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Master's Thesis

TFM TITLE: Asteroidal volatiles for the development of planetary outposts: the case of the Moon

MASTER: Màster Universitari en Enginyeria Aeronàutica

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SUPERVISOR: Joan Pau Sanchez Cuartielles

DATE: June 21, 2021

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Resumen

El descubrimiento de hielo lunar en 1998 abrió toda una serie de nuevas perspectivas para el aprovechamiento in situ de los recursos de agua en la Luna. Sin embargo, estudios más recientes sugieren que la cantidad de hielo lunar es escasa. Por otra parte, casi todas las agencias espaciales y un número considerable de iniciativas privadas abogan por la vuelta del humano a nuestro satélite natural y el establecimiento de una base lunar permanente. En este trabajo se llevará a cabo un estudio de viabilidad de la utilización de agua procedente de una fuente asteroidal en la órbita lunar, incluyendo la astrodinámica de la captura del asteroide y su inserción orbital, el enfoque de explotación y un análisis técnico de la utilización del agua asteroidal para el desarrollo lunar.
Primero, seleccionando y analizando el mejor asteroide candidato para la misión (asteroide 2010DL). Posteriormente se analizaran todos los parametros técnicos de la misión para acabar concluyendo que debido a los tipos de sistemas propulsivos actuales los tiempos de impulso que se requieren para acelerar o desacelerar la masa del asteroide 2010DL son muy superiores al tiempo de vuelo natural que tendrían las maniobras por lo que se concluye que con los sistemas propulsivos actuales no se puede desacelerar los suficientemente rápido un asteroide de 19 metros de diámetro de tipo C en su entrada a la orbita de captura de la tierra, para posteriormente proponer una solución que si que satisface tanto las limitaciones técnicas como las económicas, que es la extracción del agua directamente en la propia orbita del asteroide e ir enviando depósitos de 9500 litros de agua directamente desde el perigeo y el apogeo de la orbita del asteroide a una orbita distante retrograda de la luna.

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Overview

The discovery of lunar ice in 1998 opened a whole variety of new perspectives for in-situ resource utilization of water on the Moon. However, more recent studies suggest that the amount of lunar ice is scarce. On the other hand, almost every single space agency and a substantial number of private initiatives advocate for a human return to our natural satellite and establishement of a permanent outpost. In this work, a feasibility study of using water from an asteroidal source in lunar orbit will be carried out, including the astrodynamics of asteroid capture and orbital insertion, exploitation approach and a techno-economic analysis of asteroidal water utilization for lunar development.
First, selecting and analysing the best candidate asteroid for the mission (asteroid 2010DL). Subsequently, all the technical parameters of the mission will be analysed in order to conclude that due to the current propulsion systems, the impulse times required to accelerate or decelerate the mass of asteroid 2010DL are much longer than the natural flight time that the manoeuvres would have, so it is concluded that with the current propulsion systems it is not possible to decelerate an asteroid of 19 metres in diameter of type C fast enough on its entry into the Earth's capture orbit, It then proposes a solution that satisfies both the technical and economic constraints, which is to extract the water directly in the asteroid's own orbit and send tanks of 9500 litres of water directly from the perigee and apogee of the asteroid's orbit to a moon distant retrograde orbit.

Thank you to my advisor, Ignasi Casanova, for providing guidance and feedback throughout this project on everything related to space resources, asteroids and space mining.Thanks also to my supervisor Joan Pau Sanchez, for providing me with all the knowledge and advice necessary to develop all the astrodynamic analysis of this thesis.

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NOMENCLATURE

$A)$ **Abbreviations and Acronyms**

$CR3BP$ Circular Restricted 3 Body Problem

DRO Distant Retrograde Orbit

$JSC-1A$ Lunar regolith simulant

$MMRTG$: Multi-mission radioisotope thermoelectric generators

NEA Near-Earth Asteroid

NEO Near-Earth Object

NPV Net Present Value

PV Present Value

$PVEx$ Planetary Volatiles Extraction

ROI Return On Investment

SOI Sphere of influence

$B)$ **Symbols**

$Torr$ Unit of pressure based on an absolute scale $\approx [133.32Pa]$

$\dot{m}$ Mass flow

θ True anomaly

a Semi-major axis

ac Acceleration [m/s^2]

dt Time variation [s]

dV Velocity variation [m/s]

E Eccentric anomaly

e Eccentricity

F Force [N]

g_o Gravity of Earth 9.80665 [m/s^2]

I_{sp} Specific impulse [s]

M Mean anomaly

m Mass [Kg]

m_0 Initial mass [Kg]

Q Aphelion distance

q Perihelion distance

t Time

tof Time of flight [s]

v_e Exhaust velocity [m/s]

w Perigee argument

INTRODUCTION

As space exploration advances, more and more resources are required in space, the possibility of travelling to the moon again, but this time to establish a lunar base, requires an exhaustive search for possible sources of resources, and more specifically for water. Water is one of the most indispensable elements in space and is necessary for the existence of human life. Water can also be used as fuel for spacecraft and many other functions. However, the lack of water on the moon means that all the water that is to be sent there to start creating a lunar colony is transported directly from earth. This is a huge energy cost, since this water, in order to be taken from the earth, has to be carried by launchers that in order to leave the earth must burn many tons of fuel. However, there is another option that is starting to gain momentum these days, and this is obtaining water directly from asteroids, and specifically from C-type asteroids which are estimated to have up to 20% water content in the form of hydrated minerals. These asteroids account for 75% of all known asteroids, so there is a huge amalgam of opportunities. Therefore, in the first chapter of this paper, the different types of asteroids in the solar system will be introduced. In chapter two, a technical analysis of the mission will be made, first of the capture and orbital modification of the asteroid to exploit it directly in a stable orbit of the moon, and then proposing a new type of mission that fits better to the current technological constraints. In chapter three, a brief economic analysis will be made in which the costs of the mission will be analysed in terms of the cost of taking a litre of water from an asteroid to the moon compared to the cost of transporting a litre of water from the earth to the moon. And the current legal frameworks for space mining will be mentioned.

CHAPTER 1. STATE OF THE ART

1.1. Water Supply Chain in Space

Water (H_2O) is an ubiquitous molecule in the universe and very common in our Solar System. Three known sources for using water as a process fluid in Space manufacturing are the Moon, Asteroids and Earth [2]. This resource is one of the most important resources for space exploration as it is a critical element for the survival of human beings as well as having very versatile uses, from being used as fuel for spacecraft to being used in the construction of lunar bases, agriculture or even as radiation shields.

1.1.1. Uses of water

Water can be used for many purposes, of which some of the most important are:

1.1.1.1. Propellant

H_2O is composed by two hydrogen atoms for every oxygen atom. By applying electrolysis hydrogen and oxygen can be split. Once oxygen and hydrogen are separated they can be stored in tanks and burned together in a rocket engine to provide the needed thrust. If spacecrafts could stop at a lunar base to refuel, they no longer would need to bring all the propellant as they take off, this would make spacecrafts significantly lighter and cheaper to launch. This point is important since Earth's atmosphere and gravitational pull requires of tons of fuel during rocket launch that could be significantly reduced by creating a sustainable source of fuel in space.

1.1.1.2. Sustain human presence in space

Water is essential for life support in space since it is needed for drinking and it can be used for breathing by extracting oxygen from it. Nowadays drinkable water is fully recycled from the astronauts pee, therefore there is no a need of huge quantities of water for drinking. The same is not true for the oxygen needed to breathe. The largest cost of spaceflights is launching oxygen from the Earth, where the most of the oxygen launched into space is used for rocket propellants and for fuel cells. Human bodies burn carbohydrates for energy from which some of the carbon from them is oxidized in human cells to make carbon dioxide and later on breathing it out into the air. The carbon dioxide can be split to recycle the oxygen during the mission. The split is done naturally by plants as part of their photosynthesis process, this allow the recycling not only of oxygen but also of carbon.

1.1.1.3. Radiation shield

Water can be used as a shielding material for interplanetary space missions, it would be better for radiation protection than metals since nuclei are the things that block cosmic rays, and water molecules, made of three small atoms, contain more nuclei per volume than a metal. This is where the concept Water Walls comes into play using water shielding no only as a protection system but a combination of life-support and waste-processing systems with radiation shielding.

1.1.1.4. Agriculture

Plants require a lot of water. The most of the water used for space plants can be recycled because plants transpire through their leaves and the water can be captured by using air dehumidifiers. Notwithstanding always will be small loses of water that need to be replaced.

1.1.2. Moon water

The lack of water on the Moon is almost total. Moon is rich in oxygen since many lunar rocks are oxides.However, extracting this oxygen requires a lot of energy and machinery. Hydrogen can also be found trapped in the lunar regolith due to the solar wind and at the lunar poles trapped in the cold trap areas.

1.1.3. Asteroid water

From the meteorites data and the samples returned by the hayabyusa2 and ORISIR-REx mission more and more is becoming known about the existence of water on asteroids in the form of ice or in the form of hydrated minerals. The hydrated minerals in asteroids are thought to have originated when melting ice reacted with rocks in the early solar system to create this type of hybrid material.

Near-Earth Asteroids could be a rich source for water harvesting being these asteroids more accessible the the surface of the Moon

1.1.4. Earth water

On Earth water exists in the air as water vapor, in rivers, lakes, icecaps, glaciers, in the ground as soil moisture and in aquifers. About 71% of Earth surface is water-covered. Being the most of the water in the oceans. With such as huge amount of water in the Earth it could be logic to think about it as the main water source for the solar-system. Notwithstanding to escape from Earth's gravity requires such a huge amount of energy which is prohibitively expensive.

1.2. Asteroids

Asteroids are metallic, rocky bodies without atmospheres that orbit the Sun but are too small to be classified as planets.

1.2.1. Asteroid types by reflectance spectrum

These are defined by its albedo, which is a measure of how much light that hits a surface is reflected without being absorbed. The most of the asteroids fall into the following three categories:

- **The C-type**: Carbonaceous, are the most common asteroids (about 75% of known asteroids) . Being the ones of the most ancient objects in the solar system. Composition is thought to be similar to the Sun. Since are carbon based, they are very dark with an albedo of 0.03-0.09. Commonly located in the outer regions of the main belt. Since they are far away from the Sun, they have not been so altered by the heat of the sun. As most have not reached temperatures above 50 degrees Celsius, it is believed that they contain at least around 22% water in form of hydrated minerals.

- **The S-type**: Silicaceous, accounts for about 17% of all known asteroids. With an albedo of 0.1-0.22. Metallic iron mixed with magnesium-silicates. Commonly located in the inner regions of the asteroid belt.

- **The M-type**: Metallic, (nickel-iron). The composition of these asteroids is related with the distance to de sun. Relatively bright with an albedo of 0.1-0.18. Commonly located in the middle regions of the main belt.

1.2.2. Asteroid classifications

- **Main Asteroid Belt**: Majority of known asteroides orbit within the asteroid belt between Mars and Jupiter.
- **Trojans**: These asteroids share orbit with a larger planet withount colliding because they gather around Lagrangian points L4 and L5.
- **Near-Earth Asteroids**: These objects have orbits that pass close by to the Earth.

1.2.3. Near Earth Asteroids

By convention a celestial body is considered as a near-Earth object (NEO) if its perihelion is smaller than 1.3 AU and the aphelion is bigger than 0.983 AU. Asteroids and comets fall within this definition, with asteroids being the predominant object. When the near-Earth object is an asteroid it is called near-Earth asteroid (NEA). The population of NEAs are divided in three sub-populations: Atiras, Atens, Apollos and Amors. Those sub-populations are named after the first discovered asteroid from the corresponding sub-population. Some

NEAs are highly interesting for researchers because they can be explored at low velocities with respect to Earth (ΔV) and the small gravity of the body presenting opportunities for investigation and exploitation making them attractive targets for future explorations [14].

The number of known NEAs is increasing continuously from the last twenty years.

Table 1.1: NEAs sub-populations characterization. Own work.

Group	Definition	Description
NEAs	q $<$1.3AU	Near-Earth Asteroids
Atiras	(a $<$1.0 AU) and(Q $<$0.983 AU)	Entirely contained within Earth's orbit
Atens	(a $<$1.0 AU) and(Q $>$0.983 AU)	Earth-crossing
Apollos	(a $>$1.0 AU) and(q $<$1.017 AU)	Earth-crossing
Amors	(a $>$1.0 AU) and(1.017 $<$q $<$1.3 AU)	Orbit within Earth-Mars

Figure 1.1 shows the different types of sub-populations of NEAs depending on its orbital parameters.

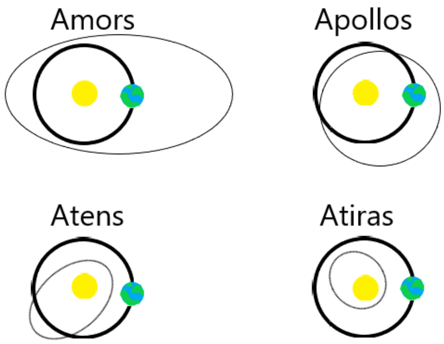

Figure 1.1: NEAs classification by orbit parameters. Own work.

Generally NEAs are smaller than the asteroids in the main belt. The largest NEA is Ganymed and it is about 32 km in diameter. The most of the NEAs have a diameter below 2 km.

1.2.4. Asteroid capture

The concept consists of detecting an easily accessible asteroid, light enough to be able to alter its trajectory in three phases: first to capture it in a near-Earth orbit, second to control and correct the orbit, and third to leave the asteroid in the desired orbit. This action is quite complex since it requires a total control of velocities and rotations of the bodies in the whole trajectory. Once the asteroid is in the desired place it can starts to be exploited in order to get the needed resources from it. The right place to locate those asteroids would be into stable orbits since it would reduce the complexity of the mission.

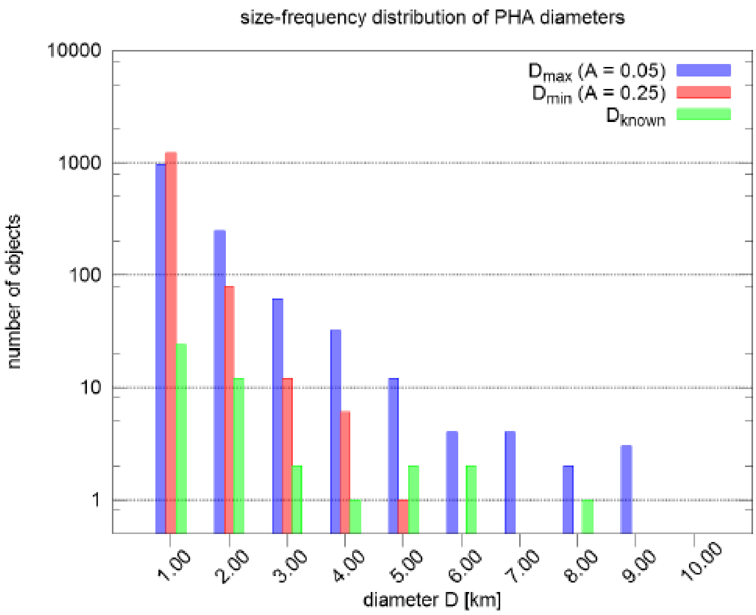

Figure 1.2: Graph of the size distribution of potentially hazardous objects depending on their diameter. The diameters are estimated based on absolute magnitudes and two different albedos: A=0.05 (blue), A=0.25 (red); the green boxes represent objects with known diameters. [1]

1.2.5. Asteroid Geologic processes

Sun was formed at the center of the solar nebula in a collapsing cloud of gas. During sun formation a hot disk was formed around the sun. Both the sun and the disk had mostly the same composition helium and hydrogen plus some other elements.As the disk and the gas cloud started to turn into solid grains some compounds condensed. Iron and rocky minerals were the only compounds of the granite forming near the Sun, whereas the the ones which formed beyond the orbit of Mars contained carbon and water. After an unknown process the grins melted and formed drop-like chondrules. The same elemental abundance as the Sun is in the grains loaded with carbon and water.

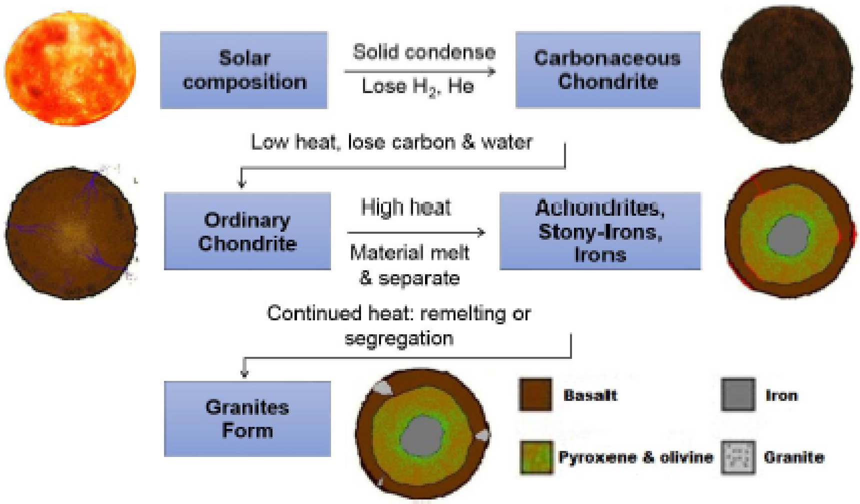

Figure 1.3: Geological processes in Asteroids during the formation. [2]

The primitive Asteroids were formed when grains and chondrules clustered together because of the gravity over time. After the creation of the asteroids impacts realeased pieces of them into Space, some of those pieces became located in orbits that intersects with Earth, and those become meteorites.

1.3. Ongoing missions to asteroids

There are already a number of space missions that have focused on asteroid analysis. Some of the most relevant are:

1.3.1. Fobos-Grunt

Is an asteroid sample-return mission operated by the Russian space agency(Roscosmos). It was launched on on 8 November 2011 aiming to rendezvous Phobos. This mission is the result of the development of a new generation of space vehicles for planetary exploration. And the project is targeted to sample return from Phobos (PSR) with an extended capability of some Main belt asteroids, NEOs and comets rendezvous.

Spacecraft properties:

- Launch mass 13,505 kg
- Dry mass 2,300 kg
- Power (1 to 1.3) kW.

Figure 1.4: Fobos-Grunt spacecraft. [3]

Mission objectives:

- Investigation of ancient matter pertinent to asteroid class bodies with remote sensing, in situ techniques and the most challenging goal of delivering samples to Earth for laboratory studies;

- Study of physical–chemical properties of the Phobos surface and inner structure, with relationship to orbital and rotational motions;
- Study of Martian environment at the Phobos orbit and dusty torus (if any);
- Mars exploration on the inbound phase of PSR mission and during on-Phobos operations;
- Measurements in interplanetary space and enroute to Mars.

This mission failed on orbit when malfunction stranded the spacecraft in Earth orbit. The spacecraft burned up in Earth's atmosphere.

1.3.2. OSIRIS-REx

OSIRIS-REx is the third major planetary science mission of NASA's New Frontiers Program. OSIRIS-REx is an acronym for "Origins, Spectral Interpretation, Resource Identification, Security-Regolith Explorer." The mission's goal is to collect a sample of at least 59.5 grams from asteroid 101955 Bennu and bring it back to Earth.

It was launched on Sept. 8, 2016, the OSIRIS-REx spacecraft traveled to a near-Earth asteroid called Bennu , and collected a sample of the of rocks and material from the surface that it will return to Earth in 2023. The mission will help scientists investigate how planets formed and how life began, as well as improve our understanding of asteroids that could impact Earth.

Spacecraft properties:

- Launch mass 2110 kg.
- Dry mass 880 kg.
- Dimensions (2.44 x 2.44 x 3.15)m
- Power (1.226 to 3) kW.

Mission objectives:

- Return and Analyze a Sample
- Map Bennu's Global Properties
- Document the Sample Site
- Study the Yarkovsky Effect
- Improve Asteroid Astronomy

The mission is not over yet, it is expected to be completed in 2023 when spacecraft returns to earth.

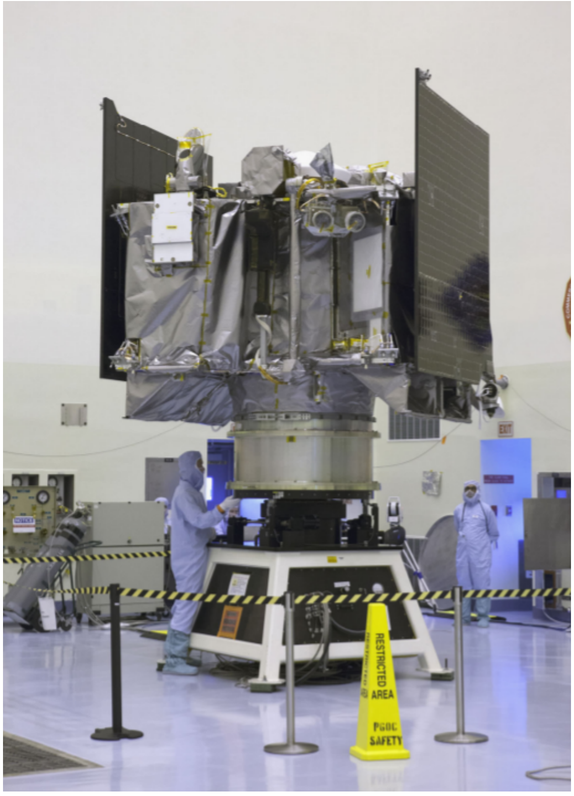

Figure 1.5: OSIRIS-Rex spacecraft. [4]

1.3.3. Hayabusa2

Is an asteroid sample-return mission operated by the Japanese space agency(JAXA). It was launched on 3 December 2014 and rendezvoused in the NEA 162173 Ryugu on 27 June 2018. It surveyed the asteroid and took samples leaving the asteroid in 2019 and returning the samples to Earth in 2020. The spacecraft is now on an extended mission to the small asteroid 1998 KY26.

Some of the spacecraft properties were: Spacecraft properties:

- Launch mass 610 kg.
- Dry mass 490 kg.
- Dimensions (1 x 1.6 x 1.25)m of the bus + (6 x 4.23)m for solar panel.
- Power (1.4 to 2.6) kW.

Mission objectives:

- Asteroid Rendezvous and Sample Return

Hayabusa2 is the continuity of Japan's original Hayabusa mission, which was the first spacecraft to sample an asteroid and also the first mission to successfully land on and take off from an asteroid. On 13 June 2010, it returned samples from asteroid 25143 Itokawa to Earth.

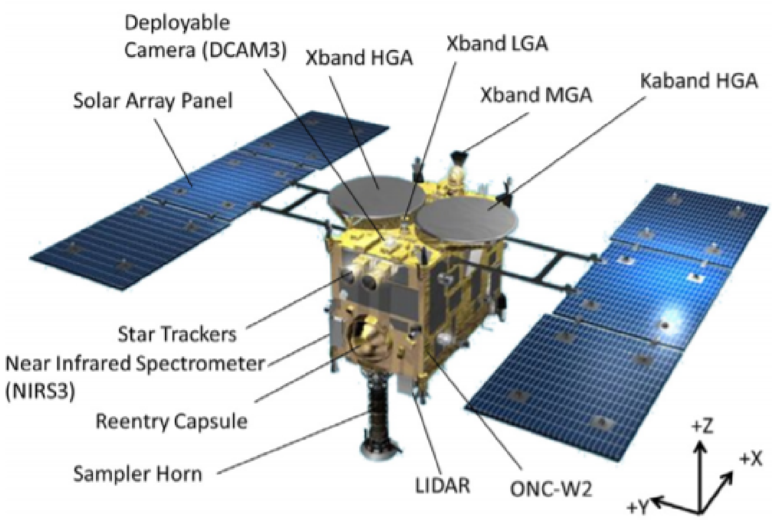

Figure 1.6: Hayabusa 2 spacecraft. [5]

The Hayabusa2 mission is similar to NASA's OSIRIS-REx mission to the asteroid Bennu. OSIRIS-REx successfully collected a sample from Bennu in October 2020 and will bring it back to Earth in 2023. Both missions explored C-type asteroids, which are believed to be the rocky building blocks of the early solar system.

CHAPTER 2. TECHNICAL ANALYSIS

2.1. Summary of the analysis

The main objective of the technical analysis is to determine the feasibility to modify the orbit of the targe asteroid from its orbit to a Moon distant retrograde orbit by estimating the: ΔV, mass of fuel and thrusting time required to perform each of the manoeuvres involved in the mission.

2.1.1. Software

The whole technical analysis has been developed by means of the following frameworks and libraries:

- Candidate asteroids: **JPL Small-Body Database Search Engine** using the online interface.
- Preliminary analysis of candidate asteroids: **JPL Small-Body Mission-Design Tool** using the online interface.
- Moon ephemerides: **HORIZONS** which is tool that provides ephemerides for solar-system bodies.
- Rendezvous to target asteroid: **Pykep** which is Scientific library written in C++ and exposed to Python developed at the European Space Agency to provide basic tools for astrodynamics research.
- The rest of the mission: **Poliastro** wich is Python library which provides an elegant and powerful API to solve for astrodynamics and orbital mechanics problems.

2.2. Target Asteroids

2.2.1. Constraints to choose the best candidates

In order to develop the best possible analysis, some constraints have been applied to the search for candidate asteroids that has allowed to reduce the list of potentially good asteroids for the mission from 1,075,180 to 4 bodies.

Orbital parameters: To reduce the time and energy consumption of the mission, it was decided to reduce the list of asteroids to only Near-Earth asteroids (perihelion distance less than 1.3 AU), thus reducing the list of candidate asteroids from 1,075,180 to 25,617 bodies.

Albedo: Dark bodies with an albedo between 0.03-0.09 are required, since this albedo corresponds to bodies of spectral classes C, which have the highest amount of volatiles. This reduces the list of candidate asteroids from 25617 to 460 bodies.

Size: Bodies with a diameter of less than 30 meters, to facilitate the operation of the spacecraft. This reduces the list of candidate asteroids from 460 to 4 bodies.

When choosing the asteroids its important to check that the sigma related to diameter and albedo is small.

2.2.2. Target candidates

Target candidates asteroids for the mission were obtained from the 'Jet Propulsion Laboratory Small-Body Database Search Engine' applying the constraints mentioned in the previous subsection. Asteroids with best properties for the mission are shown in the following table:

Table 2.1: Best NEAs candidates for the mission. Own work.

Object fullname	SPK-ID	Albedo	Sigma (Abedo)	Diameter (Km)	Sigma (Diameter)	Inclination (deg)	Period (years)
(2010 TN4)	3547937	0.062	0.098	0.018	0.006	3.03	2.26
(2010 DL)	3508142	0.080	0.018	0.019	0.001	2.57	1.57
(2010 FW9)	3512703	0.075	0.014	0.024	0.001	3.77	2.10
(2010 YD)	3554416	0.060	0.068	0.026	0.009	0.44	2.92

The asteroids targeted in this analysis 2.1 are different from the common asteroids analysed in other papers (such asteroid 2006 RH120 in paper [15]) for exploitation because the albedo of those asteroids is much higher than what is sought in this study. It would make the mission easier to target smaller asteroids than the ones shown in 2.1, but such small C-type asteroids are very dark are really complicated to observed.

Table 2.1 shows that three of the target candidates possess characteristics that contribute interesting features for the mission:

- 2010 TN4: It has a smallest diameter facilitating its orbit manoeuvring.
- 2010 DL: It has the closest orbital period to earth.
- 2010 YD: It has the smallest albedo, which increases the chances that it is a C-type asteroid (at least superficially), which increases the chances that it contains hydrated minerals.

From figure B.5 it can be seen in a qualitative way that asteroid 2010DL has the most similar orbit to the Earth orbit than the rest of candidate asteroids.

2.2.3. Preliminary analysis

The variable that defined which of the candidate asteroids would be finally chosen for the mission is the ΔV needed to rendezvous from the Earth, since this is the parameter to be minimized in order to reduce the mission costs.

As a first approximation for the analysis of the required ΔV, the 'JPL Small-Body Mission-Design Tool' was used to obtain the porkchop plots shown in the appendix B.1, B.2, B.3, B.4. From porkchop plots we can get which are the best launch and arrival windows to minimize needed ΔV with low thrust assuming constant acceleration.

Table 2.2: First estimation of needed ΔV to rendezvous candidate asteroids. Own work.

Object fullname	SPK-ID	Departure date	Arrival date	Minimum estimated ΔV
2010 TN4	3547937	2035-09-04	2037-05-31	6.8 km/s
2010 DL	3508142	2031-12-14	2033-03-23	5.1 km/s
2010 FW9	3512703	2029-05-08	2032-01-23	6.9 km/s
2010 YD	3554416	2037-02-10	2040-08-28	7.5 km/s

After the preliminary study of the four asteroids candidates for the mission, it has been determined that the best option is the asteroid 2010 DL, since it is the one that requires the less energy to rendezvous assuming an energy saving of up to 25% with respect to the second candidate asteroid that would require the least consumption for the rendezvous, which would be the 2010 TN4.

Target asteroid mass: There is no enough information about the geometry of this small asteroid, therefore it will be assumed to be closed to an sphere.

$$Vol = \frac{4}{3}\pi(\frac{D}{2})^3 \rightarrow Vol = 3591.364m^3 \tag{2.1}$$

Where 'Vol' is the volume in $[m^3]$ and 'D' is the diameter in $[m]$.

To determine the density of the asteroid, it is needed to resort to enlightened guesswork [16].

However, as density depends on the asteroid spectral type. Some approximations can be made. In this case, for type C, the best guess [17] is to estimate the density of asteroid 2010 DL as 1.38 $\frac{g}{cm^3} = 1380\frac{Kg}{m^3}$

Consequently, mass can be estimated:

$$Density = \frac{Mass}{Vol} \tag{2.2}$$

$$Mass = 4.956 * 10^6 kg \tag{2.3}$$

Thus asteroid 2010 DL mass is 4956082.32 kg. In figure B.6 it can be seen the orbit of the asteroid 2010DL and the orbit of Earth.

2.3. Mission

Mission will be divided into 4 stages:

- Stage 1: Rendezvous, the spacecraft will departure from the earth to finish wrapping the asteroid.
- Stage 2: Interplanetary cruise, Hohmann transfer from asteroid orbit to Earth orbit.
- Stage 3: Enters to Earth SOI, hyperbolic trajectory and capture into the Earth capture orbit.
- Stage 4: Lunar Orbit Injection, Hohmann transfer from the apogee of the capture orbit to the Lunar DRO.

Assumptions: The rotation of the asteroid about its own axis is considered to be zero to simplify the calculation process in this thesis. Future work should focus on how to wrap the asteroid while it rotates on its own axis, since not knowing the exact geometry of the asteroid complicates the calculation of the center of mass once the spacecraft wraps the asteroid.

2.3.1. Reference frame

The whole mission has been evaluated applying the Heliocentric mean (IAU 1976) ecliptic coordinates. In this frame the origin of the coordinates is the center of the sun with the x-axis pointing in the direction of the mean equinox of J2000 and the xy-plane in the ecliptic of J2000 according to the IAU 1976/1980 obliquity model.

2.3.2. Stage 1: Asteroid rendezvous

During this stage, the spacecraft will be launched from a parking orbit around the Earth to rendezvous the asteroid as shown is figure 2.3.2.

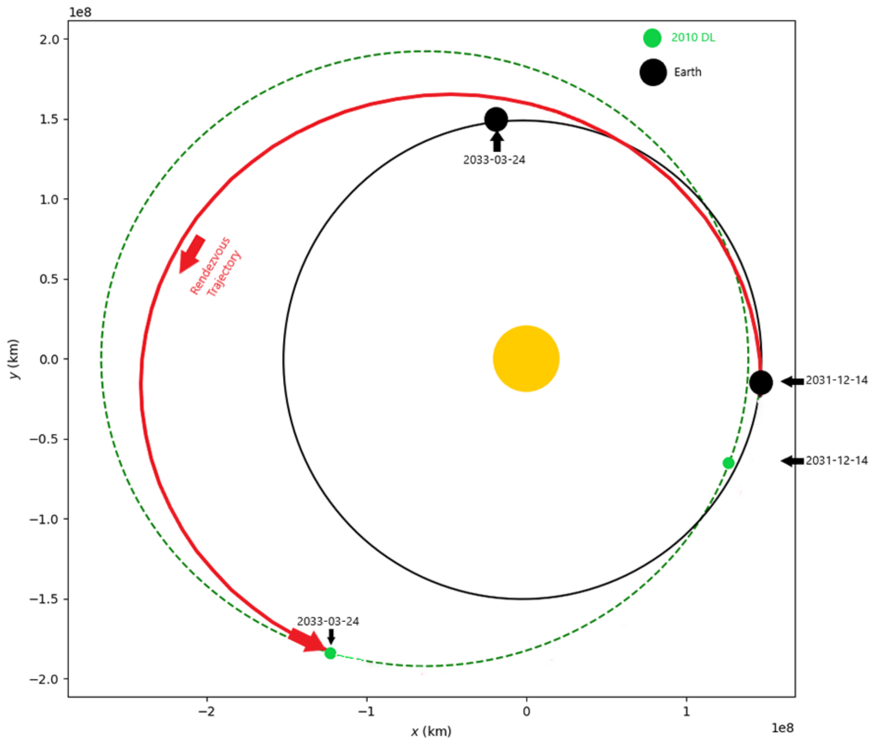

Figure 2.1: Earth-2010DL Rendezvous trajectory. Own work.

2.3.2.1. *Getting asteroid parameters*

To get the ephemeris of the asteroid it has been used Astroquery (tool for querying astronomical databases), returning the osculating Orbit. The ephemeris of the asteroid given by the JPL database are the ones corresponding to the date in which that asteroid was discovered, which in this case in 2010.

2.3.2.2. *Orbit propagation*

Once the ephemeris of the asteroid are extracted from the database. It is needed to propagate the position of the asteroid to the needed epoch.

To do so, it is needed to deal with the long term three dimensional propagation of the NEA.

The algorithm used to propagate the position of the asteroid through its elliptic orbit uses the mean motion and depends on the geometric shape of the orbit [18].

$$M = M_0 + \frac{\mu^2}{h^3}(1-e^2)^{\frac{3}{2}}t \tag{2.4}$$

Where 'M' is the mean anomaly, 'μ' is the gravitational parameter, 'h' is the angular momentum $h = |h| = |rx\dot{r}|$, 'e' is the eccentricity and 't' is the time.

2.3.2.3. Launch window

To decide the precise time period during which the spacecraft can be launched to rendezvous the asteroid at minimum energy cost porkchop plots have been used B.2. Porkchop plots are charts that shows contours of equal characteristic energy(C_3 against combinations of launch date and arrival date for a particular interplanetary flight.)

Concluding as the most optimal launch window to rendezvous the target asteroid 2.3.

Table 2.3: Rendezvous launch window. Own work.

Epoch	Maneuver phase
2031-12-14	Launching from Earth
2033-03-24	Arrival to asteroid

2.3.2.4. Lambert problem

Knowing the position of the Earth r_1 and the asteroid r_2 at a given epoch and the transfer time it is possible to determine the orbital segment joining r_1 and r_2 in such Δt.

Lambert proposed that the orbital transfer time Δt from r_1 to r_2 is determined by:

$$\sqrt{\mu} = \Delta t = f(a, r_1 + r_2, c) \tag{2.5}$$

Where 'Δt' is the time of flight between 'r_1' and 'r_2', 'r_1' is the initial position, r_2 is the final position, 'a' is the semi-major axis and 'c' is the chord between r_1 and r_2.

So Δt is independent of the orbit's eccentricity and just depends of the position 1, position 2, semi-major axis and the length of the chord joining r_1 and r_2.

The following second-order differential equation of motion must be solved:

$$\ddot{r} + \frac{\mu}{r^3}r = 0 \tag{2.6}$$

Where '' is the acceleration, 'r' is the position and 'μ is the gravitational parameter.

With boundary conditions: 2 vector and 2 scalars:

$$r(t_1) = r1 \tag{2.7}$$

$$\Delta t = t2 - t1 \tag{2.8}$$

$$r(t_2) = r2 \tag{2.9}$$

Plus the specification of long or short path.

Therefore the amount of time required for the transfer for elliptical orbits with Lagrange time equation is:

$$\Delta t = \sqrt{\frac{a^3}{\mu}}[2\pi \cdot n_{rev} + \alpha_e - sin(\alpha_e) - t_m(\beta_e - sin(\beta_e))] \tag{2.10}$$

Where:

$$sin(\frac{\alpha_e}{2}) = \sqrt{\frac{r_1 + r_2 + c}{4a}} \tag{2.11}$$

$$sin(\frac{\beta_e}{2}) = \sqrt{\frac{r_1 + r_2 - c}{4a}} \tag{2.12}$$

and 't_m is the transfer method, which has a value of +1 if it is counter clock wise or -1 if it is clockwise.

When solving the Lambert problem some issues arise such as:

- Singularities: When the true anomaly is equal to 180º the plane of the transfer orbit in not determined.
- Differences among the three types of conic section: Universal Methods and case by case treatment (different equations for different orbits).
- Conic Degeneracies: When true anomaly is equal to 0º or 360 º.
- Choice of the initial guess.
- Numerical divergence.
- Loss of precision.

Lambert solver A Lambert solver can be defined as a procedure that returns, for a gravitational field of strength μ all the possible velocity vectors v_1 and v_2 in a transfer time Δt. The ingredient of the algorithm is to choose a variable to iterate upon and thus invert the time of flight curve, the iteration method, the starting guess to use the iteration method and the reconstruction methodology to compute v_1 and v_2 from the value returned by the iterations. The solver used in this analysis is the Izzo solver [19], since it simplify the problem suggesting efficient approximations to the final solution.

The main difference of the Izzo solver compared to the rest of the Lambert solvers is the use of the Householder iterative scheme as a root finder for the time-of-flight curves.

$$T(x) - T^* = 0 \tag{2.13}$$

The Izzo solver implements the following equation:

$$x_{n+1} = x_n - f(x_n)\frac{f'^2(x_n) - f(x_n)f''(x_n)/2}{f'(x_n)(f'2(x_n) - f(x_n)f''(x_n)) + f'''(x_n)f^2(x_n)/6} \tag{2.14}$$

where f is $T(x) - T^*$ and the derivatives are indicated as f',f" and f"'. 'T' is the time of flight.

2.3.2.5. Celestial bodies position in the referee frame

Table 2.4: Earth position and velocity at launch time. Own work.

Earth	**x**	**y**	**z**
Position [km]	2.18e+07	1.46e+08	3.97e+03
Velocity [km/s]	-29.96	4.38	-9.70e-04

Table 2.5: Asteroid 2010DL position and velocity at arrival time. Own work.

2010DL	**x**	**y**	**z**
Position [km]	1.85e+08	-1.06e+08	-5.76e+05
Velocity [km/s]	4.95	23.77	1.05

2.3.3. Stage 2: Interplanetary Cruise

Interplanetary cruise is one step on the Patched Conics.Being the Patched Conics a method to estimate ballistic interplanetary trajectories. This method is mainly ruled by the two following assumptions:

- At one time, there is only 1 central body that acts on the spacecraft.
- Inside every sphere of influence, we can apply the two-body equations.

In the book [1] two main maneuvers as listed are recommendable for the interplanetary cruise, Hohmann transfer and Bi-elliptic transfer. For this mission the maneuver to realize the interplanetary cruise will be the Hohmann transfer since even in some cases bi-elliptic maneuvers consumes less fuel than the Hohmann transfer in this case the ratio final to initial semi-major axis is much smaller than 11.94.

2.3.3.1. Hohmann transfer

Hohmann transfers are typically the most efficient transfer a spacecraft can make to change the size of an orbit. This maneuver consider two tangential impulses ΔV_T,two coplanar Orbits with the same focus and both the initial and final orbit must be circular.

Figure A is a scheme the whole maneuver. In this maneuver the total ΔV is computed as a function of the initial and final radius.

$$a_{trans} = \frac{r_0 + r_f}{2} \tag{2.15}$$

Where 'r_0' is the radius of the initial orbit, 'r_f' is the radius of the final orbit and 'a_{trans}' is the semi-major axis of the transfer orbit.

$$e_{trans} = \frac{r_0 - r_f}{r_0 + r_f} \tag{2.16}$$

Where 'e_{trans}' is the eccentricity of the transfer orbit.

$$\Delta t_{trans} = \frac{T_{trans}}{2} = \pi\sqrt{\frac{a_{trans}^3}{\mu}} \tag{2.17}$$

$$\Delta V_1 = \sqrt{\frac{2\cdot\mu}{r_0} - \frac{\mu}{a_{trans}}} - \sqrt{\frac{\mu}{r_0}} \tag{2.18}$$

$$\Delta V_2 = \sqrt{\frac{\mu}{r_f}} - \sqrt{\frac{2\cdot\mu}{r_f} - \frac{\mu}{a_{trans}}} \tag{2.19}$$

Where 'ΔV_1' is the variation of velocity that the spacecraft must have in order to leave the initial orbit and enter into the transfer orbit, 'ΔV_2' is the variation of velocity the spacecraft must have in order to leave the transfer orbit and enter into the final orbit,

$$\Delta V = |\Delta V_1| + |\Delta V_2| \tag{2.20}$$

Where 'ΔV' is the absolute value of the total acceleration the spacecraft must produce in order to execute the whole maneuver.

In this case none of the orbits are circular but if the maneuver is started at the perigee of the asteroid orbit and ended in the apogee of the Earth orbit this problem is solved. As shown in the Hohmann trajectory figure 2.3.3.1..

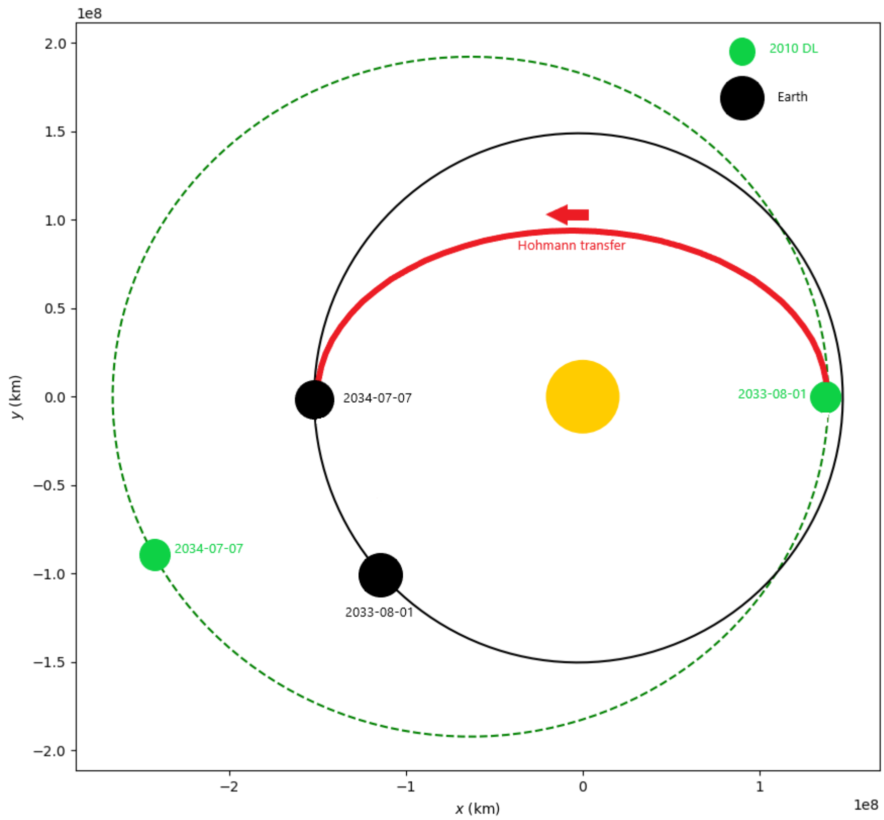

Figure 2.2: Hohmann transfer from asteroid 2010DL orbit to Earth orbit. Own work.

The starting point of the Hohmann transfer is the perigee of the asteroid 2010DL orbit. Being the initial radius 138.86e+06 km and the ending point is the apogee of the Earth orbits, being the final radius 152.5e+06 km. And the module of the initial velocity 35.43 km/s while the module of the final orbit 29.31 km/s.

Table 2.6: Position and velocity of asteroid 2010DL when starting hohmann transfer. Own work.

2010 DL	**x**	**y**	**z**
Position [km]	5.06e+06	1.39e+08	5.67e+06
Velocity [km/s]	-35.40	1.40	-0.65

Table 2.7: Position and velocity of Earth when ending hohmann transfer. Own work.

Earth	**x**	**y**	**z**
Position [km]	-6.21e+05	-1.53e+08	3.37e+04
Velocity [km/s]	2.93e+01	-8.83e-02	-1.30e-03

2.3.4. Stage 3: Planetary hyperbolic capture

At Arrival Planet SOI, the spacecraft arrives with an hyperbolic excess velocity ν_{inf} therefore a ΔV_p must be applied to enter the spacecraft into the capture orbit. The scheme of the whole stage is shown in figure A.5.

Sphere Of Influence: In astrodynamics is the oblate-spheroid.shaped region around a celestial body where the primary gravitational influence on an orbiting object is that body. In the patched conic approximation, used in estimating the trajectories of bodies moving between neighbourhood masses using a two body approximation. To determine the exact radius of the Earth sphere of influence it has been used Laplace Radius:

$$r_{SOI} = a(\frac{m}{M})^{\frac{2}{5}} = 924647Km \tag{2.21}$$

Where 'r_{SOI}' is the radius of the sphere of influence, 'a' is the semi-major axis of the smaller object, 'M' is the mass of the bigger body and 'm' is the mass of the smaller body,

Capture orbit: To define the best capture orbit, radius at the perigee and eccentricity must be well defined to optimize the spacecraft trajectory. [13]

$$r_{poptimum} = \frac{2\mu_2(1-e)}{v_\infty^2(1+e)} \tag{2.22}$$

Where '$r_{poptimum}$' is the optimum perigee radius for the capture orbit that minimizes the ΔV, 'v_{inf}' is the velocity of the spacecraft at the entrance to the Earth sphere of influence.

Optimum perigee radius of the capture orbit depends on the gravitational parameter of the earth, the velocity at the border of the SOI and the eccentricity. From these three variables the first two have already been computed. ...

$$v_\infty = -\Delta V_2 \tag{2.23}$$

Thus, eccentricity must be defined. As it is an open problem some constraints have been imposed such as (r_a<r_{SOI}) and (r_p<Distance Earth-Moon).

Approach Hyperbola:

$$e = \frac{r_a - r_p}{r_a + r_p} \tag{2.24}$$

$$v_p = \sqrt{v_\infty^2 + \frac{2\mu_2}{r_p}} \tag{2.25}$$

$$v_{capture} = \sqrt{\frac{\mu_2}{r_p}(1+e)} \tag{2.26}$$

$$\Delta V_p = v_{capture} - v_p \tag{2.27}$$

Where 'r_a' is the apogee radius, 'r_p' is the perigee radius, 'v_p' is the velocity at the perigee of the capture orbit and '$v_{capture}$' is the capture velocity.

By applying those equations for a range of possible perigee radius that goes from 10,000 km to 326,600 km (Average distance Earth-moon).

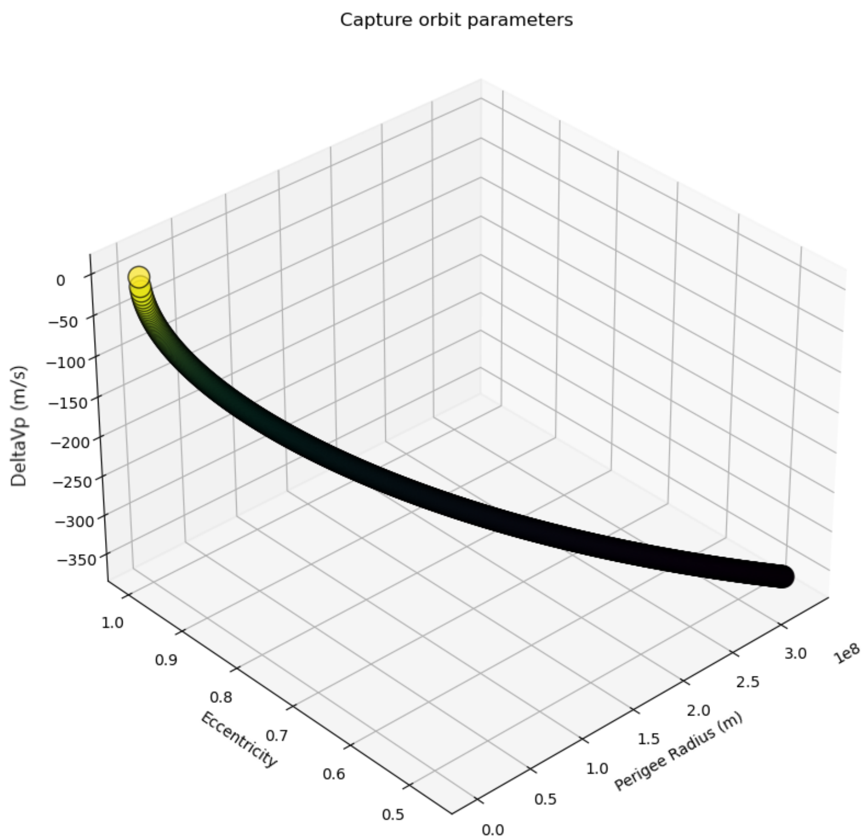

Figure 2.3: DeltaVp in function of capture orbit eccentricity and perigee radius. Own work.

From 2.3 it can be appreciated the clear relation between the needed ΔV_p needed to capture the asteroid into the desired orbit with the eccentricity and perigee radius of that orbit. Decreasing the needed ΔV_p as the perigee radius decrease while the eccentricity increase.

As the main constraint in the mission is the fuel consumption, the lower the radiusof the perigee of the captura orbit and the highest the apogee radius the better for the fuel consumption but some constraints must be applied to be realistic. To be conservative and to minimise as much as possible that the capture orbit interferes with other satellite orbits around the earth, the perigee radius has been taken as 100.000 km and to maximise the eccentricity the apogee radius has been tried to be as close as possible with certain limits of tolerance to the limits of the sphere of influence, so the apogee radius of the capture orbit will be 900.000 km.

Table 2.8: Capture orbit parameters. Own work.

Element	Value
e	0.8
r_p	100e+03 km
r_a	900e+03 km
T	40.72 Days

Table 2.9: Hyperbolic trajectory parameters. Own work.

Element	Value
e	1.11
a	450.85e03 km
b	218.14e03 km
β	25.82º
w	76.35º

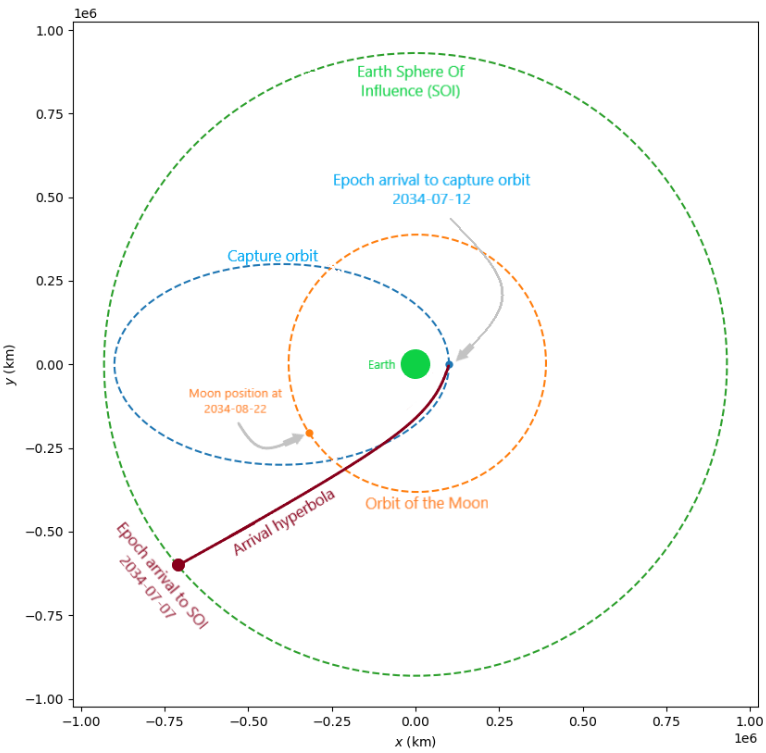

Figure 2.4: SOI, Arrival hyperbola, capture orbit and orbit of the moon. Own work.

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Figure 2.4 shows at scale the arrival hyperbolic trajectory that the spacecraft will execute once entering to the Earth SOI and that will end at the perigee of the capture orbit.

As it appears in this paper [20], it would be interesting to perform a deeper analysis of this phase of the mission applying the 3-body problem to obtain a more realistic value of what would be the delta V required to capture the asteroid in the capture orbit, taking into account not only the gravitational effects of the earth but also the gravitational effects of the moon, however, the effect of applying the 3-body problem in this phase would be that the delta V required to capture the asteroid would be lower. In any case, the calculations used in this section will give us a more restrictive value than the real value, so to check the feasibility of this operation is sufficient.

2.3.5. Stage 4: Lunar Orbit Injection

To determine the target orbit around the moon the Circular Restricted 3 Body Problem is applied.

Considering a non-inertial co-moving frame of frame between Earth and Moon with the origin of the frame at the center of mass of the system. Being the x-axis the one toward the moon, y-axis lies in the orbital plane to which z-axis is perpendicular. In this rotating frame of reference Earth and Moon appear to be at rest with the force of gravity balanced by the fictitious centripetal force required to hold in its circular path around the system center of mass.

In this frame appear the equilibrium points (figure 2.5). These are the location in the space where our spacecraft have zero velocity and zero acceleration appearing permanently at rest relative to the Earth and the Moon. [6]

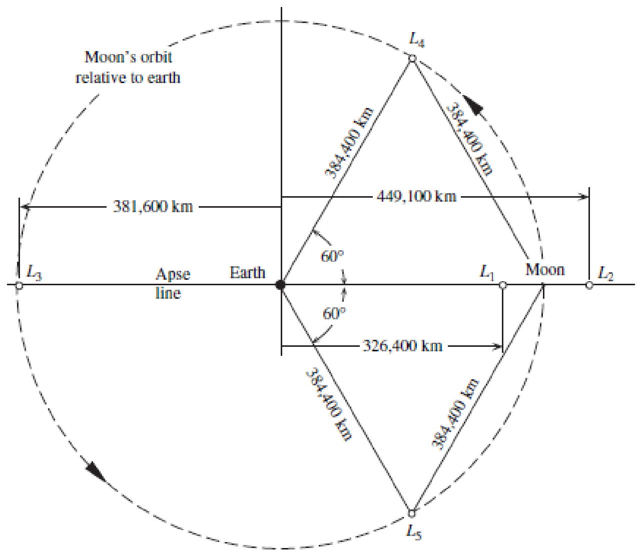

Figure 2.5: Location of the five lagrange points of the earth–moon system. [6]

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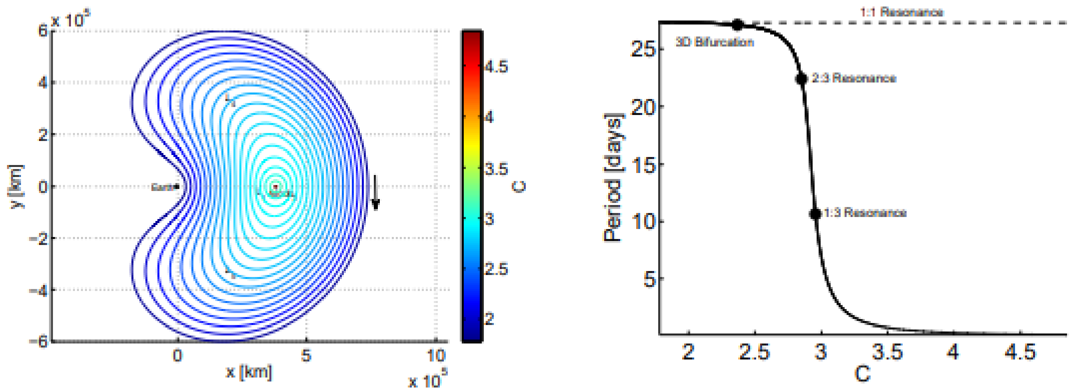
Distant Retrograde Orbit

Distant Retrograde Orbits are orbits which shows a periodical solution of CR3BP. These type of orbits were discovered by the French astronomer Hénon. DRO's amplitude can varies in a large range, being DRO with small amplitude regarded as the circumlunar orbit. When talking about DRO of large amplitude the classical orbital elements will fail. Therefor for this analysis the studied DRO will have an amplitude large enough to encompass the cislunar-libration point L1 and the translunar libration point L2. [21] These orbits does not escape from Moon or impact with Moon for a long time despite its stability has been destroyed by the Sun's gravitation. To compute the transferences orbit Lyapunov orbit are used which are orbits tangential to DRO. There are two different transfer types:

- Lyapunov orbit at L1 for quick transfer to DRO.
- Lyapunov orbit at L2 for low energy transfer to DRO.



(a) Geometry and Jacobi constant (b) Orbital period

Figure 2.6: DRO properties in the Earth-Moon CR3B problem. [7]

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Hohmann from the apogee of the capture orbit to the DRO

To transfer our spacecraft to the decided Lunar DRO it has been used the hohmann transfer. Starting the maneuver from the apogee of the Earth capture orbit and ending it at the point of the DRO that intersect the y-axis at y=0 behind the libration point L2.

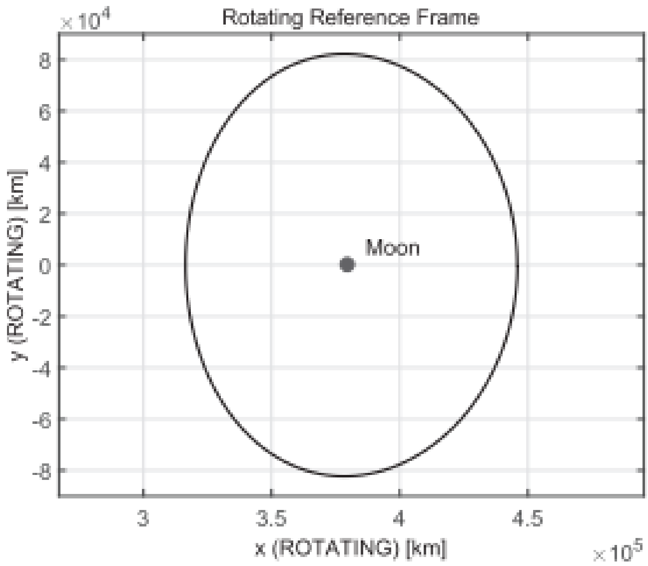

Figure 2.7: DRO in the rotating Earth-Moon reference frame [8]

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Being the initial radius of the hohmann transfer 900e+03km and the final radius 445e+03km.

Figure 2.8 shows the trajectory that the spacecraft would follow to go from the apogee of the transfer orbit to be injected one of the distant retrograde orbits of the moon.

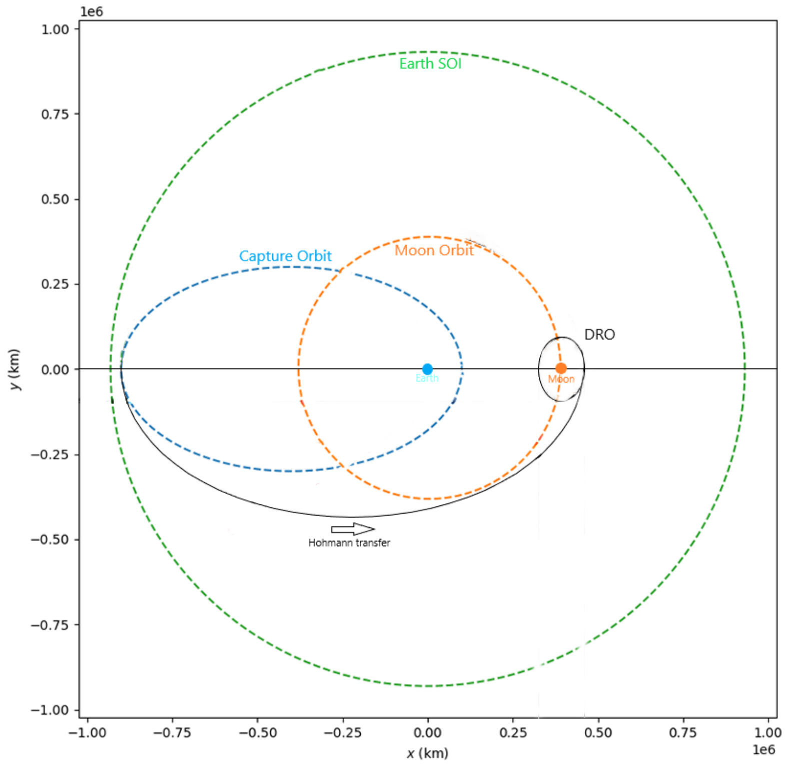

Figure 2.8: Hohmann transfer to the Lunar DRO. Own work.

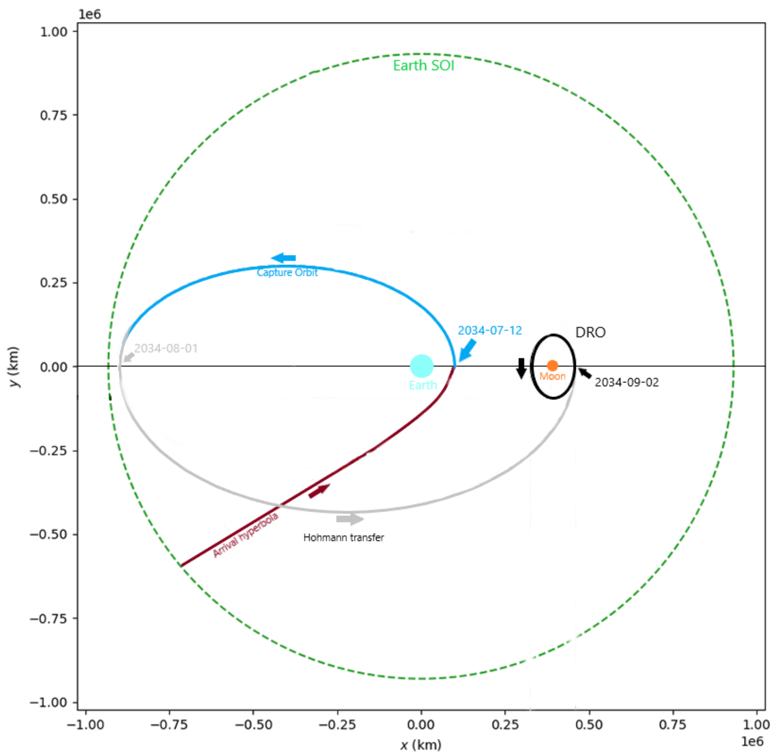

Figure 2.9: Total trajectory followed by the spacecraft from entry into the terrestrial SOI to injection into the lunar DRO. Own work.

2.4. Impulsive Maneuvers

The ideal Rocket equation:

$$\frac{\Delta m}{m_0} = 1 - e^{-\frac{\Delta V}{I_{sp} g_{SL}}} \tag{2.28}$$

Where 'Δm' is the variation of mass of the spacecraft which is the one corresponding to the burned fuel, 'm_0' is the initial mass of the spacecraft full of fuel, 'ΔV' is the needed velocity variation, 'I_{sp}' is the specific impulse of the used propellant and , 'g_{SL}' is the gravitational acceleration. From the previous equation it can be computed the mass of fuel needed to reach the required ΔV.

Table 2.10: Isp per type of propellant. [6]

Propellant	**Isp(S)**
Cold Gas	50
Monopropellant hydrazine	230
Solid propellant	290
Nitric acid/monomethyl hydrazine	310
Liquid oxygen/liquid hydrogen	455
Ion propulsion	$>$3000

Hence, the amount of needed fuel mass will be defined by the ΔV, the initial mass and the type of propellant used. For the rendezvous to the asteroid the initial mass will be just the mass of the spacecraft plus the fuel, but once the spacecraft has wrapped the asteroid the initial mass for each maneuver will have to take into account the mass of the asteroid. So the use of chemical propellant to move the asteroid will require excessive amounts of fuel that will not be available, therefore the only type of propellant that will allow the entire mission to be carried out will have to be ionic propelant.

Consequently the type of propellant used will defined the needed impulsive time to achieve the ΔV.

$$F = m \cdot ac \tag{2.29}$$

$$v_e = I_{sp} \cdot g_0 \tag{2.30}$$

$$\Delta T = \frac{m_0 \cdot v_e}{thrust} \cdot (1 - e^{-\frac{\Delta V}{v_e}}) \tag{2.31}$$

Where 'F' is the force in $[N]$, 'm' is the mass in $[kg]$, 'ac' is the acceleration in $[m/s^2]$, 'v_e' is the exhaust velocity in $[m/s]$, '$thrust$' is the thrust produced by the propulsion system.

The needed impulsive time to reach the desired ΔV must be less than the time of flight of the maneuver. As a low thrust propellant will be used for the mission there will be some maneuvers in which the impulsive time will be slighly longer than the time of flight of the maneuver, in those cases the spacecraft will start to impulse before starting the manoeuvre and will continue after finishing the maneuver.

Table 2.11: Similar spacecrafts. Own work.

Spacecraft	**Launch mass** [kg]	**Dry mass** [kg]
Hayabusa2	610	490
NEAR Shoemaker	805	487
NASA's Galileo	2560	1880
Dawn	1217	747
New horizons	478	410

The properties of the engine chosen for the consumption a time of burn is a Hall-effect thruster such as X3 ion thruster. Which is a type of ion thruster in which the propellant (Xenon) is accelerated by electric and magnetic fields. These thrusters are safer and more fuel efficient than engines used in traditional chemical rockets.

If chemical propellants were used to try to move such a heavy mass the amount of fuel that would be needed would massive massive so chemical propellant would make the mission unfeasible.

Main properties of the thruster used are that produces a thrust of 5.4 N and a specific impulse of 5000 s.

2.5. Results

Table 2.12: Final ΔV needed for each step of the mission. Own work.

Epoch	Phase	Total Δ V [km/s]
2031-12-14	Stage 1 - Launch from Earth	3.800
2033-03-24	Stage 1 - Arrival to asteroid	1.250
2033-08-01	Stage 2 - Hohmann transfer starts	0.679
2034-07-07	Stage 2 - Arrival to Earth SOI	0.664
2034-07-12	Stage 3 - Enters to capture orbit	0.222
2034-08-01	Stage 4 - Leaves capture orbit	0.125
2034-09-02	Stage 4 - Arrival to Lunar DRO	0.150

Initial spacecraft mass is 2000 kg.

Table 2.13: Consumed fuel and time of impulse needed. Own work.

Stage	ΔV [km/s]	Consumed fuel [kg]	tof [days]	Burn time [days]
1 - Rendezvous to asteroid	5.05	1.96	466.16	20.56
2 - Interplanetary cruise	1.34	1341.29	340.76	14096.27
3 - Planetary capture	0.22	224.03	4.92	2354.44
4 - Lunar orbit injection	0.27	276.91	31.99	2910.21

From these results it can be seen that the proposed mission is unfeasible since the needed time of thrusting needed to accelerate the asteroid to the needed ΔV is much higher than the time of the trajectory, even in the capture stage it would be needed 478 times a longer maneuver in order to have enough time to decelerate the asteroid to the desired velocities. As low thrust engines are been used the times of impulse are the biggest constraint in this mission due to the huge amount of mass to move.

2.5.1. Proposed solution

In order to make feasible the mission it would be needed to reduce the most of the mass involved in the mission, that is why the proposed solution is to not deflect the whole asteroid to orbit around the moon but directly extract the water from the asteroid and only send the processed water to the moon.

Therefore send to rendezvous the asteroid a swarm of spacecrafts of 700 kg each to drill the asteroid and extract the water from the asteroid until fill a deposit of 9500 kg of water each.

Once each spacecraft has full water tanks, it will wait until it is at apogee or perigee of the asteroid's orbit 2010DL to begin the transfer to the lunar orbit. In the case of starting the Hohmann transfer from the apogee of the orbit the needed ΔV is lower than departing from the perigee and it is 1.035 km/s.

Applying this solution the whole mission is feasible, the total amount fuel consumed represents just the 0.63% from the total launch mass and the most restricted maneuver which

Table 2.14: Consumed fuel and time of impulse needed in the proposed solution. Own work.

Stage	**DeltaV [km/s]**	**Consummed fuel [kg]**	**tof [days]**	**Burn time [days]**
1 - Rendezvous to asteroid	5.05	0.68	466.16	7.20
2 - Interplanetary cruise - from perigee	1.34	2.76	340.76	28.99
2 - Interplanetary cruise - from apogee	1.04 The	2.13	337.90	22.39
3 - Planetary capture	0.22	0.46	4.92	4.84
4 - Lunar orbit injection	0.27	0.57	31.99	5.99

would be the capture into the capture orbit needs a time of impulse lower than the time of flight of the maneuver.

2.6. Asteroid In-Situ water extraction

Commonly, space mining can be executed applying one of the following extraction techniques.

1. Surface mining: By mean of a auger the surface of the asteroid is scraped off, in asteroids composed of rubble piles this approach is no feasible.
2. Shaft mining: A spacecraft equipped with drills bore into asteroids to extract the materials within.
3. Magnetic rakes: A magenet is used to gather the metal elements of the asteroid. In this case, as the asteroid in study in a type-c this approach doesn't makes sense.
4. Heating: By applying heat, volatiles are extracted from the asteroid. This technique works quite good for C-type asteroids.
5. Mond process: Process used to extract the nickel and iron of an iron rich asteroid by passing carbon monoxide over the asteroid at a temperature between 50 and 60º for nickel and higher for iron. This process does not works for water extraction.

This work will focus on surface mining, in which a spacecraft will wrap the asteroid and once the asteroid and the spacecraft become one, the spacecraft will begin extracting resources from the asteroid. Specifically, this work is only focused on the extraction of volatiles from the asteroid and not on other types of resources, so the extraction methods and technologies studied will be processes focused only on the extraction of volatiles.

2.6.1. Mine

In the mining context, the common steps followed are to mine the feedstock and then transport it to a a processing plant where the desired part of the feedstock is extracted. Unfortunately this process consumes too much energy to be carried out on the surface of an asteroid, that is why some alternative approaches such as Planetary Volatiles Extraction (PVEx) arises. PVEx is a design concept that combines mining and extraction and eliminates the energy-intensive and time-consuming transport step between the mining zone to the processing plant. [10] PVEx uses a bunch of process to capture bulk.

Hence, for this work it has been decided to deal with extracting volatiles in situ and leaving dry regolith behind in order to significantly reduce the system complexity. Planetary volatiles extractors are a double-walled coring auger with a heated inner wall that drills to a target depth obtaining a core sample that is heated up and afterwards, thanks to the condenser the volatiles are captured.

In this section it will be analysed three of the latest and best options to extract the needed volatiles from the target asteroid.

- Sniffer: Deep fluted auger with perforated stem. Material is heated up, volatiles flow through holes up the hollow auger stem to the cold trap.

- MISWE: Deep fluted auger captures sample and retracts into tube. The tube/auger is preload against the ground. Auger is heated and volatiles flow through the holes, up the annular space and into a cold trap.
- Corer: Double wall corer with outer insulating auger and inner perforated and conductive tube. Material within inner tube is heated, volatiles flow through holes and up the annular space into a cold trap.

These three options were considered to execute the In-Situ water extraction, lets consider the pros and contras of each one of those options after the results shown in Zacny works [10].

Being the experimental setup tested at 5 torr vacuum and using JSC-1A, which is a terrestrial material synthesized in order to approximate the chemical, mechanical, or engineering properties of, and the mineralogy and particle size distributions of, lunar regolith.

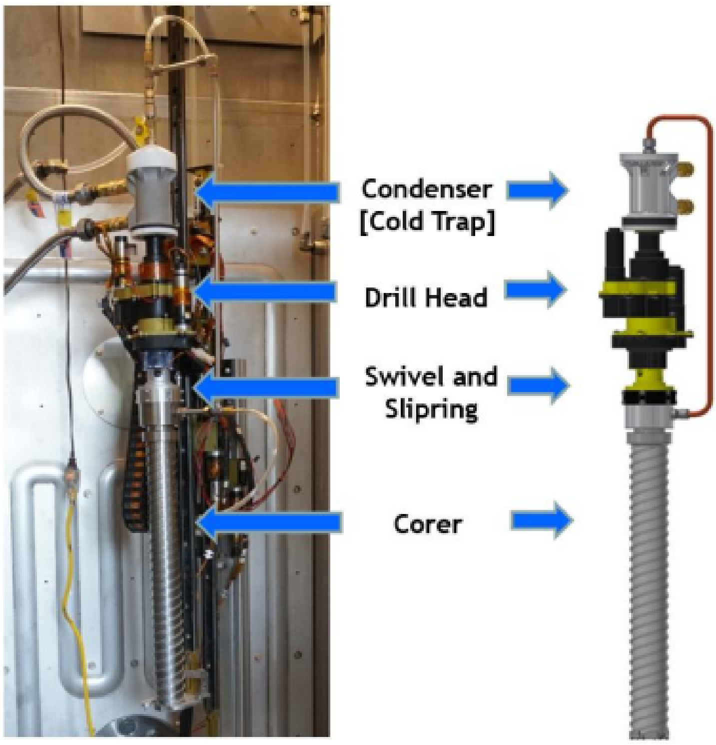

Figure 2.10: PVEx Corer breadboard assembly. [9]

Sniffer Functioning: A hollow auger with perforated walls that drills into volatile-rich regolith via heaters embedded within the auger itself, when the ice is in contact with the Sniffer it melts and sublimes releasing water vapor that flows through the holes up to the hollow auger ending in a condenser. Then volatiles are pumped from a borehole into a condenser.

Setup: It was used a 3D printer auger with several holes, a size of cm diameter and 15 cm long placed inside a frozen JSC-1A with 6wt% and wt% with varied power and time.

Figure 2.11: Sniffer experiment setup. [10]

Obtaining that the best result was of 5.5 g and it couldn't be repeated. The majority of heat used went to heat up and sublime surrounding soil. This sublimation caused lofting of soil particles which fell around circular bin. High temperatures caused the insulation to melt. It has some risks such as auger can freeze and holes can be clogged with the drilled material. Sniffer shows a low efficiency and a low complexity.

MISWE Functioning: Is a closed auger that retains material on its flutes, once it capture a volatile-rich material the auger is retracted back into a sleeve. The sleeve and the auger are then pushed against the ground to seal the auger inside the sleeve, while the auger is heated and volatiles sublimes they flow up to the auger through a swivel joint ending into a condenser. Once the water is extracted the auger is removed from the sleeve and spun at high speed to eject the dry soil via centrifugal action.

Setup: There was used a conventional auger with pins for heat spread. The auger made of sintered 316 stainless metal. With a diameter of 6.4 cm and a length of 13 cm long.

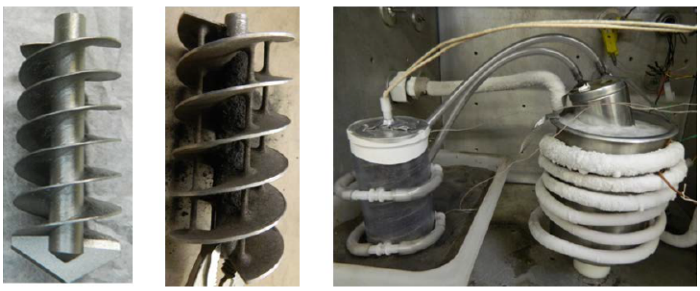

Figure 2.12: MISWE experiment setup, left images shows a conventional auger, middle image shows an alternative auger with pins and the third image shows a chamber setup. [10]

The results from this experiment were than both the auger and the vent hole were clogged, the water extraction efficiency was about 71%. It shows a high efficiency and a medium complexity.

Corer Functioning: Double-walled coring auger with heaters on the inside that penetrates the subsurface and captures a core inside, in that process the inner wall of the auger is then heated, therefore heating the regolith core what makes that the ice sublimes and volatiles flow up the auger through a swivel joint ending into a condenser. Once the volatiles are recovered the Corer is retracted leaving the dry regolith core behind.

Setup: There was used a double-wall coring auger of 2.5 cm and 5 cm in diameter. And perforated conductive metal tube made of sintered copper.

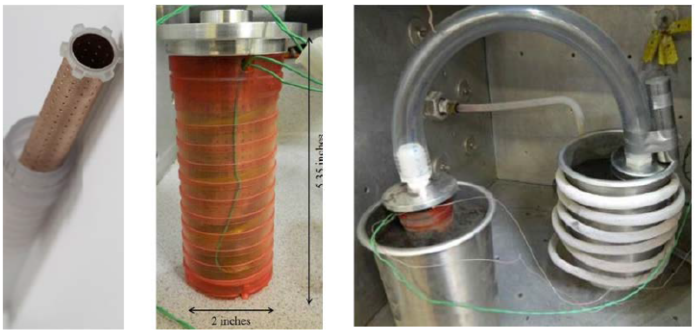

Figure 2.13: Corer experiment setup. [10]

Results showed that corer can extract up to 87% of water needing 60 watt of power for 40min. It has some risks such as it can be clogged with the drilled material. It shows a very high efficiency and a low complexity.

Table 2.15: PVEx performance comparison. [10]

	Sniffer	**Miswe**	**Corer**
Average energy effieciency [Whr/g]	36	3.55	2.2
Std. Dev energy effiencincy [Whr/g]	30	1.35	0.8
Average water recovery [%]	1.2	34.5	65
Std. Dev water recovery [%]	1.7	12.5	17
Ranking	3	2	1

Therefore for this mission the PVEx that will be used is the PVEx-Corer since it is the best one from the drilling and volatile extraction stand point. It has the a very high efficiency and a low complexity.

In order to provide enough energy for the PVEx Corer would be needed tho Multi-mission radioisotope thermoelectric generators(MMRTGs). It could be estimated that PVEx-Corer could be able to extract 30 kg of water per day.

Both MMRTGs would suppose a mass of 80 kg.

2.6.2. ISRU quantification

Taking into account that the whole asteroid has a total mass of $4.956 \cdot 10^6$ kg and it is of class C. A conservative prediction would be to estimate the amount of water on the asteroid at 10% of the total mass. This would mean that there would be $495 \cdot 10^3$ kg of water on asteroid 2010 DL. Since experiments Corer extractor can extract up to 87% of water. Hence theoretically corer would be able to extract up to 431179.16 kg of water from the 2010 DL asteroid, tanking into account that it extracts 30 kg per day it would need 14372.64 days to extract all the asteroid water. Since from the technical analysis it was checked that the optimal amount of water per spacecraft to be carried from the asteroid to the lunar DRO was 9500 kg, the corer would just need 316.67 days to extract it. Considering that the asteroidal perdiod around the sun is 1.57 years, per each orbital revolution the corer would be able to obtain 17191.5 kg of water.

- Corer can extract 30kg/day of water.
- To extract 9500 kg it needs 316.67 days.
- Per each asteroidal orbital revolution it can be extracted up to 17191.5 kg of water.

CHAPTER 3. ECONOMICAL AND LEGAL ANALYSIS

3.1. Economical analysis

The aim of this chapter is to determine if the cost of the mission compare to the money that would save to send water from the earth to the moon is worth it or not.

Performing the analysis of the economical viability of this mission can be reduced to compute the Net Present Value(NPV). [22]

NPV in this mission depends on:

- **Cost**: To launch and conduct the mission.
- **Mass**: Amount of mined material
- **Time**: It takes to accomplish.

3.1.1. Current context of sending water to the moon

The current price of sending 1kg of material from the Earth to the Lunar surface is within the range of 10,000$ to 35,000$. [23]. In order to consider as economically viable the whole mission of asteroidal mining it is seek at least a 25% cost saving to offset operational risk.

3.1.2. Mission cost

To determine the cost of mining the asteroid the following parameters must been taken into account:

- Hardware cost
- Hardware mass
- Launch cost

Launch cost: The development of commercial launch systems has substantially reduced the cost of space launch. NASA's space shuttle had a cost of about $1.5 billion to launch 27,500 kg to Low Earth Orbit (LEO), $54,500/kg. SpaceX's Falcon 9 now advertises a cost of $62 million to launch 22,800 kg to LEO, $2,720/kg.

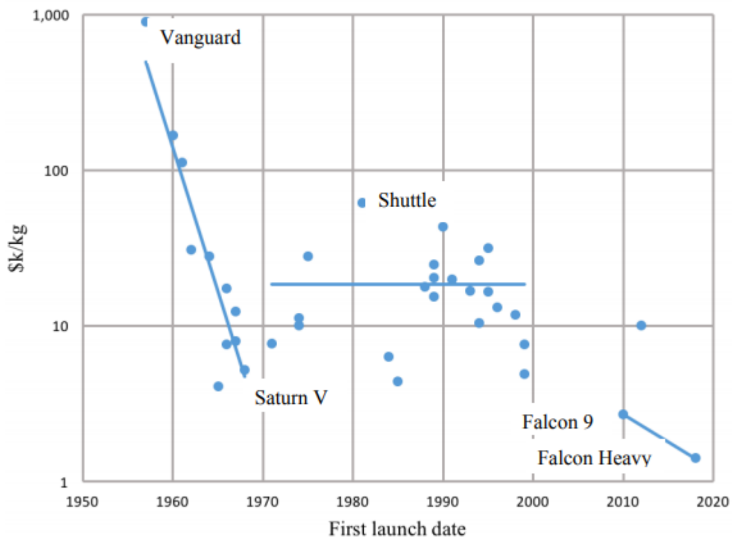

Figure 3.1: Launch cost per kilogram to LEO versus first launch date. [11]

To determine the cost of launching the spacecraft from the earth to the parking orbit the price can be assumed as \$5000/kg [24] for non reusable rockets and \$1400/kg for the latest reusable rockets. As the initial mass of each spacecraft is 700 kg the total launch cost range would be between \$980k to \$3.5M.

Table 3.1: A1. Launch cost to LEO in current dollars. [11]

System	First launch date	$k/kg
Zenit 3SL	1999	7.6
Falcon 9	2010	2.7
Vega	2012	10
Falcon Heavy	2018	1.4

The advent of reusable rockets has meant a huge reduction in the launch cost per kg.

$$LC = LM \cdot LCKG \tag{3.1}$$

Being LC the launch cost, LM the initial mass of each spacecraft and LCKG the launch cost per kg. Let's assume a conservative cost, about the average cost of the latest reusable rockets, which would be \$2.05k.

$$LC = \$1.435M \tag{3.2}$$

Trajectory cost: Cost derived from the amount of fuel needed to execute all the orbital maneuvers of the mission. The total amount of fuel consummed by each spacecraft is 4.47 kg of Xenon. The price per 1kg of xenon is \$1138 [25] , therefore:

$$FC = MF \cdot FCKG \tag{3.3}$$

Where FC is the fuel cost, MF is the mass of fuel and FCKG is the price per kilogram.

Being the total fuel cost (FC) of the mission from the earth parking orbit to the moon orbit \$5086.86.

Hardware cost and operational cost: The hardware cost is $100,000/kg [23] and the approximate mass in hardware is 500kg. Being the total cost in hardware $50 M. Assuming an annual cost per operations of $5.7M as average, that translating that number into a relative cost it would be $21.5k/kg [26].

$$HC = MH \cdot HCKG \tag{3.4}$$

Being HC the Hardware cost, MH the mass of hardware and HCKG the hardware cost per kg. So the hardware cost (HC) is $ 50 M.

$$OC = LM \cdot OCKG \tag{3.5}$$

Being OC the Hardware cost, LM the initial mass of the spacecraft and OCKG the operational cost per kg. So the operational cost (OC) is $ 15.05 M.

$$TC = LC + FC + HC + OC \tag{3.6}$$

Total cost of the mission is $ 66.49 M.

Taking into account the total amount of water carried by each spacecraft (9500kg), hence, the cost of obtaining and transporting 1kg of water from the target asteroid to the moon is $6999.

$$Revenue = 9500kg \cdot \$(10,000 \rightarrow 35,000) = (\$95 \rightarrow \$332.5)M \tag{3.7}$$

$$NPV = Revenue - TC = (\$28.51 \rightarrow \$266)M \tag{3.8}$$

$$ROI = \frac{Revenue - Investment}{Investment} \tag{3.9}$$

In case of selling the water at the same price that it cost to sent it from the earth to the moon each spacecraft carrying 9500 kg of water to the moon would generate a profit within the range of $(\$28.51 \rightarrow \$266)M$ which means a that in the worse case the ROI (Rate of investment) would be 42% and in the best case would be of 399%.

3.1.3. Economical conclusion

Currently the price of sending 1kg of water from earth to the moon goes from $10000 to $35000 while the price of sending it from the asteroid 2010DL would be $6999. which would be a reduction in cost per kg of water between the 30% to 76.67%, this reduction in cost is enough to offset operational risk.

3.2. Legal framework

This chapter is dedicated to briefly shed light on the legal status of asteroidal resources. Ownership of space started to appear with the first missions that achieve to get out of earth atmosphere. Creating space law as a new branch of international law in 1959. In that year the General Assembly of the United Nations created the 'Committee on the Peaceful Uses of Outer Space' [27]. Since the creation of this committee there has been issued and ratified by multiple countries five principles for outer space.

- **1967 - The Outer Space Treaty**: Treaty on Principles Governing the Activities of States in the Exploration and Use of Outer Space, including the Moon and Other Celestial Bodies.
- **1968 - The Rescue Agreement**: Agreement on the Rescue of Astronauts, the Return of Astronauts and the Return of Objects Launched into Outer Space.
- **1972 - The Liability Convention**: Convention on International Liability for Damage Caused by Space Objects.
- **1975 - The Registration Convention**: Convention on Registration of Objects Launched into Outer Space.
- **1979 - The Moon Treaty**: Agreement Governing the Activities of States on the Moon and Other Celestial Bodies.

From those five treaty the one most widely adopted is the first one 'The Outer Space Treaty' adopted by 110 parties while the Moon treaty just has been adopted by 18 parties.

In this chapter it only be explained the Outer Space Treaty and the moon agreement since those are the most relevant for asteroidal mining.

3.2.1. Outer Space Treaty

Forms the basis for international space law. The main points of this treaty are the prohibition of placing nuclear weapons in space, limits the use of the Moon and all other celestial bodies to peaceful purposes only and establishes that space shall be free for exploration and use by all nations but no nation may claim sovereignty of outer space or any celestial body.

- Article II: Outer space, including the Moon and other celestial bodies, is not subject to national appropriation by claim of sovereignty, by means of use or occupation, or by any other means.
- Article IV: Limits the use of the Moon and other celestial bodies to peaceful purposes, and expressly prohibits their use for testing weapons of any kind, conducting military maneuvers, or establishing military bases, installations, and fortifications
- Article VI: The activities of non-governmental entities in outer space, including the Moon and other celestial bodies, shall require authorization and continuing supervision by the appropriate State Party to the Treaty.

- Article IX: A State Party to the Treaty which has reason to believe that an activity or experiment planned by another State Party in outer space, including the Moon and other celestial bodies, would cause potentially harmful interference with activities in the peaceful exploration and use of outer space, including the Moon and other celestial bodies, may request consultation concerning the activity or experiment.

3.2.2. Moon Treaty

This treaty try to provide the necessary legal principles for governing the behavior of states, international organizations and individuals who explore celestial bodies other than Earth, as well as administration of the resources that exploration may yield.

- Article I: The Moon should be used for the benefit of all states and all peoples of the international community.
- Article XI.V: framework of laws to establish an international cooperation regime, including appropriate procedures, to govern the responsible exploitation of natural resources of the Moon.
- Article VII.I: Bans altering the environmental balance of celestial bodies and requires that states take measures to prevent accidental contamination of the environments of celestial bodies, including Earth
- Article XI.VII: The orderly and safe use of the natural lunar resources with an equitable sharing by all state parties in the benefits derived from those resources.
- Article XI: The placement of personnel or equipment on or below the surface shall not create a right of ownership.
- Article VI: There shall be freedom of scientific research and exploration and use on the Moon by any party without discrimination of any kind.

3.2.3. Others

3.2.3.1. Commercial Space Launch Competitiveness Act of 2015

This law was passed in 2015 to allow US industries to engage in the commercial exploration and exploitation of space resources without asserting sovereignty or ownership of any celestial body. This act includes the extension of indemnification of US launch providers for extraordinary catastrophic third-party losses [28].

3.2.3.2. Luxembourg space law

In 2017 Luxembourg issued a new law regarding space mining with the objective of set Luxembourg as the main hub for European space mining companies [29]. Therefore the Grand Duchy is the first European country, and the second worldwide, to offer a legal framework on the exploration and use of space resources, ensuring that private operators can be confident about their rights on resources they extract in space. The Luxembourg law also lays down the regulations for the authorization and the supervision of private space exploration missions, including both exploration and utilization of space resources.

CONCLUSIONS

This work has sought to create a new paradigm in the new methods of obtaining water that we can employ for the development of future space colonies, going beyond the tempting idea of just obtaining resources from the earth or directly exploiting the scarce water that exists on the lunar surface and putting the focus on the large amount of resources that exist in the solar system (in particular from asteroids) and of which we can take advantage and democratise the use of water in space by making it cheaper. This would facilitate the construction of lunar establishments and bases and would allow us to have larger quantities of water on the lunar surface that would serve both for life support and spacecraft refuelling.

This study has verified the technical feasibility of capturing a 19-metre diameter C-type asteroid, extracting volatiles from its surface and sending the extracted water in a spacecraft to a distant retrograde orbit around the moon. What limits the transport capacity of the mission are the propulsion systems that currently exist, since if these are chemical, the specific impulse they have is very low and therefore require excessive amounts of fuel, while the ionic ones have a very high specific impulse, their thrust is too low and the times they need to accelerate or decelerate medium-heavy masses are too high for the distances at which the near earth asteroids are located.As propulsion systems with higher specific impulse and thrust are developed, the amount of water that spacecraft can carry in these flight times can be increased. It may even be possible to transport the entire asteroid by capturing it and modifying its orbit until it orbits the moon to be exploited there, but this is not possible with current technology.

Defining a safe amount of water to be transported from the asteroid to the moon as 9500 litres of water. The tool that would allow to obtain this water from the asteroid surface would be a double-walled coring auger with heaters that would penetrate the subsurface of asteroid 2010DL capturing regolith that would be heated making it to sublime and capturing the water into a condenser. The time the spacecraft would need to obtain the 9500L of water would be approximately 317 days. The total mission time from leaving Earth's parking orbit to arrival to a lunar orbit with the water extracted from the asteroid will take at least 22 months.

This type of mission has great economic potential, as it would allow the production of water in space between 30% and 77% cheaper than it is nowadays, so this economic saving is sufficient to offset operational risk of the mission.

There is no legal framework for the steps explained in this paper, as this is a relatively new topic on which international legislation has only begun to take its first steps, but there is still a long way to go to establish an international legal framework to regulate the correct and free exploitation of asteroidal resources. Having a correct legal framework that limits and defines the correct methodologies for asteroid exploitation can lay the legal and juridical foundations for a new type of industry, the space mining industry, which could represent a before and after in the history of space exploration, as it could be able to provide enormous quantities of resources that could be used by human colonies on the lunar surface and in the rest of the solar system.

Future work

This thesis performs a preliminary technical analysis of exploiting asteroidal water for lunar development. The results obtained open a wide variety of options and improvements which can be studied in the future. Examples of these research topics are:

- Capture the asteroid. During the asteroid capture in this thesis a assumption was made to simplify the problem, and it was that the asteroid wasn't rotating on itself. But this assumption is not realistic, since this asteroid will most likely present a rotation. Therefore in future work it would be interesting to study how to approach a rotating asteroid, since to compensate for the rotation once the spacecraft wrapps the asteroid the center of mass of both bodies together must be known, however the exact shape of the asteroid will not be known until it is very close to it.

- Perform an analysis of how to send water from a lunar distant retrograde orbit to the lunar surface.

- To detail how the extraction of water directly from the asteroid would be performed, since this project has not defined the different processes that would be carried out in order to maximize the extraction of water from the asteroid.

- The use of the water that is being transported as propulsion system by heating it.

BIBLIOGRAPHY

[1] V. Badescu. *Asteroids: Prospective energy and material resources — Chapter 17 Near Earth Asteroids - Prospection, Orbit Modification, Mining and Habitation — Section 4 Orbital Maneuvers*, volume 9783642391682. Springer-Verlag Berlin Heidelberg, Polytechnic University of Bucharest, Candida Oancea Institute, 2014. xi, 9, 24

[2] Volker Hessel, Nam Nghiep Tran, Sanaz Orandi, Mahdieh Razi Asrami, Michael Goodsite, and Hung Nguyen. Continuous-flow extraction of adjacent metals—a disruptive economic window for in situ resource utilization of asteroids? *Angewandte Chemie International Edition*, 60(7):3368–3388, 2021. xi, 5, 10

[3] M.Ya. Marov, V.S. Avduevsky, E.L. Akim, T.M. Eneev, R.S. Kremnev, S.D. Kulikov, K.M. Pichkhadze, G.A. Popov, and G.N. Rogovsky. Phobos-grunt: Russian sample return mission. *Advances in Space Research*, 33(12):2276–2280, 2004. Mercury, Mars and Saturn. xi, 11

[4] DS Lauretta, SS Balram-Knutson, E Beshore, WV Boynton, C Drouet d'Aubigny, DN DellaGiustina, HL Enos, DR Golish, CW Hergenrother, ES Howell, et al. Osiris-rex: sample return from asteroid (101955) bennu. *Space Science Reviews*, 212(1):925–984, 2017. xi, 13

[5] Yuichi Tsuda, Makoto Yoshikawa, Masanao Abe, Hiroyuki Minamino, and Satoru Nakazawa. System design of the hayabusa 2—asteroid sample return mission to 1999 ju3. *Acta Astronautica*, 91:356–362, 2013. xi, 14

[6] Howard D. Curtis. Chapter 2 - the two-body problem. In Howard D. Curtis, editor, *Orbital Mechanics for Engineering Students (Third Edition)*, pages 59–144. Butterworth-Heinemann, Boston, third edition edition, 2014. xi, xiii, 30, 33

[7] Lucia Capdevila, Davide Guzzetti, and Kathleen Howell. Various transfer options from earth into distant retrograde orbits in the vicinity of the moon. *Advances in the Astronautical Sciences*, 152:3659–3678, 01 2014. xi, 31

[8] Davide Conte, Marilena Di Carlo, Koki Ho, David B. Spencer, and Massimiliano Vasile. Earth-mars transfers through moon distant retrograde orbits. *Acta Astronautica*, 143:372–379, 2018. xi, 31

[9] V Vendiola, K Zacny, P Morrison, A Wang, B Yaggi, A Hattori, and A Paz. Testing of the planetary volatiles extractor (pvex). In *Earth and Space 2018: Engineering for Extreme Environments*, pages 467–480. American Society of Civil Engineers Reston, VA, 2018. xi, 38

[10] P. D. Morrison, K. A. Zacny, V. R. Vendiola, and A. Paz. Results and lessons learned from testing of the planetary volatiles extractor (pvex) and related isru concepts. *LPI Contribution*, page 1, 2019. xi, xiii, 37, 38, 39, 40

[11] Harry Jones. The recent large reduction in space launch cost. 48th International Conference on Environmental Systems, 2018. xi, xiii, 44

[12] Howard D. Curtis. Chapter 3 - orbital position as a function of time. In Howard D. Curtis, editor, *Orbital Mechanics for Engineering Students (Third Edition)*, pages 145 – 186. Butterworth-Heinemann, Boston, third edition edition, 2014. xi, 57, 58, 60

[13] Howard D. Curtis. Chapter 8 - interplanetary trajectories. In Howard D. Curtis, editor, *Orbital Mechanics for Engineering Students (Third Edition)*, pages 405–457. Butterworth-Heinemann, Boston, third edition edition, 2014. xi, 26, 60

[14] Rui Xu, Pingyuan Cui, Dong Qiao, and Enjie Luan. Design and optimization of trajectory to near-earth asteroid for sample return mission using gravity assists. *Advances in Space Research*, 40(2):220 – 225, 2007. 8

[15] J. Cano, J. Cunill, A.J. Diaz, A. Golemis, S. Gupta, D. Innes, D. Maiden, K. March, H. Rael, J. Shawe, V. Sierra, A. Torrents, E.Z. Rossi, P. Machuca, R. Neves, and J.P.S. Cuartielles. Artemis: A complete mission architecture to bridge the gap between humanity and near-earth asteroids. In *Proceedings of the International Astronautical Congress, IAC*, volume 2018-October, 2018. 16

[16] B. Carry. Density of asteroids. *Planetary and Space Science*, 73(1):98–118, Dec 2012. 17

[17] G.A. Krasinsky, E.V. Pitjeva, M.V. Vasilyev, and E.I. Yagudina. Hidden mass in the asteroid belt. *Icarus*, 158(1):98–105, 2002. 17

[18] D. (1) Farnocchia, D.B. (1) Cioci, and A. (2) Milani. Robust resolution of kepler's equation in all eccentricity regimes. *Celestial Mechanics and Dynamical Astronomy*, 116(1):21–34, 2013. 21

[19] Dario Izzo. Revisiting lambert's problem. *CELESTIAL MECHANICS DYNAMICAL ASTRONOMY*, 121(1):1 – 15, 2015. 22

[20] J.P. Sanchez and C.R. McInnes. On the ballistic capture of asteroids for resource utilisation. In *62nd International Astronautical Congress 2011, IAC 2011*, volume 6, pages 4907–4921, Advanced Space Concepts Laboratory, University of Strathclyde, 2011. 28

[21] Xu Ming and Xu Shijie. Exploration of distant retrograde orbits around moon. *Acta Astronautica*, 65(5):853–860, 2009. 30

[22] Mark Joseph Sonter. The technical and economic feasibility of mining the near-earth asteroids. *Acta Astronautica*, 41(4-10):637–647, 1997. 43

[23] David Kornuta, Angel Abbud-Madrid, Jared Atkinson, Jonathan Barr, Gary Barnhard, Dallas Bienhoff, Brad Blair, Vanessa Clark, Justin Cyrus, Blair DeWitt, Chris Dreyer, Barry Finger, Jonathan Goff, Koki Ho, Laura Kelsey, Jim Keravala, Bernard Kutter, Philip Metzger, Laura Montgomery, Phillip Morrison, Clive Neal, Erica Otto, Gordon Roesler, Jim Schier, Brandon Seifert, George Sowers, Paul Spudis, Mark Sundahl, Kris Zacny, and Guangdong Zhu. Commercial lunar propellant architecture: A collaborative study of lunar propellant production. *REACH*, 13, 2019. 43, 45

[24] M.D. (1) Hogue, R.P. (2) Mueller, P.E. (3) Hintze, L. (4) Sibille, and D.J. (5) Rasky. Regolith-derived heat shield for planetary body entry and descent system with in situ fabrication. In *Earth and Space 2012 - Proceedings of the 13th ASCE Aerospace Division Conference and the 5th NASA/ASCE Workshop on Granular Materials in Space Exploration*, number Earth and Space 2012 - Proceedings of the 13th ASCE Aerospace Division Conference and the 5th NASA/ASCE Workshop on Granular Materials in Space Exploration, pages 526–536, (1)NASA, Electrostatics and Surface Physics Laboratory, NE-S-1, 2012. 44

[25] 3) Kieckhafer, A. (1 and 3) King, L.B. (1. Energetics of propellant options for high-power hall thrusters. *Journal of Propulsion and Power*, 23(1):21–26, 2007. 44

[26] Pablo Calla, Dan Fries, and Chris Welch. Asteroid mining with small spacecraft and its economic feasibility, 2019. 45

[27] S Neil Hosenball. The united nations committee on the peaceful uses of outer space: Past accomplishments and future challenges. *J. Space L.*, 7:95, 1979. 46

[28] US Congress. Hr 2262-us commercial space launch competitiveness act. *The United States of America*, 2015. 47

[29] Francis Lyall and Paul B Larsen. *Space law: a treatise*. Routledge, 2017. 47

APPENDICES

APPENDIX A. KEPLERIAN ORBITS

Kepler's Equation allows to know the position of an object in an orbit for a given time by relating this time with an angular position in a two-body Keplerian orbit. [12].

Starting with orbit equation from first law of Kepler:

$$r = \frac{h^2}{\mu}\frac{1}{1+e\cdot cos(\theta)} \tag{A.1}$$

This equation describes the path of the body m_2 around m_1 relative to m_1. Being 'e': Eccentricity, 'h': Angular momentum,'μ': Gravitational parameter,'θ': True anomaly.

Therefore time since periapsis:

$$h = r\cdot v_{\perp} = r\cdot(r\cdot\dot{\theta}) \tag{A.2}$$

So we can relate time and angular position:

$$h = \frac{h^4}{\mu^2}\frac{1}{(1+e\cdot cos(\theta))^2}\frac{d\theta}{dt} \tag{A.3}$$

$$\frac{\mu^2}{h^3}\int_{tp=0}^{t} dt = \int_0^{\theta}\frac{dx}{(1+e\cdot cos(x))^2} \tag{A.4}$$

⇒Thus for circular orbits (e=0):

$$\frac{\mu^2}{h^3}t = \theta \tag{A.5}$$

⇒ For elliptical orbit (0 < e < 1):

By second Kepler law we see that the period of the orbit only depend on the semi-major axis, describing third Kepler law:

$$T = 2\pi\sqrt{\frac{a^3}{\mu}} \tag{A.6}$$

Developing all these equations we get:

$$\frac{\mu^2}{h^3}t = \frac{1}{(1-e^2)^{\frac{3}{2}}}M \tag{A.7}$$

Where M is the **Mean anomaly M** that is related with the orbital period T as:

$$M = \frac{2\pi}{T}t = nt \tag{A.8}$$

From this schema we see the orbital parameters where E is the **Eccentric anomaly E** and is related with the **Mean anomaly M** by using Kepler equation:

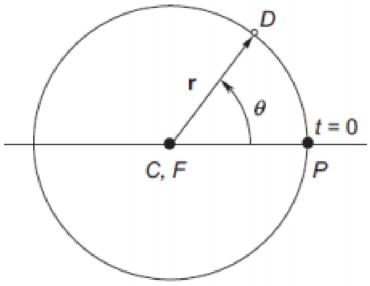

Figure A.1: Circular orbit. [12]

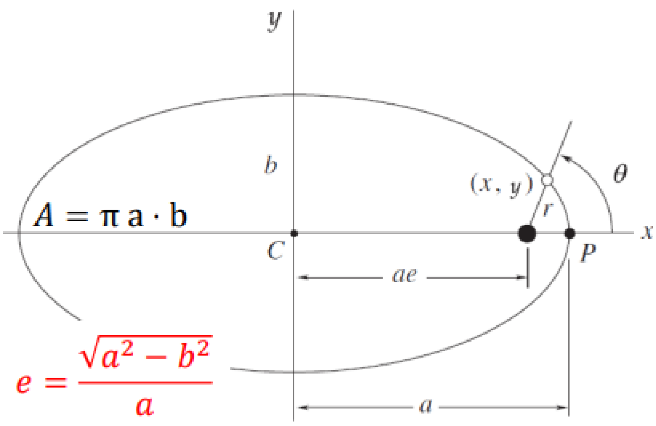

Figure A.2: Elliptical orbit. [12]

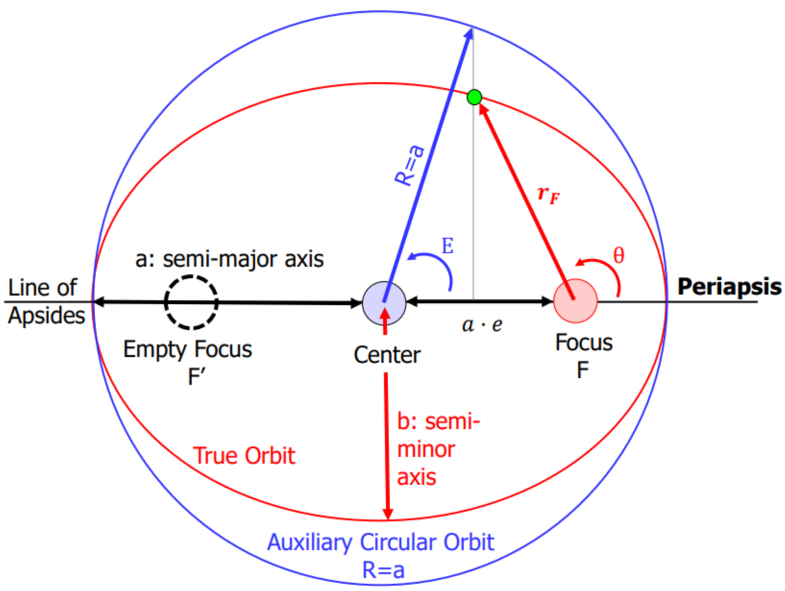

Figure A.3: Orbital Anomalies. [12]

$$M = E - sin(E) \tag{A.9}$$

And θ is the **True anomaly** θ that can be related with **eccentric anomaly E** by:

$$tan(\frac{E}{2}) = \sqrt{\frac{1-e}{1+e}} tan(\frac{\theta}{2}) \tag{A.10}$$

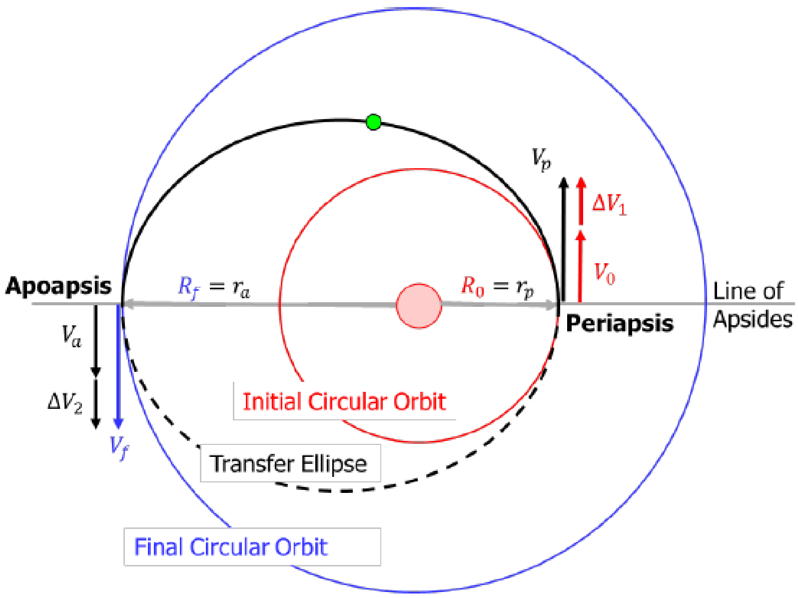

Figure A.4: Hohmann transfer scheme. [12]

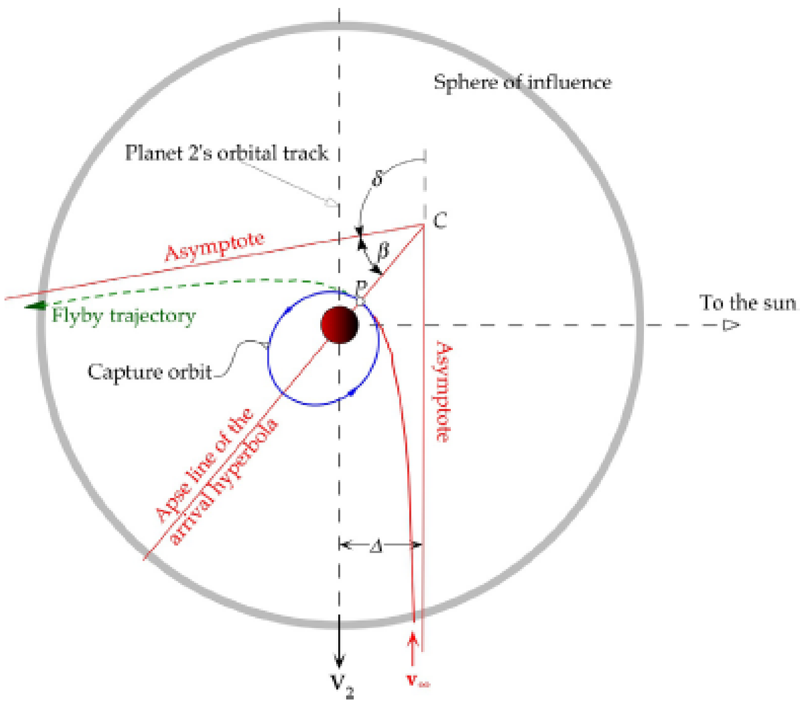

Figure A.5: Spacecraft entering to the Earth SOI sphere [13]

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APPENDIX B. PRELIMINARY ANALYSIS FOR TARGET ASTEROID SELECTION

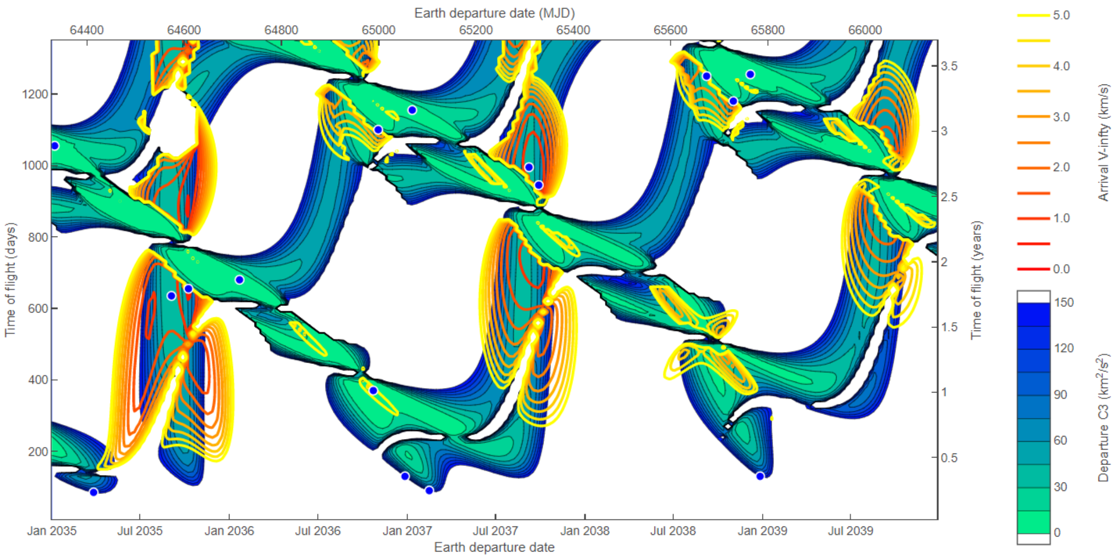

Figure B.1: Porkchop plot asteroid 2010 TN4 (From Jan 2035 to Jan 2040). Own work.

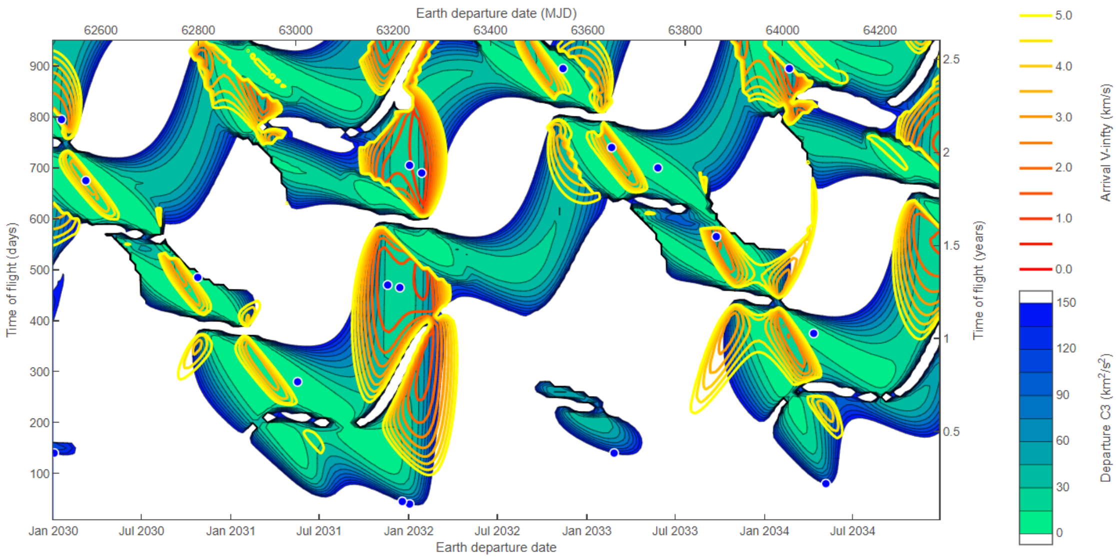

Figure B.2: Porkchop plot asteroid 2010 DL (From Jan 2030 to Jan 2035). Own work.

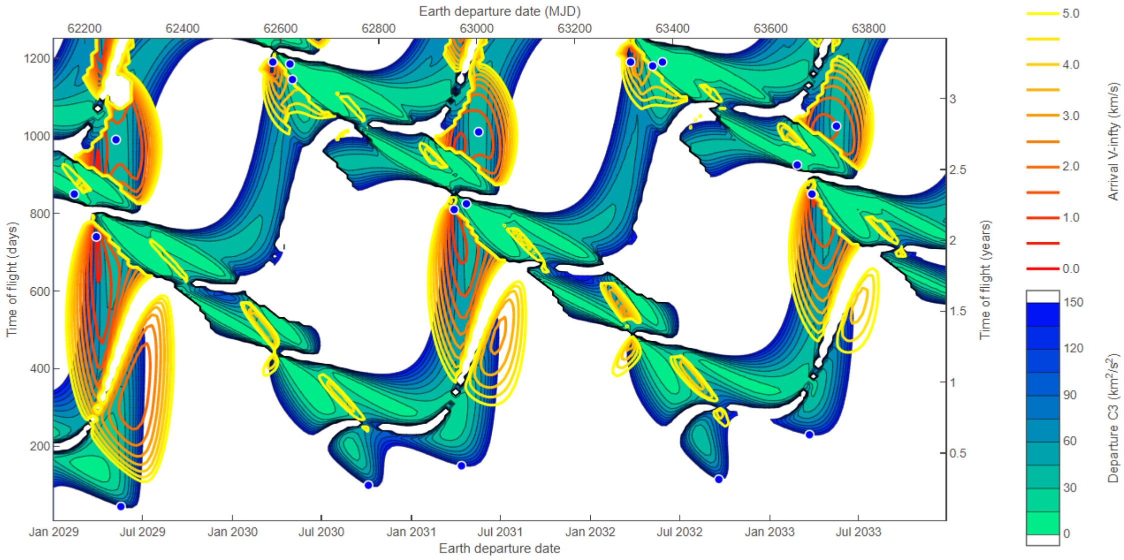

Figure B.3: Porkchop plot asteroid 2010 FW9 (From Jan 2029 to Jan 2034). Own work.

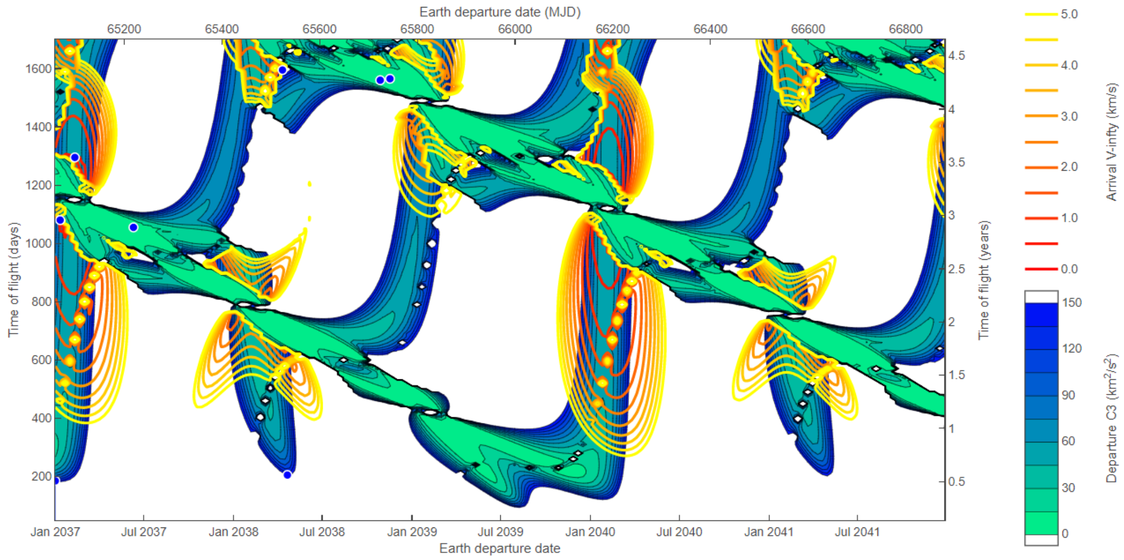

Figure B.4: Porkchop plot asteroid 2010 YD (From Jan 2037 to Jan 2042). Own work.

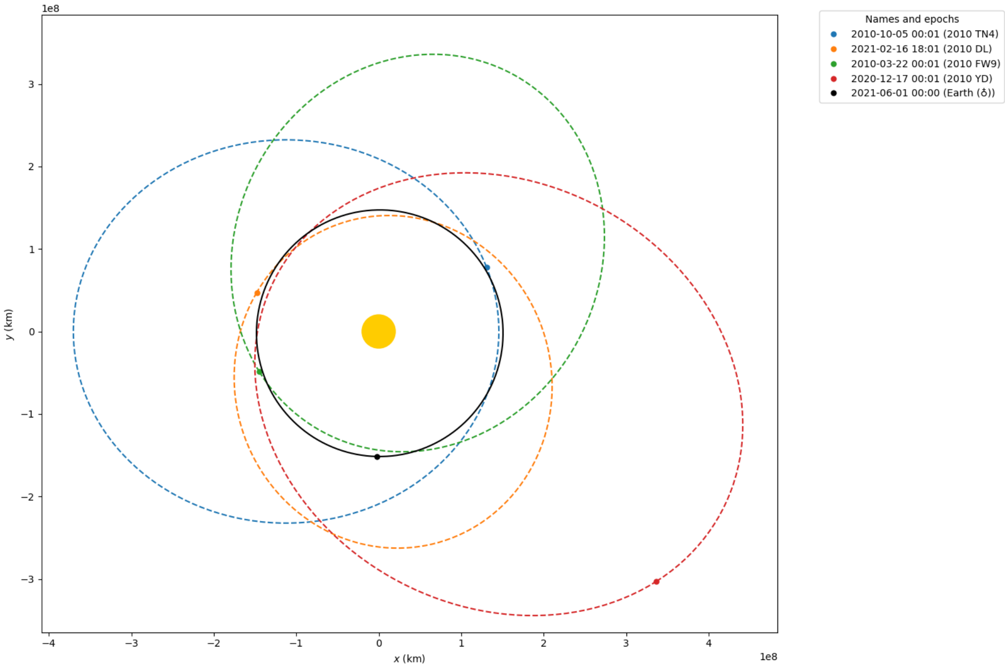

Figure B.5: Orbits of the 4 candidate NEAs. Own work.

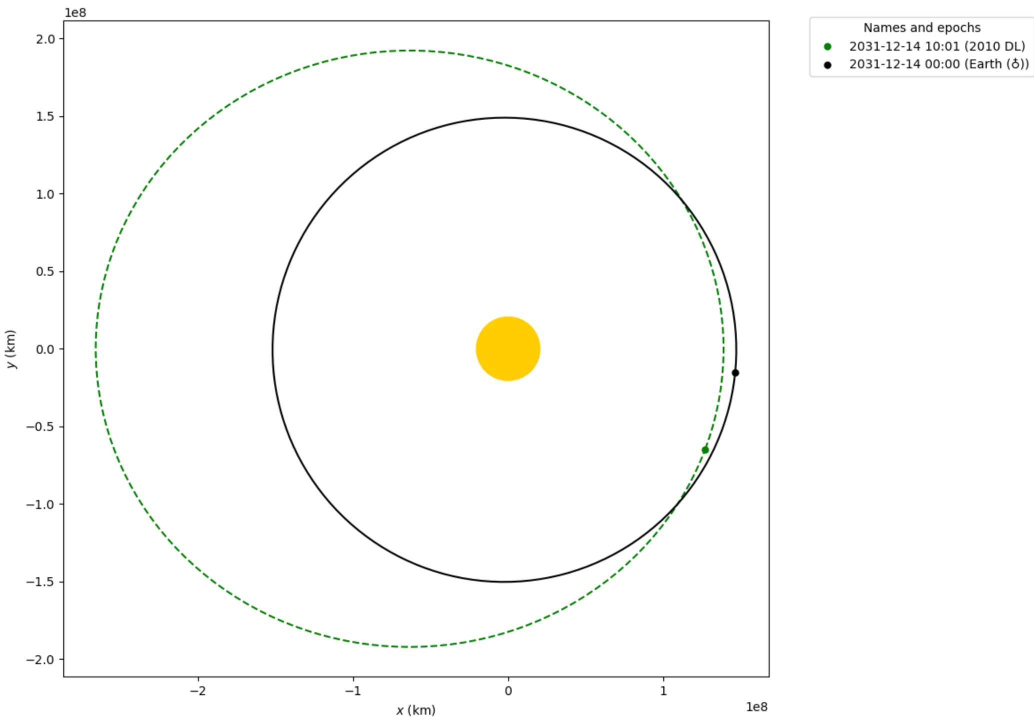

Figure B.6: Target asteroid and Earth orbit. Own work.

APPENDIX C. MISSION TRAJECTORIES

C.1. Results

C.1.1. Rendezvous to asteroid 2010 DL

ΔV_a= [-3752.69152788 592.59426083 99.61642593] m / s
ΔV_b=[-620.41890686 -219.59264112 1058.85500955] m / s

TotalΔV=5.047219498273518 km / s
Total tof = 40276867.33715284 s = 466.1674460318615 d

C.1.2. Hohmann transfer to Earth orbit

C.1.2.1. From perigee:

ΔV_a=[-7.63899242e-14 -6.23771068e+02 2.70437856e+02] m / s
ΔV_b=[7.47045150e-14 6.10008658e+02 -2.64471121e+02] m / s
TotalΔV_a=0.6798727669437402km/s
TotalΔV_b=0.6648725717593993 km/s
TotalΔV= 1.3447453383293286 km / s
Total tof = 29441498.640619297 s = 340.75808611827887 d

C.1.2.2. From apogee:

ΔV_a= [5.86540435e14 4.78946612e+02 2.07648769e+02]m/s
ΔV_b= [5.76607069e14 4.70835403e+02 2.04132130e+02]m/s

TotalΔV=1.0352049966139225 km / s
Total tof = 29195400.761534646 s = 337.9097310362806 days

C.1.3. Hyperbolic capture

ΔV_p=-0.22211747055775596 km/s
tof from SOI enter to capture orbit = 4.9178 days

C.1.4. Lunar orbit injection

ΔV_a = [-8.16691260e-03 -1.14674961e+02 4.97170850e+01] m / s
ΔV_b = [9.78286951e-03 1.37365273e+02 -5.95544215e+01] m / s

TotalΔV_a= 0.12498854062651987 km/s
TotalΔV_b= 0.14971956265721834 km/s

TotalΔV = 0.2747081026379342 km / s
Total tof = 2763676.941790654 s

APPENDIX D. CODE

The whole code used for this work is in the following github repository or go to the next url: https://github.com/CarlosIbCu/SpaceMiningResearchNEAs.git

It is a private repository, so if you wanna have access to it you must ask for it to the author Carlos Ibañez.