DEIMOS-2 INITIAL ORBIT ACQUISITION OPERATIONS

Mar Luengo Cerrón [1], Fernando González Meruelo [2], Carlos Díaz Urgoiti [3]
Annalisa Mazzoleni [4], Fabrizio Pirondini [5], Miguel Belló Mora [6]

[1] *Deimos Imaging, Parque Tecnológico de Boecillo, Edificio Galileo, Módulo Gris, Oficina 103, Boecillo – Valladolid (Spain) +34 983548923, mar.luengo@deimos-imaging.com*
[2] *Deimos Imaging, fernando.gonzalez@deimos-imaging.com*
[3] *Deimos Imaging, carlos.diaz@deimos-imaging.com*
[4] *Deimos Imaging, annalisa.mazzoleni@deimos-imaging.com*
[5] *Deimos Imaging, fabrizio.pirondini@deimos-imaging.com*
[6] *Elecnor Deimos Space, miguel.bello@elecnor-deimos.com*

Abstract: *DEIMOS-2 is an European fully-private satellite capable of providing sub-metric pan-sharpened imagery. It is owned and operated by Deimos Imaging (Spain), a subsidiary of UrtheCast Corp. (Canada). Successfully launched on June 19, 2014 from Yasny Launch Base (Russia), on board the Dnepr rocket, it was accurately inserted in its injection orbit. On-board DEIMOS-2, a Hall Effect Propulsion Subsystem (HEPS) is in charge of providing manoeuvring capabilities. The Deimos Imaging team main task during the first months of the satellite life was to manoeuvre the satellite from the injection orbit to the target operational orbit, a sun synchronous frozen orbit with a mean altitude of 620 km and LTAN at 10:30. This involved a 20-km altitude raising manoeuvring campaign and the corresponding significant change in inclination. Considering all orbital, platform, payload, power and operational conditions, the manoeuvring campaign strategy was defined in order to decrease the inclination in the first stage, and increase the semi-major axis while decreasing the eccentricity in the second stage. As a result, more than one thousand in-plane and out-of-plane manoeuvres were performed in a timespan of only three months.*

Keywords: *DEIMOS-2, Hall-Effect thruster operation, LEO, orbit acquisition.*

1. Introduction

The DEIMOS-2 mission is fully owned and operated by Deimos Imaging, a Spanish private company subsidiary of UrtheCast Corp. (Canada). The DEIMOS-2 satellite was successfully launched on June 19th, 2014 from the Yasny Launch Base (Russia), on board the Dnepr rocket together with other 32 satellites from 17 different countries. Beyond any positive expectation, the first images were acquired and produced within just 12 hours from separation and LEOP was completed without any blocking issues after a week. Commissioning activities and Calibration and Validation (CAL/VAL) phases started immediately after, only one week after launch.

The mission is fully dedicated to commercial Earth Observation. The electro-optical payload is a push-broom camera with TDI sensors, with a panchromatic and 4 multi-spectral bands: Red (R), Blue (B), Green (G) and Near-Infrared (NIR), capable of providing 75-cm pan-sharpened imagery with a nominal swath of 12-km wide at nadir.

DEIMOS-2 is an agile spacecraft, whose mobility implies roll manoeuvres within a range of ± 45º and pitch manoeuvres within ± 30º with less than 0.15º of pointing error. It is also able to concatenate several single strip images in a multi-pointing operation as well as perform 60º-pitch tilting manoeuvres in less than 90 sec, to acquire single-pass stereo images.

Once in orbit, the main activity during the first months was to manoeuvre the satellite from its injection orbit to the desired orbit. The mission analysis study aimed at placing the desired orbit in a Sun Synchronous and Frozen Orbit (SSFO) with a mean altitude of 619.6 km, LTAN at 10:30 and 14+13/16 orbits per day. As the injection orbit and the target orbit were not the same, (see section 3.4 for further details) the orbit acquisition phase involved an almost 20-km altitude raising manoeuvring campaign and a 0.03° inclination lowering campaign to acquire SSFO conditions. This was a great challenge, as the on-board Hall Effect Propulsion System (HEPS) is capable to provide the satellite with only 10 mN nominal thrust. Due to its low thrust level, more than one thousand manoeuvres including out-of-plane and in-plane ones were performed during a time span of three months.

In order to define the operational strategy for the manoeuvring campaign, several constraints were considered, some of them as a result of the commissioning activities and others coming from orbital evolution considerations.

After several propulsion system tests during commissioning, it was noticed that HEPS needed some maintenance activities between different sets of manoeuvres (see extensive description of these activities in section 4.4.1) in order to guarantee a stable and predictable thrust level.

Taking into consideration the platform constraints, on one hand it was mandatory to maintain battery voltage at a pretty high level and on the other hand it was completely forbidden for the payload (which is formed by an Electro-Optic Subsystem, EOS) to point to the Sun, fact which blocked the possibility of manoeuvres taking place at certain regions of the orbit.

Evaluating all these constraints together with the natural orbit evolution (e.g. arg. of perigee, perigee drift), the final strategy for the manoeuvre campaign was decided: it was agreed that the campaign should be split in two different stages. The first one, aimed to decrease the mean inclination of the orbit for which the out-of-plane manoeuvres needed to be performed. For this stage, the descending node was chosen as the optimal point in order to achieve the maximum inclination change without interfering with the charging of the battery during sunlit periods. The second phase, with the objective to increase semi-major axis while decreasing eccentricity, required manoeuvres in the apogee.

The objective of this paper is to provide a comprehensive overview of the DEIMOS-2 Initial Orbit Acquisition operations. Starting with a description of the HEPS and its operational modes, this paper encompasses the operational orbit selection, the initial conditions and constraints to be considered for the definition of the manoeuvring campaign, the actual execution of the manoeuvring campaign and the current orbit evolution as result of the manoeuvres performed.

2. Hall Effect Propulsion Subsystem (HEPS)

2.1. Description

On board DEIMOS-2, a Hall Effect Propulsion System (HEPS) is in charge of providing the satellite with the manoeuvring capability. Its performance is summarized in Table 1.

Table 1.: Main characteristics of DEIMOS-2 HEPS.

Parameter	Requirement
Fuel Mass	3 kg ± 5%
Thrust	≥ 10 mN
Specific Impulse	> 1,000 s
Total Delta-V	> 50 m/s (actual value around 100 m/s)

HEPS consists on Thruster Head Unit (THU), Power Processing Unit (PPU) and Xenon Feeding Unit (XFU) with a design based on redundancy except for the anode components.
THU is the equipment that accelerates propellant gas to generate thrust, being the thrust vector parallel to the EOS line of sight (see Figure 1). Electric power is supplied to the THU by the PPU and propellant gas by the XFU.

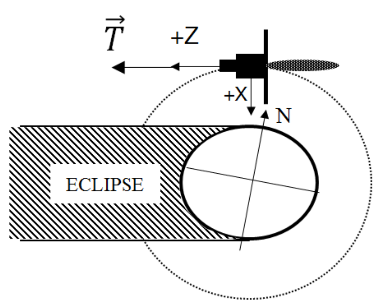

Figure 1.: Thrust vector aligned with EOS line of sight.

The function of XFU is the accurate control of the gas xenon propellant supply to thruster head. The XFU requirements are the following:

- Storage of the propellant for a period of more than 7 years.
- Prevention of unexpected internal or external leakage.
- Independent control of the flow rate on anode and cathode.

The system consists of fuel tanks, isolation valves along with micron-sized orifices, and pressure transducers. HEPS can be divided into the high pressure side and the low pressure one. In the high pressure part, the main fuel tank stores most of the available Xenon at a pressure of 80 bar. In the low pressure side, two accumulator tanks operate at a nominal pressure of 1.5 bar, feeding with Xenon the cathodes and the anode independently. The mass flow through the system is controlled by the pressure of each of the tanks. For further information about the design of a similar system, see [1].

2.2. Operational Modes

HEPS has two different operational modes that can be selected to control the flow rate through the anode and the cathode:

- HW Control (AMUX): based on a MUX setting, allows performing single long manoeuvres (up to 10 minutes) by means of controlling the pressure tanks in the HEPS interface board. With the telemetry analysis of a test manoeuvres sequence, it is noticed it is not possible to perform one long manoeuvre per orbit because the pressure tanks balance is not positive at the end of each orbit (i.e. the pressure in the accumulator tank of the anode is lower at the beginning of each manoeuvre than in the previous one).

- SW Control: based on OBC control through three different pressure sensors in each of the accumulator tanks. The reference pressure for any of them can be set by telecommand and it can be increased and the decreased depending on the needs. Long manoeuvres are not allowed in this mode because of a saturation of the CAN controller.

3. Mission Analysis

3.1. Initial Orbit Selection

The key drivers for the selection of DEIMOS-2 orbit were the following:

- The orbit shall be sun-synchronous (SSO), so as to obtain images with the same illumination conditions over 7-years lifetime at least. The local time shall be also good for optimal image recording, around 10:30 ± 30’.

- It shall be frozen, meaning that every image acquisition is executed at the same mean altitude, with very small variations during mission lifetime. This feature is important for the homogeneity of the coverage and the preservation of image resolution over the same place (See Figure 2).

- It shall have a Repeat Cycle (RC) of a few days in order to guarantee a short revisit time. The requirement for global coverage at all latitudes with the nominal Field of Regard (FoR) of ± 30º was 7 days at maximum.

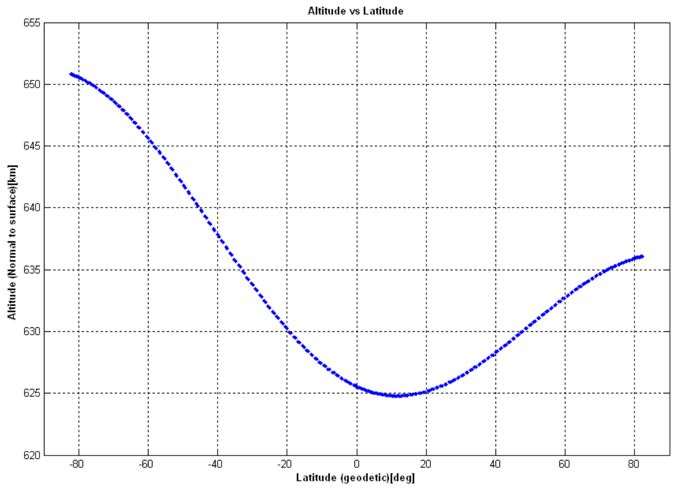

Figure 2.: Expected altitude profile of the reference orbit over the Earth ellipsoid over one RC.

Taking into account all the aforementioned requirements, the mission analysis proved that initial reference altitudes between 610 km and 620 km allow achieving global coverage in less than 7 days with an extended FoR of ±45º, without applying orbit altitude control and at least over the 7-years lifetime. For further details consult [2].

Thanks to these results, the designed optimal orbit resulted in a SSFO orbit, having H_{ref} equal to 619.6 km and 14+13/16 orbits/day. Around this orbit, a series of lower SSFO orbits are also valid, but it is not to be discarded that an initial lower altitude can imply a series of correcting altitude manoeuvres by the end of nominal lifetime.
Table 2 illustrates the main characteristics of the designed orbit.

Table 2.: Main characteristics of the design reference orbit.

Orbit	Href (km)	Orbits/day	RC (days)	LTAN
Design reference	619.6	14+13/16	16	10:30

3.2. Orbit control

Initially, one of the objectives of the mission analysis was the design of a robust orbit, that is without applying any nominal orbit maintenance, able to support satellite operations even in case of main engine failure during the nominal phase.

Since the initial reference orbit altitude is 619.6 km, it is foreseen that the natural evolution of the semi-major axis during the satellite lifetime will allow the payload to work without any degradation in terms of GSD.

After making sure that an orbit-decaying mission would have been feasible, the next problem was to avoid manoeuvres to maintain LTAN within a band from 10:00 UTC to 11:00 UTC approximately. A small delta inclination added to the SSO inclination at beginning of life is able to trigger the achievement of SSFO just at the middle of the mission lifetime minimizing the LTAN delta throughout the 7 years. Based on this strategy, the target nominal inclination of the launcher injection orbit was selected to be slightly different with respect to the SSO inclination.

3.3. Injection orbit

The Dnepr rocket was going to carry 32 satellites, many of them without propulsion, so an agreement with ISC Kosmotras ratified that DEIMOS-2 would have to manoeuvre to reach its final orbit. With such decision, the HEPS became a critical element for the commissioning phase.

3.4. Mission Analysis Summary

To summarize, 4 types of orbit were characterised before launch:

- The initial reference orbit: design reference SSFO
- The target nominal orbit: the initial reference orbit + Di
- The launcher target orbit: the launcher target orbit close to the target nominal orbit
- The injection orbit: the injection orbit with launcher injection errors

Consequently, the DEIMOS-2 operations team had been preparing to implement orbital change manoeuvres to compensate:

1. The launcher injection errors, i.e. the errors of injection orbit with respect to the launcher target orbit.
2. The gap between the launcher target orbit and the target nominal orbit.

4. Initial Manoeuvring Campaign (Orbit acquisition)

4.1. Injection conditions analysis

The Dnepr rocket placed DEIMOS-2 in a very good orbit; referring to Table 3 and Table 4, the error in inclination was basically null and the error in semi-major axis was small and even favourable to the next orbit rise. Errors are intended as the differences in orbital parameters of the injection orbit with respect to the launcher target orbit.

The error in eccentricity was not considered in the preliminary budget since it can be compensated by in-plane acquisition manoeuvres, and neither the error in RAAN was taken into account since it lies well within the LTAN margins.

The Delta-V at injection was approximately null (see Table 4), so all the effort was dedicated to reach the target nominal orbit. This favourable situation let the saving of ~9 m/s.

Table 3.: Dnepr 3σ injection errors versus injection errors at DEIMOS-2 launch. Errors are referred to the launcher target orbit parameters.

Dnepr injection errors at Href = 600 km		
	3σ	**operational**
Delta sma [km]	± 5.5	+0.264
Delta inclination [deg]	± 0.045	+0.001
Delta eccentricity	~ 6e-4	1e-4
Delta RAAN [deg]	± 0.060 (14 s)	-
Delta argp [deg]	-	+3.73

Table 4.: Theoretical Delta-V to compensate 3σ injection errors versus applied Delta-V at DEIMOS-2 separation.

Delta-V at injection		
	allocated (3σ errors)	**operational**
Delta-V_{sma} (m/s)	2.97	0
Delta-V_{incl} (m/s)	5.93	0.13
Total Delta-V_{inj} (m/s)	***8.91***	***0.13***

4.2. Target orbit acquisition

As previously outlined, the launcher left the satellite in an orbit with lower altitude and higher inclination with respect to the target, situation that would have driven the satellite to very bad conditions in terms of orbital performance. Thus, a manoeuvring campaign was necessary to obtain good orbital conditions.

To fulfil also the payload requirements in terms of resolution and line rate, coverage and revisit time:

- The altitude should be incremented.
- Eccentricity and argument of perigee should be addressed towards frozen conditions.

To generate a higher right ascension of the ascending node (RAAN) rate with respect to the sun synchronous rate, such as to obtain "almost" sun synchronous initial conditions:

- The inclination should be lowered.

On the contrary, a too high RAAN rate would have lead the LTAN to exceed its upper boundary (11:00 UTC approximately) rapidly.

Out-of-plane manoeuvres were planned first to lower the inclination 40 days after launch, since it was necessary to wait for the perigee positioning around 90º. Once reached the target inclination, at the 70th mission day in-plane manoeuvres started officially, in the most suitable orbit conditions to increment semi-major axis, while lowering eccentricity and argument of perigee.

4.3. Constraints

Before deciding the manoeuvring campaign strategy, it was needed to evaluate which were the constraints to be considered in terms of platform, payload activities and orbit evolution. It is worth mentioning the hereafter explained constraints were analysed for each of the one thousand performed manoeuvres, as the mean argument of perigee and sun position were continuously changing. Furthermore, the fact of the daily re-calculation of the constraints' effect, introduced an extreme complexity in the allocation of the manoeuvres.

4.3.1. Spacecraft power budget

Considering that thruster manoeuvring is the operational mode which demands more power from the satellite, it was required to analyse the power balance at the end of each orbit to confirm the ability of performing one manoeuvre per orbit. Additionally, in order to preserve the long-term capacity of the battery, a DoD of less than 20% is mandatory.

With these constraints and after the commissioning activities in which several test manoeuvres were performed, different values for battery charge current were set during the manoeuvring campaign to allow a complete charge of the battery between each orbit. Furthermore, as solar panels are fixed with respect to satellite platform, spacecraft attitude is very important to maximize the power generated by solar panels. As consequence, the main conclusion considering power budget as a constraint is out-of-plane manoeuvres shall be performed during eclipse.

4.3.2. Sun incidence angle in the payload

The satellite's payload is composed by an Electro-Optical Subsystem (EOS) which must be subjected to a very accurate range of temperatures. Additionally, payload cannot point directly to the Sun not to damage the sensors.

Based on pre-launch analysis, both constraints were satisfied if the angle between the Sun, as seen from the spacecraft, and the boresight of the EOS was higher than 5º. Nevertheless, flight data obtained during LEOP and early Commissioning activities proved that even when the condition was fulfilled, temperature in the mounting base of the secondary mirror reached values close to the maximum allowed for that component. Analysis of telemetry and Flight Dynamics data showed that the Sun was reaching directly to the base, heating it up.

As a consequence, a minimum angle of 65º between the optical axis of the EOS and the Sun was imposed during the complete manoeuvre to safely execute it. This meant that a greater part of the orbit - spanning 130º in true anomaly- was not going to be available to perform in-plane manoeuvres. And even more, this restriction was not fixed with respect to the Earth, but it was moving as the Sun moved southwards relatively to the Earth as autumn's Equinox approached.

Although manual review of each manoeuvre has been performed not to fall in the forbidden zone, Elecnor Deimos' Flight Dynamics Software "fly4EO" was updated in order to check for this constraint to be satisfied for every manoeuvre before delivering it to the Mission Control Subsystem.

As result of a theoretical analysis, it was computed that the sun angle between EOS and Sun vector shall be higher than 69º (including margins), to avoid any issue. Considering the manoeuvring campaign was performed during summer season, the following figure shows which was the forbidden area to perform in-plane manoeuvres due to this constraint. This scheme was analysed every day as Sun position, Earth precession and mean orbital elements variation directly affected the determination of the forbidden area.

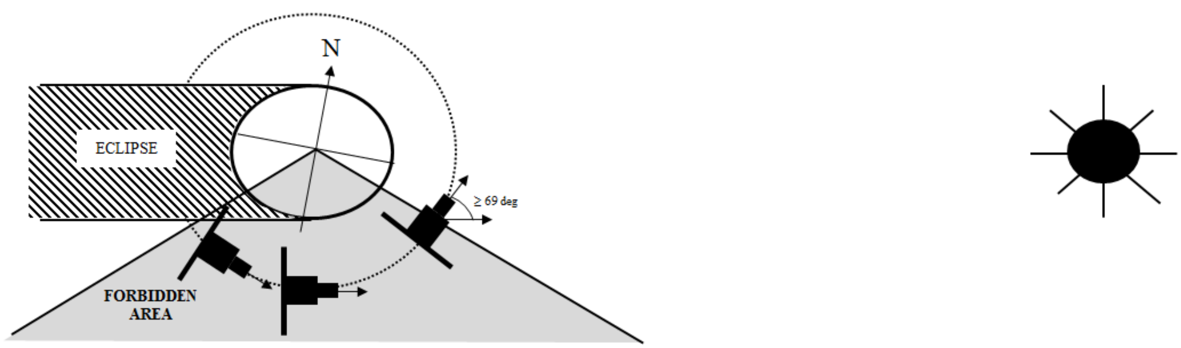

Figure 3.: Forbidden area to perform +DV in-plane manoeuvres considering the sun angle incidence in the line of sight of EOS must be less than 69º.

4.3.3. Propulsion System Performance

After commissioning activities, it was determined the propulsion system needed maintenance activities after a set of performed manoeuvres.

During the manoeuvring campaign, telemetry of each manoeuvre was analysed in order to check its correct execution in terms of voltage, current and temperature through all the hardware components of the propulsion subsystem. It was detected not all the manoeuvres were successfully performed due to not enough flow rate through the cathodes probably caused by propellant particles blocking the cathodes orifice. This issue was easy to predict some manoeuvres in advance, as the thruster telemetry showed degraded conditions while the mass flow decreased.

Due to this behaviour, the estimated duration of the manoeuvre campaign was not accurately predicted as, in addition to the HEPS extra maintenance needed, the thruster operational mode was continuously changed to look for the optimal conditions to ignite the cathode.

4.3.4. Payload operation

The CAL/VAL operation and the initial commercial activities were performed simultaneously with the manoeuvring campaign. Due to the commercial nature of DEIMOS-2 mission, manoeuvres could sometimes be displaced from the optimum point in order to allow payload activities (e.g. image acquisition, download). The available ground stations to download data are all located in the Northern hemisphere (Spain, Sweden and Canada). Additionally, medium latitudes are the more demanded zones in terms of target locations, which shall be considered in the orbits in which the satellite crosses this area as the satellite cannot manoeuvre and acquire simultaneously.

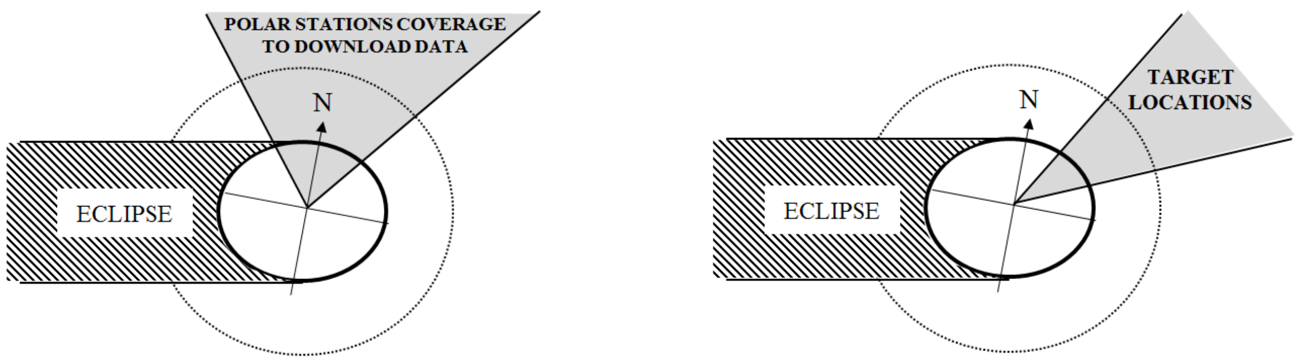

Figure 4.: Zones in which payload operation is more demanded.

4.3.5 Perigee drift

+DV in-plane manoeuvres shall be performed in the apogee in order to obtain the maximum efficiency in the effect of decreasing eccentricity and increasing semi-major axis. Perigee drift (around-3.5º/day) determines the timespan to perform these manoeuvres, fulfilling the mentioned constraints.

4.3.6 Summary of constraints

The following table shows a summary of the described constraints and the consequences in the design of the manoeuvring campaign for each of them.

Table 5.: Summary of the constraints

Constraint	Description	Consequence
Satellite power budget	Battery DoD must be less than 20% Power balance shall be positive in each orbit	Out-of-plane manoeuvres can only be performed in eclipse
Sun incidence angle in payload	Avoid overheating of the EOS	There is a forbidden region at the exit of eclipse
Propulsion System Performance	Maintenance activities are needed after a set of manoeuvres	Estimated manoeuvring campaign time can vary
Payload activities	Not possible to download data and acquire desired targets if manoeuvres are performed in medium latitudes	Some manoeuvres shall be moved with respect to the optimal orbit position
Perigee drift	Natural perigee drift determines where to	Perigee drift determines the timespan in

	perform optimal manoeuvres	which manoeuvres in the apogee fulfils the other constraints.

4.4. Strategy

The calibration of propulsion system started one week after the separation. This phase allowed the operations engineers to evaluate the constraints previously mentioned and to decide the best strategy to reach the target orbit.

The following figure shows a scheme of the objectives and activities for each phase. Two different phases were identified taking into account the perigee drift, which determined when to start the in-plane manoeuvres. The apogee was estimated to be out of the forbidden region one month after the beginning of the manoeuvring campaign so, during that time span, out-of-plane manoeuvres could be performed to decrease the inclination. It is worth mentioning the strategy was defined considering as target orbit the one obtained as result of the initial mission analysis, with H_{ref} equal to 619.6 km. Consequently, the predicted Delta-V and number of estimated manoeuvres were calculated based on this orbit.

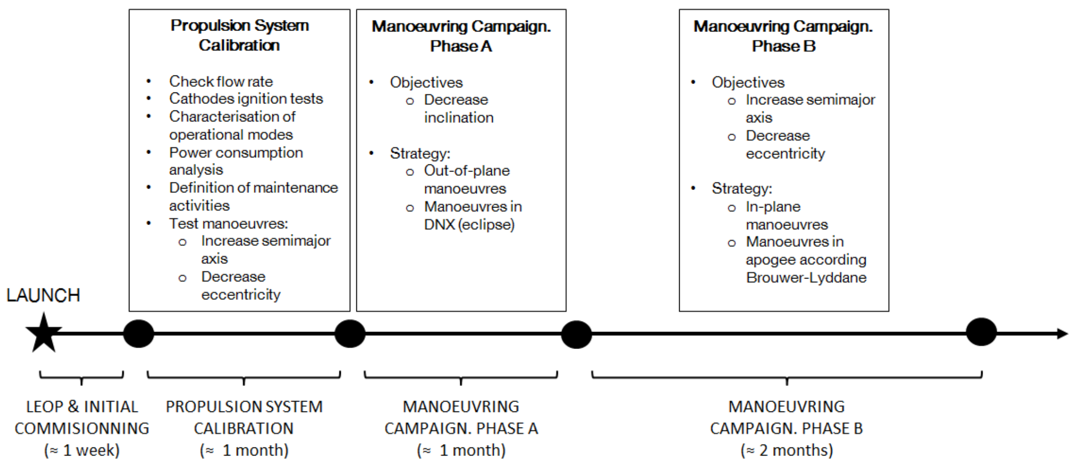

Figure 5.: Scheme of the objectives and activities of different phases of the maneuvering campaign.

4.4.1. Propulsion system calibration

Before determining any strategy, a campaign of tests to calibrate the propulsion system was performed once the LEOP and initial commissioning were done. The main objective of this phase was to characterise the thruster operation in terms of power consumption, required timing of opening and closing of valves, pointing misalignment and in general, all the parameters that define thruster performance.

During the test campaign, an unexpected low flow rate was detected when the valves of the propulsion system were used during several consecutive days. After an in-depth analysis, it was concluded that the orifices downstream the cathodes and the anode were slightly blocked by the particles that the valves generated in their cycles of continuous opening and closing. This low flow rate caused the manoeuvres to be autonomously cancelled by the satellite OBC because the mass flow was not enough to ignite the cathode. Consequently, a set of maintenance activities were defined when the low mass flow was detected by means of the analysis of the pressure decay rate.

These activities consist in increasing the cathodes accumulator tank pressure and then returning the reference pressure to its nominal value by opening the cathode valve. In this way, the obstructing particles can be blown out. This activity introduced an uncertainty in the estimated time needed to complete the manoeuvre campaign and a slight waste of fuel when the valves are opened for this purpose.

Another result of this phase was the definition of the forbidden regions to perform the manoeuvres to avoid the camera sun pointing. As it has been mentioned in the previous sections, the initial calculation showed the minimum angle between the boresight of the EOS and the Sun should be less than 5º in order to not heat the optical components. However, during this campaign it was noticed manoeuvres that fulfilled this constraint heated the second mirror of the optics because it is mounted in an exposed zone. As result, it was defined a more severe constraint which ended in a forbidden cone bigger than the initial one.

Regarding the power consumption, a higher battery charge current was defined during this initial stage, in order to be able to perform a manoeuvre in each orbit with a final positive power balance. This change was agreed with the manufacturer demonstrating that it was not going to degrade the power subsystem performance. After the manoeuvre campaign was finished, the battery charge current was set back to nominal value in order to lower at maximum the battery degradation.

The manoeuvres performed during this initial stage were executed in the apogee according to Brouwer-Lyddane theory, in order to take advantage of the tests and use the manoeuvres to increase the semi-major axis and decrease the eccentricity.

4.4.2. Manoeuvring campaign. Phase A.

After analysing the mentioned constraints and considering the results of the previous tests, the decided strategy was divided in two different stages.

The first stage of the manoeuvring campaign (phase A), aimed at decreasing the inclination, performing as much manoeuvres as possible in the descending node, which was in eclipse.

In this phase, the only constraints to take into account were the power balance and the ones imposed by the own propulsion system, as EOS sun pointing was always fulfilled because all the manoeuvres were performed in eclipse.

Initial calculations for this phase showed it was needed a delta of inclination of -0.031º, which could be obtained in the worst case with a total of 424 4-minute manoeuvres, and in the best case with a total of 318 6-minute manoeuvres. Considering the different propulsion system modes and the defined maintenance activities, the predicted duration of this phase stage was between 20 and 30 days.

4.4.3. Manoeuvring campaign. Phase B.

As explained in the introduction of this section, considering that the drift-rate for the mean argument of perigee was -3.5º/day, and that the mean arg. of perigee at the beginning of the manoeuvre campaign was 162º, the predicted arg. of perigee by the end of phase A (around one month as result of estimations) would be between 60º and 90º.

This meant that, by the end of phase A, EOS sun pointing constraints could be fulfilled by thrusting at most 15º before reaching the apogee during the first 20 days, and then perfectly aligned with the apogee for the next 40 days, when the perigee would be again in a conflictive position due to payload requirements. As the computed Delta-V needed to increase the semi-major axis is 7.2 m/s (880 4-minute or 660 6-minute manoeuvres), this meant that phase B should last at most 60 days.

During the last two weeks of phase B, re-analysis of the long-term evolution of the orbit would be used to fine-tune the last manoeuvres, to correct for errors accumulated during the whole campaign.

4.5. Execution of the campaign

With respect to the allocated Delta-V budget to compensate the gap from the launcher target orbit, the real Delta-V for orbit acquisition does not differ too much, even if the in-plane Delta-V resulted to be slightly less than the allocated one, before launch. The out-of-plane Delta-V during real operations was instead a bit more than expected. Finally the total Delta-V from the separation point to the reference orbit was estimated as 10.3 m/s. However, it is worth to point out that the target semi-major axis was not acquired as there was a need to start the commercial phase before reaching the target orbit. The final intermediate orbit was evaluated and considered suitable for the nominal lifetime of the mission.

The following tables show the main parameters of the manoeuvring campaign compared with the predicted ones. Table 6 shows the foreseen delta of semi-major axis and inclination and the operational delta, i.e. the real delta achieved after the manoeuvring campaign. Table 7 introduces the comparison between the Delta-V allocated and the real one and, in Table 8 the summary of the orbital parameters is shown for the orbit types described in paragraph 3.4.

Table 6.: Errors in semi-major axis and inclination of the launcher target orbit and injection orbit, wrt the target nominal orbit (predicted and operational respectively).

Orbit Acquisition errors		
	Predicted	**Operational**
Delta sma [km]	+14.8	+11.0
Delta inclination [deg]	-0.031	-0.032

Table 7.: Delta-V budget at orbit acquisition.

Orbit Acquisition Delta-V		
	Predicted	**Operation real**
Delta-V_{sma} (m/s)	8.0	6.0
Delta-V_{incl} (m/s)	4.1	4.3
Total Delta-V_{acq} (m/s)	**13.9** (considering 15% margin)	**11.4**

Table 8.: Summary of the mean orbital parameters in ToD of the injection orbit, the injection operational orbit, and the target orbit.

	launcher target	injection	target nominal
Semi-major axis (km)	6982.97	6983.23	6994.24
Href (km)	604.83	605.10	616.10
Inclination (deg)	97.991	97.992	97.960
Argument of perigee (deg)	0.71	356.98	90
LTAN	10:30	10:29	10:30

Referring to Table 7, the Delta-V for the orbit acquisition phase was about 14 m/s considering the predicted gaps in semi-major axis and inclination and adding a 15% safe margin. In the operational case, it was calculated that HEPS provided 11.4 m/s of Delta-V of which the 10% spent for HEPS test and maintenance.

The campaign was designed to achieve the goals mentioned in paragraph 4.2, taking into account the restrictions imposed by the platform (i.e. power budget, propulsion subsystem degraded functioning), payload (i.e. avoid sun pointing of the camera) and the current orbit (i.e. drift of the argument of the perigee).
Considering all these restrictions, the campaign was finally divided in the two phases previously explained:

- First phase, composed of more than 250 out-of-plane manoeuvres, aimed at decreasing the mean inclination, performed at the descending node, during one month.
- Second phase, composed of more than 600 in-plane manoeuvres, aimed at increasing the semi-major axis while decreasing the eccentricity, performed mainly during two months.

After a total change of mean inclination of approx. -0.032º, semi-major axis of +11 km and eccentricity of -0.016, the final orbit was achieved by 1st November, 2014.

Table 9.: Comparison between predicted and operational out-of-plane manoeuvres and spent time for Phase A.

		Predicted	Operational
Delta inclination [deg]		-0.031	-0.032
Number of manoeuvres	**8-min manoeuvres**	-	4
	6-min manoeuvres	-	123
	4-min manoeuvres	-	143
Total number of out-of-plane manoeuvres		212/8 min 318/6 min 424/4 min	270
Total time for out-of-plane manoeuvres [days]		30	28

Table 10.: Comparison between predicted and operational in-plane manoeuvres and spent time for Phase B.

		Predicted	Operational
Delta semi-major axis [km]		13.3	10
Number of manoeuvres	**8-min manoeuvres**	-	14
	6-min manoeuvres	-	190
	4-min manoeuvres	-	441
Total number of in-plane manoeuvres		440/8 min 660/6 min 880/4 min	645
Total time for in-plane manoeuvres [days]		60	59

As illustrated in the figure below, the prediction of the mean local solar time (MLST) over 10 years is acceptable. Indeed, the most probable evolution of the orbit can be considered that of 50% confidence level (CL) of solar activity, having the 11:15 as maximum value.

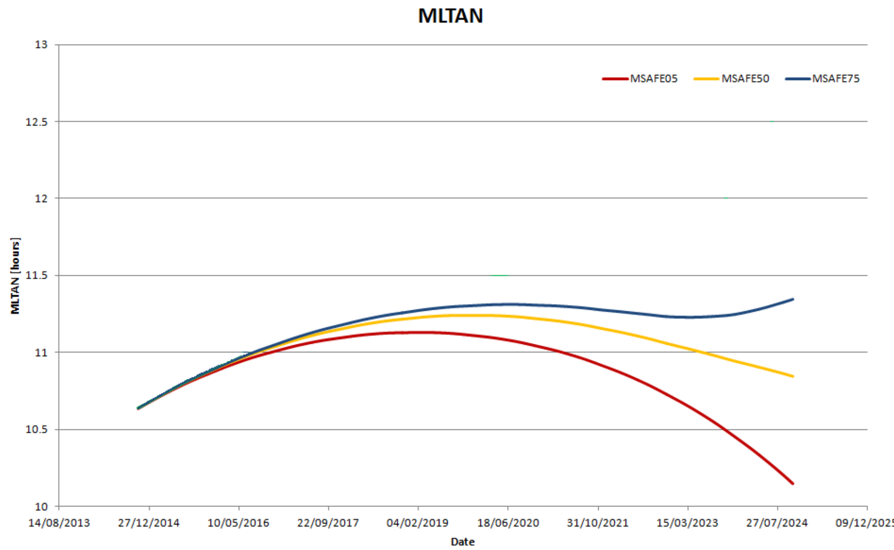

Figure 6.: Mean local solar time prediction at the end of the maneuvering campaign , for 5%, 50%, 75% solar activity confidence level, over 10 years (extended mission lifetime).

Figures below give evidence to the executed campaign. Altitude and inclinations gaps introduced by manoeuvres are clearly visible. Furthermore, with a LTAN of 10:30, the inclination is expected to decrease of 0.038 deg/year on average, due to Sun and Moon gravitational attractions and Earth tides. This is important to foresee MLST behaviour. The eccentricity and perigee lowering are then illustrated like eccentricity factor in Figure 7.

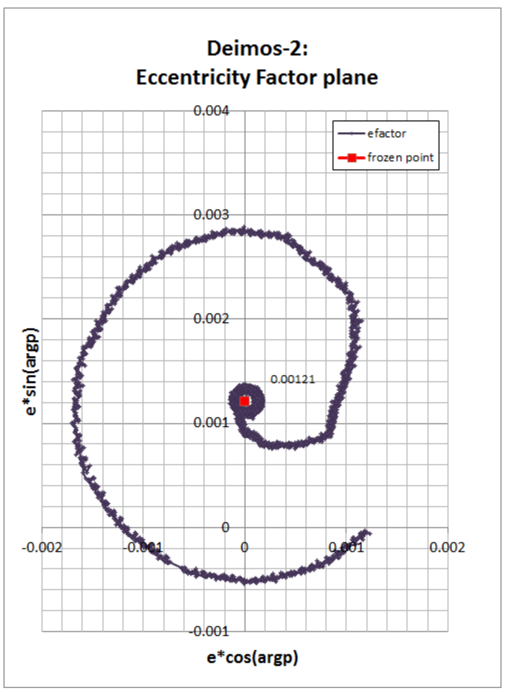

Figure 7.: DEIMOS-2 orbit eccentricity factor according to Kozai mean elements theory. Frozen conditions achievement.

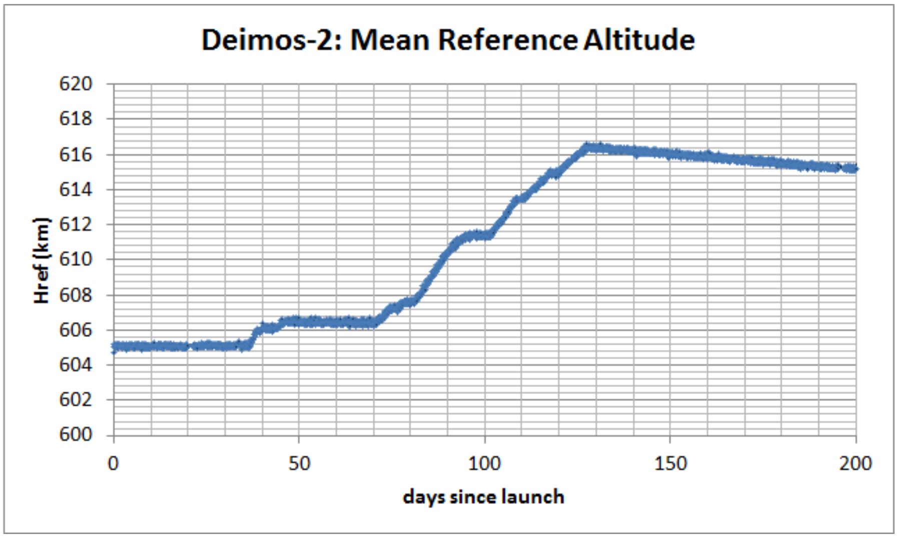

Figure 8.: DEIMOS-2 mean reference altitude during Commissioning phase.

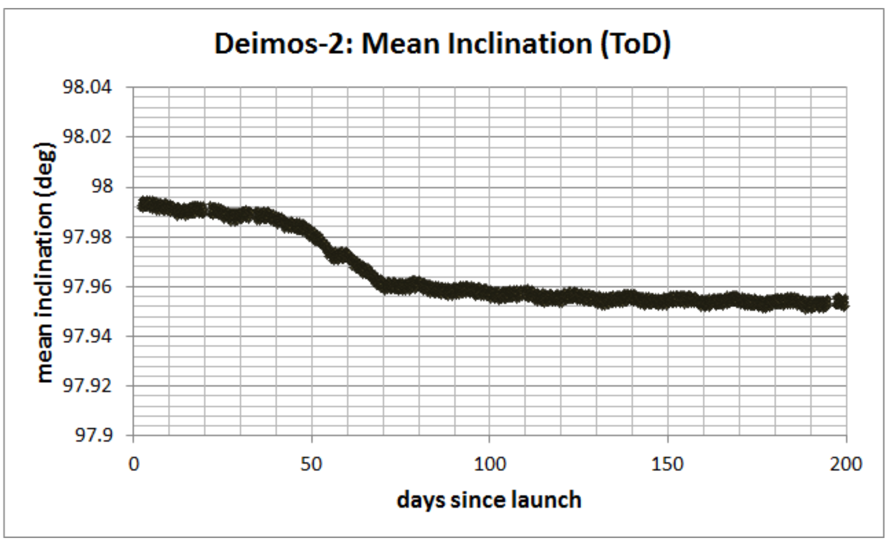

Figure 9.: DEIMOS-2 mean inclination (True of Date) during Commissioning phase.

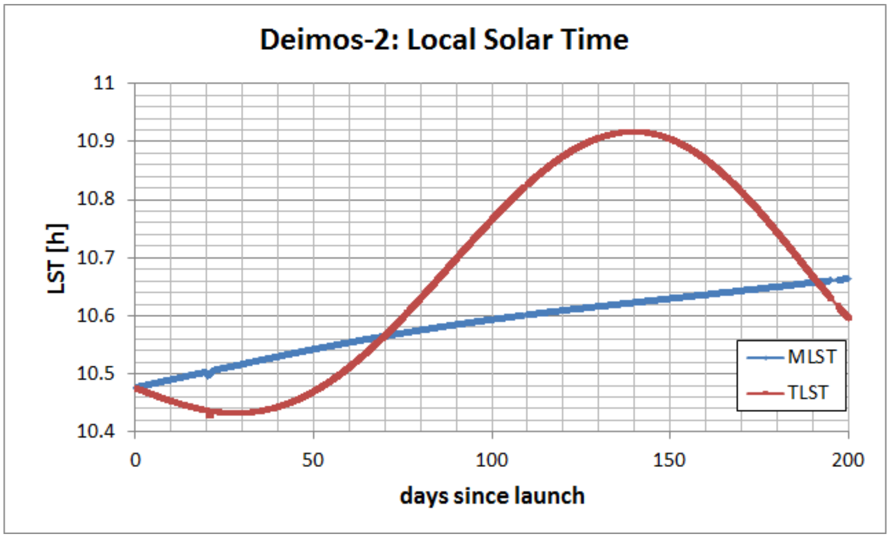

Figure 10.: DEIMOS-2 mean and true local solar time during commissioning phase.

4.6. Fuel consumption analysis

The total amount of propellant used during the campaign, including propellant used during the propulsion system maintenance activities, is approx. 300g, of a total of 3.2 kg available on-board at the beginning of the mission. Using the data currently available, we have that:

- Satellite Dry Mass at Launch (m_0): 294.3 kg
- Satellite Fuel Mass at Launch (m_{0p}): 3.2 kg
- Fuel Mass used so far ($m_{0p} – m_p$): 305 g
- Remaining Fuel Mass (m_p): 2.895 kg

This means that DEIMOS-2 still has around 90% of its original fuel. This corresponds to an available $\delta v > 90\ m/s$, enough for orbit maintenance (if needed), collision avoidance and end-of-life disposal activities.

5. Conclusions

The initial orbit acquisition campaign represented a true challenge for the DEIMOS-2 Operations team, not only from the technical point of view, but also from the planning side, as close to one thousand manoeuvres were performed while early operation activities took place.

With the clear and common objective in mind of reaching the target orbit, the DEIMOS-2 team, and the satellite manufacturer team cooperated synergically leading to a successful three-month campaign. This achieved team building allowed a quick and efficient reaction of the involved members to the issues detected, and they were solved promptly.

For the three-month campaign the DEIMOS-2 team’s know-how has been strengthening at spacecraft bus level as well as at mission planning and flight dynamics level. Furthermore, a set of thoroughly in-orbit validated operational procedures for the Propulsion Subsystem were developed and they are currently in use.

Thanks to the propellant-saving policy over the commissioning phase, more than 90% of the launch propellant is currently available for manoeuvres including possible orbit corrections, collision avoidance and end-of-life disposal activities.

To summarise, after increasing the orbit altitude more than 10 km and decreasing the inclination 0.032 degrees, the objective to place DEIMOS-2 satellite in an orbit where little-to-none station keeping activities are foreseen was successfully achieved, fact which involved around one thousand manoeuvres within a period of three months.

6. References

[1] “Development of Xenon feed system for a 300-W Hall-Thruster”. Y. Kim, S. Kang, Y.H. Jeong, J. Seon, J.H Wee, H. Yoon, J. Lee, M. Seo, W. Soo. 31st International Electric Propulsion Conference , Ann Arbor, Michigan, USA

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