MASTER THESIS

Mission Analysis for Google Lunar X-Prize Participants: Options For Going To The Moon

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Master in Aerospace Science and Technology

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ABSTRACT

Designing and launching a landed lunar mission implicated until now a significant development time and cost, which was only available for national space agencies.

The principal goal of one of the current competitions of the X-Prize Foundation sponsored by Google, the Google Lunar X-Prize, is to reduced such cost and time in order to make it available for the private sector.

On the base of the Google Lunar X-Prize competition, the present study proposes an analysis of the possible ways of reducing these costs, especially for teams being newly registered.
Several parameters and scenarios are defined in order to design a cost and time efficient landed lunar mission.

It has been concluded that the first priority at this early state is to obtain a launch contract which will define the spacecraft trajectory and hardware constraints.

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List of Abbreviations & Symbols

Abbreviations

Comsat	Communication satellite
DOI	Descent Orbit Insertion
ESA	European Space Agency
GEO	Geostationary Orbit
GLXP	Google Lunar X-Prize
GMAT	General Mission Analysis Tool
GTO	Geostationary Transfer Orbit
JAXA	Japan Aerospace Exploration Agency
JPL	Jet Propulsion Laboratory
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LLO	Low Lunar Orbit
LOI	Lunar Orbit Insertion
LOX(LO2)	Liquid Oxygen XPF
Mascon	Mass concentration
MMH	Monomethilhydrazine
MON	Mixed Oxides of Nitrogen
NASA	National Aeronautics and Space Administration
UDMH	Unsymmetrical Dimethylhydrazine
WSB	Weak Stability Boundary

Symbols used in text

$m_{propellant}$	spacecraft propellant mass [kg]
m_{dry}	spacecraft dry mass [kg]
Δv	effort needed to carry out an orbital manoeuvre [m/s]
I_{sp}	specific impulse [s]
g_o	acceleration at the earth's surface [m/s²]
Ω	right ascension of ascending node [°]
ω	argument of perigee [°]
i	inclination [°]

INTRODUCTION

The Google Lunar X-Prize is a $30 millions international competition, which prize is offered to the first privately funded team which achieves to send a robot to the moon, make it travel over a distance of 500 meters while transmitting to the earth a real-time high definition video. The final deadline is on the very last days of 2014.

The competition has been announced in 2007 and 21 teams have registered since then. However, the registration process is open until the last day of 2010, still giving the possibility to new teams to participate.

This study aims at analysing and defining the parameters permitting to optimisation of cost and time for these newly registered teams.

Analysing the mission parameters requires the definition of comparison factors. In [1], several options of weighting and rating of criteria are proposed for the determination of the importance of mission analysis factors.
In the present study, a GLXP mission analysis with short term delay, the most important criteria, proposed hereunder are listed from the most important to the less important.

1. Cost
2. Time
3. Spacecraft mass
4. Complexity

These criteria will be used for the analysis of each mission stage, from the launching phase, through the lunar transfer, to the lunar landing.

A first step in the spacecraft design will permit to define the total mass as well as propellant amount required for the total trajectory. Some launching cost estimations will be proposed in order to quantify the price.

After making several choices and defining some possible scenarios, their trajectories will be computed using GMAT, an open-source mission analysis tool developed by NASA, in order to illustrate and provide more concrete results to the study.

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CHAPTER 1

1 BEST OPTIONS RESEARCH

1.1 Mission stages

1.1.1 Launching phase

Out of all phases, launching an assembly composed of a payload, a cruise stage and a landing module from ground to LEO or GTO is probably the most constrained part of a lunar mission. Its excessive cost and geographical constraint restrict significantly the possibilities but permit, on the other hand, easier decision making.
In this section are investigated options and factors having the most favourable impact on the spacecraft launching phase.

1.1.1.1 Launchers

Investigating possibilities to launch its own payload to space rapidly leads to 2 options:
- the fabrication of its own launching system
- buy the launch services of a space agency or private company

The fabrication of a launcher on its own requires either to manufacture the launcher, or to transform an existing ballistic rocket. It is also necessary to define a launching site having a favourable position prior to the earth-moon trajectory. Transportation of the launcher on site and legal issues concerning the approval of the regional authorities for the launching phase may also turn out to be complicated.

Some GLXP teams are considering alternative launching ways to the rocket type. The balloon-launch solution [2] is probably the most interesting one, for which an atmospheric balloon is used to raise the lunar rocket assembly to about 18km of altitude. At this altitude the rocket assembly, containing the lunar lander and payload fires in order to reach LEO. Although it still requires the development of a rocket, the propellant amount may be significantly reduced. However, such options appears not conceivable in terms of time and complexity for short time development.

The transformation of an existing ballistic rockets is also an interesting option, but is based on existing military rocket. In the case of the GLXP, the overall spacecraft assembly, when considering all mission stages would require a rocket of the size of an intercontinental missile. It is most likely that such military equipment is not sold to private groups for personal transformation and use. Actually, such modified missiles are already available as space launcher but under the control of agencies, as for example the Russian Start-1 launcher formerly RT-2PM Sovietic intercontinental missile, the Israeli Shavit launcher based on Jericho II intermediate-range ballistic

missile or the well-known Soyuz launcher adapted on the base of the ancient R-7 Semyorka intercontinental ballistic missile.
Considering a 3 year period, it appears that such option, regarding to development in terms of cost, time and experience as well as the fabrication and tests of the launcher would not be feasible.

On the other hand, buying the services of a national space agency dramatically reduces the investment of time and team members but has, consequently, a large financial cost. The idea of such possibility is to take benefit of a planned launch in order to integrate the lunar assembly as a sub-payload.
Sub-payloads are also called piggy-back payloads. A piggy-back payload takes benefit of the excessive launching capability of a launcher, especially available for smaller private or academic satellites. Figure 1.1 gives an example of sub-payload integration on a launcher.

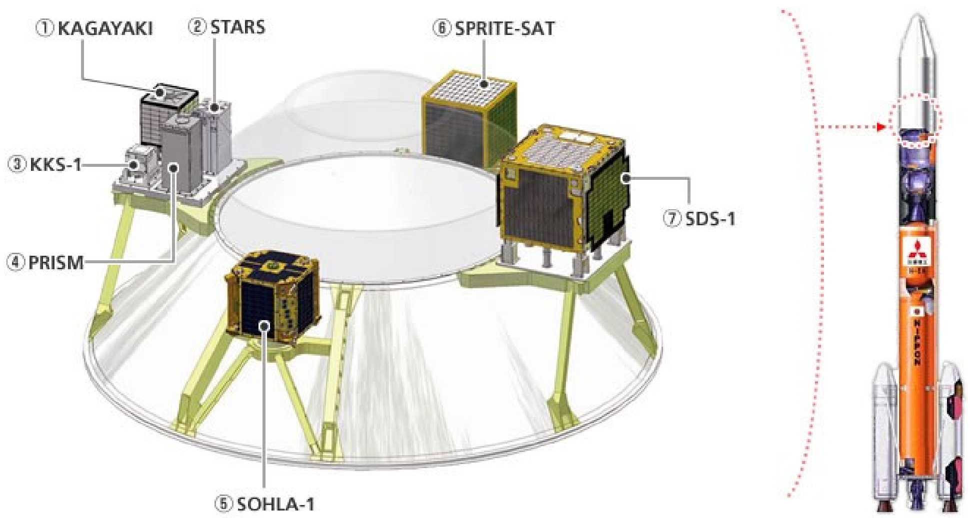

Figure 1.1 : Example of piggy-back: sub-payloads of the JAXA launcher H-IIA N° 15 [3]

<u>Lunar Mission Piggy-back launch:</u>
In the case of a GLXP team, a piggy-back launch directly aiming the moon would be the best case, because of the favourable launch inclination and ephemeris but also for the launcher's integrated stage for translunar insertion. Unfortunately, the low number of lunar missions and the expected launch cost may limit their availability to GLXP teams.

In table 1 are listed future lunar missions planned until 2014. Of course, expected dates of launch are very often postponed and a regular check for any launch information update should therefore be done. Table 1 may be completed by other missions currently in their proposal phase, especially concerning 2013 and 2014.

***Table 1** : Planned lunar missions*

COUNTRY	MISSION	EXPECTED YEAR OF LAUNCH	INFORMATION & REFERENCE
China	Chang'e 2	2010	www.spacechina.com
USA	GRAIL	2011	nasascience.nasa.gov/missions/grail
USA	LADEE	2012	nasascience.nasa.gov/missions/ladee
Russia	Luna-Glob 1	2012	www.roscosmos.ru
Russia	Luna-Glob 2	2012	www.roscosmos.ru
India	Chandrayaan-2	2013	www.chandrayaan-i.com
China	Chang'e 3	2013	www.spacechina.com
USA	ILN Node 1	2013	nasascience.nasa.gov/missions/iln
USA	ILN Node 2	2014	nasascience.nasa.gov/missions/iln

LEO & GTO Piggy-back launch:

The two types of launch occurring the most frequently are either LEO or GTO launches, which main payload are usually commercial satellites. When injected on its LEO or GTO orbit, the spacecraft will not take benefit of any launcher integrated stage for its translunar injection and will then require its own propulsion stage.

Although there are many ways and definitions of cost for space transportation [4], it is, in this case, most interesting to have a cost estimation with respect to the launching phase, by essentially analysing different launcher types that could be encountered. The cost per kilogram of payload is, in this case, a necessary metric in order to have a cost approximation.

Cost estimation of launch vehicles can be found in the literature [5], although not very often. An interesting and detailed cost analysis is developed in a launcher cost survey [6] from a space consulting company. It is proposed 2 ways of calculating the price per kilogram of payload; a generic metric and a specific metric. The generic metric provides an estimation using estimated launch costs and the published payload capacity. The specific metric computes the cost by the use of the exact cost of the launcher and the exact payload mass.

The data in tables 15, 16 and 17 in the annexes are generic estimation of the cost per kilogram per launcher, for the period 1990 to 2000. Prices are based on the dollar and euro values of the year 2000 and do not include costs of apogee kick motors or other payload injection means.
Falcon 1 and future Falcon 9 launchers have been added to existing data. These launchers are provided by the private company Space X, offering low costs launches since 2009 for commercial payloads.

When synthesizing the costs as in table 2[1], several observations are made.
It can be observed that the price per kilogram of large launchers is lower than for smaller ones, obviously because of larger payload capacities. The costs are also significantly lower in non-Western countries than for European or American launchers. It is also observed that the price between LEO and GTO launches is almost doubled for small launchers and tripled for large launchers.

[1] *Falcon launchers are omitted because of economical situation difference and dollar value between 2000 and 2009*

***Table 2** : LEO vs. GTO & western vs. non-western countries costs comparison*

	LEO		GTO	
	Western Countries	**Non-Western Countries**	**Western Countries**	**Non-Western Countries**
Small Launchers	22700 €/kg	11100 €/kg	50000 €/kg	24500 €/kg
Medium Launchers	13300 €/kg	6400 €/kg	32400 €/kg	26300 €/kg
Large Launchers	8800 €/kg	4800 €/kg	24500 €/kg	18100 €/kg

Naturally, launch costs greatly depend on the project requirements and negotiations with the space agency and the economical situation must be taken into account.
As represented in the table 18 of annexes for 2008 and 2009 [7] the number of LEO launches is higher than GTO (or orbits higher than LEO) launches.

Finally, it appears that the cheapest launch possibilities are LEO launches proposed by non-Western countries, for a cost of about 5000 €/kg. However, future test and commercial launches of newly developed launchers will also offer interesting low cost launch opportunities. These launchers are represented in table 3.

***Table 3** : Future launchers*

LAUNCHER	LAUNCHING SITE	EXPECTED DATE OF LAUNCH	TYPE
Falcon 9	Vandenberg & Omelek Island (USA)	2011	Medium
Vega	Kourou (French Guiana)	2010	Small
Angara 1.1	Plesetsk (Russia)	2011	Small
GSLV-III	Sriharikota (India)	2010	Heavy
Long March 6	China	2013	Small
HII-B	Tanegashima (Japan)	2010	Heavy
Neptune 30 (Interorbital Systems (IOS), launcher manufacturer)	Tonga (South Pacific Ocean)	2010	Small

Within the framework of this study, both LEO and GTO will be considered and their respective price per kilogram will be approximated to 10000 and 25000 €.

1.1.1.2 Launching sites

Although it cannot be directly chosen since it is defined in the same time than the launching epoch and the launcher, the launching site is a key parameter for the whole trajectory determination. Each site is characterized by its position with respect to the equator as well as its orbital inclination range.
In figure 1.2 is represented the geographical site of spaceports on a world map.

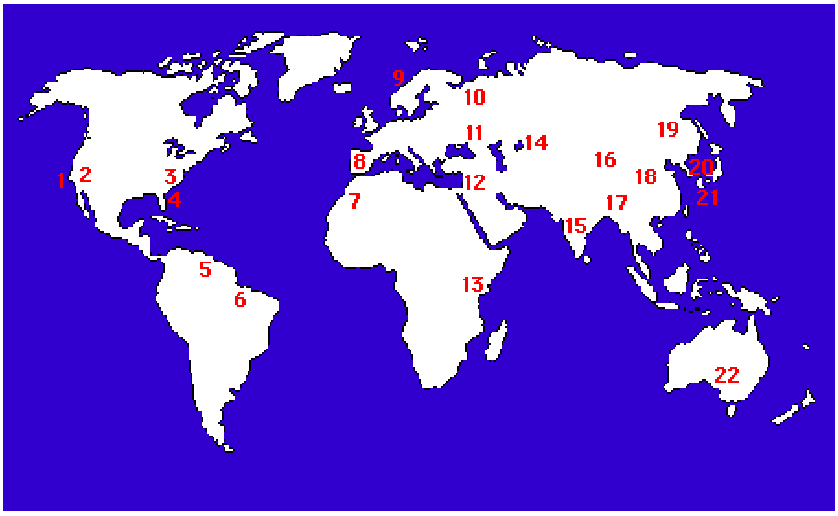

Figure 1.2 : Spaceports around the World [8]

According to the red numbering, launching sites are described in table 19 of annexes, providing latitude position, orbital inclination range, launchers used and commercial availability.

1.1.1.3 Launch scheduling

As it has already been introduced, the allocated time until the GLXP final deadline for teams registering within 2010, is of about 5 years. Therefore, the time management for each development phase requires a very specific attention.
A critical issue concerns the launch scheduling. Depending on the contractor, delays from contract signature until launch are not the same. A telecommunication company well-known from space agency launch services and familiar with their launching platforms will benefit of much shorter launching delays than occasional users. At Arianespace [9,10,11], these delays vary between 6 months and 2 years.
Although piggyback payloads may require shorter delays than launchers main payload, GLXP teams should book their launch as early as it can be, as well as take into consideration that the launching day is most likely to be postponed.
Furthermore, each launching platform has its own specifications for payload integration. Since a lunar assembly has a considerable volume and weight, the spacecraft and its modules should be developed according to the launcher's payload location.
Therefore, the research of a launch contract is a high level priority in order to respect delays, especially in the case of a GLXP team.
Even more than delays and the platform, the launch contract will provide the inclination, exact launching day for ephemeris, altitude and type of parking orbit (LEO or GTO) which are key parameters for trajectory computation.

1.1.2 Manoeuvring before transfer

1.1.2.1 Spacecraft positioning before translunar kick

When the spacecraft is launched, it can either be directly transferred into a translunar orbit or stay on a parking orbit and manoeuvred until being in the appropriate conditions to be transferred.

A trajectory coplanar to the moon plane can only occur when the spacecraft inclination is situated in the interval of the moon orbit inclination, which maximum is of 18.2° to 28.5° varying over a period of 18.6 years, as represented in figure 1.3.

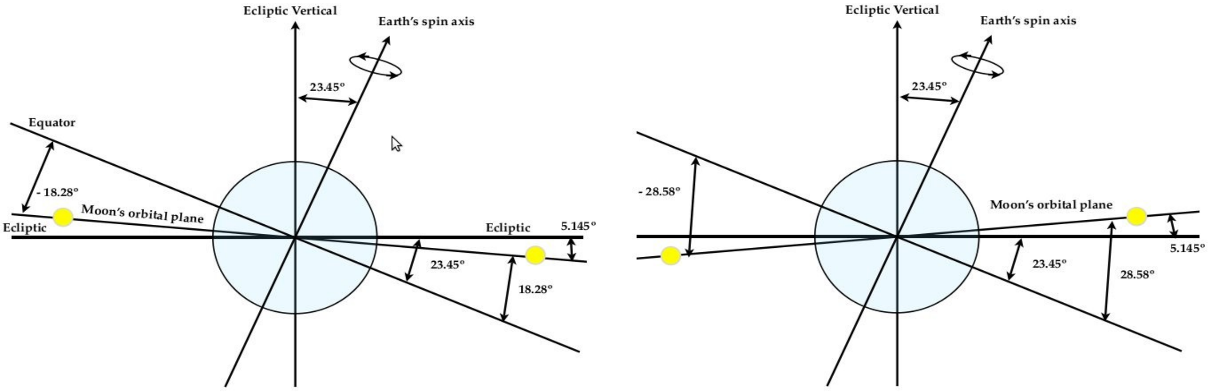

Figure 1.3 : Moon orbit inclination

The moon declination angle variation for 2011, 2012, 2013 and 2014 is shown in figures 6.1, 6.2, 6.3 and 6.4 of annexes, calculated with JPL ephemeris calculator Horizons [12]. It can be observed that the declination angle is actually decreasing, from about 24° in January 2011 to 18.6° in December 2014.

There are 2 manoeuvring options which have been analysed in the for Ariane 5 GTO launches [13]. Both cases are defined by the difference of right ascension of ascending node[2], Ω, of the initial spacecraft orbit with the moon orbit.

The favourable case, represented in figure 1.4, occurs when the moon orbit line of nodes is coinciding, or almost coinciding, with the spacecraft orbit line of nodes.
In this case, the moon is in the plane of the spacecraft orbit and a lunar transfer without inclination and Ω changes is possible. If the lines are not exactly coinciding, a mid-course kick on the moon sphere of influence, at lunar altitude of about 64000 km, is performed in order to correct the inclination difference. It is therefore defined ΔΩ intervals for which mid-course manoeuvres can be performed:

- $134° \le \Delta\Omega \le 207°$ (- 46° and 27° variation over 180°)
- $315° \le \Delta\Omega \le 29°$ (- 45° and 29° variation over 360°)

[2] *Angle between the ascending node with the Vernal equinox*

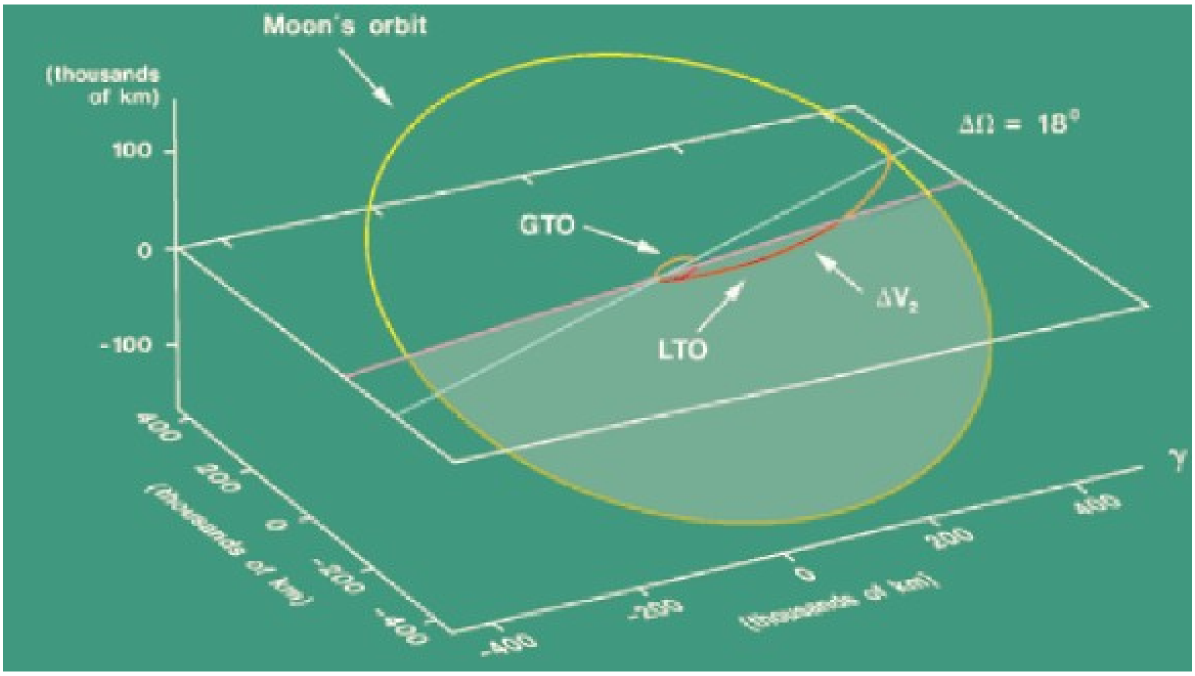

Figure 1.4 : GTO transfer with mid-course trajectory correction [13]

In the case of GTO Ariane 5 launches, the perigee is situated in the equatorial plane and therefore the argument of perigee[3], ω, is not to be modified. Such launch would represent one of the rare launches permitting a direct transfer without main positioning manoeuvres.
The only drawback is that the lines of nodes coincide twice a year, restricting significantly the team schedule. It has to be checked when such Ariane 5 launches will occur in the coming years.

If the $\Delta\Omega$ is not within the interval defined above, the spacecraft is either maintained in a parking orbit until the conditions are favourable again or manoeuvred in order to be transferred. The first case is the cheapest solution although the spacecraft is maintained on the parking orbit for several days. The second case may be chosen if the spacecraft is to be rapidly transferred although the manoeuvring Δv could exceed the translunar kick Δv.
In this case, the manoeuvres are the changes of inclination, Ω and ω.
The first one requires to bring temporarily the orbit plane into the equatorial plane. The plane change is particularly expensive at large orbit inclinations.
The second one, on the base of the first one and only relevant for GTO orbits, requires to temporarily circularise the orbit.

[3] *Angle between the orbit perigee and the ascending node*

1.1.3 Lunar transfer trajectories

When comparing different lunar transfer trajectories, the goal is to find a solution minimizing the injection velocity resulting in a lower Δv since the lowest injection velocity requires the lowest amount of propellant.
Lunar transfer trajectories depend first on the spacecraft thrusting type, which is either continuous or impulsive.
The continuous thrusting type relies on the use of electric thrusters. This thrusting type follows very different types of trajectories compared to those of impulsive thrusting. In the case of a lunar transfer, a spiralling trajectory, as in the case of ESA Smart-1, is considered, implying a very long transfer duration of about 20 months, and a higher complexity regarding spacecraft monitoring and attitude control.
This type of thrusting and its related trajectories, are not considered in this document since complexity and transfer time are critical issues

When talking about impulsive Δv for lunar missions, corresponding to the use of traditional rocket engines, 4 types of trajectories are considered:

- Hohmann transfer
- Bielliptic transfer
- Bielliptic transfer with flyby addition (trielliptic)
- Lunar transfers through Lagrangian points L1 and L2

The determination of the first 3 transfers can be approximated using the patched conic method particularly useful for Δv cost estimations. The patched conics method does not provide accurate results, especially for the earth-moon system for which the moon's sphere of influence is large with respect to the earth-moon distance, but nevertheless gives useful approximations.

The last transfer possibilities, lunar transfers through Lagrangian points L1 and L2, involves a three-body dynamics which determination is significantly more complicated and cannot be approximated analytically.

1.1.3.1 Hohmann transfer

As it is represented in figure 1.5, the Hohmann transfer is a direct transfer characterised by an elliptical orbit between 2 circular coplanar orbits with a sweep angle of π. In a earth-moon configuration, the spacecraft requires an acceleration thrust shot on LEO or higher orbit and a deceleration thrust shot on LLO.

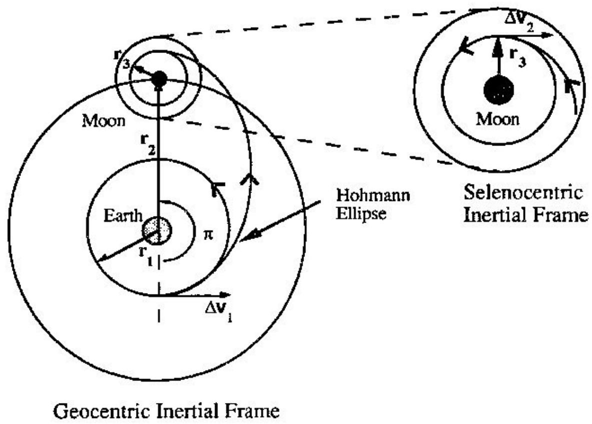

Figure 1.5 : Hohmann transfer (in the sun- earth rotating coordinates) [14]

Compared to other direct transfers, the Hohmann transfer is the one requiring the lowest Δv but having the longest duration. When increasing the Δv on the geocentric orbit, the trajectory will gradually pass from an elliptic form to a parabola, then a hyperbola and tending to a straight for an infinite Δv value.
The Hohmann transfer can have a Δv value as low as 3.95 km/s and a transfer time of at least 5 days.

1.1.3.2 Bielliptic transfer

The bielliptic transfer is characterised by patching 2 half ellipses of different size together along their semi-major axis. On figure 1.6, a first kick Δv_3 injects the spacecraft into a lunar transfer orbit of very high eccentricity, with an apogee of radius R. At apogee (for smaller impulse intensity), the second kick Δv_4 puts the spacecraft into the second transfer ellipse. The largest is the intermediate radius R, the lowest will be the Δv_4 cost. The 3rd kick Δv_5 finally injects the spacecraft into the final moon orbit.
In the case of a simple change of orbit involving a primary body and a satellite of infinitesimal mass, the bielliptic transfer presents a lower Δv cost than Hohmann transfer at the condition that the ratio of the final orbit radius to the initial orbit radius is greater than 15,84 [14]. In the case of a final lunar injection Δv, this ratio is no more respected because of the lunar gravity field.

1.1.3.3 Trielliptic transfer (bielliptic transfer with lunar flyby)

A lunar flyby addition permits a Δv cost reduction of the bielliptic transfer. For that, a third free kick is obtained by the gravity assist effect of the moon, resulting in a trielliptic trajectory. As represented on figure 1.6, the first half ellipse seen on the bielliptic transfer is divided in 2 ellipses when adding a lunar flyby.
A variant would be to take benefit of the lunar flyby on the Ellipse 3 instead of Ellipse 2, but would consequently increase the transfer time.

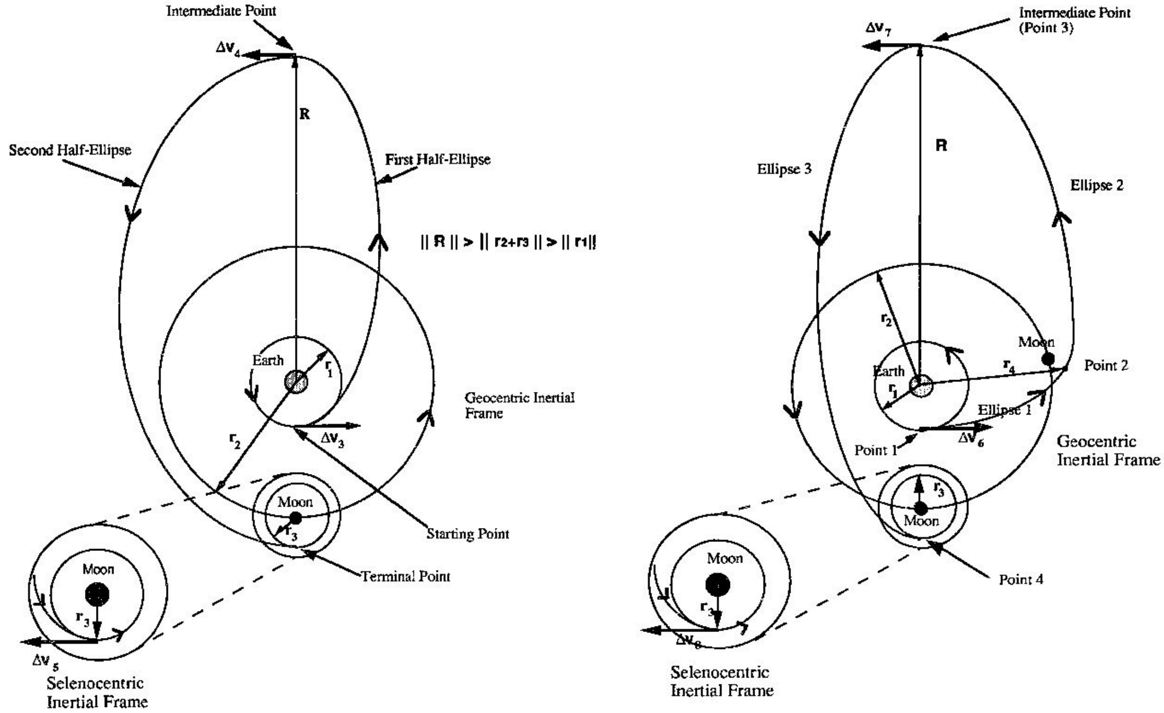

Figure 1.6 : Bi-elliptic (left) & Trielliptic trajectory(right) (sun- earth rotating coordinates) [14]

1.1.3.4 Lunar transfer through libration points

As a reminder, Lagrangian or libration points correspond to the stationary solutions of the circular restricted three body problem. In a system where 2 massive bodies orbit in a circular orbit around their mutual centre of mass, there exist 5 positions where a body of negligible mass can be placed and maintained at almost negligible energy cost in its position relatively to the 2 first massive bodies.
These points are represented on figure 1.7.

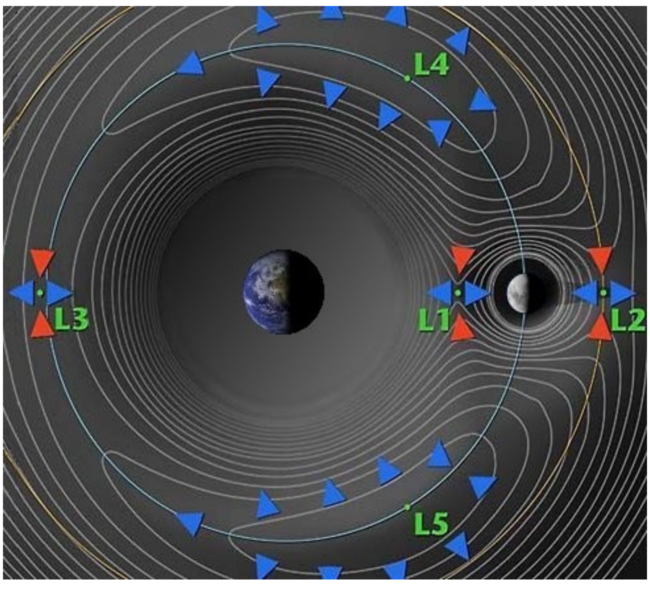

Figure 1.7 : Libration points of the earth-moon system [15]

These points represents interesting paths in order to achieve low energy transfers .
The equilibrium permitting a stationary position around L1, L2 or L3 is unstable and requires a constant station-keeping for a spacecraft situated in this position. However, lower energy is required for this spacecraft for approach and for trajectory redirection in the case such libration points are used as a path.

For trajectories to the moon, earth-moon L1 and L2 and sun-earth L2 are of interest as transit points, although resulting orbits are significantly more complex than Hohmann or bielliptic type transfers.
In the case of earth-moon transfer, a trajectory through the earth-moon L1 would have the shape represented on figure 1.8.

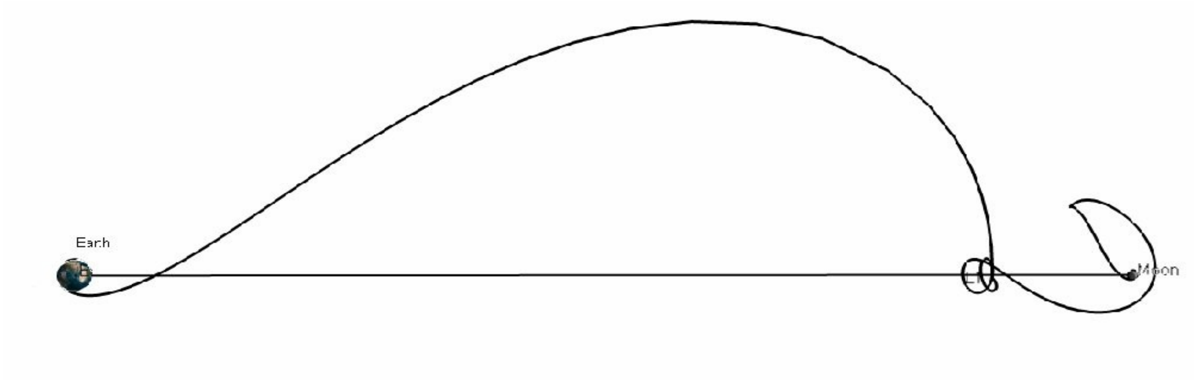

Figure 1.8 : Example of translunar trajectory through L1 (earth-moon rotating coordinates) [16]

This trajectory is characterized by a transfer from earth to earth-moon L1, Lissajous orbiting about L1, the lunar insertion and finally the descent until reaching the lunar surface.
In the case illustrated in figure 1.8 [16], extra manoeuvres to proceed to lunar descent are particularly expensive in Δv, representing almost the double of Δv of what is needed for LOI of a Hohmann transfer (1.55 instead of 0.85 km/s). The transfer time is of about 33 days, where 22 days are spent on the Lissajous orbit around L1.
Trajectories through earth-moon L2 and L1 are currently under investigation [17] for future lunar orbiting missions.

A trajectory passing by the sun-earth L2 libration point permits to save Δv for lunar periapsis arrival kick, permitting ballistic injection decreased of some hundreds of m/s [16] compared to a Hohmann transfer [18]. An example of such trajectories is illustrated in figure 1.9.

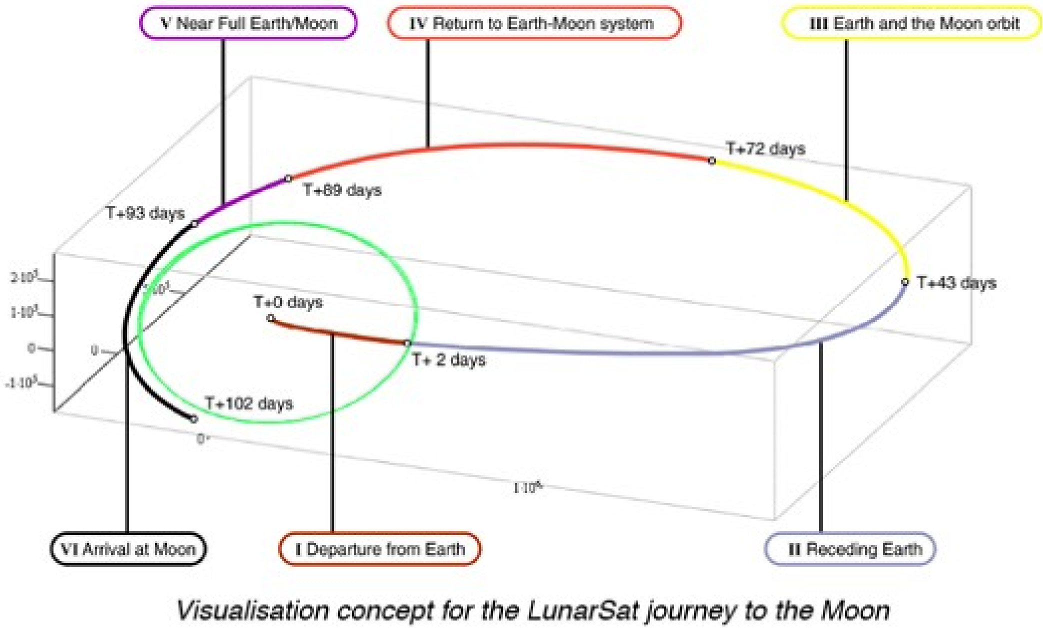

Figure 1.9 : Trajectory of LunarSat journey through L2 (earth inertial coordinates) [19]

However, a spacecraft using such trajectories requires a considerable time to reach the LLO, about 100 days, where a Hohmann transfer requires a minimum of 5 days. Furthermore, lunar transfer orbit injection is highly sensitive to magnitude and direction thrust vector. A special attention is to be paid to monitoring and attitude control at libration point maneuvering.

1.1.3.5 Lunar trajectories comparison

For the trajectories described in the last sections, it is used the same criteria, out of the mass, that have been defined in the study introduction, which are:

- cost
- time
- complexity

Cost is directly related to the Δv for each trajectory.

In a matter to quantify the Δv impact on launch costs, their corresponding values [14], propellant mass and calculated cost for different spacecraft dry masses are represented in table 20 of annexes.
In order to obtain a direct comparison, bielliptic and trielliptic trajectories have been estimated with an intermediate radius R of about 1.5 x 10^6 km which corresponds to the distance from earth to sun-earth Lagrangian point L2, situated in the WSB [20]. Trajectories through earth-moon L1 and L2 have voluntarily been omitted since they are still not well evaluated for lunar landed missions.

It is shown that, out of the bielliptic transfer which is more expensive, the approximate launch price for the only translunar kick propellant is of about € 13.5 millions.
In table 4 are attributed grades, according to the 4 trajectory types, to each of the parameters mentioned above, and is also provided an overall grade for each trajectory.

Table 4 : Overall grade of trajectories

Trajectory type	Δv	Transfer time	Complexity	Overall grade
Hohmann transfer	OOO	OOO	OOO	Good
Bielliptic transfer	O	O	OO	Bad
Triellitpic transfer	OOO	O	OO	Medium
Transfer through L2	OOO	O	O	Bad

The Δv values of [14] in table 20 are approximative. They represent the theoretical best case that can be obtained for a lunar transfer and for which initial conditions are difficult to be gathered. Such conditions can be found when repeating the trajectory computation while varying the initial epoch, day by day, over a very large time period.
Their corresponding costs are also approximative and do not permit to see significant differences between a Hohmann, a trielliptic orbit and a WSB transfer. Therefore, the grades given in the table 4 above for these 3 types are the same.
Considering the transfer time, the Hohmann transfer clearly has the advantage, several days instead of several months. The same is observed concerning the complexity, of computation and attitude control, for which the WSB transfer is the worst.
Finally, it can be considered that bielliptic and WSB transfers are not adequate options within the present case, where cost and complexity impact are too important. The trielliptic transfer, although a little bit better, still have a transfer time rather long.
The best overall grade is given to the Hohmann transfer, appearing as the optimal solution and, thus, considered as the best option in this study.

1.1.4 Moon orbiting, descent & landing

Lunar approach and landing are the last critical manoeuvres before the rover performs its run on the lunar ground.
The following examples are based on the landing strategy of Luna 17 mission [23] and of Apollo missions [16, 21]
The first one had the following descent scheme:

- Lunar capture on a circular orbit at 90 km of altitude and inclination of 141°
- De-orbiting from 90 to 20 km before final descent.

The Apollo landing phases, described in more details and represented in figure 1.10, 1.11 and 1.12, are the following:

- Lunar orbit insertion at 111 km of altitude
- Descent on an elliptical orbit, from 111 km to 15 km
- Braking manoeuvre from 15 km to 2.1 km
- Approach, from 2.1 km to 152.4 m
- Vertical landing, 152.4 m to touch down

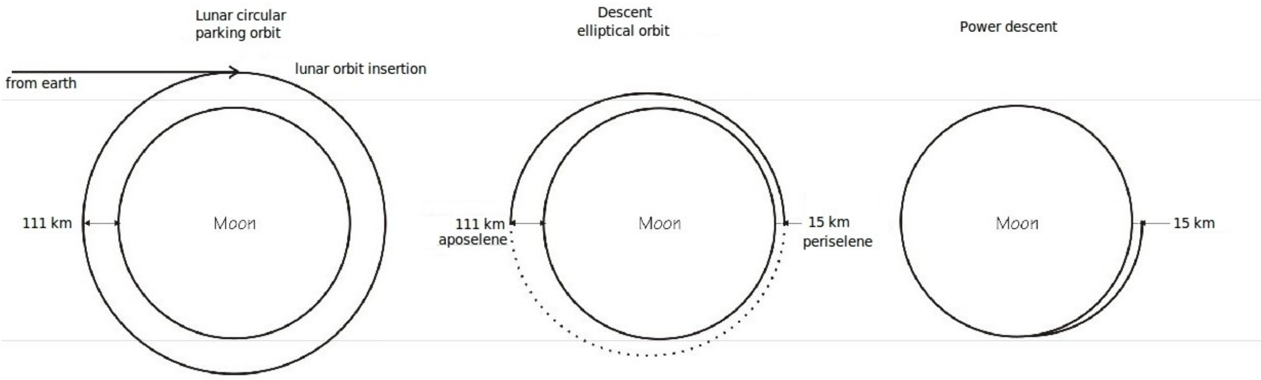

Figure 1.10 : Lunar orbit injection & descent [16]

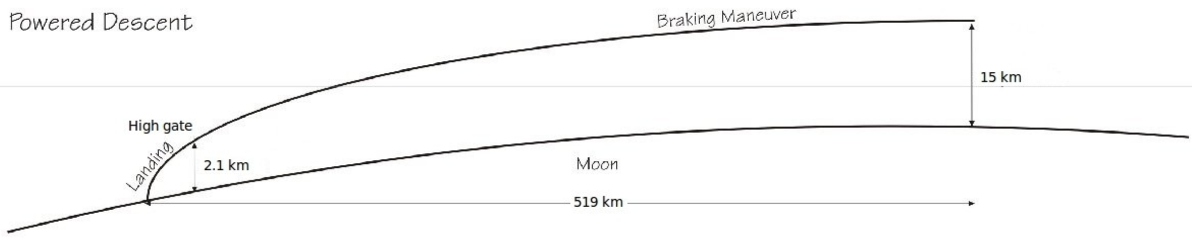

Figure 1.11 : Braking manoeuvre from 15km of altitude [16]

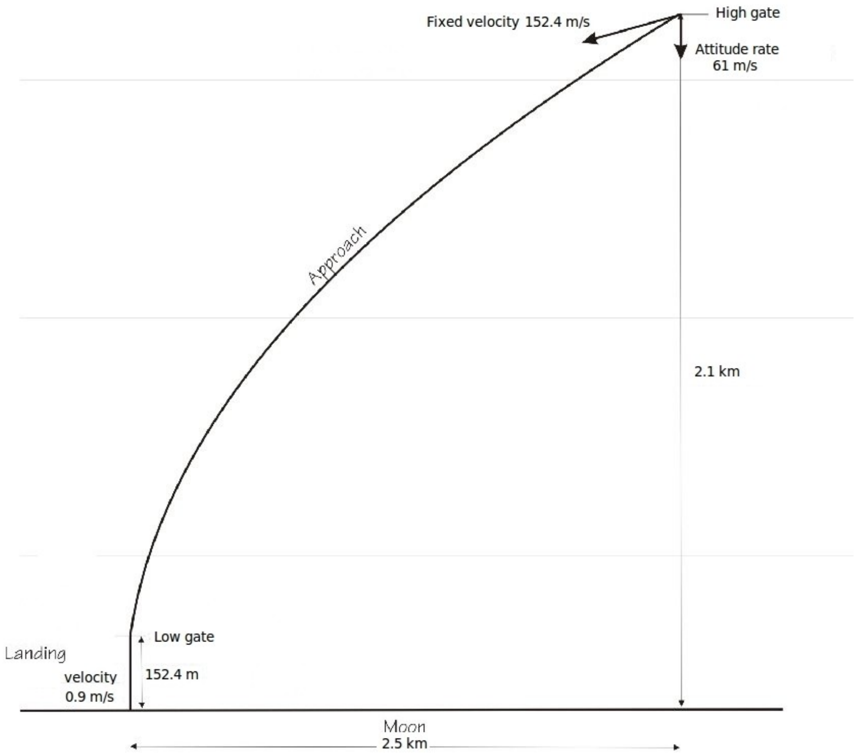

Figure 1.12 : Approach & final landing [16]

In the landing scheme described above, which is usually taken as basis in lunar landing studies, the altitude before descent is of about 15 km, altitude generally used because of the lunar ground topology and guidance errors.
However, the de-orbit cost to reach this altitude is rather high.
It has been investigated [22] that, using optimisation loops for descent computation, the total de-orbiting and descent cost is minimum at a perilune altitude of about 61 km.

Although trajectory models built with GMAT, presented in a later section, will provide simplified descent procedure, it is of interest to define the impact of lunar altitude at lunar capture on the descent Δv cost.

1.1.4.1 Landing site

The landing site has a great importance since it is where the spacecraft will have its first contact with the lunar surface and where the rover will have to perform its 500 meters.
In order to define an appropriate landing site, several criteria are to be considered:

- *Lunar inclination*: the spacecraft should reach the lunar orbit without large geocentric inclination changes, resulting in higher Δv costs.
- *Topography*: the ground should be smooth enough for rover displacements
- *Lighting conditions*: spacecraft should be landed in lighted zones for energy supply through solar panels
- *Telecommunication*: landing on the far-side should be avoided since the moon's body acts as a shield for radio transmissions from the earth.
- *Heritage bonus prize*: the first team that captures images of a past lunar mission wins a bonus prize.

According to the last criteria, and stated in the GLXP guidelines, the landing site is to be defined and submitted to the GLXP committee in order to be approved.
Two different cases can be considered, defining if it is wanted or not to land near a past lunar mission site. If considering that this criteria is of less importance, then the landing site can be defined according to the trajectory of less Δv.
On the contrary, landing near former lunar mission sites permits to take benefits of the mission data but also putting an additional constraint on the trajectory computation and cost.
The illustration of figure 1.13 represents the landing/operation and of past lunar missions.
The most convenient site would be where 1 or 2 past missions sites are nearby, improving the probability to observe them, and where the topography is as flat as possible as the rover will have to travel over a minimum distance of 500 meters.

Several possibilities show favourable conditions such as the Surveyor III/Apollo 12 site (Apollo 12 main objective was the site visiting of Surveyor III probe), Apollo 11, Surveyor V or Luna17/Lunokhod 1.

The latter will be an interesting option, since it appear to rely next to a particularly flat region. Furthermore, Lunokhod 1 [23] was one of the 2 Soviet lunar rovers. It travelled over a distance of about 10.5 km north and south of its landing point and, thus, now proved that the ground state is good enough for rover exploration.

The Luna 17 landing site is situated at 38.28°N, 35.00°W and the current lander position is estimated at 35.19°N, 38.28°W, north from the landing point.
The area is represented in details in figure 1.14.

It is most likely that, according to GLXP judging panel, these past missions sites must be approached by the teams rover and not directly landed on site. This will depend greatly on the landing accuracy.
However, it seems most probable that each of these historical sites will have a defined security perimeter in which it will be prohibited to land.

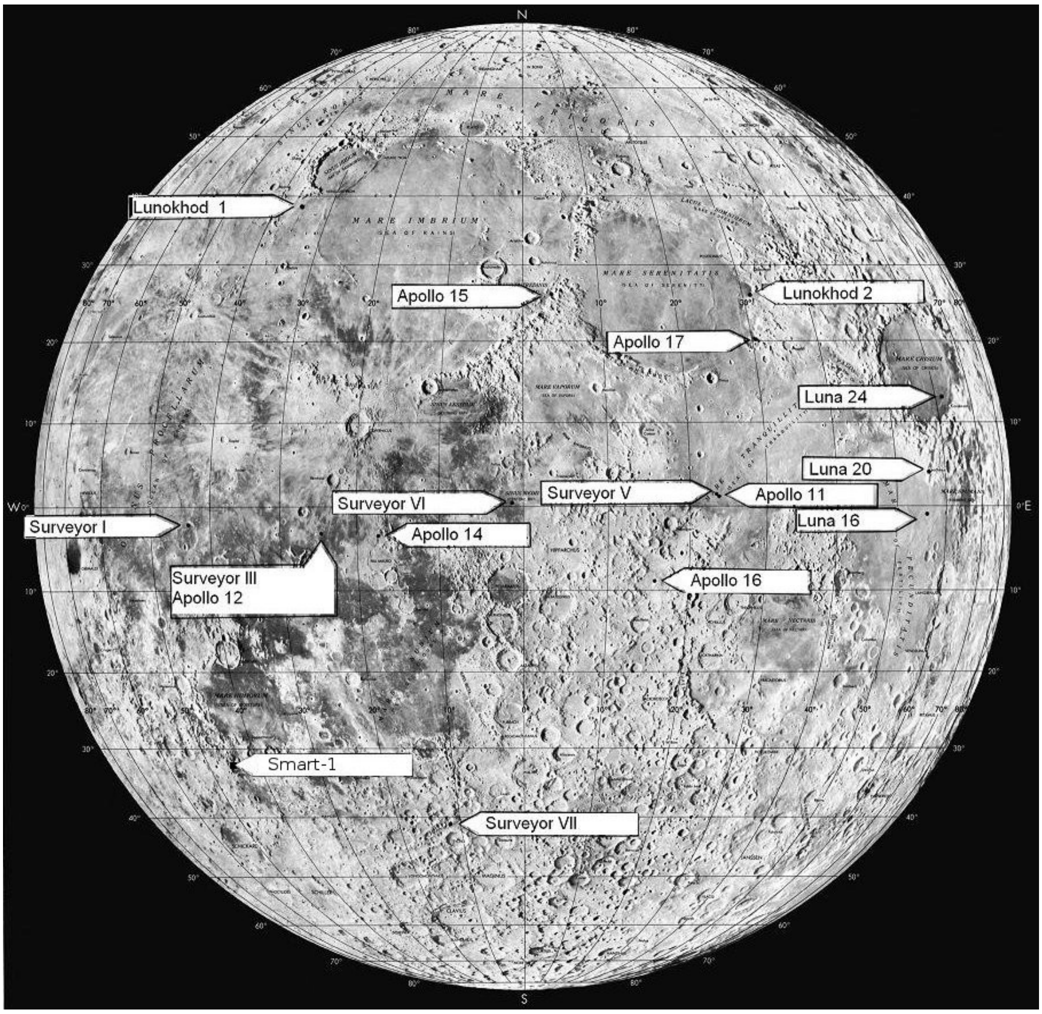

Figure 1.13 : Lunar landing sites of past missions [25]

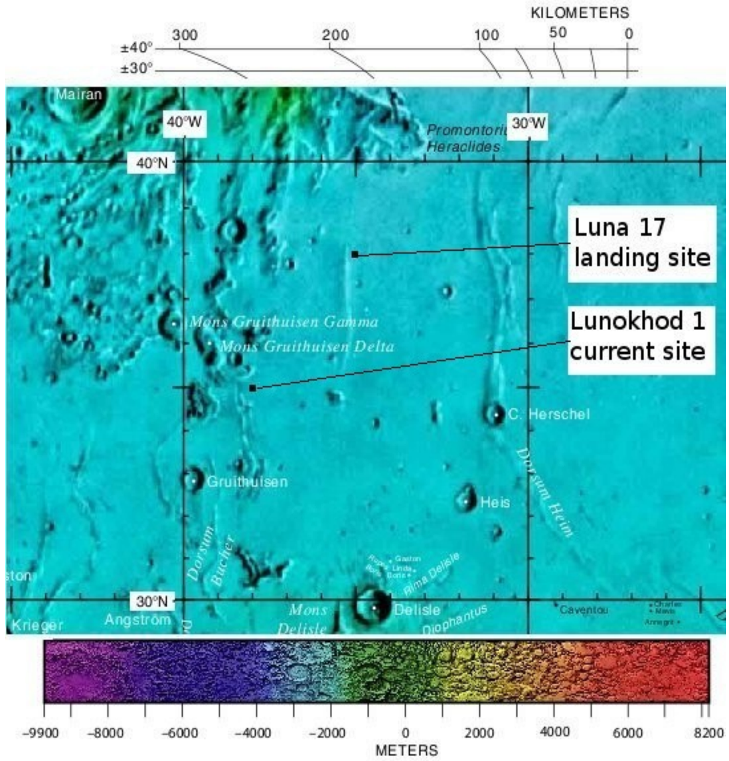

Figure 1.14 : Luna 17 / Lunokhod 1 area topography & distance [26]

1.1.4.2 Low lunar orbit injection

Independently of the chosen trajectory transfer, the low lunar orbit in which the spacecraft is injected will be defined by its inclination, altitude and coast time before descent.
The moon has a particular gravitational field because, among others, of the presence of mascons (mass concentration) inducing gravitational irregularities and significantly affecting lunar orbits. This perturbation has been discovered at the end of Apollo 16 mission when the sub-satellite PFS-2 crashed on the moon's surface only 35 days after its release from orbiting Apollo spacecraft.
Even if these irregularities rather affect long duration orbiting, their consideration for short duration orbiting, followed by a lunar landing, will enhance the precision of the latter manoeuvre.
Fortunately, specific orbit inclinations, for which this perturbation has no effect, have been defined [27] with the help of precise lunar gravitational models. These orbits are called "Frozen Orbits" and are very useful to maintain a spacecraft indefinitely orbiting around the moon. They have the particularity to have a constant eccentricity, inclination and periapsis direction (in average). Divided in families, they are the solutions of each family representing eccentricity dips, which value is minimized. Their inclinations are: 27.6°, 53.4°, 78.2°, and 86°. It appears that these orbits are more stable when maintained at a low altitude.

Although typical informatics tools for trajectories computation, like the one used in this study (LP-165 with a degree and order of 4 [28]), do not have a lunar gravitation model accurate enough to take mascons into consideration (degree 17 required).
It is, however, worth describing their effect for later steps following this study, where higher landing accuracy will be needed.

1.2 Spacecraft

1.2.1 Propulsion systems for lunar transfer, trajectory adjustments & lunar landing

A spacecraft aiming the moon requires significant propulsion systems for its TLI phase, trajectory adjustment and lunar landing. At the current state of investigation and technology, chemical thrusters are considered for propulsion.

As it has been said earlier, electric thrusters and their related trajectory types are not considered in this document because of their complexity and rather long transfer time.

1.2.1.1 Chemical propulsion

The chemical propulsion principle relies on combustion of fuel with oxidiser.
Among chemical thrusters, 2 types are considered: mono-propellant & bi-propellant.
Mono-propellant propulsion has the particularity that the oxidiser is already bound in the fuel molecule, resulting in a simpler, lighter and more reliable propulsion system, but presents lower thrust (0.1 to 100N) and lower efficiency than bi-propellant thrusters. Mono-propellant thrusters are generally used for attitude control.

Bi-propellant thrusters, traditionally used in rockets. Although their I_{sp} is much lower than electric propulsion systems (250 to 500 sec.), they can achieve high thrust (4N to > 400N) at specific manoeuvre points according to the adopted trajectory. Their propellants depend on the engine. They are generally N_2O_4/UDMH, N_2O_4/MMH, MON/MMH, LOX/LH2 or LOX/kerosene. The 2 last ones are generally used for launchers propulsion system.

Although their relative density is low, the high amount of propellant significantly increases the spacecraft mass. The price per litre [29] and bipropellant ratio of fuel to oxidiser are given in table 21 and 22 in the annexes.

1.2.1.2 Recommended propulsion system for GLXP

When analysing the performance of mono- and bipropellant thrusters, the propellant mass for a Hohmann transfer corresponding to each case are represented in table 5 and obtained on the base of equation 1.1 on page 39.

Table 5 : Monopropellant versus bi-propellant thrusters

Mono-propellant thruster	**Propellant mass**	3'094.68	*kg*
	dry mass	600.00	*kg*
	Isp (hydrazine)	220	*s*
	Gravity	9.81	*m/s²*
	Delta-v	3'923	*m/s*

Bi-propellant thruster	**Propellant mass**	1'379.61	*kg*
	Spacecraft mass	600.00	*kg*
	Isp (N2O4/MMH)	335	*s*
	Gravity	9.81	*m/s²*
	Delta-v	3'923	*m/s*

An I_{sp} difference of 1/3rd more than doubles the resulting propellant mass between both thruster types.
It is also to keep in mind that bipropellant thrusters require twice as much equipment as monopropellant, increasing cost, volume and complexity.
If on a first approach the bipropellant option appears cheaper regarding the propellant aspect, a further check is required in order to define if bipropellant complexity cost compensates the propellant savings in terms of design, equipment and volume.

Therefore, 2 cases are considered:

- bipropellant thrusting for propulsion transfer and monopropellant for attitude control and lunar landing
- monopropellant thrusting for transfer propulsion, attitude control and lunar landing

The first case was used in most of the past lunar missions, giving the advantage of a great experience.
The second case may provide a slight advantage in terms of design time, especially for the GLXP competition where delays are fixed.

Fuel mass estimation for lunar transfer, attitude control, descent and landing phases is given in section 1.2.2.2 .

1.2.2 Mass estimation

1.2.2.1 Spacecraft dry mass estimation

The conception of the spacecraft and its rover can be done in many ways. Since GLXP judging panel offers the freedom for rovers to perform their run on, below or above the lunar ground, many configurations are possible. Consequently the mass of the spacecraft, of the lander and the required propellant amount will be significantly different.
Therefore, the mass estimation carried out in this section is very approximate and follows guidelines.

A mass break down estimation of the spacecraft is needed and in which will be included the rover mass. Allocation guides and thumb rules can be found when investigating mass break down of former planetary missions as well as lunar missions in literature.
According to past missions [30], small scientific payload satellites and telecommunication satellites breakdown are analysed in table 23 of Annexes. In

order to estimate a mass distribution for GLXP spacecraft, each subsystem is commented and its value tuned resulting in a convenient model for lunar landing mission.
It is obtained that the payload represents between 8 to 14% of the dry mass of the spacecraft.
In the GLXP case, the payload is represented by the rover. Since the experience of the past planetary exploration missions involved mobile robots using wheels, it is assumed that, for mass estimation simplicity, the GLXP team will be similar.

A good example for the rover mass estimation is the Sojourner rover of the NASA Mars Pathfinder mission (1996). According to its payload (3 cameras, laser detection system, x-ray spectrometer, abrasion and adherence experiments, accelerometers and potentiometers), its mass is of about 10.5 kg.
Although it was a rover developed for Mars, where ambient conditions are different, the only difference with a lunar rover would come from the telecommunication subsystem and the payload. Concerning other subsystems, it is assumed that the overall complexity is the same. Some subsystem complexity balances other ones, such as thermal control and power generation, since the former is easier to achieve on Mars but problematic on the moon and inversely for the latter.
GLXP rover payload is composed at least of a high definition video camera and its uplink data transmitter. Therefore, it is reasonable to estimate that, as the lowest value, its mass is approximatively the same as Sojourner. Of course, since the rover can adopt many shape and include other payloads, a heavier variant is to be planned. Its maximum value is estimated at about 50 kg.
When combining best case and worst case payload percentage (8% and 14%) to minimum and maximum evaluated masses of the craft it is obtained the following spacecraft dry mass, represented in table 6.

Table 6 : Spacecraft dry mass estimation

Worst case	Unfavourable typical case	Favourable typical case	Best case
$\frac{50\cdot100}{8}=625\quad kg$	$\frac{50\cdot100}{14}=357\quad kg$	$\frac{10\cdot100}{8}=125\quad kg$	$\frac{10\cdot100}{14}=71\quad kg$

It can be seen that the difference between the worst case mass is of about 9 times the best case value.

1.2.2.2 Propellant & wet spacecraft mass estimation

According to values obtained in table 6, mass estimation of propellant is calculated following equation 1.1.

$$m_{fuel}=m_{dry}\cdot\left(e^{\frac{\Delta v}{I_{sp}\cdot g_o}}-1\right) \tag{1.1}$$

Results of calculation figuring in tables 24, 25, 26 and 27 of annexes are resumed in table 7.

Values are obtained according to the following parameters:

- Hohmann transfer from LEO (Δv = 3.923 km/s) and GTO (Δv = 1.7 km/s)
- Descent of Δv = 1.7 km/s
- Mono- and bipropellant thrusting and all-monopropellant thrusting variants
- N_2O_4/MMH as fuel combination for bipropellant thrusting (I_{sp} = 335 s)
- Hydrazine for monopropellant thrusting (I_{sp} = 220 s)
- Launch cost per kg is of 10000 € for LEO launches and 25000 € for GTO launches

Table 7 : Spacecraft mass and launching cost estimation for LEO and GTO transfer, for both mono- and bipropellant and all monopropellant thrusting variants.

LEO	Mono- and bipropellant thrusting			All-monopropellant thrusting		
	Propellant mass	Wet mass	Total cost (launch and propellant)	Propellant mass	Wet mass	Total cost (launch and propellant)
Best case	445 kg	516 kg	5198854 €	892 kg	963 kg	9630000 €
Favourable typical case	783 kg	908 kg	9146169 €	1571 kg	1696 kg	16960000 €
Unfavourable typical case	2236 kg	2593 kg	26116868 €	4487 kg	4844 kg	48440000 €
Worst case	3915 kg	4540 kg	45727120 €	7855 kg	8480 kg	84800000 €

GTO	Mono- and bipropellant thrusting			All-monopropellant thrusting		
	Propellant mass	Wet mass	Total cost (launch and propellant)	Propellant mass	Wet mass	Total cost (launch and propellant)
Best case	272 kg	262 kg	6569954 €	272 kg	343 kg	8575000 €
Favourable typical case	479 kg	461 kg	11560151 €	479 kg	600 kg	15000000 €
Unfavourable typical case	1368 kg	1316 kg	33000392 €	1368 kg	1725 kg	43125000 €
Worst case	2395 kg	2305 kg	57800734 €	2395 kg	3020 kg	75500000 €

Several observations can be done:

1. The launching cost is quantified in tens of millions of dollars.
2. Although the total cost varies from a GTO transfer to a LEO transfer, the cost difference is not significant since these values are estimations. Actually, the last remark is obvious since the main differences between both transfers are the orbital parameters. The distance from earth to moon and the trajectory type (Hohmann) stays the same as well as the transfer energy.
3. From the mono-and bipropellant variant to the all monopropellant variant, the total cost is almost doubled for a LEO transfer (factor 1.9) and multiplied by a factor 1.3 for a GTO transfer. This cost difference is smaller for a GTO transfer since its translunar Δv kick is much lower than for a LEO transfer, the lunar descent stays unchanged.
4. For a LEO transfer the propellant mass represents 93% of the wet spacecraft mass where for a GTO transfer it represents a 79%.

Taking into consideration that piggyback payload mass are generally between 50 up to 500 kg [31,32,33], the mono- and bipropellant variant has a significant weight and cost advantage over the all monopropellant variant since the difference is measured in millions of euro and, thus, appears as a good option within this study.

1.3 Possible transfer scenarios

Now that the most important parameters have been enumerated and described, some transfer scenarios can be imagined.
Considering the most realistic cases, 2 options are held up:

- transfer where the lunar landing site is defined (Luna17)
- transfer where the descent occurs directly after lunar capture.

Both options, independent of the type of launch (LEO or GTO), are described below.

1.3.1 Defined landing site

This case corresponds to how a classic lunar mission with lander would be defined.
A lunar landing site is considered appropriate in terms of manoeuvring and lunar ground state. Its proximity to a past lunar mission site offers the advantages of having existing data and the possibility of winning the Heritage bonus prize.

When launched, either in LEO or GTO, coast time and inclination change are required in order to transfer the spacecraft at the correct epoch for lunar rendez-vous and in a lunar inclination permitting to reach the landing site latitude.

Advantages:

- The landing site is known as well as its topography
- Possibility of winning Heritage bonus prize
- Computation of one single descent scheme but at various epochs

Drawbacks:

- Geocentric manoeuvres may be large according to initial earth parking orbit
- Coast time could be rather large according to lunar rendez-vous and lunar lighting conditions on the near-side

1.3.2 Direct descent after lunar capture

In this case, propellant savings are privileged.
According to the launching parameters (inclination, altitude, epoch) it is decided to choose a lunar landing inclination which minimises the spacecraft inclination change and land the spacecraft on this same inclination.

Advantages:

- Δv of geocentric inclination change is optimised
- Total transfer time is minimised → spacecraft can be directly landed right after its lunar capture

Drawbacks:

- Lunar site is unknown (no data of previous lunar mission)
- Multiple descent schemes have to be computed according to multiple possible landing site possibilities.

CHAPTER 2

2 TRAJECTORIES COMPUTATION

In this section are illustrated examples of the scenarios defined previously (1.3.1 & 1.3.2) at specific initial conditions and using the open source software GMAT (General Mission Analysis Tool) currently developed by NASA.
Added to these scenarios computations, a variation of lunar altitude where the spacecraft is injected is done between several parameters in order to minimise the overall Δv cost.

On beforehand, some first guesses are analytically calculated in order to be inserted as initial conditions in the numerical models as well as to have a comparison base for results interpretation.

2.1 Initial conditions data

In this section are given the main data used as initial conditions. Several first guesses values are calculated and parameters for lunar descent and landing are presented.
Δv in-plane orbit changes are calculated for:

- LEO to circular LLO
- GTO to circular LLO

Δv out-of-plane orbit changes are calculated for:

- Inclination change manoeuvre (geocentric)

2.1.1 LEO & GTO transfers

Data used for these analytical calculations is presented in table 8.

Table 8: Data used in GMAT models

Denomination	GMAT abbreviations	Value
LEO radius about the earth	RMAG	6555 km
LEO semi-major axis	SMA	6555 km
LEO eccentricity	ECC	~0
GTO radius at apogee	RadApo	42071 km
GTO radius at perigee	RadPer	6555 km
GTO eccentricity	ECC	0.72
GTO semi-major axis	SMA	24313 km
LLO altitide	-	100 km
Epoch	Epoch	26659.0000 (Julian calendar) → 01 January 2014 12.00h
Argument of perigee	AOP	0°
Right ascension of ascending node	RAAN	306°
Inclination	INC	6°
True anomaly	TA	54°

Results from first guess calculations figuring in annexes are resumed in table 9.

Table 9: Δv first guess results for Hohmann transfers

Type Of Transfer	Earth			Moon	Δv_1	Δv_2	Δv_{tot}
	r	r_{aE} (Periapsis)	r_{pE} (Apoapsis)	r_m			
LEO to circular LLO	6555 km	-	-	1837 km	3.14 km/s	0.68 km/s	3.82 km/s
GTO to circular LLO	-	42071 km	6555 km		0.68 km/s	0.68km/s	1.36 km/s

2.1.2 Inclination change manoeuvre

Results from first guess calculations figuring in the annexes are synthesised in table 10.

Table 10: Δv first guess results for inclination change

Type Of Orbit	LEO orbit radius	GTO Apoapsis	GTO Periapsis	V_{cp}	V_{ep}	Δv to 18.2°	Δv to 28.5°
LEO	6555 km	-	-	7.80 km/s	-	2.46 km/s	3.84 km/s
GTO	-	6555 km	42071 km	-	1.60 km/s	0.51 km/s	0.79 km/s

2.1.3 Descent and landing data

Data used for lunar landing are approximately the same as for Apollo missions and represented in table 11. It represents a general case which is to be refine to obtain higher accuracy models in later studies.

Table 11: Data for descent and landing

Denomination	Value
LLO altitude	100 km
De-orbiting altitude	15 km
Defined landing site coordinates	35°N 35° W (Luna 17)
Time of descent (15 km to touchdown)	1000 seconds

2.2 GMAT Models & Results

In this section are presented the models developed with GMAT for the trajectory analysis and on the base of the first guess calculated in the section above.
Resulting of the computation of these models, figures representing lunar trajectories as well as the tables resuming Δv values are described and discussed in the following sub-sections.

2.2.1 GMAT Models

All trajectories have been computed with the same initial conditions:

- orbital elements (except eccentricity for GTO orbits)
- epoch

The computing procedure for each trajectory is the following:

- Computation of right ascension of ascending node, *Ω,* argument of periapsis, *ω,* and inclination, *i*, in order to determine lunar injection conditions
- Computation until lunar sphere of influence at 64000 km of the moon
- Back propagation from previous lunar injection conditions until the targeted point on the lunar sphere of influence
- Positioning of the spacecraft around the moon before de-orbiting (with or without inclination change)
- De-orbiting from 100 km until 15 km
- Final landing from 15 km until touch down
- Propagation of 2 orbits at initial conditions

It resumes, in a more readable way, the source code processed by GMAT, which is available at the very end of this document in the annexes.

The first step is proceeded by 2 loops computing the inclination, right ascension of ascending node and argument of periapsis required for the Hohmann transfer to reach the moon. The first loop propagates the spacecraft to a point in the vicinity of the moon. The second loop refines the first one, by targeting a define lunar altitude and inclination (or altitude only) using the computed values of the first loop.
Then the mid-course manoeuvre is computed from both side, earth and moon, to meet on a point on the lunar sphere of influence.

The geocentric coast time is used, in this case, as waiting time for having adequate lunar rendez-vous positioning and lunar lighting conditions.

The duration of lunar coast depends on the landing site. It can be rather large, depending on the lighting condition since it is not wanted to land the spacecraft in a lunar night.
On the contrary, the coast time lasts several orbits, or less, in the case the spacecraft is directly landed after lunar capture.

2.2.2 Lunar transfer from LEO

LEO transfers with favourable right ascension of ascending node conditions are represented in figures 2.1 and 2.2, where in the first case the spacecraft is directly landed and in the second the spacecraft is landed on Luna 17 site.

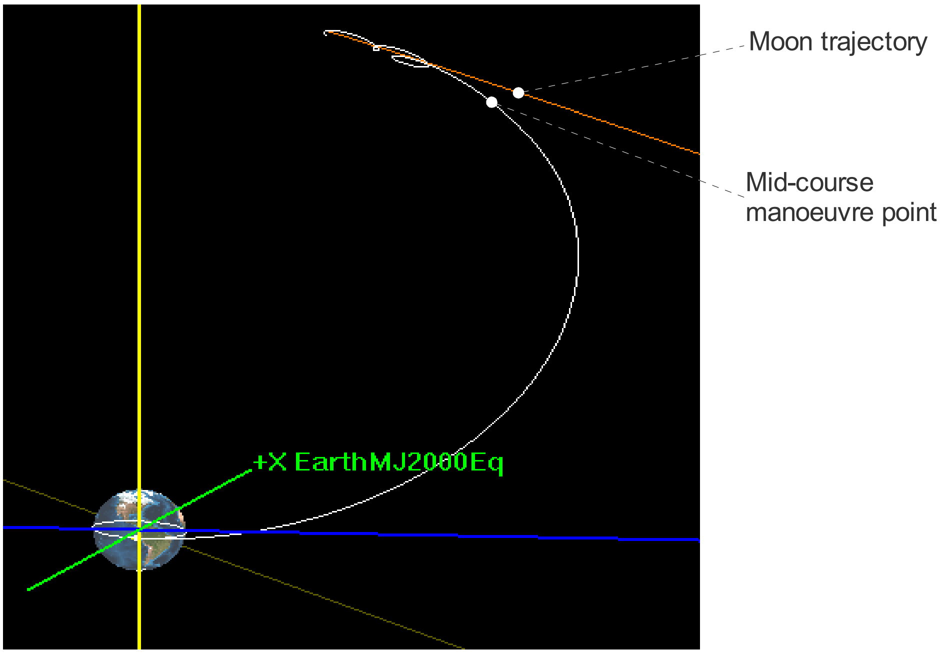

Figure 2.1 : Hohmann transfer from LEO with direct descent (earth inertial frame)

From an initial circular orbit of 6° of inclination, the spacecraft is transferred and its inclination is corrected at the mid-course point, situated on the moon's sphere of influence.
The loop spiralling along the moon's orbit represents the spacecraft orbiting in the earth inertial reference frame.
When landed on the Luna 17 site, the coast time about the moon is extended in order for the spacecraft to reach the landing site.

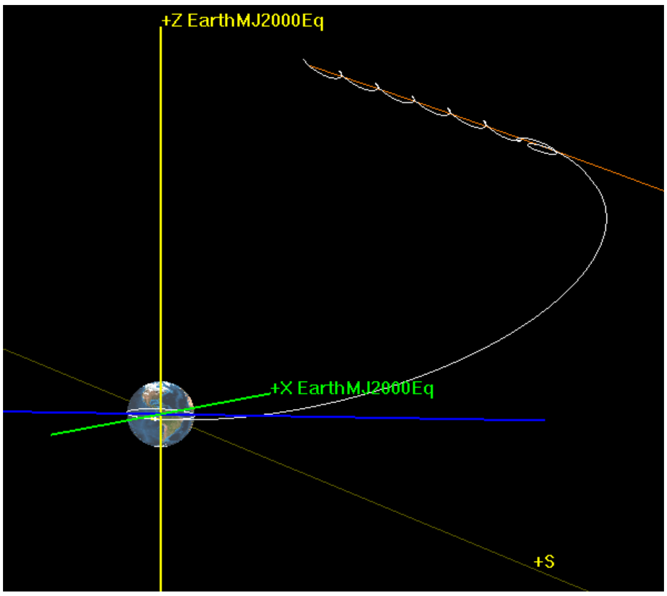

Figure 2.2 : Hohmann transfer from LEO, landed on Luna17 site (earth inertial frame)

2.2.3 Lunar transfer from GTO

In figure 2.3 and 2.4 are represented lunar transfer from a GTO orbit with favourable Ω, the right ascension of ascending node conditions. The initial inclination and epoch are the same as those used from a LEO orbit.

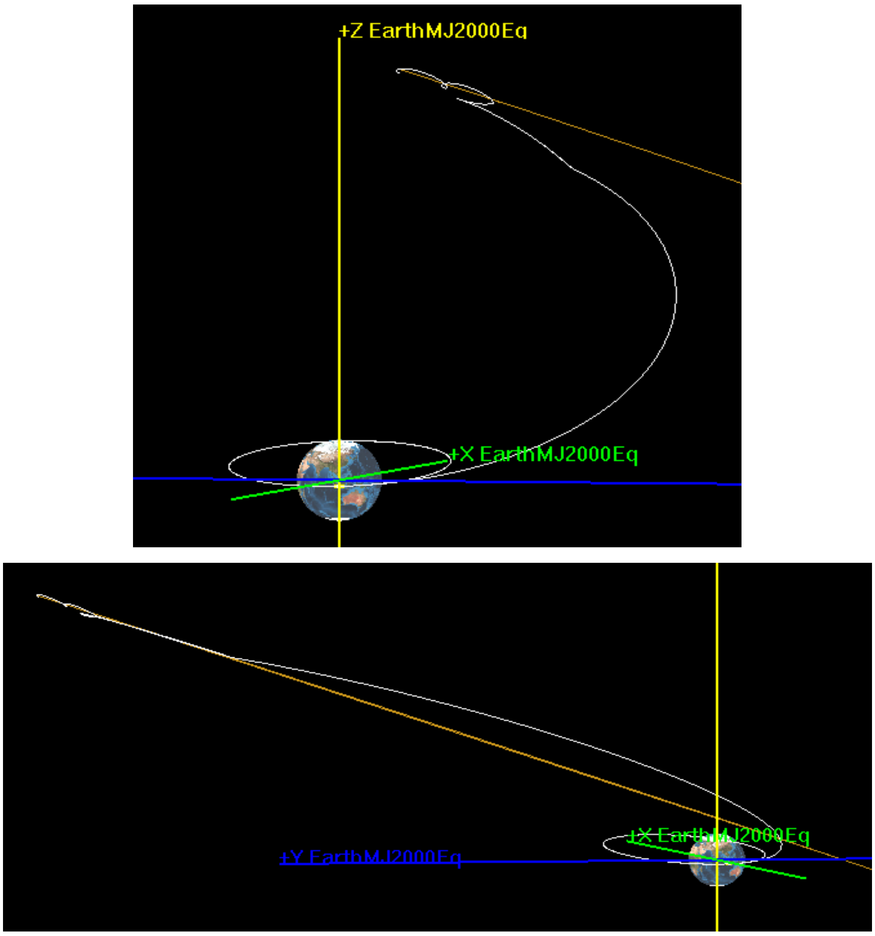

Figure 2.3 : Hohmann transfer from GTO with direct descent (earth inertial reference frame)

The bend of the mid-course manoeuvre is more clearly distinguishable as in the LEO transfers. Both representations in figure 2.3 permit to have a better overview of the trajectory and manoeuvre.
As it has been remarked in the LEO transfers, the spacecraft requires additional orbits about the moon in order to reach the Luna 17 site.

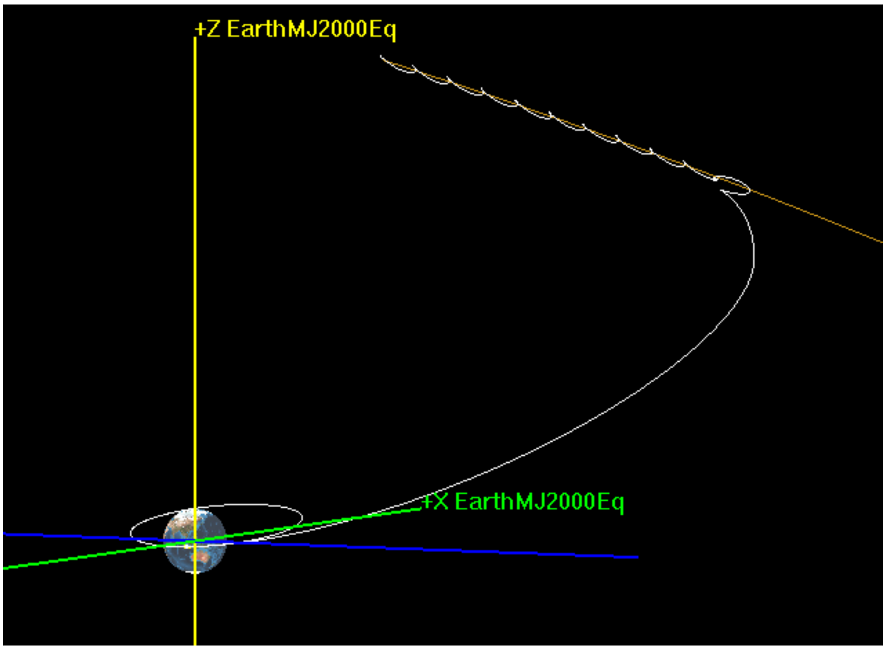

Figure 2.4 : Hohmann transfer from GTO, landed on Luna17 site (earth inertial frame)

2.2.4 Lunar descent & landing

Both landing schemes, direct descent and near Luna 17, are shown in figures 2.5 and 2.6 in a closer representations.

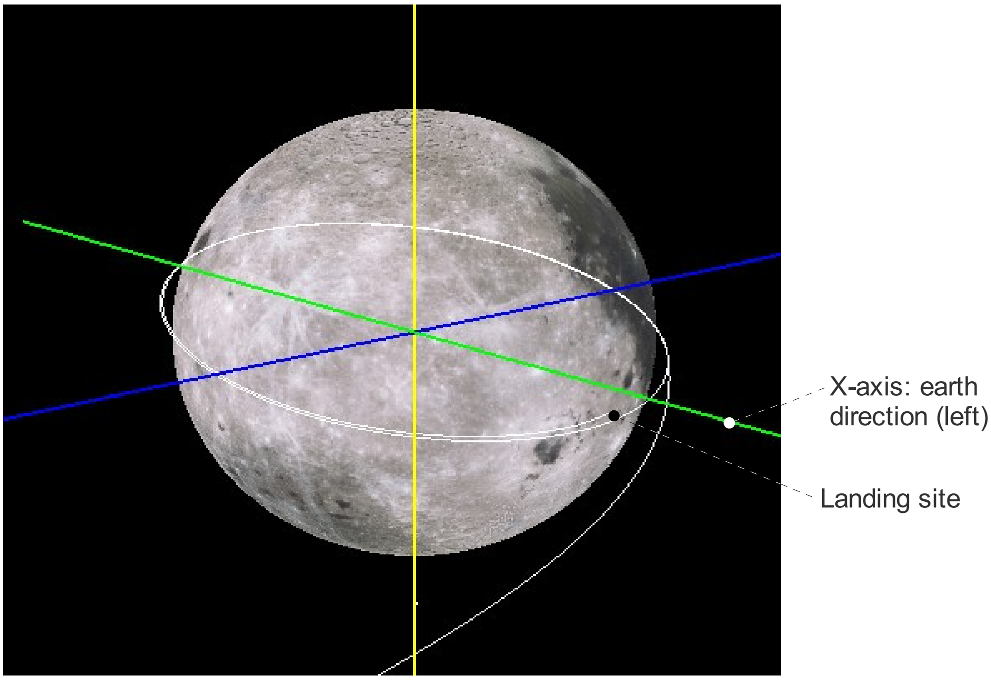

Figure 2.5 : Direct descent after lunar capture

The plane change to reach the minimum inclination of 35° is clearly seen in figure 2.6.

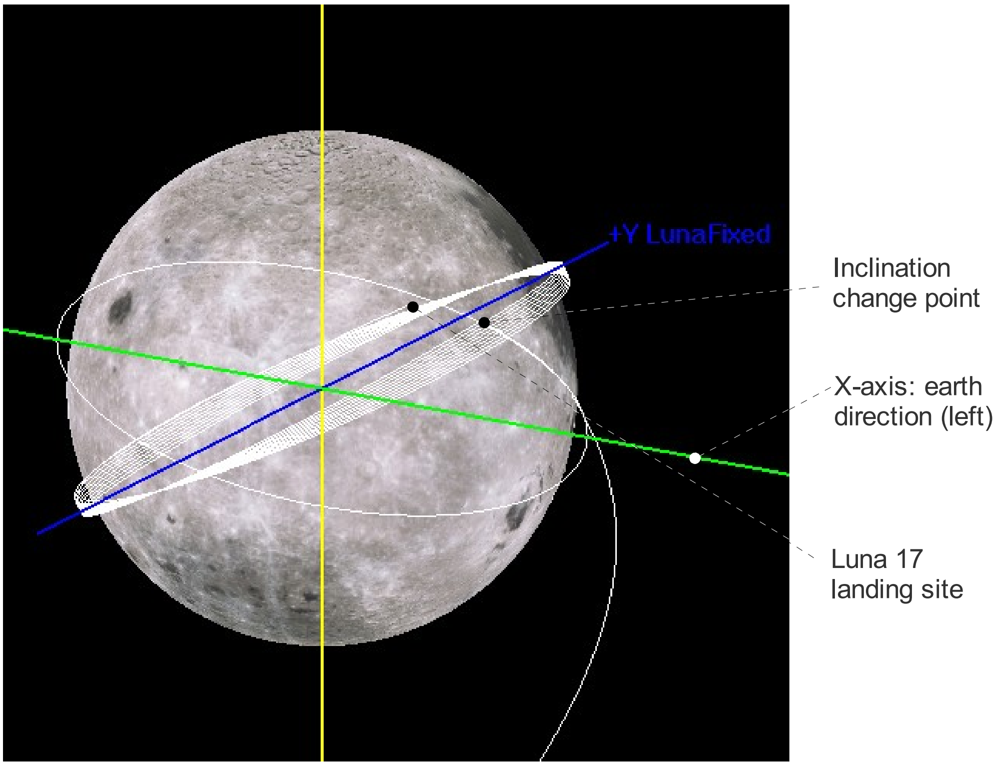

Figure 2.6 : Large inclination change for the landing on Luna 17 site

As described earlier, the lunar descent and landing, which sequence of events is represented in figure 2.7 , is based on Apollo missions landings.
Just after lunar capture (1), the spacecraft orbits the moon for more than a revolution until 180° of longitude (2) where its perigee is lowered to 15 km altitude (3) at 0° longitude.

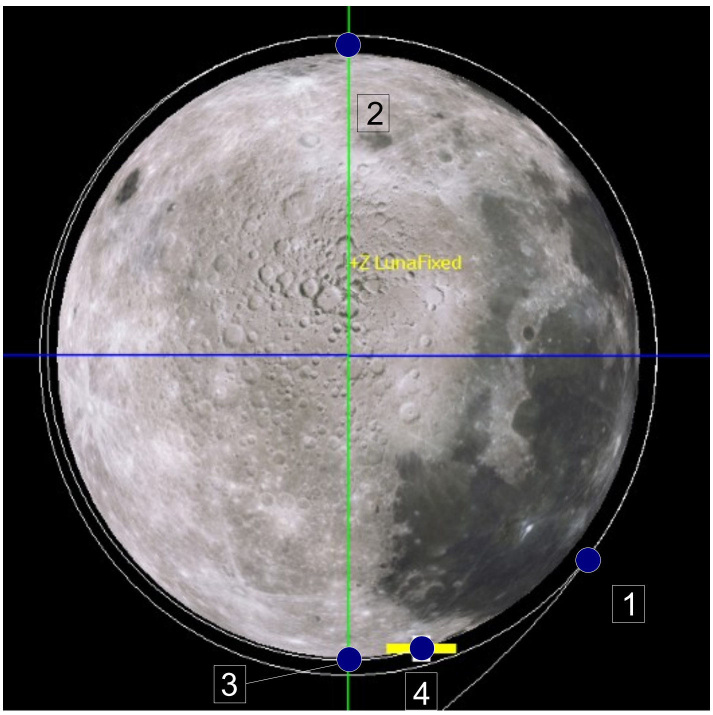

Figure 2.7 : Lunar landing sequence of events (lunar fixed frame)

Since only the Δv values and basic trajectory shapes are required in this study, the detailed descent portion presented in figure 1.12, has been simplified into a single arc between 0° longitude and landing site (points 3 to 4 in the figure).
In the case where the spacecraft is landed near Luna 17 site, the orbiting between lunar capture and perigee lowering (1 to 2) is extended until the spacecraft flies over the landing site.

2.2.5 Trajectory data analysis

Results analysis
Results of model computations are presented in table 12. According to both kind of initial orbits, LEO and GTO, as well as landing sites, direct descent after lunar capture or Luna 17 site, the Δv values of each event, the total Δv and transfer time are given. Sticking to Apollo missions design, the lunar capture altitude is of 100 km.

Table 12 : Detailed Δv and transfer time for LEO and GTO transfer

Launching orbit	Descent Type	Lunar capture altitude [km]	Total transfer time [days]	Total delta-V [km/sec]	Delta-V translunar kick [km/sec]	Delta-V mid-course manoeuvres [km/sec]	Delta-V lunar injection kick [km/sec]	Delta-V deorbiting until 15km lunar altitude [km/sec]	Delta-V from 15km until touchdown [km/sec]
LEO	Luna 17, lunar inc. = 35°	100	5.35	7.132	3.132	0.065	2.120	0.039	1.776
	Direct descent		4.95	5.856	3.132	0.065	0.820	0.040	1.799
GTO	Luna 17, lunar inc. = 35°		6.11	4.891	0.677	0.237	2.162	0.040	1.776
	Direct descent		5.38	3.574	0.677	0.237	0.820	0.040	1.801

A first glance at the total Δv values shows, as expected, lower costs for GTO transfers than for LEO transfers, which difference of about 1.3 km/s for direct descent and of about 2.2 km/s for descent near Luna 17 results from a greater translunar kick although transfer from GTO have a higher mid-course Δv cost. Between a direct landing and landing near Luna 17 site, the difference is of about 1.3 km/s for both LEO and GTO transfers which corresponds to the lunar inclination change taken into account in the lunar injection kick column.
Although the transfer time are all approximately between 5 and 6 days, shorter transfers of about a half day are obtained with LEO transfers.
Values for de-orbiting and landing until touch down do not vary significantly from one case to another.

Variation of lunar capture altitude
When varying the altitude at lunar capture, which influence on direct transfers is resumed in table 13, the impact on the total Δv is rather small. The difference is noticeable at the second decimal. The de-orbiting Δv cost decreases with the decreasing lunar altitude while lunar injection kick Δv increases.
Similar differences are obtained with trajectories landed on Luna17 site.

Table 13 : Impact of lunar capture altitude.

Launching orbit	lunar capture altitude [km]	Total transfer time [days]	Total delta-V [km/sec]	Delta-V translunar kick [km/sec]	Delta-V mid-course manoeuvres [km/sec]	Delta-V lunar injection kick [km/sec]	Delta-V deorbiting until 15km lunar altitude [km/sec]	Delta-V from 15km until touchdown [km/sec]
LEO Direct descent	30	4.9	5.845	3.132	0.066	0.831	0.008	1.808
	65		5.857			0.826	0.024	1.810
	100		5.856			0.820	0.040	1.799
GTO Direct descent	30	5.4	3.569	0.677	0.241	0.831	0.008	1.810
	65		3.577			0.825	0.024	1.810
	100		3.574			0.820	0.040	1.801

For further study, an improved model with a descent optimisation loop and topographic map, permitting closer zooming, should be used in order to obtain a more accurate Δv and trajectory shape. This model would have to consider additional Δv costs due to gravity and thrust vector losses[34], which should increase the total de-orbiting and descent Δv to about 2.3 km/s.

Lighting conditions at arrival

When analysing the lighting conditions at arrival of both landing sites, the sun line orientation (line passing through the earth) represented in the figures of 2.8 indicates that the spacecraft is landed at lunar sunrise.

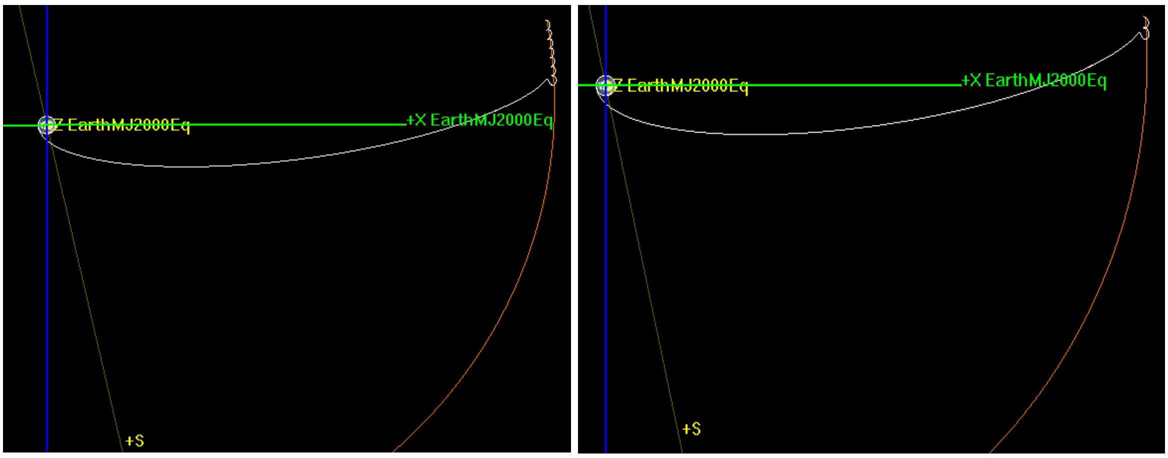

Figure 2.8 : sun's direction on Luna17 (left) and at direct descent (right) (earth inertial frame)

The $+S$ indication near the sun line in the figures indicates the sun direction with respect to the earth.

Transfers with unfavourable right ascension of ascending node at initial conditions

As it has been remarked previously, cases characterised by unfavourable conditions of right ascension of ascending node at launch (29° ≤ ΔΩ ≤ 134° and 207° ≤ ΔΩ ≤ 315°) requires positioning manoeuvres increasing significantly the overall trajectory Δv.

A LEO and a GTO transfer model with the aforementioned conditions have been done in order quantify the Δv difference with the trajectories with mid-course correction already analysed.

Their computing procedure, in figure 2.9, are slightly more complicated than for mid-course correction trajectories.

LEO transfer computing procedure

a) Geocentric coast on a circular orbit

b) Research of i, Ω and ω targeting a point near the Moon (2-body propagation: Earth and spacecraft)
c) Research of i, Ω and ω , on the base of the previous computed values, targeting a lunar altitude and inclination or altitude only (3-body propagation: Earth, Moon and Spacecraft)

d) Back propagations in time, from epoch just before lunar transfer to initial conditions in order to perform to i and Ω manoeuvres:
- equatorialization of orbit to reach initial Ω values
- coast of defined time to obtain defined Ω at next manoeuvre
- inclination kick at perigee to obtain the initial i

e) Lunar capture kick
f) Selenocentric coast
g) Perigee lowering from LLO to 15km altitude
h) Final descent from 15 km to touch down

GTO transfer computing procedure

a) GTO orbit
b) Phasing loops on a lower eccentricity orbit
c) circularization of orbit at apogee

d) Research of i, Ω and ω targeting a point near the Moon (2-body propagation: Earth and spacecraft)
e) Research of i, Ω and ω , on the base of the previous computed values, targeting a lunar altitude and inclination or altitude only (3-body propagation: Earth, Moon and Spacecraft)

f) Back propagations in time, from epoch just before lunar transfer to initial conditions in order to perform to i, Ω and ω manoeuvres:
- equatorialization of orbit to reach initial ω and Ω values
- coast of defined time to obtain defined ω at next manoeuvre
- inclination kick at perigee to obtain the initial I
- *coast of defined time to obtain* defined Ω at next manoeuvre
- insertion on an elliptic orbit (perigee lowering) to perform phasing loops
- perigee kick to obtain required initial GTO orbit

g) Lunar capture kick
h) Selenocentric coast
i) Perigee lowering from LLO to 15km altitude
j) Final descent from 15 km to touch down

Figure 2.9 : Computing procedure of LEO and GTO transfer with unfavourable Ω conditions

The details of their manoeuvring before transflunar kick are represented in figures 2.10 and 2.11.
In the LEO case, a first kick is performed from its initial orbit in order to put it in the equatorial plane (1). The spacecraft has its inclination changed again after a defined cost time permitting to reach the translunar kick point at the right Ω value (2). The burn is performed when the spacecraft reaches the translunar kick point (3).

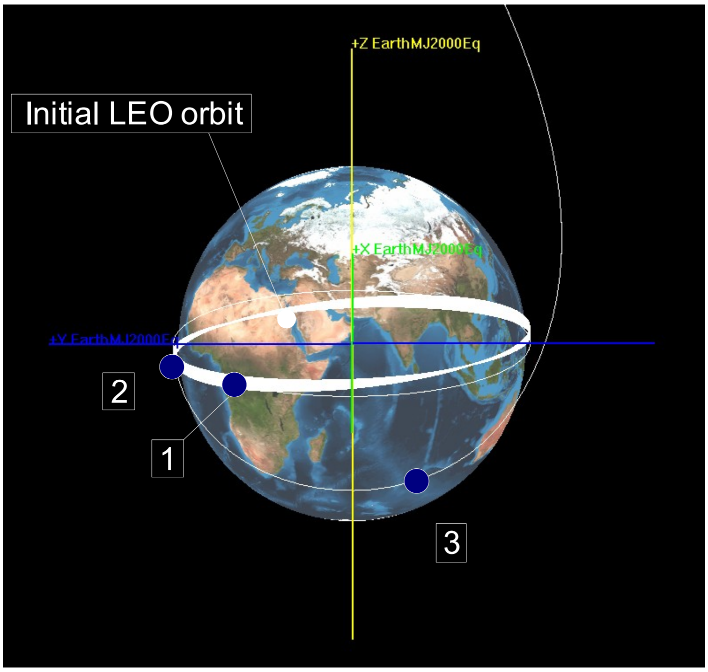

Figure 2.10 : Unfavourable Ω; manoeuvres on LEO before lunar transfer (earth inertial frame)

From an initial GTO orbit, it is performed an apogee lowering (1) which permits to decrease the orbit period and, thus, meet the Ω position in the available time interval. After a coast time, the orbit is circularized at apogee (2) and then equatorialized (3). A second coast is performed in order to reach the ω position required at translunar kick. It follows a last inclination change kick (4) in order to meet all positioning conditions for translunar kick (5).

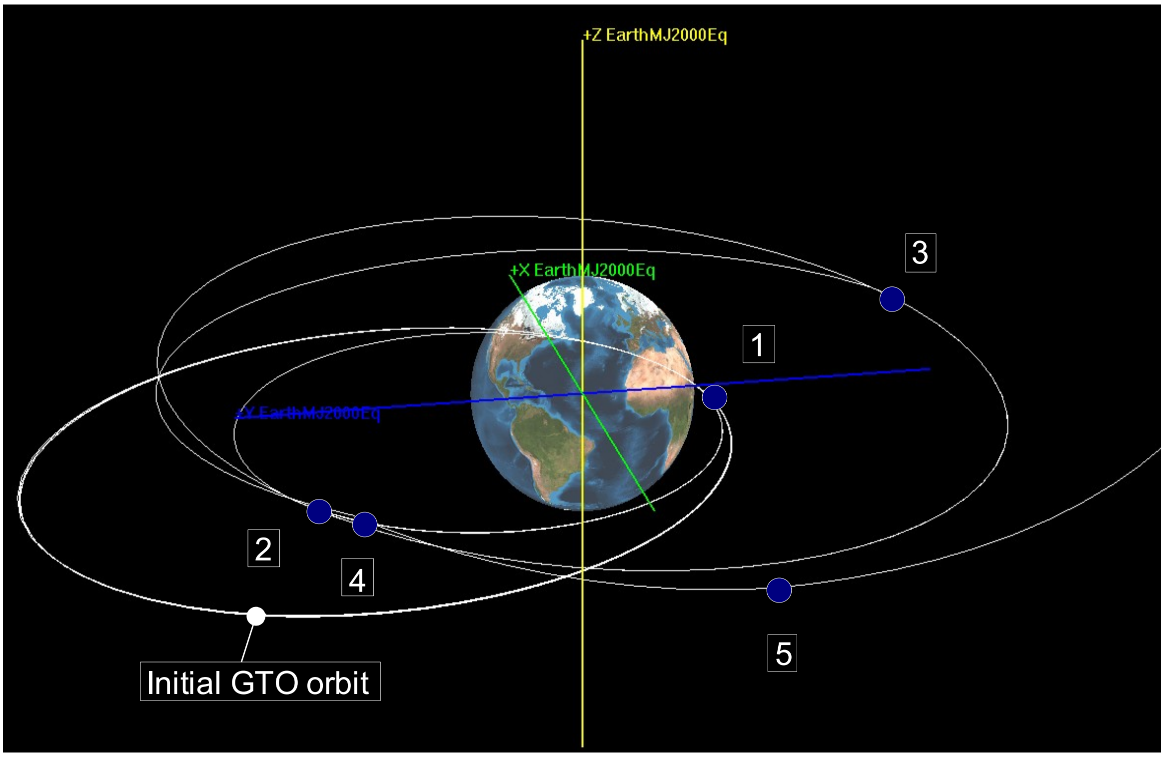

Figure 2.11 : Unfavourable Ω; manoeuvres on GTO before lunar transfer (Earth inertial frame)

The resulting Δv costs are represented in table 14.

Table 14 : Detailed Δv and transfer time for LEO and GTO transfer

Launching orbit	Lunar capture altitude [km]	Total transfer time [days]	ΔΩ [°]	Total delta-V [km/sec]	Delta-V positioning manoeuvres [km/sec]	Delta-V translunar kick [km/sec]	Delta-V lunar injection kick [km/sec]	Delta-V deorbiting until 15km lunar altitude [km/sec]
LEO, direct descent	100	5.92	41	9.699	4.029	3.135	0.811	0.040
GTO, direct descent		7.07	33	7.214	3.186	1.532	0.772	0.040

As expected, the positioning manoeuvres increase dramatically the overall Δv costs, from 5.9 to 9.7 km/s for the LEO transfer and from 3.6 to 7.2 km/s for the GTO transfer. The positioning manoeuvring Δv, of about 4 and 3 km/s, are greater than their translunar kick values.

Although the trajectories with mid-course correction requires more restricted launch conditions, their Δv savings are, however, significantly smaller and thus fully compensates the coast time required to meet their specific conditions.

3 SYNTHESIS

Many parameters and several scenarios have been analysed in this study. Best cases and worst cases have been defined in order to give a general direction to newly registered teams in the Google Lunar X-Prize competition.
These parameters are synthesised below to provide a quick reminder of the different options and choices made in this document.

It is known that a piggy-back launch is the most easiest way to launch the spacecraft into orbit, although their maximum payload mass is limited. The launch cost per kilogram depends on the launching organism wherever it is situated in the world. Although there are 2 possible initial orbit types, LEO and GTO, the overall cost would be approximatively the same.

Propulsion systems have been compared, for which a combination of bi-propellant system, for translunar kick, and mono-propellant system, for lunar landing, has been retained because of its lower Δv cost and propellant amount

The Hohmann transfer, compared to the bi-elliptic, tri-elliptic and WSB transfers, has an optimal complexity and transfer time combination, although its transfer energy, quantified by the velocity difference, Δv, is almost the same as the 3 other options.

Manoeuvres before lunar injection greatly depend on the Keplerian elements of the initial orbit.
The launching inclination should ideally be within the moon inclination interval. If it is the case, a transfer is almost manoeuvre-free, or has a small mid-course manoeuvre, if the spacecraft is launched with optimal conditions of:

- argument of perigee and right ascension of ascending node, for GTO transfers
- right ascension of ascending node only for LEO transfers

Some rare GTO Ariane 5 launches provides such conditions.
None-optimal conditions would require expensive manoeuvring which may double the trajectory total Δv cost.

The spacecraft will adopt a lunar landing procedure similar to those of the Apollo missions.
The spacecraft has the possibility to be landed directly after lunar capture or on a defined landing site, chosen in this study nearby the Luna 17 site and requiring consequently another lunar inclination change.

Mass breakdowns have been defined in order to evaluated the dry spacecraft and propellant masses and their impact on launch cost, according to both GTO and LEO launches. The total launch cost will most probably be situated between 10 and 20 millions of euros.

Finally, trajectories have been computed in order to illustrate the different scenarios. Hohmann transfer with a mid-course manoeuvre appear as the most cost and time effective options and refined numerical models should be develop in order to increase the trajectories accuracy.

4 CONCLUSION

At the end of this study, one has a better overview of the possibilities to develop a reduced cost and short delay landed lunar mission.

Although such situation involve serious restrictions, decision making is more direct concerning other aspects and the attention can be redirect on other critical factors.

The launching phase has been simplified to the research of a piggy-back launch proposed by a space agency or a private company. However, the implication of the launching parameters on the next transfer phases are very critical. A launch contract should, therefore, be defined as early as possible in order to set main constraints. These constraints are the mission schedule and its impact on the project time distribution, the hardware more specifically concerning the launcher slot adapter design and, the most important, the initial orbit conditions of epoch and orbital elements.
The definition of the launch conditions is, therefore, to be defined as a top-level priority.

Other suggestions of further steps for later state of study can be outlined.
From the proposed spacecraft mass break-down, a basic list of hardware and their requirements can be defined in which several solutions are compared in terms of performance and cost. The same should be done for the rover, for which the transportation mode on the lunar ground may have a great impact on its final mass.
Finally, more accurate trajectory models have to be developed. A more precise lunar gravity model is to be used and optimisation sequences are necessary in order to compute accurate lunar landing. Although GMAT is an appropriate tool for an early state study, as the one developed in the current document, higher details of graphics and zoom possibilities are necessary to improve the lunar landing.

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ANNEXES 1: LAUNCHING PHASE

Table 15 : Small launchers (payload mass until ~2,300 kg) [6]

Launcher	Athena 2	Cosmos	Pegasus XL	Rockot	Shtil	Start	Taurus	Falcon 1[4]
Country	USA	Russia	USA	Russia	Russia	Russia	USA	USA
LEO Capacity [kg]	2,065	1,500	443	1,850	430	632	1,380	570
GTO Capacity [kg]	590	0	0	0	0	0	448	0
Reference Site & Inclination	Cap Canaveral (USA), 28.5°	Plesetsk (Russia), 62.7°	Cap Canaveral, USA, 28.5°	Plesetsk (Russia), 62.7°	Barents Sea, 77-88°	Svobodny (Russia, closed), 51.8°	Cap Canaveral, USA, 28.5°	Vandenberg & Omelek Island, USA
Estimated Launch Price [5]	29,000,000 € (~$ 24,000,000)	15,800,000 € (~$ 13,000,000)	16,400,000 € (~ $ 13,500,000)	16,400,000 € (~ $ 13,500,000)	242,000 € (~$ 200,000 +Navy past participation)	9,100,000 € (~$ 7,500,000)	23,000,000 € (~$ 19,000,000)	4,500,000 € [6] (~$ 6,700,000)
Estimated LEO Cost per Kg	14,000 € (~$ 11,622)	10,500 € (~$ 8,667)	37,000€ (~$ 30,474)	8,800 € (~$ 7,292)	563 € (~$ 465 +Navy past participation)	14,000 € (~$ 11,687)	17,000 € (~$ 13,768)	7,900 € (~$ 11,745)
Estimated GTO Cost per Kg	49,300 € (~$ 40,678)	N/A	N/A	N/A	N/A	N/A	51,400 € (~$ 42,411)	N/A

[4] *Privately funded launcher*
[5] *In year 2000, 1 € = $ 0.8252*
[6] *Today's rate of exchange of about $1 = 0.67 €*

Table 16 : Medium & intermediate launchers (payload mass from 2,300 to 11,300 kg) [6]

Launcher	Ariane 44L (retired)	Atlas 2AS	Delta 2 (7920/5)	Dnepr	Long March 2C	Long March 2E	Soyuz	Falcon 9 [7]
Country	Europe	USA	USA	Russia	China	China	Russia	USA
LEO Capacity [kg]	10,200	8,618	5,144	4,400	3,200	9,200	7,000	10,450
GTO Capacity [kg]	4,790	3,719	1,800	0	1,000	3,370	1,350	4,540
Reference Site & Inclination	Kourou (French Guyana), 5.2°	Cap Canaveral, USA, 28.5°	Cap Canaveral, USA, 28.5°	Baikonur (Kazakhstan), 46.1°	Taiyuan (China), 37.8°	Taiyuan (China), 37.8°	Baikonur (Kazakhstan), 46.1°	Cape Canaveral (USA, from 2011), 28.5°
Estimated Launch Price [8]	136,000,000 € (~$ 112,500,000)	118,000,000 € (~$ 97,500,000)	66,650,000 € (~$ 55,000,000)	18,200,000 € (~$ 15,000,000)	27,300,000 € (~$ 22,500,000)	60,600,000 € (~$ 50,000,000)	45,500,000 € (~$ 37,500,000)	29,500,000 € [9] (~$ 44,000,000)
Estimated LEO Cost per Kg	13,300 € (~$ 11,029)	13,700 € (~$ 11,314)	13,000 € (~$ 10,692)	4,000 € (~$ 3,409)	8,500 € (~$ 7,031)	6,600 € ($ 5,435)	6,500 € (~$ 5357)	2,821 € (~$ 4,210)
Estimated GTO Cost per Kg	28,500 € (~$ 23,486)	31,800 € (~$ 26,217)	37,000 € (~$ 30,556)	N/A	27,300 € (~$ 22,500)	18,000 € (~$ 14,837)	33,700 € (~$ 27,778)	6,493 € (~$ 9,692)

Table 17 : Large launchers (payload mass of 11,300kg and more) [6]

Launcher	Ariane 5G	Long March 3B	Proton	Zenit 2	Zenit 3SL
Country	Europe	China	Russia	Ukraine	Multinational
LEO Capacity [kg]	18,000	13,600	19,700	13,740	15,876
GTO Capacity [kg]	6,800	5,200	4,630	0	5,250
Reference Site & Inclination	Kourou (French Guiana), 5.2°	Xichang (China), 28.5°	Baikonur (Kazakhstan), 46.1°	Baikonur (Kazakhstan), 46.1°	Odyssey Launch Platform (Pacific Ocean), 0°
Estimated Launch Price [8]	200,000,000 € (~$ 165,000,000)	72,700,000 € (~$ 60,000,000)	103,000,000 € (~$ 85,000,000)	51,500,000 € (~$ 42500,000)	103,000,000 € (~$ 85,000,000)
Estimated LEO Cost per Kg	11,100 € (~$ 9,167)	5,350 € (~$ 4,412)	5,200 € (~$ 4,302)	3,750 € (~$ 3,093)	6,500 € (~$ 5,354)
Estimated GTO Cost per Kg	29,400 € (~$ 24,265)	14,000 € (~$ 11,538)	22,200 € (~$ 18,359)	N/A	19,600 € (~$ 16,190)

[7] *Privately funded launcher*
[8] *In year 2000, 1 € = $ 0.8252*
[9] *Today's rate of exchange of about $1 = 0.67 €*

Table 18 : LEO and over LEO orbit launch distribution [7]

Launcher		**2008**					**2009**			
		Total launches	LEO launches	Over LEO launches	Deep space		Total launches	LEO launches	Over LEO launches	Deep space
Chang Zheng		11	7	4	-		-	-	-	-
R-7		10	8	2	-		13	12	1	-
Proton		10	-	10	-		10	-	10	-
Ariane 5		6	1	5	-		7	1	6	-
Zenit		6		6	-		4	--	4	-
CZ		-	-	-	-		6	4	2	-
Delta 2		5	4	1	-		8	5	2	1
Delta 4		-	-	-	-		3	-	3	-
STS		4	4	-	-		5	5	-	-
PSLV		3	2	1	-		2	2	-	-
KSLV-1		-	-	-	-		1	1	-	-
Kosmos 3M		3	3	-	-		1	1	-	-
Atlas 5		2	-	2	-		5	1	4	-
Dnepr		2	2	-	-		1	1	-	-
Pegasus XL		2	2	-	-		-	-	-	-
Falcon 1		2	2	-	-		1	1	-	-
H-2A		1	-	1	-		2	2	-	-
H-2B		-	-	-	-		1	1	-	-
Rokot/Briz KM		1	1	-	-		3	3	-	-
Tsyklon 3		-	-	-	-		1	1	-	-
Minotaur 1		-	-	-	-		1	1	-	-
Safir		-	-	-	-		1	1	-	-
Taurus-XL		-	-	-	-		1	1	-	-
Unha 2		-	-	-	-		1	1	-	-
Total		**68**	**36**	**32**	**-**		**78**	**45**	**32**	**1**

Table 19 : Launching sites around the world [8]

N°	Site Name	Country	Latitude	Orbital Inclination (min-max)	Launchers	Commercial Availability
1	Vandenberg	California, USA	34 ° N	51° - 145°	Delta II, Falcon 1, Falcon 9 (from 2011)	Yes
2	Edwards	California, USA	34° N	-	Military basis, Space Shuttle landing site	No
3	Wallops Island	Virginia, USA	37° N	-	Sounding rockets for suborbital flights	No
4	Cape Canaveral	Florida, USA	28° N	28° - 57°	Delta II, Delta IV, Atlas V, SpaceX Falcon9	Yes
N/A	Omelek Island	Republic of the Marshall Islands, USA control	9° N	-	Falcon 1, Falcon 9 (from 2011)	Yes
5	Kourou	French Guiana	5° N	5° - 100°	Vega, Ariane 5, Soyuz-2	Yes
6	Alcântara	Brazil	2° S	2° - 100°	VLS-1, sounding rockets	No
7	Hammaguir	Algeria	30° N	-	Sounding rockets	-
8	Torrejón Air Base	Spain	-	-	Military base	No
9	Andoya	Norway	69° N	-	Sounding rockets for suborbital flights	-
10	Plesetsk	Russia	62° N	62° - 83°	Soyuz, Cosmos-3M, Rockot, Tsyklon	Yes
11	Kapustin Yar	Russia	48° N	48° - 51°	Military missiles	No
12	Palmachim	Isreal	31° N	142° - 144°	Shavit	No
13	San Marco platform	Kenya (operated by Italy)	2.9° S	-	No more available	-
14	Baikonur	Kazakhstan (operated by Russia)	46° N	49° - 99°	Soyuz(2,-U,-FG), Zenit (-2M & -3M), Proton-M, Dnepr-1	Yes
15	Sriharikota	India	13° N	0 – 140°	PSLV, GSLV (Polar-, Geostationary Satellite Launch Vehicle)	Yes
16	Jiuquan	China	41° N	55° - 76° (56° - 40°)	Long March 2C (CZ-2C), Long March 2D (CZ-2D), Long March 2F (CZ-2F))	No
17	Xichang	China	28° N	28° - 36°	Long March 3B (CZ-3B), Long March 3A (CZ-3A)	No
18	Taiyuan	China	38° N	99°	Long March (CZ-2C/SD, CZ-4A, CZ-4B, CZ-4C)	No
19	Svobodny	Russia	51° N	51° - 110°	(Start-1)	Closed
20	Uchinoura	Japan	31° N	29° - 75°	M-V	Momentarily inactive
21	Tanegashima	Japan	30° N	99°	H-IIA	yes
22	Woomera	Australia	30° S	82° - 84°	-	Not used

Graphic 5.1 :Moon declination angle variation from 2011 to 2014

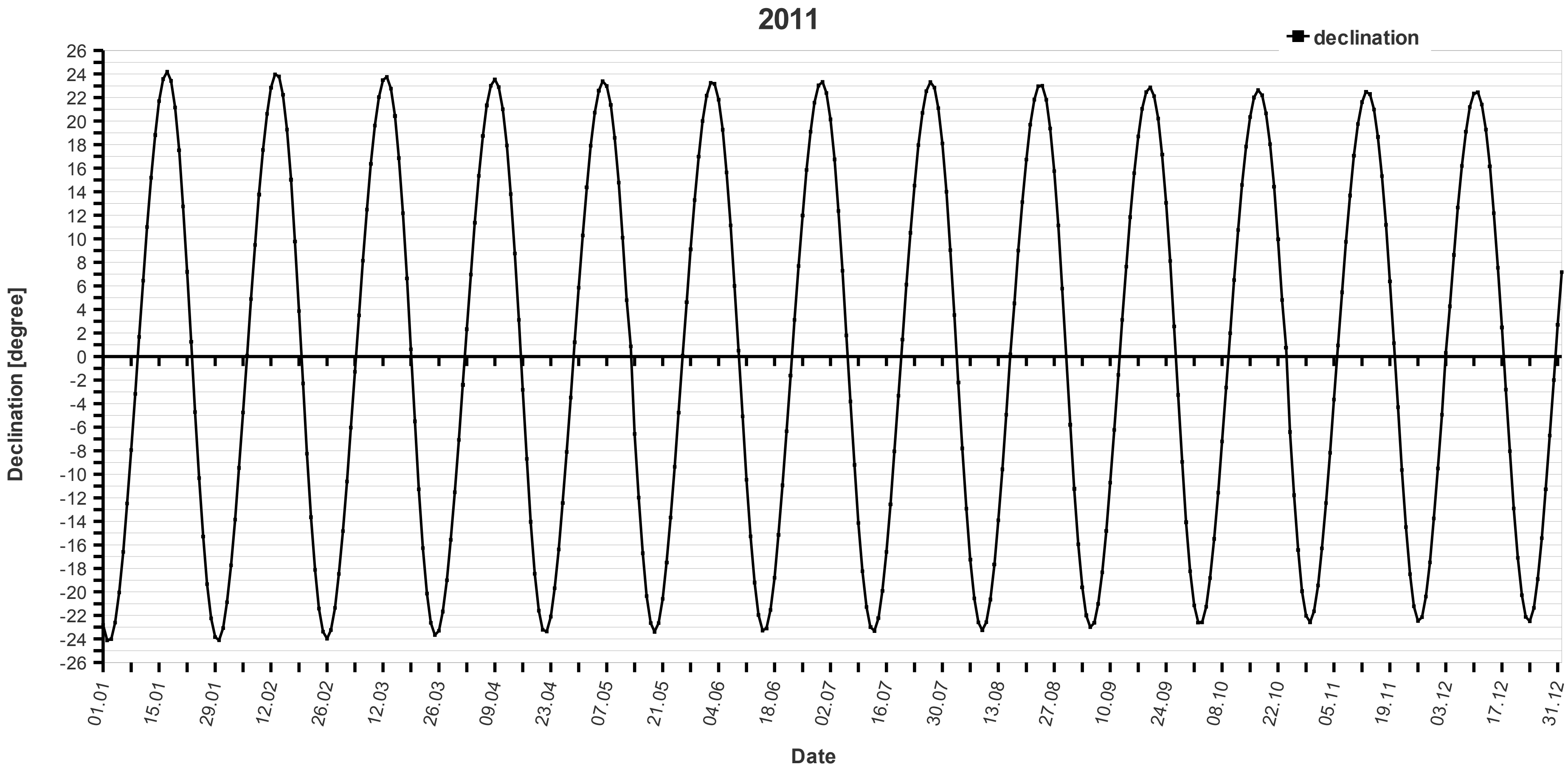

Figure 6.1 Moon Declination 2011

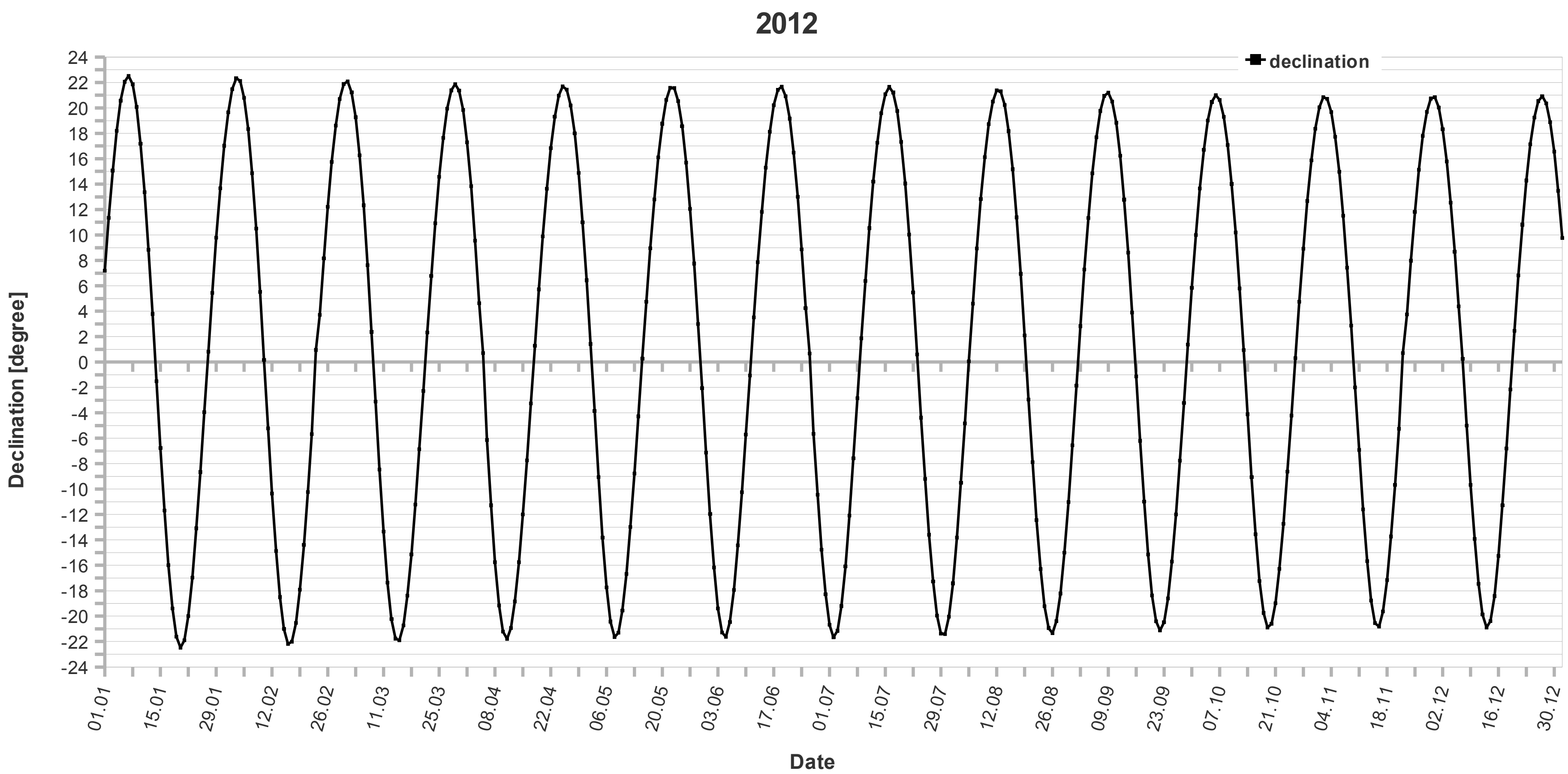

Figure 6.2 Moon Declination 2012

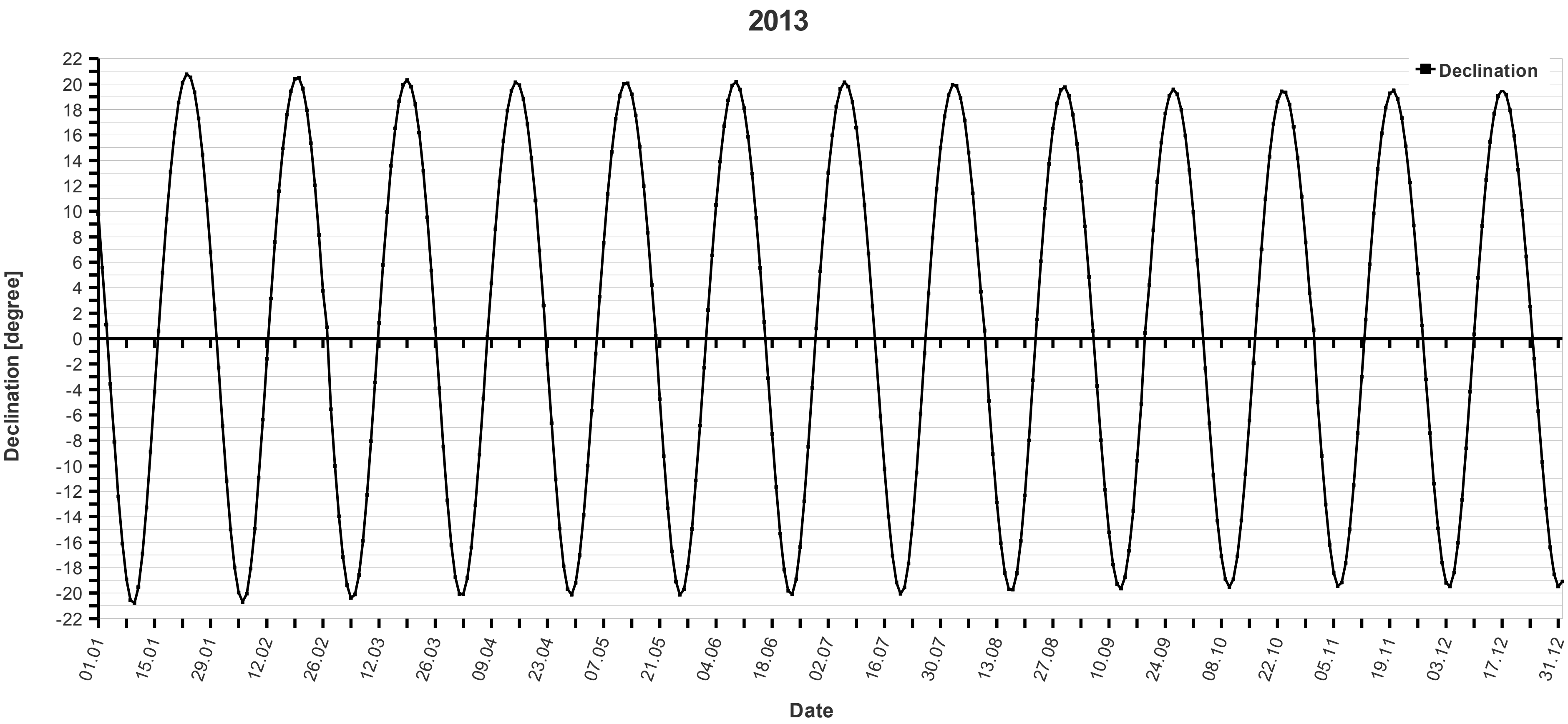

Figure 6.3 Moon Declination 2013

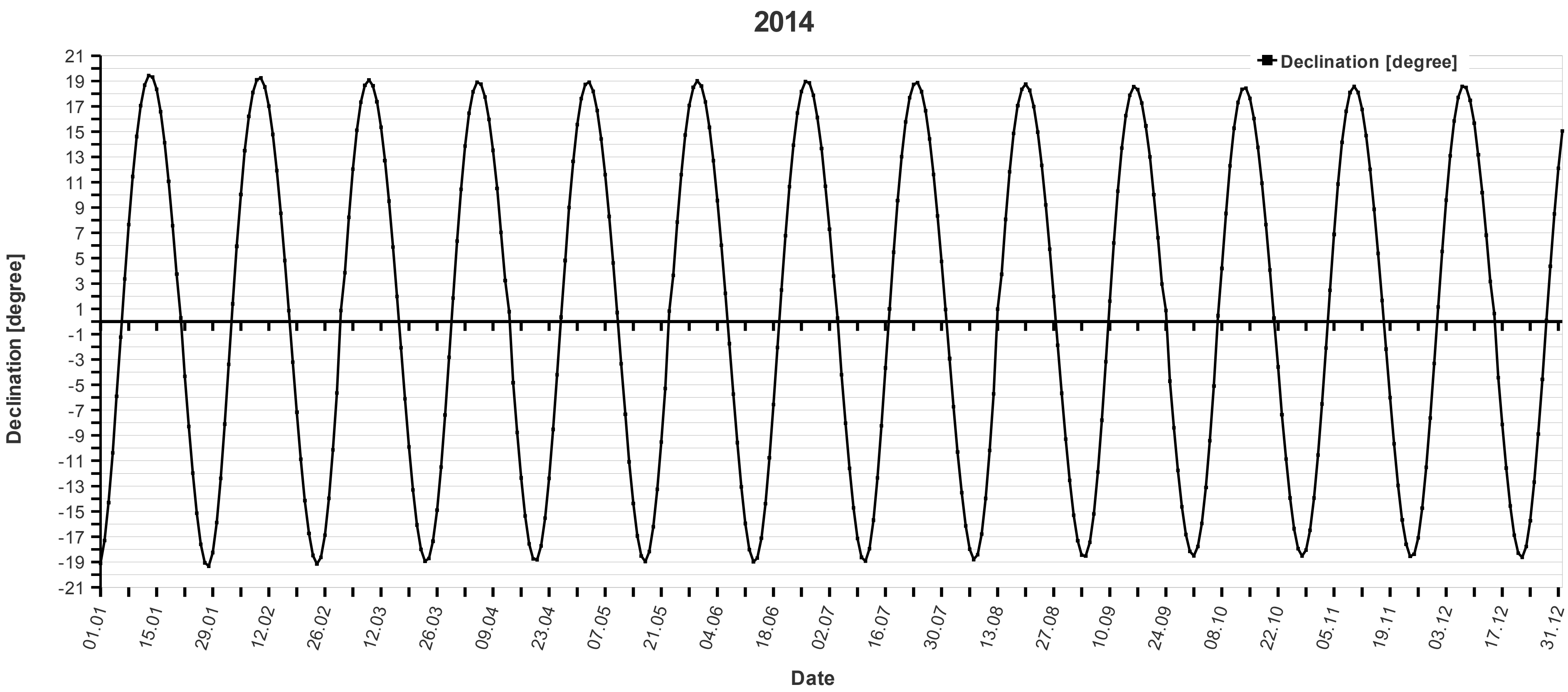

Figure 6.4 Moon Declination 2014

ANNEXE 3: TRAJECTORIES TO THE MOON

Table 20: Impulsive thrusting with bipropellant thrusters of I_{sp}=~335 s

Transfer trajectory	Spacecraft dry mass [kg]	Δv from LEO to LLO [m/s]	Transfer time [days]	Propellant mass [kg]	Cost at launch (at ~10'000 €/kg)[€]
Hohmann	600	3928	≥5	1382.62	13'826'243
Bielliptic	600	4153	~ 91	1523.12	15'231'193
Trielliptic (Bielliptic with lunar flyby)	600	3921	~ 95	1378.41	13'784'058
WSB (Sun-Earth L2)	600	3867	~ 100	1346.16	13'461'630

Table 21: Propellant price / litre [35, 36]

	Hydrazine	MMH	N_2O_4	LO_2	LH_2	Kerosene
Price per Litre	€150.4	€130.7	€39.14	€0.20	€0.29	€0.07

Table 22: Oxidizer to fuel ratio [37]

	N_2O_4 / UDMH	N_2O_4 / MMH	MON / MMH	LO_2 / LH_2	LO_2 / Kerosene
Oxidizer to Fuel Ratio	2.61	2.16	2.27	6.00	2.56

ANNEXE 4: SPACECRAFT MASS ESTIMATION

Table 23: Spacecraft mass breakdown

Subsystem	Small scientific satellites mass breakdown [10,11]	GEO telecom satellites mass breakdown	Interplanetary missions mass breakdown (Ranger, Surveyor, IMP , Mariner; 1960's to 1970's)	Planetary missions [38]	Changes to GLXP requirements	Technologies	Worst case GLXP mass breakdown	Best case GLXP mass breakdown
Structure	24,1%[12]	17,1%	29,8%	29,0%	Reinforced structure is needed for lunar landing Additional mass is required for rover harness	- Loy density alloys - Composite materials	32%	30%
Thermal	1,8%	5%		3,0%	Thermal environment of lunar mission similar to GEO missions	- Conduction heat transfer - Phase changing heat transfer	6%	5%
Telemetry & Data	4,8%	3,1%	12,3%	19% (cabling included, 7%)	Accurate telemetry subsystem required for precise spacecraft trajectory guidance and landing	- Commercial of the shelves electronics - Space environment resistant electronics	5,5%	5%
Propulsion	6,9% [13]	8,7%	13,3%	13,0%	Increased propulsion is needed for large Δv of lunar transfer, lunar orbit injection and landing	- Monopropellant or bipropellant thruster	21,5%	20%
Attitude, Control & Guidance	7%	6,9%		9,0%	Accurate attitude subsystem required for precise spacecraft trajectory guidance and landing	- Monopropellant thruster	11%	10%
Power	14%	29,5%	24,7%	19,'%	Although high power is required for larger distance data transfer, payload and spacecraft do not require constant power supply such as encounter in telecommunication satellites.	- Lithium ion battery - Solar panels	16%	16%
Payload	27,3%	29,7%	8,1%	11,0%	Rover	-	8%	14%

[10] *Average values, the sum of mass break down values does not equal 100% mass of the spacecraft.*
[11] *Missions are: Orsted , Freya, SAMPLEX, ANS and Viking*
[12] *Includes launcher slot adapter and cabling*
[13] *Not all scientific satellites had a propulsion system*

Table 24: Propellant mass estimation, transfer from LEO, mono- & bipropellant thrusting

		Worst case	Unfavourable case	Favourable case	Best case	
Lunar Landing	**Propellant mass**	748.95	427.80	149.79	85.08	*kg*
	dry mass	625.00	357.00	125.00	71.00	*kg*
	Isp (hydrazine)	220.00	220.00	220.00	220.00	*s*
	Gravity	9.81	9.81	9.81	9.81	*m/s²*
	Delta-v	1,700.00	1,700.00	1,700.00	1,700.00	*m/s*
Lunar transfer	**Propellant mass**	3,166.09	1,808.47	633.22	359.67	*kg*
	Spacecraft mass	1,373.95	784.80	274.79	156.08	*kg*
	Isp (N2O4/MMH)	335.00	335.00	335.00	335.00	*s*
	Gravity	9.81	9.81	9.81	9.81	*m/s²*
	Delta-v	3,928.00	3,928.00	3,928.00	3,928.00	*m/s*
Wett total spacraft mass at launch		4,540.04	2,593.27	908.01	515.75	*kg*

Table 25: Propellant mass estimation, transfer from GTO, mono- & bipropellant thrusting

		Worst case	Unfavourable case	Favourable case	Best case	
Lunar Landing	**Propellant mass**	748.95	427.80	149.79	85.08	*kg*
	dry mass	625.00	357.00	125.00	71.00	*kg*
	Isp (hydrazine)	220.00	220.00	220.00	220.00	*s*
	Gravity	9.81	9.81	9.81	9.81	*m/s²*
	Delta-v	1,700.00	1,700.00	1,700.00	1,700.00	*m/s*
Lunar transfer	**Propellant mass**	930.82	531.68	186.16	105.74	*kg*
	Spacecraft mass	1,373.95	784.80	274.79	156.08	*kg*
	Isp (N2O4/MMH)	335.00	335.00	335.00	335.00	*s*
	Gravity	9.81	9.81	9.81	9.81	*m/s²*
	Delta-v	1,700.00	1,700.00	1,700.00	1,700.00	*m/s*
Wett total spacraft mass at launch		2,304.77	1,316.48	460.95	261.82	*kg*

Table 26: Propellant mass estimation, transfer from LEO, all monopropellant thrusting

		Worst case	Unfavourable case	Favourable case	Best case	
Lunar Landing	**Propellant mass**	748.95	427.80	149.79	85.08	*kg*
	dry mass	625.00	357.00	125.00	71.00	*kg*
	Isp (hydrazine)	220.00	220.00	220.00	220.00	*s*
	Gravity	9.81	9.81	9.81	9.81	*m/s²*
	Delta-v	1,700.00	1,700.00	1,700.00	1,700.00	*m/s*
Lunar transfer	**Propellant mass**	7,106.17	4,059.05	1,421.23	807.26	*kg*
	Spacecraft mass	1,373.95	784.80	274.79	156.08	*kg*
	Isp (N2O4/MMH)	220.00	220.00	220.00	220.00	*s*
	Gravity	9.81	9.81	9.81	9.81	*m/s²*
	Delta-v	3,928.00	3,928.00	3,928.00	3,928.00	*m/s*
Wett total spacraft mass at launch		8,480.12	4,843.85	1,696.02	963.34	*kg*

Table 27: Propellant mass estimation, transfer from GTO, all monopropellant thrusting

		Worst case	Unfavourable case	Favourable case	Best case	
Lunar Landing	**Propellant mass**	748.95	427.80	149.79	85.08	*kg*
	dry mass	625.00	357.00	125.00	71.00	*kg*
	Isp (hydrazine)	220.00	220.00	220.00	220.00	*s*
	Gravity	9.81	9.81	9.81	9.81	*m/s²*
	Delta-v	1,700.00	1,700.00	1,700.00	1,700.00	*m/s*
Lunar transfer	**Propellant mass**	1,646.43	940.44	329.29	187.03	*kg*
	Spacecraft mass	1,373.95	784.80	274.79	156.08	*kg*
	Isp (N2O4/MMH)	220.00	220.00	220.00	220.00	*s*
	Gravity	9.81	9.81	9.81	9.81	*m/s²*
	Delta-v	1,700.00	1,700.00	1,700.00	1,700.00	*m/s*
Wett total spacraft mass at launch		3,020.38	1,725.24	604.08	343.12	*kg*

ANNEXE 5: DATA & FIRST GUESS CALCULATIONS FOR NUMERICAL MODEL

From LEO

LEO to lunar circular orbit transfer data:

Denomination	Abbreviation	Value
Translunar kick	Δv1	-
Lunar insertion kick	Δv2	-
LEO radius about the earth	r_E	6555 km (184 + R_E)
Moon radius about the earth	R	4 x 10^5 km
LLO radius about the moon	r_M	1837 km (100 + R_M)
Moon semi-major axis	a_m	384399 km
Earth's mean radius	R_E	6371 km
Moon's mean radius	R_M	1737 km
Earth gravitational parameter	μ_Earth	3.986 x 10^5 km^3/s^2
Moon gravitational parameter	μ_Moon	4.903 x 10^3 km^3/s^2

Referring to the Hohmann transfer representation in figure 1.5, the respective equations for Δv calculation are:

$$\Delta v_{LEOtot} = \Delta v_1 + \Delta v_2 \qquad (6.1)$$

When defining the Δv at each point we have:

$$\Delta v_1 = v_1 - v_{p1} \qquad (6.2)$$

and

$$\Delta v_2 = v_2 - v_{p2} \qquad (6.3)$$

When defining the first kick Δv_1 :

$$v_1 = \sqrt{2 \cdot \left(\frac{\mu_{Earth}}{r_E} - \frac{\mu_{Earth}}{r_E + R + r_M} \right)} \qquad (6.4)$$

$$v_1 = \sqrt{2 \cdot \left(\frac{3.968 \cdot 10^5}{6555} - \frac{3.968 \cdot 10^5}{6555 + 4 \cdot 10^5 + 1837} \right)} = 10.94 \ \frac{km}{s} \qquad (6.5)$$

$$v_{p1} = \sqrt{\frac{\mu_{Earth}}{r_E}} \qquad (6.6)$$

$$v_{p1} = \sqrt{\frac{3.986 \cdot 10^5}{6555}} = 7.80 \ \frac{km}{s} \qquad (6.7)$$

$$\Delta v_1 = 10.94 - 7.80 = 3.14 \ \frac{km}{s} \qquad (6.8)$$

When defining the lunar orbit insertion kick Δv_2 :

$$v_2=\sqrt{v^2{}_{\infty/m}+\frac{2\mu_m}{r_M}} \quad (6.9)$$

with

$$v_{\infty/m}=\|v_{rM}-v_{Moon/Earth}\| \quad (6.10)$$

where

$$v_{rM}=\sqrt{\mu_{Earth}\cdot\left(\frac{2}{R+r_M}-\frac{1}{a_M}\right)} \quad (6.11)$$

as well as

$$v_{Moon/Earth}=\ 1\quad km/s \quad \text{Corresponding to the velocity of the moon relative to the earth} \quad (6.12)$$

It is obtained:

$$v_{rM}=\sqrt{3.986\cdot10^5\cdot\left(\frac{2}{4\cdot10^5+1837}-\frac{1}{384399}\right)}=\ 0.97\quad\frac{km}{s} \quad (6.13)$$

$$v_{\infty/m}=\|0.97-1\|=\ 0.03\quad\frac{km}{s} \quad (6.14)$$

$$v_2=\sqrt{0.03^2+\frac{2\cdot4903}{1837}}=\ 2.31\quad\frac{km}{s} \quad (6.15)$$

Corresponds to the velocity on the lunar circular orbit

$$v_{p2}=\sqrt{\frac{\mu_{Moon}}{r_M}} \quad (6.16)$$

$$v_{p2}=\sqrt{\frac{4903}{1837}}=\ 1.63\quad\frac{km}{s} \quad (6.17)$$

Finally:

$$\Delta v_2=2.31-1.63=\ 0.68\quad\frac{km}{s} \quad (6.18)$$

It is obtained a total Δv for a LEO transfer of:

$$\Delta v_{LEOtot}=3.14+0.68=\ 3.82\quad\frac{km}{s} \quad (6.19)$$

From GTO

GTO to lunar elliptical orbit transfer data:

Denomination	Abbreviation	Value
Translunar kick	Δv1	-
Lunar insertion kick	Δv2	-
GTO radius at apogee	r_{aE}	42071 km (35700 + R_E)
GTO radius at perigee	r_{pE}	6555 km (184 + R_E)
Moon radius about the earth	R	4 x 10^5 km
Moon semi-major axis	a_m	384399 km
LLO orbit radius	r_{pM}	1837 km (100 + R_M]
Earth's mean radius	R_E	6371 km
Moon's mean radius	R_M	1737 km
Earth gravitational parameter	μ_{Earth}	3.986 x 10^5 km³/s²
Moon gravitational parameter	μ_{Moon}	4.903 x 10^3 km³/s²

In this case, the transfer occurs between an elliptical orbit around the earth, the GTO, and an elliptical orbit around the moon.

$$\Delta v_{TOT} = \Delta v_1 + \Delta v_2 \tag{6.20}$$

When defining the Δv at each point we have:

$$\Delta v_1 = v_1 - v_{p1} \tag{6.21}$$

and

$$\Delta v_2 = v_2 - v_{p2} \tag{6.22}$$

When defining the first kick:

$$v_1 = \sqrt{2 \cdot \left(\frac{\mu_{Earth}}{r_{pE}} - \frac{\mu_{Earth}}{r_{pE} + R + r_{pM}} \right)} \tag{6.23}$$

$$v_1 = \sqrt{2 \cdot \left(\frac{3.986 \cdot 10^5}{6555} - \frac{3.986 \cdot 10^5}{6555 + 4 \cdot 10^5 + 1837} \right)} = 10.94 \ \frac{km}{s} \tag{6.24}$$

$$v_{p1} = \sqrt{2 \cdot \left(\frac{\mu_{Earth}}{r_{pE}} - \frac{\mu_{Earth}}{r_{pE} + r_{aE}} \right)} \tag{6.25}$$

$$v_{p1} = \sqrt{2 \cdot \left(\frac{3.986 \cdot 10^5}{6555} - \frac{3.986 \cdot 10^5}{6555 + 42071} \right)} = 10.26 \ \frac{km}{s} \tag{6.26}$$

$$\Delta v_1 = 10.94 - 10.26 = 0.68 \ \frac{km}{s} \tag{6.27}$$

The second kick has been defined previously for LEO calculations, $\Delta v_2 = 0.64$ km/s. It is obtained the total Δv for a GTO transfer:

$$\Delta v_{GTOtot} = 0.68 + 0.68 = 1.36 \frac{km}{s} \tag{6.28}$$

Inclination Change Manoeuvre

Denomination	Abbreviation	Value
GTO radius at apogee	r_{aE}	42071 km (35700 + R_E)
GTO radius at perigee	r_{pE}	6555 km (184 + R_E)
LEO radius about the earth	r_E	6555 km (184 + R_E)
Inclination change angle	θ	18.2° to 28.5°
Earth gravitational parameter	μ_{Earth}	3.986012 x 10^5 km³/s²
Moon gravitational parameter	μ_{Moon}	4.903 x 10^3 km³/s²

In figure 1.3 are represented the maximal and minimal inclination angles between earth and lunar equatorial planes, varying between 18.2° to 28.5°.
When considering a plane change manoeuvre, the worst case would be when the spacecraft is launched in the equatorial plane and manoeuvred until being in the lunar plane.
According to equations below, Δv for inclination variations about the earth are shown in table 28:
In LEO and GTO (initial and final orbit have the same orbital elements except inclination):

$$\Delta v_{Inc} = 2 \cdot v_p \cdot \sin\frac{\theta}{2} \tag{6.29}$$

Circular parking orbit

$$v_c = \sqrt{\frac{\mu_{Earth}}{r_E}} \tag{6.30}$$

Elliptical parking orbit

$$v_e = \sqrt{2 \cdot \left(\frac{\mu_{Earth}}{r_{aE}} - \frac{\mu_{Earth}}{r_{pE} + r_{aE}}\right)} \tag{6.31}$$

Table 28: Inclination out-of-plane manoeuvres about the earth

Type Of Orbit	LEO orbit radius	GTO Apoapsis	GTO Periapsis	V_c	V_e	Δv to 18.2°	Δv to 28.5°
LEO	6555 km	-	-	7.80 km/s	-	2.46 km/s	3.84 km/s
GTO	-	6555 km	42071 km	-	1.60 km/s	0.51 km/s	0.79 km/s

ANNEXE 6: GMAT DATA

Table 29: Δv and transfer time results from GMAT models

Launching orbit	Descent Type	lunar capture altitude [km]	Total transfer time [days]	Total delta-V [km/sec]	Delta-V translunar kick [km/sec]	Delta-V mid-course manoeuvres [km/sec]	Delta-V lunar injection kick [km/sec]	Delta-V deorbiting until 15km lunar altitude [km/sec]	Delta-V from 15km until touchdown [km/sec]
LEO (coast time = 1 day)	Luna 17/Lunokhod 1, 35°N 35°W, lunar inc. = 35°	30	5.32	7.260	3.132	0.067	2.159	0.007	1.896
		65	5.33	7.161	3.132	0.066	2.139	0.023	1.800
		100	5.35	7.132	3.132	0.065	2.120	0.039	1.776
	Direct descent	30	4.94	5.845	3.132	0.067	0.831	0.008	1.808
		65	4.94	5.857	3.132	0.066	0.826	0.024	1.810
		100	4.95	5.856	3.132	0.065	0.820	0.040	1.799
GTO (coast time = 1.5 day)	Luna 17/Lunokhod 1, 35°N 35°W, lunar inc. = 35°	30	6.06	5.135	0.677	0.244	2.202	0.008	2.005
		65	6.08	4.928	0.677	0.241	2.182	0.024	1.806
		100	6.11	4.891	0.677	0.237	2.162	0.040	1.776
	Direct descent	30	5.37	3.569	0.677	0.244	0.831	0.008	1.810
		65	5.37	3.577	0.677	0.241	0.825	0.024	1.810
		100	5.38	3.574	0.677	0.237	0.820	0.040	1.801

ANNEXE 7: GMAT SCRIPTS

To run the scripts, copy-paste the models below in a text editor (Notpad or Gedit) and save the file using .script extension.
GMAT is easily installed in its beta version in MS-Windows operating systems.

LEO Transfer Model

```
% LEO Transfer Model
% Mission Analysis for Google Lunar X-Prize Participants: Options For Going To The Moon
% Boris Maitre
% UPC-MAST Master's thesis
% 17.06.2010

%----------------------------------------
%---------- Spacecrafts
%----------------------------------------

Create Spacecraft Sat;
GMAT Sat.DateFormat = A1ModJulian;
GMAT Sat.Epoch = '26660.00002354609';
GMAT Sat.CoordinateSystem = EarthMJ2000Eq;
GMAT Sat.DisplayStateType = ModifiedKeplerian;
GMAT Sat.RadPer = 6552.999934470055;
GMAT Sat.RadApo = 6553.000065530066;
GMAT Sat.INC = 6.000000000000086;
GMAT Sat.RAAN = 306;
GMAT Sat.AOP = 314.1905560090334;
GMAT Sat.TA = 99.88774485900801;
GMAT Sat.DryMass = 200;
GMAT Sat.Cd = 2.2;
GMAT Sat.Cr = 1.8;
GMAT Sat.DragArea = 15;
GMAT Sat.SRPArea = 1;

Create Spacecraft InitSat;
GMAT InitSat.DateFormat = A1ModJulian;
GMAT InitSat.Epoch = '26660.00002354609';
GMAT InitSat.CoordinateSystem = EarthMJ2000Eq;
GMAT InitSat.DisplayStateType = ModifiedKeplerian;
GMAT InitSat.RadPer = 6552.999993447055;
GMAT InitSat.RadApo = 6553.000006553061;
GMAT InitSat.INC = 6.000000000000086;
GMAT InitSat.RAAN = 306;
GMAT InitSat.AOP = 314.1905228086426;
GMAT InitSat.TA = 99.88777805939871;
GMAT InitSat.DryMass = 200;
GMAT InitSat.Cd = 2.2;
GMAT InitSat.Cr = 1.8;
GMAT InitSat.DragArea = 15;
GMAT InitSat.SRPArea = 1;

Create Spacecraft InitSat2;
GMAT InitSat2.DateFormat = A1ModJulian;
GMAT InitSat2.Epoch = '26660.00002354609';
GMAT InitSat2.CoordinateSystem = EarthMJ2000Eq;
GMAT InitSat2.DisplayStateType = ModifiedKeplerian;
```

```
GMAT InitSat2.RadPer = 6552.999993447055;
GMAT InitSat2.RadApo = 6553.000006553061;
GMAT InitSat2.INC = 6.000000000000086;
GMAT InitSat2.RAAN = 306;
GMAT InitSat2.AOP = 314.1905228086426;
GMAT InitSat2.TA = 99.88777805939871;
GMAT InitSat2.DryMass = 200;
GMAT InitSat2.Cd = 2.2;
GMAT InitSat2.Cr = 1.8;
GMAT InitSat2.DragArea = 15;
GMAT InitSat2.SRPArea = 1;

%----------------------------------------
%---------- ForceModels
%----------------------------------------

Create ForceModel EarthProp_ForceModel;
GMAT EarthProp_ForceModel.CentralBody = Earth;
GMAT EarthProp_ForceModel.PrimaryBodies = {Earth};
GMAT EarthProp_ForceModel.PointMasses = {Sun};
GMAT EarthProp_ForceModel.Drag = None;
GMAT EarthProp_ForceModel.SRP = Off;
GMAT EarthProp_ForceModel.ErrorControl = RSSStep;
GMAT EarthProp_ForceModel.GravityField.Earth.Degree = 4;
GMAT EarthProp_ForceModel.GravityField.Earth.Order = 4;
GMAT EarthProp_ForceModel.GravityField.Earth.PotentialFile = 'JGM2.cof';

Create ForceModel EarthMoonProp_ForceModel;
GMAT EarthMoonProp_ForceModel.CentralBody = Earth;
GMAT EarthMoonProp_ForceModel.PrimaryBodies = {Earth};
GMAT EarthMoonProp_ForceModel.PointMasses = {Sun, Luna};
GMAT EarthMoonProp_ForceModel.Drag = None;
GMAT EarthMoonProp_ForceModel.SRP = Off;
GMAT EarthMoonProp_ForceModel.ErrorControl = RSSStep;
GMAT EarthMoonProp_ForceModel.GravityField.Earth.Degree = 4;
GMAT EarthMoonProp_ForceModel.GravityField.Earth.Order = 4;
GMAT EarthMoonProp_ForceModel.GravityField.Earth.PotentialFile = 'JGM2.cof';

Create ForceModel MoonProp_ForceModel;
GMAT MoonProp_ForceModel.CentralBody = Luna;
GMAT MoonProp_ForceModel.PrimaryBodies = {Luna};
GMAT MoonProp_ForceModel.PointMasses = {Sun, Earth};
GMAT MoonProp_ForceModel.Drag = None;
GMAT MoonProp_ForceModel.SRP = Off;
GMAT MoonProp_ForceModel.ErrorControl = RSSStep;
GMAT MoonProp_ForceModel.GravityField.Luna.Degree = 4;
GMAT MoonProp_ForceModel.GravityField.Luna.Order = 4;
GMAT MoonProp_ForceModel.GravityField.Luna.PotentialFile = 'LP165P.cof';


%----------------------------------------
%---------- Propagators
%----------------------------------------

Create Propagator EarthProp;
GMAT EarthProp.FM = EarthProp_ForceModel;
GMAT EarthProp.Type = RungeKutta89;
GMAT EarthProp.InitialStepSize = 60;
GMAT EarthProp.Accuracy = 9.999999999999999e-12;
GMAT EarthProp.MinStep = 0.001;
```

```
GMAT EarthProp.MaxStep = 2700;
GMAT EarthProp.MaxStepAttempts = 50;

Create Propagator EarthMoonProp;
GMAT EarthMoonProp.FM = EarthMoonProp_ForceModel;
GMAT EarthMoonProp.Type = RungeKutta89;
GMAT EarthMoonProp.InitialStepSize = 60;
GMAT EarthMoonProp.Accuracy = 9.999999999999999e-12;
GMAT EarthMoonProp.MinStep = 0.001;
GMAT EarthMoonProp.MaxStep = 2700;
GMAT EarthMoonProp.MaxStepAttempts = 50;

Create Propagator MoonProp;
GMAT MoonProp.FM = MoonProp_ForceModel;
GMAT MoonProp.Type = RungeKutta89;
GMAT MoonProp.InitialStepSize = 60;
GMAT MoonProp.Accuracy = 9.999999999999999e-12;
GMAT MoonProp.MinStep = 1e-09;
GMAT MoonProp.MaxStep = 2700;
GMAT MoonProp.MaxStepAttempts = 50;

%----------------------------------------
%---------- Burns
%----------------------------------------

Create ImpulsiveBurn TOI;
GMAT TOI.Origin = Earth;
GMAT TOI.Axes = VNB;
GMAT TOI.VectorFormat = Cartesian;
GMAT TOI.Element1 = 0;
GMAT TOI.Element2 = 0;
GMAT TOI.Element3 = 0;

Create ImpulsiveBurn IncMod;
GMAT IncMod.Origin = Earth;
GMAT IncMod.Axes = VNB;
GMAT IncMod.VectorFormat = Cartesian;
GMAT IncMod.Element1 = 0;
GMAT IncMod.Element2 = 0;
GMAT IncMod.Element3 = 0;

Create ImpulsiveBurn LOI;
GMAT LOI.Origin = Luna;
GMAT LOI.Axes = VNB;
GMAT LOI.VectorFormat = Cartesian;
GMAT LOI.Element1 = 0;
GMAT LOI.Element2 = 0;
GMAT LOI.Element3 = 0;

Create ImpulsiveBurn Descent;
GMAT Descent.Origin = Luna;
GMAT Descent.Axes = VNB;
GMAT Descent.VectorFormat = Cartesian;
GMAT Descent.Element1 = 0;
GMAT Descent.Element2 = 0;
GMAT Descent.Element3 = 0;

Create ImpulsiveBurn DOI;
GMAT DOI.Origin = Luna;
GMAT DOI.Axes = VNB;
```

```
GMAT DOI.VectorFormat = Cartesian;
GMAT DOI.Element1 = 0;
GMAT DOI.Element2 = 0;
GMAT DOI.Element3 = 0;

Create ImpulsiveBurn Brake;
GMAT Brake.Origin = Luna;
GMAT Brake.Axes = VNB;
GMAT Brake.VectorFormat = Cartesian;
GMAT Brake.Element1 = 0;
GMAT Brake.Element2 = 0;
GMAT Brake.Element3 = 0;

%----------------------------------------
%---------- Variables, Arrays, Strings
%----------------------------------------

Create Variable RAAN RAAN1 AOP INC Epoch DeltaV DeltaVearth DeltaVinc DeltaVmoon
DeltaVdescent;
Create Variable DeltaVtouchdown Epoch1 Epoch2 Epoch3 CoastTime ECC SMA TOI1 VmagLOI
INC2;
Create Variable RadPer RadApo Xint Yint Zint;

%----------------------------------------
%---------- Coordinate Systems
%----------------------------------------

Create CoordinateSystem EarthMJ2000Eq;
GMAT EarthMJ2000Eq.Origin = Earth;
GMAT EarthMJ2000Eq.Axes = MJ2000Eq;
GMAT EarthMJ2000Eq.UpdateInterval = 60;
GMAT EarthMJ2000Eq.OverrideOriginInterval = false;

Create CoordinateSystem EarthMJ2000Ec;
GMAT EarthMJ2000Ec.Origin = Earth;
GMAT EarthMJ2000Ec.Axes = MJ2000Ec;
GMAT EarthMJ2000Ec.UpdateInterval = 60;
GMAT EarthMJ2000Ec.OverrideOriginInterval = false;

Create CoordinateSystem EarthFixed;
GMAT EarthFixed.Origin = Earth;
GMAT EarthFixed.Axes = BodyFixed;
GMAT EarthFixed.UpdateInterval = 60;
GMAT EarthFixed.OverrideOriginInterval = false;

Create CoordinateSystem EarthToMoon;
GMAT EarthToMoon.Origin = Luna;
GMAT EarthToMoon.Axes = ObjectReferenced;
GMAT EarthToMoon.UpdateInterval = 60;
GMAT EarthToMoon.OverrideOriginInterval = false;
GMAT EarthToMoon.XAxis = R;
GMAT EarthToMoon.ZAxis = N;
GMAT EarthToMoon.Primary = Luna;
GMAT EarthToMoon.Secondary = Earth;

Create CoordinateSystem LunaFixed;
GMAT LunaFixed.Origin = Luna;
GMAT LunaFixed.Axes = BodyFixed;
GMAT LunaFixed.UpdateInterval = 60;
GMAT LunaFixed.OverrideOriginInterval = false;
```

```
%----------------------------------------
%---------- Solvers
%----------------------------------------

Create DifferentialCorrector DC1;
GMAT DC1.ShowProgress = true;
GMAT DC1.ReportStyle = 'Normal';
GMAT DC1.TargeterTextFile = 'DifferentialCorrectorDefaultDC.data';
GMAT DC1.MaximumIterations = 200;
GMAT DC1.UseCentralDifferences = false;

%----------------------------------------
%---------- Plots and Reports
%----------------------------------------

Create OpenGLPlot EarthOGL;
GMAT EarthOGL.SolverIterations = Current;
GMAT EarthOGL.Add = {Sat, Earth, Luna, Sun};
GMAT EarthOGL.OrbitColor = [ 251658239 32768 251691263 253401294 ];
GMAT EarthOGL.TargetColor = [ 8421440 8421440 8421440 8421440 ];
GMAT EarthOGL.CoordinateSystem = EarthMJ2000Eq;
GMAT EarthOGL.ViewPointReference = Earth;
GMAT EarthOGL.ViewPointVector = [ 30000 30000 0 ];
GMAT EarthOGL.ViewDirection = Earth;
GMAT EarthOGL.ViewScaleFactor = 1;
GMAT EarthOGL.ViewUpCoordinateSystem = EarthMJ2000Eq;
GMAT EarthOGL.ViewUpAxis = Z;
GMAT EarthOGL.CelestialPlane = Off;
GMAT EarthOGL.XYPlane = Off;
GMAT EarthOGL.WireFrame = Off;
GMAT EarthOGL.Axes = On;
GMAT EarthOGL.Grid = Off;
GMAT EarthOGL.SunLine = On;
GMAT EarthOGL.UseInitialView = On;
GMAT EarthOGL.DataCollectFrequency = 1;
GMAT EarthOGL.UpdatePlotFrequency = 50;
GMAT EarthOGL.NumPointsToRedraw = 0;
GMAT EarthOGL.ShowPlot = true;

Create OpenGLPlot EarthMoonOGL;
GMAT EarthMoonOGL.SolverIterations = Current;
GMAT EarthMoonOGL.Add = {Sat, Luna, Earth, Sun};
GMAT EarthMoonOGL.OrbitColor = [ 255 1864014030 1743054 1743054 ];
GMAT EarthMoonOGL.TargetColor = [ 897613888 8421440 8421440 8421440 ];
GMAT EarthMoonOGL.CoordinateSystem = EarthToMoon;
GMAT EarthMoonOGL.ViewPointReference = Luna;
GMAT EarthMoonOGL.ViewPointVector = [ 30000 0 0 ];
GMAT EarthMoonOGL.ViewDirection = Luna;
GMAT EarthMoonOGL.ViewScaleFactor = 1;
GMAT EarthMoonOGL.ViewUpCoordinateSystem = EarthToMoon;
GMAT EarthMoonOGL.ViewUpAxis = Z;
GMAT EarthMoonOGL.CelestialPlane = Off;
GMAT EarthMoonOGL.XYPlane = Off;
GMAT EarthMoonOGL.WireFrame = Off;
GMAT EarthMoonOGL.Axes = On;
GMAT EarthMoonOGL.Grid = Off;
GMAT EarthMoonOGL.SunLine = On;
GMAT EarthMoonOGL.UseInitialView = On;
GMAT EarthMoonOGL.DataCollectFrequency = 1;
GMAT EarthMoonOGL.UpdatePlotFrequency = 50;
```

```
GMAT EarthMoonOGL.NumPointsToRedraw = 0;
GMAT EarthMoonOGL.ShowPlot = true;

Create ReportFile LunarDATA;
GMAT LunarDATA.SolverIterations = Current;
GMAT LunarDATA.Filename = './output/ReportFile1.txt';
GMAT LunarDATA.Precision = 16;
GMAT LunarDATA.Add = {Sat.A1ModJulian, Sat.ElapsedDays, Sat.Luna.Latitude,
Sat.Luna.Longitude, Sat.LunaFixed.RAAN, Sat.Luna.Altitude, Sat.LunaFixed.INC};
GMAT LunarDATA.WriteHeaders = On;
GMAT LunarDATA.LeftJustify = On;
GMAT LunarDATA.ZeroFill = Off;
GMAT LunarDATA.ColumnWidth = 20;

Create OpenGLPlot DescentOGL;
GMAT DescentOGL.SolverIterations = Current;
GMAT DescentOGL.Add = {Sat, Luna};
GMAT DescentOGL.OrbitColor = [ 251658239 33023 ];
GMAT DescentOGL.TargetColor = [ 8421440 8421440 ];
GMAT DescentOGL.CoordinateSystem = LunaFixed;
GMAT DescentOGL.ViewPointReference = Earth;
GMAT DescentOGL.ViewPointVector = [ -2000 0 0 ];
GMAT DescentOGL.ViewDirection = Earth;
GMAT DescentOGL.ViewScaleFactor = 1;
GMAT DescentOGL.ViewUpCoordinateSystem = LunaFixed;
GMAT DescentOGL.ViewUpAxis = Z;
GMAT DescentOGL.CelestialPlane = Off;
GMAT DescentOGL.XYPlane = Off;
GMAT DescentOGL.WireFrame = Off;
GMAT DescentOGL.Axes = On;
GMAT DescentOGL.Grid = Off;
GMAT DescentOGL.SunLine = Off;
GMAT DescentOGL.UseInitialView = On;
GMAT DescentOGL.DataCollectFrequency = 1;
GMAT DescentOGL.UpdatePlotFrequency = 50;
GMAT DescentOGL.NumPointsToRedraw = 0;
GMAT DescentOGL.ShowPlot = true;

Create ReportFile EarthDATA;
GMAT EarthDATA.SolverIterations = Current;
GMAT EarthDATA.Filename = './output/ReportFile2.txt';
GMAT EarthDATA.Precision = 16;
GMAT EarthDATA.Add = {Sat.A1ModJulian, Sat.EarthToMoon.X, Sat.EarthToMoon.Y,
Sat.EarthToMoon.Z, Sat.Earth.Altitude, Sat.EarthMJ2000Eq.AOP, AOP, Sat.EarthMJ2000Eq.INC,
INC, Sat.EarthMJ2000Eq.RAAN, RAAN};
GMAT EarthDATA.WriteHeaders = On;
GMAT EarthDATA.LeftJustify = On;
GMAT EarthDATA.ZeroFill = Off;
GMAT EarthDATA.ColumnWidth = 20;

Create ReportFile VariablesDATA;
GMAT VariablesDATA.SolverIterations = Current;
GMAT VariablesDATA.Filename = './output/ReportFile3.txt';
GMAT VariablesDATA.Precision = 16;
GMAT VariablesDATA.Add = {Sat.A1ModJulian, DeltaV, DeltaVearth, DeltaVmoon, DeltaVdescent,
DeltaVtouchdown, DeltaVinc, Sat.EarthMJ2000Eq.INC};
GMAT VariablesDATA.WriteHeaders = On;
GMAT VariablesDATA.LeftJustify = On;
GMAT VariablesDATA.ZeroFill = Off;
GMAT VariablesDATA.ColumnWidth = 20;
```

```
%----------------------------------------
%---------- Mission Sequence
%----------------------------------------

%*********** Sequence 0 *************

% sequences 0 to 2 are not plotted in order to have the final trajectory only
Toggle EarthMoonOGL EarthOGL DescentOGL Off;

% initialization of variables
GMAT Epoch2 = Sat.A1ModJulian;
GMAT INC2 = Sat.INC;
GMAT RadPer = Sat.RadPer;
GMAT RadApo = Sat.RadApo;

% propagation until periapsis
Propagate EarthProp(Sat) {Sat.Earth.Periapsis};

% initialization of variables
GMAT RAAN = Sat.RAAN;
GMAT AOP = Sat.AOP;
GMAT INC = Sat.INC;
GMAT InitSat = Sat;

%************** Sequence 1 **************
% First translunar kick loop
%****************************************

% parking orbit target orientation and TLI maneuver to align line of apsides with moon at lunar
encounter
Target DC1;
  Vary DC1(TOI.Element1 = 3.15, {Perturbation = 0.0001, MaxStep = 0.01, Lower = -5, Upper = 5});
  Vary DC1(Sat.AOP = AOP, {Perturbation = 0.2, MaxStep = 15, Lower = -360, Upper = 360});
  Vary DC1(Sat.RAAN = RAAN, {Perturbation = 0.2, MaxStep = 15, Lower = -360, Upper = 360});
  Vary DC1(Sat.INC = INC, {Perturbation = 0.2, MaxStep = 15, Lower = -720, Upper = 720});

  % save RAAN, AOP and INC values for later use
  GMAT RAAN = Sat.RAAN;
  GMAT AOP = Sat.AOP;
  GMAT INC = Sat.INC;

  Maneuver TOI(Sat);
  Propagate EarthProp(Sat) {Sat.Earth.Apoapsis};

  Achieve DC1(Sat.EarthToMoon.X = 0, {Tolerance = 0.1});
  Achieve DC1(Sat.EarthToMoon.Y = -8000, {Tolerance = 0.1});
  Achieve DC1(Sat.EarthToMoon.Z = 0, {Tolerance = 0.1});
EndTarget;  % For targeter DefaultDC


%************** Sequence 2 **************
% Second translunar kick loop : refining sequence 1
%****************************************

% save current orbital elements and epoch in Initsat
GMAT Sat = InitSat;

Target DC1;
  Vary DC1(TOI.Element1 = TOI.Element1, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -3,
Upper = 5});
```

```
   Vary DC1(Sat.AOP = AOP, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper =
9.999999e300});
   Vary DC1(Sat.RAAN = RAAN, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper
= 9.999999e300});
   Vary DC1(Sat.INC = INC, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper =
9.999999e300});

   Maneuver TOI(Sat);
   Propagate EarthMoonProp(Sat) {Sat.ElapsedDays = 2.0};
   Propagate MoonProp(Sat) {Sat.Luna.Periapsis};

   Achieve DC1(Sat.Luna.Altitude = 100, {Tolerance = 1.0});

   GMAT Epoch3 = Sat.A1ModJulian;
   GMAT InitSat2 = Sat;
EndTarget;  % For targeter DefaultDC


%************** Sequence 3 **************
% Targeting a point on the moon's sphere of incluence using sequence 2 values
%****************************************

% trajectory is plotted
Toggle EarthMoonOGL EarthOGL DescentOGL On;

% back to conditions before lunar transfer
GMAT Sat = InitSat;

Target DC1;
   Vary DC1(TOI.Element1 = TOI.Element1, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -3,
Upper = 5});
   Vary DC1(Sat.AOP = AOP, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper =
9.999999e300});
   Vary DC1(Sat.RAAN = RAAN, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper
= 9.999999e300});

   % save RAAN and AOP values for later use
   GMAT RAAN = Sat.RAAN;
   GMAT AOP = Sat.AOP;

   Maneuver TOI(Sat);
   Propagate EarthMoonProp(Sat) {Sat.A1ModJulian = 26662.8095802};

   Achieve DC1(Sat.EarthToMoon.X = 65743, {Tolerance = 0.1});
   Achieve DC1(Sat.EarthToMoon.Y = -139444, {Tolerance = 0.1});
EndTarget;  % For targeter DefaultDC

% calculation of Delta-V for lunar transfer
GMAT DeltaVearth = sqrt( TOI.V^2 + TOI.N^2 + TOI.B^2 );

% initialization of variables
GMAT Xint = Sat.EarthToMoon.X;
GMAT Yint = Sat.EarthToMoon.Y;
GMAT Zint = Sat.EarthToMoon.Z;
GMAT Epoch = Sat.A1ModJulian;
```

```
%************** Sequence 4 **************
% Back propagation from lunar insertion point to previous targeted point on moon's sphere of
influence
%****************************************

GMAT Sat = InitSat2;

Target DC1;
   Vary DC1(TOI.V = 0, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -5, Upper = 5});
   Vary DC1(TOI.N = 0, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -5, Upper = 5});
   Vary DC1(TOI.B = 0, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -5, Upper = 5});

   Maneuver TOI(Sat);
   Propagate BackProp MoonProp(Sat) {Sat.A1ModJulian = 26662.8095802};

   Achieve DC1(Sat.EarthToMoon.X = Xint, {Tolerance = 0.1});
   Achieve DC1(Sat.EarthToMoon.Y = Yint, {Tolerance = 0.1});
   Achieve DC1(Sat.EarthToMoon.Z = Zint, {Tolerance = 0.1});
EndTarget;  % For targeter DefaultDC

% calculation of Delta-V for lunar transfer
GMAT DeltaVinc = sqrt( TOI.V^2 + TOI.N^2 + TOI.B^2 );

%************** Sequence 5 **************
% Lunar capture
%****************************************

% back to conditions at the end of lunar transfer
GMAT Sat = InitSat2;

% lunar capture kick
Target DC1;
   Vary DC1(LOI.V = -0.4, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -4, Upper = 3.14});

   Maneuver LOI(Sat);
   Propagate MoonProp(Sat) {Sat.Luna.Apoapsis};

   Achieve DC1(Sat.Luna.ECC = 0, {Tolerance = 0.0001});
EndTarget;  % For targeter DefaultDC

% calculation of Delta-V for lunar capture
GMAT DeltaVmoon = sqrt( LOI.V^2 + LOI.N^2 + LOI.B^2 );


%************** Sequence 6 **************
% Perigee lowering to 15 km
%****************************************

% poisitioning at longitude = 180° before de-orbiting
Propagate MoonProp(Sat) {Sat.Luna.Longitude = 180};

% saving spacecraft velocity vector magnitude at 100km lunar altitude and inclination
GMAT VmagLOI = Sat.LunaFixed.VMAG;
GMAT INC = Sat.LunaFixed.INC;

% kick for lowering from 100 km
Target DC1;
   Vary DC1(DOI.V = -0.4, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -4, Upper = 4});
   Maneuver DOI(Sat);
   Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
```

```
   Achieve DC1(Sat.Luna.Altitude = 15, {Tolerance = 0.1});
   Achieve DC1(Sat.LunaFixed.INC = INC, {Tolerance = 0.1});
EndTarget;  % For targeter DC1

% kick for orbit insertion at 15km
Target DC1;
   Vary DC1(Brake.V = 0.4, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -4, Upper = 4});
   Maneuver Brake(Sat);
   Achieve DC1(Sat.Luna.ECC = 0, {Tolerance = 0.0001});
EndTarget;  % For targeter DC1

% calculation of Delta-V for lowering kick
GMAT DeltaVdescent = sqrt( DOI.V^2 + DOI.N^2 + DOI.B^2 ) + sqrt( Brake.V^2 + Brake.N^2 +
Brake.B^2 );


%************** Sequence 7 **************
% Propagating until flying over landing site
%****************************************

% definition of Epoch1 variable for descent time calculation
GMAT Epoch1 = Sat.A1ModJulian;

% incrementation of Epoch with descent time in order to have the landed Epoch
GMAT Epoch1 = Epoch1  + 0.01157407402; % 1000 seconds

Target DC1;
   Vary DC1(Descent.V = -0.1, {Perturbation = 0.0001, MaxStep = 0.5, Lower = -100, Upper = 100});

   Maneuver Descent(Sat);
   Propagate MoonProp(Sat) {Sat.Luna.Longitude = 15};

   Achieve DC1(Sat.Luna.Altitude = 0, {Tolerance = 0.1});
   Achieve DC1(Sat.A1ModJulian = Epoch1, {Tolerance = 0.1});
   Achieve DC1(Sat.Luna.Latitude = Sat.Luna.Latitude, {Tolerance = 0.5});
EndTarget;  % For targeter DC1

% calculation of Delta-V for descent
GMAT DeltaVtouchdown = sqrt( Descent.V^2 + Descent.N^2 + Descent.B^2 ) + VmagLOI;

% calculation for Total Delta-V
GMAT DeltaV = DeltaVearth + DeltaVinc+ DeltaVmoon + DeltaVdescent + DeltaVtouchdown;
Report VariablesDATA Sat.A1ModJulian DeltaV DeltaVearth DeltaVinc DeltaVmoon DeltaVdescent
DeltaVtouchdown;

%************** Sequence 8 **************
% Propagating 2 GTO orbits at initial conditions
%****************************************

% setting the spacecraft at initial conditions of epoch, RAAN and AOP
GMAT Sat = InitSat;
GMAT Sat.RAAN = RAAN;
GMAT Sat.AOP = AOP;

% propagating 2 orbits
Propagate BackProp EarthProp(Sat) {Sat.Earth.Periapsis};
Propagate BackProp EarthProp(Sat) {Sat.Earth.Periapsis};
```

GTO Transfer Model

```
% GTO Transfer Model
% Mission Analysis for Google Lunar X-Prize Participants: Options For Going To The Moon
% Boris Maitre
% UPC-MAST Master's thesis
% 17.06.2010

%----------------------------------------
%---------- Spacecrafts
%----------------------------------------

Create Spacecraft Sat;
GMAT Sat.DateFormat = A1ModJulian;
GMAT Sat.Epoch = '26660.00002354609';
GMAT Sat.CoordinateSystem = EarthMJ2000Eq;
GMAT Sat.DisplayStateType = ModifiedKeplerian;
GMAT Sat.RadPer = 6551.999934470081;
GMAT Sat.RadApo = 42071.00000000009;
GMAT Sat.INC = 6.000000000000086;
GMAT Sat.RAAN = 315.9999999999999;
GMAT Sat.AOP = 0;
GMAT Sat.TA = 99.88774800306118;
GMAT Sat.DryMass = 200;
GMAT Sat.Cd = 2.2;
GMAT Sat.Cr = 1.8;
GMAT Sat.DragArea = 15;
GMAT Sat.SRPArea = 1;

Create Spacecraft InitSat;
GMAT InitSat.DateFormat = A1ModJulian;
GMAT InitSat.Epoch = '26660.00002354609';
GMAT InitSat.CoordinateSystem = EarthMJ2000Eq;
GMAT InitSat.DisplayStateType = ModifiedKeplerian;
GMAT InitSat.RadPer = 6551.999993447066;
GMAT InitSat.RadApo = 42071.00000000015;
GMAT InitSat.INC = 6.000000000000086;
GMAT InitSat.RAAN = 315.9999999999999;
GMAT InitSat.AOP = 1.207418269725733e-06;
GMAT InitSat.TA = 99.88775248425233;
GMAT InitSat.DryMass = 200;
GMAT InitSat.Cd = 2.2;
GMAT InitSat.Cr = 1.8;
GMAT InitSat.DragArea = 15;
GMAT InitSat.SRPArea = 1;

Create Spacecraft InitSat2;
GMAT InitSat2.DateFormat = A1ModJulian;
GMAT InitSat2.Epoch = '26660.00002354609';
GMAT InitSat2.CoordinateSystem = EarthMJ2000Eq;
GMAT InitSat2.DisplayStateType = ModifiedKeplerian;
GMAT InitSat2.RadPer = 6552.999993447029;
GMAT InitSat2.RadApo = 42070.99999999995;
GMAT InitSat2.INC = 6.000000000000086;
GMAT InitSat2.RAAN = 315.9999999999999;
GMAT InitSat2.AOP = 0;
GMAT InitSat2.TA = 99.88775248425245;
GMAT InitSat2.DryMass = 200;
GMAT InitSat2.Cd = 2.2;
```

```
GMAT InitSat2.Cr = 1.8;
GMAT InitSat2.DragArea = 15;
GMAT InitSat2.SRPArea = 1;
%----------------------------------------
%---------- ForceModels
%----------------------------------------

Create ForceModel EarthProp_ForceModel;
GMAT EarthProp_ForceModel.CentralBody = Earth;
GMAT EarthProp_ForceModel.PrimaryBodies = {Earth};
GMAT EarthProp_ForceModel.PointMasses = {Sun};
GMAT EarthProp_ForceModel.Drag = None;
GMAT EarthProp_ForceModel.SRP = Off;
GMAT EarthProp_ForceModel.ErrorControl = RSSStep;
GMAT EarthProp_ForceModel.GravityField.Earth.Degree = 4;
GMAT EarthProp_ForceModel.GravityField.Earth.Order = 4;
GMAT EarthProp_ForceModel.GravityField.Earth.PotentialFile = 'JGM2.cof';

Create ForceModel EarthMoonProp_ForceModel;
GMAT EarthMoonProp_ForceModel.CentralBody = Earth;
GMAT EarthMoonProp_ForceModel.PrimaryBodies = {Earth};
GMAT EarthMoonProp_ForceModel.PointMasses = {Sun, Luna};
GMAT EarthMoonProp_ForceModel.Drag = None;
GMAT EarthMoonProp_ForceModel.SRP = Off;
GMAT EarthMoonProp_ForceModel.ErrorControl = RSSStep;
GMAT EarthMoonProp_ForceModel.GravityField.Earth.Degree = 4;
GMAT EarthMoonProp_ForceModel.GravityField.Earth.Order = 4;
GMAT EarthMoonProp_ForceModel.GravityField.Earth.PotentialFile = 'JGM2.cof';

Create ForceModel MoonProp_ForceModel;
GMAT MoonProp_ForceModel.CentralBody = Luna;
GMAT MoonProp_ForceModel.PrimaryBodies = {Luna};
GMAT MoonProp_ForceModel.PointMasses = {Sun, Earth};
GMAT MoonProp_ForceModel.Drag = None;
GMAT MoonProp_ForceModel.SRP = Off;
GMAT MoonProp_ForceModel.ErrorControl = RSSStep;
GMAT MoonProp_ForceModel.GravityField.Luna.Degree = 4;
GMAT MoonProp_ForceModel.GravityField.Luna.Order = 4;
GMAT MoonProp_ForceModel.GravityField.Luna.PotentialFile = 'LP165P.cof';


%----------------------------------------
%---------- Propagators
%----------------------------------------

Create Propagator EarthProp;
GMAT EarthProp.FM = EarthProp_ForceModel;
GMAT EarthProp.Type = RungeKutta89;
GMAT EarthProp.InitialStepSize = 60;
GMAT EarthProp.Accuracy = 9.999999999999999e-12;
GMAT EarthProp.MinStep = 0.001;
GMAT EarthProp.MaxStep = 2700;
GMAT EarthProp.MaxStepAttempts = 50;

Create Propagator EarthMoonProp;
GMAT EarthMoonProp.FM = EarthMoonProp_ForceModel;
GMAT EarthMoonProp.Type = RungeKutta89;
GMAT EarthMoonProp.InitialStepSize = 60;
GMAT EarthMoonProp.Accuracy = 9.999999999999999e-12;
GMAT EarthMoonProp.MinStep = 0.001;
```

```
GMAT EarthMoonProp.MaxStep = 2700;
GMAT EarthMoonProp.MaxStepAttempts = 50;

Create Propagator MoonProp;
GMAT MoonProp.FM = MoonProp_ForceModel;
GMAT MoonProp.Type = RungeKutta89;
GMAT MoonProp.InitialStepSize = 60;
GMAT MoonProp.Accuracy = 9.999999999999999e-12;
GMAT MoonProp.MinStep = 1e-09;
GMAT MoonProp.MaxStep = 2700;
GMAT MoonProp.MaxStepAttempts = 50;

%----------------------------------------
%---------- Burns
%----------------------------------------

Create ImpulsiveBurn TOI;
GMAT TOI.Origin = Earth;
GMAT TOI.Axes = VNB;
GMAT TOI.VectorFormat = Cartesian;
GMAT TOI.Element1 = 0.7;
GMAT TOI.Element2 = 0;
GMAT TOI.Element3 = 0;

Create ImpulsiveBurn IncMod;
GMAT IncMod.Origin = Earth;
GMAT IncMod.Axes = VNB;
GMAT IncMod.VectorFormat = Cartesian;
GMAT IncMod.Element1 = 0;
GMAT IncMod.Element2 = 0;
GMAT IncMod.Element3 = 0;

Create ImpulsiveBurn LOI;
GMAT LOI.Origin = Luna;
GMAT LOI.Axes = VNB;
GMAT LOI.VectorFormat = Cartesian;
GMAT LOI.Element1 = 0;
GMAT LOI.Element2 = 0;
GMAT LOI.Element3 = 0;

Create ImpulsiveBurn Descent;
GMAT Descent.Origin = Luna;
GMAT Descent.Axes = VNB;
GMAT Descent.VectorFormat = Cartesian;
GMAT Descent.Element1 = 0;
GMAT Descent.Element2 = 0;
GMAT Descent.Element3 = 0;

Create ImpulsiveBurn DOI;
GMAT DOI.Origin = Luna;
GMAT DOI.Axes = VNB;
GMAT DOI.VectorFormat = Cartesian;
GMAT DOI.Element1 = 0;
GMAT DOI.Element2 = 0;
GMAT DOI.Element3 = 0;

Create ImpulsiveBurn Brake;
GMAT Brake.Origin = Luna;
GMAT Brake.Axes = VNB;
GMAT Brake.VectorFormat = Cartesian;
```

```
GMAT Brake.Element1 = 0;
GMAT Brake.Element2 = 0;
GMAT Brake.Element3 = 0;
%----------------------------------------
%---------- Variables, Arrays, Strings
%----------------------------------------

Create Variable RAAN RAAN1 RAAN2 AOP AOP2 INC Epoch DeltaV DeltaVearth DeltaVinc;
Create Variable DeltaVmoon DeltaVdescent DeltaVtouchdown Epoch1 Epoch2 Epoch3 CoastTime
ECC SMA TOI1;
Create Variable VmagLOI INC2 RadPer RadApo Xint Yint Zint;


%----------------------------------------
%---------- Coordinate Systems
%----------------------------------------

Create CoordinateSystem EarthMJ2000Eq;
GMAT EarthMJ2000Eq.Origin = Earth;
GMAT EarthMJ2000Eq.Axes = MJ2000Eq;
GMAT EarthMJ2000Eq.UpdateInterval = 60;
GMAT EarthMJ2000Eq.OverrideOriginInterval = false;

Create CoordinateSystem EarthMJ2000Ec;
GMAT EarthMJ2000Ec.Origin = Earth;
GMAT EarthMJ2000Ec.Axes = MJ2000Ec;
GMAT EarthMJ2000Ec.UpdateInterval = 60;
GMAT EarthMJ2000Ec.OverrideOriginInterval = false;

Create CoordinateSystem EarthFixed;
GMAT EarthFixed.Origin = Earth;
GMAT EarthFixed.Axes = BodyFixed;
GMAT EarthFixed.UpdateInterval = 60;
GMAT EarthFixed.OverrideOriginInterval = false;

Create CoordinateSystem EarthToMoon;
GMAT EarthToMoon.Origin = Luna;
GMAT EarthToMoon.Axes = ObjectReferenced;
GMAT EarthToMoon.UpdateInterval = 60;
GMAT EarthToMoon.OverrideOriginInterval = false;
GMAT EarthToMoon.XAxis = R;
GMAT EarthToMoon.ZAxis = N;
GMAT EarthToMoon.Primary = Luna;
GMAT EarthToMoon.Secondary = Earth;

Create CoordinateSystem LunaFixed;
GMAT LunaFixed.Origin = Luna;
GMAT LunaFixed.Axes = BodyFixed;
GMAT LunaFixed.UpdateInterval = 60;
GMAT LunaFixed.OverrideOriginInterval = false;


%----------------------------------------
%---------- Solvers
%----------------------------------------

Create DifferentialCorrector DC1;
GMAT DC1.ShowProgress = true;
GMAT DC1.ReportStyle = 'Normal';
GMAT DC1.TargeterTextFile = 'DifferentialCorrectorDefaultDC.data';
```

```
GMAT DC1.MaximumIterations = 200;
GMAT DC1.UseCentralDifferences = false;

%----------------------------------------
%---------- Plots and Reports
%----------------------------------------

Create OpenGLPlot EarthOGL;
GMAT EarthOGL.SolverIterations = Current;
GMAT EarthOGL.Add = {Sat, Earth, Luna};
GMAT EarthOGL.OrbitColor = [ 16777215 32768 1743054 ];
GMAT EarthOGL.TargetColor = [ 8421440 8421440 8421440 ];
GMAT EarthOGL.CoordinateSystem = EarthMJ2000Eq;
GMAT EarthOGL.ViewPointReference = Earth;
GMAT EarthOGL.ViewPointVector = [ 0 30000 30000 ];
GMAT EarthOGL.ViewDirection = Earth;
GMAT EarthOGL.ViewScaleFactor = 15;
GMAT EarthOGL.ViewUpCoordinateSystem = EarthMJ2000Eq;
GMAT EarthOGL.ViewUpAxis = Z;
GMAT EarthOGL.CelestialPlane = Off;
GMAT EarthOGL.XYPlane = Off;
GMAT EarthOGL.WireFrame = Off;
GMAT EarthOGL.Axes = On;
GMAT EarthOGL.Grid = Off;
GMAT EarthOGL.SunLine = Off;
GMAT EarthOGL.UseInitialView = On;
GMAT EarthOGL.DataCollectFrequency = 1;
GMAT EarthOGL.UpdatePlotFrequency = 50;
GMAT EarthOGL.NumPointsToRedraw = 0;
GMAT EarthOGL.ShowPlot = true;

Create OpenGLPlot EarthMoonOGL;
GMAT EarthMoonOGL.SolverIterations = Current;
GMAT EarthMoonOGL.Add = {Sat, Luna, Earth};
GMAT EarthMoonOGL.OrbitColor = [ 16777215 1743054 1743054 ];
GMAT EarthMoonOGL.TargetColor = [ 1065386048 8421440 8421440 ];
GMAT EarthMoonOGL.CoordinateSystem = EarthToMoon;
GMAT EarthMoonOGL.ViewPointReference = Luna;
GMAT EarthMoonOGL.ViewPointVector = [ 30000 0 0 ];
GMAT EarthMoonOGL.ViewDirection = Luna;
GMAT EarthMoonOGL.ViewScaleFactor = 1;
GMAT EarthMoonOGL.ViewUpCoordinateSystem = EarthToMoon;
GMAT EarthMoonOGL.ViewUpAxis = Z;
GMAT EarthMoonOGL.CelestialPlane = Off;
GMAT EarthMoonOGL.XYPlane = On;
GMAT EarthMoonOGL.WireFrame = Off;
GMAT EarthMoonOGL.Axes = On;
GMAT EarthMoonOGL.Grid = Off;
GMAT EarthMoonOGL.SunLine = Off;
GMAT EarthMoonOGL.UseInitialView = On;
GMAT EarthMoonOGL.DataCollectFrequency = 1;
GMAT EarthMoonOGL.UpdatePlotFrequency = 50;
GMAT EarthMoonOGL.NumPointsToRedraw = 0;
GMAT EarthMoonOGL.ShowPlot = true;

Create ReportFile LunarDATA;
GMAT LunarDATA.SolverIterations = Current;
GMAT LunarDATA.Filename = './output/ReportFile1.txt';
GMAT LunarDATA.Precision = 16;
GMAT LunarDATA.Add = {Sat.A1ModJulian, Sat.ElapsedDays, Sat.Luna.Latitude,
```

```
Sat.Luna.Longitude, Sat.LunaFixed.RAAN, Sat.Luna.Altitude, Sat.LunaFixed.INC};
GMAT LunarDATA.WriteHeaders = On;
GMAT LunarDATA.LeftJustify = On;
GMAT LunarDATA.ZeroFill = Off;
GMAT LunarDATA.ColumnWidth = 20;

Create OpenGLPlot DescentOGL;
GMAT DescentOGL.SolverIterations = Current;
GMAT DescentOGL.Add = {Sat, Luna};
GMAT DescentOGL.OrbitColor = [ 16777215 1743054 ];
GMAT DescentOGL.TargetColor = [ 8421440 8421440 ];
GMAT DescentOGL.CoordinateSystem = LunaFixed;
GMAT DescentOGL.ViewPointReference = Earth;
GMAT DescentOGL.ViewPointVector = [ 2000 -1000 3000 ];
GMAT DescentOGL.ViewDirection = Earth;
GMAT DescentOGL.ViewScaleFactor = 1;
GMAT DescentOGL.ViewUpCoordinateSystem = LunaFixed;
GMAT DescentOGL.ViewUpAxis = X;
GMAT DescentOGL.CelestialPlane = Off;
GMAT DescentOGL.XYPlane = Off;
GMAT DescentOGL.WireFrame = Off;
GMAT DescentOGL.Axes = On;
GMAT DescentOGL.Grid = Off;
GMAT DescentOGL.SunLine = Off;
GMAT DescentOGL.UseInitialView = On;
GMAT DescentOGL.DataCollectFrequency = 1;
GMAT DescentOGL.UpdatePlotFrequency = 50;
GMAT DescentOGL.NumPointsToRedraw = 0;
GMAT DescentOGL.ShowPlot = true;

Create ReportFile EarthDATA;
GMAT EarthDATA.SolverIterations = Current;
GMAT EarthDATA.Filename = './output/ReportFile2.txt';
GMAT EarthDATA.Precision = 16;
GMAT EarthDATA.Add = {Sat.A1ModJulian, Sat.EarthToMoon.X, Sat.EarthToMoon.Y,
Sat.EarthToMoon.Z, Sat.Earth.Altitude, Sat.EarthMJ2000Eq.AOP, AOP, Sat.EarthMJ2000Eq.INC,
INC, Sat.EarthMJ2000Eq.RAAN, RAAN};
GMAT EarthDATA.WriteHeaders = On;
GMAT EarthDATA.LeftJustify = On;
GMAT EarthDATA.ZeroFill = Off;
GMAT EarthDATA.ColumnWidth = 20;

Create ReportFile VariablesDATA;
GMAT VariablesDATA.SolverIterations = Current;
GMAT VariablesDATA.Filename = './output/ReportFile3.txt';
GMAT VariablesDATA.Precision = 16;
GMAT VariablesDATA.Add = {Sat.A1ModJulian, DeltaV, DeltaVearth, DeltaVmoon, DeltaVdescent,
DeltaVtouchdown, DeltaVinc, AOP2, RAAN2, INC2};
GMAT VariablesDATA.WriteHeaders = On;
GMAT VariablesDATA.LeftJustify = On;
GMAT VariablesDATA.ZeroFill = Off;
GMAT VariablesDATA.ColumnWidth = 20;
```

```
%----------------------------------------
%---------- Mission Sequence
%----------------------------------------

%************** Sequence 0 **************

% sequences 0 to 2 are not plotted in order to have the final trajectory only
Toggle EarthMoonOGL EarthOGL DescentOGL Off;

% propagate spacecraft until perigee
Propagate EarthProp(Sat) {Sat.Earth.Periapsis};

% initialization of variables
GMAT RAAN = Sat.RAAN;
GMAT AOP = Sat.AOP;
GMAT INC = Sat.INC;
GMAT InitSat = Sat;

%************** Sequence 1 **************
% First translunar kick loop
%****************************************

% parking orbit target orientation and TLI maneuver to align line of apsides with moon at lunar
encounter
Target DC1;
   Vary DC1(TOI.Element1 = 0.7, {Perturbation = 0.0001, MaxStep = 0.01, Lower = -3, Upper = 5});
   Vary DC1(Sat.AOP = AOP, {Perturbation = 0.2, MaxStep = 15, Lower = -360, Upper = 360});
   Vary DC1(Sat.RAAN = RAAN, {Perturbation = 0.2, MaxStep = 15, Lower = -360, Upper = 360});
   Vary DC1(Sat.INC = INC, {Perturbation = 0.2, MaxStep = 15, Lower = -720, Upper = 720});

   % save RAAN, AOP and INC values for later use
   GMAT RAAN = Sat.RAAN;
   GMAT AOP = Sat.AOP;
   GMAT INC = Sat.INC;

   Maneuver TOI(Sat);
   Propagate EarthProp(Sat) {Sat.Earth.Apoapsis};

   Achieve DC1(Sat.EarthToMoon.X = 0, {Tolerance = 0.1});
   Achieve DC1(Sat.EarthToMoon.Y = -8000, {Tolerance = 0.1});
   Achieve DC1(Sat.EarthToMoon.Z = 0, {Tolerance = 0.1});
EndTarget;  % For targeter DefaultDC


%************** Sequence 2 **************
% Second translunar kick loop
%****************************************

% save current orbital elements and epoch in Initsat
GMAT Sat = InitSat;

Target DC1;
   Vary DC1(TOI.Element1 = 0.7, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -3, Upper = 5});
   Vary DC1(Sat.AOP = AOP, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper =
9.999999e300});
   Vary DC1(Sat.RAAN = RAAN, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper
= 9.999999e300});
   Vary DC1(Sat.INC = INC, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper =
9.999999e300});
```

```
   Maneuver TOI(Sat);
   Propagate EarthMoonProp(Sat) {Sat.ElapsedDays = 2};
   Propagate MoonProp(Sat) {Sat.Luna.Periapsis};

   Achieve DC1(Sat.Luna.Altitude = 100, {Tolerance = 1.0});

   GMAT Epoch3 = Sat.A1ModJulian;
   GMAT InitSat2 = Sat;
EndTarget;  % For targeter DefaultDC


%************** Sequence 3 **************
% Targeting a point on the moon's sphere of incluence
%****************************************

% trajectory is plotted
Toggle EarthMoonOGL EarthOGL DescentOGL On;

% back to conditions before lunar transfer
GMAT Sat = InitSat;

Target DC1;
   Vary DC1(TOI.Element1 = 0.7, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -3, Upper = 5});
   Vary DC1(Sat.AOP = AOP, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper =
9.999999e300});
   Vary DC1(Sat.RAAN = RAAN, {Perturbation = 0.1, MaxStep = 15, Lower = -9.999999e300, Upper
= 9.999999e300});

   % save RAAN and AOP values for later use
   GMAT RAAN = Sat.RAAN;
   GMAT AOP = Sat.AOP;

   Maneuver TOI(Sat);
   Propagate EarthMoonProp(Sat) {Sat.A1ModJulian = 26663.19948121};

   Achieve DC1(Sat.EarthToMoon.X = 66620, {Tolerance = 0.1});
   Achieve DC1(Sat.EarthToMoon.Y = -141354, {Tolerance = 0.1});
EndTarget;  % For targeter DefaultDC

% calculation of Delta-V for lunar transfer
GMAT DeltaVearth = sqrt( TOI.V^2 + TOI.N^2 + TOI.B^2 );

% initialization of variables
GMAT Xint = Sat.EarthToMoon.X;
GMAT Yint = Sat.EarthToMoon.Y;
GMAT Zint = Sat.EarthToMoon.Z;
GMAT Epoch = Sat.A1ModJulian;


%************** Sequence 4 **************
% Back propagation from lunar insertion point to previous targeted point on moon's sphere of
influence
%****************************************

GMAT Sat = InitSat2;

Target DC1;
   Vary DC1(TOI.V = 0, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -5, Upper = 5});
   Vary DC1(TOI.N = 0, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -5, Upper = 5});
   Vary DC1(TOI.B = 0, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -5, Upper = 5});
```

```
  Maneuver TOI(Sat);
  Propagate BackProp MoonProp(Sat) {Sat.A1ModJulian = 26663.19948121};

  Achieve DC1(Sat.EarthToMoon.X = Xint, {Tolerance = 0.1});
  Achieve DC1(Sat.EarthToMoon.Y = Yint, {Tolerance = 0.1});
  Achieve DC1(Sat.EarthToMoon.Z = Zint, {Tolerance = 0.1});
EndTarget;  % For targeter DefaultDC

% calculation of Delta-V for lunar transfer
GMAT DeltaVinc = sqrt( TOI.V^2 + TOI.N^2 + TOI.B^2 );


%************** Sequence 5 **************
% Lunar capture
%****************************************

% back to conditions at the end of lunar transfer
GMAT Sat = InitSat2;

% lunar capture kick
Target DC1;
  Vary DC1(LOI.V = -0.4, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -4, Upper = 3.14});

  Maneuver LOI(Sat);
  Propagate MoonProp(Sat) {Sat.Luna.Apoapsis};

  Achieve DC1(Sat.Luna.ECC = 0, {Tolerance = 0.0001});
EndTarget;  % For targeter DefaultDC

% calculation of Delta-V for lunar capture
GMAT DeltaVmoon = sqrt( LOI.V^2 + LOI.N^2 + LOI.B^2 );


%************** Sequence 6 **************
% Perigee lowering to 15 km
%****************************************

% propagation until longitude = 180° before descent
Propagate MoonProp(Sat) {Sat.Luna.Longitude = 180};

% kick for lowering from 100 km
GMAT VmagLOI = Sat.LunaFixed.VMAG;
GMAT INC = Sat.LunaFixed.INC;

Target DC1;
  Vary DC1(DOI.V = -0.4, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -4, Upper = 4});
  Maneuver DOI(Sat);
  Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
  Achieve DC1(Sat.Luna.Altitude = 15, {Tolerance = 0.1});
  Achieve DC1(Sat.LunaFixed.INC = INC, {Tolerance = 0.1});
EndTarget;  % For targeter DC1

% kick for orbit insertion at 15km
Target DC1;
  Vary DC1(Brake.V = 0.4, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -4, Upper = 4});
  Maneuver Brake(Sat);
  Achieve DC1(Sat.Luna.ECC = 0, {Tolerance = 0.0001});
EndTarget;  % For targeter DC1
```

```
% calculation of Delta-V for lowering kick
GMAT DeltaVdescent = sqrt( DOI.V^2 + DOI.N^2 + DOI.B^2 ) + sqrt( Brake.V^2 + Brake.N^2 +
Brake.B^2 );


%************** Sequence 7 **************
% Direct landing phase
%****************************************

% definition of Epoch1 variable for descent time calculation
GMAT Epoch1 = Sat.A1ModJulian;

% incrementation of Epoch with descent time in order to have the landed Epoch
GMAT Epoch1 = Epoch1  + 0.01157407402;

Target DC1;
   Vary DC1(Descent.V = -0.1, {Perturbation = 0.0001, MaxStep = 0.5, Lower = -100, Upper = 100});

   Maneuver Descent(Sat);
   Propagate MoonProp(Sat) {Sat.Luna.Longitude = 15};

   Achieve DC1(Sat.Luna.Altitude = 0, {Tolerance = 0.1});
   Achieve DC1(Sat.A1ModJulian = Epoch1, {Tolerance = 0.1});
   Achieve DC1(Sat.Luna.Latitude = Sat.Luna.Latitude, {Tolerance = 0.5});
EndTarget;  % For targeter DC1

% calculation of Delta-V for descent
GMAT DeltaVtouchdown = sqrt( Descent.V^2 + Descent.N^2 + Descent.B^2 ) + VmagLOI;

% calculation for Total Delta-V
GMAT DeltaV = DeltaVearth + DeltaVinc+ DeltaVmoon + DeltaVdescent + DeltaVtouchdown;
Report VariablesDATA Sat.A1ModJulian DeltaV DeltaVearth DeltaVinc DeltaVmoon DeltaVdescent
DeltaVtouchdown;

%************** Sequence 8 **************
% Propagating 2 GTO orbits at initial conditions
%****************************************

% 2 GTO orbits

% setting the spacecraft at initial conditions of epoch, RAAN and AOP
GMAT Sat = InitSat;
GMAT Sat.RAAN = RAAN;
GMAT Sat.AOP = AOP;

% propagating 2 orbits
Propagate BackProp EarthProp(Sat) {Sat.Earth.Periapsis};
Propagate BackProp EarthProp(Sat) {Sat.Earth.Periapsis};
```

Sequences for Luna 17 landing

The following sequence are to be replaced in the earlier codes in order to obtain the Luna 17 landing.

```
%************** Sequence 6 **************
%Inclination change until 35°
%****************************************

% double propagation until descending node in the equatorial plane
Propagate MoonProp(Sat) {Sat.Luna.Latitude = 0};
Propagate MoonProp(Sat) {Sat.Luna.Latitude = 0};

% saving values semi-major axis and eccentricity values
GMAT SMA = Sat.Luna.SMA;
GMAT ECC = Sat.Luna.ECC;

Target DC1;
   Vary DC1(LunarInc.V = 0, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -3, Upper = 5});
   Vary DC1(LunarInc.N = -0.7, {Perturbation = 0.0001, MaxStep = 0.5, Lower = -3, Upper = 5});
   Vary DC1(LunarInc.B = 0, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -5, Upper = 5});

   Maneuver LunarInc(Sat);
   Propagate MoonProp(Sat) {Sat.Luna.Periapsis};

   Achieve DC1(Sat.LunaFixed.INC = 35, {Tolerance = 0.1});
   Achieve DC1(Sat.Luna.SMA = SMA, {Tolerance = 0.01});
   Achieve DC1(Sat.Luna.ECC = ECC, {Tolerance = 0.1});
EndTarget;  % For targeter DefaultDC

% calculation of Delta-V for lunar inclination change
GMAT DeltaVmoon = DeltaVmoon + sqrt( LunarInc.V^2 + LunarInc.N^2 + LunarInc.B^2 );


%************** Sequence 7 **************
% Perigee lowering to 15 km
%****************************************

% 4 orbit (LEO) and 8 orbit (GTO) propagation and positioning at longitude = 120° before de-orbiting
Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
%Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
%Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
%Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
%Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
Propagate MoonProp(Sat) {Sat.Luna.Longitude = 120};

% kick for lowering from 100 km

GMAT VmagLOI = Sat.LunaFixed.VMAG;
GMAT INC = Sat.LunaFixed.INC;

Target DC1;
   Vary DC1(DOI.V = -0.4, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -4, Upper = 4});
   Maneuver DOI(Sat);
   Propagate MoonProp(Sat) {Sat.Luna.Periapsis};
   Achieve DC1(Sat.Luna.Altitude = 15, {Tolerance = 0.1});
   Achieve DC1(Sat.LunaFixed.INC = INC, {Tolerance = 0.1});
```

```
EndTarget;  % For targeter DC1
 % kick for orbit insertion at 15km
Target DC1;
   Vary DC1(Brake.V = 0.4, {Perturbation = 0.0001, MaxStep = 0.2, Lower = -4, Upper = 4});
   Maneuver Brake(Sat);
   Achieve DC1(Sat.Luna.ECC = 0, {Tolerance = 0.0001});
EndTarget;  % For targeter DC1

% calculation of Delta-V for lowering kick
GMAT DeltaVdescent = sqrt( DOI.V^2 + DOI.N^2 + DOI.B^2 ) + sqrt( Brake.V^2 + Brake.N^2 +
Brake.B^2 );


%************** Sequence 8 **************
% Landing phase : Luna 17 site (35°N/35°W)
%***************************************

% definition of Epoch1 variable for descent time calculation
GMAT Epoch1 = Sat.A1ModJulian;

% incrementation of Epoch with descent time in order to have the landed Epoch
GMAT Epoch1 = Epoch1  + 0.01157407402; % 1000 seconds

Target DC1;
   Vary DC1(Descent.V = -0.1, {Perturbation = 0.0001, MaxStep = 0.5, Lower = -100, Upper = 100});
   Vary DC1(Descent.N = -0.1, {Perturbation = 0.0001, MaxStep = 0.5, Lower = -100, Upper = 100});

   Maneuver Descent(Sat);
   Propagate MoonProp(Sat) {Sat.Luna.Longitude = -35};

   Achieve DC1(Sat.Luna.Altitude = 0, {Tolerance = 0.1});
   Achieve DC1(Sat.A1ModJulian = Epoch1, {Tolerance = 0.1});
   Achieve DC1(Sat.Luna.Latitude = 35, {Tolerance = 0.5});
EndTarget;  % For targeter DC1

% calculation of Delta-V for descent
GMAT DeltaVtouchdown = sqrt( Descent.V^2 + Descent.N^2 + Descent.B^2 ) + VmagLOI;

% calculation for Total Delta-V
GMAT DeltaV = DeltaVearth + DeltaVinc+ DeltaVmoon + DeltaVdescent + DeltaVtouchdown;

Report VariablesDATA Sat.A1ModJulian DeltaV DeltaVearth DeltaVinc DeltaVmoon DeltaVdescent
DeltaVtouchdown;

%************** Sequence 9 **************
% Propagating 2 GTO orbits at initial conditions
%***************************************

% 2 GTO orbits

% setting the spacecraft at initial conditions of epoch, RAAN and AOP
GMAT Sat = InitSat;
GMAT Sat.RAAN = RAAN;
GMAT Sat.AOP = AOP;

% propagating 2 orbits
Propagate BackProp EarthProp(Sat) {Sat.Earth.Periapsis};
Propagate BackProp EarthProp(Sat) {Sat.Earth.Periapsis};
```