TRANSFER AND RENDEZ-VOUS STRATEGIES FOR THE DEPLOYMENT AND THE SERVICING OF AN INHABITED SPACE STATION AT EARTH-MOON L2

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The next step for human exploration in the solar system could be the deployment of an inhabited station at Earth-Moon Libration points (EML), as a gateway for further destinations such as the Moon (lunar surface settlement), Mars or asteroids, according to International Space Exploration Coordination Group (ISECG) roadmap [1] and several recent publications [5]. In this context, this paper examines how to design a low cost mission, using the natural dynamics for Station integration, crew rotations, cargo delivery and disposal. Preliminary studies lead us to select a Halo orbit around EML2 to locate the space station. Then, the entire trajectory, from the selection of the departure Low Earth Obit to the rendez-vous strategy in EML2, was analyzed with several possible transfer types (direct, indirect, lunar flyby or weak stability boundary). Actually, optimization criteria strongly depend on the mission phase. For instance, when crew transit is considered, mission duration has to be minimized, while in the case of cargo transportation, we rather seek to optimize the global delta-v. This paper presents the results (in term of duration and cost) obtained for the two strategies we selected: lunar flybys for the crew and weak stability boundaries trajectories for cargo. We carefully considered the constraints for rendez-vous in EML2 and evaluated their impact on the performances. Moreover, we assessed the impacts of the model selection (ephemeris, four bodies versus restricted circular three bodies problem...). The main contribution of this project lies in the global optimization of the entire mission from LEO to EML2 and return with regards to two criteria (mission duration and delta-v), with a focus on the rendez-vous feasibility in EML2.

I. INTRODUCTION

Solar system exploration represents one of the last frontiers for human desire of conquest and discovery. Human spaceflight programs have enabled the access to Low Earth Orbits (LEO), even for private spaceships now, but no crew has had the opportunity to visit the Moon vicinity again since Apollo missions. Space agencies participating in ISECG (International Space Exploration Coordination Group [1]) have elaborated a long-range exploration strategy, starting from the ISS (International Space Station) up to expanding human presence to Mars surface, as a final goal. Locating an inhabited space station at Earth-Moon Lagrangian (EML) point seems to be a promising option to ease connection between targeted destinations like the Moon, Mars or NEOs (Near-Earth Objects) and provide services to those missions.

At ISAE/Supaero a research project to design such a space station has been initiated [2], which mainly focuses on the station architecture [3] and the entire trajectory optimization. Several scenarios to reach EML neighborhood from the Earth (Station deployment, crew rotation, cargo transportation) have been identified, corresponding to different types of transfers (lunar flyby, weak stability boundary trajectories). Trajectory optimization in the three-body problem is a very well known subject, largely described in literature since Farquhar's first publication [4]. Nevertheless, the rendez-vous problem in the vicinity of the Lagrangian points has rarely been addressed (except [21], [6]). Yet, in the studied scenarios, rendez-vous strategy is a crucial step in the optimization of the transfer from LEO to EML. Preliminary results related to this project have recently been published [7]. The main goal of the current paper is to present complementary results, and particularly a methodology developed to optimize the rendez-vous of a cargo with an inhabited space station located at Earth-Moon Lagrangian point 2 (EML2). The main contribution of this study is to explore the optimization of an entire trajectory from LEO to EML and above all, to suggest methodology to design the rendez-vous in the Lagrangian point vicinity in human spaceflight context.

Section II describes the project context with mission analysis results and space station architecture. In Section III, the theoretical background of the three-body problem is reminded. Section IV deals with the results obtained for the transfer optimization for the various studied scenarios. It is completed by the fourth part, focusing on the rendez-vous. In the last part, project perspectives are evoked.

II. THOR STATION CONTEXT

Previous space agencies projects [8] and ISEGC roadmap [1] clearly highlight the importance of Lunar Lagrangian points in the solar system exploration roadmap. Operating a space station located at EML2 could generate numerous benefits to strengthen international cooperation in science and technology and guarantee a safe and permanent human presence in space outside the Earth cradle. Such a space station could:

- Support exploration teams based on Moon surface

- Relay communications with Earth
- Shield the crew in case of SPE (Solar Particle Events) and protect them against GCR (Galactic Cosmic Rays)
- Procure food, water and fuel
- Embed scientific instruments (like telescopes)
- Offer medical support

Taking into account all those functional needs, an inhabited space station, named THOR (Trans-lunar Human explORation) has been designed ([2], [3]). The architecture is mainly composed of seven cylindrical modules based on ATV (Automated Transfer Vehicle) proportions (a mass of twenty tons, a diameter of five meters and a length of ten meters each), completed by two spheres. Each cylindrical module support a specific function (as room, offices, kitchen, medical center, cult area...) while the spheres are added to ease displacements inside the station and offer windows on space, like the Cupola on board the ISS. The following picture depicts the THOR station functions allocation.

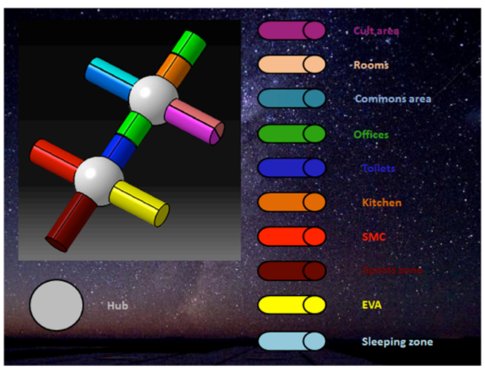


Figure 1: THOR station configuration [9]

Mission analysis led to the conclusion that the optimal location for the THOR station could be a Halo orbit around EML2 point. Concepts of Lagrangian point and Halo orbit will be briefly reminded in part III. Actually, Lagrangian points are easily accessed from Earth, Moon or Mars with minimum launch window constraints and low fuel consumption for station keeping. Compared to LEO, there is no artificial debris hazard. Moreover, a decision matrix presented in [2], based on criteria such as crew access from Earth, deployment and resupply efficiency, access to lunar location, communications station keeping, exploration capabilities, long term strategy risk and human factor (it will be the first time for a crew to test life behind the Moon, without a permanent visual contact with the planet Earth) justifies the choice of EML2 as a space outpost where to set a Deep Space-Habitat.

THOR life-profile decomposition brought the conclusion that three important phases had to be carefully designed: the station deployment, the crew transportation (there and back) and the cargo transfer. Even if the expected performances (in term of duration and fuel consumption) vary from one phase to the other, the main legs of the cargo and the crew trajectories remain the same: Launch, station keeping in LEO (1), transfer (2), rendez-vous (3), station keeping on the Halo orbit (4) and return (5) as it is described on the following figure.

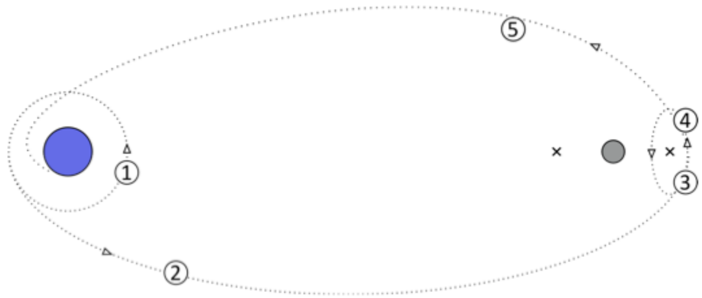


Figure 2: Trajectories main legs

The station deployment phase consists in transporting the seven modules and the two spheres from earth surfaces to the EML2. In this case, the return is not considered, but the main challenge is to find the optimal assembly scenario: is it better to integrate the module in LEO, at EML2 or somewhere else. The preliminary baseline considers that the resupply vehicle frequently delivers cargo to the Station. The operational lifetime is projected to be fifteen years.

This paper focuses on methodology and results obtained for transfer and rendez-vous optimization while duration and delta-v (velocity increment) are considered. The optimization scenario at system level of the global station mission is detailed in [10].

III. THEORETICAL BACKGROUND

This paragraph outlines the dynamic model of the motion of the studied elements (resupply cargo, station module, station, crew vehicle). The element, named spacecraft, moves under the gravitational effect of two massive bodies, within the framework of an Earth-Moon transfer. The mathematical model representing the Earth-Moon or Sun-Earth dynamical environment is the Circular Restricted Three-Body Problem (CR3BP). This model is used to introduce the concepts of libration points, libration orbits and invariant manifolds. The CR3BP leads to obtaining quick and efficient quantitative results for transfers between Earth and libration orbits [16], [12].

In the CR3BP, the primaries are two massive bodies, m_1 and m_2, animated by a circular coplanar motion around their common center of mass. The spacecraft is a particle of negligible mass m3 (the third body). In this paper, the CR3BP is used to describe both the Sun-Earth-Spacecraft and Earth-Moon-Spacecraft systems. The motion of the spacecraft is defined in the rotating reference frame centered at the center of mass of the system (see **Figure 3**).
The units of the systems are normalized in order to simplify the expressions. Masses, distances and time are normalized respectively by the sum of the primaries' masses, the distance between them and their angular velocity around their barycenter.

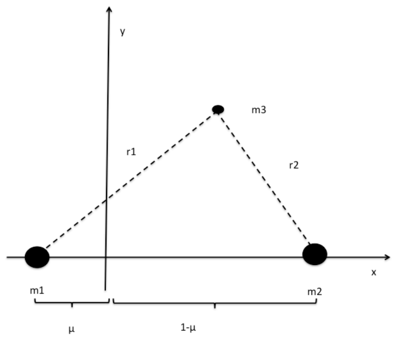


Figure 3: CR3BP in normalized units

The equations are written in the synodic frame, centered on the center of mass of the system and with the x-axis directed from m_1 to m_2 and the y-axis in the plane of the primaries' motion. The spacecraft state is defined with the position and velocity, as $\bar{x} = \{x, y, z, \dot{x}, \dot{y}, \dot{z}\}$, governed by the following system of equations [13]:

$$\begin{cases} \ddot{x} - \dot{y} = -\bar{U}_x \\ \ddot{y} + 2\dot{x} = -\bar{U}_y \\ \ddot{z} = -\bar{U}_z \end{cases} \quad \text{(I)}$$

With $\bar{U}(r_1, r_2) = -\frac{1}{2}\left[(1-\mu)r_1^2 + \mu r_2^2\right] - \frac{1-\mu}{r_1} - \frac{\mu}{r_2}$

$$r_1^2 = (x+\mu)^2 + y^2 + z^2$$

$$r_2^2 = (x-1+\mu)^2 + y^2 + z^2$$

$$\mu = \frac{m_1}{m_1 + m_2}$$

Where μ is the primaries mass ratio, $\bar{U}$ is the effective potential of the system, and r1 and r2 represent the distance from the third body to the larger and smaller primaries, respectively. The system of equations (I) admits an integral of motion [14], commonly used in the form of the Jacobi integral, defined as:

$$C(x, y, z, \dot{x}, \dot{y}, \dot{z}) = -2\bar{U} - (x^2 + y^2 + z^2) \quad \text{(II)}$$

The system of equations (I) is characterized by five equilibrium points or Lagrangian (or Libration) points. They are denoted EML1 to EML5 in the Earth-Moon system. Three of them (from EML1 to EML3) are collinear and located on the Earth – Moon axis, the last two ones are positioned at 60° leading and 60° trailing on the Moon orbit (as smaller primary body). **Figure 4** presents the Libration points location without respect of the Celestial bodies' size and the distances scale.

Four main types of trajectories can be found for a spacecraft orbiting around a Libration point: the Lyapunov orbits, the Lissajous orbit, the Halo orbits and Quasi-Halo orbits, defined as follows:

- Lyapunov orbits are planar periodic orbits in the orbital plane of the primaries (xy-plane). Exact Lyapunov orbits only exist in the CR3BP.
- Lissajous orbits are three-dimensional quasi-periodic orbits with an in- and out-of-plane oscillation.
- Halo orbits are three-dimensional periodic orbit. Farquhar named them like this after their shape they look like when seen from Earth [15]. Exact halo orbits can only be computed in the CR3BP.
- Quasi-halo orbits are quasi-periodic orbits around a Halo orbit. They are intermediate between Lissajous and Halo orbits.

Some examples of trajectories are provided on **Figure 5**. They are characterized by the maximal elongation along y-axis (A_y) and z-axis (A_z).

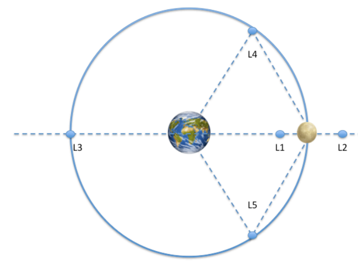


Figure 4: Lagrangian points' location in the Earth-Moon system

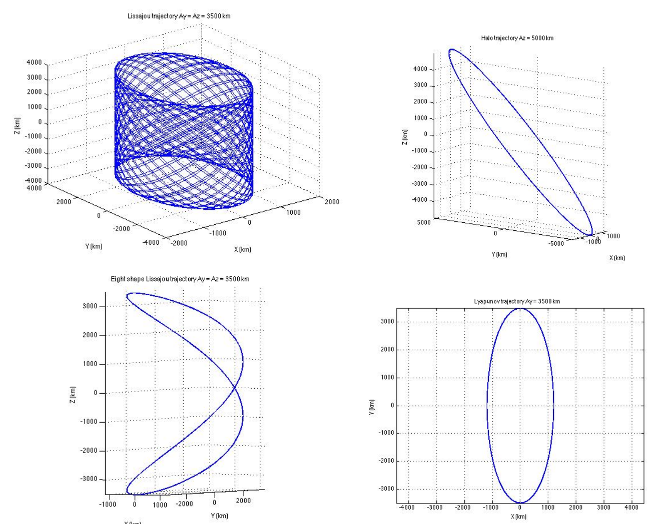


Figure 5: (a) Lissajous trajectory with Ay = Az = 3500 km (b) Halo trajectory Az = 5000 km (c) Eight shape Lissajous trajectory with Ay = Az = 3500 km (d) Lyapunov trajectory with Ay = 3500 km

Orbit around a Lagrangian point has to be considered as the asymptotic limit when time grows of the solution of the system of equations (I). Each solution can be characterized by its Jacobi constant C_h (II), defined by a five dimensional energy manifold as:

$$M(C_h) = \{(x, y, z, \dot{x}, \dot{y}, \dot{z}) / C(x, y, z, \dot{x}, \dot{y}, \dot{z}) = C_h\}$$

Invariant manifolds provide dynamical channels beneficial to the design of energy efficient spacecraft trajectories [14]. They are often referred to as tubes since they exhibit tube-like shapes when projected onto the 3-dimensional position space.

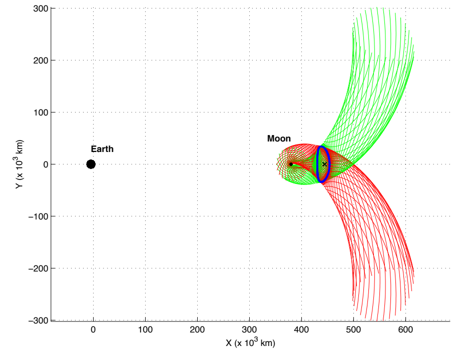


Figure 6: Stable (green) and unstable (red) manifolds of an EML2 southern halo orbit (blue) with an amplitude Az = 5000 km [7]

Mission analysis results led the selection of a Halo orbit around EML2 to set the THOR station. The model used to compute the trajectory (orbit and manifold) is detailed in the next part.

IV. TRANSFERT OPTIMIZATION

IV.I Transfer strategy

To deploy, maintain and operate a space station located on Halo orbit around EML2, a global trajectory has to be selected. Even launch and station keeping (in LEO or at EML) are critical legs; in this part, the focus is set on the transfer while the next one deals with the rendez-vous. As far as transfer strategy is concerned, a wide literature already exists and enlightens that four main strategies are possible: the direct transfer, the indirect transfer, the lunar flyby and the Weak Stability Boundary transfer.

Direct transfer consists in displacing a spacecraft between two space bodies with two direct ballistic maneuvers. It is the most fuel-consuming strategy since it does not take benefit of the manifolds. The indirect transfer strategy main goal is to deposit the spacecraft at an optimized point to enter the manifold and let it glide until it reaches the Halo orbit. In the Lunar flyby strategy, the manifold entrance point is in the Moon vicinity so as to benefit from its slingshot effect to get into the manifold towards the Halo orbit. The Weak Stability Boundary transfer strategy uses the gravitational influence of the Sun to lower the required fuel. For such a transfer, an extension of the C3RBP is needed: as a first approach, two patched Three-Body problems (Sun-Earth-Spacecraft and Earth-Moon-Spacecraft) are used to account for the influence of the Sun, Earth and the Moon.

The strategies have mainly been evaluated thanks to two main criteria: duration (total time of flight) and delta-v. Comparison of those four strategies has been performed and results are provided in [7]. Main conclusions are:

- Since the travel is symmetric, it is enough to focus only on one way. The return will be deducted by using the same trajectory but travelling on the unstable manifold.
- Since the duration criteria is the most important in case of human spaceflight, the crew vehicle trajectory shall be sized using a lunar flyby strategy.
- Since the consumption is the most significant criteria for cargo scenario, weak stability boundary transfer is recommended.

As a consequence, the strategy adopted in the present study is a two impulsive LEO-to-halo transfer with a first maneuver to escape Earth and a second maneuver in the vicinity of the Moon to inject the spacecraft on the stable manifold of the halo orbit. Return trajectories are also considered, with identical notations. Transfer trajectories for crew vehicle, with lunar-fly strategy, have already been detailed in [7]. As a consequence, the paper focuses on the cargo transfer trajectory, with a weak stability boundary transfer model.

IV.II Transfer optimization methodology

The following methodology has been developed for a cargo vehicle that will resupply the station, in orbit at least for six months on a Halo orbit around EML2. The cargo transfer will start from a circular LEO in the (xy) plane (no inclination). The influence of the LEO altitude (h_{LEO}) is great on the overall cost. Therefore, h_{LEO} is fixed, equal to 200km. The departure point angular position is not fixed but rather used as one of the optimization parameter. The Halo orbit is entirely defined by two parameters: Az and m. Az corresponds to the maximum out-of-plane amplitude in the +z direction of the considered orbit, in kilometers. In this study, Az will be set equal to 5000km, 8000km and 30000km. For a given Lagrangian point, halo orbits are divided into two families, which are mirror images across the xy-plane. When the Az is in the +z direction, the halo orbit is a member of the northern family (m=1), while if Az is in the -z direction; the Halo orbit belongs to southern family (m=3) [14]. The transfer trajectory is summarized on **Figure 7**.

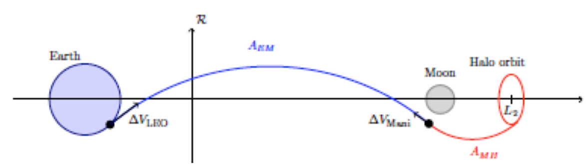


Figure 7: Transfer trajectory definition [7]

The transfer optimization algorithm consists in minimizing the total transfer delta-v along the trajectory. The transfer is separated into two branches: A_{EM} (the arc in the Sun-Earth system) and A_{MH} (the arc in the Earth-Moon system). The methodology relies on two CR3BP models overlapping in the vicinity of the Moon and aims at reducing the velocity gap to jump from first arc to the second one. It is based on backward computation that starts from the expected Halo orbit to the LEO. The algorithm has five main steps that will be described just afterwards:

- Step 1: Halo orbit computation
- Step 2: Computation of the arc in the Earth-Moon system.
- Step 3: Computation of the arc in the Sun-Earth system
- Step 4: Delta-v optimization
- Step 5: Transfer trajectory reconstruction based on the selection of the best transfer trajectories

Step 1: The Halo orbit, being entirely described through A_z and m, is computed with a third-order approximation developed by Richardson [17], used as first guess. Then, according to K. Howell [18], a differential correction process is then implemented.

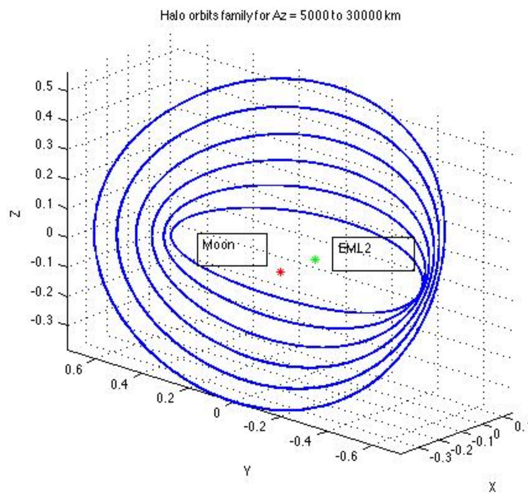


Figure 8: Halo orbits family for Az = 5000 to 30000 km

Step 2:
The A_{MH} arc is generated from a distance d_M of a given angular position (θ) on the Halo orbit, in the initial stable direction, by propagating the equations of motion (I). According to literature, d_M is chosen in the [1km; 100km] range for which the linear approximation is valid. The position (θ) on the Halo orbit varies from 0° to 360°. Each set (Az, θ, d_M) gives a specific trajectory from the Halo orbit. The arc is then propagated until the intersection with the selected Poincare section, defined by the angle ϕ_{EM} (represented on **Figure 9**). For optimization purpose, ϕ may varying from 0° to 360°.

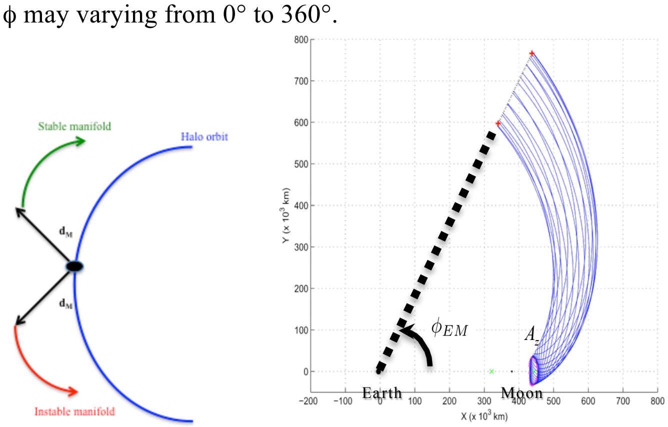


Figure 9: (a) d_M definition, (b) ϕ_{EM} definition [7]

Step 3
The A_{EM} arc is generated from the connection point in the Moon vicinity, defined by the angle ϕ_{EM}, and depends on the velocity at the entrance of the manifold in the Sun-Earth system. As a consequence step 3 and step 4 are interrelated.

Step 4:
For transfer, the total delta-v results from two maneuvers

- The initial delta-v (ΔV_{LEO}) to quit LEO and be injected on the A_{EM} arc
- The intermediate delta-v (ΔV_{Mani}) to quit the A_{EM} arc and join the A_{MH}

The initial delta-v corresponds to a random value of the escape velocity for h_{LEO}, in a range of +/- 10%. The velocity has to be tangential to the orbit and is targeted using a numerical iterative differential correction process [11], from the final point on the A_{MH}. The intermediate delta-v (ΔV_{Mani}) is obtained by genetic algorithm optimization. It shall minimize the velocity gap between the final point on the A_{MH} and the first point on A_{EM} and shall be collinear to the velocity at entrance of the manifold from the Poincare section to the LEO.

Step 5:
Solutions for the design parameters (A_z, θ, d_M, ϕ_{EM}) that produce the lowest total transfer delta-v are then selected and the entire trajectory is then reconstructed [7].

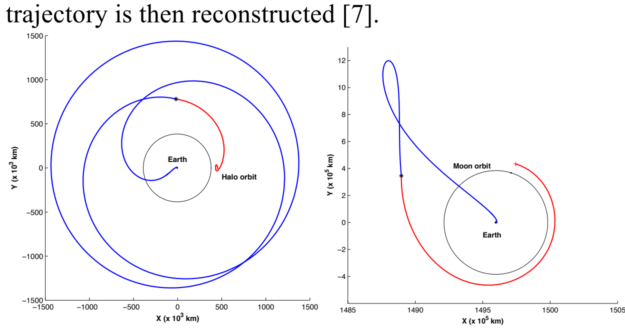


Figure 10: (a) Earth to Halo orbit trajectory in the Earth-Moon rotating frame for A_z=8000 km (b) Same trajectory in the Sun-Earth rotating frame [7]

Figure 10 represents an example of global transfer trajectory from the Earth to the EML2 vicinity. The blue arc corresponds to the Earth departure leg, computed in the Sun-Earth CR3BP, while the red one sis the arrival near EML2 in the Earth-Moon CR3BP.

IV.III Transfer optimization results
Thanks to the application of the previous methodology, the cost of the transfer trajectories from LEO to EML2 is computed for a maximal elongation A_z equal to 5000 km, 8000 km and 30000 km with the following assumptions: d_M = 50 km and h_{LEO} = 200 km. The optimization parameters are ϕ_{EM} between 0° and 360°, initial delta-v and the angular position of the Poincaré map. A synthesis of the results (total delta-v along the transfer trajectory versus the time of flight) is presented on **Figure 11**.

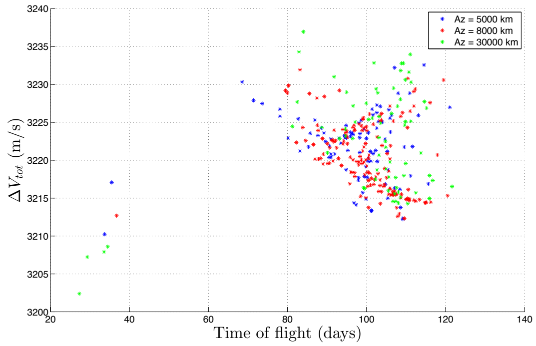


Figure 11: Total required delta-v versus time of flight (in days) for A_z = 5000 km, 8000km and 30000 km [7]

Most trajectories are grouped in a "nominal family" (same order of magnitude than the results published in the literature) with a range [3210; 3235] m/s and a time of flight of about [70,120] days. Moreover, some unexpected trajectories (named the "exotic family") are found for a time of flight lower than 35 days. A deeper analysis of this trajectories family shows that there is significant gap in the orientation of the velocity at the manifold insertion point. Nevertheless, the norm of the velocity increment (ΔV_{Mani}) satisfies the connection conditions. Further analyses were carried out to check the relevance of such solutions, since they could be very promising trajectories, in particular for inhabited vehicle transfer. It seems that the Earth leg (the AEM arc) could correspond to a very flat ellipse (with a very low eccentricity). In the new studies, the first part of the trajectory – from LEO to the insertion into the manifold – is computed in the BCR4BP (Bi-Circular Restricted Four-Body Problem) of the Sun-Earth-Moon system and then optimized. The process consists in computing first a peculiar trajectory in a CR3BP model in order to have a good starting point for the optimization calculation with the Four-Body mode. The optimization is performed with a Chebyshev interpolation of the best points of the CR3BP trajectory so as to minimize the discrepancies. Because obtained results in the BCR4BP degrade the time of flight and the total delta-v along the transfer trajectory, investigations are still undergoing to compare optimization process. As a consequence, the CR3BP remains the baseline model for transfer. Nevertheless, those exotic trajectories performances are so promising, that the effort will be kept to further study them.

IV.IV Transfer optimization limitations

These transfer trajectory optimization results from some simplifying assumptions. This methodology leads to some interesting results that allowed deciding the mission feasibility. Nevertheless, in order to get more accurate performances for the time of flight and the delta-v, some complementary analyses shall be performed to:

- Evaluate the impact of ephemeris versus theoretical celestial bodies' position
- Optimize the LEO altitude; h_{LEO} should become a design parameter.
- Analyze the robustness of selected trajectories, and particularly for the exotic family
- Model the entire trajectory in the four-body problem

V. RENDEZ-VOUS RESULTS

The main goal of this part is to describe the proposed methodology to plan a rendez-vous in EML2 between the Thor space station and the delivery cargo or crew vehicle. The different phases and maneuvers of a typical rendez-vous mission from the launch until the docking have already been extensively studied. Those phases are usually denoted launch, transfer, orbital injection, phasing and proximity maneuvers (including homing, closing and final approach), as for example in the case of the ATV Jules Verne [19][20]. In THOR resupply context, the three main rendez-vous phases have to be modified and adapted to non-keplerian orbits around unstable Lagrangian points (here, EML2).

A focus is set on a Halo-to-Halo transfer, while assuming that THOR is already orbiting around EML2. But every type of rendez-vous (Lissajous to Halo, Lyapunov to Halo, …) should be investigated. Two different Halo orbits are never coplanar; studies have been performed on optimal transfers between unstable orbits around Lagrangian points using Weak Stability Boundary and Invariant Manifolds (see part IV). Even if the targeted Halo orbits are the same (same A_z, same m), small discrepancies in the launch can generate large differences at the arrival at EML2. That is the reason why; it is assumed that the cargo will not arrive directly on THOR orbit

The most critical part of the rendez-vous mission lies in the proximity operations phase when the distance between the chaser (here the cargo) and he target (here THOR station) is below a small distance. Safety is the overriding design consideration for automated missions towards inhabited facility. To avoid collision and accident, corrections maneuvers must be performed before this final step, that is why the cargo trajectory must be computed with a very high accuracy. The results presented, in this part, start from the end of the transfer and finish before the proximity operations.

V.I. Rendez-vous in EML2 definition

At this stage, it is supposed that the Thor space station is already on its Halo orbit and the delivery cargo tends to join it. A Halo orbit can be characterized by

- A_z, the maximum out-of-plane amplitude in the + z direction of the considered orbit
- -m, the orbit family (m = 1 Northern for orbit, m = 3 for Southern orbit)

The additional important element is the orbital position of the space station at the rendez-vous time.

For this study, the rendez-vous is defined as the phase during which the cargo leaves its initial Halo orbit (A_{z_cargo}, $m_{\_cargo}$) and reaches the THOR halo orbit (A_{z_THOR}, $m_{\_THOR}$). The performances will be characterized by the required delta-v and the duration.

V.II. Halo orbit model

The rendez-vous maneuver main goal is to ensure that the chaser approach the target within a very close distance. Rendez-vous requires a precise match of the orbital velocities of the two bodies, allowing them to remain at a constant distance through station keeping maneuvers in order to allow docking or berthing. This constant distance will be suppressed during proximity maneuvers until docking. Therefore, the determination of highly accurate trajectories in the vicinity of the translunar libration point is very important. The linearized model will not suffice. For this project, the analytical solutions for quasi-periodic orbits about EML2 that Farquhar [23] has obtained using the method of Lindstedt - Poincaré are compared to linearized model and the one of Richardson.

Next figure presents the result of the comparison of the Halo obtained with Farquhar model (red plot) and Richardson model (blue plot), for Az = 30000 km and m=3. The green star in the center is EML2.

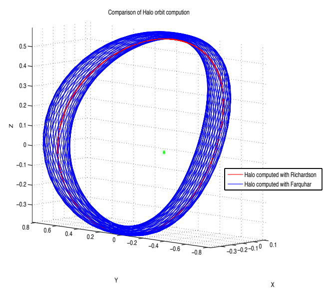


Figure 12: Comparison of Halo obtained with Farquhar model and Richardson model

It can be concluded that the Halo orbit model type has large consequences on the accuracy of the position knowledge. An imprecise model will generate degraded rendez-vous performances. The Farquhar orbit model is interesting since it takes into account natural uncertainties (like for example the Sun's gravitational effect on the Earth-Moon system). Nevertheless, for consistency reasons with the transfer model (see IV), the Richardson model is applied.

V.III. Rendez-vous strategies

Very few literature can be found about rendez-vous in the vicinity of a Lagrangian point, except in [6], [21] and [23].

The strategy will depend on when the rendez-vous starts: either the cargo is still considered as traveling on the manifold (MOI) or the cargo has already been inserted on its Halo orbit (HOI). For a real mission, there should be no difference, since except in CR3BP, exact Halo orbit does not exist. The cargo keeps on traveling on its trajectory that will converge at the infinite time to the Halo orbit. Nevertheless, numerical representations impose to consider two different models: one for the Halo orbit (as described in V.II) and one for the manifold (as explained in III).

Considering that A_z and m can fully characterize a Halo orbit and that the starting time has an influence on the rendez-vous, eight rendez-vous strategies have been identified. The following table provides a synthesis. NA means that the phase's difference between the cargo and the station is not considered.

Type	A_z	M	Phase
HOI	$A_{z\_THOR} = A_{z\_cargo}$	$m_{\_THOR} = m_{\_cargo}$	Different phases
HOI	$A_{z\ THOR} \neq A_{z\ cargo}$	$m_{\ THOR} = m_{\ cargo}$	NA
HOI	$A_{z\ THOR} = A_{z\ cargo}$	$m_{\ THOR} \neq m_{\ cargo}$	NA
HOI	$A_{z\ THOR} \neq A_{z\ cargo}$	$m_{\ THOR} \neq m_{\ cargo}$	NA
MOI	$A_{z\_THOR} = A_{z\_cargo}$	$m_{\_THOR} = m_{\_cargo}$	Different phases
MOI	$A_{z\_THOR} \neq A_{z\_cargo}$	$m_{\_THOR} = m_{\_cargo}$	NA
MOI	$A_{z\_THOR} = A_{z\_cargo}$	$m_{\_THOR} \neq m_{\_cargo}$	NA
MOI	$A_{z\_THOR} \neq A_{z\_cargo}$	$m_{\_THOR} \neq m_{\_cargo}$	NA

Table 1: Rendez-vous strategies

The previous synthesis shows that performing the rendez-vous between a cargo and the THOR station always correspond to an heteroclinic connection between two Halo orbits, by finding the intersection between their manifolds (the unstable manifold for the cargo and the stable manifold for the station), except when only the phase has to be changed. The focus is set on the HOI configuration, with different A_z.

The methodology developed describes how to model the rendez-vous between two different Halo orbits, as a heteroclinic connection.

V.IV. Rendez-vous methodology

The process is consistent with previous transfer study, since the main step lies in the intersection of two manifolds thanks to a Poincaré section. It falls into five main steps:

- Step 1: To compute the Cargo Halo orbit and unstable manifold
- Step 2: To compute the Thor station Halo orbit and stable manifold
- Step 3: To find the optimal intersection between both manifolds thanks to a Poincaré section
- Step 4: To compute the cargo entire rendez-vous trajectory from its Halo orbit to the Station orbit
- Step 5: To estimate the rendez-vous performances (total delta-v, duration)

In this process, the design parameters let free for the optimization are: d_M (the distance between the orbit and the manifold) and the Poincaré section position.

Then, when the cargo is on the THOR orbit, the maneuver left to compute is the one that will reduce the phase difference, as proposed in [21]. The total delta-v for the rendez-vous is computed as:

$$\Delta v_{rdv} = \Delta v_1 + \Delta v_2 + \Delta v_3 \text{ (III)}$$

where

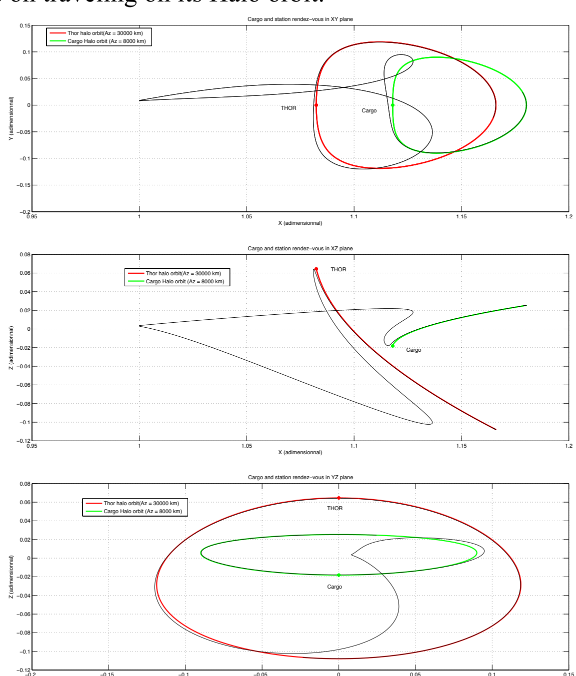
- Δv_1 is the necessary burst to enter the unstable manifold from the cargo initial Halo orbit
- Δv_2 is the burst to leave the unstable manifold to get on the stable manifold
- Δv_3 is the necessary burst to leave the stable manifold to join the Thor Halo orbit

The cost function for the optimization process is a combination of the distance between the two manifolds and the velocity gap at this point, since Δv_2 is the most expensive maneuver. The distance has to be as low as possible:

- First to explore only physical and feasible trajectories
- Then to limit the rendez-vous duration, since it is a direct consequence of this distance.

V.V. Rendez-vous preliminary results

Trajectories are computed for several departure points on the cargo initial Halo orbit and several arrival points on the Thor Halo orbit. The **Figure 13** provides an example of a Halo-to-Halo rendez-vous strategy, with the THOR station orbit defined by Az = 30000 km, m = 1 and cargo orbit defined by A_z = 8000km, m = 3. The cargo is first rotating on its Halo orbit (green leg), than escapes on the unstable manifold (first black leg) thanks to a first impulsive maneuver. At the intersection between cargo unstable manifold and station stable manifold, on the Poincaré section, the cargo enters the station stable manifold thanks to a second impulsive maneuver and then glides until it reaches the Station orbit. During that phase, the Thor station keeps on traveling on its Halo orbit.



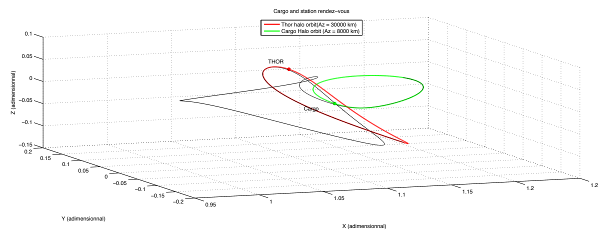


Figure 13: Example of rendez-vous strategy with Cargo and Thor station

The trajectory presented on the **Figure 13** seems to be very promising as far as distance between the two manifolds is considered, but is really a bad case for velocity gap, since it has been obtained by only minimizing the distance gap. Hereunder are some results obtained for this type of configuration:

PM position (adimensionnal)	A_z	M	Δv_2
x= 1	A_{z_THOR} = 30000km A_{z_cargo} = 8000km	$m_{\_THOR}$ = 1 $m_{\_cargo}$ = 3	2.292 km/s
x= 1	A_{z_THOR} = 30000km A_{z_cargo} = 8000km	$m_{\_THOR}$ = 3 $m_{\_cargo}$ = 3	2.352 km/s

PM stands for Poincaré map (defined with (IV))

The cost of this strategy is not affordable. A huge amount of energy is required since the velocities are quite opposite along the x-axis. The vehicle velocity has first to be decreased and then accelerated to the aimed value.

Strategies that minimize a cost function as (III), combination of distance and velocity have been tested. The table presented here after provides some examples:

PM position (adimensionnal)	A_z	m	Δv_2
x= 1	A_{z_THOR} = 30000km A_{z_cargo} = 8000km	$m_{\_THOR}$ = 1 $m_{\_cargo}$ = 3	0.495 km/s
x= 1.072	A_{z_THOR} = 30000km A_{z_cargo} = 8000km	$m_{\_THOR}$ = 3 $m_{\_cargo}$ = 3	0.235 km/s

As a consequence, it can be concluded that the position of the Poincaré section is a key design parameter. It is then suggested to perform a two-stages optimization:

- First to start with a nominal trajectory as an initial guess that minimizes the cost function.
- Then to minimize the distance between the manifold at Poincaré section, thanks to small variation of the position of the departure and arrival points.

To improve the first step of the optimization process, it has been decided to modify the intersection plane definition and to consider the Halo orbit maximal elongation (Az) as a design parameter. The Poincaré section description is consequently enhanced by a new definition, given as:

$$\{(x, y, z, \dot{x}, \dot{y}, \dot{z})/\phi = \phi_{PM}\} \text{ (IV)}$$

with ϕ_{PM} varying between 0° and 360°.

Figure 14 presents the Poincaré section definition.

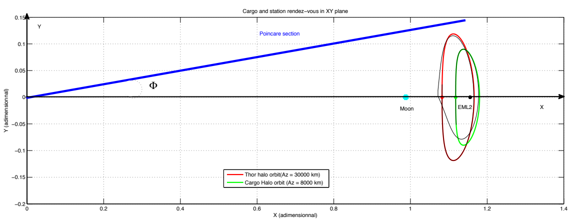


Figure 14: Poincare section definition for rendez-vous optimization

ϕ_{PM}excursion is limited to an internal realm of the Earth-Moon system [14]. A_{z_cargo} and A_{z_THOR} are assumed to be different, so as to take into account the launch and transfer maneuvers discrepancies. Nevertheless, the simulation campaign demonstrated that the difference between both elongations has a bad impact on Δv_2. Moreover, as already discussed in [7], Halo orbits with an Az equal to 8000km are less expensive than Halo orbits with an Az equal to 30000km. From now on, A_{z_cargo} is set equal to 7500 km and A_{z_THOR} to 8000km.

Angular position on the Halo orbit is defined thanks to the pseudo orbit center position (black star), as presented on Figure 15 :

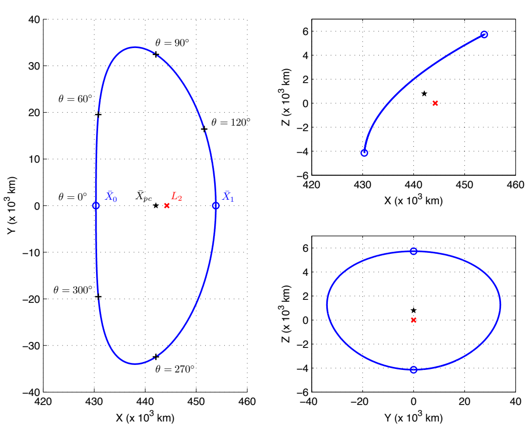


Figure 15: Angular position on the Halo orbit definition [7]

The graphs on Figure 16 provide the evolution of total delta_v during the rendez-vous as a function of the angular position on the departure Halo orbit and of the arrival Halo orbit. The example corresponds to $\phi_{PM} = 0°$.

For cargo initial Halo orbit (A_{z_cargo} = 7500 km, m =3) and a THOR Halo orbit (A_{z_THOR} = 8000km, m =3), the best results give a total delta_v of 0.0026 km/s for ϕ_{PM} equal to 64°, that correspond to theta_cargo =78° and theta_Thor = 282°. This trajectory is selected as the initial guess for the second step of the optimization.

On Figure 17 the blue area corresponds to the optimal configuration for (theta_Thor, theta_cargo), while the red area is the forbidden zone for rendez-vous. If the target position on Thor Halo orbit is in the range of [0°,90°], the angular departure position on the cargo Halo orbit should be in the [110°, 360°] to minimize the fuel consumption.

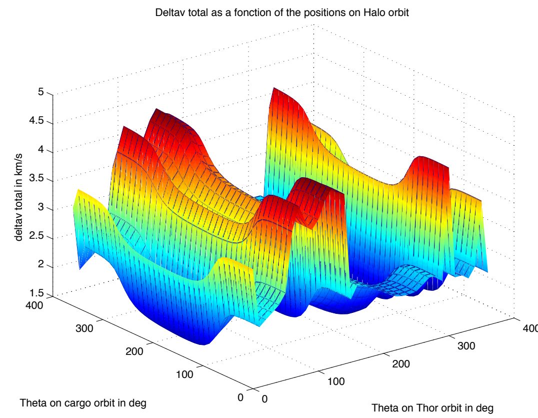


Figure 16: Delta-v total as a function of the angular positions on the halo orbit

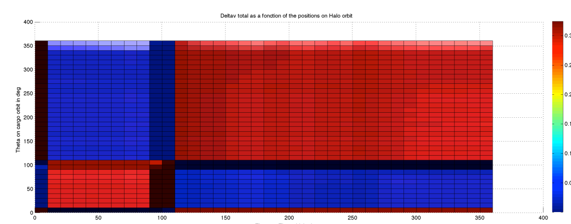


Figure 17: Delta-v total as a function of the angular positions on the halo orbit for ($\phi_{PM} = 64°$)

The Poincaré section angular position and the maximal elongation of the cargo initial Halo orbit being fixed, the influence of elongation of the Thor station Halo orbit is studied.

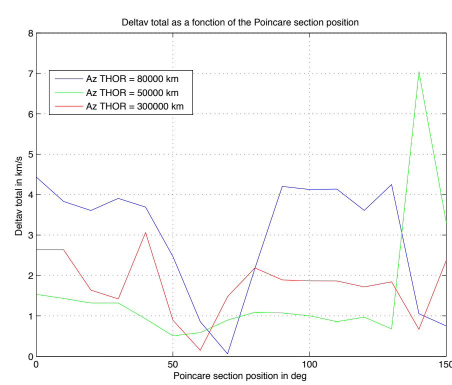


Figure 18: Delta-v total as a function of the Poincaré section angular position, A_{z_Thor} = 5000 km, 8000 km and 30000 km and A_{z_cargo} = 7500 km

Figure 18 presents the evolution of the total delta-v as a function of A_{z_Thor}. An elongation set equal to 8000 km seems to lead to

the minimal fuel consumption. A more systematic analysis, with a wider range of A_{z_Thor}. could confirm this preliminary conclusion.
The initial guess is by consequence defined by:

- A_{z_Thor} = 8000 km, m = 3
- A_{z_Thor} = 7500 km, m = 3
- ϕ_{PM} = 64°
- Theta_cargo = 78°
- Theta_Thor = 282°

Figure 19 illustrates this initial guess trajectory for rendez-vous between the cargo and Thor.

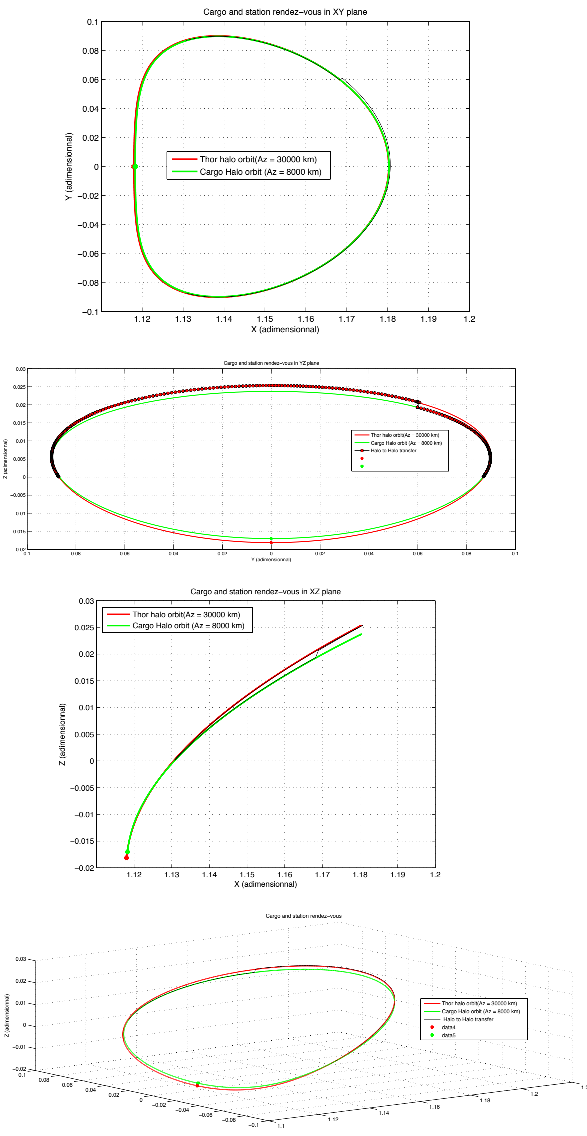


Figure 19: Initial guess rendez-vous trajectory

In this case, the expected performances are

- deltav_tot = 0.0025 km/s
- transfer duration on the manifold = 9 days.
- distance gap between both manifolds: 439 km.

Those results are really encouraging as far as deltav_tot is concerned, but the distance gap is not realistic and intermediate maneuvers should be scheduled to safely ensure transfer between both manifolds. As a consequence, more extended optimization around this initial guess is recommended:

- First to evaluate the impact of parameters (Az_Thor, ϕ_{PM}) but also ephemeris versus theoretical position of the celestial bodies (Earth, Moon, Sun)
- Secondly, to reduce the distance gap at intersection between both manifolds.

At this second step of the optimization process, Theta_cargo and Theta_Thor will be let free in the [0,360°] range. Moreover, POINT (Program to Optimize INterplanetary Trajectories) software, developed by ASTOS team [16], will be used to validate and refine the preliminary results. ASTOS team has developed this tool. It implements, for example, the Three Bodies problem, flyby trajectories and the patched conic methodology. As a consequence, it is well adapted to interplanetary mission. It also proposes several optimization algorithms and particularly, the genetic algorithm.

V.VI. Remarks and future work

In this last section, the rendez-vous strategy for a cargo, departing from a Halo orbit around EML2 towards an inhabited space station, travelling on another Halo orbit around EML2 has been explored. An initial guess trajectory has been selected so as to prepare an automatic optimization campaign.
In order to complete this work, further analyses should be performed:

- To forbid some arrival position on the Thor allow orbit so as to reduce collisions risk,
- To take into account that communication with Earth ground control should be continuous during those critical operations
- To study the MOI scenario (when the rendez-vous starts from the cargo manifold and not from the orbit)
- To apply this methodology to other types of orbits (Lyapunov, Lissajous)
- To verify the feasibility of the maneuvers compared to propulsion technologies
- To compare optimization maneuvers

VI. CONCLUSION AND PROSPECTS

This paper deals with the transfer of a resupply cargo from its low Earth orbit to its rendez-vous with an inhabited space station located on halo orbit around the EML2. It main contribution lies in the rendez-vous strategy, a topic that is few developed in the literature. First, a methodology to optimize the transfer from LEO to EML2 has been presented, based on two CR3BP models and a genetic algorithm. Promising results were obtained that are a good compromise between delta-v and time of flight.
Secondly, the focus has been set on the rendez-vous between the cargo and the station. Several potential scenarios have been identified. Among them, the selected one corresponds to a departure of the cargo from a Halo orbit of the southern category with a maximal elongation different from the targeted station Halo orbit. The transfer takes place between the unstable

manifold of the cargo initial orbit and the stable manifold of the station Halo orbit. The design parameters for the optimization are the Poincaré section angular position, the departure angular position on the initial cargo orbit and the arrival departure angular position on the Thor orbit. Promising results have been obtained and an initial guess trajectory has been selected. Next effort will be put on performing automatic optimization around this trajectory to see the influence of additional physical data - like the position of the celestial bodies – and design parameters and to explore the other scenarios. Optimization methodologies have also to be explored to refine and compare the results.

This study confirms, from an astrodynamic point of view, the feasibility of human spaceflight exploration mission. The vicinity of the Lagrangian point is a really interesting step on the journey towards Mars or the Asteroids.

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