Preliminary Trajectory Design of a CubeSat Mission to a Near-Earth Object

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“The important thing is not to stop questioning. Curiosity has its own reason for existence”

Albert Einstein
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Resumo

Cometas e asteroides oferecem conhecimento sobre formação planetária, exploração de recursos espaciais, e técnicas de mitigação de colisão para defesa planetária. Por outro lado, espera-se que o paradigma dos CubeSats, que levou a uma redução em mais de uma ordem de grandeza dos custos de entrada para missões à órbita terrestre baixa, se estenda a missões interplanetárias na década de 2020. Este trabalho consiste numa análise preliminar de trajetórias para um CubeSat que parte de uma órbita de transferência geossíncrona rumo a um objeto próximo da Terra, sendo assumido que a duração da missão não excede 3 anos, e que a massa inicial do CubeSat é 16 kg. O objetivo é maximizar o rácio entre a massa final e inicial da sonda para cada um dos conceitos de trajetória analisados. A partida da Terra é modelada mediante múltiplas manobras de levantamento de apogeu, efetuadas por um estágio high-thrust, resultando numa trajetória de escape parabólica. A transferência interplanetária baseia-se no método patched conics, e é modelada através de: manobras impulsivas à partida da Terra e à chegada ao alvo, realizadas pelo estágio high-thrust, terminando num flyby ou rendezvous; ou uma transferência com propulsão low-thrust contínua, fornecida pelo CubeSat, terminando num rendezvous. A transferência com propulsão low-thrust implica uma massa inicial para a sonda menor e menos dependente da data de lançamento, logo, tende a oferecer menor custo e maior flexibilidade de lançamento.

Palavras-chave: CubeSat interplanetário, design preliminar de trajetória, objeto próximo da Terra, levantamento de apogeu, Problema de Lambert, trajetória low-thrust
Abstract

Comets and asteroids offer insight into planetary formation, space resources exploitation, and collision mitigation techniques for planetary defence. On the other hand, the CubeSat paradigm, which led to a reduction in entry-level costs of more than an order of magnitude for low Earth orbit missions, is expected to be extended to interplanetary missions in the 2020s. The present work performs a preliminary trajectory analysis for a CubeSat mission departing from a geosynchronous transfer orbit to a near-Earth object, assuming a maximum mission duration of 3 years and an initial CubeSat mass of 16 kg. The goal is maximising the spacecraft final-to-initial-mass ratio for each of the trajectory concepts assessed. The Earth departure is modelled as multiple finite apogee raising manoeuvres, enabled by a high-thrust stage, leading to a parabolic escape. The interplanetary transfer is based on the patched conics method, and is modelled via: Lambert Problem impulsive manoeuvres at the Earth departure and target arrival, performed by the high-thrust stage, ending in a flyby or rendezvous; or a continuous low-thrust transfer, powered by the CubeSat, ending in a rendezvous. The low-thrust transfer concerns a smaller and less launch date dependent initial spacecraft mass, thus it tends to offer a lower launch cost, more piggyback flight opportunities, and greater launch date flexibility.

Keywords: interplanetary CubeSat, preliminary trajectory design, near-Earth object, apogee raising, Lambert Problem, low-thrust trajectory
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Nomenclature

Greek symbols

\( \Delta E_T \) Eccentric anomaly span of the thrust activation arcs of all the ARs except the last one

\( \Delta E_{T_{\text{e}}} \) Eccentric anomaly span of the thrust activation arc of the last AR

\( \Delta t_{\text{ED}} \) ED duration

\( \Delta t_{\text{IT}} \) IT duration

\( \Delta t_{\text{max}} \) Maximum mission duration

\( \Delta t_{\text{NEO}} \) NEO operations duration

\( \Delta t_{\text{wait}} \) Waiting time from the end of the ED until the start of the IT

\( \Delta \nu \) Integral of the thrust acceleration over time including DVM

\( \Delta \nu_{\text{dep,arr}} \) Lambert Problem IT impulsive departure/arrival Delta-v

\( \lambda \) Magnetic latitude

\( \mu \) Gravitational parameter

\( \Omega \) Longitude of the ascending node

\( \omega \) Argument of periapsis

\( \rho \) Density

\( \sigma \) Standard deviation

\( g_0 \) Standard gravity acceleration: 9.806 65 m\(^2\)/s

\( R_E \) Earth equatorial radius: 6 378.137 km

\( \text{AU} \) Astronomical unit: 149 597 870 691 m

Roman symbols

\( a \) Orbital semimajor axis

\( D \) NEO diameter determined from the dimension of a sphere with similar volume
DVM  Delta-v margin

d_{E-\text{NEO}}  Earth-NEO distance

E  Eccentric anomaly OR Energy

e  Orbital eccentricity

\mathcal{E}  Specific energy

F  Particle flux

i  Orbital inclination

I_{sp}  Specific impulse

m_\ast  CubeSat scientific payload mass

m_0  SC mass at injection into the initial GTO

m_{CS}  CubeSat mass

m_f  CubeSat mass at arrival to the target NEO

m_p  Propellant mass

N_{IT}  Lambert Problem IT number of complete revolutions

n_{seg}  Number of thrusting segments for the continuous thrust IT

N_g  Number of generations concerning the evolution of the archipelago islands used in the continuous thrust IT optimisation

N_p  Population size for the archipelago islands used in the continuous thrust IT optimisation

OSM  Overscaling margin

P  Power

PM  Propellant mass margin

r  Orbital radius

\hat{\text{ED}}  ED normalised radiation dose

T  Thrust

\text{topo}_{ID}  Integer which identifies the archipelago topology used in the continuous thrust IT optimisation

\mathcal{t}_0  Instant at which the SC is injected into the initial GTO

\mathcal{t}_{\text{launch}}  Launch date

\mathcal{t}_{\text{arr,NEO}}  Date at which the SC arrives to the NEO
\( t_{\text{dep}_E} \) IT starting date

\( U \) Integer defined by the MPC ranging from 0 to 9 which quantifies the uncertainty in a perturbed orbital solution for a minor planet. 0 indicates a very small uncertainty, and 9 an extremely large uncertainty [1]

\( \vec{V}^I_c \) SC velocity relative to the Sun at the start of the Lambert Problem IT after the instantaneous departure manoeuvre

\( \vec{V}^f_{\text{arr}_\text{NEO}} \) SC velocity relative to the Sun at the end of the Lambert Problem IT before the instantaneous arrival manoeuvre

\( \vec{V}^f_{\text{arr}_\text{NEO}} \) NEO velocity relative to the Sun at the end of the Lambert Problem IT

\( \vec{V}_E \) Earth velocity relative to the Sun at the start of Lambert Problem IT

\( v_\infty \) Hyperbolic excess speed

\( v_{\text{FB}_\text{rel}} \) SC speed relative to the NEO at flyby

**Subscripts**

* Refers to a mass calculated for null overscaling margin (OSM = 0)

0 Initial GTO orbit

\( a \) Apoapsis

\( \text{arr} \) Arrival to the NEO and end of the IT

\( \text{CS} \) CubeSat

\( \text{dep} \) Deparure from Earth and start of the IT

\( E \) Earth

\( e \) Earth escape OR Electrons

\( p \) Periapsis OR Propellant OR Protons

range Refers to the set of values analysed for the variable having this subscript

\( \text{SC} \) Spacecraft

\( t \) HTS propellant tank

**Superscripts**

\( ' \) SC velocity after a certain manoeuvre OR mass for null propellant margin (PM = 0)

\( '' \) Delta-v for null Delta-v margin (DVM = 0)
Glossary

ACO  Ant Colony Optimisation
AR  Apogee Raising manoeuvre
COBYLA  NLopt Constrained Optimisation BY Linear Approximations
COTS  Commercially available Off-The-Shelf components
DART  Double Asteroid Redirection Test mission
ED  Earth Departure: from injection into the initial GTO until crossing the Earth SOA
ESA  European Space Agency
FEEP  Field Emission Electrical Propulsion
GACO  PyGMO extended Ant Colony Optimisation
GCR  Galactic Cosmic Radiation
GCRS  Geocentric Celestial Reference System
GEO  Geostationary (Equatorial) Orbit
GSO  GeoSynchronous Orbit
GTO  Geosynchronous Transfer Orbit
HTS  High-Thrust Stage
HYDROS-L  HTS stage used for the mission; scaled-up version of the HYDROS-M propulsion system
ICRF  International Celestial Reference Frame
IHS  PyGMO Improved Harmony Search
IT  Interplanetary Transfer
JPL  Jet Propulsion Laboratory
LEO  Low Earth Orbit
LTPS  Low-Thrust Propulsion System
MPC  Minor Planet Centre

MPCORB  Minor Planet Centre ORBit database

M-ARGO  Miniaturised Asteroid Remote Geophysical Observer

PDF  Probability Density Function

NASA  National Aeronautics and Space Administration

ND  Not Defined

NEA  Near-Earth Asteroid

NEO  Near-Earth Object

SC  SpaceCraft

SOA  Sphere Of Activity

SPE  Solar Particle Event

TC  Trajectory Concept

TRL  Technology Readiness Level
Chapter 1

Introduction

As the primitive building blocks of the solar system formation process, comets and asteroids offer insight into the chemical mixture from which the planets formed some 4.6 billion years ago. There has also been growing interest in the exploration of these bodies for profit, either in the context of an Earth or space economy. Moreover, studying them is relevant for developing collision mitigation techniques for planetary defence. Near-Earth objects (NEOs), asteroids or comets whose perihelion distance is smaller than 1.3 AU, are particularly attractive targets for space missions given their proximity to Earth.

On the other hand, CubeSats have reduced entry-level costs for space missions in low Earth orbit (LEO) by more than an order of magnitude, and the paradigm is expected to be extended to interplanetary missions as innovative miniaturised technologies are developed to overcome the severe technical challenges of deep-space.

The present work assesses potential trajectory concepts for a CubeSat mission from a geosynchronous transfer orbit (GTO) to a near-Earth object. There are two main objectives: firstly, for each concept, optimising trajectories for maximum ratio between final and initial spacecraft (SC) mass; secondly, comparing the optimised results in order to determine the advantages and disadvantages of each trajectory concept.

1.1 Near-Earth Objects

NEOs are asteroids or comets whose perihelion distance is smaller than 1.3 AU. Comets originally formed in the cold outer planetary system, while most of the rocky asteroids formed in the warmer inner solar system between the orbits of Mars and Jupiter. [2]

NEOs could provide insight into the solar system formation some 4.6 billion years ago, and the origin of life. In effect, it is believed that the giant outer planets (Jupiter, Saturn, Uranus and Neptune) and the inner planets (Mercury, Venus, Earth and Mars) formed from an agglomeration of billions of comets and asteroids, respectively. [2]

Due to their proximity to Earth, NEOs could also be exploited for raw materials, thus facilitating space exploration. Asteroids are rich in minerals and metals, which are useful to build space structures;
and comets in water and carbon-based molecules, which are essential to life, and could be used for propellant production [3]. In addition, although not presently cost effective, it has been argued that the high concentration of platinum group metals (platinum (Pt), rhodium (Rh), osmium (Os), iridium (Ir), palladium (Pd) and rhenium (Re)) in some asteroids represents a business opportunity from an Earth economy point of view [4, 5].

Finally, despite NEOs having brought to Earth the building blocks of life, they present a major threat to life itself in case of a high energy impact. Consequently, efforts have been mounted to increase detection and mitigation capabilities. NASA (National Aeronautics and Space Administration)’s proposed Near-Earth Object Surveillance Mission (NEOSM) [6] and ESA (European Space Agency)’s ground-based NEO Survey Telescope (NEOSTEL) [7] are examples of detection efforts. NASA’s Double Asteroid Redirection Test (DART) mission [8] and ESA’s Hera mission [9] are examples of mitigation efforts.

1.2 CubeSats

1.2.1 From Sputnik 1 to Satellite Constellations

At the beginning of the space age, all satellites were “small” due to limited launch vehicle capability [10]. The first artificial satellite, Sputnik 1, launched in 1957 by the Soviet Union, was a 58 cm metal sphere weighing approximately 84 kg [10]. Explorer 1, launched by the United States in 1958 after Sputnik 1 and 2, was 2.0 m tall and 15 cm in diameter, weighing nearly 14 kg [10]. As launch performance increased and the launch cost per unit mass decreased, scale gains supported the rise of larger satellites. This culminated in the average mass of commercial communications satellites peaking in 2012 at 4.4 t [11].


Commercially available standard spacecraft platforms were practically unknown until the emergence of microsatellites1 in the early 1980s. Such satellites adopted a radically different design approach to reduce costs — focusing on available and existing technologies by using properly qualified commercial off-the-shelf (COTS) components. This motivated a design-to-cost “small satellite mission philosophy” where there are strict cost and schedule constraints, often combined with a single mission objective in order to reduce complexity. The appearance of the CubeSat standard was the culmination of that approach.

1 Small spacecraft could be classified as picosatellites (0.1 kg–1 kg), nanosatellites (1 kg–10 kg) or microsatellites (10 kg–100 kg), amongst other classifications [20].
1.2.2 The CubeSat Standard

CubeSat is a standard for small spacecraft made of multiples of 1 U — a standard volume of $10 \text{ cm} \times 10 \text{ cm} \times 11.35 \text{ cm}$ — which use COTS for their electronics and structure. The standard [21] was conceived in 1999 by professors Jordi Puig-Suari (California Polytechnic State University, CalPoly) and Bob Twiggs (Stanford University) to allow graduate students to conceive, design, implement, test, and operate in space a complete spacecraft in a “reasonable” amount of time (duration of their studies) [10].

Regarding mass, CubeSats tend to be nanosatellites, but could also be picosatellites or microsatellites, and the standard originally predicted a maximum mass per unit of 1.33 kg [21]. The volume ranges from 0.25 U to 27 U, but only the 1 U, 1.5 U, 2 U, 3 U and 6 U form factors have been standardised [21, 22]. Due to their small mass and size, CubeSats are usually launched as a secondary payload or even from the International Space Station (ISS). Shared rides limit the achievable orbits, but led to a reduction in entry-level costs for LEO missions of more than an order of magnitude, what has been crucial for bringing new players into the space sector [23].

The six first CubeSats were launched on a Russian Eurockot in 2003, and, during the following decade, the majority of the activity concerned university projects. However, as performance limitations regarding size, available power, and down-link bandwidth were overcome, interest extended to companies, the military and space agencies. The boom started in 2013, during which 88 nanosatellites were launched, corresponding to 2.5 times more than in 2012, and totalling 65% of those launched from 1998 to 2012 [20]. The driving force has been the commercial Earth observation (EO) sector, launching massive satellite constellations of nano and microsatellites to reduce revisit times. The same revolution is now taking place in the field of telecommunications, where there are plans underway to deploy massive constellations of LEO satellites to provide worldwide Internet coverage (notably SpaceX’s Starlink [24]), Internet of Things (IoT) services, and machine-to-machine (M2M) communications. All this contributing to the evolution of the space segment from monolithic to distributed systems. [10]

The interested reader is encouraged to consult the nanosatellites database of [20].

1.2.3 CubeSats Systems Architectures and Launch Opportunities

ESA General Studies Programme on lunar and interplanetary CubeSat mission concepts involve either mother-daughter system architectures or a completely stand-alone system executing its own mission [23]. The mother-daughter architecture alleviates the technical challenges of propulsion, long-range communication and deep-space environment survivability because the host spacecraft provides power, accommodation, communications to Earth ground stations, and eventually local inter-satellite links between the deployed CubeSats. However, a stand-alone deep-space CubeSat (i.e. without a mothercraft), requiring a smaller volume, could rely on piggyback flight opportunities, and therefore tends to be cheaper.

However, there are spacecraft designated CubeSats whose volume is not a multiple of 1 U.

3Satellite revisit period is the time elapsed between observations of the same point on Earth by a satellite.
In terms of launch opportunities, there is a lack of dedicated small launchers that can cost-effectively inject nano or microsatellites into orbits with specific energy similar or larger than that of GTO [23]. Consequently, piggyback flight opportunities must be found, either on a launcher upper stage carrying a primary spacecraft, or on the primary spacecraft itself as part of its mission. A range of interplanetary missions which could provide piggyback flight opportunities is available in [23]. However, these opportunities are rare, and the constraints on deployment can significantly influence the Delta-v and transfer window needed to reach the final mission destination, and hence the feasibility of such a mission.

Although not considered for this work, it is worth noting that there are launchers dedicated to small satellites currently operational [25] and numerous under development [26–28], but these mainly target orbits lower than GTO. The exception is Rocket Lab’s Electron launcher which, together with the Photon cis-lunar spacecraft, plans to insert NASA’s Cislunar Autonomous Positioning System Technology Operations and Navigation Experiment (CAPSTONE) CubeSat into a lunar trajectory in 2021 [29].

### 1.3 Interplanetary CubeSat Missions

Achieving state-of-art astronomy via interplanetary CubeSat missions has become possible due to advances across all subsystems. Examples of some notable missions are described below.

NASA’s Mars Cube One (MarCO) [30], launched in 2018 as part of the InSight mission [31], is the only mission which have tested CubeSats in deep-space. The two 6 U 13.5 kg MarCO CubeSats served as communications relays during InSight’s landing on Mars, and costed 18.5 M$, about one order of magnitude cheaper than an equivalent conventional spacecraft.

Multiple CubeSat missions to NEOs have also been conceptualised. In the context of ESA’s Asteroid Impact Mission (AIM) [23], the concepts culminated in Hera mission, to be launched in 2024 and arrive to the binary asteroid system Didymos in 2026. Hera features two 6 U CubeSats, APEX (Asteroid Prospection Explorer) [32] and Juventas [33], whose main goals are characterising Dimorphos (smallest body of the system) and assessing the consequences of the DART impact4.

In addition, there are proposals for stand-alone CubeSats which, contrarily to MarCO, APEX and Juventas mother-daughter architectures, use their own propulsion to reach their final destination. A relevant example is ESA’s Miniaturised Asteroid Remote Geophysical Observer (M-ARGO) [34], whose launch is scheduled for 2023–2025. Considering a NEO rendezvous as a design driver, a bottom-up approach was followed to investigate how much Delta-v could be achieved within a 12 U CubeSat, while still accommodating a science payload of 1 U–2 U, and downlinking the science data back to Earth at a reasonable data rate [23].

M-ARGO is supposed to depart from the Sun-Earth L2 Lagrange point, and therefore requires a shared launch with an interplanetary mission. However, missions to the Moon could also be used as a launchpad for interplanetary CubeSats. For example, NASA’s Artemis 1 mission5, scheduled for launch...

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4NASA’s DART mission, to be launched in 2021 and collide in 2022 with Dimorphos, will try to demonstrate the kinetic impactor deflection technique [8].
5NASA’s Artemis program [35], enabled by the Space Launch System (SLS) and the Orion spacecraft, pretends to return humans to the Moon by 2024, and establish sustainable exploration of the satellite by 2028.
in 2021, will carry a secondary payload of 13 6 U CubeSats\(^6\) together with the Orion spacecraft. The CubeSats will conduct science and technology investigations, and 3 of them, NEA Scout [39], CU-E\(^5\) [40] and the Team Miles CubeSat [41], seek to reach interplanetary orbits.

Finally, there are CubeSats which piggyback on missions to GTO. For instance, the Mars Atmospheric Radiation Imaging Orbiter (MARIO) [42, 43] is a 32 kg 16 U CubeSat which aims to reach Mars from a super-synchronous orbit, using a high-thrust propulsion system for the Earth departure and a low-thrust one for the interplanetary transfer.

Examples of the development of innovative miniaturised technologies, and a summary with more detailed information about the aforementioned interplanetary CubeSat missions are presented in Appendix A.1.

### 1.4 Mission Architecture

#### 1.4.1 Spacecraft Architecture

The SC proposed here comprises two elements, a high-thrust stage (HTS) and a 12 U CubeSat (CS) without a high-thrust propulsion system. A CubeSat featuring both high-thrust and low-thrust, such as the MARIO CubeSat (see Section 1.3), could potentially be lighter than the proposed SC. However, the MARIO CubeSat volume of 16 U, for a payload of 1 U, suggests that a hybrid propulsion architecture would not satisfy the 12 U volume constraint introduced in Section 1.4.4. In fact, ESA’s 12 U M-ARGO CubeSat (see Section 1.3), only has room for 1 U of payload. Consequently, accommodating an additional high-thrust propulsion systems in the proposed CubeSat would hardly be feasible.

#### 1.4.2 Mission Phases and Trajectory Concepts

The mission could be divided into 5 phases: launch; Earth departure (ED), from injection into the initial GTO until exiting the Earth sphere of activity (SOA); waiting phase; interplanetary transfer (IT); and NEO operations. The IT starting date could be adjusted without the waiting phase by extending the ED, but that would imply crossing the Van Allen belts more times, and hence a significant increase in the radiation dose.

The ED is modelled as multiple finite apogee raising manoeuvres, enabled by the HTS, leading to a parabolic escape. The IT is based on the patched conics method, and is modelled via: Lambert Problem impulsive manoeuvres at the Earth departure and target arrival, performed by the HTS, ending in a flyby (TC1) or rendezvous (TC2); or a continuous low-thrust transfer, powered by the CubeSat, ending in a rendezvous (TC3).

TC1, TC2 and TC3 are the only 3 trajectory concepts (TCs) explored in this work. Low-thrust Earth departures were not analysed owing to their excessive Delta-v budget, duration and radiation dose [23, 43]. Hyperbolic escape trajectories were not studied since these would not be compatible with the waiting phase. A continuous thrust IT followed by a flyby was not considered because the maximum

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\(^6\) 7 CubeSats were developed by NASA [36], 3 by international space agencies [37] and 3 by finalists of the Cube Quest Challenge [38].
relative flyby speed of 100 m/s (see Section 1.4.4) represents a small fraction of the CubeSat low-thrust propulsion system Delta-v capacity (about 2.4 km/s from Table 3.5).

### 1.4.3 CubeSat Scientific Payload Estimation

The CubeSat scientific payload selection goes beyond the scope of this work, but its mass was estimated for the various TCs in order to determine the respective mass breakdowns. These are more useful for comparing different TCs than the final-to-initial-mass ratio. In effect, for the continuous thrust ITs (TC3), the final mass includes the CubeSat low-thrust propulsion system (LTPS) which enables the IT, and hence, for the same final-to-initial-mass ratio, the Lambert Problem ITs (TC1 and TC2) could have a higher payload ratio.

The payload mass was calculated from the payload mean density and volume as

\[ m_* = \rho_* V_* , \]  

(1.1)

where \( \rho_* \) was supposed equal to the CubeSat mean density of 1.33 kg/U introduced in Section 1.4.4. The payload volume was set to \( V_{*LP} = 9 \) U for the Lambert Problem ITs, and to \( V_{*CT} = 1 \) U for the continuous thrust ITs. The reasons are as follows:

- \( V_{*CT} = 1 \) U was assumed equal to that of ESA’s M-ARGO CubeSat \([34]\) (see Section 1.3) because it features the same LTPS, departs from a parking orbit around the Sun-Earth L2 point (which from an energy perspective resembles the waiting phase introduced in Section 1.4.2), and also targets a NEO.

- The volume which in the continuous thrust ITs is reserved for the LTPS could be used for payload in the Lambert Problem ITs, therefore \( V_{*LP} = V_{*CT} + V_{LTPS} \). Given the M-ARGO CubeSat LTPS volume of \( V_{LTPS} = 8 \) U \([44]\), it was assumed that \( V_{*LP} = 9 \) U.

Based on the aforementioned volumes and Eq. (1.1), the CubeSat payload mass was set to \( m_{*LP} \approx 12.0 \) kg for the Lambert Problem ITs, and to \( m_{*CT} \approx 1.3 \) kg for the continuous thrust ITs.

### 1.4.4 Global Constraints

The mission global constraints and respective motivations are:

- Earliest launch date 2023-01-01.0, and latest NEO arrival date 2031-01-01.0\(^7\): timely preparing a mission for an earlier launch would be difficult, and predictions concerning GSO satellites demand for the 2030s are not reliable; the NEO arrival occurs in the current decade.

- Maximum mission duration \( \Delta t_{max} = 3 \) a: maximum IT duration for the M-ARGO mission \([45]\) (see Appendix A.1) which supports a low operations cost.

- Maximum NEO periapsis radius \( r_{p_{max}} = 1.3 \) AU: NEO definition \([2]\).

\(^7\)In this work, the date YYYY-MM-DD.0 refers to the instant YYYY-MM-DDT00:00:00 UTC (Universal Time Coordinated).
Minimum NEO diameter $D_{\text{min}} = 15 \text{ m}$: for the same albedo, smaller diameter implies lower brightness, and therefore it is harder for the CubeSat to detect the NEO; NEOs of smaller diameter are categorised as “small” in the NEA Scout mission [46] (see Appendix A.1).

Maximum Minor Planet Centre (MPC) $U$ parameter $U_{\text{max}} = 6$: NEOs with higher orbit uncertainty were not considered in the NEA Scout mission [46].

Maximum speed of the CubeSat relative to the NEO at flyby $v_{\text{FB rel max}} = 100 \text{ m/s}$: speed taken from the NEA Scout mission [46] which allows for thorough characterisation of the target asteroid upon arrival.

Maximum Earth-NEO distance for the NEO operations $d_{\text{E-NEO max}} = 1 \text{ AU}$: distance taken from the NEA Scout mission [46] which results in a reasonable communication range for a CubeSat antenna.

Minimum duration for the NEO operations $\Delta t_{\text{NEO min}} = 0.5 \text{ a}$: duration of the M-ARGO mission NEO operations [34] which ensures a thorough analysis of the NEO and transmission of the data to Earth.

CubeSat size $V_{\text{CS}} = 12 \text{ U}$: limits the technological and financial effort such that a Portuguese company would be able to independently develop the CubeSat; provides sufficient room for scientific payload in TC3.

CubeSat mass $m_{\text{CS}} = 16 \text{ kg}$: results from assuming a mean density of 1.33 kg/U (equal to the maximum mass per unit of the original CubeSat standard); this density is small in comparison with those of the M-ARGO (21.6 kg / 12 U $\approx 1.8 \text{ kg/U}$ [34]), NEA Scout (12 kg / 6 U = 2 kg/U [46]) and MARIO (32 kg / 16 U = 2 kg/U [42]) interplanetary CubeSats.

Maximum initial SC mass $m_{0_{\text{max}}} = 100 \text{ kg}$: implies that the SC could still be classified as a microsatellite [20].

The above constraints were summarised in Table 1.1.

The duration of the waiting phase and respective constraint are

$$\Delta t_{\text{wait}} = t_{\text{dep,E}} - (t_0 + \Delta t_{\text{ED}}) \geq 0,$$

where $t_{\text{dep,E}}$ is the IT starting date, $\Delta t_{\text{ED}}$ is the ED duration, and the injection date is [47, 48]

$$t_0 \approx t_{\text{launch}} + 30 \text{ min},$$

where $t_{\text{launch}}$ is the launch date.

---

*Integer defined by the MPC ranging from 0 to 9 which quantifies the uncertainty in a perturbed orbital solution for a minor planet. 0 indicates a very small uncertainty, and 9 an extremely large uncertainty [1].

*21.6 kg / 12 U $\approx 1.8 \text{ kg/U}$ for the 6 solar panels configuration, and 22.3 kg / 12 U $\approx 1.9 \text{ kg/U}$ for the 8 panels version.
The total mission duration is

$$
\Delta t = (t_{arr \text{NEO}} - t_{\text{launch}}) + \Delta t_{\text{NEO}},
$$

(1.4)

where $t_{arr \text{NEO}}$ is the NEO arrival date, and $\Delta t_{\text{NEO}}$ is the duration of the NEO operations. The NEO arrival date is

$$
t_{arr \text{NEO}} = t_{\text{dep } E} + \Delta t_{\text{IT}},
$$

(1.5)

where $\Delta t_{\text{IT}}$ is the IT duration. The NEO operations duration is negligible for a flyby, and for a rendezvous is limited by either the maximum mission duration or the maximum Earth-NEO distance for the NEO operations. Consequently, the maximum NEO operations duration and respective constraint are

$$
\Delta t_{\text{NEO max}} = \min \{ \Delta t_{\text{max}} - (t_{arr \text{NEO}} - t_{\text{launch}}), t_{\text{break}} (t_{arr \text{NEO}}, \text{NEO}) - t_{arr \text{NEO}} \} \geq \Delta t_{\text{NEO min}},
$$

(1.6)

where $t_{\text{break}}$ is the date at which the Earth-NEO distance reaches the defined limit for the NEO operations ($d_{E-\text{NEO max}} = 1$ AU), and was determined with the ephemerides models introduced in Section 2.3.

Table 1.1: Mission global constraints.

<table>
<thead>
<tr>
<th>Description</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Earliest launch date (and NEO arrival date)</td>
<td>2023-01-01.0</td>
</tr>
<tr>
<td>Latest NEO arrival date (and launch date)</td>
<td>2031-01-01.0</td>
</tr>
<tr>
<td>Maximum mission duration ($\Delta t_{\text{max}}$)</td>
<td>3 a</td>
</tr>
<tr>
<td>Maximum NEO periapsis radius ($r_{p_{\text{max}}}$)</td>
<td>1.3 AU</td>
</tr>
<tr>
<td>Minimum NEO diameter ($D_{\text{min}}$)</td>
<td>15 m</td>
</tr>
<tr>
<td>Maximum MPC U’ parameter ($U_{\text{max}}$)</td>
<td>6</td>
</tr>
<tr>
<td>Maximum speed of the CubeSat relative to the NEO at flyby ($v_{FB_{\text{rel}}}_{\text{max}}$)</td>
<td>100 m/s</td>
</tr>
<tr>
<td>Maximum Earth-NEO distance for the NEO operations ($d_{E-\text{NEO max}}$)</td>
<td>1 AU</td>
</tr>
<tr>
<td>Minimum NEO operations duration ($\Delta t_{\text{NEO min}}$)</td>
<td>0.5 a</td>
</tr>
<tr>
<td>CubeSat size ($V_{CS}$)</td>
<td>12 U</td>
</tr>
<tr>
<td>CubeSat mass ($m_{CS}$)</td>
<td>16 kg</td>
</tr>
<tr>
<td>Maximum initial spacecraft mass ($m_{0_{\text{max}}}$)</td>
<td>100 kg</td>
</tr>
</tbody>
</table>

1.4.5 Geosynchronous Transfer Orbit

1.4.5.1 Overview

A satellite in a geosynchronous orbit (GSO) has a period of one sidereal day (23 h 56 min [49]), which is the Earth rotation period relative to the fixed stars. The circular GSO without inclination is named geostationary equatorial orbit (GEO), as for that case the satellite hovers approximately over one location on the Earth equator (at an altitude of $h_{\text{GEO}} = 35,786$ km [49]).

In order to reach a GSO (or GEO), satellites are frequently inserted into a geosynchronous (or
geostationary) transfer orbit (GTO). This highly elliptic orbit has a perigee altitude of a few hundred kilometres, and, if GEO is the goal, an apogee altitude equal to \( h_{\text{GEO}} \). A direct insertion is usually not performed because the increased payload capability to GTO (in comparison to GSO) offsets the additional mass of the propulsion system necessary for the circularising apogee burn \([50]\). In addition, GEO could be reached via sub-synchronous or super-synchronous transfer orbits \([51]\), whose periods are shorter and longer than a sidereal day, respectively.

It is worth noting that departing from a GTO is not energetically as favourable as from a lunar transfer or deep-space orbit. Moreover, even though deep space missions are not as frequent, there are numerous missions under development targeting the Moon during the 2020s \([52]\). Studying a departure from a lunar transfer orbit, and eventually a lunar gravity assist, would therefore be relevant. However, that goes beyond the scope of the present work.

In terms of demand, according to Shagun Sachdeva, one of the co-authors of the Northern Sky Research 2019 report \([53]\), “GEO commercial satcom satellite demand is expected to stabilise at an average of 13–15 satellite orders annually” during the 2020s. There is therefore potential for piggyback flight opportunities, with 5 CubeSat missions to GTO planned for 2020–2022 \([20]\) as of 2020-03-05.

### 1.4.5.2 Initial Orbit Definition

The definition of the initial GTO orbit is closely related to the piggyback flight opportunity, which in turn respects a certain launch provider and launch site. For this work, standard Ariane 5 and Ariane 6\(^{10}\) launches from the Kourou Europe Spaceport to GTO have been considered. The osculating parameters at injection\(^{11}\) in the geocentric celestial reference system\(^{12}\) (GCRS), retrieved from the Ariane 5 and Ariane 6 user manuals \([47, 48]\), are presented in Table 1.2 (equal values for both launchers).

<table>
<thead>
<tr>
<th>Element</th>
<th>Description</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>( Z_{p0} )</td>
<td>Perigee altitude</td>
<td>250 km</td>
</tr>
<tr>
<td>( Z_{a0} )</td>
<td>Apogee altitude</td>
<td>35,943 km(^{13})</td>
</tr>
<tr>
<td>( a_0 )</td>
<td>Semimajor axis</td>
<td>24,396 km</td>
</tr>
<tr>
<td>( e_0 )</td>
<td>Eccentricity</td>
<td>0.7283</td>
</tr>
<tr>
<td>( i_0 )</td>
<td>Inclination</td>
<td>6°</td>
</tr>
<tr>
<td>( \Omega_0 )</td>
<td>Longitude of the ascending node</td>
<td>GMST ((t_{\text{launch}} - 3\text{s}) + \text{long}_{\text{launch}} + 120°\ (\text{TBC}))</td>
</tr>
<tr>
<td>( \omega_0 )</td>
<td>Argument of perigee</td>
<td>178°</td>
</tr>
</tbody>
</table>

The semimajor axis was computed from \( a_0 = (r_{p0} + r_{a0}) / 2 \) for \( r_{p,a0} = R_E + Z_{p,a0} \), where \( R_E = 6,378.137 \text{ km} \) \([56]\) is the Earth equatorial radius. The eccentricity was calculated from Eq. (2.9). In

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\(^{10}\) Updated version of Ariane 5, whose key driver for development has been exploitation cost, with first flight planned for 2022 \([54]\).

\(^{11}\) The osculating parameters are determined from the position and velocity at injection, which is defined as the end of the upper stage shutdown \([48]\).

\(^{12}\) Earth centred inertial (ECI) coordinate system whose axes are parallel to those of the international celestial reference system (ICRS). GCRS closely approximates (to within about 0.02″) the system defined, at the epoch 2000-01-01.5 (epoch J2000), as having the origin at the Earth centre of mass, the ZZ axis normal to the mean equator, the XX axis pointing in the vernal equinox direction, and the YY axis such that the frame is right-handed. \([55]\)

\(^{13}\) \( Z_{a0} \) is not equal to the GEO radius to compensate for orbit perturbations.
the second to last row, GMST \( (t_{\text{launch}} - 3\ s) \) is the Greenwich Mean Sidereal Time at 3 s before lift-off, \( \text{long}_{\text{launch}} = -52.775^\circ \) [57] is the longitude of the Guiana Space Centre, and the 120° correspond to the increase in longitude of the ascending node during the ascent. The spacecraft was supposed to be inserted at the perigee \( (E_0 = 0) \), although the eccentric anomaly is not given in the user manuals. The longitude of the ascending node was also considered null \( (\Omega_0 = 0) \) because, for a parabolic escape (see Section 1.4.2) and disregarding perturbations (see Section 2.4.4), rotating the initial orbit about the Earth rotation axis does not impact on the ED duration and Delta-v.

Ariane launchers were considered for two main reasons. Firstly, as more than half of the communications satellites in orbit have been launched by the Ariane family into GTOs [48], numerous piggyback launch opportunities are expected. Secondly, Arianespace features a rideshare launch service solution for Ariane 6 (see 5.4.3 of [48]) which guarantees a safe and reliable separation of multiple small satellites (micro, mini or nanosatellites).

For the defined orbit, the maximum payload is 9.5 t for Ariane 5 [47], 4.5 t–5.0 t for Ariane 62, and 11.5 t for Ariane 64 [48]. This means the maximum initial SC mass of 100 kg (see Table 1.1) amounts to 0.9%–2.2% of the maximum payload capacity.

### 1.5 Software: PyKEP and PyGMO

This work makes use of PyKEP [58] and PyGMO\textsuperscript{14} [60], open-source scientific libraries developed at ESA to provide basic tools for astrodynamics research. Both were coded in C++ for computational speed, and exposed to Python for user-friendliness. PyKEP and PyGMO have been extensively validated by ESA’s Advanced Concepts Team, namely in the context of the Global Trajectory Optimisation Challenge (GTOC) [61].

Algorithmic efficiency is a main focus of PyKEP, and an efficient Keplerian propagator (pykep.propagate.lagrangian), Taylor-integrator (pykep.propagate.taylor), and solver for the multiple revolutions Lambert Problem (pykep.lambert_problem) are at the core of the library.

PyGMO (Python Parallel Global Multiobjective Optimizer) enables easy distribution of massive optimisation tasks over multiple CPUs (central processing units) according to the “generalised island model” paradigm [62], whose detailed description is available in [63].

\textsuperscript{14}Until the version 2.13.0, PyGMO and PaGMO [59], which is the C++ library on which PyGMO is based, were in the same project.
Chapter 2

Mission Design Concepts

2.1 Patched Conics

An object travelling through space is subject to the gravity field created by all other bodies. However, it is often the case that its trajectory is essentially defined by the influence of one single central body. This led Hohmann to devise the patched conics method in 1925. The method considers that a complex n-body problem could be divided into multiple 2-body problems (spacecraft and central body), and thus the trajectory is a sequence of patched conics [64].

The region where the gravity field of a body dominates is referred to as its sphere of activity (SOA) or sphere of influence. Although it is not a clear-cut concept, its radius is usually expressed via (11.8) of [64],

\[ r \approx \left(\frac{m}{M}\right)^{2/5} R. \]  

For a body orbiting the Sun, \( R \) is the semimajor axis of its orbit, and \( m \) and \( M \) are the orbiting body and Sun mass, respectively.

From Eq. (2.1), the Earth SOA radius is \( r_{SOA} \approx 9.25 \cdot 10^5 \text{km} \approx 0.006 \text{AU} \) [64]. Consequently, in preliminary trajectory design, from the interplanetary phase point of view, the Earth SOA is often approximated as the Earth centre of mass position. The same approximation applies to the SOA of NEOs as these are considerably less massive than Earth. In fact, Earth is about 6400 times\(^1\) heavier than Ceres, the largest and most massive known asteroid.

2.2 Lambert Problem

Posed in the 18th century by the Swiss polymath Johann Lambert, the Lambert Problem concerns the determination of an orbit which connects two points in space for a certain time of flight [66]. The problem is usually solved considering an unperturbed orbit in a given inverse-square-law centre of force [66], i.e.

\(^1\)Calculated from the ratio between the gravitational parameters of Earth (given by pykep.MU_EARTH) and Ceres (available in the JPL small body database [65]).
it is the boundary value problem for Keplerian dynamics.

The first solution was derived by the Franco-Italian mathematician and astronomer Joseph-Louis Lagrange, who demonstrated that the aforementioned time of flight depends on three quantities (Lambert Theorem): the semimajor axis of the connecting ellipse, the chord length between the two positions, and the semiperimeter of the connecting triangle between the focus and the two positions [50]. A detailed exposition could be found in [50, 67].

The demand for a robust and fast Lambert solver suitable for a wide range of initial conditions motivated the development of new algorithms during the Space era [68]. Lancaster and Blachard [69] presented a unified form of the Lambert Theorem, valid for elliptic, hyperbolic and parabolic orbits, based on an iterative process which only required the computation of one inverse trigonometric or hyperbolic function per iteration. Based on Lancaster and Blanchard’s approach, Gooding [66] designed a procedure which could achieve high precision with only three iterations for all geometries. In 2013, such procedure was considered the most accurate and fastest Lambert solver by Arora and Russel [70], who also claimed improvements on the original algorithm.

For the present work, Izzo’s implementation [68] has been selected. It builds upon Lancaster and Blachard work by iterating on the same variable, but, due to new found relations, proposes a new iteration scheme. The resulting algorithm is stated to be faster and numerically as accurate as Gooding’s method [68], and the respective Lambert solver concerns the PyKEP function lambert_problem.

### 2.3 Ephemerides

Ephemerides define the position and velocity of celestial bodies as a function of time (i.e. their trajectory), thus being essential for astronomy and spacecraft celestial navigation. For this work, the Jet Propulsion Laboratory (JPL) low-precision ephemerides were used, through the class pykep.planet.jpl.lp, to model the Earth motion; and the Minor Planet Centre Orbit database, by means of pykep.planet.mpcorb, to model the NEOs motion. The aforementioned classes have the method eph which returns the position and velocity vectors for a certain date.

The JPL low-precision ephemerides “are often used in observation scheduling, telescope pointing, and prediction of certain phenomena as well as in the planning and design of spacecraft missions” [71]. They provide the Keplerian elements of the eight major planets and Pluto via adjusting, for a certain time-interval, their positions and velocities to linear functions of time. Consequently, they are useful for preliminary trajectory searches where speed is prioritised over high accuracy, but are not adequate to design realistic trajectories during more advanced mission stages.

The MPC Orbit database (MPCORB) [72] contains orbital elements for minor planets, comets and natural satellites, and has been used for the M-ARGO mission (see Section 1.3) preliminary trajectory searches [34].

Both the JPL low-precision ephemerides and MPCORB are derived from the JPL (high-precision) Development Ephemerides (DE), which are given relative to the International Celestial Reference Frame (ICRF) 2.0 (described in [73]). The positions and velocities of the Sun, Moon and planets re-
result from a numerically integrated dynamical model of the Solar System, and are stored as Chebyshev polynomial coefficients fit in segments of 32 days [74]. The dynamical model accounts for multi-body effects of the Sun, Moon and planets, and includes perturbations from a set of asteroids representing 90% of the total mass of the main belt [74].

2.4 Earth Departure

2.4.1 Oberth Effect and Multi-burn Apogee-raising

The ED of all TCs features multiple apogee raising manoeuvres. This section illustrates the importance of such manoeuvres having in mind the Oberth effect.

The Oberth effect states that the specific energy $E$ of a spacecraft is more efficiently increased if the manoeuvre is performed at a higher speed $v$, i.e. at smaller radius $r$. This point is illustrated by the following relation, valid for an instantaneous Delta-v parallel to the pre-burn velocity:

$$\Delta E = \Delta \left( \frac{v^2}{2} - \frac{\mu}{r} \right) = \left( \frac{v + \Delta v}{2} \right)^2 - \frac{v^2}{2} = \left( v + \frac{1}{2} \Delta v \right) \Delta v,$$

where $\mu$ is the gravitational parameter of the central body. The specific energy increase $\Delta E$ is defined by the initial and target orbits, hence the Delta-v is minimised if the manoeuvre is applied at the point of highest speed, i.e. at the perigee. There the potential energy of the released propellant is minimum, and thus the SC energy increase is maximum.

As no propulsion system is able to generate infinite thrust, an instantaneous manoeuvre at the perigee is unrealistic, and there is a spatial distribution of the burn. The required Delta-v is therefore higher than that of the impulsive manoeuvre, and the respective difference is usually referred to as gravity losses. In order to minimise these losses, a multiple-burn apogee-raising strategy is preferable to a single-burn direct escape [50]. In fact, the shorter the manoeuvre, the closer to the perigee it could be applied, and hence the greater its efficiency. However, as the manoeuvre burning time tends to zero, the number of required burns tends to infinity, and so does the ED duration and radiation dose, thus the ED Delta-v budget cannot be arbitrarily close to the impulsive one.

2.4.2 Direct and Two-burn Escape

This section demonstrates that lowering the perigee is not beneficial for decreasing the ED Delta-v. For this purpose, the two following manoeuvres were studied:

- Direct escape: one single accelerating manoeuvre of magnitude $\Delta v_D$ is performed at the perigee of the initial GTO, the most advantageous point given the Oberth effect.

- Two-burn escape: firstly, a decelerating manoeuvre of magnitude $\Delta v_1$ is applied at the apogee of the initial GTO, and, secondly, an accelerating manoeuvre of magnitude $\Delta v_2$ is applied at the new perigee, both totalling $\Delta v_{2B} = \Delta v_1 + \Delta v_2$. Since $\Delta v_1$ lowers the perigee, the Oberth effect
concerning $\Delta v_2$ is more powerful than that of $\Delta v_D$ (i.e., for the same Delta-v, the specific energy increase linked to $\Delta v_2$ is larger than that respecting $\Delta v_D$).

Denoting with a prime the orbital speeds after the burns, the above Delta-v magnitudes are:

$$\Delta v_D = |v'_{p_0} - v_{p_0}| = |v (r_{p_0}, \xi_{dep}) - v (r_{p_0}, \xi_0)|;$$

(2.3)

$$\Delta v_1 = |v'_{a_0} - v_{a_0}| = |v (r_{a_0}, \xi_I) - v (r_{a_0}, \xi_0)|;$$

(2.4)

$$\Delta v_2 = |v'_{p_I} - v_{p_I}| = |v (r_{p_I}, \xi_{dep}) - v (r_{p_I}, \xi_I)|.$$  

(2.5)

The subscript 0 denotes the initial orbit, $I$ concerns the intermediate orbit obtained after the first burn in the two-burn escape, $\text{dep}$ respects the departure hyperbola, and $p$ and $a$ stand for perigee and apogee, respectively. The speed as a function of the orbital radius $r$ and specific energy $\xi$ is given by (2.15) of [64],

$$v (r, \xi) = \sqrt{\frac{2}{r} \left(\frac{\mu}{r} + \xi\right)}.$$  

(2.6)

The initial and intermediate orbits specific energies were calculated from (2.36) of [64],

$$\xi_{0,I} = -\frac{\mu E}{2 a_{0,I}};$$

(2.7)

and the escape specific energy from (2.40) of [64],

$$\xi_{\text{dep}} = \frac{v_{\infty,\text{dep}}^2}{2},$$

(2.8)

where $v_{\infty,\text{dep}}$ is the escape hyperbolic excess speed. Via (2.34) and (2.35) of [64],

$$r_{p,a} = a (1 \pm e),$$

(2.9)

the initial orbit perigee and apogee radii (the latter being equal to the apogee radius of the intermediate orbit) were written as a function of the initial orbit semimajor axis and eccentricity, and the intermediate orbit perigee radius in terms of the intermediate orbit semimajor axis and eccentricity. In addition, since the intermediate orbit semimajor axis is $(r_{a_0} + r_{p_I}) / 2$, from Eq. (2.9), it was expressed in terms of the initial orbit semimajor axis and eccentricity, and intermediate orbit eccentricity. Finally, for simplifying the Delta-v expressions, it was worth introducing the normalising speed

$$v_0 = \sqrt{\frac{\mu E}{a_0}},$$

(2.10)

which corresponds to the orbital speed of a SC in a circular orbit with radius equal to the semimajor axis of the initial orbit.
The above leads to the following Delta-v magnitudes:

$$\Delta v_D / v_0 = \sqrt{\frac{2}{1-e_0} + \left(\frac{v_{\infty\text{dep}}}{v_0}\right)^2} - \sqrt{\frac{1+e_0}{1-e_0}}$$  \hspace{1cm} (2.11)

$$\Delta v_1 / v_0 = \text{sgn} \left(e_I - e_0\right) \left(\frac{\sqrt{1-e_0} - \sqrt{1-e_I}}{\sqrt{1+e_0}}\right);$$  \hspace{1cm} (2.12)

$$\Delta v_2 / v_0 = \sqrt{\frac{2 (1+e_I)}{(1+e_0) (1-e_I)}} + \left(\frac{v_{\infty\text{dep}}}{v_0}\right)^2 \frac{1+e_I}{\sqrt{(1+e_0) (1-e_I)}}.$$  \hspace{1cm} (2.13)

The sign function was included in Eq. (2.12) so that it remains valid for a first accelerating burn which increases the perigee (resulting in an intermediate orbit with lower eccentricity than the initial one).

In order to compare the direct with the two-burn escape, the ratio between the two Delta-v budgets, $\Delta v_{2B} / \Delta v_D$, was plotted in Fig. 2.1 as a function of the ratio between the hyperbolic excess speed and reference speed ($v_{\infty\text{dep}} / v_0$), and the ratio between the eccentricities of the intermediate and initial orbits ($e_I / e_0$). The graph shows that raising the perigee is never advantageous ($e_I / e_0 < 1 \Rightarrow \Delta v_{2B} / \Delta v_D > 1$), as predicted by the Oberth Effect; and lowering it is beneficial for a hyperbolic excess speed larger than the reference speed ($e_I / e_0 > 1$, $v_{\infty\text{dep}} / v_0 > 1 \Rightarrow \Delta v_{2B} / \Delta v_D < 1$). For the semi-major axis of the initial GTO selected for this work (see Table 1.2), the reference speed is $v_0 \approx 4.0 \text{ km/s}$. As a consequence, lowering the perigee is not useful for the parabolic escape trajectories ($v_{\infty\text{dep}} = 0$) of the EDs.

In any case, lowering the perigee altitude for the Earth escape would hardly be advantageous as, even for a null perigee altitude (for which $e_I / e_0 \approx 1.01$) and hyperbolic excess speed $\sqrt{2\mu_S / 1\text{AU}} \approx 42.1 \text{ km/s}$ (for which $v_{\infty\text{dep}} / v_0 \approx 10.4$), roughly enough to escape the Solar System, the Delta-v gain would only be 0.4% (about 10 times smaller the Delta-v margin defined in Section 3.1). Furthermore, a low perigee would imply significant drag, thus a two-burn escape from GTO would not be worth it.

Figure 2.1: Comparison between the two-burn and direct escape Delta-v budgets.
2.4.3 Injection Accuracy

Target orbital elements of a given injection orbit could only be reached within a certain accuracy, and therefore the ED Delta-v has a stochastic component. This section demonstrates that the magnitude of such component is negligible.

For each excess hyperbolic speed, the stochastic component was assessed from the standard deviation of an ED Delta-v population. This was determined inputting into Eq. (2.11)² the following semimajor axis and eccentricity populations:

- Semimajor axis population following a normal distribution of mean \( a_0 \approx 24\,396\,\text{km} \) and standard deviation \( \sigma(a_0) = 40\,\text{km} \).

- Eccentricity population following a normal distribution of mean \( e_0 \approx 0.728\,3 \) and standard deviation \( \sigma(e_0) = 4.5 \cdot 10^{-4} \).

The mean values were given in Table 1.2, and the standard deviations were taken from the Ariane 5 and Ariane 6 user manuals [47, 48] (equal values for both launchers).

The ratio between the Delta-v standard deviation and the objective Delta-v (i.e. the Delta-v for the objective semimajor axis and eccentricity) as a function of the excess hyperbolic speed is presented in Fig. 2.2 for a population size (for each excess hyperbolic speed) of \( 10^8 \). As the ratio is of the order of 0.1 %, the stochastic component of the ED is covered by the Delta-v margin of 5 % (defined in Section 3.1).

Figure 2.2: Effect of the injection accuracy on the impulsive ED Delta-v assuming normal distributions of population size \( 10^8 \) for the semimajor axis and eccentricity.

²Eq. (2.11) does not account for non-instantaneous manoeuvres, but it is sufficient to determine the relevance of the injection accuracy.
2.4.4 Orbital Perturbations

Orbital perturbations should be taken into account for high-fidelity mission design, but were disregarded for this work. This section presents some considerations regarding the ED perturbations.

The only significant perturbations for an Earth bound trajectory are\(^3\) [76]: drag for altitudes lower than 430 km; Earth oblateness effect through the zonal coefficient \(J_2\) for altitudes between 430 km and \(36 \cdot 10^3\) km; and the Sun and Moon third body effects for altitudes higher than \(36 \cdot 10^3\) km.

Drag was not modelled because near the perigee, the point of highest drag, the (overestimated) drag-to-thrust ratio is of the order of 0.1%. This was calculated as follows:

- The atmospheric density \(\rho \approx 6.1 \cdot 10^{-11}\) kg/m\(^3\) for the initial GTO perigee altitude of \(Z_{p_{0}} = 250\) km (from Table 1.2) was obtained with the Matlab program from [3], which implements the 1976 United States standard atmosphere model [77].
- The perigee speed for a parabolic orbit, whose perigee altitude is equal to that of the initial GTO, was computed from Eq. (2.6) (for a null specific energy), what resulted into \(v \approx 11\) km/s.
- The SC diameter \(D_t \approx 45\) cm was taken from Fig. 3.1 for a Delta-v budget equal to the impulsive parabolic escape Delta-v of 770 m/s (given by Eq. (2.11)).
- The SC cross section area was estimated as \(S = \frac{\pi}{4} D_t^2 \approx 0.16\) m\(^2\), given the cylindrical shape of the propellant tank.
- The drag coefficient \(C_D = 4\) was retrieved from the highest value presented in Table 8-3 of [49], which presents typical values for LEO satellites.
- With the above assumptions, the drag is \(D = C_D \frac{1}{2} \rho v^2 S \approx 2.0\) mN; and, for the selected HTS (see Section 3.4.1), the thrust is 1.2 N. Consequently, the (overestimated) drag-to-thrust ratio is 0.2%.

The Earth oblateness effect was also disregarded. From [76], at the Earth surface, the magnitude of the \(J_2\) effect is about 3 orders of magnitude smaller than the effect of the Earth gravitational parameter \(\mu_E\). Moreover, since the force respecting \(J_2\) is proportional to \(r^{-4}\) [49], instead of \(r^{-2}\) as the \(\mu_E\) effect, the effect of \(J_2\) will be even smaller for higher altitudes.

The Sun and the Moon third body effects were also excluded from the ED trajectory. In fact, they cause an acceleration roughly 4 orders of magnitude smaller than that of the \(\mu_E\) effect at the GEO altitude of about \(36 \cdot 10^3\) km [76]. However, their importance increases with the orbital radius, and, near the Earth SOA boundary, the Sun gravity is relevant.

2.5 Radiation Environment

The radiation environment is an important limiting factor of the lifetime of operational SC subsystems. The following sections provide a means of assessing the radiation dose of the ED trajectories. It should nevertheless be noted that no constraint is defined for this work.

\(^3\)For high-fidelity trajectory design, solar radiation pressure, other third body effects (namely Jupiter gravity) and general relativity should also be considered [75].
2.5.1 Radiation Sources

Radiation hazards could be divided into trapped radiation, solar particle events (SPEs) and galactic cosmic rays (GCR) [49].

Trapped radiation consists of electrons and ions (predominantly protons) having energies greater than 30 keV. It is distributed non-uniformly within the magnetosphere, and the energetic electrons preferentially populate a pair of toroidal regions centred on the magnetic shells $L = 1.3$, on which the protons are also focussed, and $L = 5$ (see Eq. (2.14) for the definition of $L$). These two regions are often named the inner and outer Van Allen belts. [49]

SPEs occur in association with solar flares, and are rapid increases in the flux of particles, with energy from about 1 MeV to above 1 GeV, lasting from hours to days. Their frequency is linked to the 11 year sunspot cycle, peaking within a year or two of the sunspot maximum, and being much lower during the few years surrounding the sunspot minimum. [49]

GCR consists of particles which reach the vicinity of Earth from outside of the Solar System. It poses a serious hazard because a single particle could cause a malfunction in common electronic components such as random access memory, microprocessors or power transistors. [49]

GCR constraints were not assessed for a number of reasons. Firstly, due to the highly elliptical initial orbit, the shielding effect provided by Earth [78] (Earth body, atmosphere and magnetic field) only applies to a small region near the perigee, what decreases the dependence of the GCR radiation dose on the departure trajectory. Secondly, the effects of GCR are usually attenuated by better parts, improved shielding, and process redundancy as described in Table 8-8 of [49]; and the maximum mission duration constraint limits the probability of single-event phenomena\(^4\). Thirdly, while performing a mission during a solar maximum would decrease the effects of GCR due to the heliosphere shielding effect [79] (“solar modulation”), the increase in trapped radiation due to more frequent SPEs would imply a larger ED radiation dose. For these reasons, only trapped radiation was studied.

2.5.2 Radiation Flux Models

The trapped electron and proton energy fluxes were modelled using the curves of Figure 13 of [80], which were reproduced\(^5\) in Fig. 2.3. $L$ respects the magnetic L-shell (surface generated by rotating a magnetic field line around the Earth best-fitting dipole axis [49]), and approximately satisfies equation (8-3) of [49],

$$ r = R_E L \cos^2 \lambda, $$

(2.14)

where $r$ is the orbital radius, and $\lambda$ is the magnetic latitude\(^6\).

\(^4\)The effects of GCR are termed single-event phenomena (SEP) if caused by a single passing particle [49].

\(^5\)Firstly, the electrons and protons curves reference points of Figure 13 of [80] were determined using Microsoft Paint by extracting the respective pixel coordinates, and subsequently converting them (via the coordinates of the origin, the right-most point of the horizontal axis, and the upper-most point of the vertical axis) to the appropriate units. Secondly, the electrons curve was drawn with two cubic splines which share the 6th curve point, and the protons curve with one linear segment and one cubic spline which share the 2nd curve point, considering “not-a-knot” boundary conditions (third derivative continuous in the second and second to last nodes) for the 3 cubic splines.

\(^6\)Magnetic latitude is a parameter analogous to geographic latitude, but defined relative to the geomagnetic dipole of the International Geomagnetic Reference Field (IGRF) [81].
The curves of Figure 13 of [80], on which Fig. 2.3 was based, were derived from the NASA CCMC (Community Coordinated Modeling Center) AE-8 and AP-8 models [82], whose online tool is available in [83]. AE-8 and AP-8 are particularly suitable for the wide-ranging trajectory of this work because they are the only models covering the full spatial and spectral range of the radiation belts\footnote{Alternatively, a combination of different models would be required to cover the full spatial and spectral range (see Table 1 of [84]).} [84], although for some part through extrapolation. Their particle fluxes are stored as functions of the energy (in the range $[0.04; 7]$ MeV for electrons, and $[0.10; 400]$ MeV for protons), magnetic L-shell, and magnetic field intensity normalised to the field line minimum at the magnetic equator [82].

Both AE-8 and AP-8 are subdivided into two models: AE8MIN and AE8MAX represent the time-averaged electron flux during solar minimum and solar maximum, while AP8MIN and AP8MAX describe the respective proton fluxes. Insight into the model choice (MIN or MAX) and assumptions regarding the normalised magnetic field intensity for the points of Fig. 2.3 are not specified in [80]. However, so as not to underestimate the radiation dose, selecting AE8MAX and AP8MIN as well as assuming unitary normalised magnetic field intensity would have been reasonable. This because the AE8MAX flux is larger than that of AE8MIN, the AP8MIN flux is larger than that of AP8MAX\footnote{The protons flux at solar minimum is larger because, for greater solar activity, the atmospheric density increases, what leads to more trapped protons colliding with the atmosphere at low altitude, and therefore to a decrease in the protons flux [49].}, and both electron and proton fluxes are higher for smaller normalised magnetic field strength [83].

### 2.5.3 Radiation Dose Estimation

The effect of trapped radiation for a given ED trajectory is quantified through the respective radiation (absorbed) dose\footnote{The SI unit is the “gray” (1 Gy = 1 J/kg).}, which is the amount of energy deposited per unit mass. This was assessed via the
following adimensional quantity used in [80]:

\[
\tilde{\text{RD}}_{e,p} = \int_{E_{\text{range}e,p}}^{E_D} F_{e,p}(E_{\text{range}e,p}, L, \frac{B}{B_0}) \, dt;
\]

(2.15)

\[
\text{RD}_{0e,p} = \int_{0}^{F_{e,p}(E_{\text{range}e,p}, r, \frac{B}{B_0})} dt.
\]

(2.16)

The subscripts \(e\) and \(p\) stand for electrons and protons, respectively, \(\tilde{\text{RD}}\) is the normalised ED radiation dose, the integral in the numerator is a measure of the ED radiation dose, the denominator \(\text{RD}_{0}\) quantifies the radiation dose for one complete revolution in the initial GTO, and \(F\) is the energy omnidirectional flux for the energy range \(E_{\text{range}}\). From a spacecraft design perspective, only electrons and protons with energy larger than 0.5 MeV and 0.1 GeV, respectively, are relevant [80], and therefore only those were considered in Fig. 2.3, and Eqs. (2.15) and (2.16). The magnetic L-shell for the electrons and protons was computed from Eq. (2.14), setting the magnetic latitude to \(\lambda_e = 62^\circ\) and \(\lambda_p = 38^\circ\), respectively, for the reasons advanced in the following section. The integrals were calculated with the trapezoidal rule (see, for example, [85]) considering a time step of 10 s.

The dimensional radiation doses could be obtained from the shield thickness \((l_{sh})\) and density \((\rho_{sh})\) through

\[
\text{RD}_{e,p} = \frac{\tilde{\text{RD}}_{e,p} \text{RD}_{0e,p}}{l_{sh} \rho_{sh}}.
\]

(2.17)

**Magnetic Latitude Estimation**

The magnetic latitude could be computed with the aacgmv2 [86] toolbox from the date, latitude, longitude and altitude\(^\text{10}\) of a given trajectory point. However, applying this procedure to the whole ED would require a long computation time. As a result, the procedure was bypassed by assuming constant magnetic latitude.

In the study on which the definition of the normalised radiation doses were based [80], the magnetic latitude is supposed null, but that would not be optimal for this work. By comparing the normalised radiation dose for a constant magnetic latitude in the range \([0;90]^\circ\) (step \(1^\circ\)) with that of a variable magnetic latitude computed with the aforementioned procedure, it was concluded that \(\lambda_e = 62^\circ\) and \(\lambda_p = 38^\circ\) are the optimum magnetic latitudes which minimise the error in the electrons and protons normalised radiation doses, respectively. The optimum constant magnetic latitudes, as well as the respective electrons and protons normalised radiation doses absolute and relative errors \((\Delta \tilde{\text{RD}} = \tilde{\text{RD}}_{\lambda=ct} - \tilde{\text{RD}}_{\lambda\not=ct})\), are presented in Table 2.1 for various eccentric anomaly spans of the pre-escape perigee burns \((\Delta E_T\) defined in Section 4.1), respecting a launch at 2023-01-01.0. Excluding the protons normalised radiation dose for large values of \(\Delta E_T\), the relative error for the optimum constant magnetic latitudes does not exceed 12%. The results do not depend on the longitude of ascending node since the considered Earth magnetic field model is axisymmetric.

It was also concluded that the dependence of the optimum magnetic latitudes on the launch date is

\(^{10}\text{The latitude, longitude and altitude could be determined from the GCRS coordinates with the Astropy [87] toolbox.}\)
negligible. For example, for a launch between 2016-01-01.0 and 2024-01-01.0\textsuperscript{11} with a step of 1 a, for $\Delta E_T = 90^\circ$, there was no variation of the optimum magnetic latitudes, relative to the values for a launch at 2023-01-01.0, within a precision of 1°. Furthermore, the absolute and relative errors did not exceed those of Table 2.1.

The gains in computation time by considering constant magnetic latitudes are significant, especially considering the large number of EDs whose radiation dose should be evaluated\textsuperscript{12}. For instance, for $\Delta E_T = 30^\circ$, it was of the order of 10 s for constant magnetic latitude, and of the order of 100 min for variable magnetic latitude (excluding the time to determine the input ED trajectory, i.e. only the time to obtain the normalised radiation doses).

Table 2.1: Optimum constant magnetic latitudes and respective absolute and relative errors for various eccentric anomaly spans of the pre-escape perigee burns, respecting a launch at 2023-01-01.0.

<table>
<thead>
<tr>
<th>$\Delta E_T$ ($^\circ$)</th>
<th>5</th>
<th>30</th>
<th>60</th>
<th>90</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\lambda_e$ ($^\circ$)</td>
<td>62</td>
<td>62</td>
<td>62</td>
<td>62</td>
</tr>
<tr>
<td>$</td>
<td>\Delta RD_e</td>
<td>$</td>
<td>0.037</td>
<td>0.072</td>
</tr>
<tr>
<td>$</td>
<td>\Delta RD_e</td>
<td>/RD_e$ (%)</td>
<td>0.48</td>
<td>6.0</td>
</tr>
<tr>
<td>$\lambda_p$ ($^\circ$)</td>
<td>38</td>
<td>38</td>
<td>38</td>
<td>37</td>
</tr>
<tr>
<td>$</td>
<td>\Delta RD_p</td>
<td>$</td>
<td>3.6</td>
<td>0.36</td>
</tr>
<tr>
<td>$</td>
<td>\Delta RD_p</td>
<td>/RD_p$ (%)</td>
<td>12</td>
<td>7.8</td>
</tr>
</tbody>
</table>

2.5.4 Normalised Radiation Dose for Circular Orbits

So as to facilitate the interpretation of the normalised radiation dose values, these were calculated as a function of the mission duration $\Delta t_c$ in a circular orbit of altitude $h_{ct}$. The results for the electrons and protons are displayed in Fig. 2.4 and Fig. 2.5, respectively. The variation of the normalised radiation doses with the orbit altitude is more significant than with the mission duration.

\textsuperscript{11}The aacgmv2 toolbox does not allow to study dates later than 2024, but the analysed range spans 8 years as the launch date range of the present mission (see Section 1.4.4).

\textsuperscript{12}The EDs were analysed for 86 values of $\Delta E_T$ (see Section 4.1).
2.6 NEO Population

2.6.1 NEO Diameters

Knowing the NEO population diameter, intended as that of a sphere whose volume is equal to that of the NEO, is relevant as it impacts directly on the NEO size constraint defined in Section 1.4.4 ($D \geq 15$ m). For that purpose, the following databases were considered: the MPCORB database [72], the ESA Space Situational Awareness website database [88], and the JPL Small-Body Database [65]. The ESA catalogue has been selected because it provides individualised diameter estimates, from a range of literature sources and astronomical surveys, for 99.9% of the NEOs.
The MPCORB database provides values for the NEOs absolute magnitudes\(^{13}\) \((H)\), but not for their geometric albedos\(^{14}\) \((p)\). These could have been estimated to obtain the NEO diameters from [89]:

\[
D = 1.329 \cdot 10^6 p^{-\frac{1}{2}} 10^{-0.2 H}.
\]

(2.18)

Nonetheless, the estimates would not be individualised as those of the ESA catalogue. For example, the MPC magnitude-diameter conversion table [90] groups the NEOs into only 3 albedo values (0.05, 0.25 and 0.50).

The JPL estimates are individualised, but are only presented for 3.8% of the NEOs\(^{15}\). Consequently, it is not as wide-ranging as the ESA catalogue.

It is worth pointing out that ESA and JPL could provide significantly different diameters for the same NEO. For instance, for 433 Eros, the ESA database gives a diameter of 23.3 km, which is 38% larger than the JPL's 16.84 km. This is because, although both ESA and JPL use Eq. (2.18) to estimate the diameter whenever a more accurate result is not available in the literature [91, 92], the values attributed to the absolute magnitude and geometric albedo vary. In practice, the target NEO must have its diameter carefully analysed such that uncertainty is decreased to a reasonable level. Nevertheless, the values provided by ESA have been considered correct for the purpose of verifying the minimum diameter constraint.

2.6.2 NEO Population Definition

This section deals with the NEO population definition for the Lambert Problem ITs, continuous thrust ITs, and TCs global optimisation.

The NEO population definition for the Lambert Problem ITs is described in Table 2.2. The 1st row condition is reasonable because the “MPC is the single worldwide location for receipt and distribution of positional measurements of minor planets, comets and outer irregular natural satellites of the major planets” [93], the 2nd is the definition of NEO, the 3rd results from having selected the ESA database as source of the NEO diameters (see Section 2.6.1), and the 4th respects the diameter and MPC \(U\) parameter constraints introduced in Section 1.4.4. Only a minority (2.3%) of the MPCORB bodies are NEOs, almost all (99.9%) MPCORB NEOs have their diameter defined on the ESA database, and of those about two thirds (68.2%) satisfy the diameter and MPC \(U\) parameter conditions.

Optimising continuous thrust ITs for all the 15,131 NEOs considered for the Lambert Problem ITs (last row of Table 2.2) would require an unreasonably long computation time. Consequently, continuous thrust ITs were only optimised for NEOs for which there is at least one Lambert Problem IT satisfying the two

\(^{13}\)Absolute magnitude is the visual magnitude (reverse logarithmic measure of brightness) of an asteroid at zero phase angle, and at unit heliocentric and geocentric distances.

\(^{14}\)Geometric albedo is the ratio between the brightness of a body at zero phase angle and the brightness of a perfectly diffusing disk at the same position with the same apparent size as the body.

\(^{15}\)The difference is not explained by the total number of NEOs, which only differed in 1%.
The Lambert Problem IT, including the calculations of the departure Delta-v ($\Delta v_{dep}$) and arrival Delta-v for a rendezvous ($\Delta v_{arrRDV}$), is described in Section 4.2. Eq. (2.20) limits the Earth-NEO distance at the end of the IT, and results from the Earth-NEO distance constraint introduced in Section 1.4.4.

Table 2.3 contains the name, orbital elements at the epoch 2020-05-31.0 in the ICRF 2.0, diameter and MPC U parameter of the NEOs which satisfy the above constraints\textsuperscript{16}. The NEOs semimajor axis ranges from 0.94 AU to 1.10 AU, and the maximum inclination is 2.7°.

Fig. 2.6 illustrates the effect of changing the Delta-v constraint of Eq. (2.19) on the number of NEOs to be analysed (in absolute terms on the left axis, and as a fraction of the total on the right axis). The selected Delta-v limit of 2.5 km/s (red cross) implies a decrease in the NEO population size from $1.5 \cdot 10^4$ to 14 (0.093%).

In addition, only the Lambert Problem ITs satisfying Eqs. (2.19) and (2.20) were considered for the TCs global optimisation (see Section 4.4). This implies NEO population coherence amongst all the TCs global optimisation, and avoids a long global optimisation computation time for the Lambert Problem IT TCs.

Fig. 2.7 illustrates the effect of changing the Delta-v constraint of Eq. (2.19) on the number of valid Lambert Problem ITs for the NEOs of Table 2.3. The selected Delta-v limit of 2.5 km/s (red cross) implies a decrease in ITs from $5.3 \cdot 10^5$ to $6.0 \cdot 10^2$ (1.1%).

Table 2.2: NEO population definition for the Lambert Problem ITs.

<table>
<thead>
<tr>
<th>Constraints</th>
<th>Number of NEOs (fraction of the number above)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Body is on the MPCORB database</td>
<td>949,512</td>
</tr>
<tr>
<td>Body is a NEO (perihelion no larger than 1.3 AU)</td>
<td>22,217 (2.3 %)</td>
</tr>
<tr>
<td>NEO diameter is defined on the ESA database</td>
<td>22,194 (99.9 %)</td>
</tr>
<tr>
<td>NEO satisfies the diameter ($D \geq 15 \text{ m}$) and MPC U parameter ($U \leq 6$) constraints of Table 1.1</td>
<td>15,131 (68.2 %)</td>
</tr>
</tbody>
</table>

\textsuperscript{16}2014 WA366 and 2017 BN93 do not respect Eqs. (2.19) and (2.20) for ITs without complete revolutions.

\textsuperscript{17}NEO designator according to the ESA asteroid database [88].
Table 2.3: Name, orbital elements at the epoch $t_0 = 2020-05-31.0$ in the ICRF 2.0, diameter and MPC $U$ parameter of the NEOs analysed for the continuous thrust ITs and TCs global optimisation.

<table>
<thead>
<tr>
<th>NEO$^{17}$</th>
<th>$a$ (AU)</th>
<th>$e$</th>
<th>$i$ (°)</th>
<th>$\Omega$ (°)</th>
<th>$\omega$ (°)</th>
<th>$M_0$ (°)</th>
<th>$D$ (m)</th>
<th>$U$</th>
</tr>
</thead>
<tbody>
<tr>
<td>2000 SG344</td>
<td>0.977 4398</td>
<td>0.066 9574</td>
<td>0.1122</td>
<td>191.91213</td>
<td>275.3467</td>
<td>35.67924</td>
<td>30</td>
<td>4</td>
</tr>
<tr>
<td>2013 WA44</td>
<td>1.100 5139</td>
<td>0.060 4763</td>
<td>2.30208</td>
<td>56.51177</td>
<td>176.77039</td>
<td>65.59405</td>
<td>60</td>
<td>2</td>
</tr>
<tr>
<td>2014 QN266</td>
<td>1.052 6274</td>
<td>0.092 3123</td>
<td>0.48841</td>
<td>171.11204</td>
<td>61.61488</td>
<td>211.27060</td>
<td>21</td>
<td>3</td>
</tr>
<tr>
<td>2014 WA366</td>
<td>1.034 2956</td>
<td>0.071 4762</td>
<td>1.55886</td>
<td>67.10693</td>
<td>287.64904</td>
<td>148.95123</td>
<td>15</td>
<td>4</td>
</tr>
<tr>
<td>2014 YD</td>
<td>1.072 0961</td>
<td>0.086 6307</td>
<td>1.73582</td>
<td>117.64033</td>
<td>34.13668</td>
<td>273.26011</td>
<td>50</td>
<td>2</td>
</tr>
<tr>
<td>2015 DU</td>
<td>1.101 7721</td>
<td>0.087 5283</td>
<td>2.74917</td>
<td>341.7601</td>
<td>155.66271</td>
<td>213.79942</td>
<td>18</td>
<td>3</td>
</tr>
<tr>
<td>2016 TB18</td>
<td>1.077 5160</td>
<td>0.084 2967</td>
<td>1.52705</td>
<td>190.32308</td>
<td>305.62702</td>
<td>341.22381</td>
<td>40</td>
<td>3</td>
</tr>
<tr>
<td>2016 TB57</td>
<td>1.102 2162</td>
<td>0.123 1569</td>
<td>0.29836</td>
<td>294.82422</td>
<td>147.83708</td>
<td>359.03326</td>
<td>20</td>
<td>2</td>
</tr>
<tr>
<td>2017 BN93</td>
<td>1.044 5107</td>
<td>0.051 6205</td>
<td>2.12563</td>
<td>315.78744</td>
<td>23.37608</td>
<td>188.62918</td>
<td>30</td>
<td>6</td>
</tr>
<tr>
<td>2017 SV19</td>
<td>1.063 2547</td>
<td>0.040 7058</td>
<td>1.30347</td>
<td>343.83289</td>
<td>156.82527</td>
<td>24.23585</td>
<td>30</td>
<td>4</td>
</tr>
<tr>
<td>2017 YW3</td>
<td>1.094 7081</td>
<td>0.113 2672</td>
<td>2.20028</td>
<td>273.73713</td>
<td>136.95171</td>
<td>77.93202</td>
<td>17</td>
<td>5</td>
</tr>
<tr>
<td>2018 FM2</td>
<td>0.941 8296</td>
<td>0.094 6861</td>
<td>1.15816</td>
<td>143.05867</td>
<td>191.92738</td>
<td>352.46945</td>
<td>15</td>
<td>6</td>
</tr>
<tr>
<td>2018 LQ2</td>
<td>1.091 1359</td>
<td>0.057 5314</td>
<td>2.12572</td>
<td>178.31356</td>
<td>142.8243</td>
<td>206.99288</td>
<td>30</td>
<td>2</td>
</tr>
<tr>
<td>2018 PK21</td>
<td>0.988 3530</td>
<td>0.080 9127</td>
<td>1.19448</td>
<td>304.50595</td>
<td>223.08003</td>
<td>87.71059</td>
<td>22</td>
<td>4</td>
</tr>
</tbody>
</table>

Figure 2.6: Number of accessible NEOs as a function of the maximum Lambert Problem IT impulsive Delta-v for a rendezvous.
Figure 2.7: Number of valid Lambert Problem ITs for the NEOs of Table 2.3 as a function of the maximum Lambert Problem IT impulsive Delta-v for a rendezvous.
Chapter 3

Spacecraft Propulsion

3.1 Margin Philosophy

Adopting a meaningful margin philosophy is important to account for uncertainties. This section describes the margins included in the present mission analysis study.

The margins used for this work are given in Table 3.1, and were based on requirements presented in the ESA margin philosophy for science assessment studies report [94].

Table 3.1: Mission margins.

<table>
<thead>
<tr>
<th>Margin</th>
<th>Description</th>
<th>Motivating requirement of [94]</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>PM</td>
<td>Propellant margin</td>
<td>R-M1-6</td>
<td>2 %</td>
</tr>
<tr>
<td>OSM</td>
<td>Overscaling margin</td>
<td>R-M2-9</td>
<td>10 %</td>
</tr>
<tr>
<td>DVM</td>
<td>Delta-v margin</td>
<td>R-DV-11</td>
<td>5 %</td>
</tr>
</tbody>
</table>

The propellant margin accounts for propellant residuals, and affects the propellant mass $m_p$ via

$$m_p = (1 + \text{PM}) m'_p.$$ (3.1)

The prime superscript refers to a mass determined for null propellant margin.

The propellant tanks volume, and therefore propellant mass capacity, is such that the SC mass could be larger by the overscaling margin, what implies oversized propellant tanks. This margin impacts on the initial SC mass through

$$m_0 = (1 + \text{OSM}) m'_0.$$ (3.2)

The star subscript refers to a mass calculated for null overscaling margin.

The Delta-v margin covers uncertainties in mission design and system performance by updating an accurately calculated Delta-v, which should include the effect of gravity losses, to

$$\Delta v = (1 + \text{DVM}) \Delta v''.$$ (3.3)
The double prime superscript refers to a Delta-v obtained for null Delta-v margin.

### 3.2 Low-thrust and High-thrust

This section presents an overview of the two main propulsion types, low-thrust and high-thrust. High-thrust includes chemical propulsion systems and thermal rockets, and low-thrust encompasses electric propulsion and solar sails [50].

A high-thrust manoeuvre has a much shorter burn time, and therefore there is greater potential to minimise gravity losses in comparison with low-thrust systems (see Section 2.4.1).

High-thrust propulsion could be approximated as an instantaneous change in the spacecraft velocity. The longer is the manoeuvre duration, the lower is the validity of the approximation. Consequently, a Hohmann transfer could potentially be modelled with impulsive manoeuvres, but not a launch from Earth to LEO. On the other hand, low-thrust propulsion should be modelled as a continuous force acting on the spacecraft. This means the trajectory should be numerically integrated, what greatly increases the complexity of the respective boundary value problem optimisation.

In high-thrust and low-thrust systems, the kinetic energy of the expelled mass has different origins. High-thrust takes advantage of the internal energy of the propellant, whereas low-thrust accelerates the expelled mass via solar or nuclear energy. As a result, high-thrust systems tend to have a greater fuel fraction, but low-thrust demands a larger power budget.

Due to the smaller thrust-to-mass ratio, low-thrust systems usually imply a longer transfer time. However, that generally leads to greater launch date flexibility.

The advantages of both systems were summarised in Table 3.2.

<table>
<thead>
<tr>
<th>Low-thrust</th>
<th>High-thrust</th>
</tr>
</thead>
<tbody>
<tr>
<td>Smaller fuel fraction</td>
<td>Smaller gravity losses</td>
</tr>
<tr>
<td></td>
<td>Lower trajectory design complexity</td>
</tr>
<tr>
<td>Greater launch date flexibility</td>
<td>Smaller power budget</td>
</tr>
<tr>
<td></td>
<td>Shorter time of flight</td>
</tr>
</tbody>
</table>

### 3.3 Small Spacecraft Propulsion

The present section introduces the various types of small spacecraft propulsion, thus providing a basis for the HTS and CubeSat LTPS selection addressed in the following sections.

The range of propulsion system types for small spacecraft found in the “Propulsion” chapter1 of the online NASA report State of the Art of Small Spacecraft2 Technology [95] is presented in Table 3.3. $T$ stands for thrust, $I_{sp}$ for specific impulse, and TRL for Technology Readiness Level as defined in

---

1 The “Propulsion” chapter of [95] was last updated in March 2019.
2 Small spacecraft are defined in the report as having less than 180 kg.
NASA Procedural Requirements [96]. The three first systems are high-thrust, whereas the last five are low-thrust.

An overview of propulsion system types in earlier stages of development could be found in the section “On the Horizon” of the “Propulsion” chapter of the NASA online report, but have not been assessed due to low TRL.

Table 3.3: Propulsion system types for small spacecraft [95].

<table>
<thead>
<tr>
<th>System</th>
<th>$T$</th>
<th>$I_{sp}$ (s)</th>
<th>TRL</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hydrazine</td>
<td>0.5 N–30.7 N</td>
<td>200–235</td>
<td>7</td>
</tr>
<tr>
<td>Cold gas</td>
<td>10 mN–10 N</td>
<td>40–70</td>
<td>GN2/Butane/R236FA: 9</td>
</tr>
<tr>
<td>Alternative (green) propulsion</td>
<td>0.1 N–27 N</td>
<td>190–250</td>
<td>HAN: 6. ADN: 9</td>
</tr>
<tr>
<td>Pulsed plasma and vacuum arc thrusters</td>
<td>1 µN–1.3 mN</td>
<td>500–3 000</td>
<td>Teflon: 7. Titanium: 7</td>
</tr>
<tr>
<td>Electrospray propulsion</td>
<td>10 µN–120 µN</td>
<td>500–5 000</td>
<td>7</td>
</tr>
<tr>
<td>Hall effect thrusters</td>
<td>10 mN–50 mN</td>
<td>1 000–2 000</td>
<td>Xenon: 7. Iodine: 3</td>
</tr>
<tr>
<td>Gridded ion engines</td>
<td>1 mN–10 mN</td>
<td>1 000–3 500</td>
<td>Xenon: 7. Iodine: 4</td>
</tr>
<tr>
<td>FEEP$^3$</td>
<td>10 µN–400 µN</td>
<td>2 000–6 000</td>
<td>Indium: 9</td>
</tr>
<tr>
<td>Solar Sails</td>
<td>0.25 mN–0.6 mN</td>
<td>$+\infty$</td>
<td>85 m$^2$: 6. 35 m$^2$: 7</td>
</tr>
</tbody>
</table>

### 3.4 High-thrust Stage Selection

The current section details the selection of the HTS, which enables the EDs of all TCs, and the Lambert Problem ITs of TC1 and TC2.

From Eq. (2.11), the impulsive Delta-v to escape with null hyperbolic excess speed is $772 \text{ m/s}$. This large Delta-v is not compatible with cold gas, which, due to its low specific impulse, is only suitable for small Delta-v applications as the attitude control of small spacecraft [95] (e.g. 3U CubeSats).

Hydrazine could potentially be used, but “green propellants” were selected owing to their higher specific impulse, and being safer and more environmentally friendly. For example, “green propellants” have a significantly lower vapour pressure as compared to hydrazine, and therefore a lower rate of evaporation, which means they tend to be less flammable [95]. In turn, this reduces operational oversight by safety and emergency personnel.

A selection of hydrazine and “green propellants” state-of-the-art thrusters is presented in Table 3.4.

The Delta-v capacity ($\Delta v$) was calculated with the Rocket Equation using the minimum specific impulse$^4$, the CubeSat mass of 16 kg (see Table 1.1), the HTS wet mass $m_{HTS}$, and the propellant mass $m_p$.

The ratio between the Delta-v capacity and HTS wet mass $(\Delta v / m_{HTS})$ was also computed because it provides a wider picture of the thruster efficiency, accounting for its dry mass.

$^3$Field Emission Electrical Propulsion.

$^4$For the MPS thrusters, the minimum specific impulse was calculated dividing the minimum impulse by the propellant weight $g_0 m_p$, where $g_0 = 9.80665 \text{ m/s}^2$ is the standard gravity acceleration at the Earth surface, and the propellant mass $m_p$ was retrieved from the difference between the wet and dry HTS mass.

$^5$m$_{HTS}$ has been provided by Tethers Unlimited via private communication, and is higher than the 12.6 kg given in the engine datasheet of [98].
Table 3.4: Comparison between high-thrust CubeSat propulsion systems.

<table>
<thead>
<tr>
<th>HTS</th>
<th>Propellant</th>
<th>$T$ (N)</th>
<th>$I_{sp}$ (s)</th>
<th>$m_{HTS}$ (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>MPS-125-8 U [97]</td>
<td>Hydrazine</td>
<td>0.25–1.25</td>
<td>&gt; 203</td>
<td>12.1</td>
</tr>
<tr>
<td>MPS-135-8 U [97]</td>
<td>AF-M315E</td>
<td>0.25–1.25</td>
<td>&gt; 206</td>
<td>14.7</td>
</tr>
<tr>
<td>HYDROS-M [98, 99]</td>
<td>Water</td>
<td>&gt; 1.2</td>
<td>&gt; 310</td>
<td>13.9$^5$</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Thruster</th>
<th>$m_p$ (kg)</th>
<th>$\Delta v$ (m/s)</th>
<th>$\Delta v / m_{HTS}$ (m/($s$ kg))</th>
<th>TRL</th>
</tr>
</thead>
<tbody>
<tr>
<td>MPS-125-8 U [97]</td>
<td>7.0</td>
<td>570</td>
<td>47</td>
<td>8</td>
</tr>
<tr>
<td>MPS-135-8 U [97]</td>
<td>9.6</td>
<td>756</td>
<td>51</td>
<td>6</td>
</tr>
<tr>
<td>HYDROS-M [98, 99]</td>
<td>6.2</td>
<td>706</td>
<td>51</td>
<td>6+</td>
</tr>
</tbody>
</table>

The Aerojet Rocketdyne MPS thrusters resulted from the CubeSat High-Impulse Adaptable Modular Propulsion System (CHAMPS) project, which leverages the miniaturisation effort performed for previous small hydrazine thrusters to develop CubeSat monopropellant propulsion systems [95]. The Tethers Unlimited HYDROS thrusters have the highest specific impulse of the “green” propulsion systems (by a margin of 23% to the second highest) according to information available in the “Propulsion” chapter of [95]. While MPS thrusters rely on conventional chemical propulsion, HYDROS requires an electrolysis system to split the water into hydrogen and oxygen, which are subsequently used for bi-propellant combustion [100]. Both HYDROS and MPS thrusters are modular scalable systems.

An oversized version of the HYDROS-M thruster, henceforth referred to as HYDROS-L (HYDROS-Large), was selected for the high-thrust stage. Overscaling is necessary because the systems presented in Table 3.4 have a Delta-v capacity smaller than the parabolic escape impulsive Delta-v of 772 m/s, which underestimates the required Delta-v budget. HYDROS was selected because:

- From Table 3.4, HYDROS-M and MPS-135-8 U have a similarly high efficiency ($\Delta v / m_{HTS}$). However, since the specific impulse of HYDROS-M is higher, for a larger Delta-v capability, HYDROS-L will tend to have a higher efficiency than a larger version of the MPS-135.

- Using water as a propellant has been prioritised over relying on the MPS-135 propellant AF-M315E. This could eventually allow for an in-orbit water refuelling demonstration, as the next generation of HYDROS thrusters will include refuelling ports [101], thus paving the way for in-situ resource utilisation (ISRU) of NEOs.

- The HYDROS development is likely ahead of the MPS-135-8 U. Although HYDROS-M is TRL 6+ [98] (similar to the TRL 6 of MPS-135-8 U), HYDROS-C, which is a smaller version of HYDROS-M, was launched in December 2019 on NASA’s Pathfinder Technology Demonstrator 1 [102].

Nevertheless, it should be noted that HYDROS would demand a heavier power system for the electrolysis, and the standardised modular approach of the MPS thrusters could potentially imply lower costs for the clients in comparison with HYDROS.

For trajectory design purposes, the HYDROS-L thrust and specific impulse were set, respectively, to 1.2 N and 310 s, which are the minimum values for HYDROS-M given in Table 3.4. These correspond to
conservative assumptions as, the lower the thrust and specific impulse, the smaller is the thrust induced acceleration\(^6\), and therefore the larger the Delta-v budget, and the longer the ED duration (what in turn also implies a larger radiation dose).

### 3.4.1 HYDROS-L

This section addresses the scaling-up process of HYDROS-M to HYDROS-L. The ultimate goal of the scaling analysis is determining the relation between the initial SC mass (HTS plus CubeSat), and the Delta-v budget for the HTS, \( \Delta v_{\text{HTS}} \).

The aforementioned relation depends on how the propellant tank mass scales with the propellant mass. As the HYDROS-M tank resembles a cylinder [98], the propellant tank mass \( m_t \), propellant mass \( m_p \), and ratio between the two are:

\[
\begin{align*}
m_t &= \rho_t \left( 2 \pi R^2 + 2 \pi R h \right) t; \\
m_p &= \rho_p \pi R^2 h; \\
\frac{m_t}{m_p} &= 2 \frac{\rho_t}{\rho_p} \frac{t}{h} \left( 1 + \frac{h}{R} \right).
\end{align*}
\]

\( \rho_t \) and \( \rho_p \) are the density of the tank material and propellant, respectively, \( R \) is the tank radius, \( h \) is its height, and \( t \) its thickness.

HYDROS does not need a particular water tank shape [99], and its geometry is mainly driven by customer needs. As a result, for simplicity, the height-to-radius ratio was assumed constant \( (h \propto R) \). Supposing constant thickness \( (t = ct) \) is also reasonable as the water pressure would not change for the scaled version. Consequently, since the propellant and tank densities are maintained constant \( (\rho_{p,t} = ct) \), Eqs. (3.5) and (3.6) imply, respectively:

\[
\begin{align*}
m_p &\propto h^3; & (3.7) \\
\frac{m_t}{m_p} &\propto \frac{1}{h}. & (3.8)
\end{align*}
\]

Combining these equations,

\[
m_t = \frac{m_{t0}}{m_{p0}} \frac{1}{3} m_p^2,
\]

where the baseline HYDROS-M tank and propellant masses are, respectively, \( m_{t0} = 2.5 \text{ kg} \) and \( m_{p0} = 6.2 \text{ kg} \) [99].

The SC comprises the CubeSat and HTS (see Section 1.4), therefore its mass at injection is

\[
m_0 = m_{CS} + m_{HTS}.
\]

\(^6\)A lower specific impulse implies a larger propellant mass, and therefore a larger SC mass, thus leading to a smaller thrust acceleration.
The HTS mass could be written as

\[ m_{\text{HTS}} = m_{\text{HTS}}^* + m_t + m_p, \]  

(3.11)

where \( m_{\text{HTS}}^* = 5.2 \text{ kg} \) is the non-scalable HYDROS-M mass [99]. The ratio between the initial SC mass and the mass after all the HTS manoeuvres for null propellant and overscaling margins is

\[ \frac{m_0'}{m_0 - m_p'} = e^{\frac{\Delta v_{\text{HTS}}}{r_p}} = \frac{\hat{\Delta v}_{\text{HTS}}}{(1 + \text{PM}) (1 + \text{OSM})}, \]  

(3.12)

where \( \hat{\Delta v}_{\text{HTS}} \) was defined as the normalised HTS Delta-v budget.

Combining Eqs. (3.9) to (3.12) with the margins defined in Section 3.1 leads to the following equation which implicitly defines the propellant mass:

\[ m_p + A = B m_p^3 \iff m_p^3 + (3 A - B^3) m_p^2 + 3 A^2 m_p + A^3 = 0; \]  

(3.13)

\[ A = \left[ 1 + \text{PM} (1 + \text{OSM}) - \hat{\Delta v}_{\text{HTS}} \right] (m_{\text{CS}} + m_{\text{HTS}}^*); \]  

(3.14)

\[ B = \frac{m_{\text{ta}}}{m_p^{\frac{3}{2}}} \left[ (1 + \text{PM})^{-\frac{3}{2}} \hat{\Delta v}_{\text{HTS}} - (1 + \text{PM})^{\frac{3}{2}} (1 + \text{OSM}) + \text{OSM} \right]. \]  

(3.15)

The real positive root of the cubic equation, the only meaningful value for the propellant mass, was calculated with roots from the \textit{numpy} python toolbox. Subsequently, the propellant tank mass was computed from Eq. (3.9), the HTS mass from Eq. (3.11), and the initial SC mass from Eq. (3.10).

The initial SC mass, and HTS propellant mass and tank diameter as a function of the HTS Delta-v budget are presented in Fig. 3.1. The crosses respect a parabolic escape enabled by an instantaneous perigee kick (whose Delta-v for null Delta-v margin was computed from Eq. (2.11)), and therefore represent the HTS Delta-v budget for an ED without gravity losses. The curves of Fig. 3.1 were used to determine the SC mass breakdown and diameter for the various TCs after calculating the HTS Delta-v budget as described in Sections 4.1 and 4.2.
3.5 CubeSat Low-thrust Propulsion System Selection

The selection of the CubeSat LTPS, which enables the continuous thrust ITS of TC3, is discussed in the two following sections.

3.5.1 Propulsion Type Selection

This section explains why ion thrusters are the most adequate propulsion type for the CubeSat LTPS.

Pulsed plasma and vacuum arc thrusters are suitable for attitude control and fine pointing applications, thanks to their small adjustable thrust, but not for ITS. Similarly, electrospray propulsion is only adequate for small Delta-v applications.\(^7\)

FEEP features a very high specific impulse, but was not selected due to smaller thrust-to-power ratio and shorter operational life. For the Enpulsion's instantaneous frequency measurement (IFM) Nano Thruster\(^7\), the aforementioned ratio is 12 µN/W \(^8\) [103]. In comparison, for the M-ARGO 6 and 8 solar panels configurations, the ratios are 18 µN/W and 20 µN/W [34]. In other words, the IFM Nano Thruster would require a solar panel area about 50\% larger than the 6 solar panels M-ARGO for the same thrust level. In addition, FEEP operational life is usually shorter than 10 kh \(\approx 1.1\) a [104], which is only about 40\% of the maximum mission duration of 3 a (see Section 1.4.4); while the operational life of gridded ion engines is about 30 kh \(\approx 3.4\) a [104], roughly 10\% longer than the maximum mission duration.

Hall effect thrusters would be a reasonable option, with their thrust-to-power ratio being higher than that of ion engines [42]. However, difficulties in the miniaturisation of some of the components, such as the neutralisers, render the total power consumption high compared to ion engines [95]. According to Tables 4–7 and 4–9 of [95], the typical power consumption for ion engines and Hall effect thrusters is

\(^7\)Enpulsion is currently the only commercial manufacturer offering an FEEP thruster [95].

\(^8\)Value calculated based on the thruster performance graph available in [103], for a thrust of 0.3 mN achieved with the minimum power of 25 W, which results in the minimum specific impulse of 2 250 s for that thrust level.
40 W–60 W and 175 W–200 W, respectively. Electromagnetic interference of ion thrusters is also much lower compared to Hall thrusters [42]. Moreover, these have a propulsion lifetime of 10 kh [104] which strongly limits the maximum mission duration, and is about three times shorter than that of gridded ion engines (see previous paragraph).

Solar sails are propellant-less systems, and could theoretically provide an arbitrarily high Delta-v. Nevertheless, the Delta-v capacity is small in practice owing to the limited thrust (from Table 3.3, roughly one order of magnitude smaller than that of gridded ion engines) paired with mission duration constraints. Solar sails would find an application in a scenario analogous to that of the NEA Scout mission (see Table A.2), where the CubeSat departs from Earth in a hyperbolic orbit, and therefore a smaller IT Delta-v is required in comparison to an IT starting from null hyperbolic excess velocity as in this work. The ion engines thrust model is also simpler than that of solar sails.

### 3.5.2 Ion Thruster Selection

The present section specifies the reasons for electing ArianeGroup’s RIT (Radiofrequency Ion Thruster) µX as the CubeSat LTPS.

The selection of state-of-art ion engines found in the “Propulsion” chapter of [95] is presented in Table 3.5. $P_{in}$ is the LTPS required input power, and $m_{LTPS}$ its wet mass. The Delta-v capacity was calculated with the Rocket Equation for an initial mass equal to the CubeSat mass of 16 kg (see Table 1.1), and final mass given by the difference between the CubeSat mass and thruster propellant mass. For this calculation, the RIT µX variable specific impulse and propellant mass were set to 3 000 s and 2.5 kg, respectively. Such specific impulse and mass are used in the M-ARGO preliminary trajectory design study [45], respectively, for the constant specific impulse (and maximum thrust) model, and to define the maximum propellant mass for which the NEOs are considered reachable.

<table>
<thead>
<tr>
<th>Thruster</th>
<th>Manufacturer</th>
<th>Propellant</th>
<th>$T$</th>
<th>$P_{in}$ (W)</th>
<th>$I_{sp}$ (s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>BIT-3 [105]</td>
<td>Busek Iodine</td>
<td>1.15 mN</td>
<td>75</td>
<td>2 100</td>
<td></td>
</tr>
<tr>
<td>I-COUPS</td>
<td>University of Tokyo Xenon</td>
<td>300 µN</td>
<td>43</td>
<td>1 000</td>
<td></td>
</tr>
<tr>
<td>RIT µX [34, 107]</td>
<td>ArianeGroup Xenon</td>
<td>10 µN–3 000 µN</td>
<td>&lt; 50</td>
<td>300–3 500</td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Thruster</th>
<th>$m_{LTPS}$ (kg)</th>
<th>$m_p$ (kg)</th>
<th>$\Delta v$ (km/s)</th>
<th>$\Delta v / m_{LTPS}$ (km/(s·kg))</th>
<th>TRL</th>
</tr>
</thead>
<tbody>
<tr>
<td>BIT-3 [105]</td>
<td>4.4</td>
<td>1.50</td>
<td>2.0</td>
<td>0.46</td>
<td>5</td>
</tr>
<tr>
<td>I-COUPS⁹ [106]</td>
<td>9.5</td>
<td>2.50</td>
<td>1.7</td>
<td>0.18</td>
<td>9</td>
</tr>
<tr>
<td>RIT µX [34, 107]</td>
<td>5.3</td>
<td>2.5</td>
<td>2.4</td>
<td>0.94</td>
<td>5</td>
</tr>
</tbody>
</table>

I-COUPS (Ion thruster and COld gas thruster Unified Propulsion System) has the highest TRL, since it was part of the PROCYON spacecraft launched with Hayabusa-2 in 2014, but it malfunctioned during its mission, thus preventing the flyby of the target asteroid [108]. In any case, the I-COUPS’s specific impulse is much smaller compared to the other two systems, and hence it was not selected. BIT-3

---

⁹I-COUPS includes a cold gas thruster and an integrated propulsion system, therefore $\Delta v / m_{LTPS}$ is underestimated as $m_{LTPS}$ includes an internal power supply system.
was also discarded because its efficiency ($\Delta v / m_{\text{HTS}}$) is significantly smaller than that of RIT $\mu$X, and it demands a larger power consumption.

The RIT $\mu$X thrust model is presented in the following section.

**RIT $\mu$X Thrust Model**

RIT $\mu$X presents variable maximum thrust $T_{\text{max}}$ and specific impulse $I_{\text{sp}}$, depending on the available input power $P_{\text{in}}$, which in turn varies with the distance to the Sun $r_S$. For this matter, the thrust and power model of the M-ARGO CubeSat, whose reference thruster is the RIT $\mu$X, was considered for this work. The model is integrated in the *PyKEP* class `trajopt.lt_margo`, which was used to define the continuous thrust IT optimisation problem (see Section 4.3.1), and respects the following [45]:

\[
T_{\text{max}} = (26.27127 P_{\text{in}} - 708.973) \times 10^{-3};
\]

\[
I_{\text{sp}} = -0.0011 P_{\text{in}}^3 + 0.175971 P_{\text{in}}^2 + 4.193797 P_{\text{in}} + 2073.213;
\]

\[
P_{\text{in}} = \min \{0.92 (P_{\text{sw}} - \max \{13.75 - P_{\text{bmp}}\}), 120\};
\]

\[
P_{\text{bmp}} = -40.558 r_S^3 + 173.49 r_S^2 - 259.19 r_S + 141.86;
\]

\[
P_{\text{sw}} = -146.26 r_S^3 + 658.52 r_S^2 - 1059.2 r_S + 648.24.
\]

$P_{\text{bmp}}$ and $P_{\text{sw}}$ are the power produced by the body-mounted solar panel and solar wings, respectively. For a distance to the Sun of 1 AU, the available input power is 93.20 W, the maximum thrust is 1.739 mN, and the specific impulse is 3.102 s.
Chapter 4

Trajectory Design

4.1 Earth Departure

This section details the ED strategy which applies to all TCs.

The ED ARs were based on those of the MARIO CubeSat [42] (see Appendix A.1), which aims to reach Mars departing from a supersynchronous GTO, with a parabolic escape Delta-v only 1% higher than the ideal one of an instantaneous perigee kick [43]. Following such strategy for the present mission is reasonable, as it will be shown in Section 5.1, despite the thrust accelerations being somewhat different: the MARIO CubeSat thrust acceleration for the initial SC mass is $2 \cdot 1.536 \text{N} / 32 \text{kg} \approx 9.6 \text{cm/s}^2$ [42]; and, for the present mission, it could range from $1.2 \text{N} / 100 \text{kg} = 1.2 \text{cm/s}^2$ (for the maximum initial SC mass defined in Section 1.4.4) to $1.2 \text{N} / 35.9 \text{kg} \approx 3.3 \text{cm/s}^2$ (for the initial SC mass of an ED without gravity losses, given by the blue cross of Fig. 3.1).

The following characterises the MARIO mission ARs [43]:

- The thrust magnitude is supposed constant ($T = ct$).
- The thrust direction is assumed parallel to the velocity vector ($\vec{T} / T = \vec{v} / v$).
- The thrust activation region starts at $\alpha_i = -\Delta \alpha_T / 2$, and the burn duration is calculated from the time to travel from $\alpha_i$ to $\alpha_f = \Delta \alpha_T / 2$ in the pre-burn ellipsis. The reference angle $\alpha$ is either the mean or true anomaly, and $\Delta \alpha_T$ is the angular length of the thrust activation region.
- The last AR ends when the SC reaches null specific energy.

For this work, the thrust activation was defined via the eccentric anomaly ($\alpha = E$) rather than the mean or true anomaly. Consequently, the ED only depends on $\Delta E_T$, whose range was set to $[5;90]^\circ$ (step $1^\circ$). The eccentric anomaly was selected because it is readily returned by pykep.ic2par for a given position, velocity and gravitational parameter. In any case, regardless of the anomaly which is used to define the thrust activation region, the gravity losses tend to zero as its angular length decreases.
In addition, two minor improvements were made to the strategy of [43]:

- The thrust activation arcs were defined relative to the instantaneous perigee, not to the pre-burn one as in [43], and therefore account for the orbit change during the burn. In other words, thrust is active if $E \in [-\Delta E_T / 2, \Delta E_T / 2]$.

- The last thrusting arc was made approximately symmetric relative to the perigee. For the last burn of the strategy of [43], while the thrust activation arc is symmetric relative to the pre-burn perigee, the thrust application is not symmetrically split because the SC might reach null specific energy midway through the activation arc. So as to achieve symmetry, the eccentric anomaly span of the last thrust activation arc ($\Delta E_T$) was progressively increased, in steps of $1^\circ$ starting from $5^\circ$, instead of set to the eccentric anomaly span of the previous burns ($\Delta E_T$).

It should be noted that the optimum strategy does not necessarily involve symmetry of the thrusting arcs relative to the perigee. However, as illustrated by the MARIO mission results, and as will be confirmed in Section 5.1, symmetry is an accurate proxy for minimising gravity losses.

For the analysed parabolic escape, the ED duration and Delta-v do not depend on the longitude of the ascending node of the initial GTO (its only variable parameter), and therefore these are independent of the launch date.

Concerning the SC state (position, velocity and mass) propagation, the ARs were modelled via the PyKEP function `propagate_taylor`, which “propagates keplerian motion disturbed by a constant inertial thrust using the Taylor integration method”; and the SC motion during the rest of the planetary trajectory was computed via `propagate_lagrangian`, which “propagates pure keplerian motion using Lagrange coefficients and universal variables” [58]. The propagation was performed with a time step of 10 s, and the absolute and relative tolerance for `propagate_taylor` were set to $10^{-15}$.

The Delta-v concerning the aforementioned ARs affected by the Delta-v margin should be equal to that provided by the HYDROS-L HTS. As a result, an iterative process is required to find the solution of

$$
\Delta v_{ED} (m_0) = (1 + DVM) \Delta v'_{ED} (m_0) = \Delta v_{HTS} (m_0).
$$

(4.1)

The ARs Delta-v for null Delta-v margin ($\Delta v''_{ED}$) was calculated with the Rocket Equation from the SC mass at injection ($m_0$) and after the last AR, which is an output of the SC state propagation. The HTS Delta-v budget ($\Delta v_{HTS}$) was taken from Fig. 3.1.

The solution of Eq. (4.1) was found with the bisection method (see, for instance, [111]). The lower bound of the initial SC mass range was initialised to $m_{0\text{min}} \approx 35.9$ kg, which is the initial SC mass for null gravity losses (given by the blue cross of Fig. 3.1). The upper bound was initialised to the maximum

---

1 The Taylor integration method consists in approximating integrand functions by truncated Taylor series (see, for example, [109]).

2 Lagrange coefficients translate the relation between the current state and the initial conditions. Universal variables are those which could be used with any of the two-body conic sections (ellipses, parabolas and hyperbolas). Further details about Lagrange coefficients and universal variables could be found in chapters 3 and 4 of [110], respectively.
initial SC mass of $m_{0\text{max}} = 100$ kg (see Table 1.1). The iterative process was interrupted as soon as the solution range became smaller than $\Delta m_{0\text{max}} = 0.1$ kg. As a consequence, by performing the last initial SC mass update to the middle value of the final range, the maximum error is $\Delta m_{0\text{max}} / 2 = 0.05$ kg.

The bisection method only requires one simulation of the ED per iteration, whereas methods relying on derivatives (of the error function $\Delta v_{\text{ED}}(m_0) - \Delta v_{\text{HTS}}(m_0)$) require at least two simulations per iteration. Moreover, as the only convergence requirement is the initial SC mass belonging to the defined initial range, the bisection method is more robust than other methods which converge for fewer iterations.

**4.2 Lambert Problem Interplanetary Transfer**

The present section details the Lambert Problem ITs. These are part of TC1 and TC2, and were also used to determine the NEO population for the continuous thrust ITs and TCs global optimisation (see Section 2.6.2).

The Lambert Problem ITs were defined with 4 variable parameters:

- NEO belonging to the population defined in Table 2.2.
- IT starting date $t_{\text{dep}}$ between 2023-01-01.0 and 2030-01-01.0 (step 10 d).
- IT duration $\Delta t_{\text{IT}}$ between 10 d and 3 a (step 10 d).
- IT maximum number of complete revolutions calculated, as suggested by Izzo in algorithms 1 and 2 of [68], from $N_{\text{IT max}} = \lfloor \sqrt{2 \mu s / s^3 \Delta t_{\text{IT}} / \pi} \rfloor$, where $s$ is the semiperimeter of the triangle defined by the Sun, Earth at the IT start, and NEO at the IT end.

Retrograde solutions were not considered given that the initial GTO and Earth orbits are prograde.

The Earth and NEO velocities at the start and end of the IT, $\vec{V}_E$ and $\vec{V}_{\text{arr NEO}}$, respectively, were calculated with pykep.planet.jpl./jpl.eph (see Section 2.3). The SC velocity at the start and end of the IT, $\vec{V}_E$ and $\vec{V}_{\text{arr NEO}}$, respectively, were obtained with Izzo’s Lambert solver (introduced in Section 2.2). Subsequently, the departure Delta-v, arrival Delta-v for the rendezvous, relative flyby speed, and arrival Delta-v for the flyby were computed from:

\[
\Delta v_{\text{dep}} = \left| \vec{V}_E - \vec{V}_E \right|; \tag{4.2}
\]

\[
\Delta v_{\text{arr RDV}} = \left| \vec{V}_{\text{arr NEO}} - \vec{V}_{\text{arr NEO}} \right|; \tag{4.3}
\]

\[
v_{\text{FB rel}} = \min \left\{ \Delta v_{\text{arr RDV}}, v_{\text{FB rel max}} \right\}; \tag{4.4}
\]

\[
\Delta v_{\text{arr FB}} = \Delta v_{\text{arr RDV}} - v_{\text{FB rel}}. \tag{4.5}
\]

Both the arrival Delta-v and relative flyby velocity after the arrival burn are parallel to the relative flyby velocity before the arrival burn. The arrival Delta-v for a flyby is null if the arrival Delta-v for a rendezvous is smaller than the maximum relative flyby speed; and it is equal to the difference between the arrival Delta-v for a rendezvous and maximum relative flyby speed if the arrival Delta-v for a rendezvous is...
larger than the maximum relative flyby speed.

The above IT Delta-vs were combined with the ED ones to obtain the total Delta-v budgets for null Delta-v margin:

\[ \Delta v_{\text{TC1,TC2}}' = \Delta v_{\text{ED}}' + \Delta v_{\text{dep}} + \Delta v_{\text{arr, RDV}}, \]

where the computation of \( \Delta v_{\text{ED}}' \) is described in Section 4.1. The Delta-v margin (see Eq. (3.3)) was subsequently included in the above Delta-vs to obtain the HTS Delta-v budget, and the initial SC mass could then be determined from Fig. 3.1.

The Delta-v eventually needed to insert the CubeSat into a parking orbit, for the rendezvous case, was neglected given the weak gravity of NEOs. For example, the escape speed from the Didymos system, targeted by ESA’s Hera mission, is only about 20 cm/s [112].

4.3 Continuous Thrust Interplanetary Transfer

This section concerns the design of the continuous thrust ITs of TC3.

4.3.1 Problem Definition

The continuous thrust IT optimisation problem was defined with the PyKEP class trajopt.lt.margo, henceforth referred to as the M-ARGO Problem, which represents a continuous thrust IT departing from Earth (or from the Sun-Earth L1 or L2 Lagrangian points), with null hyperbolic excess speed, to a target NEO. The objective function is the final SC mass for null Delta-v margin \((m_f')\), and the respective optimisation variables are the IT starting date\(^3\), IT duration, and thrust components for each of the \(n_{\text{seg}}\) IT thrusting segments.

The M-ARGO Problem was defined with 2 variable parameters:

- NEO belonging to the population defined in Table 2.3.

- Number of thrusting segments in the set \(\{10;20;30\}\)^4.

Respecting the other input parameters:

- The IT starting date and duration ranges were set to those presented in Section 4.2, and the initial mass to the CubeSat mass affected by the overscaling margin of Eq. (3.2):

\[ m_{\text{CS OS}} = (1 + \text{OSM}) m_{\text{CS}} = 17.6 \text{ kg}. \]

\(^3\)The default date format used in pykep is the mjd2000, which refers to the number of days after 2000-01-01.0. pykep.epoch_from_string(), mjd2000 was used to convert calendar dates to that format. For the inverse conversion, the date in mjd2000 was firstly converted to Julian date with pykep.epoch()/jd, and subsequently to calendar date with the jdutil [113] Python toolbox.

\(^4\)The M-ARGO preliminary trajectory design study [45] considered 100 segments, but that would lead to a long computation time.
This assumes the HTS is jettisoned after the ED for the continuous thrust ITs so that the CubeSat does not have to carry dead mass.

- The starting point was set to the Earth ephemerides for coherence with the Lambert Problem ITs. However, in practice, after the ED, the CubeSat could be waiting (for the IT start) around the Sun-Earth L2 point as ESA’s M-ARGO.

- Earth gravity was not considered for the IT, in agreement with the patched conics method (see Section 2.1) and the Lambert Problem ITs.

- The solar electric propulsion model (described in Section 3.5) for the thrust and specific impulse dependence on the distance to the Sun was activated.

- The trajectory grid was defined as uniform, i.e. such that all the segments have the same burn time. A denser grid in the first part of the trajectory was not selected because it only improves accuracy if Earth gravity is modelled [45].

From Table 2 of the M-ARGO preliminary trajectory design study [45], assuming constant maximum thrust and specific impulse results in \( \text{err}_1 \approx 7.8\% \) of incorrectly\(^5\) ranked NEOs in terms of required CubeSat propellant mass, and in \( \text{err}_2 \approx 8.0\% \) if the Earth gravity is also neglected. Consequently, accounting for the Earth gravity would only imply \( \Delta \text{err} = \text{err}_2 - \text{err}_1 \approx 0.2\% \) more correctly ranked NEOs, which is insignificant for the population of 14 NEOs (see Table 2.3) to be analysed. As \( \text{err}_1 \gg \Delta \text{err} \), modelling the thrust and specific impulse as a function of the orbital radius is much more relevant than accounting for Earth gravity.

The tolerance (\( c_{\text{tol}} \) of pykep.trajopt.lt.margo) for the position and velocity components of the final state error was set to \( 10^{-5} \) in terms of AU and Earth mean orbital speed \( (V_E = \sqrt{\mu_S/a_E} \approx 29.8 \text{ km/s} [58]) \), respectively. Thus the maximum errors in the position and velocity components are \( 1.5 \cdot 10^3 \text{ km} \) and \( 0.30 \text{ m/s} \), respectively.

### 4.3.2 Optimisation Algorithms

Optimisation problems could be classified as [114]:

- **Continuous** or **discrete**. Continuous optimisation problems tend to be easier to solve owing to the smoothness of the constraint and objective functions.

- **Unconstrained** or **constrained**. The variables could take any real value in unconstrained optimisation, but are subject to equality or inequality constraints in constrained optimisation. Constrained problems could be further classified, according to the constraints nature, as **linear** or **non-linear**, and, depending on the constraints smoothness, as **differentiable** or **non-differentiable**.

- **Feasibility**, **single-objective** or **multi-objective**. In feasibility problems, the only goal is finding the variables which satisfy the constraints. In practice, multi-objective problems are often refor-

\(^5\)Considering the ground truth as the model which includes the Earth gravity, and considers variable maximum thrust and specific impulse.
mulated as single-objective problems by either forming a weighted combination of the different objectives or replacing some of the objectives by constraints.

- **Deterministic or stochastic.** In deterministic optimisation, it is assumed that the data for the given problem is known accurately. However, for many actual problems, due to measurement error (e.g. SC position or attitude) and uncertainty about the future (e.g. NEO position), stochastic optimisation (under uncertainty) could be useful.

According to the above, the M-ARGO Problem is a continuous non-linearly constrained single-objective deterministic problem. There are 9 optimisation algorithms available in PyGMO suitable for such problem, but 2 are not freely available, and 4 are gradient-based algorithms, thus not being adequate for the M-ARGO Problem as its objective function gradient is not known. The 3 remaining algorithms, for which optimisation archipelagos were defined, are the extended ant colony optimisation (GACO), improved harmony search (IHS), and constrained optimisation by linear approximations (COBYLA), which was taken to PyGMO from the NLopt [115] (non-linear optimisation) free/open-source library.

For GACO and IHS, the set \{5;10;50;100;500\} was analysed for the number of generations \(N_g\) to evolve, and the other input parameters were set to the default values.

For the non-population based COBYLA, the number of generations is not defined (ND). However, the optimisation was stopped when all the decision variables changed less than the NLopt’s default relative tolerance (\(\text{xtol}_r\text{l of pygmo.core.nlopt}\)) of \(10^{-8}\). This value translates into a maximum absolute tolerance of \(10^{-8} T_{\text{max}} \approx 20 \text{ pN}\) for the thrust components, \(10^{-8} \Delta t_{\text{max}} \approx 1 \text{ s}\) for the IT duration, and \(10^{-8} t_{\text{depE_{max}[mjd2000]} = 10^{-8} \cdot 11323 \text{ d} \approx 10 \text{ s}\) for the IT starting date. Such high precision is not achievable in practice, but the difference between the numerically obtained control profile and its practical implementation should be covered by the Delta-v margin.

The optimisation algorithms and respective parameters are described in Appendix A.4.1.

### 4.3.3 Archipelagos Definition

A PyGMO archipelago is a collection of island objects connected by a topology. The islands in the archipelago can exchange individuals (candidate solutions) via a process called migration. During migration, individuals are selected from the islands, and copied into a migration database, from which they can be fetched by connecting islands. The islands’ selection policy and replacement policy establish how individuals are selected from and replaced in the islands’ populations. The migration type and migrant handling policy define the migrants flow and what happens to the migrants in the database after being fetched by a destination island.

The following sections focus on the concepts highlighted above.

---

\[^{6}\text{SNOPT (sparse nonlinear optimizer), which was used in the ESA M-ARGO study [45], and WORP (we optimiza really huge problems).}\]
Island and Topology

An island object is defined by the island type, optimisation algorithm, optimisation problem and population size $N_p$. The optimisation algorithm selection was addressed in Section 4.3.2; the optimisation problem is the M-ARGO Problem; and the set of values {$5;10;50;100;500$} was studied for the islands population size. The island type was set to `pygmo.mp_island`, which relies on the Python standard `multiprocessing` module. The other island types (`pygmo.thread_island` and `pygmo.ipyparallel_island`) require `basic` thread safety$^7$, and hence are not compatible with the `none` thread safety of the M-ARGO Problem.

A topology is a weighted directed graph in which: the vertices (or nodes) are islands; the edges (or arcs) are directed connections between islands across which information flows during the optimisation process (via the migration of individuals); and the weights of the edges represent the migration probability. [60]

The analysed archipelagos are presented in Table 4.1. All archipelagos were defined as fully connected topologies (class `pygmo.fully_connected`) with unitary migration probability for all island connections. As COBYLA is not population based, archipelagos with multiple COBYLA islands were not studied. Topology 0 does not feature a global optimiser, and topologies 1 and 2 do not feature a local optimiser, thus they are supposed to function as baselines, not being expected to produce the best results. Topologies 3 and 4 pretend to isolate the synergies between the global optimiser algorithms GACO and IHS, respectively, and the local optimiser algorithm COBYLA; while topology 5 assesses the effect of combining both GACO and IHS with COBYLA.

Table 4.1: Candidate archipelago topologies.

<table>
<thead>
<tr>
<th>Archipelago ID (topoID): islands</th>
</tr>
</thead>
<tbody>
<tr>
<td>0: 1 COBYLA</td>
</tr>
<tr>
<td>1: 3 GACO</td>
</tr>
<tr>
<td>2: 3 IHS</td>
</tr>
<tr>
<td>3: 2 GACO + 1 COBYLA</td>
</tr>
<tr>
<td>4: 2 IHS + 1 COBYLA</td>
</tr>
<tr>
<td>5: 1 GACO + 1 IHS + 1 COBYLA</td>
</tr>
</tbody>
</table>

Topology type (for topoID $\neq 0$): fully connected with unitary migration probability

Selection and Replacement Policies

The best selection policy and fair replacement policy are the only selection and replacement policies implemented in PyGMO, and hence were adopted for the present work.

For constrained single-objective problems, as the M-ARGO Problem, the best selection policy and fair replacement policy select/maintain the individuals which have the highest ranking according to `pagmo::sort_population_con` [59]. This function considers that an individual with solution $x_1$ is better than one with solution $x_2$ if [59]:

- $x_1$ is feasible, and $x_2$ is infeasible;

$^7$The thread safety level in PyGMO could be: `none`, if concurrent operations on distinct objects are unsafe; `basic`, if concurrent operations on distinct objects are safe; `constant`, if read-only concurrent operations are safe. The thread safety of a PyGMO problem object prob could be determined via `prob.get_thread_safety`. [60]
Both $x_1$ and $x_2$ are infeasible, but $x_1$ satisfies more constraints than $x_2$, or $x_1$ and $x_2$ satisfy the same number of constraints, but the norm of the constraints violations is smaller for $x_1$;

Both $x_1$ and $x_2$ are feasible, but the objective function is smaller for $x_1$.

The best selection and fair replacement policies have rate as input argument. For the best selection policy, it defines the fraction of island individuals that may be selected for the migration database; for the fair replacement policy, it represents the fraction of island individuals that may be replaced [59]. The default value $rate = 1$ was used for both policies.

**Migration Type and Migrant Handling Policy**

The migration type could be [59]:

- **Point-to-point.** The islands consider individuals from only one of the connecting islands.

- **Broadcast.** The islands consider individuals from all the connecting islands.

The migrant handling policy could be [59]:

- **Preserve.** A copy of the previous candidate migrants remains in the database.

- **Evict.** The previous candidate migrants are removed from the database.

The broadcast migration and preserve policy were selected for this work.

**4.3.4 Archipelago Optimised Solution and Selection**

The archipelago evolutions were performed via `pygmo.archipelago.evolve()`, the best (highest final mass) solution of each island (its champion) was retrieved with `pygmo.archipelago.get_champions(x)`, and the respective feasibility checked with `pygmo.problem.feasibility_x(champion)`. From the champions for which the feasibility call returns True, and the Earth-NEO distance and NEO operations duration constraints are satisfied (see Section 1.4.4), the one with highest final mass corresponds to the optimised solution for the respective archipelago and NEO target.

Having in mind Sections 4.3.1 to 4.3.3, candidate archipelagos are defined by the number of thrusting segments ($n_{seg}$), topology ($topo_{ID}$), island population size ($N_p$) and number of evolutions ($N_g$). Based on the ranges defined for each of these parameters, there are $N_{n_{seg}} \cdot N_{N_p} \cdot N_{N_g} = 3 \cdot 5^2 = 75$ candidates for each of the 5 archipelagos with global optimisers ($topo_{ID} \neq 0$), and $N_{n_{seg}} \cdot N_{N_p} = 3 \cdot 5 = 15$ candidates for the archipelagos without global optimisers ($topo_{ID} = 0$). As a result, there are $5 \cdot 75 + 15 = 390$ candidate archipelagos for each NEO.

The archipelago selected for a given NEO is that whose optimised solution respects the highest final mass. A feasible continuous thrust IT is not found for a given NEO if, for all champions of all candidate archipelagos, the feasibility call returns False, or the Earth-NEO distance or NEO operations duration constraints are violated.
### 4.3.5 Final Spacecraft Mass

This section shows how the final SC mass of the continuous thrust ITs \(m_f\) could be computed from the final SC mass for null Delta-v margin \(m''_f\), which is the M-ARGO Problem objective function.

From the Delta-v margin definition (see Eq. (3.3)) and the Rocket Equation, the final mass for null propellant margin is

\[
m''_f = \frac{m_{f}^\prime}{m_{CS_{OS}}} \left(1 + DVM\right),
\]

where \(m_{CS_{OS}} = 17.6\) kg is the oversized CubeSat mass given in Eq. (4.7). The CubeSat propellant mass for null propellant margin is

\[
m'_p_{CS} = m_{CS_{OS}} - m'_f,
\]

which was inputted into Eq. (3.1) to obtain the CubeSat propellant mass accounting for the propellant margin, \(m_p_{CS}\). Finally, the mass at the end of the IT including the effect of all the margins is

\[
m_f = m_{CS} - m_{p_{CS}}.
\]

### 4.4 Trajectory Concepts Global Optimisation

#### 4.4.1 Utility Function and Variables

The goal of the global optimisation is integrating the ED and IT, which were discussed in the previous sections, maximising the final-to-initial-mass ratio \(m_f/m_0\) for each of the launch dates. For all TCs, the launch cost tends to be lower for smaller initial SC mass; and, for TC3, there is more CubeSat propellant mass available for the NEO operations for larger final CubeSat mass. Performing the optimisation as a function of the launch date is relevant because the CubeSat is not the primary payload.

The global optimisation variables are presented in Table 4.2 for each TC.

<table>
<thead>
<tr>
<th>TC</th>
<th>Number of variables</th>
<th>Variables</th>
</tr>
</thead>
<tbody>
<tr>
<td>1, 2</td>
<td>5</td>
<td>(\Delta E_T, NEO, t_{dep}, \Delta t_{IT}, N_{IT})</td>
</tr>
<tr>
<td>3</td>
<td>(4 + 3n_{seg})</td>
<td>(\Delta E_T, NEO, t_{dep}, \Delta t_{IT}, \vec{T}<em>1, \vec{T}<em>1, \ldots, \vec{T}</em>{n</em>{seg}})</td>
</tr>
</tbody>
</table>

#### 4.4.2 Pseudocodes

The pseudocodes for all the TCs are presented below. The subscript \(\text{range}\) refers to the set of values analysed for the respective variable. This set is ordered in alphabetical/increasing order, except for \(\Delta E_{T_{\text{range}}},\) which is ordered by increasing order of ED Delta-v so that the first feasible ED found is the best
one for the current IT. NEO_{range} and IT_{range} refer to the NEOs and Lambert Problem ITs, respectively, considered for the global optimisation, and their definition is described in Section 2.6.2.

**Trajectory concepts 1 and 2**

for NEO in NEO_{range}:

for \( t_{\text{launch}} \) in \( t_{\text{launch}}\_{\text{range}} \):

for \( \text{IT} = (t_{\text{dep}_E}, \Delta t_{\text{IT}}, N_{\text{IT}}) \) in \( \text{IT}_{\text{range}} \) (NEO):

\[ t_{\text{arr}_\text{NEO}} = t_{\text{dep}_E} + \Delta t_{\text{IT}} \]

if \( t_{\text{arr}_\text{NEO}} > 2031-01-01.0 \) or \( t_{\text{arr}_\text{NEO}} - t_{\text{launch}} > 3 \) a:

continue

for \( \Delta E_T \) in \( \Delta E_{T\_{\text{range}}} \):

if \( \Delta t_{\text{wait}} (t_{\text{launch}}, \Delta E_T, t_{\text{dep}_E}) < 0 \):

continue

\[ (m_f / m_0) (t_{\text{launch}}) = m_{\text{CS}} / m_0 (\Delta E_T, \text{NEO}, \text{IT}) \]

if \( (m_f / m_0) (t_{\text{launch}}) > (m_f / m_0)_{\text{max}} (t_{\text{launch}}) \):

if \( \Delta t_{\text{NEO}_{\text{max}}} (t_{\text{launch}}, t_{\text{arr}_\text{NEO}}, \text{NEO}) \geq 0.5 \) a (only for TC2):

\[ (m_f / m_0)_{\text{max}} (t_{\text{launch}}) = (m_f / m_0) (t_{\text{launch}}) \]

update best optimisation variables

break

**Trajectory concept 3**

for NEO in NEO_{range}:

\[ \text{IT}_{\text{CT}_{\text{opt}}} \text{ (NEO)} = (t_{\text{dep}_{E_{\text{opt}}}}, \Delta t_{\text{IT}_{\text{opt}}}, m_{f_{\text{opt}}}, \vec{T}_{1_{\text{opt}}}, \vec{T}_{2_{\text{opt}}}, \ldots, \vec{T}_{n_{\text{seg}_{\text{opt}}}}) \]

\[ t_{\text{arr}_{\text{NEO}_{\text{opt}}} = t_{\text{dep}_{E_{\text{opt}}}} + \Delta t_{\text{IT}_{\text{opt}}} \]

for \( t_{\text{launch}} \) in \( t_{\text{launch}}\_{\text{range}} \):

if \( t_{\text{arr}_{\text{NEO}_{\text{opt}}} > 2031-01-01.0 \) or \( t_{\text{arr}_{\text{NEO}_{\text{opt}}} - t_{\text{launch}} > 3 \) a:

continue

for \( \Delta E_T \) in \( \Delta E_{T\_{\text{range}}} \):

if \( \Delta t_{\text{wait}} (t_{\text{launch}}, \Delta E_T, t_{\text{dep}_E}) < 0 \):

continue

\[ (m_f / m_0) (t_{\text{launch}}) = m_{f_{\text{opt}}} / m_0 (\Delta E_T) \]

if \( (m_f / m_0) (t_{\text{launch}}) > (m_f / m_0)_{\text{max}} (t_{\text{launch}}) \):

if \( \Delta t_{\text{NEO}_{\text{max}}} (t_{\text{launch}}, t_{\text{arr}_\text{NEO}}, \text{NEO}) \geq 0.5 \) a:

\[ (m_f / m_0)_{\text{max}} (t_{\text{launch}}) = (m_f / m_0) (t_{\text{launch}}) \]

update best optimisation variables

break
Chapter 5

Results and Discussion

5.1 Earth Departure

This section discusses how the results of the ED, described in Section 4.1, are affected by the eccentric anomaly span of the pre-escape perigee burns (all burns except the last one), $\Delta E_T$.

Figs. 5.1 to 5.6 concern, respectively, the ED Delta-v, gravity losses\(^1\), duration, normalised radiation doses, eccentric anomaly span of the escape perigee burn (the last burn), and number of complete revolutions, all as a function of $\Delta E_T$. From $\Delta E_T = 5^\circ$ to $\Delta E_T = 90^\circ$:

- The ED Delta-v increases from 820 m/s to 1.4 km/s, and the gravity losses from 1.2% to 74% (see Figs. 5.1 and 5.2). The small magnitude of the gravity losses for $\Delta E_T = 5^\circ$ confirms that the symmetry of the thrust activation arcs relative to the perigee is an accurate proxy for minimising gravity losses (see Section 4.1). Moreover, taking into account the magnitude of the margins used for this work (see Section 3.1), modelling the ARs for $\Delta E_T < 5^\circ$ would not lead to significant gains.

- The ED duration decreases from 170 d to 12 d (see Fig. 5.3).

- The electrons normalised radiation dose decreases from 16 to 1.1, and the protons normalised radiation dose from 110 to 7.2 (see Fig. 5.4).

- The eccentric anomaly span of the escape perigee burn increases from 9° to 91° (see Fig. 5.5).

- The number of complete revolutions decreases from 104 to 7 (see Fig. 5.6).

The oscillations in the ED Delta-v and gravity losses of Figs. 5.1 and 5.2 are positively correlated with the oscillations in the eccentric anomaly span of the escape perigee burn of Fig. 5.5. This is in agreement with the Oberth effect (see Section 2.4.1) prediction of larger gravity losses for longer burns.

The ED duration and normalised radiation doses are also presented in Fig. 5.7 and Fig. 5.8 as a function of the number of complete revolutions, and in Fig. 5.9 and Fig. 5.10 as a function of the gravity losses.

\(^1\)The gravity losses are given by $\Delta v_{\text{ED}} - 1$, where $\Delta v_{\text{ED}} = \Delta v_{\text{ED}}' / \Delta v_D$ is the ratio between the actual ED Delta-v for null Delta-v margin (computed from Eq. (4.1)) and the one calculated for null gravity losses (from Eq. (2.11)).
From Figs. 5.7 and 5.8, the ED duration and normalised radiation doses are directly proportional to the number of complete revolutions. This means the ED duration and normalised radiation doses are also directly proportional amongst each other.

From Fig. 5.9, the ED duration decreases rapidly (with the gravity losses) for small gravity losses, but stabilises for higher values. In fact, from gravity losses of 1.2% ($\Delta E_T = 5^\circ$) to 13% ($\Delta E_T = 30^\circ$), the ED duration decreases 140 d (from 170 d to 25 d). However, from gravity losses of 13% ($\Delta E_T = 30^\circ$) to 74% ($\Delta E_T = 90^\circ$), the ED duration only decreases 13 d (from 25 d to 12 d).

From Fig. 5.10, the ED normalised radiation doses also decrease rapidly for small gravity losses, but stabilise for higher values. From gravity losses of 1.2% ($\Delta E_T = 5^\circ$) to 13% ($\Delta E_T = 30^\circ$), the electrons and protons normalised radiation doses decrease 14 (from 16 to 2.6) and 92 (from 110 to 18), respectively. However, from gravity losses of 13% ($\Delta E_T = 30^\circ$) to 74% ($\Delta E_T = 90^\circ$), the electrons and protons normalised radiation doses only decrease 1.6 (from 2.6 to 1.1) and 10 (from 18 to 7.2), respectively.

Having the two previous paragraphs in mind, increasing $\Delta E_T$ beyond roughly 30$^\circ$ is neither effective for shortening the ED duration nor decreasing the normalised radiation doses, and implies a significant increase in gravity losses.

Figs. 5.11 and 5.12 represent the ED electrons and protons normalised radiation doses in terms of time spent in a circular orbit, as illustrated in Figs. 2.4 and 2.5. For $\Delta E_T = 5^\circ$: the electrons normalised radiation dose corresponds to staying 1 a at an altitude of $14.7 \cdot 10^3$ km, 5 a at $15.9 \cdot 10^3$ km, or 10 a at $18.1 \cdot 10^3$ km; and the protons normalised radiation dose concerns 1 a at $5.1 \cdot 10^3$ km, 5 a at $5.9 \cdot 10^3$ km, or 10 a at $6.2 \cdot 10^3$ km.

Examples of ED trajectories for various values of $\Delta E_T$ are presented in Appendix A.2.

![Figure 5.1: ED Delta-v as a function of the eccentric anomaly span of the pre-escape perigee burns.](image)
Figure 5.2: ED gravity losses as a function of the eccentric anomaly span of the pre-escape perigee burns.

Figure 5.3: ED duration as a function of the eccentric anomaly span of the pre-escape perigee burns.
Figure 5.4: ED electrons and protons normalised radiation doses as a function of the eccentric anomaly span of the pre-escape perigee burns.

Figure 5.5: Eccentric anomaly span of the escape perigee burn as a function of the eccentric anomaly span of the pre-escape perigee burns.
Figure 5.6: Number of complete revolutions as a function of the eccentric anomaly span of the pre-escape perigee burns.

Figure 5.7: ED duration as a function of the number of complete revolutions.
Figure 5.8: ED electrons and protons normalised radiation doses as a function of the number of complete revolutions.

Figure 5.9: ED duration as a function of the ED gravity losses.
Figure 5.10: ED electrons and protons normalised radiation doses as a function of the ED gravity losses.

Figure 5.11: Circular orbit altitude and flight time respecting the ED electrons normalised radiation dose.
5.2 Continuous Thrust Interplanetary Transfer

This section presents the results of the optimisation of the continuous thrust ITs described in Section 4.3. The archipelago parameters for the best (maximum final mass) continuous thrust IT of each of the 14 analysed NEOs (see Section 2.6.2) are presented in Table 5.1. The respective IT starting date, IT duration, Earth-NEO distance at the end of the IT, CubeSat propellant mass, CubeSat propellant mass fraction, and CubeSat final mass are shown in Table 5.2.

The optimum number of thrusting segments is 20 for 2016 TB57 (ranking 10), and 10 for all the other 13 NEOs. Having more segments leads to a larger search space, and hence to the possibility of having a smaller final mass. However, the fact that the best archipelagos only have 10 segments, practically for all the NEOs, suggests that the larger search space potential is not being fulfilled.

There are no optimum archipelago topologies without a local optimiser (\(\text{topo}_{ID} \neq 1, 2\)), and the only ones without a global optimiser (\(\text{topo}_{ID} = 0\)) are those of 2017 BN93 and 2018 LQ2 (rankings 8 and 9). The archipelagos featuring both a local and a global optimiser (\(\text{topo}_{ID} = 3, 4, 5\)) performed better. This was to be expected since the local optimiser receives the solutions found by the global optimisers, refines them, and transmits the information back to the global optimisers, thus improving the quality of the optimisation [116].

In terms of the optimum IT parameters (Table 5.2), the IT starting date ranges from 2024-07-07.3 to 2030-12-18.8, the IT duration from 364 d to 762 d, the Earth-NEO distance at the end of the IT from 0.081 AU to 0.940 AU, the CubeSat propellant mass from 0.7 kg to 2.3 kg, the CubeSat propellant mass fraction from 4.4 % to 14 %, and the CubeSat final mass from 15.3 kg to 13.7 kg. All these NEOs would be classified as reachable in the M-ARGO preliminary trajectory design study\(^2\) [45], which sets 2.5 kg as the maximum CubeSat propellant mass, and therefore the ITs of TC3 are expected to be feasible.

\(^2\)The M-ARGO study is a useful reference for comparison because it analysed a continuous thrust IT of a 12 U CubeSat, with the same LTPS as the CubeSat of this work, from the Sun-Earth L2 point to a NEO (see Section 1.3).
The optimised continuous thrust IT trajectories and thrust profiles for the 8 NEOs worth targeting for TC3 (see Section 5.3) are shown in Appendix A.4.2. There is room for improving the obtained trajectories because they do not exactly resemble the optimum “bang-bang” control profile (for which the thrust is either null or maximum), theoretically predicted [45] by applying the Pontryagin maximum principle [117] to the M-ARGO (optimum control) Problem. Further optimisation of the continuous thrust ITs (decrease in the final CubeSat mass) could be achieved by considering other numbers of thrusting segments, archipelago topologies, island population sizes, or input parameters of the global optimisers GACO and IHS (including the number of generations).

Nevertheless, since the mass at the start of the continuous thrust ITs is set to the CubeSat mass, further optimisation does not imply a decrease in the initial SC mass. Moreover, the minimum CubeSat propellant mass fraction of 4.4% obtained for the present mission (see Table 5.2) is similar to the 4.6% concerning the best target NEO of the M-ARGO preliminary trajectory design study [45]. Furthermore, using the Rocket Equation for the initial CubeSat mass (16 kg), CubeSat LTPS specific impulse at 1 AU (3102 s), and smallest Lambert Problem IT Delta-v for a rendezvous (1.140 km/s), which underestimates the minimum continuous thrust IT Delta-v, leads to a CubeSat propellant mass of 0.6 kg. Consequently, given the minimum 0.7 kg of Table 5.2, the potential decrease in the required CubeSat propellant mass for the best NEO is limited to 0.1 kg.

Table 5.1: Number of thrusting segments, topology, island population size and number of evolutions for the best continuous thrust IT of each NEO.

<table>
<thead>
<tr>
<th>Ranking: NEO</th>
<th>n_{seg}</th>
<th>topo_{ID}</th>
<th>N_p</th>
<th>N_g</th>
<th>Ranking: NEO</th>
<th>n_{seg}</th>
<th>topo_{ID}</th>
<th>N_p</th>
<th>N_g</th>
</tr>
</thead>
<tbody>
<tr>
<td>1: 2000 SG344</td>
<td>10</td>
<td>4</td>
<td>100</td>
<td>10</td>
<td>8: 2018 LQ2</td>
<td>10</td>
<td>0</td>
<td>5</td>
<td>ND</td>
</tr>
<tr>
<td>2: 2017 SV19</td>
<td>10</td>
<td>3</td>
<td>100</td>
<td>100</td>
<td>9: 2017 BN93</td>
<td>10</td>
<td>0</td>
<td>500</td>
<td>ND</td>
</tr>
<tr>
<td>3: 2014 QN266</td>
<td>10</td>
<td>5</td>
<td>500</td>
<td>5</td>
<td>10: 2016 TB57</td>
<td>20</td>
<td>4</td>
<td>500</td>
<td>500</td>
</tr>
<tr>
<td>4: 2014 YD</td>
<td>10</td>
<td>5</td>
<td>500</td>
<td>50</td>
<td>11: 2018 PK21</td>
<td>10</td>
<td>3</td>
<td>500</td>
<td>500</td>
</tr>
<tr>
<td>5: 2016 TB18</td>
<td>10</td>
<td>5</td>
<td>500</td>
<td>500</td>
<td>12: 2015 DU</td>
<td>10</td>
<td>3</td>
<td>100</td>
<td>100</td>
</tr>
<tr>
<td>6: 2013 WA44</td>
<td>10</td>
<td>4</td>
<td>100</td>
<td>50</td>
<td>13: 2017 YW3</td>
<td>10</td>
<td>5</td>
<td>100</td>
<td>5</td>
</tr>
<tr>
<td>7: 2018 FM2</td>
<td>10</td>
<td>3</td>
<td>500</td>
<td>5</td>
<td>14: 2014 WA366</td>
<td>10</td>
<td>3</td>
<td>50</td>
<td>50</td>
</tr>
</tbody>
</table>

The M-ARGO study results do not include the effect of the margins presented in Section 3.1, but were normalised to take them into account. The normalised initial CubeSat mass was computed from Eq. (4.7) for an oversized CubeSat mass of \( m_{CSOS} = 20 \text{ kg} \) (M-ARGO study initial CubeSat mass). The normalised minimum CubeSat propellant mass was calculated by following the procedure of Section 4.3.5 for the aforementioned oversized CubeSat mass of \( m_{CSOS} = 20 \text{ kg} \), and CubeSat propellant mass for null Delta-v margin \( (m_{CSOS}’ - m_{f}’’) \) of 0.79 kg (M-ARGO study minimum CubeSat propellant mass).
Table 5.2: IT starting date, IT duration, Earth-NEO distance at the end of the IT, CubeSat propellant mass, CubeSat propellant mass fraction, and CubeSat final mass for the best continuous thrust IT of each NEO.

<table>
<thead>
<tr>
<th>Ranking: NEO</th>
<th>( t_{dep} ) (UTC)</th>
<th>( \Delta t_{IT} ) (d)</th>
<th>( d_{E\rightarrow\text{NEO}} ) (AU)</th>
<th>( m_{pcs} ) (kg)</th>
<th>( m_{pcs}/m_{CS} ) (%)</th>
<th>( m_f ) (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1: 2000 SG344</td>
<td>2027-12-10.2</td>
<td>364</td>
<td>0.081</td>
<td>0.7</td>
<td>4.4</td>
<td>15.3</td>
</tr>
<tr>
<td>2: 2017 SV19</td>
<td>2028-07-05.1</td>
<td>375</td>
<td>0.083</td>
<td>1.1</td>
<td>6.7</td>
<td>14.9</td>
</tr>
<tr>
<td>3: 2014 QN266</td>
<td>2027-04-07.5</td>
<td>431</td>
<td>0.763</td>
<td>1.2</td>
<td>7.4</td>
<td>14.8</td>
</tr>
<tr>
<td>4: 2014 YD</td>
<td>2024-07-07.3</td>
<td>384</td>
<td>0.337</td>
<td>1.2</td>
<td>7.8</td>
<td>14.8</td>
</tr>
<tr>
<td>5: 2016 TB18</td>
<td>2025-12-31.4</td>
<td>403</td>
<td>0.112</td>
<td>1.3</td>
<td>8.3</td>
<td>14.7</td>
</tr>
<tr>
<td>6: 2013 WA44</td>
<td>2028-05-12.0</td>
<td>417</td>
<td>0.439</td>
<td>1.5</td>
<td>9.3</td>
<td>14.5</td>
</tr>
<tr>
<td>7: 2018 FM2</td>
<td>2028-02-16.0</td>
<td>387</td>
<td>0.549</td>
<td>1.5</td>
<td>9.6</td>
<td>14.5</td>
</tr>
<tr>
<td>8: 2018 LQ2</td>
<td>2026-04-10.2</td>
<td>436</td>
<td>0.540</td>
<td>1.6</td>
<td>9.8</td>
<td>14.4</td>
</tr>
<tr>
<td>9: 2017 BN93</td>
<td>2029-12-22.2</td>
<td>762</td>
<td>0.852</td>
<td>1.6</td>
<td>9.9</td>
<td>14.4</td>
</tr>
<tr>
<td>10: 2016 TB57</td>
<td>2030-12-18.8</td>
<td>430</td>
<td>0.940</td>
<td>1.6</td>
<td>10</td>
<td>14.4</td>
</tr>
<tr>
<td>11: 2018 PK21</td>
<td>2024-09-18.6</td>
<td>630</td>
<td>0.806</td>
<td>1.7</td>
<td>10</td>
<td>14.3</td>
</tr>
<tr>
<td>12: 2015 DU</td>
<td>2029-03-14.6</td>
<td>402</td>
<td>0.420</td>
<td>1.9</td>
<td>12</td>
<td>14.1</td>
</tr>
<tr>
<td>13: 2017 YW3</td>
<td>2024-11-22.4</td>
<td>407</td>
<td>0.602</td>
<td>2.0</td>
<td>12</td>
<td>14.0</td>
</tr>
<tr>
<td>14: 2014 WA366</td>
<td>2029-11-30.8</td>
<td>677</td>
<td>0.176</td>
<td>2.3</td>
<td>14</td>
<td>13.7</td>
</tr>
</tbody>
</table>

5.3 Trajectory Concepts Global Optimisation

This section discusses the spacecraft mass budgets which result from the TCs global optimisation described in Section 4.4.

The optimum (highest final-to-initial-mass ratio) initial SC mass and NEO target as a function of the launch date are shown in Figs. 5.13 to 5.15 for the 3 TCs, and the maximum NEO operations duration in Figs. 5.14 and 5.15 for TC2 and TC3. The optimum initial SC mass and NEO target, as well as the respective global optimisation variables (excluding the thrust components for the continuous thrust ITs), are presented in Appendix A.5 as a function of the launch date ranking.

The mission is feasible, satisfying all constraints of Table 1.1, from 2023-01-01.0 to 2029-09-16.0 for TC1, from 2023-01-01.0 to 2029-09-06.0 for TC2, and from 2023-01-31.0 to 2029-02-28.0 for TC3. For launches approaching the latest NEO arrival date constraint (end of 2030), the available mission duration tends to zero, and thus feasible trajectories cease to exist. In addition, although the global optimisation considered 14 NEOs (see Section 2.6.2), only 4 of these are worth targeting for TC1 and TC2, and 8 for TC3.

There are step variations in the maximum NEO operations duration whenever the NEO arrival date changes, either because of a NEO target transition, or a change in the IT starting date for the same NEO. Otherwise, in agreement with Eq. (1.6), the maximum NEO operations duration increases linearly with the launch date when it is restricted by the maximum mission duration, and is constant when it is constrained by the date at which the Earth-NEO distance reaches the defined limit of 1 AU.
The step variations of the maximum NEO operations duration are a symptom of selecting the final-to-initial-mass ratio for the utility function of the global optimisation. In fact, a negligible increase in that ratio could lead to a completely different IT, with a potentially much shorter maximum NEO operations duration. For example, for TC2, from 2027-09-07.0 to 2027-09-17.0, a relatively small decrease in the initial SC mass from 61.6 kg to 63.9 kg leads to a decrease in the maximum NEO operations duration from 736 d to 186 d. In order to prevent such situations, the maximum NEO operations duration could be integrated in the global optimisation utility function. However, this would only be useful if NEO operations longer than the defined minimum of 6 months resulted in greater scientific return.

There are also sudden initial SC mass variations for TC1 and TC2. As the Lambert Problem ITs are enabled by the HTS, changes in their Delta-v affect the required HTS mass, and hence the initial SC mass. In contrast, the continuous thrust ITs are performed by the CubeSat LTPS, thus there are no abrupt initial SC mass variations for TC3. Nonetheless, NEO target transitions for TC3 could result in small bumps in the initial SC mass. These derive from the need for longer thrust activation arcs in order to escape from Earth soon enough to “catch” the current IT, thus postponing the transition to a higher Delta-v IT (1st, 3rd, 4th and 5th bumps) or an IT whose NEO operations duration is too short (2nd bump), or “catching” the last feasible IT (last bump).

The mass breakdowns and HTS propellant tank diameters for the optimised trajectories of the best (highest final-to-initial-mass ratio), 1st quartile, median and 3rd quartile launch dates are presented in Figs. 5.16 to 5.18 for the 3 TCs. The left and right numbers on the top of the bars concern, respectively, the initial SC mass and HTS propellant tank diameter. The numbers below are the absolute (left) and relative (right) contributions of each mass component to the initial SC mass: HTS dry mass (blue), HTS propellant mass (orange), CubeSat non-payload dry mass (green), CubeSat propellant mass (red), and CubeSat payload mass (purple).

Regarding the initial SC mass, from the best to the median launch date, it is 58 kg–65 kg for TC1, 60 kg–68 kg for TC2, and 36 kg for TC3. From the median to the 3rd quartile launch date, it increases 16 kg (24 %) to 81 kg for TC1, 17 kg (24 %) to 85 kg for TC2, and 0.9 kg (2.5 %) to 37 kg for TC3.

In other words, although TC2 provides 8 U of additional CubeSat scientific payload relative to TC3, it is heavier than TC3 by: 24 kg (68 %) for the best launch date; 32 kg (90 %) for the median launch date; and 48 kg (130 %) for the 3rd quartile launch date. Similarly, even though TC2 features a NEO rendezvous instead of a flyby as TC1, it is heavier than TC1 by: 2.9 kg (5.0 %) for the best launch date; 3.3 kg (5.0 %) for the median launch date; and 4.2 kg (5.1 %) for the 3rd quartile launch date.

From the above, the higher the launch date quantile, the larger the difference between TC1/TC2 and TC3, especially for launch dates worse than the median. The underlying reason for the much smaller initial SC mass of TC3 is the CubeSat LTPS specific impulse being about 3 100 / 310 = 10 times higher than that of the HTS. Since TC3 concerns a smaller and less launch date dependent initial SC mass, it tends to offer a lower launch cost, more piggyback flight opportunities, and greater launch date flexibility.

---

4The q-quantile launch date is defined here as the one which is worse (lower final-to-initial-mass ratio) than \(100q\)% of the analysed launch dates.
In comparison with the 32 kg MARIO CubeSat [42], which aims to reach Mars departing from a supersynchronous GTO, the TC1, TC2 and TC3 initial SC masses for the best launch dates are roughly 80\%, 90\% and 10\% heavier, respectively. This confirms the hypothesis advanced in Section 1.4.1 that a CubeSat with a hybrid propulsion architecture, featuring both high-thrust and low-thrust, but not an attached HTS, could result in a lighter initial SC mass. A more detailed analysis would be needed to account for the MARIO CubeSat advantageous (higher specific energy) initial orbit and disadvantageous (higher specific energy) destination, but the (∼10\%) gain relative to TC3 does not seem to be significant.

The payload ratios for the best and 3rd quartile launch dates are, respectively, 21\% and 15\% for TC1, 20\% and 14\% for TC2, and 3.7\% and 3.6\% for TC3. The small initial SC mass and payload ratio differences between TC1 and TC2 could, depending on the difference between the respective non-launch costs\(^5\), justify performing a rendezvous (TC2) for greater scientific return. The payload ratios being significantly larger for TC1 and TC2 than for TC3 might suggest a higher scientific return on investment, but their much larger initial SC mass could imply an infeasible launch cost or scarce piggyback flight opportunities.

The difficulty in finding suitable piggyback flight opportunities for TC1 and TC2 in comparison with TC3 is also supported by the difference in the SC dimension. For the best launch dates, the HTS propellant tank diameters for TC1 (64 cm) and TC2 (66 cm) are, respectively, 38\% and 42\% larger than that of TC3 (46 cm). Furthermore, similarly to the initial SC mass, the difference increases with the launch date quantile. For the 3rd quartile launch dates, the HTS propellant tank diameters for TC1 (76 cm) and TC2 (78 cm) are, respectively, 61\% and 65\% larger than that of TC3 (47 cm).

The Lambert Problem ITs concerning the best, 1st quartile, median and 3rd quartile launch dates of TC1 and TC2 are presented in Appendix A.3, and the continuous thrust ITs and respective thrust profiles for the 8 NEOs worth targeting in TC3 are shown in Appendix A.4.2.

\(^5\)The flight dynamics support is estimated to account for roughly 50\% of the M-ARGO mission cost [34].
Figure 5.13: Optimum initial SC mass and NEO target as a function of the launch date for TC1.

Figure 5.14: Optimum initial SC mass, NEO target and maximum NEO operations duration as a function of the launch date for TC2.
Figure 5.15: Optimum initial SC mass, NEO target and maximum NEO operations duration as a function of the launch date for TC3.

Figure 5.16: Mass breakdown and HTS propellant tank diameter for the best, 1st quartile, median and 3rd quartile launch dates of TC1.
Figure 5.17: Mass breakdown and HTS propellant tank diameter for the best, 1st quartile, median and 3rd quartile launch dates of TC2.

Figure 5.18: Mass breakdown and HTS propellant tank diameter for the best, 1st quartile, median and 3rd quartile launch dates of TC3.
Chapter 6

Conclusions

The present work performed a preliminary trajectory analysis for a CubeSat mission departing from a geosynchronous transfer orbit to a near-Earth object, assuming a maximum mission duration of 3 years and an initial CubeSat mass of 16 kg. The Earth departure was modelled as multiple finite apogee raising manoeuvres enabled by a high-thrust stage. The interplanetary transfer was based on the patched conics method, and was modelled via: Lambert Problem impulsive manoeuvres at the Earth departure and target arrival, performed by the high-thrust stage, ending in a flyby or rendezvous; or a continuous low-thrust transfer, powered by the CubeSat, ending in a rendezvous. From the best to the third quartile launch date, the interplanetary impulsive manoeuvres result in initial spacecraft masses ranging from 58 kg / 60 kg to 81 kg / 85 kg (flyby / rendezvous), and the low-thrust transfer in 36 kg to 37 kg. Since the low-thrust transfer concerns a smaller and less launch date dependent initial spacecraft mass, it tends to offer a lower launch cost, more piggyback flight opportunities, and greater launch date flexibility.

6.1 Future Work

Some topics worth exploring for the ED are:

- Considering a CubeSat with a hybrid propulsion architecture, featuring both high-thrust and low-thrust (see MARIO CubeSat in Table A.2).

- Analysing other initial orbit possibilities. In particular, transfers to the Moon could be used as a launchpad to escape from Earth with negligible Delta-v (see NEA Scout CubeSat in Table A.2).

- Performing a Moon gravity assist.

- Considering hyperbolic escape trajectories for the ED.

- Studying the ED eclipses.

- Assessing the required torque to rotate the SC during the ARs.
For the IT, the following could be investigated:

- Improving the optimisation of the continuous thrust ITs such that the thrust profiles presented in Appendix A.4.2 approach the optimum bang-bang control profile.

- Modelling the HTS IT burns as continuous manoeuvres.
Bibliography


A.1 Interplanetary CubeSat Missions

Examples of the development of innovative miniaturised technologies, taken from Table 5.2 of the National Academy of Sciences 2016 report of [118], are presented in Table A.1. A summary with more detailed information about the interplanetary CubeSat missions introduced in Section 1.3 is given in Table A.2. Special attention should be given to the M-ARGO, NEA Scout and MARIO missions, which are mentioned throughout this work.

Table A.1: Advances across some CubeSat subsystems.

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Past</th>
<th>Available</th>
<th>Emerging</th>
</tr>
</thead>
<tbody>
<tr>
<td>Attitude control</td>
<td>±10°: passive, magnetic system and hysteresis</td>
<td>±0.5°–5°: reaction wheels with limited de-saturation</td>
<td>±0.1° (± 0.01° by 2017): reaction wheels with de-saturation via propulsion systems</td>
</tr>
<tr>
<td>Orbit control</td>
<td>None</td>
<td>Differential drag, limited manoeuvring</td>
<td>Non-propulsive systems and low-capability propulsive systems</td>
</tr>
<tr>
<td>Communications</td>
<td>&lt; 9.6 Kibit/s</td>
<td>~ 1 Mibit/s</td>
<td>Up to 50 Mibit/s (100 Mibit/s–600 Mibit/s by 2017)</td>
</tr>
<tr>
<td>Thermal</td>
<td>Passive</td>
<td>Passive and electrical heaters</td>
<td>Passive heat-pipes, thermal louvers, deployable Sun shield and new active systems (e.g. micro-cryocooler)</td>
</tr>
<tr>
<td>Deployable systems</td>
<td>None</td>
<td>Solar arrays and UHF/VHF dipole antennas</td>
<td>Ka-band antennas, gossamer structures and tethers</td>
</tr>
</tbody>
</table>
Table A.2: Interplanetary CubeSat missions.

(a) MarCO and APEX.

<table>
<thead>
<tr>
<th>Name</th>
<th>Mars Cube One (MarCO) [30]</th>
<th>APEX (Asteroid Prospection Explorer) [32]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Organisation</td>
<td>NASA</td>
<td>Swiss Institute for Space Physics, OHB Sweden, Space Systems Czech, amongst others</td>
</tr>
<tr>
<td>System</td>
<td>two 6 U of 13.5 kg each</td>
<td>6 U of 12 kg</td>
</tr>
<tr>
<td>Science payload: goal OR Mission goal</td>
<td>UHF (Ultra High Frequency) Radio Receiver and X-band radio transmitter: serve as communications relays during the descent of the InSight lander to the Martian surface</td>
<td>Visible and infrared spectral imager (500 nm–2 500 nm): map the surface mineralogy of the asteroids. Mass spectrometer: determine the properties of ions and neutrals sputtered from the asteroids surfaces. Magnetometer: characterise the magnetic field as well as its interaction with the solar wind</td>
</tr>
</tbody>
</table>

(b) Juventas and M-ARGO.

<table>
<thead>
<tr>
<th>Name</th>
<th>Juventas [33]</th>
<th>M-ARGO (Miniaturised Asteroid Remote Geophysical Observer) [23, 34, 119]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Organisation</td>
<td>GOMspace</td>
<td>ESA</td>
</tr>
<tr>
<td>Mother mission (launch date): goal</td>
<td>ESA's Hera mission</td>
<td>Stand-alone mission (launch in 2023–2025, and maximum duration of 3 years): rendezvous with a near-Earth asteroid to characterise its physical properties, and assess resource exploitation</td>
</tr>
<tr>
<td>System</td>
<td>6 U of 10 kg</td>
<td>12 U of 22 kg</td>
</tr>
<tr>
<td>Science payload: goal OR Mission goal</td>
<td>Low frequency radar: characterise the internal structure. 3-axis gravimeter, accelerometers and gyro: characterise the gravity field, and determine surface and dynamical properties. Visible camera: provide context to the determined surface properties</td>
<td>Visible and infrared spectral imager (500 nm–2 500 nm): map the surface mineralogy. Laser altimeter: study the surface geography</td>
</tr>
</tbody>
</table>

4 NASA's DART mission, to be launched in 2021 and collide in 2022 with Didymos B (the smallest body of the system), will try to demonstrate the kinetic impactor deflection technique [8].
<table>
<thead>
<tr>
<th>Name</th>
<th>NEA Scout (Near-Earth Asteroid Scout) [39, 46]</th>
<th>CU-E³ (Earth Escape Explorer) [40]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Organisation</td>
<td>NASA</td>
<td>University of Colorado</td>
</tr>
<tr>
<td>Mother mission (launch date): goal</td>
<td>NASA's Artemis⁵ 1 mission (2021): first integrated flight test of NASA's Deep Space Exploration Systems; deploy 13 6 U Cube-Sats as secondary payload</td>
<td>NASA's Artemis 1</td>
</tr>
<tr>
<td>System</td>
<td>6 U of 12 kg</td>
<td>6 U</td>
</tr>
<tr>
<td>Science payload: goal OR Mission goal</td>
<td>Visible and near infrared camera (400 nm–750 nm): determine the physical properties (shape/volume, rotational properties, debris/dust field, and regolith characteristics) of a near-Earth asteroid</td>
<td>Demonstrate long-distance communications by reaching an orbit of at least 10 times the distance between Earth and the Moon</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Name</th>
<th>Team Miles CubeSat [41]</th>
<th>MARIO (Mars Atmospheric Radiation Imaging Orbiter) [42]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Organisation</td>
<td>Team Miles</td>
<td>Politecnico di Milano, Airbus Defence &amp; Space, TU Delft</td>
</tr>
<tr>
<td>Mother mission (launch date): goal</td>
<td>NASA's Artemis 1</td>
<td>Stand-alone mission (not defined): escape from Earth, perform autonomous deep-space cruise, achieve ballistic capture, and enter an operational orbit at Mars</td>
</tr>
<tr>
<td>System</td>
<td>6 U</td>
<td>16 U of 32 kg</td>
</tr>
<tr>
<td>Science payload: goal OR Mission goal</td>
<td>Test an innovative plasma thruster and on-board computer by flying autonomously to a distance of 0.60 AU, and demonstrate long-distance communications</td>
<td>Visible and near-infrared camera: characterise the thermal environment of the Mars upper atmosphere</td>
</tr>
</tbody>
</table>

⁵ NASA’s Artemis program [35], enabled by the Space Launch System (SLS) and the Orion spacecraft, pretends to return humans to the Moon by 2024, and establish a sustainable exploration of the satellite by 2028.
A.2 Earth Departure Trajectories

Examples of ED trajectories for various eccentric anomaly spans of the pre-escape perigee burns are presented in the following figures, with the thrusting arcs coloured in red. The coordinate axes have different scales.

Figure A.1: ED trajectory for $\Delta E_T = 5^\circ$.

Figure A.2: ED trajectory for $\Delta E_T = 30^\circ$. 
Figure A.3: ED trajectory for $\Delta E_T = 60^\circ$.

Figure A.4: ED trajectory for $\Delta E_T = 90^\circ$.
A.3 Lambert Problem Interplanetary Transfer Trajectories

The Lambert Problem ITs concerning the best, 1st quartile, median and 3rd quartile launch dates of TC1 and TC2 are shown in the following figures.

Figure A.5: Lambert Problem IT for the best launch date of both TC1 and TC2, and 1st quartile launch date of TC1.

Figure A.6: Lambert Problem IT for the 1st quartile launch date of TC2.
Figure A.7: Lambert Problem IT for the median launch date of both TC1 and TC2.

Figure A.8: Lambert Problem IT for the 3rd quartile launch date of both TC1 and TC2.
A.4 Continuous Thrust Interplanetary Transfer

A.4.1 Optimisation Algorithms

The 3 algorithms mentioned in Section 4.3.2 used in the optimisation of the continuous thrust ITs are described in the following sections.

Extended Ant Colony Optimisation

Real ants lay down pheromones directing each other to resources while exploring their environment. In ant colony optimisation (ACO), simulated ants record their positions and solutions quality so that in later simulation iterations more ants locate better solutions. [59]

PyGMO’s ACO version, GACO, is based on the algorithm of Schlueter et al. [120]. This ACO methodology works by the incremental construction of solutions regarding probabilistic choices according to probability density functions (PDFs). Each PDF $G^i(x)$ respects one optimisation variable $i$, and consists of a weighted sum of several one-dimensional normal distributions as described by (3) of [120]:

$$G^i(x) = \sum_{l=1}^{k} w^i_l \cdot g^i_l(x) = \sum_{l=1}^{k} w^i_l \cdot \frac{1}{\sigma^i_l \sqrt{2\pi}} e^{-\frac{(x-\mu^i_l)^2}{2\sigma^i_l^2}}, \quad \text{(A.1)}$$

where $k$ is the number of normal distributions used within $G^i(x)$, the weights $w^i_l$ are computed with a linear proportion ($w^i_1$ is $k$ times larger than $w^i_k$), the means $\mu^i_l$ are given directly by the components of the solutions saved in the solution archive (in which the most promising solutions and respective objective function values are stored), and the standard deviations $\sigma^i_l$ are calculated by exploiting the variety of solutions in the solution archive.

The incremental construction of the PDF works as follows:

- A mean $\mu^i_l$ is chosen for every dimension $i$ according to the weights ($\mu^i_1$ is $k$ times more likely to be chosen than $\mu^i_k$).
- A new solution component is created by generating random numbers around $\mu^i_l$ with deviations $\sigma^i_l$. The new solution only impacts on $G^i(x)$ if it is one of the $k$ best ranked solutions.

PyGMO’s GACO input parameters and respective default values are [59]:

- $gen = 1$ (generations): number of generation to evolve.
- $ker = 63$ (kernel): number of solutions stored in the solution archive.
- $q = 1$ (convergence speed parameter): for managing the convergence speed towards the found minima (the smaller the faster).
- $oracle = 0$ (oracle parameter): used in the oracle penalty method (see [121]).
- $acc = 0.01$ (accuracy parameter): for maintaining a minimum distance between penalty function values.
threshold = 1 (threshold parameter): when the generations reach the threshold, \( q \) is set to 0.01 automatically.

\( n_{\text{gen,mark}} = 7 \) (standard deviations convergence speed parameter): determines the convergence speed of the standard deviations values.

impstop = 100 000 (improvement stopping criterion): if a positive integer is assigned here, the algorithm will count the runs without improvements, and will be stopped if the counter exceeds impstop.

evalstop = 100 000 (evaluation stopping criterion): similar to impstop, but concerns function evaluations.

focus = 0 (focus parameter): makes the search for the optimum greedier and more focused on local improvements (the higher the greedier). If the value is very high, the search is more focused around the current best solutions.

memory = false (memory parameter): if true, memory is activated in the algorithm for multiple calls.

seed = pagmo :: random_device :: next(): seed\(^6\) used by the internal random number generator (default is random).

### Improved Harmony Search

Harmony search (HS) is an algorithm said to mimick the improvisation process of musicians. In the metaphor, the orchestra attempts to find the best harmony by means of each musician playing a harmony, i.e. each variable generates a solution with the ultimate goal of finding the global optimum. [59]

PyGMO’s IHS is based on the improved HS from Mahdavi et. all [122]. However, while the version of [122] applies to continuous unconstrained single-objective deterministic problems, PyGMO’s version was generalised to tackle discrete multi-objective stochastic problems. [59]

In HS, a new harmony (set of optimisation variables) is generated as follows [122]:

1. **Memory consideration.** A value for the decision variable is chosen from historically stored values with probability \( HMCR \) (harmony memory considering rate), or from all possible values with probability \( 1 - HMCR \).

2. **Pitch adjustment.** With probability \( PAR \) (pitch adjustment rate), a random value, following a uniform distribution in \([-bw, bw]\), is added to the value selected in the memory consideration.

3. **Harmony memory update.** If the new harmony is better than the worst current harmony, the new harmony is included in the memory, and the worst is excluded.

The key difference in improved HS is that the pitch adjustment rate and bandwidth \( bw \) are not constant. As illustrated in Fig. 2 of [122], across the generations, the pitch adjustment rate increases

\(^6\)When a pseudo-random number generator is called with the same seed, the output does not vary.
linearly, and the bandwidth decreases exponentially.

PyGMO’s IHS input parameters and respective default values are [59]:

- \( \text{gen} = 1 \) (generations): number of generation to evolve.
- \( \text{phmc}r = 0.85 \) (probability for the harmony memory considering rate).
- \( \text{ppar}_{\text{min}} = 0.35 \) and \( \text{ppar}_{\text{max}} = 0.99 \) (minimum and maximum probability for the pitch adjustment rate).
- \( \text{bw}_{\text{min}} = 10^{-5} \) and \( \text{bw}_{\text{max}} = 1 \) (minimum and maximum distance bandwidth).
- \( \text{seed} = \text{pagmo} :: \text{random} \_\text{device} :: \text{next()} \): seed used by the internal random number generator.

**Constrained Optimisation by Linear Approximations**

\( \text{NLopt} \)’s COBYLA is derived from Powell’s implementation described in [123]. The algorithm constructs successive linear approximations of the objective function and problem constraints through a simplex\(^7\) of \( N + 1 \) points (\( N \) dimensions for the optimisation variables, and one for the objective function), and optimises these approximations in a trust region\(^8\) at each step [125].

\( \text{NLopt} \)’s COBYLA slightly modifies the Powell’s implementation. The trust-region radius is not constant, being increased if the predicted improvement of the objective function is approximately correct. The simplex steps are pseudo-randomised, improving robustness by avoiding steps which do not improve conditioning, but the algorithm remains deterministic owing to a deterministic seed. In order to deal with significantly different optimisation parameters scales, there is support for different step sizes. [125]

Unlike GACO and IHS, COBYLA is not population based, and its optimisation algorithm does not require any input parameters.

---

\(^7\) A simplex, sometimes called a hypertetrahedron, is the generalisation of a tetrahedral region of space to \( n \) dimensions. The one-dimensional such region is a line segment, the two-dimensional one an equilateral triangle, and the three-dimensional one a tetrahedron. [124]

\(^8\) In mathematical optimisation, a trust region is the subset of the domain of the objective function which is approximated using a model function.
A.4.2 Trajectories and Thrust Profiles

The optimised continuous thrust IT trajectories and thrust profiles for the 8 NEOs worth targeting in TC3 (see Fig. 5.15) are shown in the following figures.

Figure A.9: Optimised continuous thrust IT trajectory for 2000 SG344.

Figure A.10: Optimised continuous thrust IT thrust profile for 2000 SG344.
Figure A.11: Optimised continuous thrust IT trajectory for 2014 QN266.

Figure A.12: Optimised continuous thrust IT thrust profile for 2014 QN266.
Figure A.13: Optimised continuous thrust IT trajectory for 2014 YD.

Figure A.14: Optimised continuous thrust IT thrust profile for 2014 YD.
Figure A.15: Optimised continuous thrust IT trajectory for 2015 DU.

Figure A.16: Optimised continuous thrust IT thrust profile for 2015 DU.
Figure A.17: Optimised continuous thrust IT trajectory for 2016 TB18.

Figure A.18: Optimised continuous thrust IT thrust profile for 2016 TB18.
Figure A.19: Optimised continuous thrust IT trajectory for 2017 SV19.

Figure A.20: Optimised continuous thrust IT thrust profile for 2017 SV19.
Figure A.21: Optimised continuous thrust IT trajectory for 2018 LQ2.

Figure A.22: Optimised continuous thrust IT thrust profile for 2018 LQ2.
Figure A.23: Optimised continuous thrust IT trajectory for 2018 PK21.

Figure A.24: Optimised continuous thrust IT thrust profile for 2018 PK21.
A.5 Trajectory Concepts Global Optimisation

The optimum eccentric anomaly span of the pre-escape perigee burns, NEO target, IT starting date and duration, Lambert Problem IT number of complete revolutions (only for TC1 and TC2), initial SC mass, maximum NEO operations duration (only for TC2 and TC3) and final-to-initial-mass ratio are presented in Tables A.3 to A.5 as a function of the launch date ranking. As discussed in Section 4.4.1, the higher the final-to-initial-mass ratio, the better the launch date.

Table A.3: Global optimisation results for TC1.

<table>
<thead>
<tr>
<th>Ranking</th>
<th>$t_0$ (UTC)</th>
<th>$\Delta E_T$ (°)</th>
<th>NEO</th>
<th>$t_{dep}$ (UTC)</th>
<th>$\Delta t_{IT}$ (d)</th>
<th>$N_{IT}$</th>
<th>$m_0$ (kg)</th>
<th>$m_f / m_0$ (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>2027-03-21.0–2028-12-30.0</td>
<td>6</td>
<td>2000 SG344</td>
<td>2029-06-08.0</td>
<td>280</td>
<td>0</td>
<td>57.6</td>
<td>27.8</td>
</tr>
<tr>
<td>2</td>
<td>2029-01-09.0–2029-02-18.0</td>
<td>8</td>
<td>2000 SG344</td>
<td>2029-06-08.0</td>
<td>280</td>
<td>0</td>
<td>57.7</td>
<td>27.7</td>
</tr>
<tr>
<td>3</td>
<td>2029-02-28.0–2029-03-10.0</td>
<td>10</td>
<td>2000 SG344</td>
<td>2029-06-08.0</td>
<td>280</td>
<td>0</td>
<td>57.9</td>
<td>27.6</td>
</tr>
<tr>
<td>4</td>
<td>2029-03-20.0</td>
<td>11</td>
<td>2000 SG344</td>
<td>2029-06-08.0</td>
<td>280</td>
<td>0</td>
<td>58.0</td>
<td>27.6</td>
</tr>
<tr>
<td>5</td>
<td>2029-03-30.0</td>
<td>12</td>
<td>2000 SG344</td>
<td>2029-06-08.0</td>
<td>280</td>
<td>0</td>
<td>58.2</td>
<td>27.5</td>
</tr>
<tr>
<td>6</td>
<td>2029-04-09.0</td>
<td>14</td>
<td>2000 SG344</td>
<td>2029-06-08.0</td>
<td>280</td>
<td>0</td>
<td>58.3</td>
<td>27.4</td>
</tr>
<tr>
<td>7</td>
<td>2029-04-19.0</td>
<td>17</td>
<td>2000 SG344</td>
<td>2029-06-08.0</td>
<td>280</td>
<td>0</td>
<td>58.4</td>
<td>27.4</td>
</tr>
<tr>
<td>8</td>
<td>2025-08-28.0–2027-03-11.0</td>
<td>6</td>
<td>2000 SG344</td>
<td>2027-10-07.0</td>
<td>320</td>
<td>0</td>
<td>58.7</td>
<td>27.3</td>
</tr>
<tr>
<td>9</td>
<td>2029-04-29.0</td>
<td>21</td>
<td>2000 SG344</td>
<td>2029-06-08.0</td>
<td>280</td>
<td>0</td>
<td>59.0</td>
<td>27.1</td>
</tr>
<tr>
<td>10</td>
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<td>28</td>
<td>2000 SG344</td>
<td>2029-06-08.0</td>
<td>280</td>
<td>0</td>
<td>59.9</td>
<td>26.7</td>
</tr>
<tr>
<td>11</td>
<td>2025-08-18.0</td>
<td>6</td>
<td>2000 SG344</td>
<td>2027-09-27.0</td>
<td>320</td>
<td>0</td>
<td>60.8</td>
<td>26.3</td>
</tr>
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<tr>
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<td>25.8</td>
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<td>2000 SG344</td>
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<td>280</td>
<td>0</td>
<td>64.7</td>
<td>24.7</td>
</tr>
<tr>
<td>15</td>
<td>2024-09-12.0–2025-07-29.0</td>
<td>6</td>
<td>2000 SG344</td>
<td>2026-10-12.0</td>
<td>330</td>
<td>0</td>
<td>65.1</td>
<td>24.6</td>
</tr>
</tbody>
</table>

For the Lambert Problem ITs, the HTS is ejected upon arrival to the target NEO, therefore the final SC mass is simply $m_f = m_{CS} = 16$ kg.
<table>
<thead>
<tr>
<th>Ranking</th>
<th>$t_0$ (UTC)</th>
<th>$\Delta E_T$ (°)</th>
<th>NEO</th>
<th>$t_{dep}$ (UTC)</th>
<th>$\Delta t_{IT}$ (d)</th>
<th>$N_{IT}$</th>
<th>$m_0$ (kg)</th>
<th>$m_f / m_0$ (%)</th>
</tr>
</thead>
<tbody>
<tr>
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<td>2000 SG344</td>
<td>2026-10-02.0</td>
<td>330</td>
<td>0</td>
<td>67.1</td>
<td>23.9</td>
</tr>
<tr>
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<td>2000 SG344</td>
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<td>280</td>
<td>0</td>
<td>70.3</td>
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</tr>
<tr>
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<td>6</td>
<td>2000 SG344</td>
<td>2026-09-22.0</td>
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<td>0</td>
<td>73.2</td>
<td>21.8</td>
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<tr>
<td>19</td>
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<td>330</td>
<td>0</td>
<td>76.2</td>
<td>21.0</td>
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<td>330</td>
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<td>0</td>
<td>76.6</td>
<td>20.9</td>
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<tr>
<td>23</td>
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