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

Vasco Amaral Grilo
vasco.amaral.grilo@tecnico.ulisboa.pt

Instituto Superior Técnico, Universidade de Lisboa, Portugal

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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

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.

2. Mission Architecture

2.1. 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 [1], which aims to reach Mars departing from a supersynchronous GTO, 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 2.4. In fact, ESA’s 12 U M-ARGO CubeSat [2], which aims to perform a rendezvous with a NEO departing from the Sun-Earth L2 point, only has room for 1 U of payload. Consequently, accommodating an additional high-thrust propulsion systems in the proposed CubeSat
would hardly be feasible.

2.2. 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 impulses 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 [3, 4]. 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 relative flyby speed of 100 m/s (see Section 2.4) represents a small fraction of the CubeSat low-thrust propulsion system Delta-v capacity.

2.3. CubeSat Scientific Payload Estimation
The CubeSat scientific payload selection goes beyond the scope of this work, but its mass was estimated for determining the SC mass breakdown. The estimates resulted in $m_{\text{LP}} \approx 12.0$ kg for the Lambert Problem ITs (TC1 and TC2), and $m_{\text{CT}} \approx 1.3$ kg for the continuous thrust ITs (TC3).

The above values were obtained supposing a payload mean density equal to the CubeSat mean density of 1.33 kg/U introduced in Section 2.4 and considering payload volumes of $V_{\text{LP}} = 9$ U and $V_{\text{CT}} = 1$ U for the Lambert Problem and continuous thrust ITs, respectively. $V_{\text{LP}}$ and $V_{\text{CT}} = 8$ U were assumed equal to the ESA’s M-Argo CubeSat payload and low-thrust propulsion system (LTPS) volumes [2, 5], respectively.

2.4. 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; timely preparing a mission for an earlier launch would be difficult, and predictions concerning geosynchronous satellites demand for the 2030s are not reliable.
- Maximum mission duration $\Delta t_{\text{max}} = 3$ a; maximum IT duration for ESA’s M-Argo CubeSat mission [5] which supports a low operations cost.
- Maximum NEO periapsis radius $r_{\text{p}} = 1.3$ AU: NEO definition [7].
- Minimum NEO diameter $D_{\text{min}} = 15$ 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 CubeSat mission [8].
- Maximum Minor Planet Centre (MPC) U parameter $U_{\text{max}} = 6$: NEOs with higher orbit uncertainty were not considered in the NEA Scout CubeSat mission [8].
- Maximum speed of the CubeSat relative to the NEO at flyby $v_{\text{FB}} = 100$ m/s; speed taken from the NEA Scout CubeSat mission [8] which allows for thorough characterisation of the target asteroid upon arrival.
- Maximum Earth-NEO distance for the NEO operations $d_{\text{E-NEO}} = 1$ AU: distance taken from the NEA Scout mission [8] which results in a reasonable communication range for a CubeSat antenna.
- Minimum duration for the NEO operations $\Delta t_{\text{NEO,min}} = 0.5$ a: duration of the M-Argo mission NEO operations [2] which ensures a thorough analysis of the NEO and transmission of the data to Earth.
- CubeSat size $V_{\text{CS}} = 12$ 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 $m_{\text{CS}} = 16$ 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 (1.8 kg/U [2]), NEA Scout (2 kg/U [5]) and MARIO (2 kg/U [1]) interplanetary CubeSats.
- Maximum initial SC mass $m_{\text{0,max}} = 100$ kg: implies that the SC could still be classified as a microsatellite [9].
2.5. Initial Geosynchronous Transfer Orbit

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 launches from the Kourou Europe Spaceport to GTO have been considered. From the respective user manuals \[10, 11\], the perigee and apogee altitudes are, respectively, 250 km and 35 943 km (equal values for both launchers). This information is sufficient, for a parabolic escape (see Section 2.2) and disregarding perturbations, to determine the ED \( \Delta v \) and duration.

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 \[11\], numerous piggyback launch opportunities are expected. Secondly, Arianespace features a rideshare launch service solution for Ariane 6 (see 5.4.3 of \[11\]) 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 \[10\], 4.5 t–5.0 t for Ariane 62, and 11.5 t for Ariane 64 \[11\]. This means the maximum initial SC mass of 100 kg (see Section 2.4) amounts to 0.9%–2.2% of the maximum payload capacity.

3. Spacecraft Propulsion

3.1. Margin Philosophy

The margins used for this work were based on requirements presented in the ESA margin philosophy for science assessment studies report \[12\].

The propellant margin \( PM = 2\% \) (requirement R-M1-6 of \[12\]) accounts for propellant residuals, and affects the propellant mass \( m_p \) via

\[
m_p = (1 + PM) m_p'.
\]

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 \( OSM = 10\% \) (requirement R-M2-9 of \[12\]), what implies oversized propellant tanks. This margin impacts on the initial SC mass \( m_0 \) through

\[
m_0 = (1 + OSM) m_0'.
\]

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

The Delta-v margin \( DVM = 5\% \) (requirement RDV-11 of \[12\]) covers uncertainties in mission design and system performance by updating an accurately calculated Delta-v, which should include the effect of gravity losses, to

\[
\Delta v = (1 + DVM) \Delta v''.
\]

The double prime superscript refers to a Delta-v obtained for null Delta-v margin.

3.2. High-thrust Stage

For the HTS, an oversized version of the Tethers Unlimited water thruster HYDROS-M \[13, 14\] (green propulsion), referred to as HYDROS-L (HYDROS-Large), was selected. Overscaling was performed because no CubeSat HTS was found to have a Delta-v capacity larger than the 772 m/s respecting the impulsive perigee kick to escape from the initial GTO, which underestimates the actual HTS Delta-v budget. The scaling-up process of HYDROS-M to HYDROS-L is addressed in the following section.

3.2.1 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, \( \Delta v_{HTS} \).

The mentioned relation depends on how the propellant tank mass scales with the propellant mass. As the HYDROS-M tank resembles a cylinder \[14\], the propellant mass \( m_t \), and propellant mass \( m_p \) are:

\[
m_t = \rho_t (2 \pi R^2 + 2 \pi R h) t; \quad (4)
\]

\[
m_p = \rho_p \pi R^2 h. \quad (5)
\]

\( \rho_t \) 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 \[15\], 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. (4) and (5) imply that

\[
m_t = m_{to} \frac{m_p}{m_{po}^2} m_p^{\frac{2}{3}}, \quad (6)
\]

where the baseline HYDROS-M tank and propellant masses are, respectively, \( m_{to} = 2.5\text{ kg} \) and \( m_{po} = 6.2\text{ kg} \) \[15\].

The SC comprises the CubeSat and HTS (see Section 2.1), therefore its mass at injection is

\[
m_0 = m_{CS} + m_{HTS}. \quad (7)
\]

The HTS mass could be written as

\[
m_{HTS} = m_{HTS}^* + m_t + m_p, \quad (8)
\]
where \( m_{\text{HTS}}^0 = 5.2 \text{ kg} \) is the non-scalable HYDROS-M mass [13]. The ratio between the initial SC mass and the mass after all the HTS manoeuvres for null propellant and overscaling margins is

\[
\frac{m_0'}{m_0} - m_{p',*} = e^{\Delta v_{\text{HTS}} / \gamma_{\text{HTS}}} = \frac{\Delta v_{\text{HTS}}}{(1 + \text{PM})(1 + \text{OSM})}. \tag{9}
\]

where \( g_0 = 9.806 \text{ m}^2/\text{s}^2 \) [10] is the standard gravity acceleration at the Earth surface, \( I_{sp} \) is the HTS specific impulse, and \( \Delta v_{\text{HTS}} \) is the normalised HTS Delta-v budget.

Combining Eqs. (6) to (9) with the margins defined in Section 3.1 leads to the following equation which implicitly defines the propellant mass:

\[
m_p + A = B \, m_p^2 \Leftrightarrow m_p^3 + (3A - B^3) \, m_p - 3A^2m_p + A^3 = 0; \tag{10}
\]

\[
A = \left[ 1 + \text{PM} (1 + \text{OSM}) - \Delta \hat{v}_{\text{HTS}} \right] ; \tag{11}
\]

\[
B = \frac{m_{p,0}^3}{m_{p,0}} \left[ (1 + \text{PM})^2 \Delta \hat{v}_{\text{HTS}} - (1 + \text{OSM}) + \text{OSM} \right] . \tag{12}
\]

The real positive root of the cubic equation, the only meaningful value for the propellant mass, was calculated with roots from the numpy python toolbox. Subsequently, the propellant tank mass was computed from Eq. (6), the HTS mass from Eq. (5), and the initial SC mass from Eq. (7).

The initial SC mass, and HTS propellant mass and diameter as a function of the HTS Delta-v budget are presented in Fig. 1. The crosses respect a parabolic escape enabled by an instantaneous perigee kick, and therefore represent the HTS Delta-v budget for an ED without gravity losses. The curves of Fig. 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.

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 [14]. These correspond to conservative assumptions as, the lower the thrust and specific impulse, the smaller is the thrust induced acceleration, and therefore the larger the Delta-v budget, and the longer the ED duration (what in turn also implies a larger radiation dose).

3.3. CubeSat Low-thrust Propulsion System

For the CubeSat LTPS, ArianeGroup’s RIT (Radiofrequency Ion Thruster) \( \mu \text{X} \) [17] was selected.

This thruster is part of the M-ARGO CubeSat [2] (described in Section 2.1), to be launched in 2023–2025. The thruster model is described in [6], and is integrated in the PyKEP class \texttt{trajopt.\_lt\_margo}, which was used to define the continuous thrust IT optimisation problem (see Section 4.3.1). 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 3102 s.

4. Trajectory Design

4.1. Earth Departure

The ED ARs were based on those of the MARIO CubeSat mission [1] (described in Section 2.1), whose parabolic escape Delta-v is only 1% higher than the ideal one of an instantaneous perigee kick [3]. The following characterises the respective ARs [3]:

- 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 \( (\alpha = E) \) because it is readily returned by \texttt{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 [3]:

Figure 1: Initial SC mass, HTS propellant mass and HTS propellant tank diameter as a function of the HTS Delta-v budget.
• The thrust activation arcs were defined relative to the instantaneous perigee, not to the pre-burn one as in [9], 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 [9], 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° starting from 5°, instead of set to the eccentric anomaly span of the previous burns (\( \Delta E_T \)).

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

\[
\Delta v_{\text{ED,AR}}^m (m_0) = (1 + \text{DVM}) \Delta v_{\text{ED,AR}}^0 (m_0) = \Delta v_{\text{HTS}} (m_0). \tag{13}
\]

The ARs Delta-v for null Delta-v margin (\( \Delta v_{\text{ED,AR}}^0 \)) 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 (\( \Delta v_{\text{HTS}} \)) was taken from Fig. 1.

The solution of Eq. (13) was found with the bisection method (see, for instance, [18]). The lower bound of the initial SC mass range was initialised to \( m_{0,\text{min}} \approx 35.9 \text{ kg} \), which is the initial SC mass for null gravity losses (given by the blue cross of Fig. 1). The upper bound was initialised to the maximum initial SC mass of \( m_{0,\text{max}} = 100 \text{ kg} \) (see Section 2.4). The iterative process was interrupted as soon as the solution range became smaller than \( \Delta m_{0,\text{max}} = 0.1 \text{ kg} \). As a consequence, by performing the last initial SC mass update to the middle value of the final range, the maximum error is \( \Delta m_{0,\text{max}} / 2 = 0.05 \text{ kg} \).

The ED trajectory for \( \Delta E_T = 5^\circ \) is shown in Fig. 2 with the thrusting arcs coloured in red (the coordinate axes have different scales).

4.2. Lambert Problem Interplanetary Transfer
The Lambert Problem ITs were solved with Izzo’s implementation [19] for 4 variable parameters:

- \( \text{NEO in the MPCORB database [20], with diameter defined in the ESA Space Situational Awareness website database [21], satisfying the diameter and MPC U parameter constraints presented in Section 2.4} \)
- \( \text{IT starting date} t_{\text{dep}} \text{ between} 2023-01-01 \text{ and} 2030-01-01 \text{ (step} 10 \text{ d).} \)
- \( \text{IT duration} \Delta t_{\text{IT}} \text{ between} 10 \text{ d and} 3 \text{ a (step} 10 \text{ d).} \)
- \( \text{IT maximum number of complete revolutions calculated, as suggested by Izzo in algorithms 1 and 2 of [19], from} \ N_{\text{IT max}} = \sqrt{2 \mu_S / s^2 \Delta t_{\text{IT}} / \pi} \), 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, \( \vec{V}_{\text{E}} \text{ and} \vec{V}_{\text{NEO}} \), respectively, were calculated with \( \text{pykep.planet.jplLsp.eph} \). The SC velocity at the start and end of the IT, \( \vec{V}_{\text{E}} \text{ and} \vec{V}_{\text{NEO}} \), respectively, were obtained with Izzo’s Lambert solver. Subsequently, the departure Delta-v, arrival Delta-v for the rendezvous, relative flyby speed, and arrival Delta-v for the flyby were computed from:

\[
\Delta v_{\text{dep}} = \left| \vec{V}_{\text{E}}^\prime - \vec{V}_{\text{E}} \right| ; \tag{14}
\]

\[
\Delta v_{\text{arr,RDV}} = \left| \vec{V}_{\text{arr,NEO}} - \vec{V}_{\text{arr,NEO}} \right| ; \tag{15}
\]

\[
\nu_{\text{FB rel}} = \min \{ \Delta v_{\text{arr,RDV}}, v_{\text{FB rel}} \}; \tag{16}
\]

\[
\Delta v_{\text{arr,Fin}} = \Delta v_{\text{arr,RDV}} - \nu_{\text{FB rel}}. \tag{17}
\]

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

\[
\Delta v_{\text{TC1,TC2}}^{\prime\prime} = \Delta v_{\text{ED,AR}}^{\prime\prime} + \Delta v_{\text{dep}} + \Delta v_{\text{arr,RDV}}. \tag{18}
\]
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. 1.

The optimum Lambert Problem IT for the best launch date of both TC1 and TC2 is presented in Fig. 3.

4.3. Continuous Thrust Interplanetary Transfer

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, IT duration, and thrust components for each of the n_{seg} IT thrusting segments.

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

- NEO for which there is at least one Lambert Problem IT rendezvous Delta-v $\Delta v_{dep} + \Delta v_{arr,RDV}$ (see Eqs. (14) and (15)) smaller than 2.5 km/s, while satisfying the Earth-NEO distance constraint introduced in Section 2.4 at the NEO arrival.

- Number of thrusting segments belonging to the set \{10;20;30\}.

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. (2):

$$m_{CS,OS} = (1 + OSM) m_{CS} = 17.6 \text{ kg}.$$  \hspace{1cm} (19)

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.

- Earth gravity was not considered for the IT, in agreement with the patched conics method (see, for example, [22]) and the Lambert Problem ITs.

- The solar electric propulsion model (see Section 3.3) 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 [6].

The tolerance ($c_{tol}$ 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 ($V_{E}$ ≈ 29.8 km/s [16]), respectively. Thus the maximum errors in the position and velocity components are $1.5 \cdot 10^{-3}$ km and 0.30 m/s, respectively.

4.3.2 Optimisation Algorithms

The 3 algorithms for which optimisation archipelagos were defined are the PyGMO’s extended ant colony optimisation [23] (GACO), improved harmony search [24] (IHS), and constrained optimisation by linear approximations [25] (COBYLA). These are the only 3 algorithms implemented in PyGMO suitable for the M-ARGO (continuous non-linearly constrained single objective deterministic) Problem.

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, but the optimisation was stopped when all the decision variables changed less than the NLopt’s default relative tolerance ($stol_{rel}$ 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 $10^{-8} t_{\text{dep},\text{max}} [\text{mjd2000}] = 10^{-8} \cdot 11323 \text{ d} \approx 10 \text{ s}$ 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.
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 previously highlighted concepts were defined as follows:

- **Island**: type `pygmo.mp_island`, which relies on the Python standard `multiprocessing` module; optimisation algorithms mentioned in Section 4.3.2; optimisation problem presented in Section 4.3.1; population size \( N_p \in \{5; 10; 50; 100; 500\} \).
- **Topology**: fully connected archipelago topologies (class `pygmo.fully_connected`) with the optimisation algorithms presented in Table 1; unitary migration probability for all island connections.
- **Selection policy**: best selection policy, which selects the individuals of highest final CubeSat mass; unitary selection rate, such that all individuals could be selected for the migration database.
- **Replacement policy**: fair replacement policy, which maintains the individuals of highest final CubeSat mass; unitary replacement rate, such that all individuals could be replaced in the migration database.
- **Migration type**: broadcast migration, which considers individuals from all the connecting islands.
- **Migrant handling policy**: preserve policy, which maintains a copy of the previous candidate migrants in the migration database.

Regarding Table 1 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>
<thead>
<tr>
<th>Archipelago ID (topoID): islands</th>
</tr>
</thead>
<tbody>
<tr>
<td>0: 1 COBYLA // 1: 3 GACO // 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>
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4.3.4 Archipelago Optimised Solution and Selection

After performing all the archipelago evolutions, the feasibility of the best (highest final mass) solution of each island (its champion) is checked. From the champions for which the feasibility call returns True, and the Earth-NEO distance and NEO operations duration constraints are satisfied (see Section 2.4), 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 (topoID), island population size \( N_p \) and number of evolutions \( N_g \). Based on the ranges defined for each of these parameters, there are 390 candidate archipelagos for each NEO. The archipelago selected for a given NEO is that whose optimised solution respects the highest final mass.

The optimised trajectory and thrust profile for the best NEO are shown in Figs. 4 and 5.

**Figure 4**: Optimised continuous thrust IT trajectory for 2000 SG344.

5. Results and Discussion

The optimum (highest final-to-initial-mass ratio) initial SC mass, maximum NEO operations duration and NEO target as a function of the launch date are shown in Figs. 6 and 7 for TC2 and TC3.
The mission is feasible, satisfying all constraints of Section 2.4 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, there are only 4 NEOs worth targeting for TC1 and TC2, and 8 for TC3.

The mass breakdowns and HTS propellant tank diameters for the optimised trajectories of the best (maximum final-to-initial-mass ratio), 1st quartile, median and 3rd quartile launch dates are presented in Figs. 8 and 9 for TC2 and TC3. 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 %) to 37 kg for TC3.

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 mul-
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.

References


