IAC–18–C1.1.11

PHASING AND RENDEZVOUS OPERATIONS ON NON-KEPLERIAN ORBITS IN THE EARTH-MOON SYSTEM

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Recent studies and investigations proved the effectiveness of non-Keplerian orbits as staging location for a lunar orbiting outpost to support future Space Exploration. NASA's Lunar Orbital Platform-Gateway and ESA's HERACLES concepts represent different answers to the outpost architectures settling, both of them looking at Near Rectilinear Halo Orbit (NRHO) family as suitable permanent location for such infrastructure. Moreover the infrastructure, to be effective, shall ensure significant operational flexibility to cope with variable logistics depending on the mission scenarios to support; frequent docking/undocking events of different module classes shall be assumed as one of the primary functionalities an infrastructure in deep space shall provide. At the time being, no proximity manoeuvring has been performed so far in the strongly perturbed non-Keplerian dynamical environment, which is intrinsically challenging.
The paper first analyses the different criticalities that arise from such dynamical environment, starting with a discussion on connecting the lunar surface to non-Keplerian orbits. Operational challenges are identified in the connection of parking orbits, belonging to the inertial lunar space, and non-Keplerian trajectories, which are defined in the Earth-Moon rotating frame. The two frames are not directly dependent, and the difference between the apparent forces plays a fundamental role in selecting parking orbits. Furthermore, the strong irregularities in the lunar gravitational field affect the lifetime of such parking orbits, posing an operational challenge in terms of timing the transfer.
The investigation then proceeds with the transfer and phasing trajectories analysis, assuming a chaser vehicle to rendezvous with an orbiting infrastructure. The need to drive the transfer design according to the operational constraints is highlighted: whenever possible operational errors are accounted in the trajectory analyses, the cheapest transfers appear as critical for operations, whereas a longer, more expensive trajectory actually gains a ΔV saving from an overall perspective which includes Trajectory Correction Manoeuvres (TCM) and both deterministic and stochastic budgets. The case study of a space vehicle, landed on the lunar surface, that shall perform phasing and rendezvous with a vehicle orbits in a NRHO, identifying timing opportunities, different rephasing possibilities, and investigating stochastic TCMs to take into account manoeuvre and navigation errors is widely discussed in the paper. The effect of the different constraints, including time of flight, surface landing site, and engine limited thrust, is analysed and critically discussed to drive mission analysis requirements.

1. Introduction

In the early years of space flight, non-Keplerian orbits in lunar environment were proposed for exploration missions [1]. After the Apollo program, however, international interest steered towards Sun-Earth libration points and asteroid or comet exploration. More recently, near-future lunar exploration missions became again a topic of interest among international industry and space agency players, and non-Keplerian orbits were increasingly used for trajectory analysis and design within such framework. The Global Exploration Roadmap [2] indicates the use of lunar environment as a key factor towards the future of human spaceflight with a crucial role in developing technologies, strategies, and techniques to enable Mars missions and interplanetary travel.

Among these capabilities, the rendezvous and docking problem has a prominent importance for future missions, which may involve large infrastructures, composed of manned and unmanned modules. Proximity manoeuvring between these modules, together with safe and efficient rendezvous trajectories, will be a must-have functionality of any exploration mission. Several contemporary studies [3, 4, 5] examined trajectories, strategies, and mission scenarios that involve renzdezvous in trans- and cis-lunar non-Keplerian orbits in the Earth-Moon system, and numerous other investigations are currently on-going [6, 7, 8, 9]. The use of stable and unstable manifolds for final approach phases [10, 11] is proposed as an efficient way of exploiting natural dynamics to achieve proximity motion, able to satisfy safety requirements, although constrained to follow the slow dynamics of the non-Keplerian orbits.

This paper proposes a high-level investigation of the possible challenges that arise from the design of a rendezvous and phasing trajectory in lunar environment. The investigation aims to couple the mission analysis and trajectory design with an operational viewpoint, showing how the inclusion of spacecraft operations in early design phases can benefit the mission budget. Section 2 describes the model, the considered orbits, and the operational assumptions for the investigation. Section 3 describes different aspect of the problem at hand, focusing on the differences and the sensitivity between lunar orbits and non-Keplerian orbits in the Earth-Moon three-body problem. An operational example is presented in Section 4, highlighting the inclusion of operational assumptions in the trajectory design procedure. Section 5 provides some final remarks on the proposed approach and possible future developments.

2. Model and Assumptions

2.1 Equations of Motion

The peculiar non-Keplerian orbits, discussed throughout the paper, arise from the combined gravitational action of the Earth and the Moon. The Circular Restricted Three-Body Problem (CR3BP) is a convenient, simplified model of the dynamics of a spacecraft, with negligible mass m, moving under to the gravitational attraction of two massive bodies, called primaries, with masses m_1 and m_2, which in turn move in relative circular orbits [12]. The dynamics of the spacecraft are conveniently described in

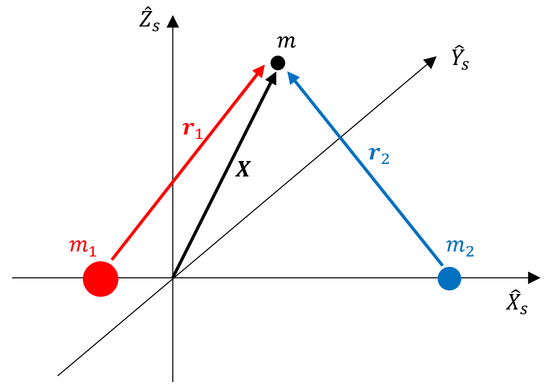

Figure 1: CR3BP reference frame

a non-inertial frame $\hat{X}_s\hat{Y}_s\hat{Z}_s$, which rotates together with the primaries, such that they always lie on the $\hat{X}_s$ and $\hat{Z}_s$ axis is directed as the angular momentum of the relative orbit between m_1 and m_2.

Equations of motion may be normalized, defining a parameter μ

$$\mu = \frac{m_2}{m_1 + m_2} \tag{1}$$

Figure 1 depicts the $\hat{X}_s\hat{Y}_s\hat{Z}_s$ rotating reference frame, where $\mathbf{r}_1$ and $\mathbf{r}_2$ represent, respectively, the non-dimensional position vector directed from m_1 and m_2 to the spacecraft, and $\mathbf{X}$ is the position of the spacecraft in the rotating frame. The potential U

$$U = \frac{1}{2}(x^2 + y^2) + \frac{1-\mu}{r_1} + \frac{\mu}{r_2} \tag{2}$$

is defined, employing dimensionless coordinates; r_1 and r_2 denote the magnitudes of $\mathbf{r}_1$ and $\mathbf{r}_2$. The equations of motion of the spacecraft may be written as

$$\ddot{x} - 2\dot{y} = U_x \tag{3}$$
$$\ddot{y} + 2\dot{x} = U_y \tag{4}$$
$$\ddot{z} = U_z \tag{5}$$

where U_x, U_y, U_z denote the partial derivatives of U. The CR3BP possesses five equilibrium points, called libration points; the three of these that lie on the $\hat{X}_s$ axis, thus called collinear points, are of prominent interest for space mission design, and are labelled L_1, L_2 and L_3.

Although the CR3BP represents a valuable tool for preliminary analyses, later stages of mission design require a high-fidelity model, which takes into account the real orbits of the Earth and the Moon, as well as other relevant accelerations acting on the spacecraft. Namely, the gravitational pull of the Sun may be non-negligible for some orbits (e.g. Near Rectilinear Halo Orbits, [13, 14]) and the irregular lunar gravity potential significantly perturbs low lunar orbits (below ~ 500 km altitude).

The results of the paper employs a dynamical model which includes the Moon, the Earth and the Sun gravities, where the true position of such massive bodies is employed to calculate the gravitational acceleration acting on the spacecraft. The JPL DE405 ephemeris database is employed to retrieve such values of position, at the epoch of each integration step. Where necessary, for low lunar orbits, a 50x50 spherical harmonics model is employed for the Moon gravity field. The effect of Solar Radiation Pressure is not included in the current investigation.

2.2 Non-Keplerian Lunar Orbits

The Earth-Moon system offers a wide variety of non-Keplerian orbits, which may be used for exploration mission. Among the wide selection space, the family of translunar Halo orbits, originating from L_2, appears to be the most favourable for future missions design, thanks to a combination of visibility, station-keeping and accessibility requirements. In particular, Near Rectilinear Halo Orbits [15] were recently proposed for a manned exploration infrastructure, and widely studied in the international literature.

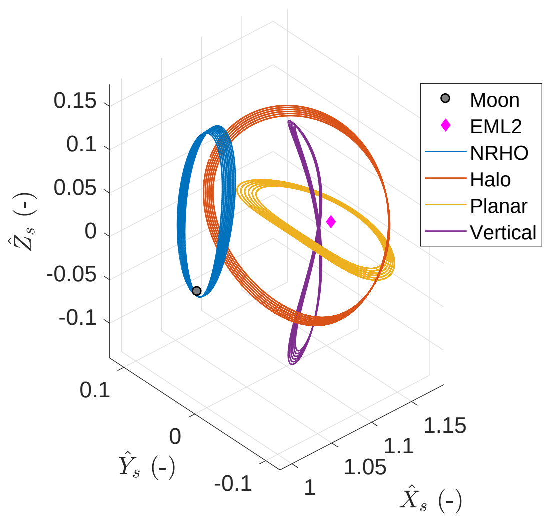

Figure 2: Some Earth-Moon L_2 non-Keplerian orbits, in the CR3BP model

Figure 2 depicts some sample non-Keplerian translunar orbits, in the Earth-Moon system, computed in the CR3BP model. Such orbits can be obtained as well in a full-ephemeris model, although losing the strict periodicity due to the Earth-Moon relative eccentricity and the Sun gravitational pull.

2.3 Operational Constraints

A significant aspect of the present study is the coupling between mission analysis, trajectory design and operational aspects; as it will be noted in the results, these aspects are often in conflict, and a pure ΔV optimisation might yield non-optimal results, or even not acceptable, from an operational point of view. Although the operational constraints for a space mission are vast, and it would be excessively complex to add them since early mission design phases, the following assumptions were employed for the study:

- Mechanisation errors lead to incorrect manoeuvres, both in magnitude and in direction. As conservative assumption, these errors are considered to be 3% in manoeuvre magnitude and 1.5° in pointing accuracy.
- Trajectory Correction Manoeuvres (TCM) are necessary to correct the resulting spacecraft dispersion, but cannot be executed prior to a given amount of time, needed for orbit determination, data processing and telecommand upload.
- Safety requirements constrain the region where rendezvous operations can take place, as underlined in [10] for the NRHO family.
- Possible constraints on the achievable inclinations, after ascent from the lunar surface, might arise from the latitude of the landing site.

Further sets of operational constraints may require, e.g., given illumination conditions, Sun aspect angles, Earth visibility during proximity operation, although these constraints might drive a priori the orbit selection in order to ease successive design phases.

3. Lunar to Libration Points Orbits

Transfer strategies for exploration missions often involve intermediate Low Lunar Orbits (LLO), which shall be connected to non-Keplerian Libration Points Orbits (LPO). Such kind of connection requires some care in trajectory design:

- LLOs are defined in the inertial lunar space, and are mainly perturbed by the lunar irregular gravity field. Above an altitude of ~ 500 km, the third-body perturbation of the Earth becomes more and more significant.

- LPOs are periodic in the Earth-Moon rotating frame, and arise from the combined Earth-Moon gravity fields.

Thus, it is not straightforward to connect the two dynamical regimes, and care must be taken in designing transfer strategies that satisfy operational requirement while minimising the transfer cost.

This Section proposes an approach to this transfer problem, highlighting the different features and peculiarities to be taken into account.

3.1 Parking Orbits

Unless a direct injection into an LPO is employed, lunar mission may take advantage of intermediate parking orbits, e.g. if a large spacecraft contains smaller satellites to be injected in different orbits, or if a vehicle ascends from the lunar surface. If an LLO is employed as parking orbit, the following consideration shall be noticed:

- The lunar gravity field affects the orbit lifetime, mainly depending on the equatorial inclination [16]; this aspect might limit operational capabilities, since the spacecraft might need additional manoeuvres if the loitering time shall be increased in contingency cases.

- The relative motion of LLOs in the Earth-Moon rotating frame can increase the transfer budget, if the nominal manoeuvre timing is missed. Although trivial, this behaviour plays a significant role in the solution space, when searching for LLO-LPO connection arcs. Figure 3 portrays the apparent motion of an LLO, as seen from the Earth-Moon rotating frame; such apparent plane rotation might significantly affect the design of a transfer to/from an LPO.

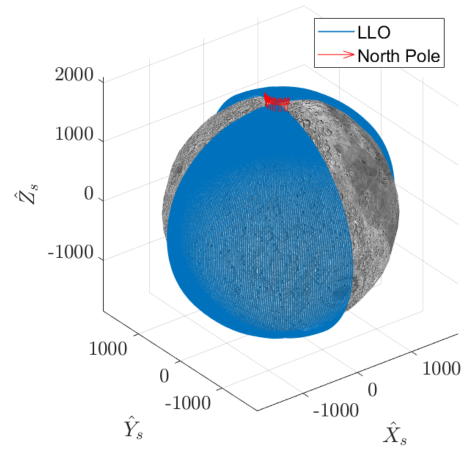

Figure 3: Apparent motion of a 100 km LLO in Earth-Moon rotating frame

3.2 Phasing Trajectories

Once the spacecraft leaves the low lunar environment, the design of phasing trajectories shall employ the non-Keplerian regime to connect and phase with a target LPO. A wide literature exist on the topic, often involving the use of manifolds to connect different orbits [17, 18, 19]. When stringent operational constraints arise, namely on transfer time and on sensitivity, different, faster strategies might be designed, at the expense of a larger ΔV budget. Recalling the results of [10], based on NRHOs in the Earth-Moon system, two main categories of phasing trajectories might be exploited:

- Direct, two impulse transfers, yield a straightforward connection between LLOs and the injection manifold of an LPO. Operationally, this kind of transfer might be challenging, if the timeline is too tight and the manoeuvre sequence requires early TCMs; furthermore, direct transfer often require larger burns, where the mechanisation error might lead to a significant dispersion on the trajectory.

- Indirect, multi-impulse transfer, allow for a greater operational flexibility in the trajectory design. Intermediate orbits may be used to phase, to correct timing errors, and to increase the transfer strategy robustness to dispersion and uncertainties. The expected drawback is an increase in the overall transfer time, which might be incompatible with some mission timing requirements.

4. Rendezvous Strategy

Taking into account the remarks of Section 3, this Section focuses on the rendezvous trajectory design, highlighting the motivations behind the operational and optimisation choices.

4.1 Stochastic optimisation

The transfer optimisation problem may be formulated as follows: given a set of parameters $\mathbf{x}$, describing the transfer itself (e.g. intermediate orbit features, epochs, times of flight, etc.), the nominal transfer budget ΔV is a function of said parameters

$$\Delta V_{nom} = f(\mathbf{x}) \tag{6}$$

and the usual optimisation objective is the minimisation of the total ΔV. When dealing with uncertainties and operational constraints, the mission budget will be the sum of the nominal ΔV, obtained from equation (6), and a stochastic ΔV, resulting from TCMs, contingency allocations, and other non-nominal manoeuvres. The actual mission ΔV will thus be

$$\Delta V_{tot} = \Delta V_{nom} + \Delta V_{stoch} = f(\mathbf{x}) + g(\mathbf{x}) \tag{7}$$

In equation (7), the deterministic function f describes the transfer geometry and the nominal, ideal ΔV; g is a stochastic function, describing the procedure employed to compute the stochastic mission budget.

The objective of the current study was to mitigate the stochastic budget, resulting from the highly non-linear environment of non-Keplerian orbits in the Earth-Moon system, by the inclusion of the stochastic function g in the optimisation process.

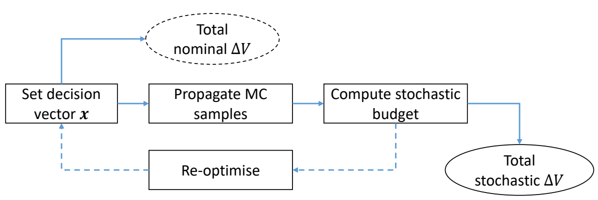

Figure 4: Stochastic optimisation work logic

Figure 4 depicts a schematic representation of the stochastic optimisation process:

- The usual optimisation approach aims at finding the decision vector, $\mathbf{x}$, that minimises the nominal ΔV; the stochastic component is added a posteriori, and included in the mission budget.
- The proposed stochastic optimisation process includes the stochastic component in the objective function. The decision vector is selected, such that the total ΔV is minimised, including the stochastic component.

The approach works as long as an estimate of the expected errors is available. In that sense, the new solution $\mathbf{x}$ might lead to a higher ΔV_{nom}, which in turn reduces the total mission budget by a further reduction of the ΔV_{stoch}.

4.2 Case Study

As case study, this Section investigates a transfer from the lunar surface to a large translunar NRHO, via an intermediate LLO. The ascent to LLO is not investigated in detail, and used exclusively to perform some assumptions on the LLO arrival timing. The chaser vehicle is thus assumed to begin the mission in a 100 km LLO, and tasked to rendezvous with a vehicle in NRHO, with a period of 7.5 days. No constraints are, at this stage, introduced on the LLO. This scenario was initially investigated in [10], and the operational challenges led to the need of the current study.

The following sequence of manoeuvres is performed:

1. After a loitering phase in LLO, a first manoeuvre is performed, to inject into a transfer arc heading to the target orbit.
2. A TCM is performed, after state determination is available; typically, at least 24 hours are needed prior to executing the TCM.
3. A second manoeuvre is performed, to rendezvous with the target vehicle.

Different, separate studies investigate proximity manoeuvres and guidance [20, 21]. Figure 5 depicts the trajectory in the Earth-Moon rotating frame, reporting the manoeuvre magnitude and time of flight for a sample transfer case. Note that the first manoeuvre, departing from the LLO, takes the majority of the deterministic ΔV budget, whereas the second rendezvous manoeuvre is one order of magnitude smaller.

In order to assess the stochastic ΔV, the TCM magnitude shall be estimated. For the current analysis, the assumptions in Section 2.3 were employed. Figure 6 depicts the trajectories obtained with a Monte-Carlo procedure, propagating the initial manoeuvre with execution errors and computing the TCM, necessary to perform rendezvous with the target. Each Monte-Carlo shot consists in:

1. Execution of the first manoeuvre with random errors (3% magnitude - 1.5°pointing, 3-σ);
2. Ballistic propagation for a given time, where orbit determination is performed;
3. Calculation of the TCM, to target the correct rendezvous point;
4. Execution of the TCM and ballistic propagation;

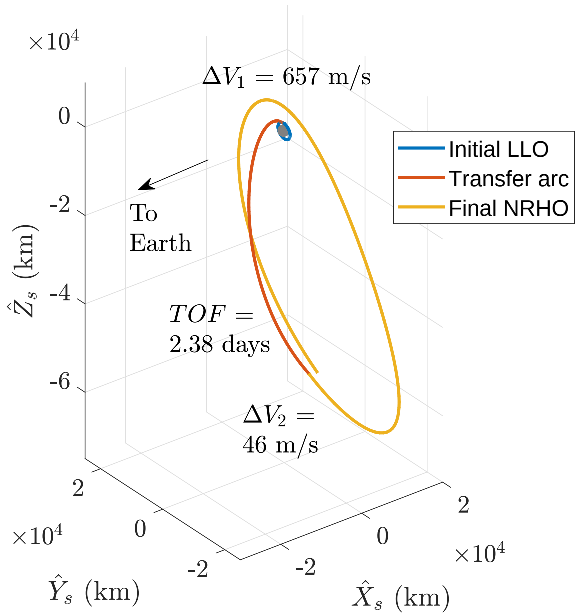

Figure 5: LLO to NRHO direct transfer case

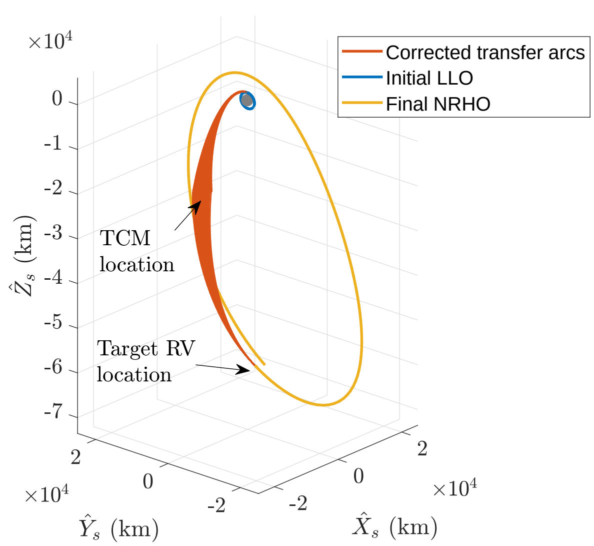

Figure 6: LLO to NRHO direct transfer TCM assessment

5. Execution of the second rendezvous manoeuvre.

The following aspects may be noted:

- The time span prior to the TCM strongly affects the magnitude of said manoeuvre. This is due to the strongly non-linear environment of the Halo family, although the behaviour might vary among different Halo orbits.
- The nominal rendezvous point (state and epoch) might be changed, after TCM computation, to achieve possible savings in ΔV_{stoch} budget. Such additional degree of freedom will be bounded by operational requirements and constraints; in the current case study, the location of the second manoeuvre was constrained to be at least 24 hours prior to the NRHO aposelene, to allow for proximity manoeuvring.

As reported in previous analyses [10, 9], execution errors of the initial manoeuvre lead to a large dispersion in the trajectory, which require an expensive TCM (more than 100 m/s) to be corrected; employing the stochastic optimisation strategy, described in Section 4.1, is able to reduce up to the 20% of the TCM magnitude.

5. Final Remarks

The paper presented an investigation of rendezvous and phasing, in non-Keplerian lunar orbits, focusing on the operational challenges that may arise from such mission and proposing a novel optimisation strategy, to mitigate the impact of the stochastic ΔV budget on the overall mission cost. The inclusion of operational aspects, in the early phases of mission analysis and design, is proven to be effective in reducing the mission ΔV. In particular, such benefit is significant in the non-Keplerian framework, where the effect of navigation and manoeuvring errors is more significant, because of the larger instabilities that might be triggered.

The results of the study also remark the effect of manoeuvre execution and orbit determination accuracy on the stochastic budget. The inclusion of these aspects, together with the available high-level constraints, as early as the trajectory design phase, results in a two-fold benefit:

- The analysis highlights the sensitivity bottlenecks of the mission design, aiding to note where mitigation measures may be employed (e.g. increasing the accuracy of the sensors, allocating additional ground passes, etc.). In later design phases, a trade-off might be performed at system level, identifying whether the proposed approach allows to achieve a system-wide cost reduction.

- The total mission ΔV may be reduced, using an optimisation process that considers the stochastic components as part of the decision vector. Such procedure relies on the knowledge of the high-level operational requirements, and, although effective, might result in a computational burden if a Monte-Carlo approach is employed.

Further design iterations are thus able to rely on a preliminary assessment of operational issues, enabling mitigation strategies, different options, and allowing a wider system view since early phases. The application to non-Keplerian translunar orbits proved to be effective in guiding the full mission design, identifying the manoeuvre dispersion bottleneck in such dynamical environment, paving the way for future design refinements.

Acknowledgements

The research was carried out at the European Space Operations Centre, under the Networking/Partnering Initiative (NPI) between Politecnico di Milano and the European Space Agency.

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