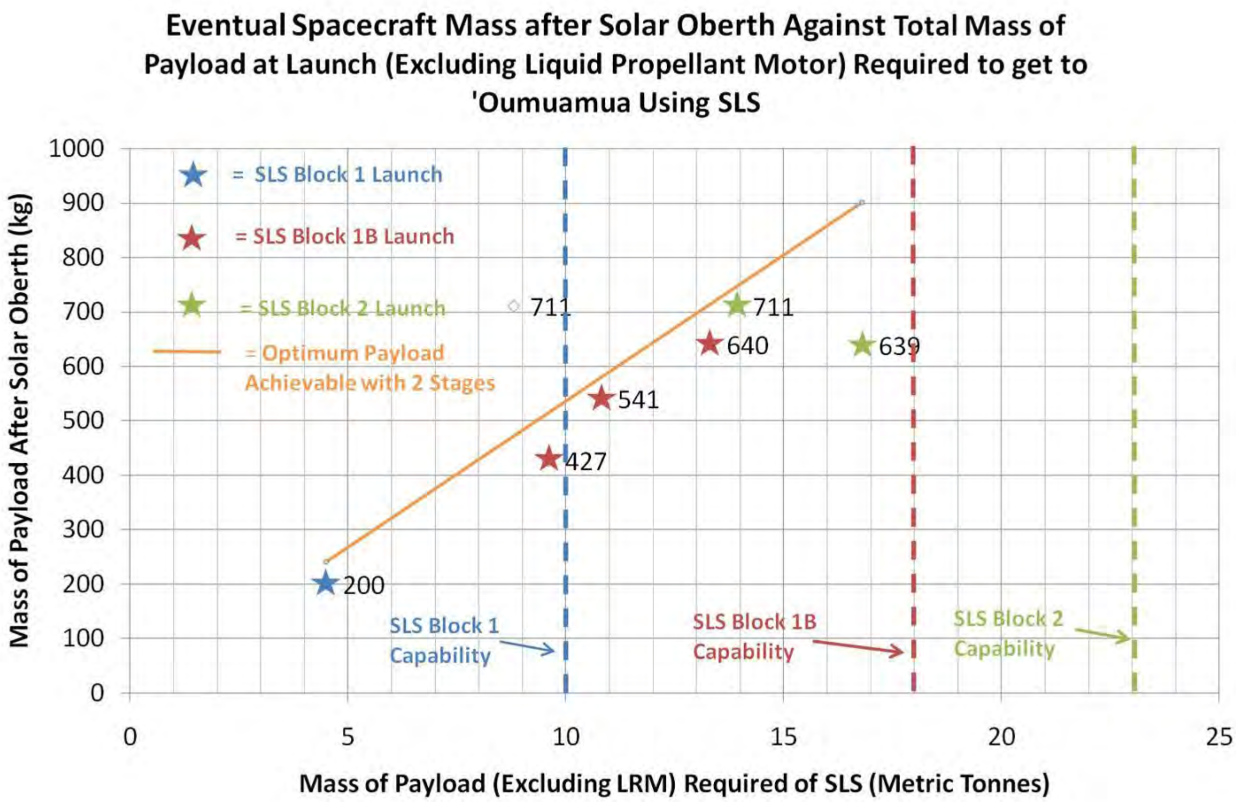


Figure 19:: Graphic of Possible Payload Masses to 'Oumuamua Using the NASA SLS..

## 5) Coast to ‘Oumuamua and Arrival

Nominally it will take 16 years after the Solar Oberth to get to ‘Oumuamua making the overall mission duration 22 years (arriving in 2052). This is a long wait and begs the question as to whether we can reduce this duration. The straightforward answer would be yes if we were to use an alternative to chemical propulsion, for example the emerging NTP, Nuclear Thermal Propulsion.  But this, like STP, has a low TRL* and hasn’t actually been flight tested yet - so let us stick to chemical for the moment.

* Technology Readiness Level https://www.nasa.gov/directorates/heo/scan/engineering/technology/txt_accordion1.html

What happens to the size of ΔV needed at the Solar Oberth as we reduce the flight duration from the Solar Oberth to ‘Oumuamua? Figure 20 shows clearly that as this duration is reduced, so the ΔV increases, moving from right to left in the plot. On the far right of this curve, it is equivalent to trajectory A, ie a 16 or so year flight and a ΔV impulse of 7.2 km/s. As we move to the left, so the flight duration reduces and as one would expect, the impulse goes up. Initially the rise is quite gradual but if we try to shorten the duration too much, say by 3 years or more, we start to enter infeasible territory.

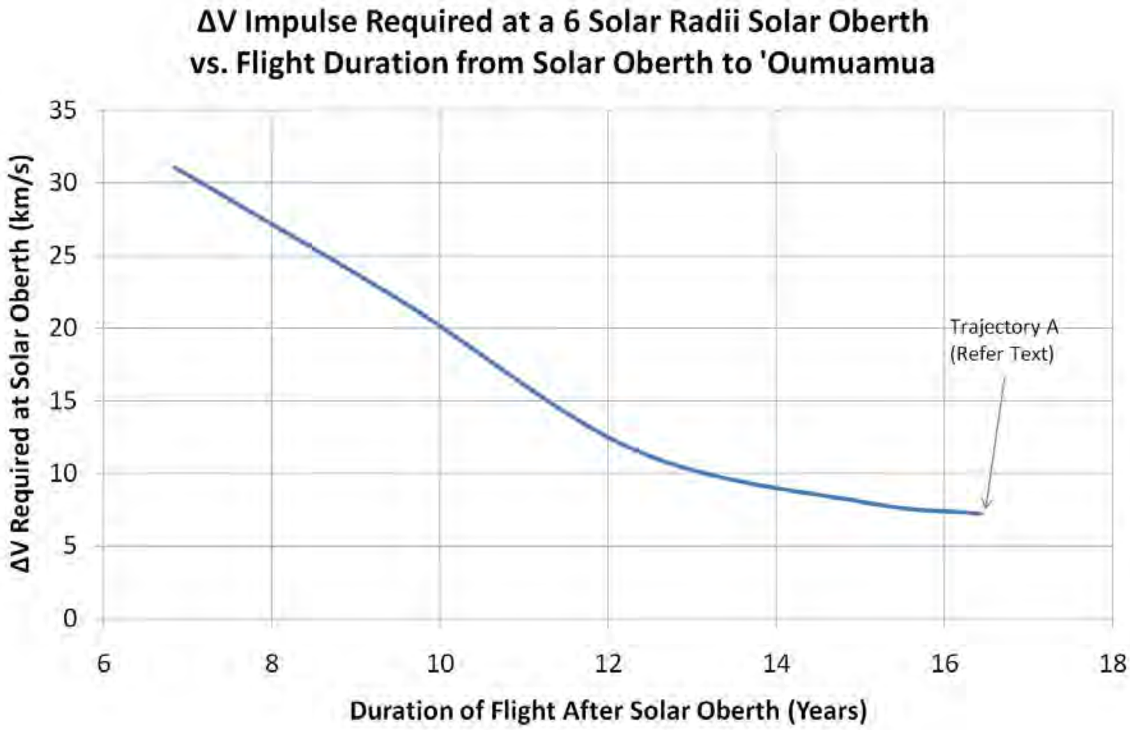


Figure 20: How ΔV at the Solar Oberth Varies with Flight Duration after the Solar Oberth.

There is an issue with arriving at ‘Oumuamua in that there is a fair degree of uncertainty in its precise direction – the asymptote direction - which it is heading in relative to the sun. There are two prime reasons for this. Firstly ‘Oumuamua was detected fairly late as it encountered the inner solar system – after perihelion in fact – and as a consequence the number of observations of it from ground telescopes and the Hubble space telescope was relatively low. Secondly when ‘Oumuamua’s trajectory was analysed, a non-gravitational force was discovered to be present, influencing and perturbing its motion Micheli et al (Ref 4). Both of these factors mean we don’t have an accurate fix on the orbit of ‘Oumuamua. One can do the calculations and we find that for an intercept distance of 200 AU, ‘Oumuamua could be laterally displaced by as much as 1 or 2 million km from the best estimate of its orbital path.

This seems to be a real issue for any single spacecraft. The solutions suggested in the i4is Interstellar Now paper (Ref 8) are -

- An onboard telescope should be able to detect an object from within a range of around (1/R) AU where R is the distance from the sun in AU. We know that R=200 AU, so we find this distance turns out at around 750,000 km. So a single spacecraft could quite easily pass ‘Oumuamua by without even noticing it!
- The other solution elaborated in the Interstellar Now paper is to send a single spacecraft which, at a significant distance from ‘Oumuamua’s predicted position, deploys a swarm of chipsats in a random scatter around ‘Oumuamua’s estimated asymptote direction. With a sufficient number of chipsats, statistically at least one should detect ‘Oumuamua and be able to communicate back to the mother craft ‘Oumuamua’s precise position and velocity. It should then be a simple question of the mother craft applying a corresponding adjustment to its velocity vector to ensure an intercept.

Clearly it would be helpful if we had solutions to this terminal guidance issue which were not so demanding upon telescopic pattern matching or as-yet unproven chipsat technology.

The next issue of Principium will examine terminal guidance in detail.

In future we will also consider the science payload possible with a number of probe scenarios - flyby, impactor and rendezvous - and configurations including single, multiple and swarm spacecraft.

## 6) Final

You may have noticed the entire mission duration is extremely extended – over 20 years. Furthermore this mission would stretch the limits of current human scientific and technological understanding. Is the scientific prize worth all this? I guess I end this whole analysis with a call to arms not just to scientists and engineers, but to the whole of humanity.

Just as a young software engineer should regularly be asked by their supervisor, "where do you see yourself in the future? What are your goals?" Should not humanity be asked the same question? However in our case there is no supervisor, there is no overseer to steer us. Is it not therefore incumbent upon us collectively to ask ourselves this question? When an opportunity arises, surely it would be folly not to grasp it and seek to climb the ladder of promotion. And who knows where interstellar travel may bring us? To a Kardashev Type III species maybe? Such grand ambitions start with small steps and as luck would have it an opportunity has indeed arisen, let us all now unite in this common goal. Objective: 'Oumuamua.

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**About The Author**

Adam was educated at a state school in Coventry and attended the University of Keele, gaining a joint honours degree in physics and maths. He worked in the '90s as a software engineer on the on-board flight program for the European Ariane 4 launch vehicle - including the production, maintenance, real-time testing and post-flight analysis, his expertise being the guidance algorithm. He is also a pianist and composer and, as a member of musical trio 'Superheroes Dream', produced a vinyl under the Coventry Tin Angel Record Label (tinangelrecords.bandcamp.com/album/waiting-or-flying). He developed his Optimum Interplanetary Trajectory Software, 'OITS' in 2017 as a personal challenge to learn the MATLAB programming environment and language. He then used it to investigate missions to interstellar objects (the work being published in Acta Astronautica) and now is a research volunteer for the 'Initiative for Interstellar Studies'.