Name: Joost Hubbard

Student ID: 210372773

Module Code: EMS610

Module Title: Spacecraft Design

Coursework Name: Interplanetary Orbit Transfers

Table of Contents

1. Introduction

Through the pursuit of comets and other stellar objects, invaluable advancements can be made in understanding the birth of our solar system. Frozen ice on a comets surface can often be dated back to the origin of the universe – providing insight into the chemical composition and conditions present during the early stages of solar system format comets and other stellar objects, invaluable advancements can be made in understanding ion. comets and other stellar objects, invaluable advancements can be made in understanding Additionally, studying the trajectories and behaviour of comets can offer clues about the dynamics of the early solar system and the processes that shaped its evolution over billions of years (Snodgrass, 2019).

Missions such as the European Space Agency's Comet Interceptor are positioned to provide insights at exceedingly short notice - a feat achieved through meticulous mission planning and prediction. By positioning a satellite in a halo around the Lagrange Point 2 (L2) point, it is ready to conduct interception manoeuvres with any incoming comets as they pass through the elliptic plane. This is particularly interesting due to the minimal station keeping requirement around L2 – allowing the satellite to remain in an operational orbit almost indefinitely.

To achieve this, three major steps were taken as detailed below and reflected in figure 1:

1. Launch the spacecraft from Earth and position it at L2, where it remains stationed until a suitable comet is identified.
2. When a comet is discovered, the satellite transitions to an orbit which intercepts with the comet on the elliptical plane – cruising on that orbit until optimum position is reached.
3. Execute the intercept, ensuring time constraints are accounted for, to allow for the gathering of data.

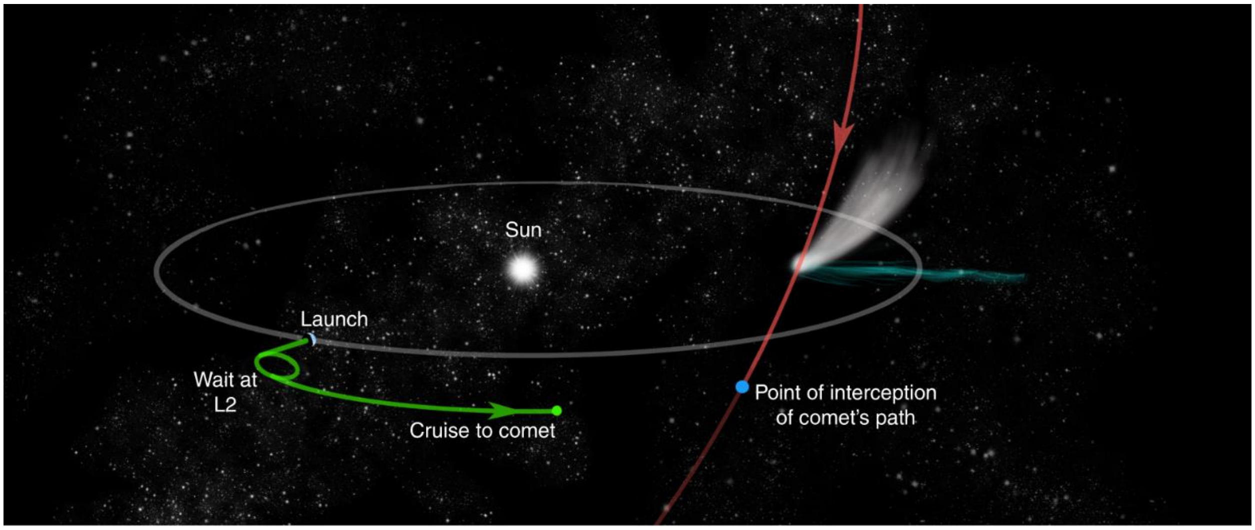


Figure 1 - Sketch of mission phases. The trajectory of the spacecraft is in green, while the comet's orbit is in red (Snodgrass, 2019).

1.1 Report Specification

In the case of this report, a similar mission will be designed and optimised – proposing the intercept of a comet in an inclined, elliptical, heliocentric orbit. The given parameters for this assignment can be found below in table 1.

Table 1 - Parameters given for calculation.

Basic Data	Universal Gravitational Unit	6.674E-11	Nm^2/kg^2
	1 Astronomical Unit	149597870	km
	Equatorial Radius of Sun	693634	km
	µsun	1.327E+11	km^3/s^2
	µearth	398600	km^3/s^2
	Earth Mass	5.97E+24	kg
	Earth Orbit Radius	1	Au
	Sun Mass	1.99E+30	kg
Spacecraft Properties	Eccentricity	0	-
	Inclination to Ecliptic	0	°
	Initial Orbit Radius	8242.5675	km
Comet Properties	Semi Major Axis	3.464737	Au
	Eccentricity	0.6405847	-
	Inclination	7.0436987	°
	Argument of Perihelion	12.694464	°
	Mass	1.1E+12	kg

2. Questions 1-5

The theory section of this report outlines the equations and background used for calculation. Each section includes a detailed explanation and references where appropriate. Answers are highlighted in green.

2.1 Part 1

'*Draw a new diagram of the Comet's orbit indicating clearly where this orbit intersects with the ecliptic plane, the two distances (i.e. as Comet approaches the Sun and then where it is moving away from the sun) from the Sun at these points and any other relevant or important features. Calculate the Earth-L2 and the L2-Comet distances at both the pre-perihelion and post-perihelion points of interception*.'

Prior to completing a new diagram, the requested distances must be calculated. These distances are those between Earth and L2 as well as the comet and L2 (at both pre and post Helion points of interception). These points of interception are also referred to as the descending and ascending nodes respectively.

2.1.1 Earth – L2 Distance

To find the distance between the Earth and the L2 point, we must first find the Earth's orbital period (T).

$$T_{Earth} = (365.24 days)\left(24\frac{hr}{day}\right)\left(60\frac{min}{hr}\right)\left(60\frac{secs}{min}\right) = 31{,}556{,}736\ seconds \quad [1.1]$$

The force between two bodies can be found by relating the universal gravitational constant (G), distance between the centre of the two bodies (r) and the masses of the two objects (m_1 and m_2).

$$F = \frac{Gm_1m_2}{r^2} \ [1.2]$$

In the case of the L2 point, both the gravitational effect of the Sun and Earth have influence. This gives the equation for the force acting on an object at L2 as a balance between the Earth and the Sun. This new equation makes use of equation [1.2] and takes into account the mass of the object at the L2 point (m_{object}). The distances between the Earth and L2 ($r_{Earth-L2}$) as well as between the Earth and the Sun ($r_{Sun-Earth}$) are also considered.

$$F_{object} = \frac{Gm_{sun}m_{object}}{(r_{Sun-Earth}+r_{Earth-L2})^2} + \frac{Gm_{earth}m_{object}}{{r_{Earth-L2}}^2} \ [1.3]$$

The force on the object can also be represented as a relationship between mass and acceleration – as per newtons second law. The centripetal acceleration of the satellite can be derived as a relationship between the period of the L2 point and the radius of the L2 orbit. It should be noted that the L2 has the same orbital period as the Earth (AURA's Space Telescope Science Institute, 2021) and as such T is that from equation 1.1.

$$F_{object} = m_{object} \times a_{object} \ [1.4] \qquad a_c = \left(\frac{2\pi}{T}\right)^2 (r_{Sun-Earth} + r_{Earth-L2}) \ [1.5]$$

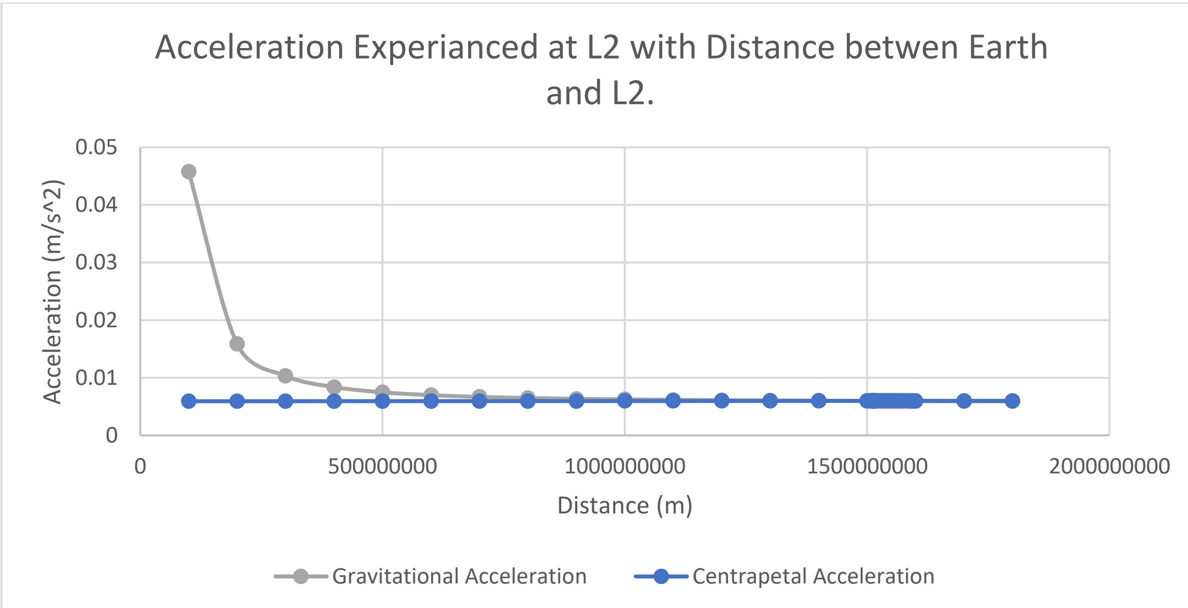
The previous equations, 1.3, 1.4 and 1.5 can be combined to derive a new relationship associated with the L2 point. The mass of the object (m_{object}) can be cancelled since it is on both sides of the equation.

$$\frac{Gm_{sun}}{(r_{Sun-Earth} + r_{Earth-L2})^2} + \frac{Gm_{earth}}{r_{earth-L2}^2} = \left(\frac{2\pi}{T}\right)^2 (r_{Sun-Earth} + r_{Earth-L2}) \ [1.6]$$

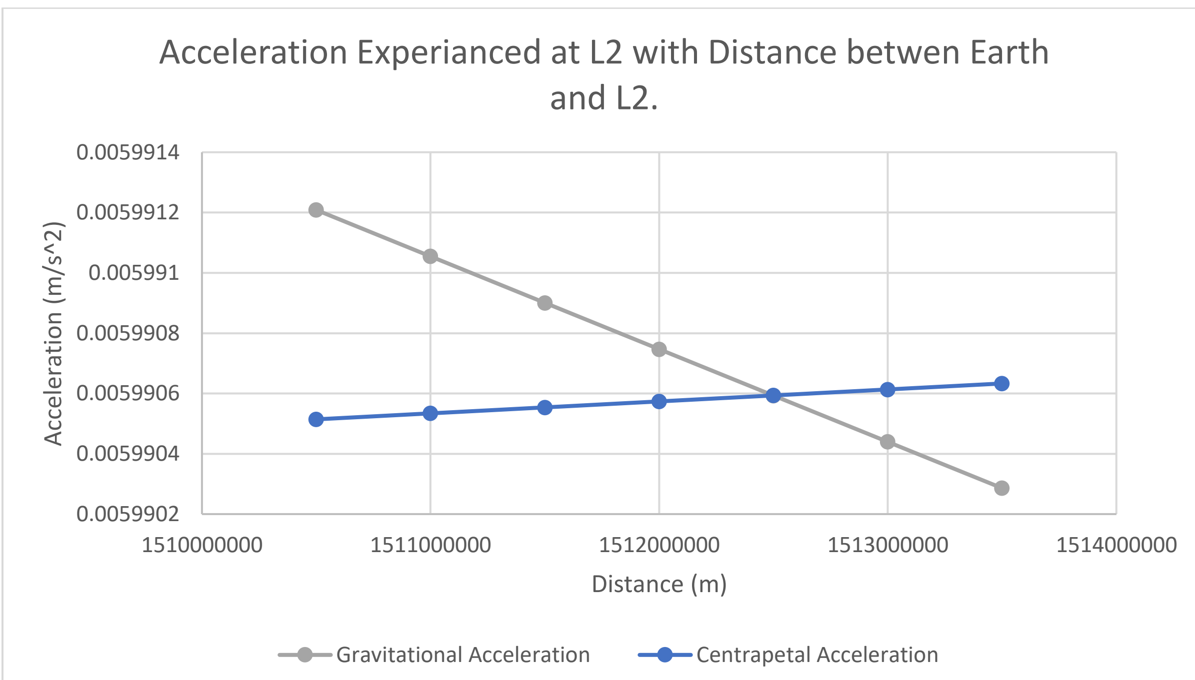
Substituting for known values gives an equation in terms of $r_{Earth-L2}$ that can be solved.

$$\frac{(6.67408 \times 10^{-11}\mathrm{Nm^2/kg^2})(1.99 \times 10^{30}kg)}{(r_{earth-L2} + 1.49598x10^{11}m)^2} + \frac{(6.67408 \times 10^{-11}\mathrm{Nm^2/kg^2})(5.97 \times 10^{24}kg)}{r_{earth-L2}^2}$$
$$= \left(\frac{2\pi}{31{,}556{,}736secs}\right)^2 (r_{earth-L2} + 1.49598 \times 10^{11}m)$$

To solve the equation for $r_{Earth-L2}$ a graphical approach was used. This involved plotting the gravitational and centripetal acceleration experienced at the L2 point – with the intercept determining the distance between Earth and the L2 point. From this, $r_{Earth-L2}$ was determined to lie 1512500 km or 0.01 AU away from Earth. This represents an 0.83% error compared to externally given figures of 1500000 km (Vepa, 2024).



Graph 1 - Acceleration experienced at l2 with distance between earth and l2. Full graph.



Graph 2 - Acceleration experienced at l2 with distance between earth and l2. Focus on intercept.

2.1.2 Comet – L2 Distance

Next, the node distances were calculated. This was done using the properties of ellipses to relate the distance of the node from the Sun (r_{node}), the semi major axis (a), the eccentricity (e) and the augment of periapsis (ω).

$$r_{node} = a(1-e^2)\left(\frac{1}{ecos\omega+1}\right) \text{ [1.7]}$$

Substituting known values into equation 1.7 gives the following equations and results for the descending ($r_{decending\ node}$) and ascending ($r_{acending\ node}$) node distances. It should be noted that all angles have been translated into radians for ease of calculation.

$$r_{decending\ node} = 518317278\ km \times (1 - 0.640584737^2)\left(\frac{1}{0.640584737 \times cos(\ \pi + 0.221560194\ rads) + 1}\right) = 814843255.2 \text{ km or } 5.45 \text{ AU}$$

$$r_{acending\ node} = 518317278\ km \times (1 - 0.640584737^2)\left(\frac{1}{0.640584737 \times cos(\ 0.221560194\ rads) + 1}\right) = 188086334.5 \text{ km or } 1.26 \text{ AU}$$

Subsequently, the distance between both nodes and the L2 point can be calculated. First, the distance between the Sun and the L2 point.

$$r_{Sun-L2} = r_{Earth-L2} + r_{Sun-Earth} \text{ [1.8]}$$

Substituting given values, it can be found:

$$1512500\ km + \ 149597870\ km = 151110370\ km$$

Using this value, the comet node to L2 distances can be determined.

$$r_{comet-L2} = r_{node} - r_{Sun-L2} \text{ [1.9]}$$

Substituting given values, it can be found:

$$r_{comet\ -\ decending\ node} = \ 814843255.2\ km - 151110370\ km = 663732885.2 \text{ km or } 4.436780318 \text{ AU}$$

$$r_{comet-\ acending\ node} = \ 188086334.5\ km - 151110370\ km = 36975964.54 \text{ km or } 0.247169058 \text{ AU}$$

2.1.3 New Diagram

Finally, a new diagram of the comet's orbit is created, highlighting important features such as:

- The inclination of the comet's orbit (i),
- The argument of periapsis (ω),
- The initial orbit of the satellite (r),
- The apoapsis of the comets orbit,
- The periapsis of the comets orbit,
- The semi major axis (a),
- The distance to the descending node of the comets orbit (b),

- The distance to the ascending node of the comets orbit (c),
- The distance between Earth and L2 (d).

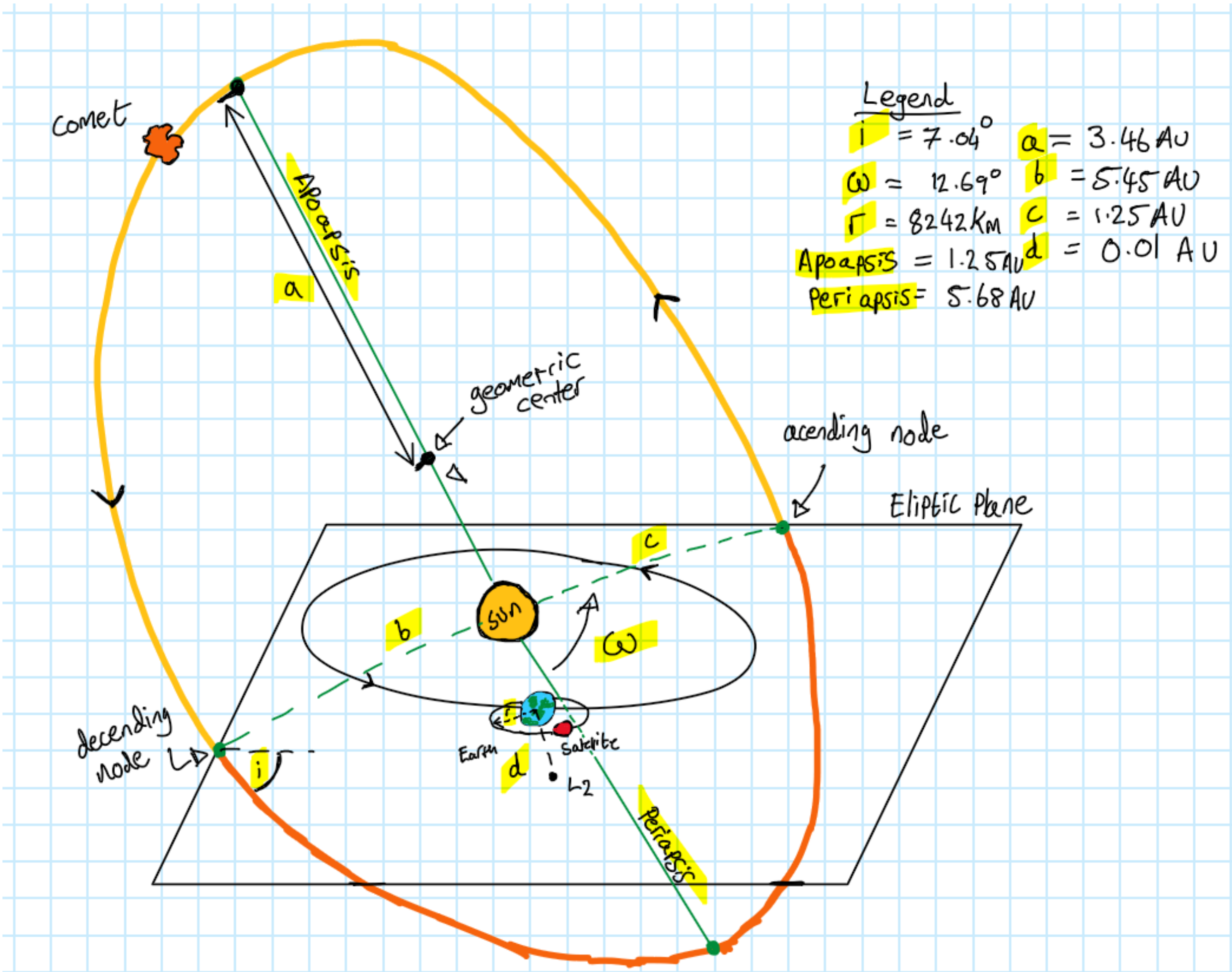


Figure 2 - New diagram of the comet's orbit.

2.2 Part 2

'*Calculate the minimum values of ΔV (in km/s) that would be required to take the satellite from its parking orbit about the Earth to the L2 location. Calculate the ΔV (magnitude and direction) that would be required to change the velocity of the spacecraft as it intersects the L2 orbit, so that it could then fly alongside the L2 in the same orbit as the L2. You should explain in words as well as writing down the equations, what manoeuvre would be performed and whether there are any options that could affect the magnitude of the ΔV.*'

To solve this problem, there are two stages of transfer that must be completed. First, the satellite must escape from earths sphere of influence, only then being able to transfer into a heliocentric orbit alongside L2. The escape stage can be completed using a hyperbolic orbit while the second must be a prograde Hohmann transfer. The satellites orbit will increase at each stage of this manoeuvre.

2.2.1 Heliocentric Hohmann Transfer from Earth to L2

For ease of calculation, the Hohmann transfer to the L2 point is completed first. To do this, the velocity at both the initial and final orbit is calculated using equation [2.1]. This makes use of a relationship between standard gravitational perimeter of a central body (μ) and the radii of an objects orbit around said body (r).

$$v = \sqrt{\frac{\mu}{r}} \text{ [2.1]}$$

Taking the initial and final orbits to that of Earth and L2 respectively, the velocity can be found by substituting in given values. It should be noted that the

$$v_{Earth} = \sqrt{\frac{1.32747\times10^{11}\ km^3/s^2}{149597870\ km}} = 29.78862043 \text{ km/s}$$

$$v_{L2} = \sqrt{\frac{1.32747\times10^{11}\ km^3/s^2}{151110370\ km}} = 29.63916477 \text{ km/s}$$

Next, the velocities at the perigee and apogee of the transfer orbit must be determined. This can be completed utilising a relationship between the specific angular momentum of the transfer orbit (h_t) alongside both the perigee and apogee radii (r_a and r_p). The perigee and apogee radii can be taken to equal to the initial and final radii respectively.

$$r_p = r_{Sun-Earth} = 149597870 \text{ km} \qquad r_a = r_{Sun-L2} = 151110370 \text{ km}$$

The specific angular momentum of the transfer orbit is given by equation 2.2.

$$h_t = \sqrt{\frac{2\mu r_a r_p}{r_a+r_p}} \text{ [2.2]}$$

Substituting values gives the angular momentum of the transfer orbit.

$$h_t = \sqrt{\frac{2\times1.32747\times10^{11}\ km^3/s^2\times 151110370 \text{ km}\times149597870 \text{ km}}{151110370 \text{ km}+149597870 \text{ km}}} = 4467507276\ km^2/s$$

The perigee and apogee velocities (v_p and v_a) are then found using equation 2.3.

$$v = \frac{h_t}{r} \text{ [2.3]}$$

Substituting in the previously calculated values, the velocity at the apogee and perigee of the transfer obit can be calculated.

$$v_p = \frac{4467507276\ km^2/s}{149597870 \text{ km}} = 29.86344175 \text{ km/s}$$

$$v_a = \frac{4467507276\ km^2/s}{151110370 \text{ km}} = 29.56453139 \text{ km/s}$$

To compute delta-v (Δv) values for the Hohmann transfer, one calculates the velocity disparities between the initial orbit and the perigee, and between the final orbit and the apogee. Adding these differences together provides the total delta-v.

$$\Delta v_{inital-\ transfer} = \left|v_p - v_{inital}\right| \text{ [2.4]}$$

$$\Delta v_{transfer-final} = \left|v_a - v_{final}\right| \text{ [2.5]}$$

$$\Delta v_{Hohmann} = \left|v_{inital-transfer} + v_{transfer-final}\right| \text{ [2.6]}$$

Using given and calculated values, equations 2.4, 2.5 and 2.6 can be utilised.

$$\Delta v_{Earth-\ transfer} = |29.86344175\ km/s\ - 29.78862043 \text{ km/s}| = 0.074821321\ km/s$$

$$\Delta v_{transfer-L2} = |29.56453139 \text{ km/s} - 29.63916477 \text{ km/s}| = 0.074633388\ km/s$$

$$\Delta v_{Hohmann} = |0.074821321\ km/s\ + 0.074633388\ km/s| = 0.149454709\ km/s$$

2.2.2 Earth Escape

Prior to conducting the Hohmann transfer from Earth to the L2 point, the satellite must escape Earths sphere of influence (SOI). This departure is conducted using a hyperbolic orbit which must be designed and then transferred to from the satellite's parking orbit. Initially, the velocity of the parking orbit must be determined – making use of equation 2.1. It should be noted that in this case the standard gravitational perimeter of Earth is used.

$$v_{parking} = \sqrt{\frac{398600\ km^3/s^2}{8242.567481\ km}} = 6.954043316\ km/s$$

Next, the specific mechanical energy associated with the escape trajectory ($\varepsilon_{\infty Earth}$) can be determined. This can be found as a function of the escape velocity of the satellite ($v_{\infty Earth}$) with respect to the Earth. This value has already been established as $\Delta v_{Earth-\ transfer}$ and as such can be used for calculation.

$$\varepsilon_{\infty Earth} = \frac{{v_{\infty Earth}}^2}{2} + \frac{\mu_{Earth}}{r_{Earth\ SOI}} \text{ [2.7]}$$

It should be noted that since the sphere of influence radius of Earth ($r_{Earth\ SOI}$) is orders of magnitude larger than the standard gravitational parameter of Earth (μ_{Earth}), the second term in equation 2.7 can be taken to be zero. Substituting in given values gives the following result:

$$\varepsilon_{\infty Earth} = \frac{0.074821321\ km/s^{\ 2}}{2} = 0.002799115\ \ km^2/s^2$$

Using this, the velocity of the hyperbolic orbit can be found.

$$v_{Hyperbolic} = \sqrt{2(\frac{\mu_{Earth}}{r_{parking}} + \varepsilon_{\infty Earth})} \text{ [2.8]}$$

Substituting in associated values gives the following equation:

$$v_{Hyperbolic} = \sqrt{2(\frac{398600\ km^3/s^2}{8242.567481\ km} + 0.002799115\ \ km^2/s^2)} = 9.834786988\ km/s$$

Similar to how the Hohmann transfer delta v is calculated, the escape delta v (Δv_{escape}) is found as the velocity difference between the initial and hyperbolic orbits.

$$\Delta v_{escape} = v_{Hyperbolic} - v_{parking} \text{ [2.9]}$$

$$\Delta v_{escape} = 9.834786988\ km/s - 6.954043316\ km/s = 2.880743672\ km/s$$

2.2.3 Total Delta V

The concluding step involves determining the total delta v required for the satellite's transition from its parking orbit to the L2 point ($\Delta v_{Earth-L2}$). This entails combining the delta v values obtained from the preceding two sections: the escape manoeuvre and the subsequent Hohmann transfer.

$$\Delta v_{Earth-L2} = \Delta v_{Hohmann} + \Delta v_{escape} \text{ [2.10]}$$

Substituting in associated values gives the following equation:

$$\Delta v_{Earth-L2} = 0.149454709\ km/s\ + 2.880743672\ km/s = \ 3.030198382\ km/s$$

2.3 Part 3

'*Calculate the ΔV (magnitude and direction) that would be required to change the velocity of the spacecraft as it intersects the cometary orbit. Again, you should explain in words as well as writing down the equations, what manoeuvre would be performed and whether there are any options that could affect the magnitude of the ΔV.*'

To complete part three, a transition from the L2 point to both the ascending and descending nodes must be completed. This is completed using a prograde Hohmann transfer where the initial orbit is that of L2 and the final is a helio-centric orbit at the same altitude as the relevant node. In both cases the transfer increases the altitude of the satellite.

2.3.1 Transfer to Descending Node

First, the velocity at the orbit associated with the descending node must be calculated using equation 2.1. It should be noted that the standard gravitational perimeter (μ) used is that of the sun.

$$v_{decending\ node} = \sqrt{\frac{1.32747 \times 10^{11}\ km^3/s^2}{814843255.2\ km}} = 12.76368443\ km/s$$

Following this, the angular momentum of the transfer orbit can be calculated from the equation 2.2. In this case, the radius of the apogee (r_a) is that between the node and the sun. The radius of perigee (r_p) is that of the L2 orbit.

$$h_t = \sqrt{\frac{2 \times 1.32747 \times 10^{11}\ km^3/s^2 \times\ 814843255.2 \text{ km} \times 151110370 \text{ km}}{814843255.2 \text{ km} + 151110370 \text{ km}}} = 5817469662\ km^2/s$$

Using this value, the velocity of the apogee and perigee of the transfer orbit can be determined from equation 2.3.

$$v_p = \frac{5817469662\ km^2/s}{151110370 \text{ km}} = 38.49814981 \text{ km/s}$$

$$v_a = \frac{5817469662\ km^2/s}{814843255.2 \text{ km}} = 7.139372664 \text{ km/s}$$

Finally, the total delta v of the Hohmann transfer can be determined. This can be made more concise by combining equations 2.4, 2.5 and 2.6.

$$\Delta v_{L2-decending\ node} = |12.76368443\ km/s - 7.139372664\ km/s| + |29.63916477\ km/s - 38.49814981\ km/s| = 14.4832968\ km/s$$

2.3.2 Transfer to Ascending Node

As before, the velocity at the orbit associated with the descending node must be calculated using equation 2.1. It should be noted that the standard gravitational perimeter (μ) used is that of the sun.

$$v_{acending\ node} = \sqrt{\frac{1.32747 \times 10^{11}\ km^3/s^2}{188086334.5\ km}} = 26.56650808\ km/s$$

Following this, the angular momentum of the transfer orbit can be calculated from the equation 2.2. In this case, the radius of the apogee (r_a) is that between the node and the sun. The radius of perigee (r_p) is that of the L2 orbit.

$$h_t = \sqrt{\frac{2 \times 1.32747 \times 10^{11}\ km^3/s^2 \times\ 814843255.2 \text{ km} \times 151110370 \text{ km}}{188086334.5 \text{ km} + 151110370 \text{ km}}} = 4716589038\ km^2/s$$

Using this value, the velocity of the apogee and perigee of the transfer orbit can be determined from equation 2.3.

$$v_p = \frac{4716589038\ km^2/s}{151110370 \text{ km}} = 31.21287465 \text{ km/s}$$

$$v_a = \frac{4716589038\ km^2/s}{188086334.5 \text{ km}} = 25.07672367 \text{ km/s}$$

Finally, the total delta v of the Hohmann transfer can be determined. This can be made more concise by combining equations 2.4, 2.5 and 2.6.

$$\Delta v_{L2-acending\ node} = |26.56650808\ km/s - 25.07672367\ km/s| + |31.21287465\ km/s - 29.63916477\ km/s| = 3.063494286\ km/s$$

2.4 Part 4

'*Calculate how close your spacecraft will need to approach the comet in order for the spacecraft motion to be dominated by the comet rather than the being dominated by the gravitational field of the sun.*'

To determine the sphere of influence of the comet, a similar method to the determination of the position of the L2 point completed in section 2.1.1 was used. In this case, it is clear that the altitude at which the spacecraft will orbit the comet is heavily influenced by the comet's proximity to the sun. This can be demonstrated by existence of a distance term in equation 1.2. To account for this, the proximity required to orbit the comet will be calculated for both the ascending and descending nodes.

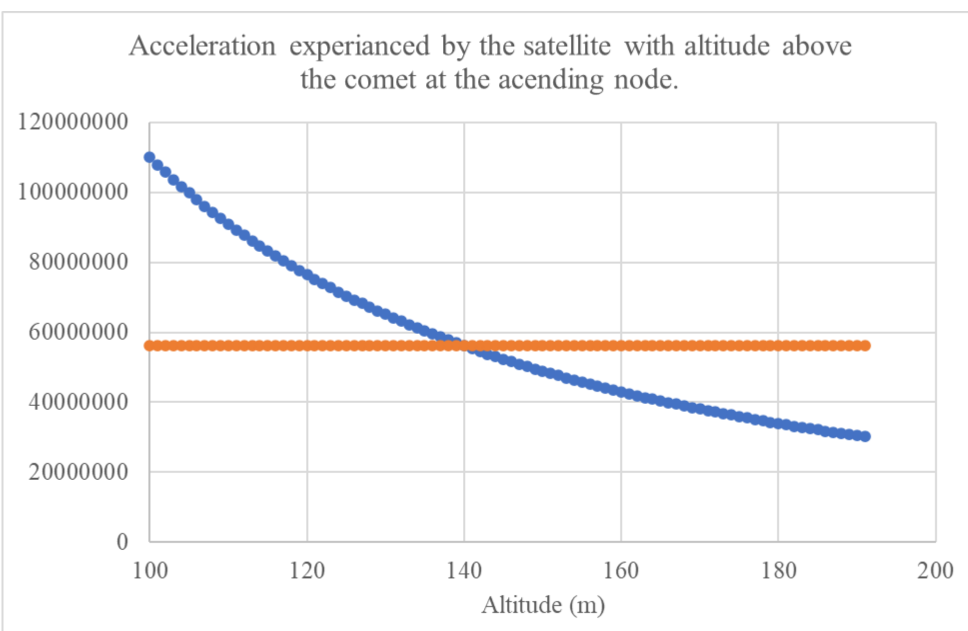
The gravitational acceleration equilibrium equation for the each node can be derived by equating the gravitational acceleration force of the comet and the Sun in terms of the radius of the satellites orbit around the sun (r_{target}). This is based on equation 1.2.

$$\frac{Gm_{sun}m_{sat}}{r_{Sun}^2} = \frac{Gm_{comet}m_{sat}}{r_{target}^2} \quad [4.1]$$

It is worth noting that $r_{sun}$is the distance between Sun and the comet minus the altitude of the satellites orbit r_{target}.

2.4.1 Ascending Node

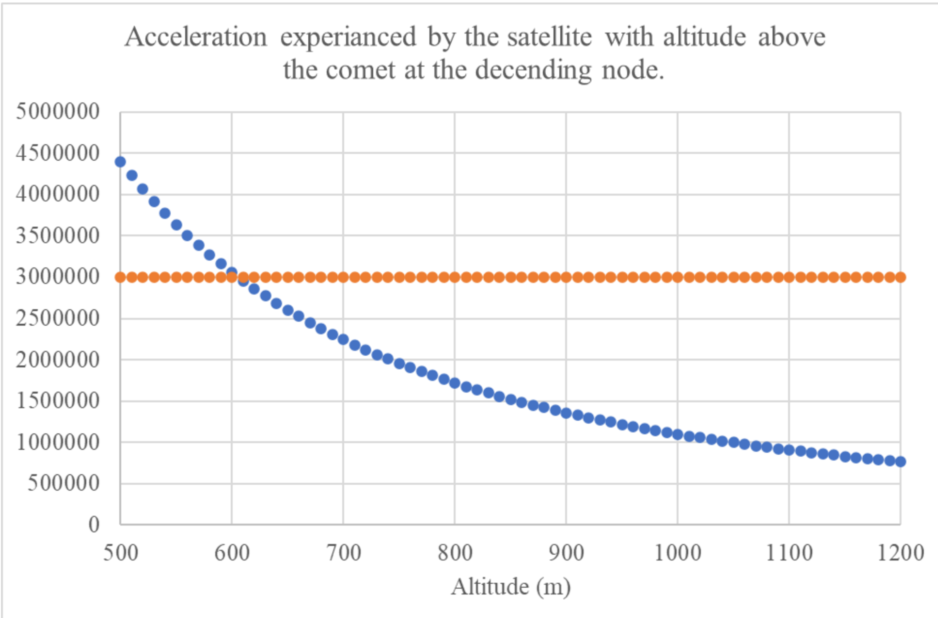
The gravitational acceleration equilibrium equation for the ascending node must be created and the solved for r_{target}. At the ascending node the spacecraft would need to fly-by the comet at less than 0.14 km. This is determined using the intercept using graph 3 and a comet altitude of 1.257279496 AU.



Graph 3 - Acceleration experienced by the satellite with altitude above the comet at the ascending node.

2.4.2 Descending Node

At the descending node the spacecraft would need to fly-by the comet at less than 0.6 km. This is determined using the intercept on graph 4 and a comet altitude of 5.446890756 AU.



Graph 4 - Acceleration experienced by the satellite with altitude above the comet at the descending node.

2.5 Part 5

'*Consider the results obtained in section C and compare which mission profile results in the lowest overall ΔV in respect to whether your solution for section B provides this overall minimum ΔV for the mission. Comment upon these findings. Extend your discussion regarding the mission selection by consideration of other features not considered thus far including other mission profiles, supported by appropriate calculations in order to establish which of these may influence the final mission selection.*'

Across this report, two differing mission profiles have been examined for the intercept of the comets orbit. Within mission profile one the satellite begins in its parting orbit, escapes earths SOI and establishes a parking orbit alongside the L2 point. The satellite then transfers from the L2 point to the descending node for the comet fly-by. The second mission profile is the same for the first two steps but then commits to a comet fly by the alternate intercept point with the elliptical plane – the ascending node.

To compare the two profiles the total delta v must be compared. This can be done by totalling the sum of the delta v across both profiles. As stated, prior, mission profile one is associated with the descending node while profile two is the ascending.

$$\Delta v_{mission\ profile\ 1} = |3.030198382\ km/s + 14.4832968\ km/s\ | = 17.513495182\ km/s$$

$$\Delta v_{mission\ profile\ 2} = |3.030198382\ km/s + 3.063494286\ km/s\ \ | = 6.093692668\ km/s$$

$$\Delta v_{mission\ profile\ 1} > \Delta v_{mission\ profile\ 2}$$

From this it can be determined that the determined that the total delta v for first mission profile is significantly higher than that of mission profile two. This is to be expected since the ascending node is much closer to the L2 orbit's heliocentric altitude and as such has a lower velocity difference.

To reduce the delta v further, a third mission profile is proposed. This approach would remove the need for positioning at the L2 point prior to the comet flyby. This could be achieved if the comments intercept could be predicted well ahead of time and a spacecraft could be launched with a within a pre-determined intercept timeframe. In this case, the reduced station keeping costs at L2 would not be necessary as the spacecraft would not be in a parking orbit prior to intercept – instead it could immediately escape earth and head towards the proposed intercept.

Another possible optimisation could be the use of a bi-elliptic transfer in place of a Hohmann for the transfer from L2 to the nodes. When the target orbit to initial orbit radii-ratio is less than 15.58 but greater than approximately 11.94, the bi-elliptical transfer is more economical if the intermediate point is placed at a sufficiently high altitude (Silber, 1959).

3. References

1. AURA's Space Telescope Science Institute, 2021. *Webb's Orbit at Sun-Earth Lagrange Point 2 (L2).* [Online]
   Available at:
   https://webbtelescope.org/contents/media/images/01F4STZH25YJH07WTN7XJYQP8P#:~:text=L2%20is%20one%20of%20five,in%20constant%20communication%20with%20Earth.
   [Accessed 28 03 2024].

2. Silber, R. H. a. R., 1959. *The bi-elliptical transfer between circular co-planar orbits,* s.l.: Army Ballistic Missilles Agency.

3. Snodgrass, C. a. J. G. H., 2019. The European Space Agency's comet interceptor lies in wait. *Nature communications,* 10(1), p. 5418.

4. Vepa, D. R., 2024. *EMS610 Coursework: Interplanetary Orbit Transfers.* London : Queen Mary university of London.