POST-LAUNCH OPTIMISATION OF THE SMART-1 LOW-THRUST TRAJECTORY TO THE MOOON

Johan Schoenmaekers

ESA/ESOC, Robert-Bosch-Str. 5, D-64293 Darmstadt, Germany
Johan.schoenmaekers@esa.int

ABSTRACT

The SMART-1 S/C was be launched in GTO as additional passenger on Ariane 5 on 27 September 2003. It is currently on its way to the moon where it will be injected into a polar orbit. The orbit control is done with "Solar Electric Propulsion" only. Before launch, the mission design had to cope with a large uncertainty on the initial orbit, as SMART-1 had no control on the Launch date and lift-off time. A worst case trajectory has therefore been the basis for the mission design (launch in december). The favorable actual launch date, as well as the better than expected performance of the solar arrays and precision of the electric propulsion system has allowed to fly a faster and less risky trajectory. The design of this trajectory is presented in the paper.

1. INTRODUCTION

SMART-1 is the first of the "Small Missions for Advanced Research in Technology" which have been introduced into the ESA scientific program. The prime objective of SMART-1 is to demonstrate the "Solar Electric Propulsion" (S.E.P.) concept as prime propulsion and as a key technology for scientific deep space missions.

After several trade-off studies, considering budgetary and mass constraints, a lunar mission has been selected. The propulsion is realised with one PPS-1350 Hall-plasma thruster, providing a force of about 73 mN with t an exhaust velocity of 16,4 km/s at the start of the mission when the solar arrays are not yet degraded by radiation. On 27 September 2003, Smart-1 was launched by Ariane-5 in a GTO and started its low-thrust transfer to the moon.

A major complication for the mission design carried out before launch [3] was that, as additional passenger, SMART-1 could not impose launch window constrains, implying that the right ascension of the ascending node of the injection orbit could take any value. To cover the worst cases (launch in December and June) a trajectory had been designed requiring a transfer up to capture of 16.5 months and 50 kg of Xenon propellant. This transfer started with a continuous thrust spiral of about 2.5 months to get the S/C outside the radiation belts a.s.a.p. and so limit the degradation of the solar arrays. Further, three lunar swing-by's were included to establish the proper conditions for an injection in the target polar orbit around the moon using low-thrust.

The launch during the night from 27 to 28 September led to an injection orbit with the apogee near the ascending node in the moon orbital plane. This special geometry allowed to remove the three lunar swing-by's and to reduce the duration of the transfer to moon capture by 3 months.

The better than expected performance of the solar arrays allowed us to operate the solar electric propulsion motor at a higher thrust level (less gravity loss) and a higher specific impulse than anticipated. Also the predictability of the thrust level was better than expected, reducing the fuel needed for navigation corrections. The fuel saved is used to lower the apocentre of the moon operational orbit from 10,000 to 3,000 km, improving the scientific merit of the mission.

The presentation on the re-design of the transfer trajectory starts with the analysis of the operational orbit around the moon. The parameters of this orbit define the pre-capture orbit, which is the target for the transfer. Subsequently, the first 6 months are presented which were solely driven by getting outside the radiation belts a.s.a.p. and limiting to 135 min the duration of the eclipses which occurred after 6 months. This sets the initial conditions for the design of the further transfer to the moon.

2. FROM CAPTURE TO OPERATIONAL ORBIT

2.1 Operational Orbit

Scientific requirements demand a polar and eccentric (height: 300 x 3,000 km) operational orbit around the moon with the perilune near the south pole. The orientation of the orbit plane around the moon polar axis is not prescribed.

The gravitational perturbation of the earth affects the evolution of the perilune and apolune depending on the argument of perilune w.r.t. the moon orbital plane. The earth gravity causes a steady decrease of this argument of perilune. When it is above 270 deg, the perilune height increases; below this value, the perilune height decreases. The semi-major-axis remains constant, so that the apolune height moves inverse to the perilune height.

At the start of the operational phase the argument of perilune w.r.t. the moon orbital plane is 294 deg (or 290 deg w.r.t. the moon equator), the perilune height 300 km and the apolune height 3,000 km. It takes 3 months until the argument of perilune w.r.t. the moon orbital plane reaches 270 deg. During this period the perilune height increases to 438 km. Within the next 3 months, the argument of perilune w.r.t. the moon orbital plane further decreases to 246 deg (or 242 deg w.r.t. the moon equator) and the perilune height decreases back to 300 km. About 2.5 months later, the S/C will impact on the moon surface.

2.2 Conditions for Capture

To limit the duration and the fuel usage for the transfer to the moon, capture should be via the Lagrangian point of the earth-moon system which is between the earth and the moon.

An almost continuous spiral with thrust nearly opposite to the velocity starts around capture and brings the S/C to the operational orbit around the moon (see Fig. 6).

The required pre-capture orbit is obtained by a backward integration of the moon spiral, starting at the operational orbit and ending when the S/C leaves backward the moon sphere of influence. For leaving the moon in the backward integration, the apolune radius has to be increased to 60,000 km.

The only free parameters in the backward integration are the start epoch in the operational orbit and the orientation of the operational orbit plane around the moon polar axis. A backward transfer via the Lagrangian point between the earth and the moon to an elliptic orbit around the earth is only possible for a limited range of orientations (120 deg).

The resulting orbit around the moon has an apogee radius around 300,000, a perigee radius above160,000 km and an inclination to the moon orbital plane around 12 deg. The apogee is almost at the ascending node in the moon orbital plane.

3. FIRST 6 MONTHS

The distance to the earth during this first 6 months after launch and the EP on/off periods are shown in Fig. 1.

3.1 Perigee raising above radiation belts

Ariane 5 has injected the S/C in GTO, with a perigee radius of 7,029 km, an apogee radius of 42,384 km, and, w.r.t. the mean equatorial system of epoch 2000.0, an inclination of 7 deg and an argument of perigee of 178 deg. The right ascension of the ascending node for the actual launch on 27 September 2004 at 23:02 UTC was 169 deg.

The major concern after launch was to get outside the radiation belts a.s.a.p, in order to minimize the degradation of the solar arrays and the startracker CCD.

After a few revolutions in GTO, a spiral started with almost continuous in-plane thrust first along to the velocity and later horizontal in flight direction until the perigee radius reached 20,000 km, which was considered to be the upper limit of the radiation belts. At this point, which was reached on 7 January 2004, the apogee radius was increased to 57,500 km.

3.2 Eclipses at Launch + 6 months

The near mid-night launch resulted in a series of short (< 25 min) eclipses by the earth near perigee straight after launch and a series of long eclipses by the earth near apogee 6 months later in march 2004. For power reasons, the electric propulsion had to be switched off during (no power) and for about 1 hour after the eclipses (battery recharging).

In order to limit the duration of the apogee eclipses 6 months after launch to the maximum allowable duration of 135 min, the apogee radius during the eclipses must be limited to 67,500 km.

Up to 13 March 2004, the date of the longest eclipse, the apogee was further raised to 67,500 km by small tangential thrust arcs near perigee. This also further increased the perigee to 20,700 km.

3.3 Initial conditions for further transfer

During the first 6 months the oblateness of the earth changed the right ascension of the ascending node w.r.t. the mean equatorial system of epoch 2000.0 by -20 deg to 149 deg and the argument of perigee by 37 deg to 215 deg. Relative to the orbit plane of the moon, the final inclination was 33 deg and the final argument of perigee 178 deg.

4. FURTHER TRANSFER TO THE MOON

To get a suitable orbit for moon capture, the apogee has to be increased from 67,500 km to 300,000 km, the perigee from 20,700 km to about 160,000 km and the inclination w.r.t. the orbit plane of the moon reduced from 33 deg to 12 deg. Thanks to the favorable launch date, the argument of perigee w.r.t. the orbit plane of the moon has already the required value of 178 deg.

The strategy to achieve these changes is based on a Hohmann transfer. Up to about mid September thrust arcs near perigee pump up the apogee to a radius near 300,000 km. Subsequently, apogee thrust arcs increase the perigee to a radius of about 170,000 km. The apogee thrust arcs are tilted out of plane to rotate the orbital plane to the final inclination of about 12 deg. This is shown in figures Fig. 2 to Fig. 5.

4.1 Moon Resonances

A considerable part of the perigee raising is obtained by including moon resonances. From an apogee radius of 200,000 km onwards, the moon starts to significantly perturb the orbit when the moon is in the vicinity of the apogee at the same time that the S/C is at apogee. If the moon is ahead of the S/C, the moon will exercise a perturbing force on the S/C in velocity direction. This results in an increase of the perigee height. If the moon is behind the S/C, the perigee is reduced. An optimal perigee increase is obtained when the moon is about 15 deg behind the S/C when the S/C is at apogee. This angle can be controlled by adjusting the orbit periods prior to the resonance.

As shown in Fig. 4, moon resonances are included 1, 2 and 3 moon revolutions before capture. The first and weakest one is at an apogee radius of 230,000 km on 19 August 2004. The next and stronger one occurs 4 S/C revolutions later on 15 September 2004 at an apogee radius of 290,000 km. Finally the strongest resonance is on 12 Oktober 2004, after a further 3 S/C revolutions and at an apogee radius of 324,000 km. Capture occurs 2 revolutions later on 15 November 2004.

4.2 Mitigation of eclipses in moon orbit

The right ascension of the ascending node w.r.t. the orbit plane of the moon is of no relevance to get capture. However, it affects the date of the capture (an increase of 13 deg delays capture by 1 day) and, more important, its value influences the orientation of the operational orbit around the moon and hence affects the eclipse periods in this orbit.

The fuel optimal transfer would hardly change the ascending node. Unfortunately this leads to a period of eclipses by the moon during the moon spiral, making electric propulsion operations nearly impossible. The orientation of the moon operational orbit around the moon polar axis has therefore been biased from the fuel optimal value by about 59 deg in clockwise direction. This shifts the moon eclipse periods forward by 2 months, so that no eclipses occur during the moon spiral.

To get this orientation, the right ascension of the ascending node w.r.t. the moon orbital plane has to be lowered during the transfer phase. As can be seen in Fig. 5, this moves the plane change point for changing the inclination from apogee towards higher true anomalies. Furthermore, this rotation brings the apogee above the moon orbital plane, such that, in addition, a rotation of the line of apsides is needed to reduce the argument of perigee back to 180 deg in the moon orbital plane. This is obtained by shifting the perigee and apogee thrust arcs towards the part of the orbit between apogee to perigee. The extra fuel cost for this is 1.5 kg of Xenon.

4.3 Risk evaluation

Before capture, the S/C is continuously below the moon altitude. An non-anticipated interruption of the electric propulsion can therefore never lead to a collision with the moon. Also a close encounter potentially resulting in an earth escape orbit is not possible.

The nominal transfer trajectory does not include electric propulsion arcs between the last moon resonance andcapture. This provides us 2 revolutions and 27 days to determine and correct previously accumulated navigation errors.

In case at capture the electric propulsion does not start as planned, the S/C will remain in a moon bounded orbit for at least 1 month, giving us sufficient time to recover. This is illustrated in Fig. 6.

5. REFERENCES

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2 SMART-1: Consolidated Report om Mission Analysis, Issue 1.2, J. Cano, M. Hechler, D. Horas, M. Khan, J. A. Pulido, J. Schoenmaekers (ESOC), July 2001, S1-ESC-RP-5506

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Fig. 1. From launch to long eclipse (Thrust: dotted line).

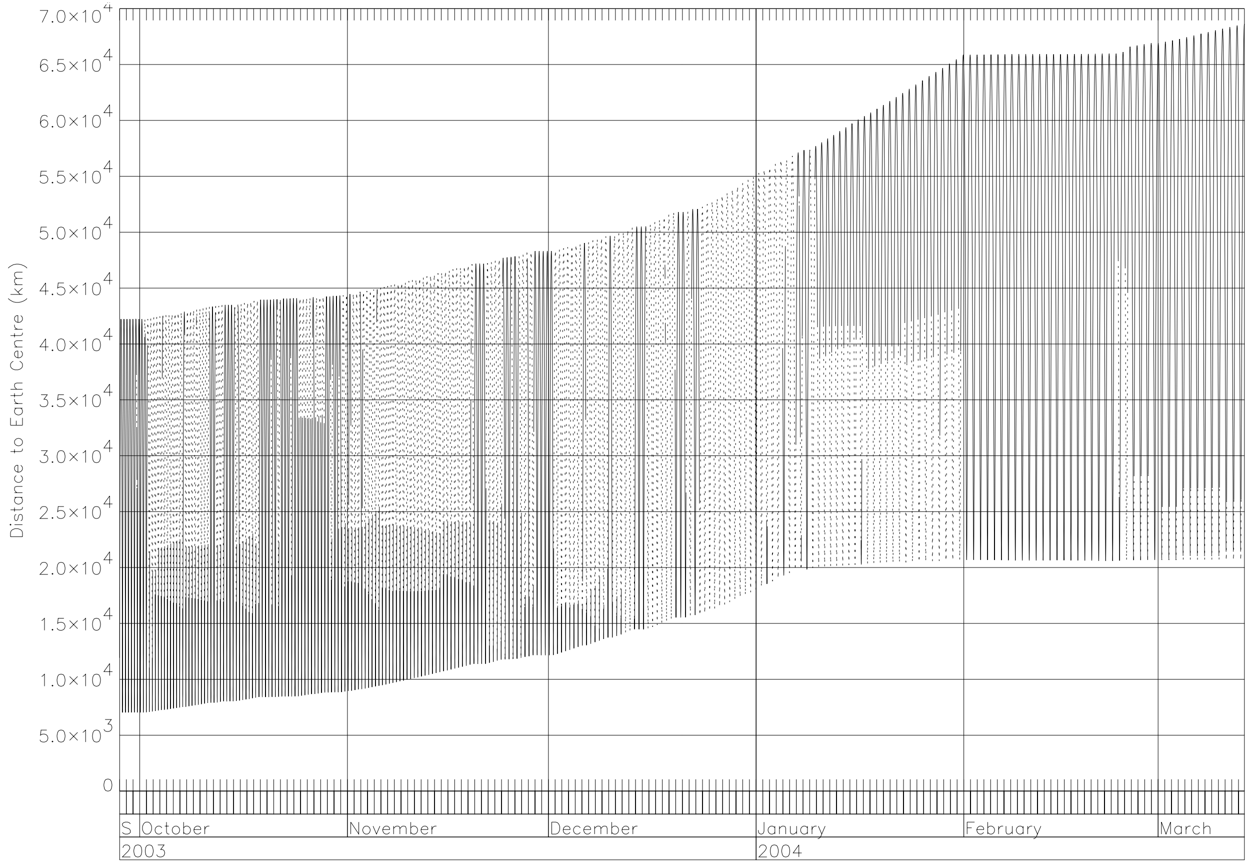

Fig. 2. From long eclipse to moon capture (Thrust: dotted line; Diamond: resonance).

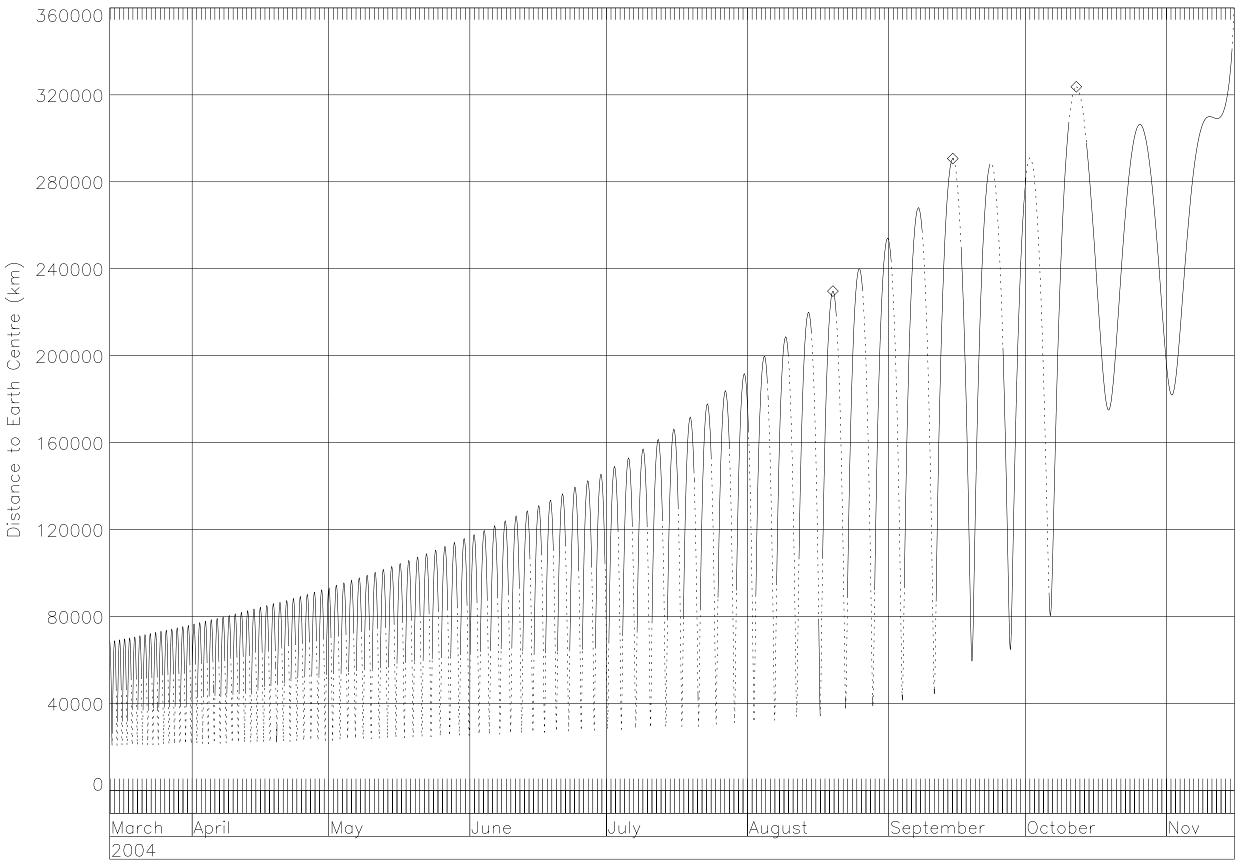

Fig. 3. From long eclipse to moon capture (Thrust: dotted line; Diamond: resonance; Apogee: plus sign).

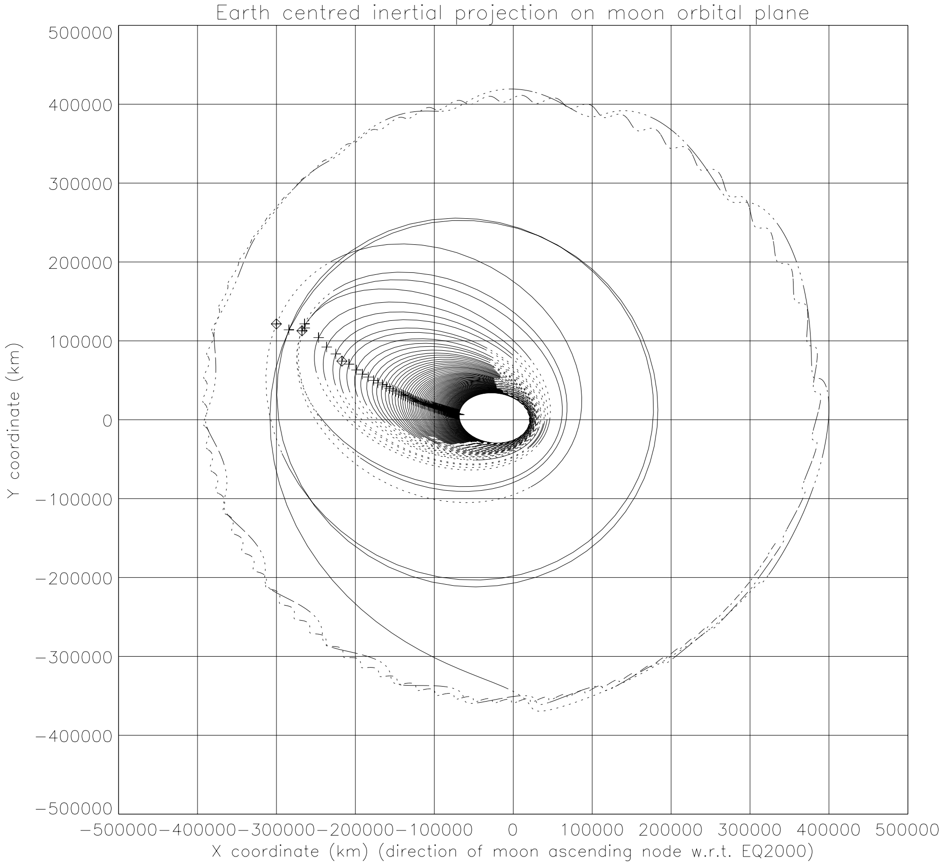

Fig. 4. From long eclipse to moon capture (Thrust: dotted line; Diamond: resonance; Apogee: plus sign).

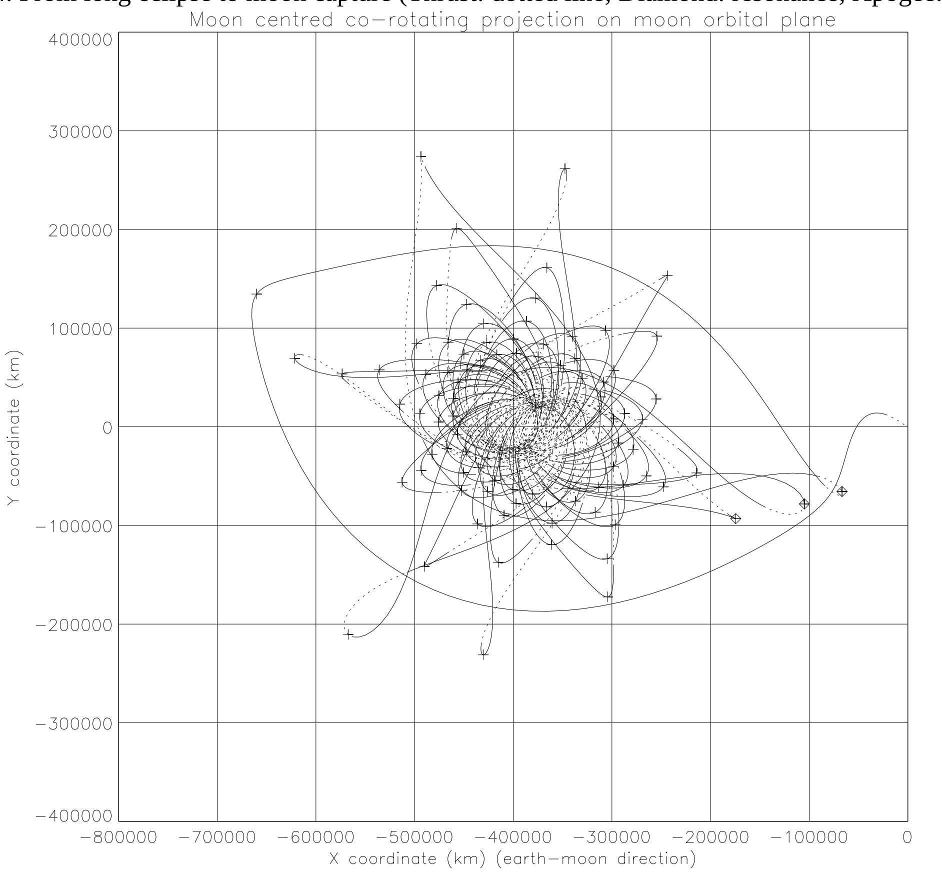

Fig. 5. From long eclipse to moon capture (Thrust: dotted line; Diamond: resonance; Apogee: plus sign).

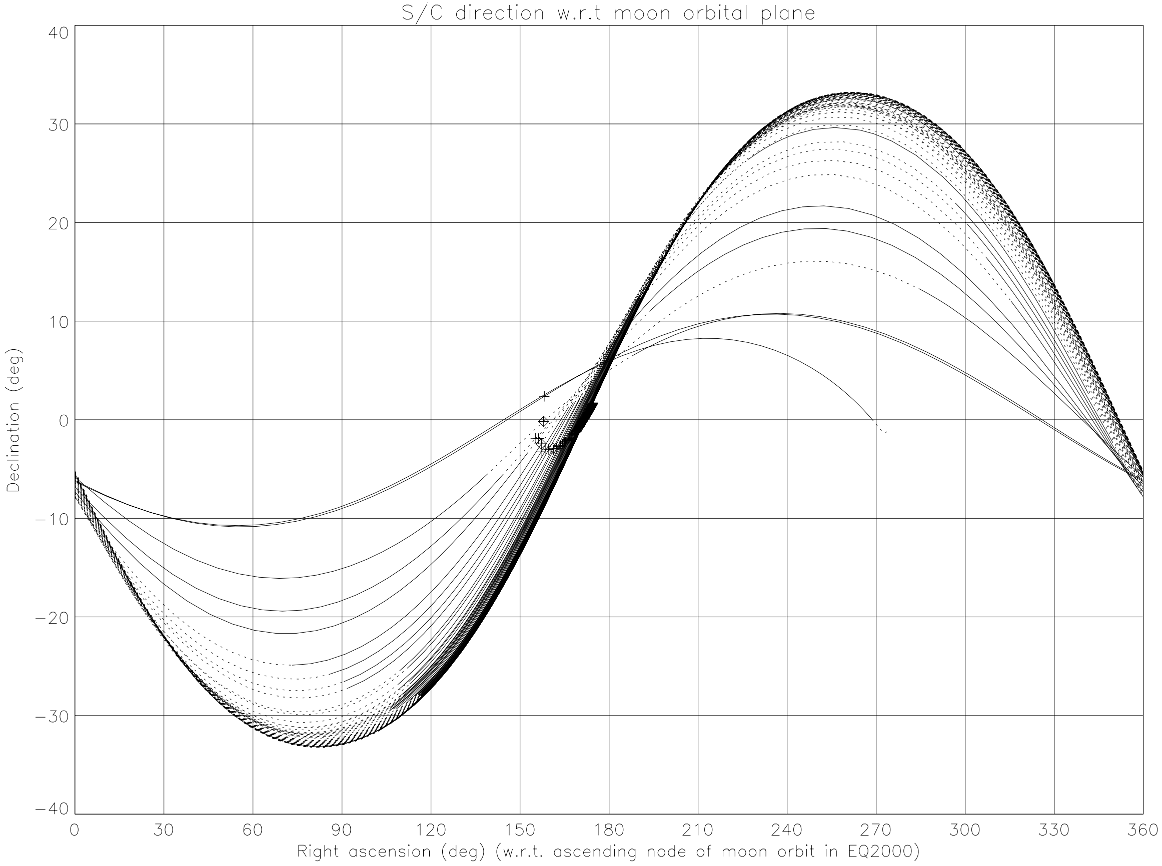

Fig. 6. Moon spiral (Thrust: dotted; Full line: no thrust after capture).