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Lunar Transfer Trajectories

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Acronyms and Abbreviations

AMSAT	Radio Amateur Satellite Corporation
ARP	Argument Of Perigee
FDS	Flight Dynamic System
GTO	Geostationary Transfer Orbit
LEO	Lunar Exploration Orbiter
LTO	Lunar Transfer Orbit
RAAN	Right Ascension of Ascending Node
RIT	Radiofrequency Ion Thruster
TCM	Trajectory Correction Manoeuvre
TLI	Trans Lunar Injection
WSB	Weak Stability Boundary

Zusammenfassung

Nach einer kurzen Darstellung der Bahngeometrie im Erde-Mond System erfolgt ein Vergleich verschiedener Transfermöglichkeiten. Als Grundlage für die AMSAT-Mond Mission folgt daraus ein WSB Transfer. Nach einem kurzen Überblick über die Implementierungsmöglichkeiten werden mit einem Ariane 5 GTO als Startorbit und einem annähernd polaren 100 x 100 km Zielorbit am Mond die Rahmenbedingungen für den AMSAT-Mond Transfer gesetzt. Für die maximal zulässige Raumfahrzeug Gesamtmasse von 650 kg ergibt sich eine Transferzeit von 130 Tagen bei einem ΔV von ca. 1800 m/s (inklusive Sicherheit).

Unter Berücksichtigung von Manöverfehlern des 400 N Triebwerks beim TLI von ±1% (unkalibriert 1-2%, kalibriert <1% möglich) ergibt sich eine Absenkung des Apogäums in der WSB Region um 365000 km, oder eine Beschleunigung auf Fluchtgeschwindigkeit. Um dies zu vermeiden und gleichzeitig die hohe Anzahl an Van-Allen-Gürtel Passagen zu reduzieren, die sich bei einer bis zu 27-tägigen Wartezeit im GTO ergeben würden, werden sogenannte Phasing Orbits eingeführt. Diese ergeben sich durch Aufteilung des TLI in drei einzelne Manöver deren Wert mit jedem Manöver kleiner wird. Dadurch reduziert sich der Positionsfehler im Apogäum der Transferbahn auf -52000 km oder +60000 km und die Anzahl der Van-Allen-Gürtel Passagen auf elf (drei GTO Umläufe für Bahnbestimmung und Manöverprobe). Unter Berücksichtigung von jeweils drei Bahnkorrekturmanövern auf dem Weg zur WSB und von der WSB zum Mond sollte der Transfer somit möglich sein.

Für die Sichtbarkeit während des Transfers wurden Berechnungen mit der Bodenstation Weilheim durchgeführt. Für diese Station wäre in den Phasing Orbits und dem anschließenden Transfer ein täglicher Kontakt mit mindestens 9-10 Stunden Kontaktzeit möglich. Die maximale Elevation dieser Passagen liegt bei ca. 40°. Allein für die LEOP Phase und die Zeit im GTO muss ein erweitertes Bodenstationsnetzwerk genutzt werden, welches noch genauer zu spezifizieren ist.

Die Ergebnisse des chemischen Transferszenarios werden im Anschluss einem elektrischen Transfer gegenübergestellt. Die Ergebnisse wurden mit der Bahnoptimierungssoftware InTrance berechnet. Neben den Änderungen am Missionszenario aufgrund des elektrischen Transfers werden auch die einzelnen Subsystemveränderungen des Raumfahrzeugs beschrieben. Das Ergebnis ist ein 152 Tage Transfer mit Abweichungen vom Zielorbit von 5000 km und 14 m/s Relativgeschwindigkeit. Das ΔV beträgt 5633 m/s, für welches aufgrund des hohen spezifischen Impulses von 3714 s jedoch nur eine Treibstoffmasse von 82 kg benötigt wird. Mit 20 % Sicherheit ergibt sich somit bei 98 kg ein Treibstoffmassenanteil von 15% an der Gesamtmasse von 647.6 kg. Trotz des größeren Massenbedarfs einzelner Subsysteme aufgrund des elektrischen Transferszenarios erhöht sich die Nutzlast von 14.5 kg auf 51 kg.

Scope

Within the frame of the AMSAT-Moon Phase 0 study, an extended analysis of possible transfer trajectories to the moon has been performed. This covers direct transfers, like Hohmann or Bi-Elliptical, and indirect transfers using piggy-back launches on an Ariane 5. After a brief description of the geometrical problem on how to go to the Moon, the different transfers will be described and the advantages and disadvantages will be outlined. The sections on Hohmann and Bi-elliptical transfer result mainly from literature research with short ΔV estimations and additional considerations, while the piggy-back scenarios are the results of the AMSAT mission analysis.

Concerning the chosen Weak Stability Boundary Transfer for the AMSAT mission a manoeuvre error analysis is performed added by a visibility analysis for the transfer. Additionally an electrically propelled transfer scenario from GTO is analyzed, including a complete spacecraft bus system redesign with the chemical spacecraft bus as baseline.

1. The Moon

1.1 Orbit Shape

With the perigee and apogee altitude listed in Table 1 the mean semimajor axis results to a = 384000 km. From the large difference of about 10% between perigee and apogee distance a larger eccentricity compared to other circular orbits results for the lunar orbit to e = 0.0554. Furthermore the sidereal and synodic periods are given in Table 1, as well as the common center of mass. As this still lies below Earth surface, the two bodies are treated as a planet-satellite system, instead of a double-planet system

Parameter	Abbr.	Unit	Value
Perigee	r_p	[km]	363104
Apogee	r_a	[km]	405696
Mean Semimajor Axis	a	[km]	384400
Eccentricity	e		0.0554
Siderial Period	T_{sid}	[d]	27.321
Synodic Period	T_{syn}	[d]	29.530
Mass Moon	m_{Moon}	[kg]	7.3477×10^{22}
Mass Earth	M_{Earth}	[kg]	5.9736×10^{24}
Common Centre of Mass	R	[km]	4670.78

Table 1 Lunar orbit parameters

1.2 Inclination

The geometry of the different inclinations of orbital plane, rotation axis, ecliptic, etc between Moon and Earth is shown in Figure 1.

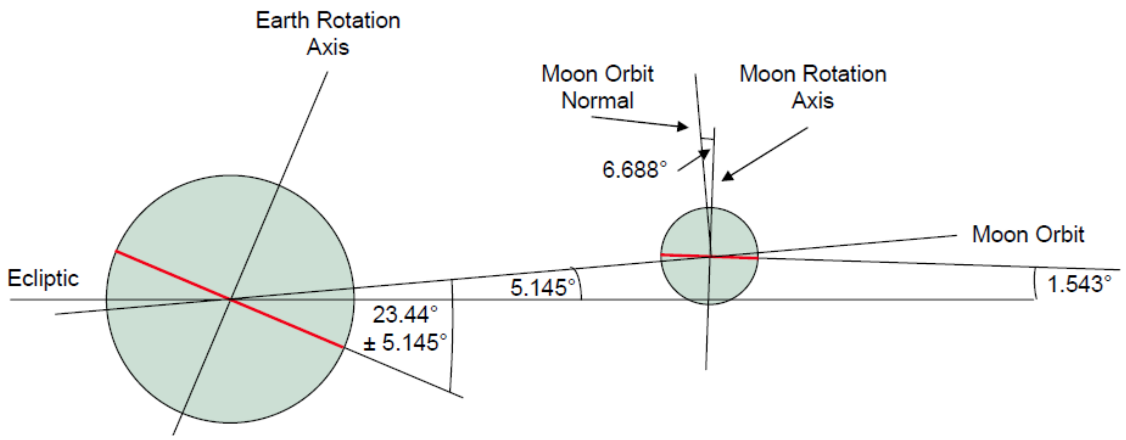


Figure 1 Earth-Moon system geometry

The mean inclination of the lunar orbit to the ecliptic is 5.145°. The Moon's rotation axis is inclined to Moon's orbital plane normal vector by 6.688°. As the precession of the rotation axis and the orbital plane are having the same rate (only 180° out of phase), the angle between the ecliptic and the lunar equator has a constant value of 1.543°.

The angle enclosing the equatorial plane of the Earth and the orbital plane of the Moon varies within a range of 23.44° ± 5.145° as precession is not in phase for Earth rotation axis and Moon orbital plane. Hence the angle changes with the precession period of the orbital nodes of 18.6 years (see Section 1.3). The variation is plotted in Figure 2 showing a maximum of 28.58° in early 2025 and a minimum of 18.29° in late 2015.

Concluding from those findings, depending on the launch date and launch site, the inclination change to reach a lunar orbit can change within approximately 10° possibly resulting in a huge ΔV difference.

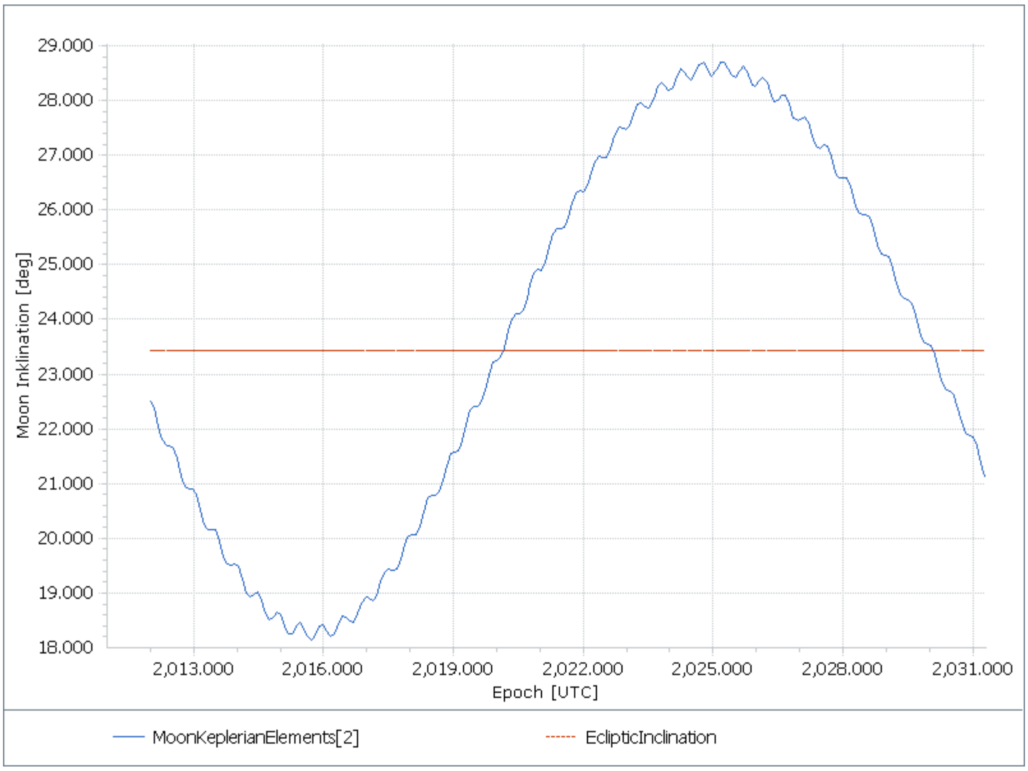


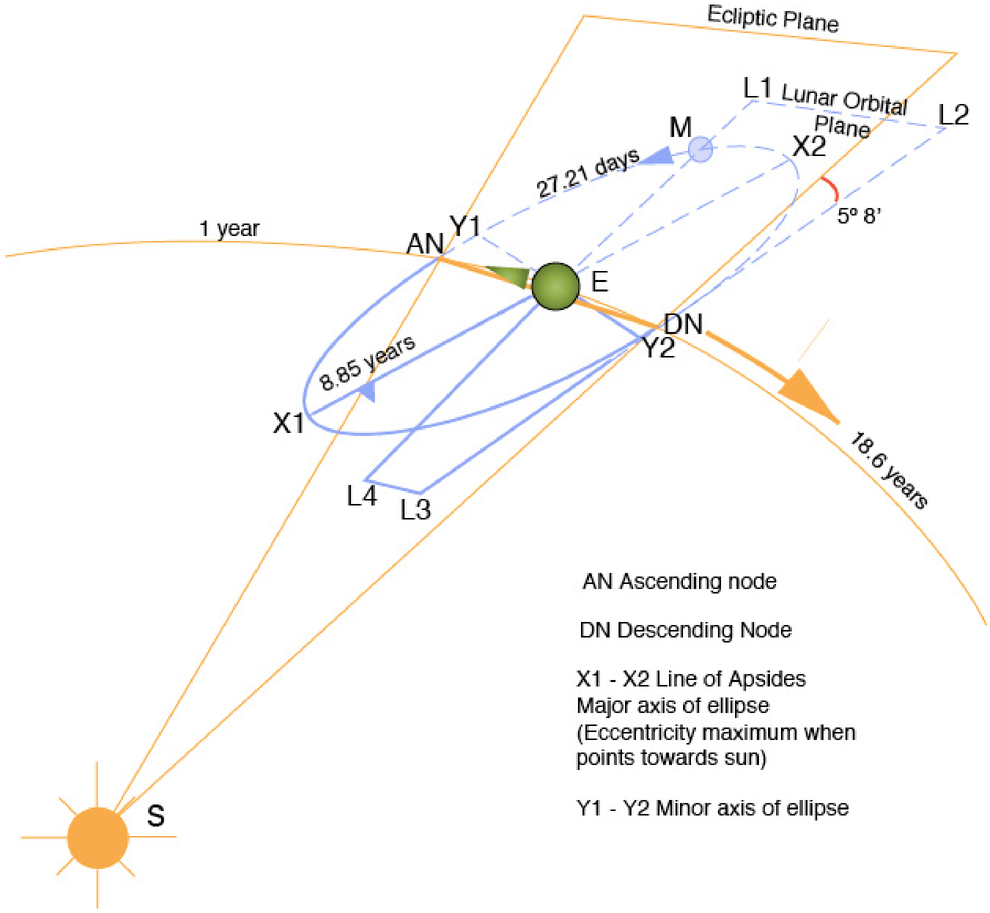
Figure 2 Lunar orbit inclination to Earth equator

1.3 Nodes

The line of Nodes of the lunar orbit has a retrograde motion and rotates clockwise (viewed form celestial north) along the ecliptic with a rate of 19°21'/year (Figure 3). Hence the nutation period amounts to 18.6 years [1].

1.4 Line of Apsides

The line of apsides on the other hand has a progressive motion, thus moving counterclockwise in the plane of the moon's orbit, completing one revolution within 8.85 years [1] (Figure 3).



AN Ascending node

DN Descending Node

X1 - X2 Line of Apsides
Major axis of ellipse
(Eccentricity maximum when points towards sun)

Y1 - Y2 Minor axis of ellipse

Figure 3 Precession of lunar orbit

2. Chemical Transfer Scenarios

The following analysis considers different transfer strategies from Earth to Moon. Lunar Orbit Insertion (LOI) is, however, not analyzed in detail.

2.1 Direct Transfer

This is the classical transfer scenario (Figure 4) used for all Luna and Apollo missions from 1960 to 1980 and is in the lowest ΔV-consuming way equal to a Hohmann transfer.

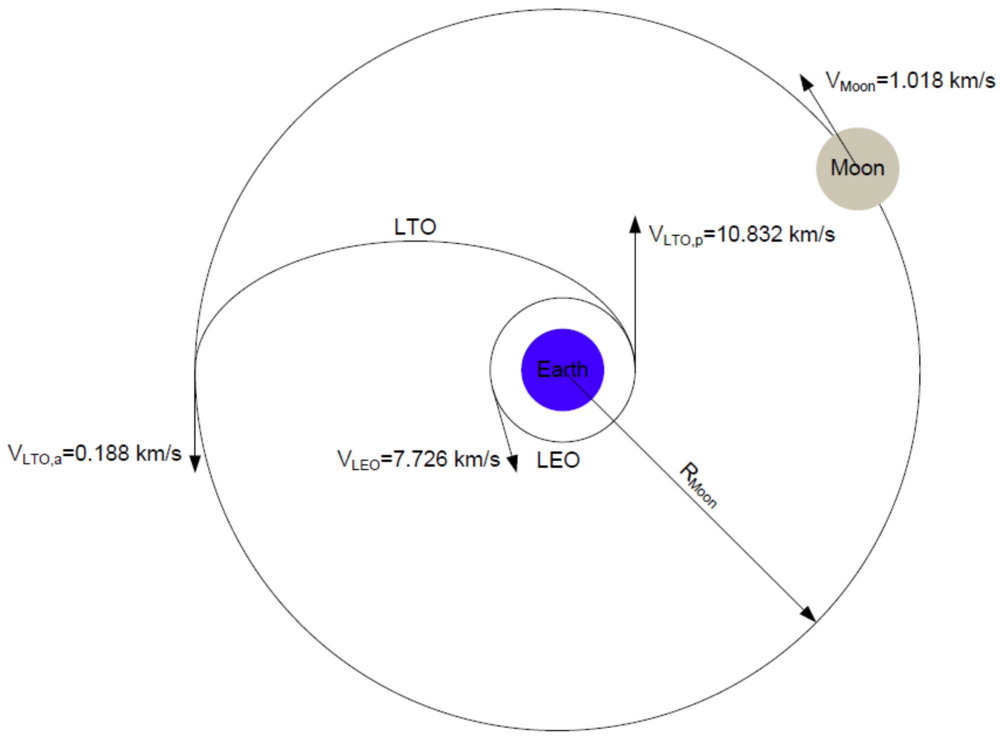


Figure 4 Hohmann transfer to Moon (Velocities for 300 km LEO and 384400 km circular Moon orbit)

The lunar probe is injected into a circular LEO of roughly 300 km altitude. Thus the ΔV_1 for Trans Lunar Injection (TLI) amounts from the differential velocity of the circular 300 km LEO and the perigee velocity of the transfer ellipse with the apogee at 384400 km to 3.106 km/s.

$$\Delta V_1 = V_{LTO,p} - V_{c,LEO} = \sqrt{\mu \cdot \left(\frac{2}{r_{LTO,p}} - \frac{1}{a} \right)} - \sqrt{\frac{\mu}{r_{LEO}}} \qquad (1)$$

It has to be mentioned that the ΔV_1 depends on the LTO apogee, which varies with the lunar orbit radius and thus the time of arrival giving the current lunar distance from Earth. However the influence is rather small such that the ΔV_1 varies in a range of ±5 m/s. The influence of the LEO altitude in comparison is larger, leading to a variation of approximately ±25 m/s for ±100 km altitude difference.

This first manoeuvre to inject the spacecraft on a trajectory towards the Moon can also be applied by the launch vehicle. Different launchers like Ariane 5, Soyuz or Dnepr consider C3=0 injections, almost equal to direct lunar injection.

For lunar capture another manoeuvre (ΔV_2) has to be applied in the periselene. This manoeuvre results again from the difference of the Moon's orbital velocity and the spacecraft velocity in the LTO apogee to 0.83 km/s.

$$\Delta V_2 = V_{Moon} - V_{LTO,a} = \sqrt{\frac{\mu}{r_{Moon}}} - \sqrt{\mu \cdot \left(\frac{2}{r_{LTO,a}} - \frac{1}{a} \right)} \tag{2}$$

Again the manoeuvre is affected by the current Earth-Moon distance, defining the LTO apogee height, thus varying between 813 and 849 m/s. Hence the apogee height has a larger impact on ΔV_2 of approximately ±19 m/s.

The overall ΔV demand for the direct transfer hence results to 3.936 km/s. Both manoeuvres could also be split up in a number of smaller manoeuvres to compensate for manoeuvre execution errors. Those occur mainly due to uncalibrated thrusters. Thus by increasing the number of manoeuvres also the error for the next manoeuvre execution can be reduced. Typically execution errors for an apogee engine lie in a range of 5% for the first manoeuvre, which can then be reduced by calibration to 1%.

The typical transfer time for the scenario presented in Figure 4, results from the half period of the elliptical transfer orbit to 4.58 days.

$$T = 2\Pi \sqrt{\frac{a^3}{\mu}} \tag{3}$$

To reduce transfer time the apogee of the LTO could be increased by a slightly higher ΔV_1. To achieve a 434400 km apogee for example, which is equal to a 50000 km higher apogee compared to the mean semi major axis apogee height, the increment would amount to 11 m/s. Thus arrival at the lunar orbit would occur earlier as the long transfer time due to low spacecraft velocity in the vicinity of LTO apogee would be spared. Hence transfer times can range between 2-5 days. The disadvantage of the shorter transfer time is of course the higher ΔV_2, which has to be applied due to higher spacecraft velocity and non parallel velocity vectors.

The launch window for the direct transfer depends on the angular difference between the launch site latitude λ and the lunar declination δ (w.r.t. Earth equator). As mentioned above, the declination can range within δ = 23.44° ± 5.145°.

As long as λ > δ the RAAN can be adjusted by the launch time within 12 hours such that the nodal line of the LEO points in the same direction as the right ascension of the Moon. By proper selection of the LTO argument of perigee (ARP), which is equal to a certain LEO true anomaly, and thus the timing of TLI manoeuvre, the declination of the Moon can be adjusted. According to the selected true anomaly again the RAAN has to be corrected. Thus the spacecraft can be launched towards Moon twice per day. The lunar right ascension and declination have of course to be targeted approximately five days in advance according to the duration of the Hohmann transfer.

If λ < δ the declination cannot be adjusted as no LEO orbital plane intersection with the Moon exists. Thus the launch window of twice per day holds only true for all right ascensions during one lunar orbit revolution, for which the declination is smaller than the launch site latitude.

For the worst case lunar rendezvous is only possible in the nodes of the lunar orbital plane and hence twice per month. For other rendezvous right ascensions a plane change manoeuvre is required.

As the maximum Moon declination is 29°, launches from higher latitudes are preferable for the direct transfer, e.g. Cape Canaveral or Baikonur.

2.2 Indirect Transfers (Piggy-back launches)

To save launch costs, the launch can be performed using the structure for auxiliary payloads (ASAP) of Ariane 5, injecting the prime- and the co-passenger into a GTO. One major disadvantage of this constellation is the difference in inclination of 18° to 29° between Earth Equator and Lunar orbit. As mentioned above rendezvous with Moon in such cases is only possible twice a month within the lunar orbit nodes. Otherwise an inclination change manoeuvre is required. To minimize the ΔV costs for the required plane change different indirect transfer scenarios are developed.

2.2.1 Short Transfer from GTO

Usually the moon can only be reached by direct lunar transfer orbit from GTO without a plane change, if the line of nodes of the GTO and those of the Moon orbit coincide. The line of nodes, however, is almost similar to line of apsides of the standard Ariane 5 GTO (w_p = 178°). This constellation occurs due to the fact that satellites launched into GTO require short eclipse times for power reasons, which is why the GTO perigee is placed in the center of the Earth shadow. By this illumination condition the Sun-Earth line coincides with the apsidal and nodal line of the GTO. As described above the direct transfer is then only possible if the nodal line of the Moon lies along the Sun-Earth line, which occurs just twice per year due to rotation of Earth around the Sun. Hence the launch window of the direct transfer is very much restricted and in case of being co-passenger unrealistic.

According to [2] one possible solution to this problem is a mid-course correction manoeuvre for the inclination change, which can be applied at the border of the moon's sphere of influence at 66000 km from Moon. An example of such a transfer is shown in Figure 5. But still the waiting time for the Moon to arrive at its node can amount up to one lunar month.

In order to avoid a large manoeuvre for inclination correction, the declination of the Moon is the major parameter to decide on the feasibility of such a strategy. The declination, however, is directly related to the nodal line angular difference ΔΩ between GTO and Moon. An appropriate ΔΩ would lie in a range of 134° < ΔΩ < 207° or 315° < ΔΩ < 29° [2].

The overall ΔV of this transfer type is equal to the Hohmann transfer, added by the mid-course manoeuvre ΔV. With the angular difference from the example in Figure 5 of ΔΩ = 18° the inclination change manoeuvre ΔV shall be estimated. Assuming a GTO inclination of i_{GTO} = 7° (standard Ariane 5) and a mean lunar orbit inclination of i_M = 24° to Earth's equator, the inclination difference amounts to Δi = 17°. This inclination difference is equal to the maximum declination δ achieved at an angular difference of ΔΩ = 90°. For the angular difference of ΔΩ = 18° in the example the declination is exactly one fifth of the maximum declination and hence δ = 3.4°. For calculating the necessary ΔV_i the following equation can be used.

$$\Delta V_i = 2 \cdot V \cdot \sin\left(\frac{i}{2}\right) \tag{4}$$

Applying the inclination change at the boarder of the lunar sphere of influence at 66000 km from the Moon, the velocity on the ellipse can be calculated to 657.6 m/s and hence the inclination change results to ΔV_i = 39 m/s. It has to be noted that this value represents the pure plane change.

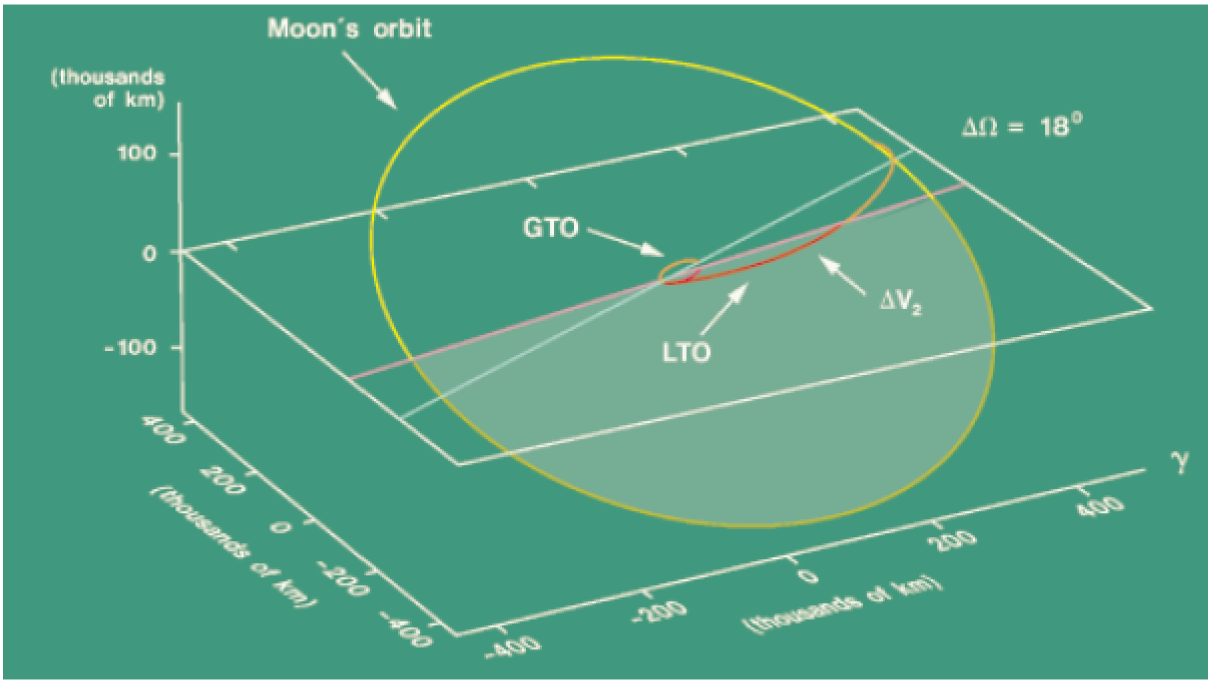


Figure 5 Direct transfer from GTO including mid-course manoeuvre [3]

2.2.2 Long Transfer from GTO (Bi-elliptic Transfer)

In case of a large node difference between GTO and Moon (29° < ΔΩ < 134° or 207° < ΔΩ < 315° [2]) the ΔV demand for plane change manoeuvres exceeds reasonable barriers. Thus a strategy has to be found to reduce the ΔV cost of the plane change.

As velocity changes are less costly for low spacecraft velocities the apogee is the preferable place for such a manoeuvre. Furthermore, if the apogee distance of the orbit is increased the spacecraft velocity is even lower, which in turn reduces the overall ΔV costs further.

Increasing the apogee of the LTO to a distance of 1 Mio km requires only a slightly higher ΔV of 58 m/s at TLI compared to an apogee at mean Moon distance. Performing the plane change at the apogee, the return leg can be selected such that the lunar rendezvous is performed. An example of such a transfer is shown in Figure 6, for a node difference of ΔΩ = 90°. Assuming again an inclination difference of 17° the overall plane change ΔV_i, performed in the LTO apogee at 1 Mio km, amounts to 21 m/s. If such an inclination difference would be compensated with a midcourse manoeuvre within a short transfer the plane change ΔV_i would amount to 194.4 m/s. Thus although the TLI manoeuvre is slightly higher, the overall ΔV is much lower. In the example 115 m/s could be spared.

The desired reduction of plane change costs leads however to a heavily increased flight time of 50 days and up to one month is required for the Moon to be present at spacecraft arrival [3]. The arrival conditions are similar to the direct transfer, although arrival now takes place closer to LTO perigee. Thus the spacecraft has a higher velocity such that the final ΔV is reduced.

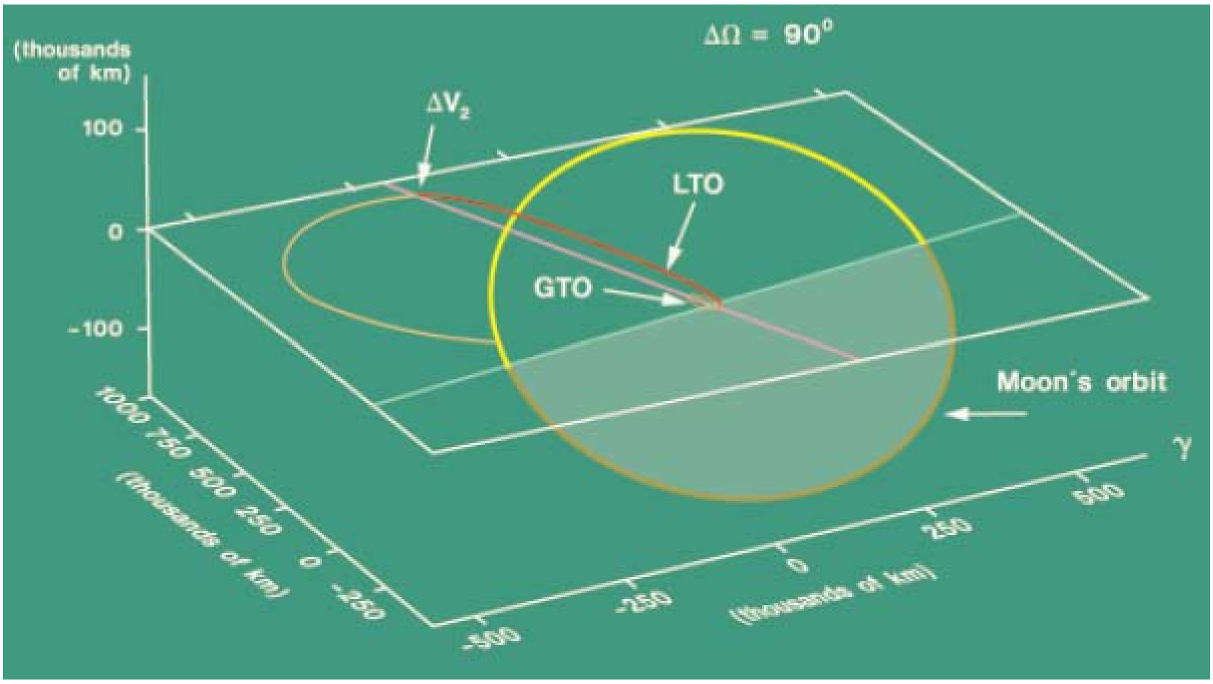


Figure 6 Bi-elliptic transfer from GTO [3]

Hence being launched Piggy-back into a GTO at a random launch time, a transfer to the Moon is possible in every case either with a short transfer from GTO using a Mid-Course manoeuvre, or via a bi-elliptic transfer to reduce the ΔV demand of the inclination change manoeuvre at cost of higher transfer time in case of a high nodal difference between GTO and Moon's orbit.

The time to perform the TLI manoeuvre, however, can amount to one lunar revolution period (~27 days) in the worst case for both scenarios, as a proper position of the Moon at space-craft's arrival is a requirement.

2.2.3 Weak Stability Boundary Transfer

For a lunar transfer scenario performing TLI after piggy-back launch into GTO there is still another option to reduce the ΔV further. By using the vicinity of the Lagrangian points, so called weak stability boundary (WSB) regions, the orbital energy of the transfer orbit can be increased such that the ΔV upon arrival at Moon can be reduced. Such a transfer to the Moon was already performed by the Japanese Hiten mission [4]. The transfer trajectory is shown in Figure 7.

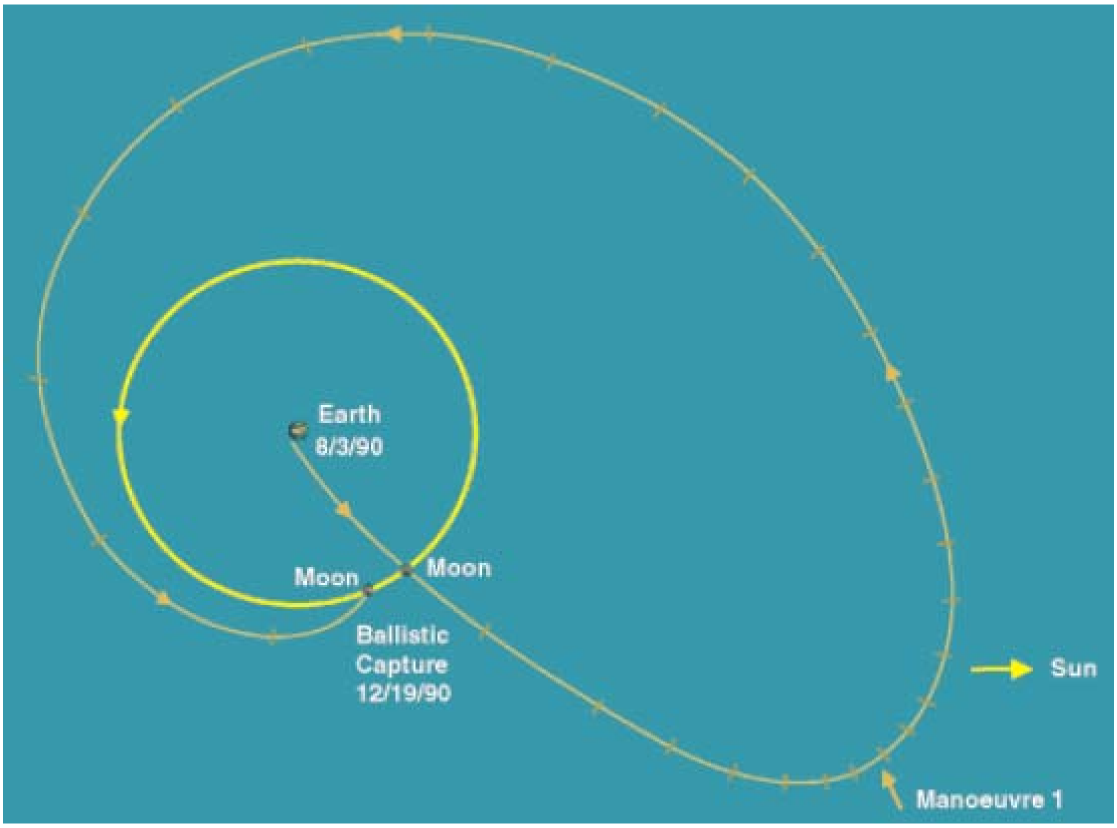


Figure 7 Hiten transfer trajectory [3]

For such a transfer scenario the apogee of the LTO has to be increased to roughly 1.4 Mio. km, in the direction of the Earth-Sun L1 WSB region. In this region the solar perturbation can increase the orbital energy of the transfer trajectory. The importance of the orientation of the LTO w.r.t. Earth and Sun is shown in Figure 8. According to this picture the orbital energy is only increased in Quadrant 2 and 4, for which the gravity gradient is directed along the apogee velocity vector. Luckily the initial orientation of the GTO apogee after launch is always directed towards the sun to achieve minimal eclipse time (see 2.2.1). As the transfer to the WSB region takes roughly 45 days the LTO apogee lies perfectly centered within the fourth quadrant, as one quadrant is passed within 90 days roughly due to Earth's rotation around the Sun. Thus the piggy-back GTO launch offers almost optimal conditions for the WSB transfer.

Upon arrival in the LTO apogee the spacecraft velocity is even smaller compared to the bi-elliptical transfer. Thus the inclination change manoeuvre costs are slightly reduced to 15 m/s such that the transfer is even more beneficial concerning the plane change ΔV.

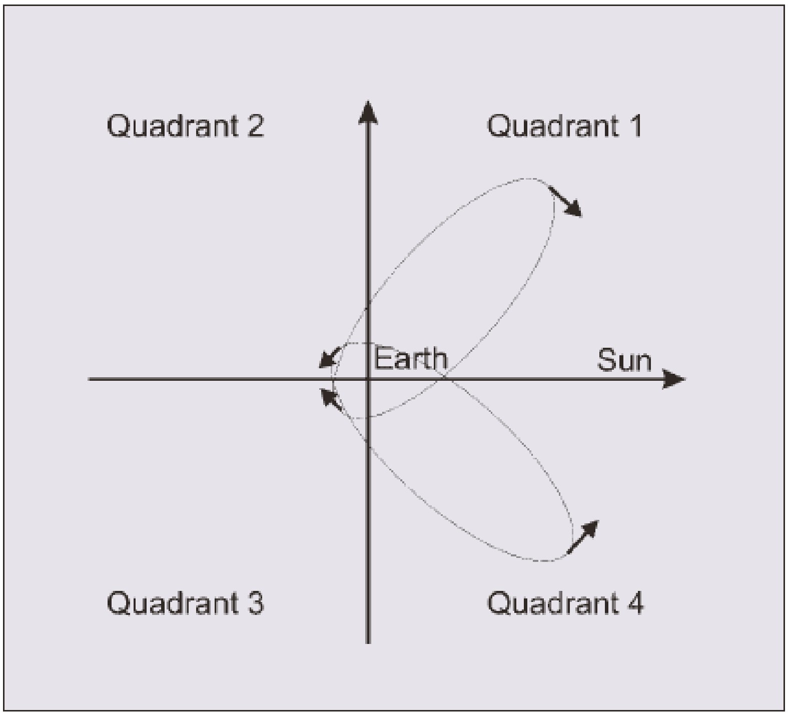


Figure 8 Field-line directions of the Sun's gravity gradient [3]

Much more interesting in this transfer concerning the ΔV, however, is the fact that lunar capture conditions are much more favorable due to slower relative velocities between Moon and spacecraft and can even be achieved without any manoeuvre. Such a ballistic capture, shown in Figure 7, can be achieved if the WSB region of the Earth-Moon system is used too. Thus the overall transfer ΔV could be reduced by 500-600 m/s compared to the Hohmann transfer.

The largest disadvantage of the WSB transfer scenario is, however, the heavily enlarged overall flight time, which might extend to 100-150 days. Additionally the transfer geometry requires a certain right ascension of the moon upon arrival of the spacecraft. As the transfer duration is mainly driven by the altitude of the apogee in the WSB region a certain TLI epoch has to be selected. Hence a waiting period of up to one lunar orbit revolution (~27 days) results prior to TLI as the piggy-back launch prohibits an influence on the launch window.

The waiting period prior to TLI poses additional problems to the mission, as waiting for 27 days in the GTO would result in 60 GTO revolutions. Thus the electronic equipment needs to be designed to withstand 120 Van-Allen-Belt passages. Additionally, due to J2 perturbation the GTO would drift in that time, changing the orientation of the transfer orbit.

3. Transfer Trajectory Design Algorithm

According to SEEFELDER [2] an algorithm to design Hohmann type and bi-elliptical transfer trajectories could be developed including the following steps:

1. Computation of Moon Ω_1 and GTO RAAN Ω_2
2. Angular difference calculation $\Delta\Omega = \Omega_2 - \Omega_1$
3. Strategy Decision:
    - Long Transfer:
        i. $29° \le \Delta\Omega \le 134°$
        ii. $207° \le \Delta\Omega \le 315°$
    - Short Transfer:
        i. $134° \le \Delta\Omega \le 207°$
        ii. $315° \le \Delta\Omega \le 29°$
4. Computation of two complete GTO revolutions
5. Addition of ΔV_0 in case of phasing orbits
6. Computation of n phasing revolutions with $0 \le n \le 4$
7. Addition of ΔV_1 components to enter LTO
8. Forward computation of LTO until arriving at application point of ΔV_2
9. Selenocentric state vector computation for V_{inf} , i = 90° and periselene altitude of 100 km
10. Backward computation of selenocentric state vector to position of ΔV_2 application
11. Transformation into geocentric equatorial reference frame
12. Imposition of matching conditions and computation of ΔV_2 from velocity difference

The calculation should afterwards be optimized by a gradient optimization method. Thereby the magnitude of the overall ΔV is minimized using the following optimization variables:

- Components of ΔV to inject from GTO into phasing orbit
- Components of ΔV to inject from phasing orbit into LTO
- Phasing time
- Time from LTO injection until mid-course of apogee manoeuvre
- Time from mid-course or apogee manoeuvre until lunar injection
- Right ascension of the lunar orbit

Furthermore the algorithm could be improved to also calculate WSB transfers. Therefore the apogee distance needs to be increased to 1.4 Mio. km in the first place. Additionally the shape of the arrival trajectory at the Moon has to be changed from a hyperbolic to an elliptic one for ballistic capture and proper LOI [2].

4. Transfer Trajectory Design within FreeFlyer

Within the Phase 0 study of the AMSAT-Moon project, an algorithm was developed by DLR RY-SARA, to calculate WSB trajectories to the Moon for varying launch windows. The main ideas for the calculation tool were extracted from BELBRUNO [5]. The algorithm was developed for the mission analysis software STK Astrogator. As DLR RB-RT does not own this software, but the similar software tool FreeFlyer, the methodology of the STK algorithm was adopted and afterwards implemented into FreeFlyer. The main difference compared to the algorithm proposed by SEEFELDER [2] lies within forward propagation of the whole trajectory.

According to the algorithm the transfer scenario builds up on three main manoeuvres:

1. Trans Lunar Injection (TLI)
2. Mid Course Correction (MCC)
3. Lunar Orbit Insertion (LOI)

Each of those manoeuvres thereby has to fulfill some initial conditions. For the TLI manoeuvre two conditions exist. The first requires the TLI manoeuvre to be executed in the GTO perigee and to be large enough to reach an LTO apogee in the Earth-Sun L1 WSB region of about 1.4 Mio. km, while the second requires an appropriate initial orbital phase of the Moon to ensure rendezvous of Moon and spacecraft upon arrival.

The MCC manoeuvre is executed in the LTO apogee, mainly for fine adjusting the lunar rendezvous and the inclination change. As the spacecraft velocity is very small in this point, the manoeuvre is rather small and all three velocity components are allowed to be varied. The goal of the following LTO part is to end within the sphere of influence of the Moon such that capture by an LOI manoeuvre is possible.

If the first two manoeuvres are properly adjusted, the spacecraft will rendezvous with the Moon at crossing of its trajectory. In case of a ballistic capture only a small manoeuvre needs to be performed for LOI to be in an initial lunar orbit and not to be affected any further by the Earth-Moon WSB region. The manoeuvre to target the final orbital altitude w.r.t. Moon and the eccentricity of orbit still needs to be applied.

5. AMSAT-Moon

5.1 Initial Conditions and Algorithm Fine Tuning

The AMSAT-Moon mission is intended to be a low cost mission, executed in a corporation of AMSAT Germany and the DLR. The idea is to use the existing spacecraft bus of a former AMSAT mission, the P3-D bus. This spacecraft bus was already used in an earlier Earth observation mission flying on a High Elliptical Orbit (HEO).

The low cost mission is thus realized by saving spacecraft bus development costs, as well as integration cost, as integration shall be performed by the AMSAT group. The goal of a Concurrent Engeneering Facility (CEF) study within Phase 0 was to adopt the existing bus to the new mission requirements.

On major cost reduction driver of the former AMSAT missions was the usage of Ariane 5 piggy-back launches. As the P3-D bus is developed for such a launch it is also intended for the AMSAT-Moon mission to bring the spacecraft into GTO. Hence also the initial conditions for the transfer scenario are set by the Ariane 5 injection into GTO (Table 2).

Parameter	Value
Perigee altitude	250 km
Apogee altitude	35943 km
Semimajor axis	24499.637 km
Eccentricity	0.727419
Inclination	7°
Argument of Perigee	178°
RAAN	Depending on launch date

Table 2 Nominal GTO after Ariane 5 launch

However, the usage of an Ariane 5 piggy-back launch allows no influence on the launch date selection. This is the reason why the WSB transfer is selected. Additionally a mass budget constraint is set by the launch adapter for the Ariane 5 piggy-back launch available at AMSAT, named SBS. It is space qualified by the earlier P3-D mission to support a 650 kg spacecraft.

Additional requirements are to launch the spacecraft between 2012 and 2014, within the typical Ariane 5 launch window lasting from 23:30 to 0:30 UTC. After launch the spacecraft shall be able to withstand the radiation environment for 27 days in the GTO. The final lunar orbit to perform scientific investigations at the Moon is required to have the dimension of 100 x 100 km.

The developed algorithm (chapter 4) was thus fine tuned to the initial requirements. First of all the suggested GTO was configured and the RAAN corrected due to the location of the Sun. For the right point in time to place the TLI manoeuvre, an angle of roughly 130° between Moon, Earth and Sun was set according to BELBRUNO [5]. LTO outbound transfer, LTO apogee manoeuvre and LTO inbound transfer are then performed as described above. The final aim point of the LTO inbound trajectory was, however, due to targeting in FreeFlyer/STK optimized in subsequent steps, with different distances of spacecraft to Moon. After capture the spiraling in the lunar system down to the 100 x 100 km final orbit was performed.

5.2 Transfer Scenario Results

A first design of the WSB transfer resulting from the algorithm implemented in FreeFlyer is shown for an initial GTO on 25th February 2012 00:00:00 UTC, up to lunar capture on 31st May 2012 04:24:49 UTC, with a ΔV of 737.5 m/s for TLI and the apogee distance of 1.376 Mio km. While the plot in the X-Y-plane (Figure 9) shows the common WSB profile up to lunar rendezvous, the X-Z- plot (Figure 10) shows especially the inclination change by the MCC. The launch date is selected arbitrarily within the given timeframe from 2012 to 2014. The overall transfer time amounts to 100 days and the overall ΔV including capture and margins sums up to roughly 1.8 km/s. To accomplish such a ΔV a propellant mass of approximately 300 kg has to be carried along being equal to 44 % of the total spacecraft mass of 650 kg.

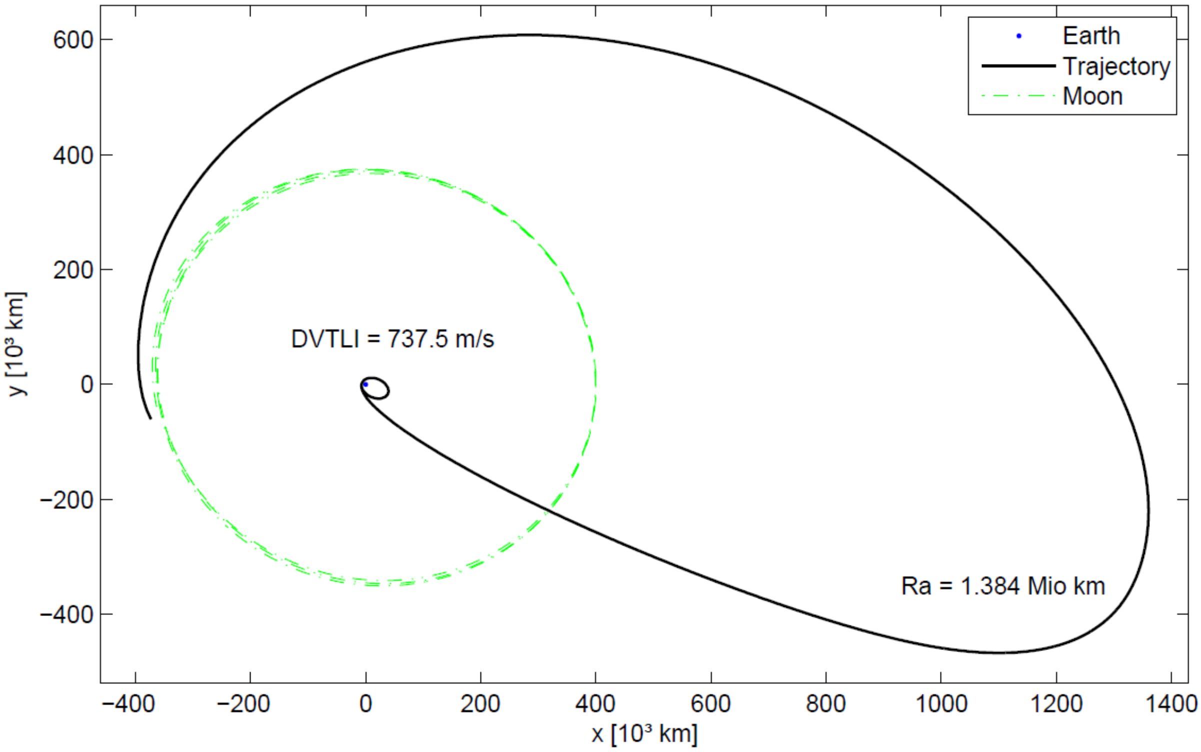


Figure 9 X-Y plot of WSB transfer (MJE2000, initial GTO on 25th February 2012 00:00:00 UTC)

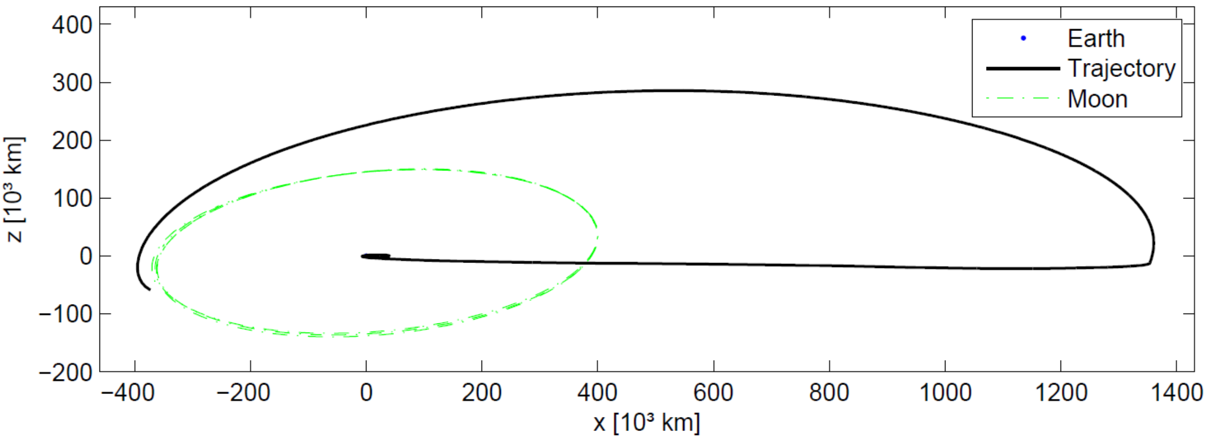


Figure 10 X-Z plot of WSB transfer (MJE2000, initial GTO on 25th February 2012 00:00:00 UTC)

5.3 Manoeuvre Errors

Considering the presented transfer (Figure 9) different problem arose. First of all the waiting time in GTO of up to 27 days leads to up to 60 orbits through the Van-Allen-Belt or 120 passages respectively. Hence an increased effort would be required to redesign all electronic equipment due to the resulting high radiation protection requirements.

Much more important, however, are the manoeuvre execution errors. As the thrusters onboard the spacecraft cannot be calibrated according to their performance in space, errors occur in the performed manoeuvre. For a typical 400 N main engine the error amounts to 1.5 – 2 % at first burn. After calibration due to the experienced performance, this error can be reduced to be smaller than 1 % for future manoeuvres. For a cold gas thruster system, to be possibly used for lunar orbit corrections, those errors are even worse amounting to 20 % for the first and 2-5 % for consecutive manoeuvres.

To evaluate the impact on the WSB transfer an error of ±1 % has been applied to the TLI manoeuvre (Figure 11). As it is currently the first manoeuvre to be executed after launch the error is probably even larger. For the given TLI manoeuvre of 737.5 m/s the error results to ±7 m/s. Reducing the TLI by this error, the LTO apogee is reduced by 388000 km such that the orbital energy cannot be increased as much by the Sun as intended (Figure 11). Adding the error to the initial TLI, however, increases the overall spacecraft velocity so much that escape velocity from Earth is achieved (Figure 11).

Thus a strategy has to be found, by which the error proneness of the transfer scenario can be reduced.

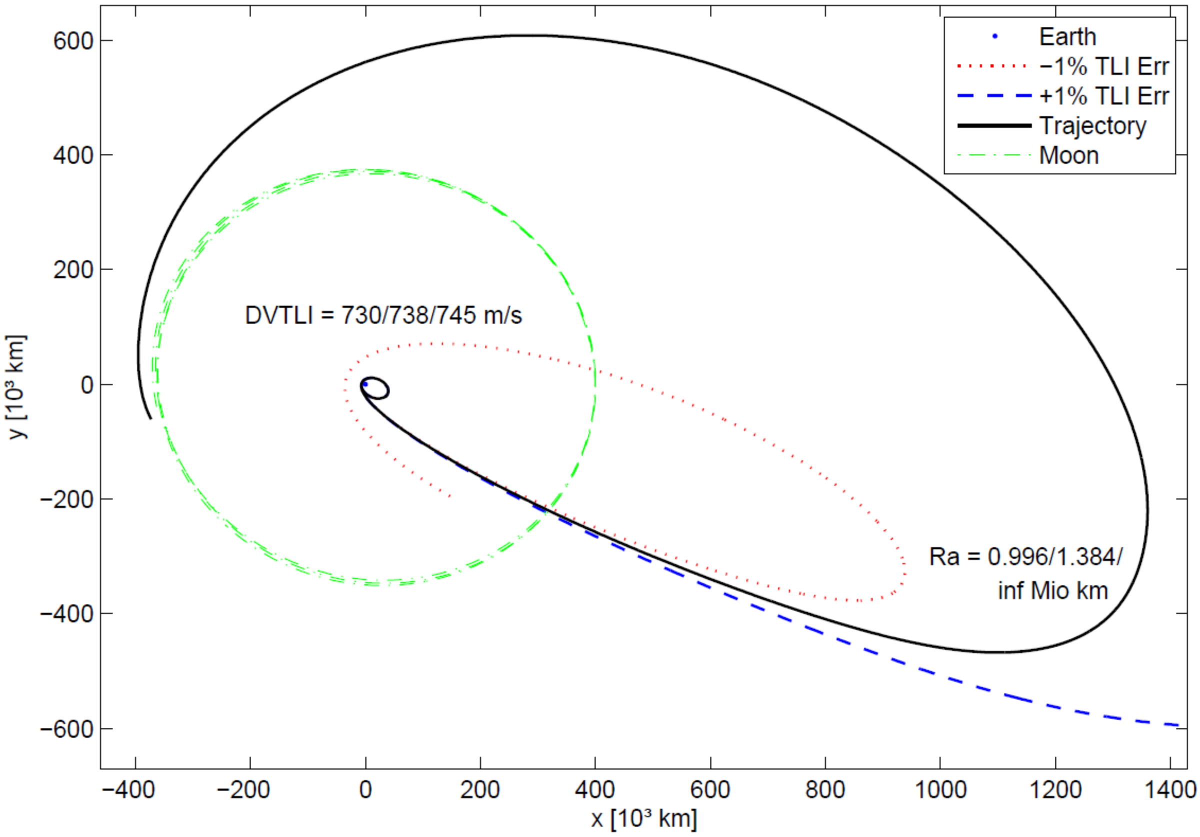


Figure 11 Manoeuvre error of ±1% at TLI (MJE2000, initial GTO on 25th February 2012 00:00:00 UTC)

5.4 Phasing Orbits

The intended strategy almost solves the manoeuvre error problem completely, shows additional benefits concerning radiation issues and does not change the overall mission scenario at all.

To reduce the manoeuvre error effect on the TLI the simplest way is to reduce the TLI manoeuvresize. This can be achieved by splitting up the TLI in three manoeuvres, whose amplitude decreases to each manoeuvre, whereas the sum equals the former TLI (Figure 12). The resulting orbits from the first two manoeuvres, called phasing orbits (PO), have considerable advantages. Since the waiting time for TLI is not spent in the GTO the Van-Allen-Belt passages are reduced and additionally the disadvantage of a drifting GTO is not present anymore. Due to the complex correlation of TLI epoch, apogee altitude in the WSB and the lunar right ascension upon arrival the presented transfer was found for a slightly shifted initial GTO on 20th February 2012 00:00:00 UTC.

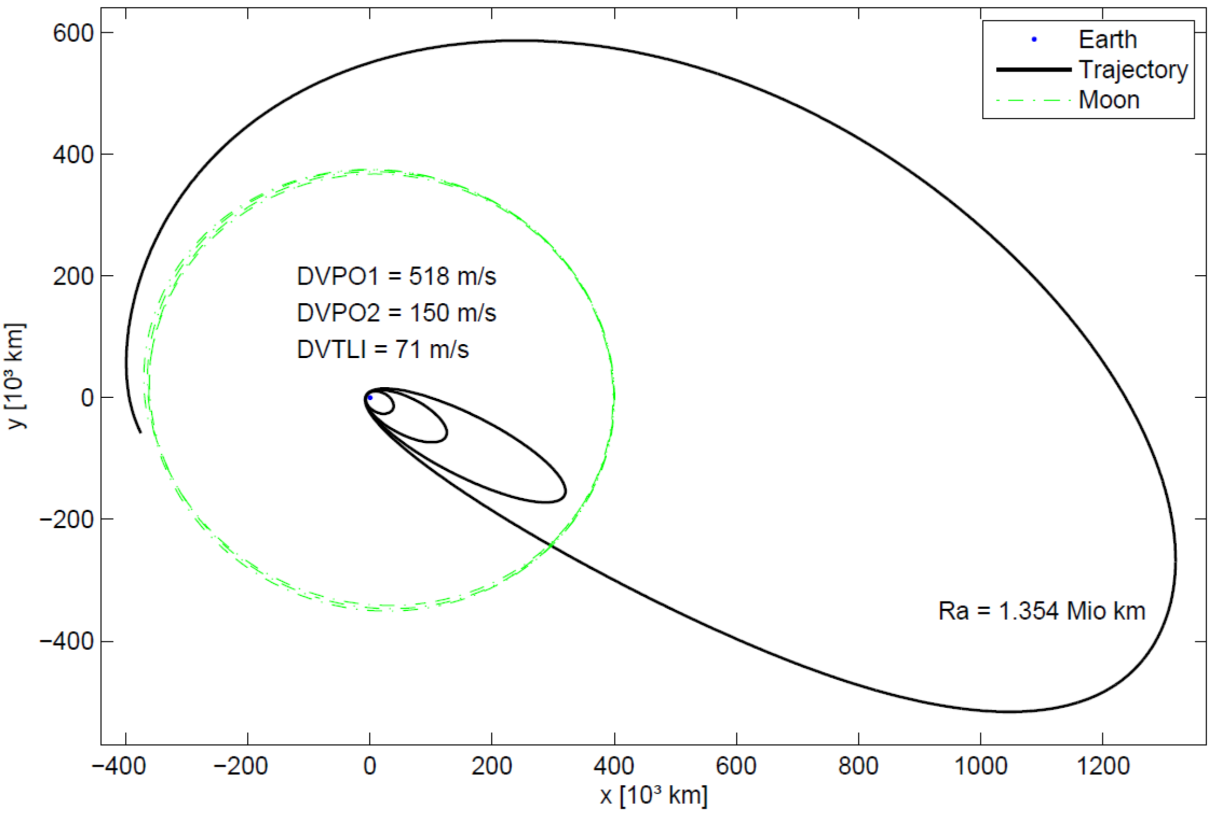


Figure 12 X-Y plot of phasing orbits and WSB transfer (MJE2000, initial GTO on 20th February 2012 00:00:00 UTC)

The manoeuvre planning strategy for the phasing orbits was however not optimized. Merely a constraint can be given that the manoeuvres should consecutively be reduced, as a smaller manoeuvre produces a smaller error. Furthermore, the strategy depends on the overall waiting time until TLI. If this is rather short, for example only three days, it might even be advantageous to extend the waiting time to an additional lunar revolution period to perform proper calibration of the thrusters. The ΔV of the final TLI is, however, depending on the period of the largest phasing orbit. The larger the orbital period, the smaller the final ΔV, which finally reduces the manoeuvre error. The resulting smaller errors can hence be easier corrected by trajectory correction manoeuvres (TCM). Additionally it has to be noticed that in some allocations interactions of the phasing orbit apogee with the Moon can occur. Thus careful manoeuvre planning has to be performed.

To approve the advantageousness of the phasing orbit strategy again manoeuvre errors are applied on the final TLI manoeuvre (Figure 12). The resulting error of ±0.7 m/s leads to a 52000 km lower (-0.7 m/s) or 60000 km higher (+0.7 m/s) apogee of the LTO (Figure 13).

The most important result is the fact that including the phasing orbit strategy, the position error due to TLI manoeuvre error lies still in the region of $6x10^4$ km for the apogee in the WSB. However, the intended transfer is still apparent with manoeuvre errors such that the phasing orbit strategy provides significant advantage compared to the basic scenario (Figure 11). By use of TCM a proper rendezvous of spacecraft and Moon should be possible.

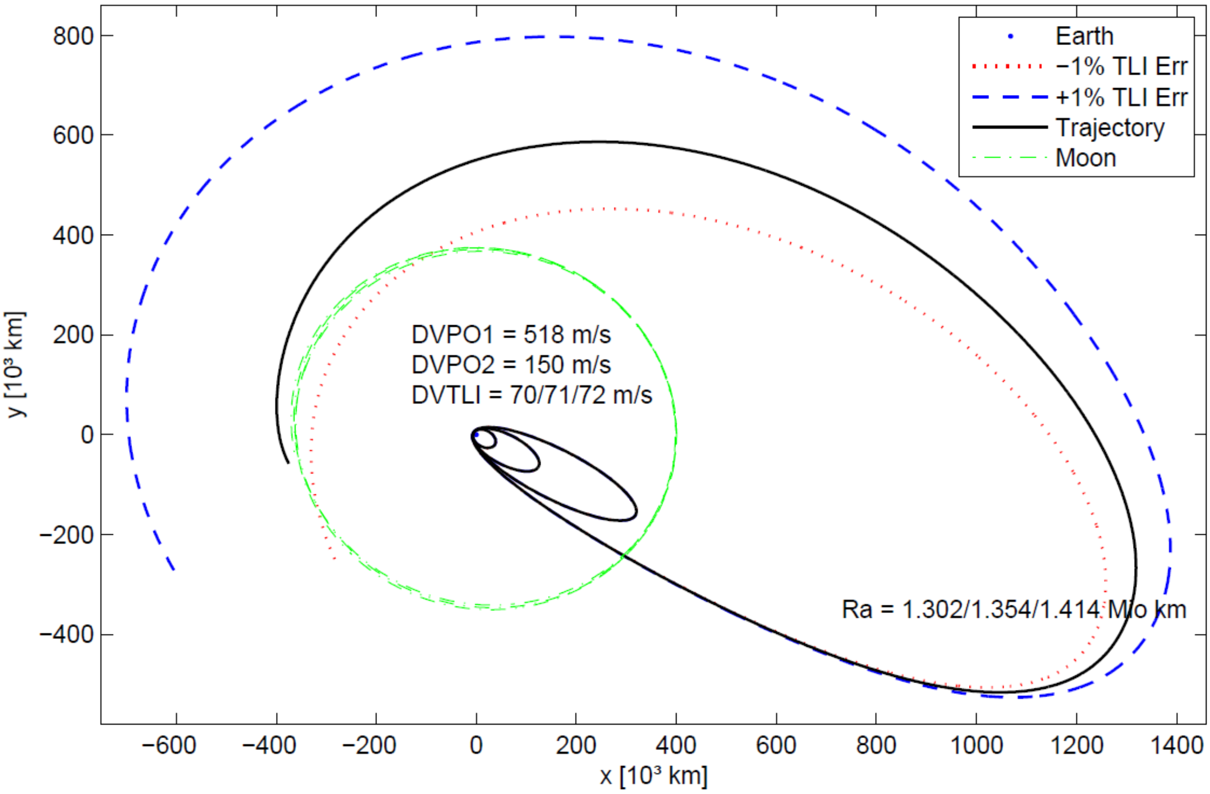


Figure 13 Manoeuvre error of ±1% at TLI (MJE2000, initial GTO on 20th February 2012 00:00:00 UTC)

5.5 Trajectory Correction Manoeuvres

After performing the TLI manoeuvre with the inevitable manoeuvre execution error, several TCMs should be performed on the way to the Moon to. Currently up to three manoeuvres are planned on the way towards the WSB and additionally up to three manoeuvres after mid-course correction on the way towards the Moon. Although the TCM will again provide errors a proper adjustment of the trajectory should be possible. An improved manoeuvre error analysis, however, needs to be performed to obtain reasonable strategy and the corresponding ΔV values. The output of such an analysis should yield detailed information about the positioning improvement within the WSB and additionally the maximum error within the TLI manoeuvre, which could be compensated. The latter could then be used to develop a detailed phasing orbit strategy.

Important input for such an analysis is the orbit determination effort, which is required to achieve a specific accuracy. The achievable orbit determination accuracy is important to perform TCMs as fast as possible after the mid course correction manoeuvre. As the spacecraft velocity is smaller closer to the WSB region a larger correction can be achieved with less ΔV.

The effort concerning orbit determination in a certain time to perform a proper TCM, and additionally how many TCMs are required to achieve the planned transfer has to be taken in the next project phase.

5.6 Visibility

The analysis, performed with FreeFlyer includes the contact times between launch on 20th February 2012 00.00.00.000 and lunar arrival on 31st May 2012 02.31.55.100 for each mission phase to the Weilheim ground station.

5.6.1 GTO

For the presented scenario (Figure 12) no contacts from Weilheim have been detected within the first revolution in GTO. Thus a larger ground station network for GTO phase has to be selected in any case. In general four revolutions in GTO are foreseen, due to operational constraints, as precise orbit determination, manoeuvre rehearsal, etc.

For contact time testing a test setup was created with 10 GTO revolutions, summing up to roughly five days of orbital period in total. For those orbits each day one contact is at least possible, but with varying contact times (Table 3). Those lie between 2.5 to 9.5 hours. Thus the statement from above to extend the ground station network for the LEOP phase in GTO is confirmed, as only one of two revolutions per day in GTO can be tracked.

Entry Epoch [UTC]	Exit Epoch [UTC]	Duration [min]
Feb 20 2012 12:26:22	Feb 20 2012 15:00:44	154
Feb 21 2012 08:31:20	Feb 21 2012 18:00:34	569
Feb 22 2012 05:38:46	Feb 22 2012 15:02:57	564
Feb 23 2012 07:01:10	Feb 23 2012 11:46:13	285
Feb 24 2012 11:17:19	Feb 24 2012 15:29:55	252

Table 3 Test setup for GTO contact times from WHM

5.6.2 Phasing Orbit One

The orbital period of phasing orbit 1 is 2.28 days. For this mission phase contacts are available on each day for 9 to 10 hours (Figure 14).

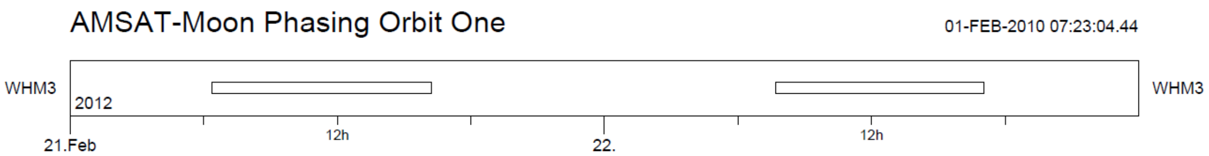


Figure 14 Contact times for Phasing Orbit one from WHM

As an example for many other passes occurring in subsequent mission phases an elevation plot for the first pass in Phasing Orbit 1 is presented (Figure 15). The plot shows the characteristic maximum elevation of around 40°, which can be achieved from the Weilheim ground station for a GTO.

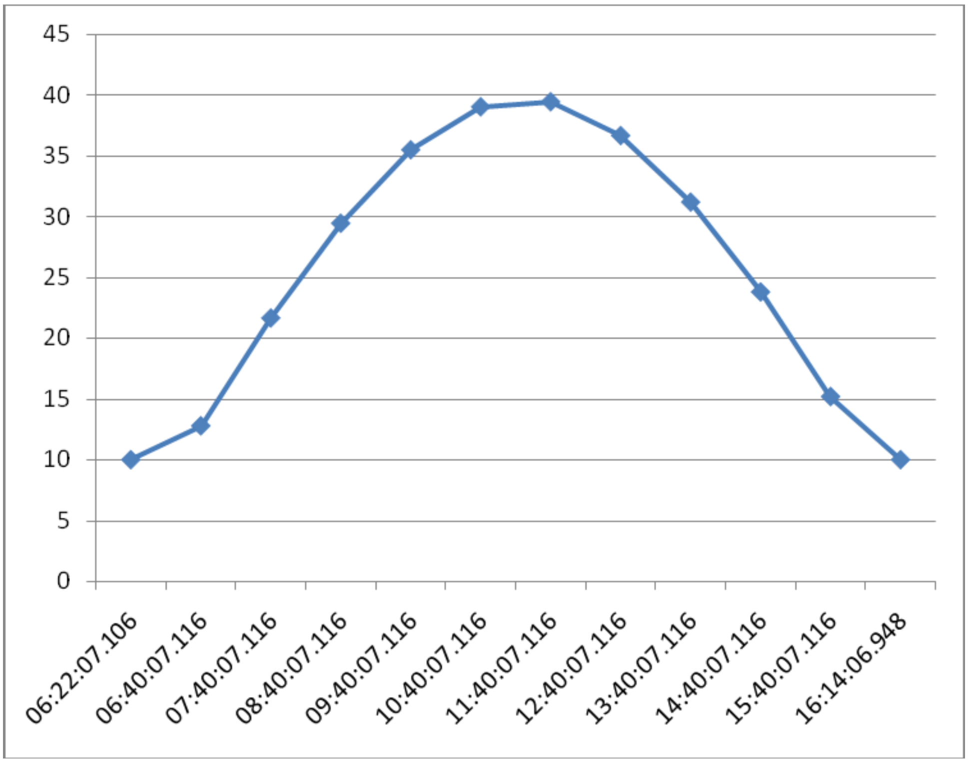


Figure 15 Elevation plot for first pass in Phasing Orbit 1

5.6.3 Phasing Orbit Two

For phasing orbit two with an orbital period of 8.82 days also each day a contact is possible, enduring 9.5 to 10 hours (Figure 15), except for the last day, for which the contact time is only eight hours.

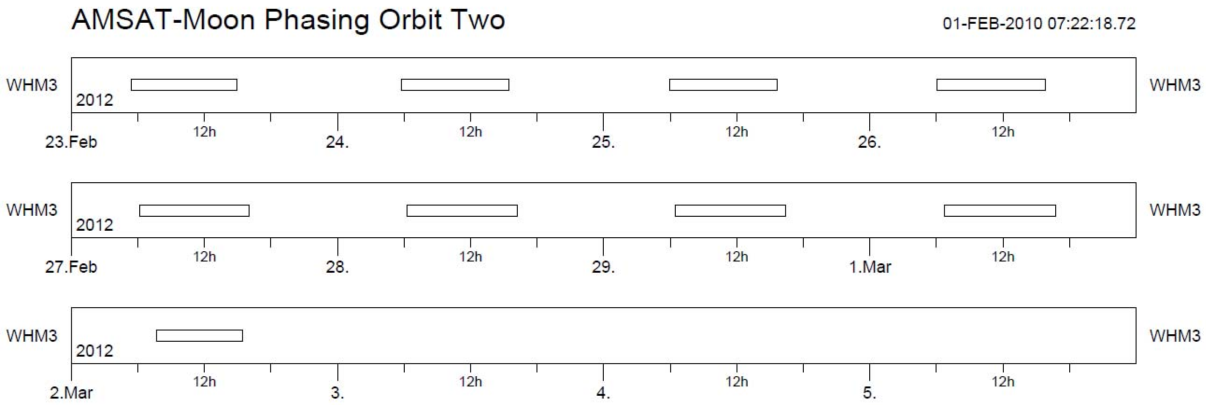


Figure 16 Contact times for Phasing Orbit two from WHM

5.6.4 LTO Outbound

For the outbound transfer towards the WSB one contact each day is possible. The contact time varies between 570 and 599 minutes (Figure 17).

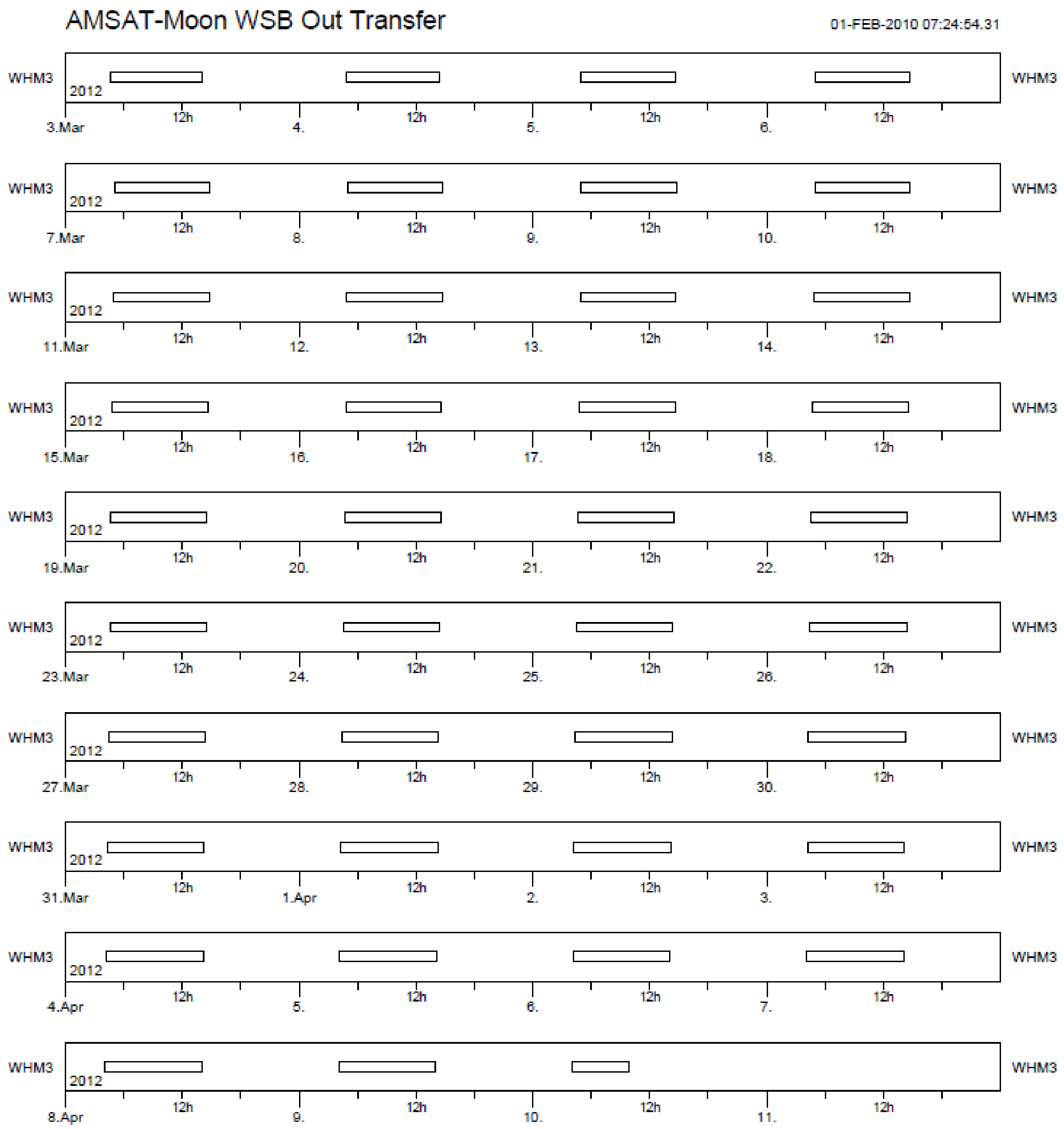


Figure 17 Contact times for WSB Outbound transfer from WHM

5.6.5 LTO Inbound

The same as for the outbound transfer is also valid for the inbound transfer. One passage per day over Weilheim, with contact times between 553 to 801 minutes (Figure 18). The last entry in Figure 17 and the first entry in Figure 18 have of course to be combined.

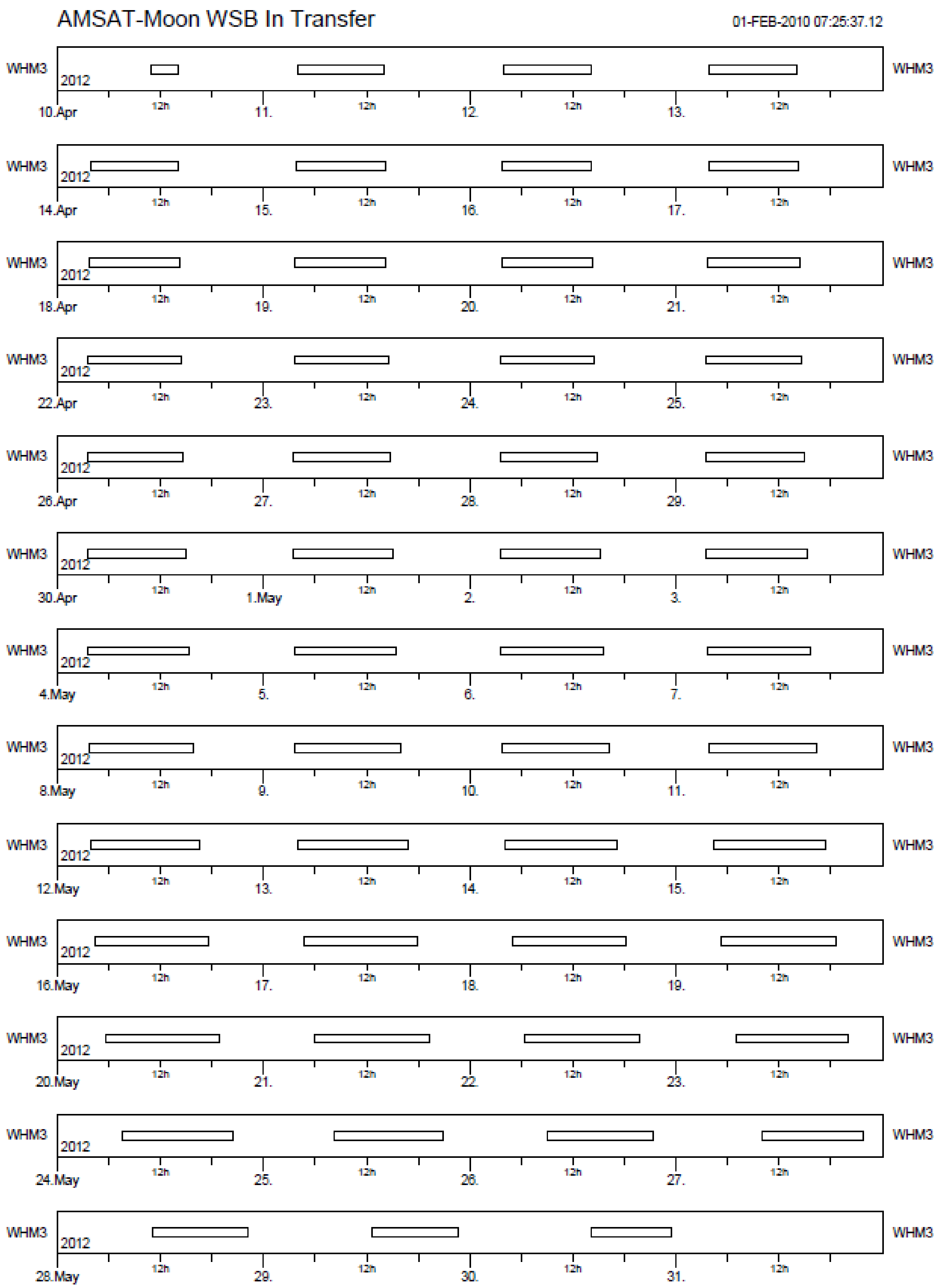


Figure 18 Contact times for WSB inbound transfer from WHM

5.6.6 Lunar Orbit

Visibility during routine operations in the lunar orbit, as well as orbit stability is analysed by KAHLE in [6].

6. Electrical Transfer Scenario

As the chemical transfer scenario requires a long transfer time and has a very high propellant ratio of 44 % also an electrical transfer was analyzed. By this ideally a lower propellant ratio shall be achieved, resulting in higher payload mass.

6.1 Optimization Software

The low-thrust trajectory optimization software InTrance developed by DACHWALD and OHNDORF [7] [8] was used to find near global optimal transfer trajectories. The software uses the combination of evolutionary algorithms and neural networks, called evolutionary neurocontrol, to find near global optima.

The big advantage of InTrance being a global optimization tool is that a dedicated initial guess is not required, like for local optimization tools. Hence only launch conditions (launch date, C3, propellant mass) and target state (rendezvous, final orbit) have to be specified. During optimization InTrance calculates the optimal steering strategy to achieve a minimal flight time and additionally the optimal values for the initial conditions, which can be specified within intervals.

During the latest software updates InTrance gained the possibility to optimize multiphase mission scenarios. Hence spiraling out from GTO up to lunar capture and spiraling within the lunar system down to the final orbit can be performed in one run such that the optimal handover point at lunar arrival is optimized too.

6.2 Mission Scenario

The initial conditions for the transfer scenario were very similar to those for the chemical transfer scenario. The overall spacecraft mass of 650 kg was kept, whereas a detailed subsystem specification will be given in the next section.

For the first mission phase, spiraling out from GTO to lunar capture, the same initial GTO was set up as for the chemical scenario (Table 2). The maximum allowed flight time for this phase was set to 120 days with allowed integration step sizes between 30 seconds and 60 minutes. The launch window was set to whole January 2012. Thus of course the best constellation concerning declination will be selected and additional effort to create trajectories for daily launch windows will have to be spent in future project phases. The end condition for this phase was merely set to be captured by Moon.

The second phase continues with the final state as initial condition and performs hence spiraling within the lunar system. An overall flight time of 70 days was allowed with integration step sizes between 10 seconds to 60 minutes. The shorter steps were allowed due to shorter orbital revolution periods around Moon. The target state to be achieved consists of different orbital elements. Primarily a semi major axis of 2000 km shall be achieved, with a final differential position and velocity of 10000 km and 50 m/s at the maximum. Furthermore the target eccentricity was set to be zero and the inclination to be 84 degrees within the lunar equatorial frame. Other elements were not targeted, as RAAN, ARP and true anomaly are unimportant for preliminary design.

Although the target conditions of semi major axis and differential velocity seem large it has to be noted that a global optimization tool is used. Hence for finding the near global optimal solution a tribute has to be paid in the accuracy of the final state. However, the best solution could be input to a local optimization tool, as it provides a very good initial guess. From the local optimization a thorough solution should then result.

6.3 Preliminary Spacecraft Design

The design of some subsystems can actually be very similar to the chemical transfer scenario from the AMSAT-Moon study. The important changes and major considerations will be outlined in the following subsections. For some subsystems merely some ideas will be formulated, as the effort of detailed analysis would be too much for a preliminary analysis. However, a mass budget of each subsystem will be given based on the results of the AMSAT-Moon CEF study and the considerations taken. The main part of subsystem redesign will occur in propulsion, power and structure. That is why those systems will be covered first.

6.3.1 Propulsion

The design of the propulsion system has of course to be changed completely, due to the selection of electrical thrusters. For the performed analysis, two Radiofrequency Ion Thrusters (RIT) are selected. More specific the RIT-22 manufactured by EADS Astrium was selected (Figure 19), whose characteristics are presented in Table 4.

Figure 19 RIT 22 [9]

Characteristic	RIT 22
Propellant	Xenon
Ionisation principle	Radio frequency excitation
Discharge chamber diameter	22 cm
Beam diameter	21 cm
Beam voltage (nom)	1200 to 2000 V
Power (nom)	5000 W at 150 mN and 4500 s Isp
Nominal thrust level	120 to 200 mN
Demonstrated thrust level	80 to 250 mN
Nominal specific impulse	3000 to 5500 s
Demonstrated specific impulse	2500 to 6400 s
Design life	> 10000 h
Overall length	23 cm
Outer diameter	30 cm

Table 4 RIT 22 characteristics [9]

Within the presented intervals of the thruster characteristics (Table 4) a specific configuration was selected with the denomination RIT-22 LO. For this configuration the thrusters have a mass of 17.79 kg each and produce a thrust of 140.7 mN at the maximum power input of

4032 W. The thrusters can be throttled with linear performance down to 65% of the maximum input power. Hence a minimum input power of 2621 W is required to operate one thruster at least. The specific impulse (I_{sp}) of the thruster has a value of 3714.37 s. For the chemical, Hohmann-like transfer from GTO to Moon and orbit circularization from 60000 x 60000 km orbit to the final 100 x 100 km orbit the ΔV was calculated to 2366 m/s. Converting this transfer to an electrical one usually a factor of 2.2 has to be applied to the chemical ΔV, leading to 5205 m/s for the electrical transfer. Entering ΔV, I_{sp} and launch mass into the rocket equation, the propellant mass results to 86.53 kg. As the circularization to 100 x 100 km orbit calculations has been really conservative, the propellant consumption entered in the setup of the optimization was set to be 85 kg at the maximum. From this propellant mass also the tank mass results. For this analysis the design was performed via a tank mass fraction of 6 % of the propellant mass. Valves, pipes and the cold gas system haven been taken over from the AMSAT-Moon CEF study.

Subsystem	Mass [kg]	Margin [%]	Total [kg]	Ratio [%]
Propulsion system	78,30	15,45%	**90,40**	**19,74%**
Thrusters	35,60	20,00%	42,72	
Tank mass	5,10	20,00%	6,12	
Thruster pressure Latch valves	0,60	20,00%	0,72	
Tank pressure latch valve	0,60	20,00%	0,72	
Pipes	3,50	20,00%	4,20	
Cold gas thrusters	1,20	5,00%	1,26	
Cold gas tank	18,80	10,00%	20,68	
Nitrogen (Cold gas propellant)	8,70	10,00%	9,57	
Pressure Latch valve	3,60	5,00%	3,78	
High pressure latch valve	0,60	5,00%	0,63	

Table 5 Propulsion system mass budget

6.3.2 Power

The second important change in subsystem design affects the power subsystem. By selecting an electrical propulsion system containing two thrusters requiring an input power of 8064 W to operate both at maximum thrust, a huge amount of power has to be provided. Thus a solar array is intended to be used. For trajectory optimization within InTrance the solar array was assumed to be a GaAs triple junction array, providing 8.2 kW output power with a power specific mass of 10 kg/kW. Such an array exists already today for the Eurostar 3000 spacecraft bus, manufactured by EADS Astrium (Figure 20). Considering Table 6 an Eurostar 3000 class XS solar array could be selected, which would yield the required power output.

Figure 20 Eurostar 3000 solar array [10]

	E3000 Class	Nr and size of panels	Solar Array Power (kW, EOL, EQX, 15 years)		
			3 panels per Wing	4 panels per Wing	5 panels per Wing
short panels	SX	3 small	6 - 8,4 "XS"		
	S	4 small		8,6 - 11,2 "S"	
long panels	L	4 large		10,8 - 14,2 "L"	
	LX	5 large			14,3 - 17,7 "XL"

Table 6 Eurostar 3000 solar array data [10]

Coinciding with the larger array also a larger battery has to be provided in order to produce thrust during solar eclipse. The maximum eclipse time is estimated for the GTO with 15 minutes umbral and 15 minutes penumbral shadow. Hence 30 minutes eclipse time has to be covered by the battery, which has to provide 8.2 kW during this time. Hence a value of 4.1 kWh is the design driver for the battery. Changing the AMSAT-Moon battery to Li-ion technology a much higher energy to weight ratio can be achieved. EADS Astrium provides a battery based on that technology with a ratio of 110 Wh/kW [11]. Hence the battery mass mounts to 41 kg with 10 % margin included.

While Battery Management Unit (BMU) and Mode Control Unit (MCU) will roughly have the same mass, Battery Charge and Discharge Regulator (BCR/BDR), as well as the Power Control and Distribution Unit (PCDU) will probably have to be redesigned in order to cope with the higher input power of the solar array. Thus 3 times the mass for the AMSAT BCR is assumed including all components mentioned above.

Having retrieved these estimates for the power subsystem elements the overall mass including margins arises to 157.4 kg (Table 7), which makes a ratio of 34.36 % of the overall spacecraft dry mass.

Subsystem	Mass [kg]	Margin [%]	Total [kg]	Ratio [%]
Power	134,27	17,22%	**157,4**	**34,36%**
Solar array	82,00	20,00%	98,40	
Battery	37,27	10,00%	41,00	
BCR	15,00	20,00%	18,00	

Table 7 Power system mass budget

6.3.3 Structure

During the AMSAT-Moon CEF study, a fraction of 11.4 % of the total spacecraft dry mass resulted for the structure. Concerning [12] a structure fraction of 12 % of the whole spacecraft dry mass is a reasonable value. Thus the structure mass of the P3-D satellite bus of 36.1 kg resulting from the AMSAT-Moon CEF study is extended by roughly 50%. Hence 18.7 kg of additional structure are applied to account for the higher amount of structure due to the larger solar arrays and the higher overall spacecraft dry mass compared to the AMSAT-Moon CEF study.

Subsystem	Mass [kg]	Margin [%]	Total [kg]	Ratio [%]
Structure	51,38	6,65%	**54,80**	**11,96%**
P3-D Structure	34,38	5,00%	36,10	
Additional Structure	17,00	10,00%	18,70	

Table 8 Structure mass budget

6.3.4 Data Handling

This subsystem will be similar to the chemical transfer scenario. The telemetry data rate will be the same, except for higher monitoring effort for solar array and electrical thrusters. But the changes are assumed to be rather small. Thus the two onboard processors and the integrated housekeeping unit build up this subsystem (Table 9).

Subsystem	Mass [kg]	Margin [%]	Total [kg]	Ratio [%]
Data Handling	3,00	20,00%	**3,60**	**0,79%**
RUDAK-IIIa	1,00	20,00%	1,20	
RUDAK-IIIb	1,00	20,00%	1,20	
IHU (Integrated Housekeeping Unit)	1,00	20,00%	1,20	

Table 9 Data Handling mass budget

6.3.5 Communication

The communication might even benefit from the electrical transfer strategy, as the maximum range to the spacecraft will not extend the distance of the Moon. Compared to the chemical transfer, where the maximum distance lies in the WSB region at roughly 1.5 million km, the distance is only one third as large. However, the subsystem was transferred equal to the AMSAT-Moon study to be on the safe side.

Subsystem	Mass [kg]	Margin [%]	Total [kg]	Ratio [%]
Communications	18,40	18,91%	**21,88**	**4,78%**
Antenna S/X-band	5,00	20,00%	6,00	
Antenna UHF (omnidirectional)	0,40	10,00%	0,44	
Antenna S-Band (omnidirectional)	0,40	10,00%	0,44	
X-Band SSPA 50W	1,20	10,00%	1,32	
X-Band SSPA PSU	5,00	20,00%	6,00	
UHF transceiver	1,00	20,00%	1,20	
S-Band receiver	2,00	20,00%	2,40	
S-Band SSPA 10W	1,00	20,00%	1,20	
S-Band SSPA PSU	1,00	20,00%	1,20	
Antenna L-band (omnidirectional)	0,40	20,00%	0,48	
L-Band receiver	1,00	20,00%	1,20	

Table 10 Communication system mass budget

6.3.6 Thermal

The thermal subsystem was not analysed in detail during the AMSAT-Moon CEF study. Hence only assumptions may be outlined here. Comparing the WSB transfer and the electrical transfer, the maximum distance from earth is higher in the WSB scenario such that slightly higher temperatures might be expected due to higher Earth and Moon Albedo for the electrical transfer. However, the thermal requirements are probably mainly driven by the scientific mission phase in lunar orbit, as the Moon albedo poses very rough constraints. Thus the transfer scenario does not have a very high impact. However, to bring the thermal subsystem mass fraction into a range proposed by [12] additional mass has been added to the subsystem, by which it is roughly doubled.

Subsystem	Mass [kg]	Margin [%]	Total [kg]	Ratio [%]
Thermal	20,39	9,91%	**22,41**	**4,89%**
Moon-CEF Thermal	10,39	9,83%	11,41	
Additional Thermal	10,00	10,00%	11,00	

Table 11 Thermal subsystem mass budget

6.3.7 GNC

The AOCS subsystem effort is hard to estimate. Of course there will be a higher attitude pointing requirement to allow a proper orientation of the thrust vector such that the intended steering strategy of the spacecraft can be achieved. How this increased effort is realized is, however, not too easy to answer. In the optimal case the electrical thruster is mounted in a way that it can be directed in different positions such that steering is possible to a certain extent without reorienting the spacecraft. If that is not possible all reorientation effort has to be performed by the AOCS system.

Subsystem	Mass [kg]	Margin [%]	Total [kg]	Ratio [%]
GNC	32,80	18,14%	**38,75**	**8,46%**
Star tracker (HYDRA) Optical Head	1,60	10,00%	1,76	
Sun sensor (CSS) Heads	0,40	10,00%	0,44	
Sun Sensor (CSS) Electronics	0,30	10,00%	0,33	
Reaction wheels (RSI 12)	20,40	20,00%	24,48	
IMU (LN-200)	1,00	5,00%	1,05	
Star tracker (HYDRA) Baffle	1,00	10,00%	1,10	
Star tracker (HYDRA) Electrical Unit	1,30	10,00%	1,43	
Reaction wheels-Redundant (RSI 12)	6,80	20,00%	8,16	

Table 12 GNC system mass budget

6.3.8 Configuration

Compared to the AMSAT-Moon configuration resulting from the CEF study, the configuration will probably change. As much less propellant is needed the large space dedicated to the propellant tanks will be available for other things. Furthermore, the structural changes due to the larger solar arrays will have an impact on the design of the spacecraft.

6.4 Results

In the following the resulting transfer trajectory achieved by the electrically propelled spacecraft will be shown. Additionally the overall mass budget for the spacecraft resulting from the above presented subsystems will be listed. The new design will of course be compared to the design result of the AMSAT-Moon CEF study. Finally the operational differences to the WSB transfer will be outlined.

6.4.1 Transfer Trajectory

The overall flight time needed is 151.51 days, where 108.03 days are required up to lunar capture (Figure 21) and the rest of 43.48 days is needed for circularization to the 100 x 100 km orbit (Figure 22).

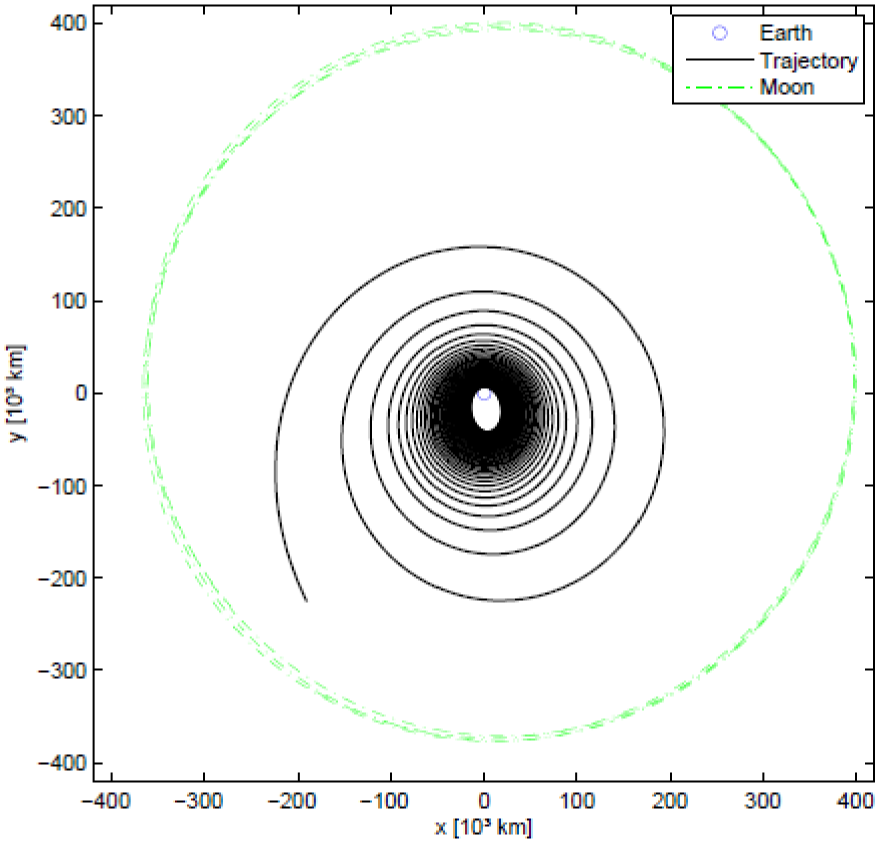


Figure 21 Trajectory from GTO to lunar capture (MJ2000, earth ecliptic inertial)

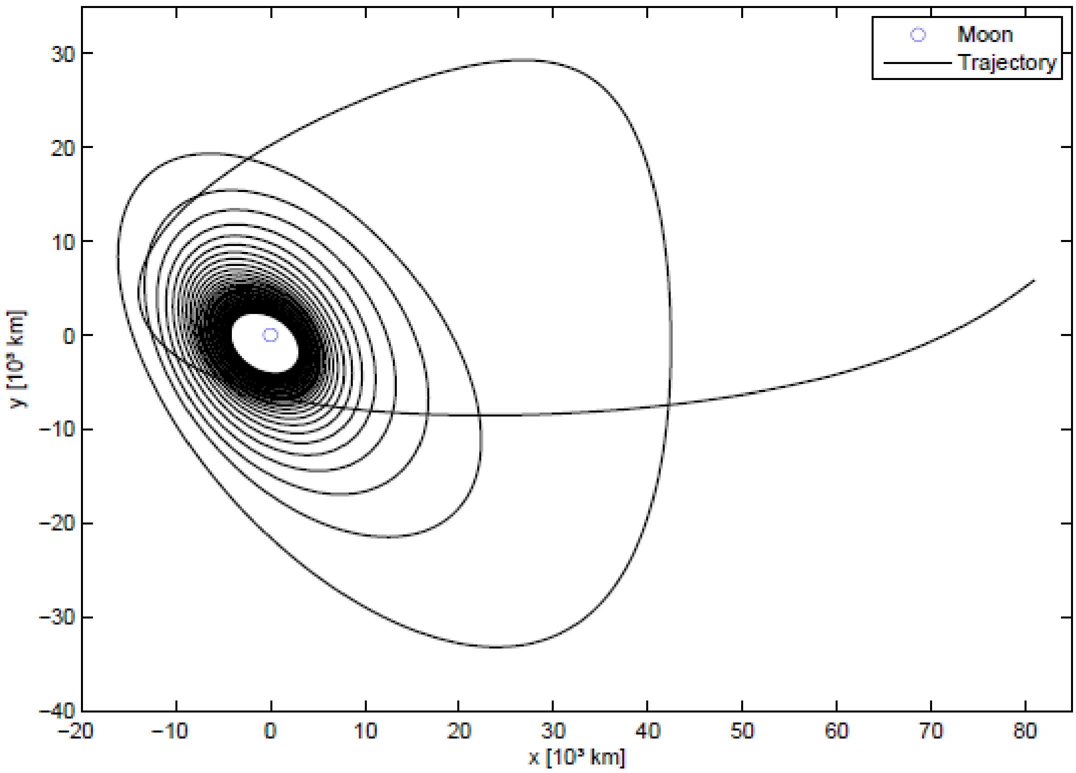


Figure 22 Trajectory from lunar capture to target orbit (MJ2000, earth ecliptic inertial)

The launch would take place on 19th of January in 2012 and hence the final arrival on 18th of June 2012. The final conditions achieved up on arrival are a semi major axis of 6549 km and a relative velocity to the final orbit of 14 m/s. Hence the targeted semi major axis for the 100 km circular lunar orbit is missed by roughly 4700 km. Although this seems much, it is a very good result for the global optimizer. Refining this solution in a local optimization software tool, should definitely allow to achieve the initially intended conditions.

For the electrically propelled transfer an overall ΔV of 5633 m/s is required. Due to the high specific impulse (3714 s) of the electrical propulsion system the propellant mass amounts to only 81.6 kg.

6.4.2 Spacecraft Mass Budget

From the detailed subsystem mass budgets within Section 6.3.1 through 6.3.7 a total subsystem mass of 458.06 kg results (Table 13). This value already contains the specific subsystem design margins, which sums up to 14.64 % in total. Thus the initial subsystem mass is 399.56 kg (Table 13). For preliminary spacecraft design an additional overall system margin of 20 % is applied such that the spacecraft dry mass results to 549.68 kg (Table 13). Hence an overall margin of 37.57 % is applied to the basic subsystem mass of 399.56 kg.

Subsystem	Mass [kg]	Margin [%]	Total [kg]	Ratio [%]
Power	134,27	20,00%	**157,40**	**34,36%**
Propulsion system	78,30	15,45%	**90,40**	**19,74%**
Structure	51,38	6,65%	**54,80**	**11,96%**
Thermal	20,39	9,91%	**22,41**	**4,89%**
Communications	18,40	18,91%	**21,88**	**4,78%**
Data Handling	3,00	20,00%	**3,60**	**0,79%**
GNC	32,80	18,14%	**38,75**	**8,46%**
Harness	15,00	20,00%	**18,00**	**3,93%**
Payload	46,02	10,44%	**50,82**	**11,10%**
Total Subsystem Mass	399,56	14,64%	**458,06**	
Overall System Margin		20,00%	**91,61**	
Spacecraft dry mass			**549,68**	
Propellant mass	81,60	20,00%	**97,92**	
Launch mass			**647,60**	**647.6**

Table 13 Spacecraft mass budget

Compared to the chemical transfer scenario with a spacecraft dry mass of 380 kg the dry mass increased to the electrically propelled transfer by roughly 170 kg. The higher dry mass results mainly by from the larger solar arrays (73 kg) and the higher payload (36 kg).

Most important, however, is the increased payload mass of 50.82 kg achieved by the electrically propelled transfer compared to 14.5 kg in the chemical transfer scenario. Hence a higher fraction of the overall spacecraft mass can be used for scientific instruments or other payload. The payload fraction lies at 11 % for the electrical transfer compared to 4.6 % with the chemical transfer.

All of this can only be achieved by the propulsion system type, as it drives the propellant mass. This is with 98 kg for the electrical mission only one third of the 297 kg required for the chemical transfer.

6.4.3 Operational Differences

The major differences that occur from the mission analysis are the increased number of solar eclipses due to spiraling out of Earth's sphere of influence from a GTO. Coinciding with this higher number of eclipses is the constrained visibility from one ground station. Hence a kind of LEOP network of a common GTO mission might be used in the beginning of the mission, where from time to time stations are excluded, when the spacecraft reaches the next key Earth-spacecraft distance during spiraling towards the Moon.

Besides visibility, and thus access to the spacecraft, and solar eclipses also the orbit determination effort needs to be increased. Due to thrusting continuously a very precise orbit determination has to be performed to allow proper orientation of the thrust vector to achieve the intended transfer trajectory.

Thus a completely new Flight Dynamic System (FDS) needs to be developed to cover all the above mentioned topics.

Nevertheless also advantages are apparent in the electrical transfer. Thus communication and ranging is much easier with the spacecraft during transfer due to the reduced maximum distance from Earth. As this laid at 1.4 Mio km for the chemical WSB transfer it is now reduced to approximately 400000 km. Hence data rates will be higher and ranging can be much easier performed.

6.5 Future Work

After having performed the presented analysis already some future steps can be outlined. The first would be to perform an extended launch window analysis for the electrical transfer. In this study the whole January 2012 was set available and the best launch date was selected by optimization the transfer trajectory within InTrance. Hence it needs to be tested if for each day in January a similar transfer is possible, or how large the differences are concerning flight time and ΔV. From the current perspective it is rather doubtful that it will have much of an impact, as the steering strategy during spiraling out to Moon can be adopted to rendezvous with the Moon at different locations.

In a second step the solution space concerning the spacecraft configuration could be analyzed more in detail. The different impacts of the number of thruster, the thruster's specific impulse, or the thruster input power generated by the solar array on the payload mass to be delivered towards Moon and the flight time should be figured out by this analysis.

The most important step, however, would be to use a local optimization tool, to refine the presented solution to reach the intended final boundary conditions.

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