NAVIGATION ANALYSIS AND MANOEUVRES DESIGN FOR THE EUROPEAN STUDENT MOON ORBITER

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ABSTRACT

This paper presents the latest results on the navigation and orbit determination analysis of the European Student Moon Orbiter (ESMO). The paper contains an investigation into the required orbit determination accuracy to inject the spacecraft into an orbit around the Moon that can be maintained without control for at least six months. A new baseline transfer is proposed together with a navigation strategy which fulfil the stringent requirement on the total propellant budget available for this challenging mission.

1. INTRODUCTION

Scheduled for launch in the 2013-2014 timeframe, the European Student Moon Orbiter (ESMO) will be the first microsatellite in lunar orbit designed entirely by students [1]. Through the utilisation of a piggyback payload launch opportunity, ESMO will exploit the benefits of a Weak Stability Boundary (WSB) transfer to reach a highly elliptical orbit around the Moon.

The primary objective of the mission is to acquire surface images of the South Pole. This will be achieved through the insertion into a polar orbit, to allow the operation of a high-resolution narrow angle CCD camera (NAC). To complement the scientific return, a 3 kg Biological Lunar Experiment (BioLEx) is intended to investigate the effects of the space environment on simple biological organisms. The very limited budget available to achieve its ambitious mission objectives severely constrains the wet mass to be below 200 kg (the current wet mass is 189 kg) and imposes an all-day piggy-back launch requirement (i.e., a transfer should be possible for any launch date and for different launchers). To help alleviate the all-day piggy-back requirement, the use of WSB offers a means to provide a higher degree of flexibility in the selection of the launch opportunity. Yet this flexibility is slated against the expense of having to use a far more complex navigation strategy.

This paper presents a navigation analysis considering trans-lunar injection errors and Moon orbit injection errors. The analysis provides an estimation of the required Δv budget for re-targeting the nominal Earth to Moon transfer and orbital insertion around the Moon. Furthermore, it gives an estimation of the maximum admissible errors in position and velocity to achieve an orbit around the Moon that ultimately satisfies the mission requirements. The most stringent requirements are those concerning the low perilune altitude and a lifetime of over six months. This paper will present a trade-off analysis between the Δv budget and mission lifetime, for high elliptical lunar orbits. Special orbits such as elliptical inclined frozen orbits, also known as Ely's orbits [5], will be included among the potential options to reduce the mission cost. Major manoeuvres are modelled in details to assess the gravity loss, the required control accuracy and the criticality of each one. A nominal set of Trajectory Correction Manoeuvres (TCMs) are then planned to compensate for errors in the trans-lunar orbit up to lunar orbit injection (LOI).

2. ORBIT DETERMINATION AND NAVIGATION ANALYSES

The motivation for this analysis is to obtain an estimation of the required orbit determination accuracy. The first step is to derive a requirement for the accuracy of the determination of the lunar orbit injection point. An error in the determination of the injection point would translate into a potentially different orbit around the Moon. A different orbit will imply either a longer or shorter lifetime. Therefore, the first analysis will estimate the lifetime of the orbit around the Moon given an error in the initial conditions of the orbital elements. A second analysis will investigate the required accuracy at different times along the transfer orbit. The required accuracy in orbit determination must be such that one can correctly predict if ESMO will be captured around the Moon in an orbit with the desired lifetime. From the first analysis, a requirement on the insertion accuracy is derived. At the injection point, the insertion accuracy is given as a function of the error in position and velocity. This error is then propagated backwards. The set of back-propagated states defines a region (or cloud) in the state space that surrounds the nominal solution. Each point inside the cloud represents a pair of position and velocity that will lead to capture if the state is propagated forward. The orbit determination accuracy must be such that it can estimate, with 99% probability, that ESMO is within the cloud. The cloud will be called capture corridor in the remainder of this paper.

Sensitivity of the Orbit around the Moon

First, the sensitivity of the orbit around the Moon to errors in the lunar injection point was assessed. A random error was introduced in the orbital elements after the injection manoeuvre. The error values ranged from 1%-5%, of the nominal value of the orbital elements. A set of modified elements was randomly generated and propagated forward in time, using the AGI Satellite Tool Kit® (STK) software, for six months, or until the spacecraft was crashing. The altitude of perilune against time can be fou nd in Fig. 1, 2 and 3 for three different percentage errors and for 10 representative samples. The data plots clearly indicate that a 1% error is already potentially reducing the lifetime of ESMO by 20 days. In this paper we deem this reduced lifetime as an acceptable compromise between mission objectives and orbit determination requirements.

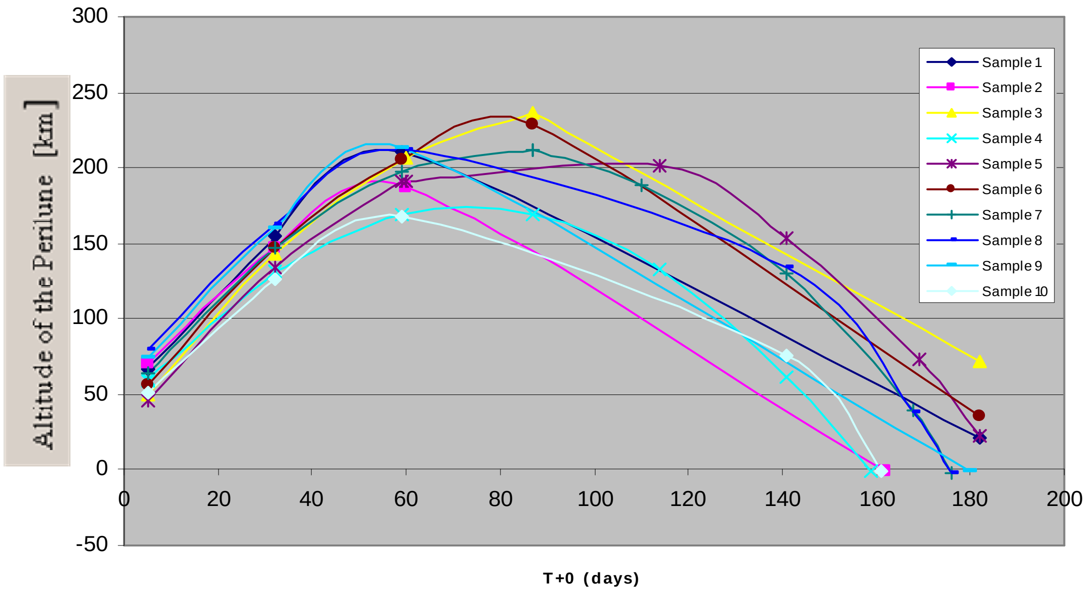


Fig. 1. Perilune lifetime with 1% error in the initial orbital elements.

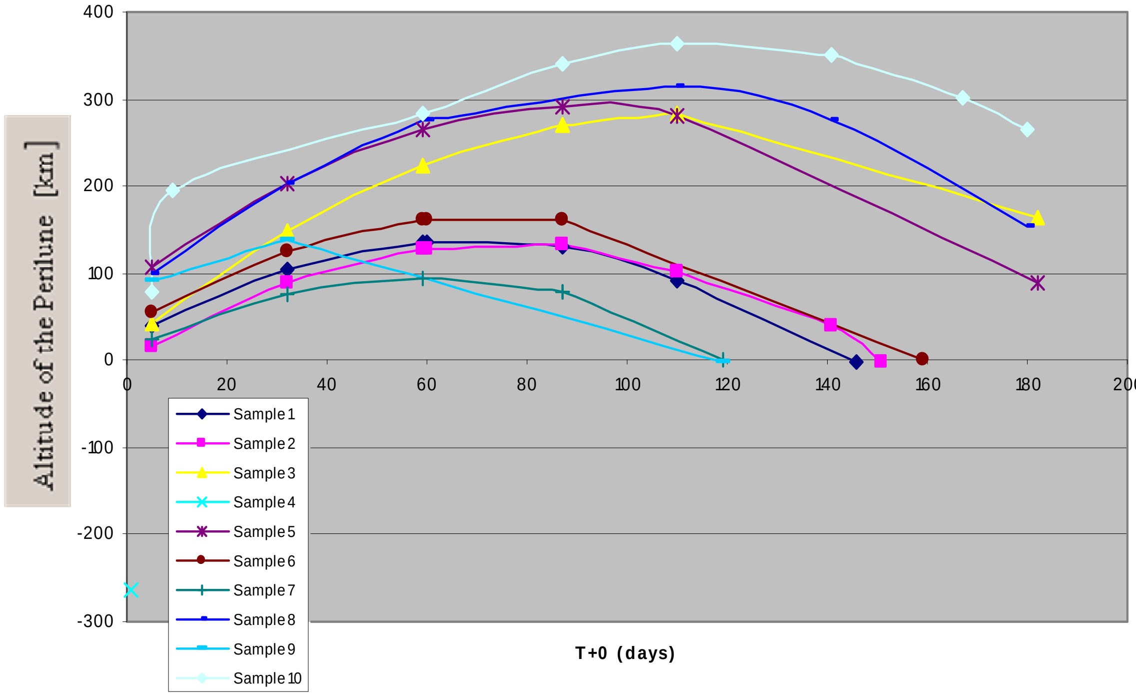


Fig. 2. Perilune lifetime for a 3% error in the initial orbital elements.

The 1% error in the orbital elements can be translated into errors in the radial, transversal and out of plane components of position and velocity at the Moon. These errors, illustrated in Fig. 4, are relative to the Earth and therefore detail the required capability of the ground station when measuring the position and velocity of ESMO.

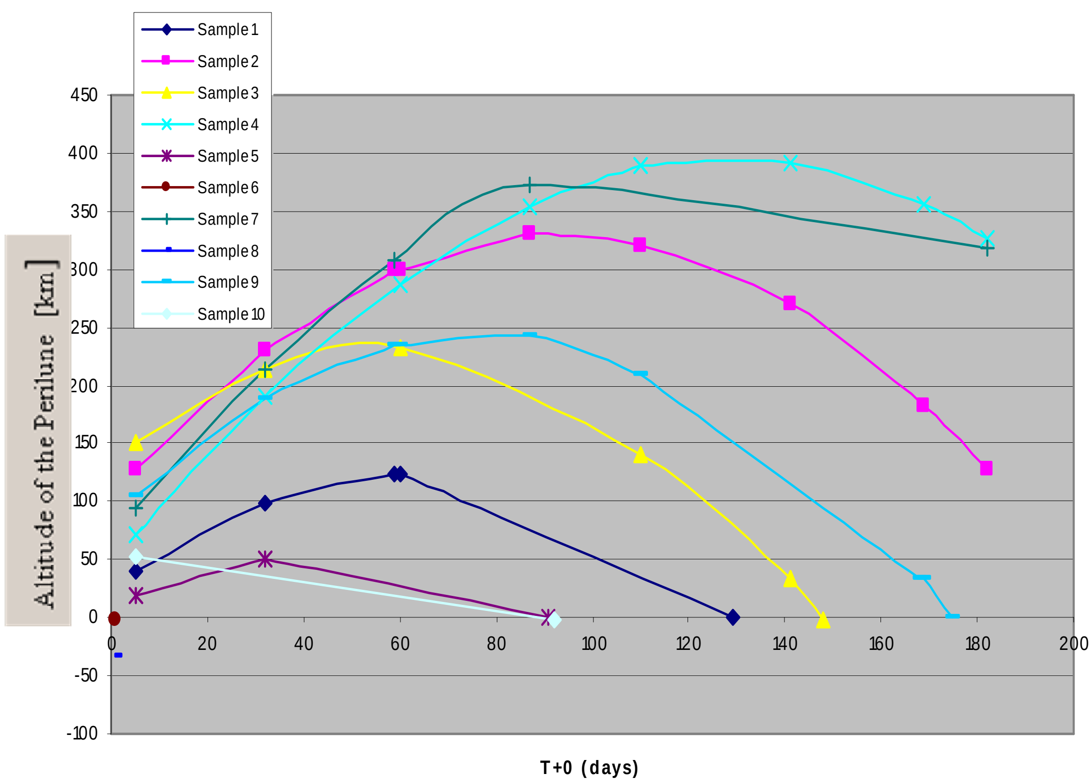


Fig. 3. Perilune lifetime for a 5% error in the initial orbital elements.

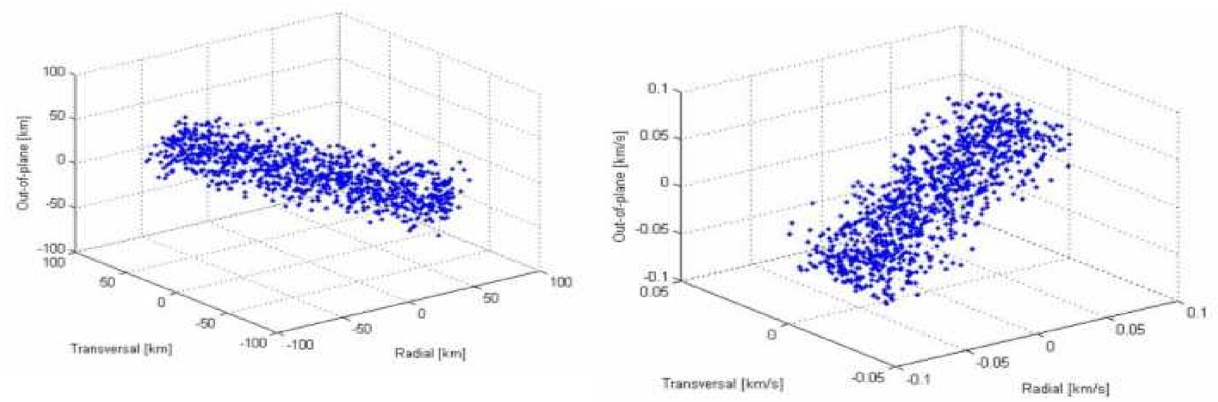


Fig. 4. Relative error in position (left) and velocity (right) projected along the radial, transversal and out of plane reference frame. The coordinate (0,0,0) represents the nominal solution.

Capture Corridor

The 1% error at lunar orbital insertion was used to define a region in the state space (position and velocity) at different times prior to the lunar orbit insertion. The region, or corridor, at $t_{insertion} - \Delta t$ defines the set of positions and velocities that ESMO must have at $t_{insertion} - \Delta t$ in order to be captured at the Moon, at time $t_{insertion}$, with at most a 1% error in the orbital elements of the lunar orbit. Note that the inaccuracies in the injection manoeuvre are not included in this analysis. The size of the corridor defines the accuracy of our knowledge of the position and velocity of ESMO. The orbit determination process must be able to discriminate whether the spacecraft is inside or outside the corridor; without which it will not be possible to predict whether or not ESMO is on course.

In order to know the size of the corridor at $t_{insertion} - \Delta t$, one needs to define at first the corridor at $t_{insertion}$. Fig. 5 is a sketch of the relative radial-transversal and out-of-plane reference frame at the nominal injection point. The axis h is perpendicular to the orbital plane of the spacecraft before insertion. One can define a displacement $\delta\mathbf{r}$ on the r-h plane and for each displaced point a set of variations $\delta\mathbf{v}$ of the nominal velocity. The displacements and velocity variations were randomly generated within a given range. The perturbed state vector $[\mathbf{r}+\delta\mathbf{r} \quad \mathbf{v}+\delta\mathbf{v}]^T$ was then propagated backward for Δt. The displacement vector $\delta\mathbf{r}$ is defined as follows:

$$\delta\mathbf{r} = \delta r\left[\cos\theta \quad 0 \quad \sin\theta\right]^T \tag{1}$$

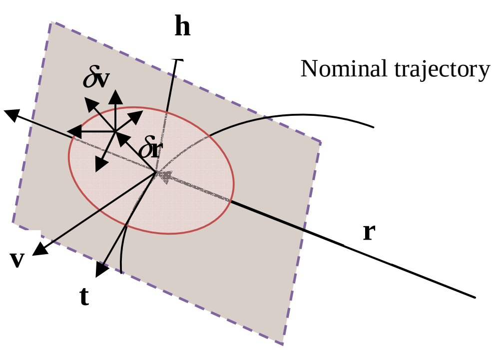
similarly, the velocity variation $\delta\mathbf{v}$ is defined as:

$$\delta\mathbf{v} = \delta v\left[\cos\vartheta\cos\phi \quad \cos\vartheta\sin\phi \quad \sin\vartheta\right]^T \tag{2}$$

where,

$$\bar{\phi} = \frac{\phi}{2\pi} \qquad \bar{\vartheta} = \frac{1}{2}\left(\cos(\vartheta + \tfrac{\pi}{2}) + 1\right)$$

The angle θ is sampled from the interval [0 2π], with uniform distribution, the quantities $\bar{\phi}$ and $\bar{\theta}$ are taken randomly within the interval [0, 1], with uniform distribution, while $\delta\mathbf{r}$ is taken from the interval [0, ε_r], with uniform distribution, where ε_r is the error on the position. With this choice it is implicitly assumed that there is 100% probability that the displacement is in that interval. Therefore there is a 100% probability that if ESMO is within the corridor then it is captured. The reverse is not true in general. For the velocities, $\delta\mathbf{v}$ is taken from the interval [0, ε_v] with uniform distribution.



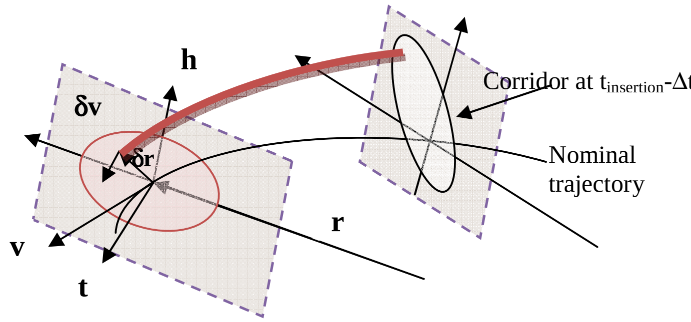


Fig. 5. Schematic of the *r*-*h* plane at the lunar orbit insertion point.

Fig. 6. Schematic of the back-propagation.

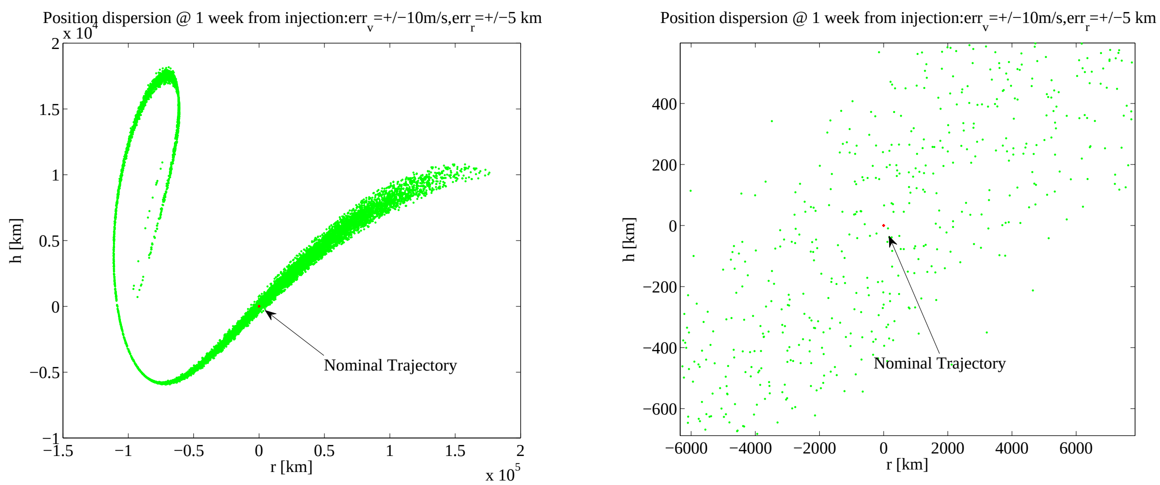


Fig. 7. Position dispersion at 1 week from lunar injection, *r*-*h* plane.

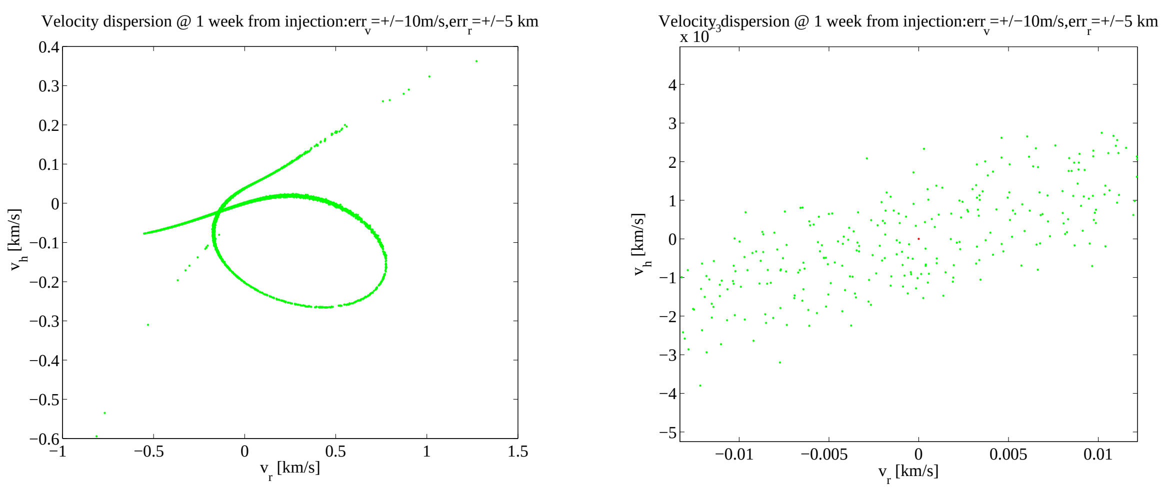


Fig. 8. Velocity dispersion at 1 week lunar injection, *r*-*h* plane.

A total of 10000 perturbed state vectors were propagated backwards for one week, two weeks and up to the WSB point (the farthest point from the Earth). The resulting positions and velocities were then projected on the r-h and r-t planes at epoch (Fig. 6 shows a sketch of a perturbed solution intersecting the r-h plane). Figs. 7-10 show the result of the backward propagation at one and two weeks. The green dots are the perturbed solutions forming the trajectory corridor, and the red dot is the reference trajectory of the existing baseline. The velocity plots give only the variation with respect to the nominal value; therefore they are centred on 0. For each figure, the right plot is a close up of the left plot around the nominal transfer. As long as ESMO is located within the trajectory corridor then orbital insertion around the Moon can be achieved.

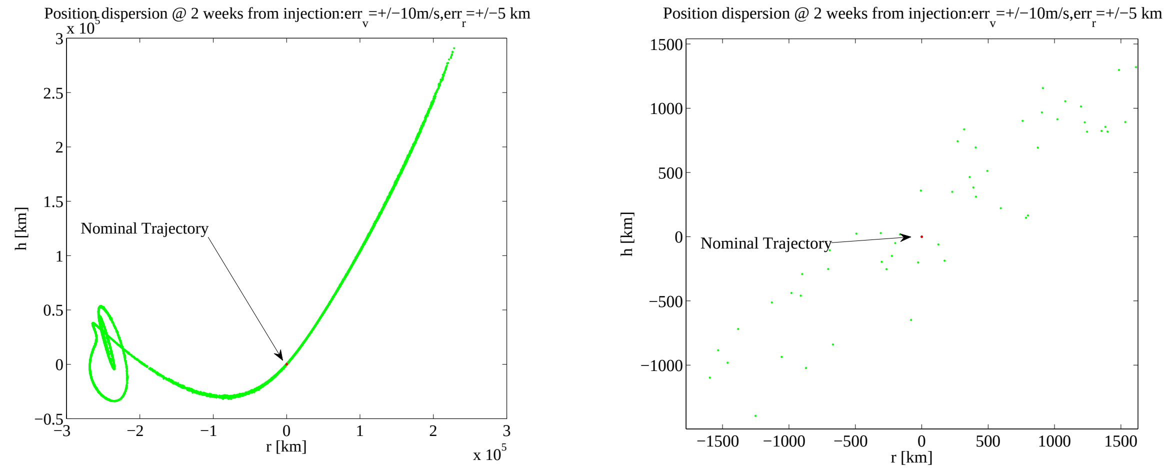


Fig. 9. Position dispersion at 2 weeks from lunar injection, *r-h* plane.

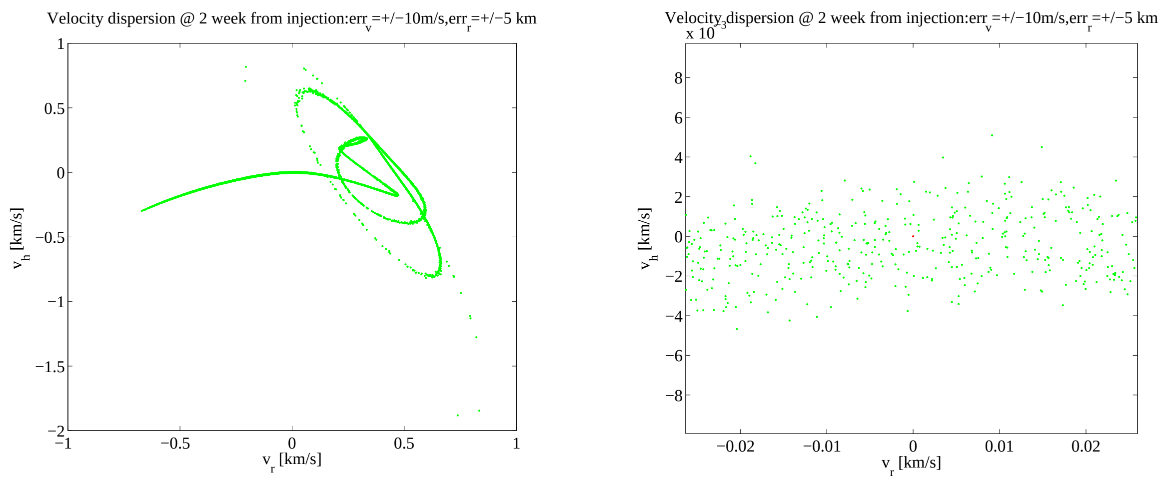


Fig. 10. Velocity dispersion at 2 weeks from lunar injection, *r-h* plane.

The figures were generated with ε_r = 5 km and ε_v = 10 m/s and represent only the projection of the corridor on the r-h plane. Similar figures can be obtained by projecting the corridor on the r-t plane. The trajectories corresponding to the curl will not reach the WSB region and do not represent feasible transfers. Furthermore, it is important to note how the corridor tends to get thinner in the normal and transversal directions while it seems to stretch along the radial direction. Based on the propagation of the corridor, and considering the required accuracy of position and velocity at the farthest point from the Earth, along the transfer trajectory, it was possible to derive the orbit determination accuracy reported in Table 1.

Table 1. Orbit determination accuracy requirements at 2 weeks from $t_{injection}$

Position	Velocity
25 km radial (range)	5 m/s radial (range rate)
10 km along track	1 m/s out of plane
10 km out of plane	1 m/s along track

Proposed Navigation Strategy

The corridor analysis also provides a way to define a robust navigation strategy. The main idea is to manoeuvre and maintain ESMO within the corridor with enough margin to accommodate the orbit determination errors. A number of Trajectory Correction Manoeuvres (TCM) needs to be inserted along the transfer, intermingled by orbit determination segments. After each orbit determination segment, a TCM is performed, if required. The orbit determination process in the paper was simulated using the ODTK® package. TCM's are used to ensure ESMO's position and velocity remains located within the trajectory corridor, thus enabling the correct lunar insertion. The action of each TCM is to reach the nominal reference trajectory. Here it is proposed an example with four TCM's along the Earth-WSB leg. The four manoeuvres aim at minimizing the distance with respect to the nominal point at the WSB region. It was assumed that the first orbit determination occurred one week after the trans-lunar injection from GTO and that the sources of errors are the trans-lunar injection burn and the typical dispersion errors of the launcher [2]. The error in the major Δv manoeuvres was assumed to be 1 m/s in every direction. It was also assumed that each TCM in itself was affected by an error that must be accounted for. This error was assumed to be 0.1 m/s in every direction. As before, a symmetric interval $[-\varepsilon \quad \varepsilon]$ around each nominal component of the Δv was considered and values were sampled, with uniform distribution, from the hypercube $[-\varepsilon \quad \varepsilon]^6$. The value of ε was set to 0.1 m/s for the TCM's and to 1 m/s the major Δv manoeuvres. Each TCM was assumed to be smaller in magnitude, and thus affected by a smaller error. It is acknowledged that these assumptions were only used as a starting point for the first iteration, and so will be updated in subsequent analyses. Following each TCM, the possible outcome of errors in both position and velocity of ESMO is measured at the next orbit determination point. It was assumed that this occurs once every three days. The sum of all the TCM's will lead to an increase in the mission Δv and propellant budget.

The example of navigation strategy proposed in this paper is given in Table 2. This corresponds to an additional increase of Δv of 84 m/s, leading to a total mission Δv of 1.1257 km/s. This is considered nominal, without additional margins. Fig. 11 shows in black the uncontrolled trajectory and in green the controlled one. The latter hits the required position at the WSB point with an accuracy sufficient to remain within the corridor.

Table 2. Example of an Orbit Determination and TCM Strategy

1st Orbit Determination	Start: 19/3/2011 05:54:30 End: 22/3/2011 05:54:30
TCM 1	Date :22/03/2011 16:42:4 Δv = 20.9 m/s
2nd Orbit Determination	Start: 2/04/2011 12:00:6 End: 5/04/2011 12:00.6
TCM 2	Date: 15/04/2011 20:18:23 Δv = 0.6 m/s
3rd Orbit Determination	Start: 19/4/2011 06:24:57 End: 22/4/2011 06:24:57
TCM 3	Date: 24/04/2011 5:29:35 Δv = 0.1 m/s
TCM 4	Date :26/04/2011 15:23:47 Δv = 51.5 m/s

This process has only been modelled for the GTO-WSB transfer, but needs to be extended across the entire WSB transfer, up to lunar orbit insertion. Therefore, an additional allocation of Δv to perform the TCM's will have to be considered within the overall mission Δv budget.

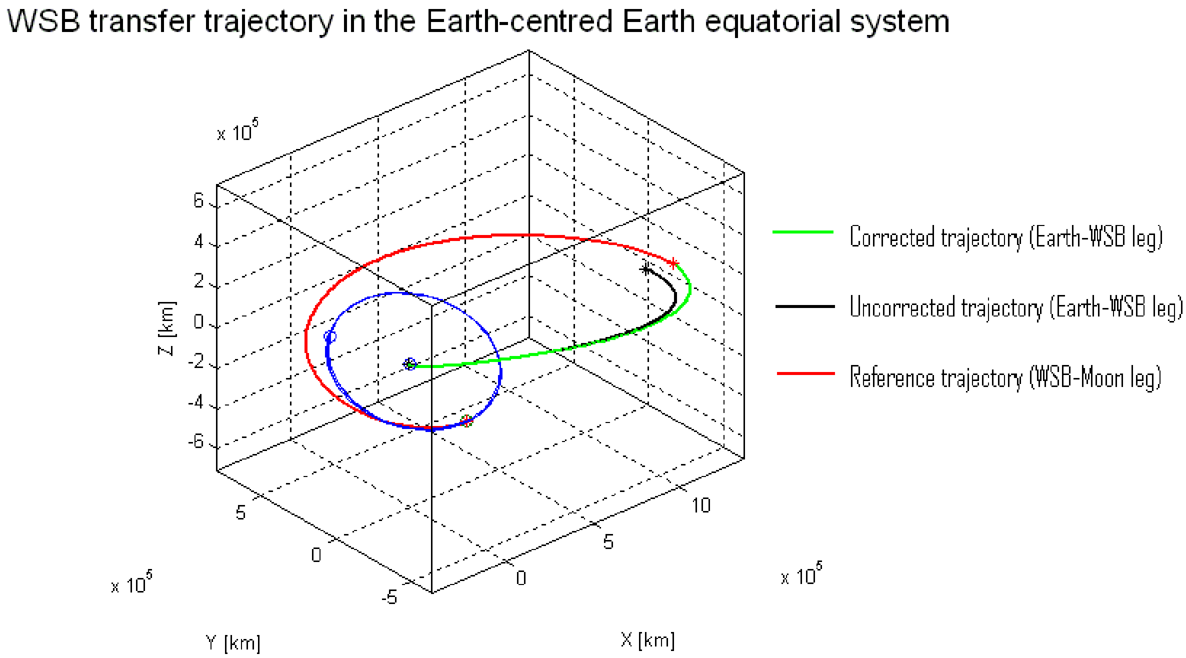


Fig. 11. Example of uncontrolled and controlled orbits.

3. NEW BASELINE OPTION

In the development of ESMO, mission cost and total mass are considered to be critical drivers. This is reflected in the requirement of a piggy-back or secondary launch opportunities into GTO and the extensive use of flight spares and non-space rated components (e.g., COTS). This imposes a constraint on the maximum size of the propellant tanks and thus on the maximum allowable Δv [4].

To account for this restriction, several options were investigated to reduce the total mission Δv. However, any adjustments in the orbital transfer and lunar insertion still had to remain compliant to the mission and system requirements. In particular, the lifetime of the orbit shall remain six months, while offering multiple passages at 200 km, or below, at periapsis, located at the South Pole. The requirement on the periapsis altitude derives from the narrow angle camera [5].

Initial trade-offs were conducted to assess where Δv could be saved. Possible locations included at launch, at GTO, and at lunar injection. The majority of the mission Δv is spent performing the transfer and the lunar insertion manoeuvre. However, the transfer Δv can be only marginally reduced. It was instead considered that a higher energetic and eccentric orbit around the Moon could lead to a major saving while still fulfilling the mission and scientific requirements.

Changes to the lunar orbit were first made by increasing the apolune altitude, which allowed ESMO to enter a higher orbit: values of 10000, 20000 and 56000 km were used. All other orbital elements were kept to the existing baseline, with the altitude of perigee constrained to 100 km to comply with the NAC requirement. The higher the apolune altitude, the quicker the orbit decayed. Entering a 10000 km orbit resulted in an orbital lifetime of approximately 4 months, while a 20000 km orbit decayed after 55 days, and a 56000 km orbit decayed under 30 days.

Since this analysis did not result in a stable orbit for six months, the authors explored the relative benefits of utilising a family of frozen orbits around the Moon. Frozen orbits offer stable liberation with no long-term, large scale variation in inclination, eccentricity and semi-major axis over a long period of time [6-8]. This results in a longer orbital lifetime and minimises and/or eliminates the need for additional station keeping and orbit control manoeuvres, therefore reducing the Δv (and associated propellant mass) needed to reach and maintain the selected orbital configuration.

Frozen orbits only occur under fixed conditions of argument of periapsis (ω = 90º or 270º) and inclination ($i \geq 39.2º$) [6,8]. The argument of perilune ω = 270º was selected to comply with the mission requirements offering perilune over the South Pole. The stability of three different families of frozen orbits was evaluated (see Table 3).

Table 3. Families of Frozen Orbits

Orbital Parameter	Case 1	Case 2	Case 3
a	6542 km	13084 km	6808 km
e	0.60	0.80	0.73
i	56.2º	56.2º	56.2º
Ω	104.99º	103.63º	98.27º
ω	270º	270º	270º
M	349.36º	345.51º	332.92º
$t_{insertion}$	56082.5799 MJD	56082.5799 MJD	56082.55082 MJD

In the table, a is the semi-major axis, e the eccentricity, i the inclination, Ω the right ascension of the ascending node (RAAN), ω the argument of the perilune, M the mean motion and $t_{insertion}$ the epoch of the injection manoeuvre. Each orbit is at much higher values of eccentricity and semi-major axis than ESMO's previous baseline lunar orbit. For each new case, the WSB transfer was re-iterated. This resulted in changes in the orbit's insertion date, right ascension of ascending node (RAAN), and mean motion.

The stability of the frozen orbits was tested by propagating the initial orbital elements with STK. It was assumed that the spacecraft was subject to the inhomogeneous gravity field of the Moon and the 3rd body effect of Earth and Sun. The Moon's gravitational force was modelled with the data gained from the Lunar Prospector Orbiter [10]. The sensitivity to the degree and order of the gravity field model was initially assessed by running STK with 20 and 60 zonal harmonic coefficients. This variation had no significant effect on the final orbital elements. Therefore, all later simulations were run using a 20th degree and 20th order gravitational model. However, since the existing gravitational models of the Moon have a degree of uncertainty, there may be some unknown discrepancy between the simulated evolution of the Keplerian elements and the experienced in-situ environment. Each simulation was run for six months.

All three cases provided an operationally stable orbit. However, Case 3 was the only orbit that adhered with the NAC requirement for the entire six months. Case 1 only offered low periline altitudes at the end of the mission, from day 145 onwards. Case 2 did not comply with the NAC requirement, although Case 2 did offer a sufficiently lower mission Δv of 0.855 km/s. There is a trade-off between the cumulative saving of Δv, altitude of perilune and mission lifetime. A comparison between the three cases in given in Table 4.

Table 4. Characteristics of the Three Families of Frozen Orbits

	Case 1	Case 2	Case 3
Stable Orbit	Yes	Yes	Yes
Compliant with NAC resolution requirement	From day 145 onwards	No	Yes, throughout the entire mission duration
Mission Δv (km/s)	0.947	0.855	0.948

It is because Case 2 is a highly eccentric orbit, with a large semi-major axis that it has the benefit of offering a low insertion Δv, while still maintaining an orbital lifetime. The reduction in Δv has a cumulative effect in reducing the required mass and volume of propellant. Due to this substantially lower Δv, Case 2 was selected for further analysis. During the trade-off process, the mission Δv was considered to be far more important than the resolution requirement of the NAC, however a good compromise between Δv cost and image resolution can be obtained with a slightly lower altitude frozen orbit. The lower altitude orbit was obtained starting from Case 2 and progressively reducing the semi-major axis in steps of 500 km. For each step, the variation in orbital lifetime was assessed against the mission Δv.

During this analysis, it was discovered that there was sensitivities to the orbital injection RAAN. Values below 100º resulted in the faster decay of the orbit with a reduction in the semi-major axis. This seems to be a characteristic of using highly elliptical frozen orbits. The progressive reduction of the semi-major axis of Case 2 lead to the identification of a stable frozen orbit with a good compromise between altitude and Δv cost. The orbital elements are: a = 10084 km, e = 0.8, i = 56.2º, Ω = 103.63º, ω = 270º, and M = 345.51º. This orbit provides coverage at a low altitude of perilune for approximately 55 days. This complies with the NAC coverage, although at all other times the altitude of perilune is varying (see Table 5). This has the benefit of offering additional flexibility in the operations of the NAC. Also of note, towards the end of the mission at day 170, there is the option to end the mission which benefits from the low altitude of perilune (37.92 km). A forced de-orbiting manoeuvre has an estimated Δv of 0.021 km/s, lowering the perilune down to the lunar surface. If not, the perilune of the orbit will naturally increase, allowing for a possible extension of the mission. This compiles with the decommissioning and de-orbiting requirement [5].

The WSB transfer had to be partially redesigned to match the new orbit around the Moon. Furthermore, the spacecraft would initially insert into a much higher, more eccentric orbit. This has the following characteristics: a = 13084 km, e = 0.8, i = 56.2º, Ω = 103.63º, ω = 270º, M = 345.51º and $t_{insertion}$= 4th July 2012 13.55.03 UTCG. This orbit combines the benefits of the frozen characteristics with a low insertion Δv. Following which, a first burn is used to lower the perilune and a second burn is required to lower the apolune. These manoeuvres would reduce the semi-major axis by 3000 km (a = 10084 km), with an additional Δv cost of 47.2 m/s. Despite having to perform an additional burn, the proposed transfer offers a mission Δv savings of 0.214 km/s. The lifetime of the orbit around the Moon can be seen in Table 5 where the altitude of the perilune is reported against the mission time. A full comparison of the old and new baseline transfers are given in Table 6. The new frozen orbit around the Moon allows the spacecraft to complete the transfer and the lunar orbit insertion with a total Δv of 0.902 km/s. This is under the 1 km/s requirement imposed by the available Δv from the propulsion system. However, the value of 0.902 km/s is considered nominal, and so does not include any margins or TCM's. This shall be added in later work.

Table 5. Perilune lifetime for the new baseline orbit around the Moon

T + (days)	Altitude of Perilune (km)	Argument of Perilune (deg)	Inclination (deg)	Eccentricity
6	106.412074	267.768	55.468	0.816841
34	366.039082	275.196	56.174	0.790743
61	796.395115	276.896	58.313	0.747587
88	1105.725242	273.287	59.853	0.716913
89	1162.787506	273.073	60.015	0.712189
116	1002.679975	267.703	60.201	0.728253
143	477.762938	266.284	59.506	0.780311
170	37.913977	271.264	59.561	0.823777
183	167.756208	274.111	60.752	0.810885

Table 6. Characteristics of the baseline transfer and the new proposed baseline transfer

WSB Transfer Comparison	Baseline	New Transfer to HE Orbit
Total Δv (plus additional orbit transfer at Moon) [m/s]	1116.29	854.85+47.2 = 902.05
Δv at Earth [m/s] (nominal escape)	747.7	748.25
Δv at WSB [m/s] (matching manoeuvre)	71.02	34.16
Δv at Moon [m/s] (plus additional orbit transfer)	297.57	72.45+47.2
Departure Date [UTCG]	25/02/2012 14:34	25/02/2012 19:03
Time of flight Earth-WSB [days]	40.82	40.76
Time of flight WSB-Moon [days]	60.31	59.08
Total time of flight [days]	101.13	99.84
Arrival Date [UTCG]	05/06/2012 17:39	04/06/2012 15:11
Arrival Orbit: semi-major axis [km]	3586	10084
Arrival Orbit: eccentricity	0.4875	0.8
Arrival Orbit: inclination [deg]	89.9	56.2

The final orbit was then assessed in terms of its ground station characteristics and eclipse duration. During LEOP, the first ground station is Kourou, following which nominal access is achieved through the Villafranca access point. Villafranca provides ground station access time during both stages of the WSB transfer, up to and including lunar orbital insertion.

Because of the thrust level delivered by the engines and the need to reduce the error in the major Δv manoeuvres, the trans-lunar injection manoeuvre was split in a number of intermediate burns. Similarly, at the Moon, the orbit insertion manoeuvre was decomposed into a few smaller size burns. This Multi-burn Strategy (MBS) is similar to the one proposed in [3]. MBS avoids performing a single manoeuvre with a high Δv and, of most significance, complies with the launch date flexibility requirement. The latter requirement is given in [5]. Each WSB transfer opportunity occurs roughly once-a-month, and therefore, depending on the exact launch date, ESMO may have to spend some additional days in an Earth parking orbit. A RAAN change may also be required as the orbit drifts due to the inhomogeneous gravity field of the Earth and to luni-solar perturbations. A worst case delay of 30 days was considered for the definition of the MBS. The trans-lunar injection manoeuvre was split into four separate manoeuvres. The first two are of similarly large magnitude. This is to raise the apogee of the GTO. After the second burn there is a wait time of 28 days, following which a small apogee manoeuvre is performed. The last burn inserts ESMO into the WSB trajectory.

The MBS adds 47.2 m/s to the total cost of the transfer compared to a single direct injection burn from GTO into the WSB transfer. The increase is due to the perturbing effect of atmospheric drag, J2 and 3rd body effects. However, utilising a MBS offers higher launch date flexibility, a reduction of the gravity losses per manoeuvre and an expected reduction of the navigation Δv. This is in comparison to the small rise in Δv. Table 7 contains a full comparison between the transfer with and without MBS at the Earth and at the Moon. The MBS at the Moon brings the spacecraft to the orbit with a = 13084 km. An additional 47.2 m/s are then required to acquire the desired final operational orbit. Hence, the total cost of the new solution is 949.3 m/s against the nominal 1116.29 m/s of the previous baseline, leading to a gain of about 167 m/s.

The complete transfer is represented in Figs. 12 and 13. In particular Fig. 13 shows a detail of the multi-burn strategy at the Earth with the spiral to progressively increase the apogee.

Table 7. Comparison between Direct transfer and Transfer with Multi-burn strategy.

Transfer with Multi-burn strategy and launcher orbit insertion at T0=4408.294 MJD2000:								
Manoeuvre	Earth manoeuvres				WSB	Moon manoeuvres		Total
	1	2	3	4	5	6	7	
ΔT [days]	0.438	1.184	27.956	0.420	40.759	59.079	3.505	1133.341
T [MJD2000]	4408.73	4409.916	4437.87	4438.29	4479.051	4538.123	4541.407	
Δv [km/s]	0.39348	0.23288	0.02223	0.14428	0.03416	0.05678	0.01829	0.9021
Direct transfer:								
Manoeuvre	Earth escape				WSB	Lunar Orbit insertion		Total
ΔT [days]	0				40.756	59.083		99.839
T [MJD2000]	4438.294				4479.051	4538.133		
Δv [km/s]	0.74825				0.03416	0.07245		0.85486

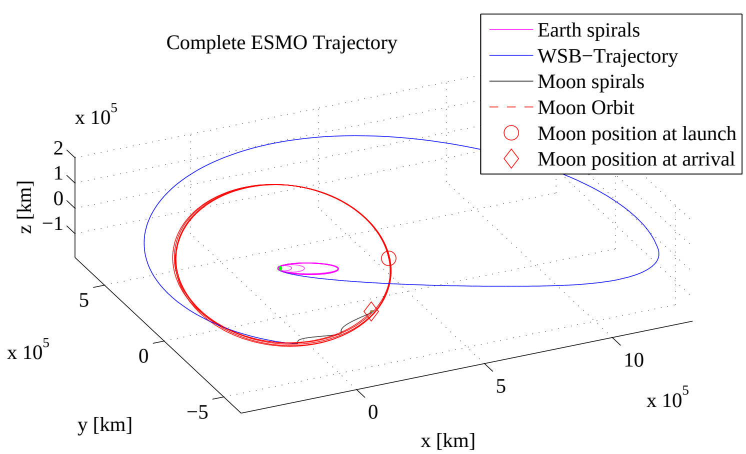


Fig. 12. Plot of the new baseline transfer with Multi-burn strategy.

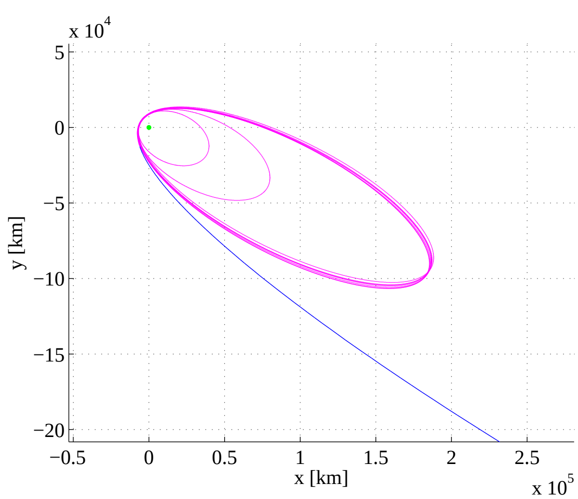


Fig. 13. Plot of the Earth spirals occurring during the Multi-burn strategy.

4. FINAL REMARKS

The paper presented a first analysis of the orbit determination requirements and a possible navigation strategy for the European Student Moon Orbiter. The proposed corridor-targeting approach yields good results at a relatively low Δv cost and with mild orbit determination accuracy. Therefore, the approach seems to be ideal for a small mission with low Δv budget. Note that the navigation analysis presented in this paper is based on a baseline transfer with a major trans-lunar injection manoeuvre. The MBS, by fractioning that manoeuvre, will lead to a reduction of the magnitude of the TCM's. Furthermore, the current planning and scheduling of the TCM's is not optimised. Future work will address the optimisation of the TCM's and an orbit determination process tailored on this small satellite. Finally, the combination of WSB transfer and highly eccentric frozen orbits provides a viable solution for relatively low-thrust, low-Isp missions.

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