MISSION ANALYSIS FOR THE DON QUIJOTE PHASE-A STUDY

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ABSTRACT

The Don Quijote Phase-A study is a definition study funded by ESA and devoted to the analysis of the possibilities to deflect a Near Earth Object (NEO) in the range of 300-800 m diameter. DEIMOS Space S.L. and EADS Astrium have teamed up within this study to form one of the three consortia that have analyzed these aspects for ESA. Target asteroids for the mission are 1989 ML, 2002 AT_4 and Apophis.

This paper presents the mission analysis activities within the consortium providing: low-thrust interplanetary rendezvous Orbiter trajectories to the target asteroids, ballistic interplanetary trajectories for the Impactor, Orbiter arrival description at the asteroids, Orbiter stable orbits characterization at the asteroid, deflection determination by means of a Radio Science Experiment (RSE) as well as the mission timelines and overall mission scenarios.

INTRODUCTION

The Don Quijote Phase-A study is a definition study funded by ESA [1] and devoted to the analysis of the possibilities to deflect a Near Earth Object (NEO) in the range of 300-800 m diameter in size. DEIMOS Space S.L. and EADS Astrium have teamed up within this study to form one of the three consortia that have analyzed these aspects for ESA in the past months. Don Quijote mission originating study is summarized in [2] and ulterior mission analysis activities performed at ESA are described in [3].

The mission is based on the use of two spacecraft: an Orbiter and an Impactor. The Orbiter will make use of solar electric propulsion (SEP) to reach the target asteroid. Once the Orbiter has arrived to the asteroid, the Impactor spacecraft will be launched to arrive to the target with relative collision velocities in excess of 10 km/s. In the meanwhile, the Orbiter will characterize the target asteroid orbit, shape, gravity field and properties to enable a precise determination of the achieved deflection when the Impactor hits the asteroid. Once the primary objective is achieved, the Orbiter is also meant to release a surface package on the asteroid for in-situ ground experimentation[1].

For the purpose of this study, the selected target asteroids are 1989 ML, 2002 AT_4 and Apophis (this one only for the Orbiter) [4].

[1] This option was assessed in the frame of a "Don Quijote Plus" scenario aiming at identifying the implications at system level of some additional secondary objectives. Here, the release experiment was analyzed in terms of release strategy and dynamics. The design of the surface package itself was not requested by ESA.

DEIMOS Space has been in charged of the mission analysis activities within the consortium providing: low-thrust interplanetary rendezvous Orbiter trajectories to the target asteroids, ballistic interplanetary trajectories for the Impactor, stable orbits characterization at the asteroid, deflection determination analyses by means of a Radio Science Experiment (RSE) as well as a wealth of other mission results as delta-V budgets, mission timelines, etc.

The results of the DEIMOS analysis contributed very significantly to the definition of requirements and constraints for a minisatellite project, currently being considered by ESA as possible in-orbit technology demonstrator.

TARGET ASTEROID PROPERTIES

The assumed properties of the three selected asteroids are summarized in the following table. It can be observed the large difference in semi-major axis and eccentricity of 2002 AT_4 and the rest of the asteroids, which is of particular importance for the design of the Orbiter. Contrary to that the inclination is smaller than for the other two asteroids.

Table 1: Assumed properties of the proposed target asteroids

Feature	2002 AT_4	1989 ML	Apophis
Period (y)	2.550	1.435	0.886
Semi-major axis	1.866	1.272	0.922
Eccentricity	0.447	0.137	0.191
Inclination (deg)	1.51	4.38	3.33
R.A. of Ascending Node (deg)	323.7	104.4	204.5
Argument of perihelion (deg)	202.8	183.3	126.4
Synodic period (y)	N/A	3.299	7.772
Maximum considered mass (kg)	7.3E+10	6.7E+11	8.3E+11

MISSION CONSTRAINTS

A set of mission constraints were defined by ESA on programmatic basis that translated on a number of constraints directly applicable to the design of the transfer trajectories to the asteroids:

- Launch of all elements of the space segment shall occur within the 2011-2017 timeframe
- The mission shall be completed by 2020 at the latest
- European launchers shall be privileged: Vega or Soyuz 2-1B trying to favor as much as possible the smaller launchers (Vega and eventually Dnepr)
- The Impactor shall be launched after the Orbiter has arrived at the asteroid
- The Orbiter shall use SEP and the Impactor chemical propulsion for their transfers
- The Orbiter shall perform 6 months pre- and post-impact asteroid analysis (asteroid CoG orbit determination campaigns) as a minimum
- A minimum variation of the asteroid CoM orbit semi-major axis of 100 m shall be achieved with the impact
- The RSE shall be such to measure the imparted deviation with a 10% accuracy

Some further constraints imposed by the Consortium on the trajectory design of the Impactor are the following:

- Maximum arrival velocities of 14 km/s at the asteroid, due to targeting camera performances
- Minimum swingby heights at Earth of 300 km, in case this astrodynamic maneuvers are used
- In case of Venus a limit on the swingby height at 4 Venus radii was imposed due to thermal reasons
- Minimum distance to the Sun at any time was set at 95% of the perihelion of Venus, also due to thermal reasons

- The angle between asteroid position and relative arrival velocity (illumination angle at asteroid) for the Impactor shall be less than 30º

All the above conditions were used to design the spacecraft transfers to the target asteroids.

ORBITER TRAJECTORY DESIGN

Due to the large energy requirements to reach asteroid 2002 AT_4 it is the mission to this asteroid the one that sizes the design of the Orbiter. This was the driver for ESA to establish the use of low-thrust for the Orbiter transfer to the proposed asteroids. Otherwise, reaching 1989 ML and Apophis is much simpler, and quasi-direct trajectories can be found.

Low-thrust trajectories were computed using different types of low-thrust engines, namely the plasma thruster PPS-1350 (thrust at 30-100 mN and specific impulse in the order of 1,800 s) and the ion engines T6 / RIT-22 (almost same performances assumed for both: thrust at 50-200 mN and specific impulse in the order of 5,400 s). Proposed launchers were Vega, Dnepr and Soyuz 2-1B. Soyuz was assumed launched from the Centre Spatial Guyanais (CSG) at Kourou (French Guyana).

A local optimizer based on a direct gradient method and a penalty function to include the mission constraints was used to obtain low-thrust mission profiles with a parabolic parameterization of the thrust laws in each thrust arc. Wherever required, more thrust arcs were included to allow a more sophisticated thrust law. Optimization parameter was the arrival spacecraft mass at the asteroids.

Many mission profiles were computed departing with the above launchers and SEP engines within the time interval of application arriving to a full set of transfer trajectories to the asteroids. However, it was soon realized that Soyuz was the only possible launcher to comply with the mission needs and the Orbiter dry masses derived at system level (in the order of 690 kg). Under such circumstances and the available launcher margins by using Soyuz, there was little gain by using a very high specific impulse engine, thus the well proven PPS-1350 was assumed for the Orbiter.

Figure 1 provides the selected mission profile to 1989 ML, which departs in July 2012 and arrives at the asteroid in May 2013 after a simple coast-thrust sequence. Some more information on the mission profile is provided in the figure.

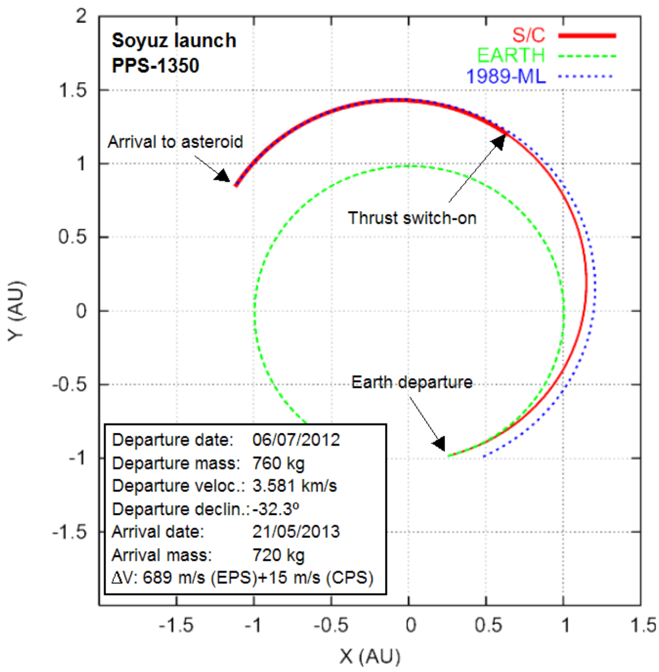

Figure 1: Orbiter mission to 1989 ML launched in 2012 by Soyuz and using a PPS-1350

Figure 2 provides the obtained mission to 2002 AT_4 which is much more complicated. The spacecraft needs to perform two revolutions about the Sun to rendezvous with the asteroid. Launch is in March 2011 and arrival in March 2015, thus a 4-year trip is needed. The thrust sequence starts with a 30-day coast phase to allow commissioning of the SEP system and is followed by a thrust-coast-thrust-coast-thrust profile to reach the asteroid. Coast arcs are present at apocentre passes as expected. A propellant load of 214 kg is needed to reach the asteroid, departing with a 1,000 kg spacecraft.

Figure 3 provides one of the obtained missions to Apophis, which also represents a simple transfer with a continuous thrust arc until reaching the asteroid almost one revolution after launch.

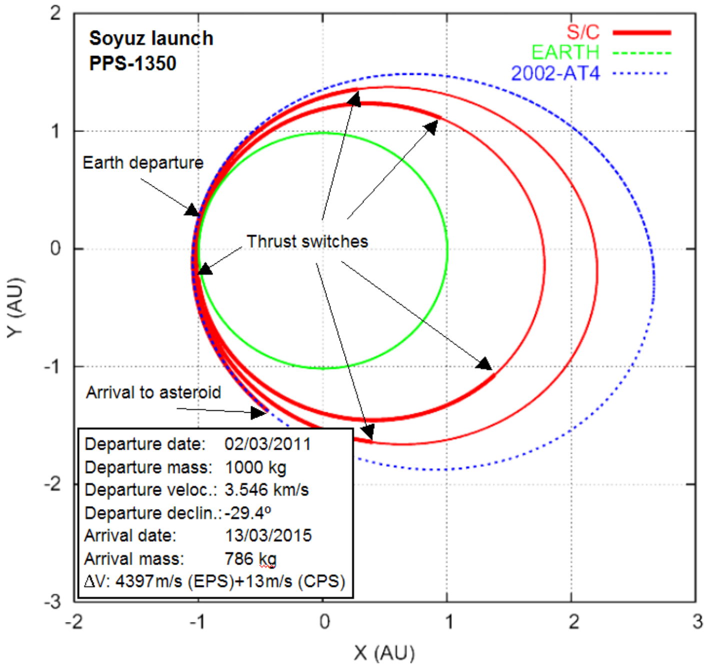

Figure 2: Orbiter mission to 2002 AT_4 launched in 2011 by Soyuz and using a PPS-1350

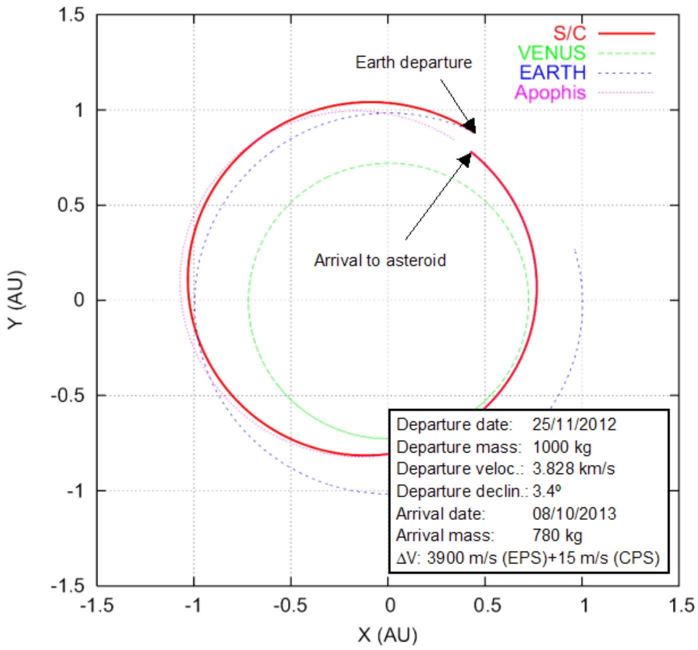

Figure 3: Orbiter mission to Apophis launched in 2012 by Soyuz and using a PPS-1350

IMPACTOR TRAJECTORY DESIGN

Impactor trajectories are mainly conditioned by the need to arrive at the asteroid with a relative collision velocity of roughly 10 km/s. The fact that 2002 AT_4 has a very eccentric orbit makes it easy to reach it with a large relative velocity, whereas the opposite applies to 1989 ML. This actually means that the sizing case for the Orbiter is now this last asteroid instead of 2002 AT_4. The situation is thus opposite than for the Orbiter.

Several mission design features were considered in this case to allow making use of a small launcher and meet the impact requirements. Among them:

- Launch into LEO to escape from it with a dedicated propulsion module
- Make use of Venus swingby in some cases (1989 ML)

The first issue allowed using the small launchers as none of the two small launchers provide with escape performances. A spiraling out sequence of maneuvers was devised to allow escaping with the minimum delta-V possible for a maximum given number of 10 burns, thus minimizing the gravity losses and achieving values below 8%.

The use of Venus swingbys proved a very valuable concept in the case of 1989 ML as it was necessary to obtain large relative arrival velocities which would have been very difficult to obtain with other techniques.

The mission search process in the time frame of application was performed by making use of a systematic search tool. Such tool allowed finding and optimizing large numbers of transfer trajectories making use of swingbys at different bodies (Earth, Mars and Venus included in the process) and deep space maneuvers to arrive or to impact to a given minor or large body. In this case, a cost function was implemented directly related to the maximization of the change in orbital energy:

$$J = m_{SC} V_{ARR} V_{PRE} \cos\varphi \qquad (1)$$

Where the first term is the spacecraft mass at impact, the second the relative arrival velocity, the third the asteroid velocity and φ the angle between both velocities. Such cost function is related to the change in orbit size by the following formula:

$$\Delta a \approx 2 \frac{a_{PRE}^2}{m_{AST}\ \mu} J \qquad (2)$$

Where a_{PRE} is the asteroid semi-major axis prior to impact, m_{AST} the asteroid mass and μ the solar gravity constant. Such formulation assumed an inelastic impact where no ejecta are produced and the S/C and the asteroid travel together after the impact.

Based on the formulation of Δa, the main conditions that allow achieving a maximum value of the optimization objective function and, as a consequence, a large change in the asteroid semi-major axis are the following:

- Large relative arrival velocity with respect to the asteroid
- Large S/C mass at asteroid arrival, provided by the combination of large S/C launch mass and low mission ΔV
- Small asteroid mass: the impact with 2002 AT_4 is favored with respect to impact with 1989 ML, since the former asteroid has a mass which is approximately one order of magnitude smaller than the latter
- Good geometry at asteroid encounter: as close as possible to the alignment of the S/C arrival velocity and the asteroid orbital velocity. This goes in normal direction with the arrival illumination condition

The result of the scanning and optimization process was composed of several databases of missions populated with large numbers of mission opportunities complying with the imposed constraints to be evaluated for future selection in conjunction with the Orbiter trajectories.

As an example of a database of missions Figure 4 to Figure 7 provide with a set of relevant parameters for a family of missions to 1989 ML after Dnepr launch and with a Venus swingby. From this group of solutions the

Impactor mission to this asteroid was selected. Figure 4 provides the group of transfer options in terms of Earth departure date vs. arrival date to the asteroid. Figure 5 represents the arrival impact velocity as function of the launch date which provides values between 8 km/s and 13 km/s. Figure 6 gives the achievable change in the asteroid orbit semi-major axis as function of the mission duration, showing that only one group allows deviations in the order of 100 m. Finally, Figure 7 gives the obtained illumination angles which in most of the cases are smaller than the 30º limit.

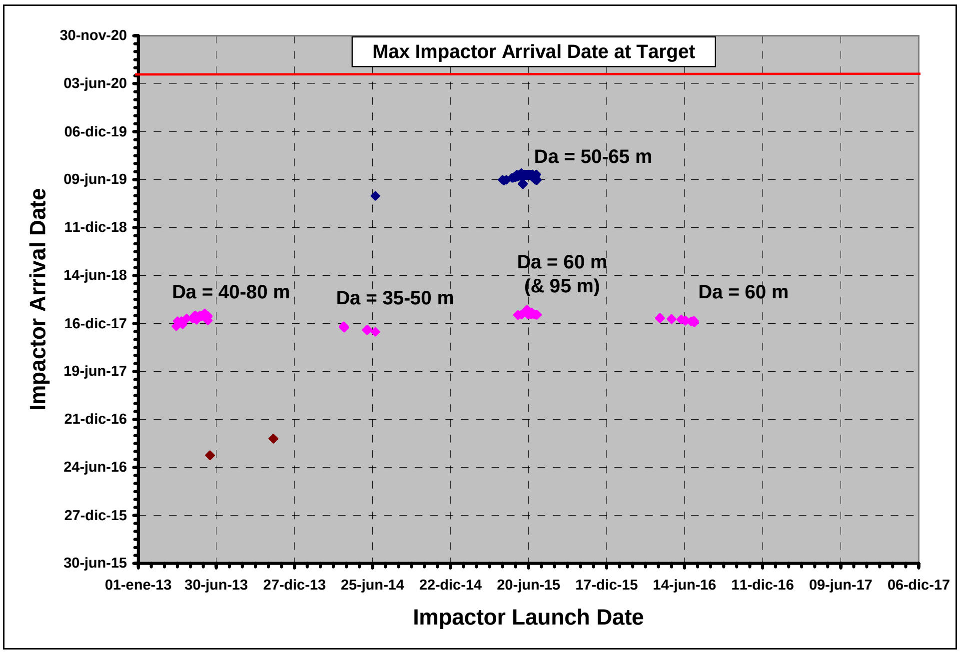

Figure 4: Impactor transfers to 1989 ML (Dnepr launch & Venus swing-by): arrival date vs. launch date

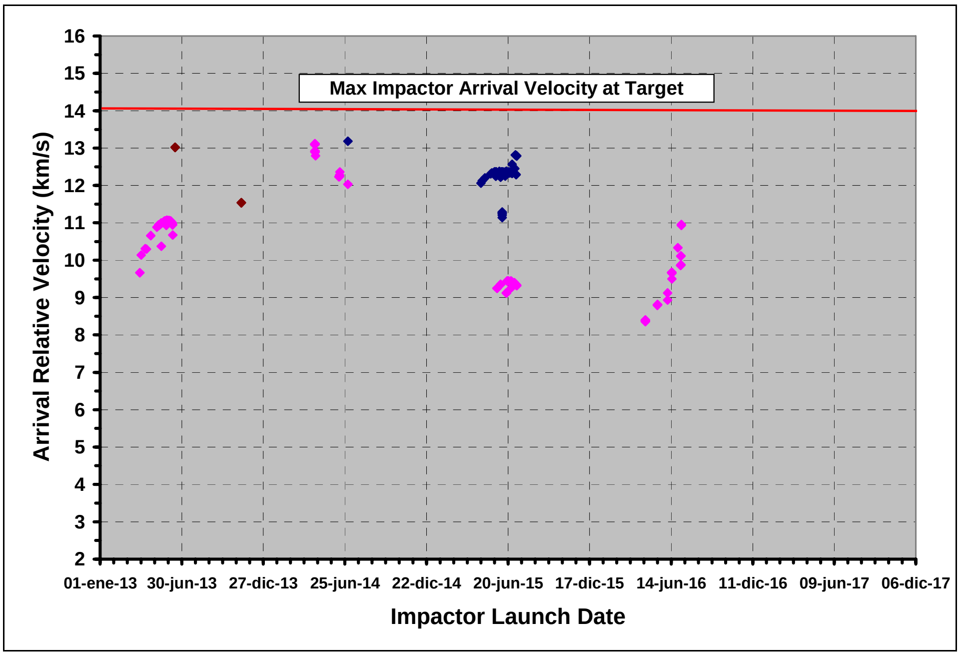

Figure 5: Impactor transfers to 1989 ML (Dnepr launch & Venus swing-by): arrival relative velocity vs. launch date

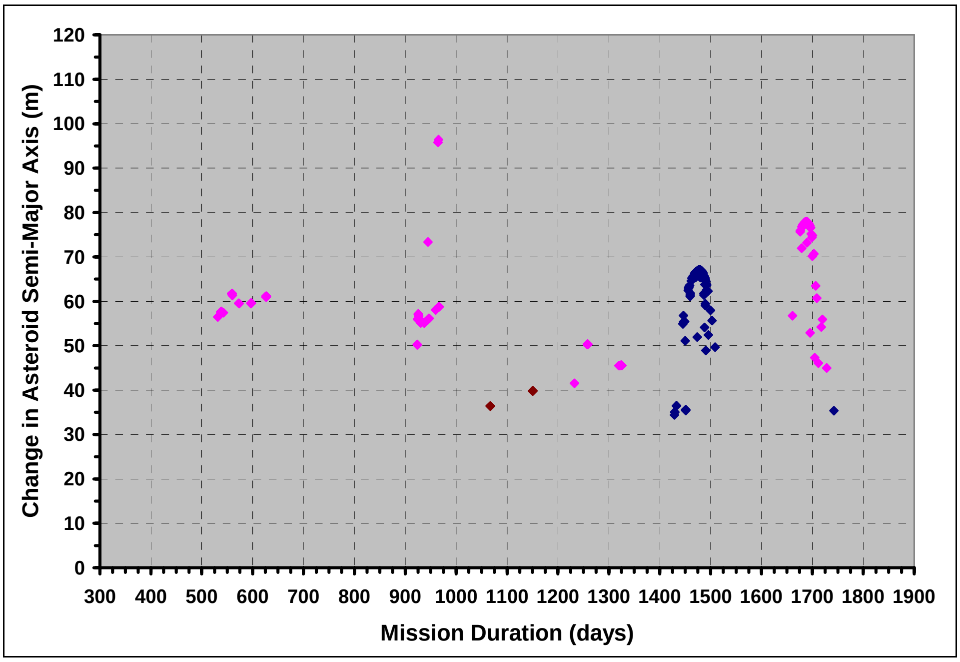

Figure 6: Impactor transfers to 1989 ML (Dnepr launch & Venus swing-by): change in asteroid semi-major axis vs. mission duration

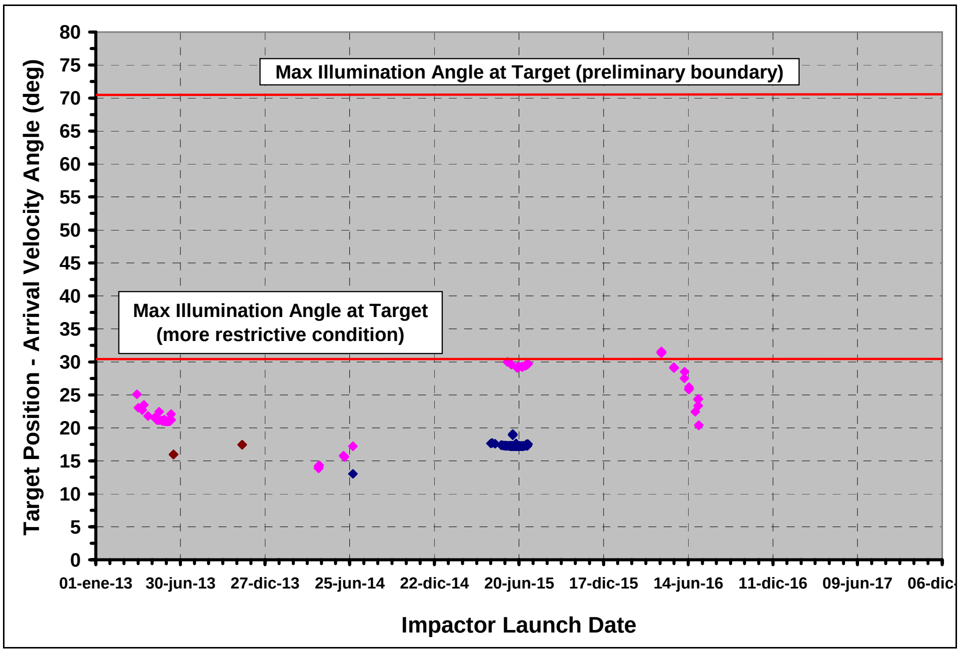

Figure 7: Impactor transfers to 1989 ML (Dnepr launch & Venus swing-by): illumination angle at the asteroid vs. launch date

Figure 8 provides the selected transfer to 1989 ML based on an escape after launch in LEO into a direct transfer to Venus in June 2015. After the swingby, two waiting revolutions about the Sun are implemented prior to the final impact branch, which occurs at 9.17 km/s in January 2018. Thus a long transfer is expected here.

In this case, the required 100 m change in the semi-major axis of the asteroid is almost achieved by making use of a Dnepr launch (Vega achieves much lower performances) and reaching 96.4 m as computed by the above formulae. Illumination angle is 29º, just below the constraint at 30º.

Figure 9 provides the transfer to 2002 AT_4, which represents a much simpler case, as the transfer is quasi-direct, departing on May 2016 and impacting on April 2017. Arrival velocity is 7.19 km/s, which is smaller than 10 km/s but enough to reach changes in the semi-major axis of the asteroid in the order of 1000-2000 km, due to its small mass. Illumination angle is 27º at impact.

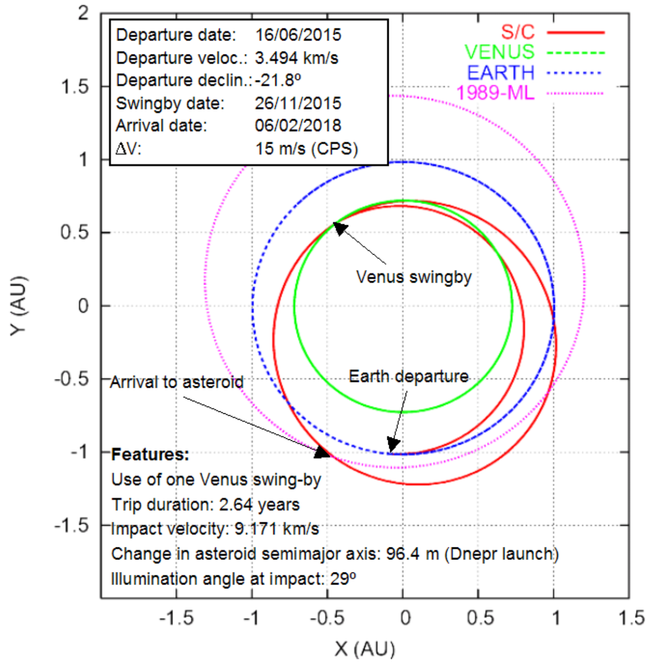

Figure 8: Impactor mission to 1989 ML launched in 2015 by Dnepr

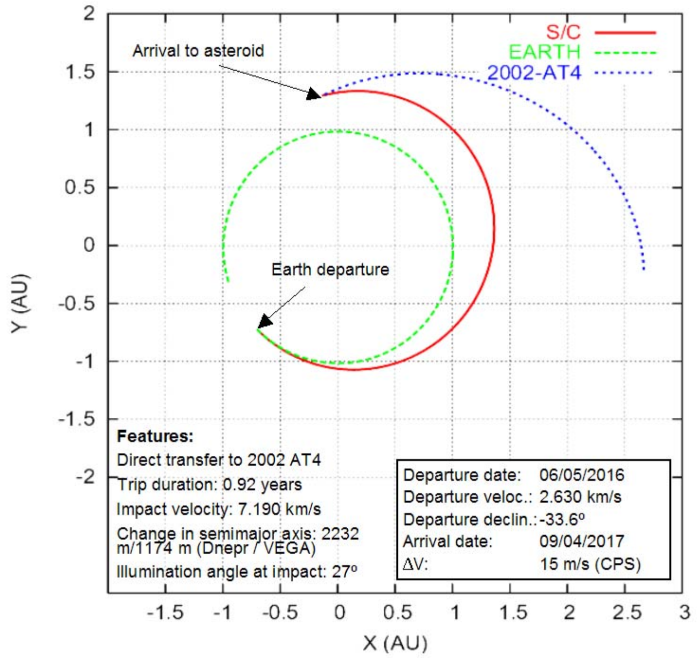

Figure 9: Impactor mission to 2002 AT_4 launched in 2016 by Dnepr or Vega

MISSION SCENARIOS

The abovementioned selected transfer cases have not been fully justified in terms of the mission constraints up to now. In this section, the coupling of the Orbiter and Impactor missions is presented such that the arrived at mission profile is fully justified.

The obtained Orbiter and Impactor families of feasible trajectories were gathered in the same type of plot to allow analyzing the impact of the mission constraints and allow the selection of the definite missions, which have been already presented in the previous sections.

Figure 10 provides the timeline for the possible missions to 1989 ML with the Orbiter transfers marked with MO# and the Impactor transfers marked with MI#. In addition, the obtained change of the asteroid's orbital semi-major axis is provided in the Impactor lines. From these values it is obvious that the only Impactor transfers compatible (almost) with the 100 m condition are the ones belonging to the MI4 group. The only Orbiter transfers compatible with that group are MO1 and MO2. The Orbiter transfer MO2 was finally selected due to its shorter transfer and to the fact that it allows longer contingency time for the spacecraft development (launch in 2012 instead of 2011).

The resulting MO2 mission was represented in Figure 1 and the selected Impactor mission in MI4 was represented in Figure 8. Both were already commented in above sections.

Figure 11 provides the same type of assessment, but this time for 2002 AT_4. No figures of achievable deviation in asteroid semi-major axis are provided as these are always above 1000 km. The diagram here is simpler due to the less number of missions to the asteroid.

The only compatible scenario is the coupling of AO1 with AI1, AI2, AI3 or AI4. The AI2 and AI4 groups were disregarded due to their late impact dates at the asteroid (close to the asteroid perihelion pass in 2020). Between AI1 and AI3 a mission in AI3 was selected due to its shorter mission duration.

The resulting AO1 mission was represented in Figure 3 and the selected Impactor mission in MI4 was represented in Figure 9. Both were already commented in above sections.

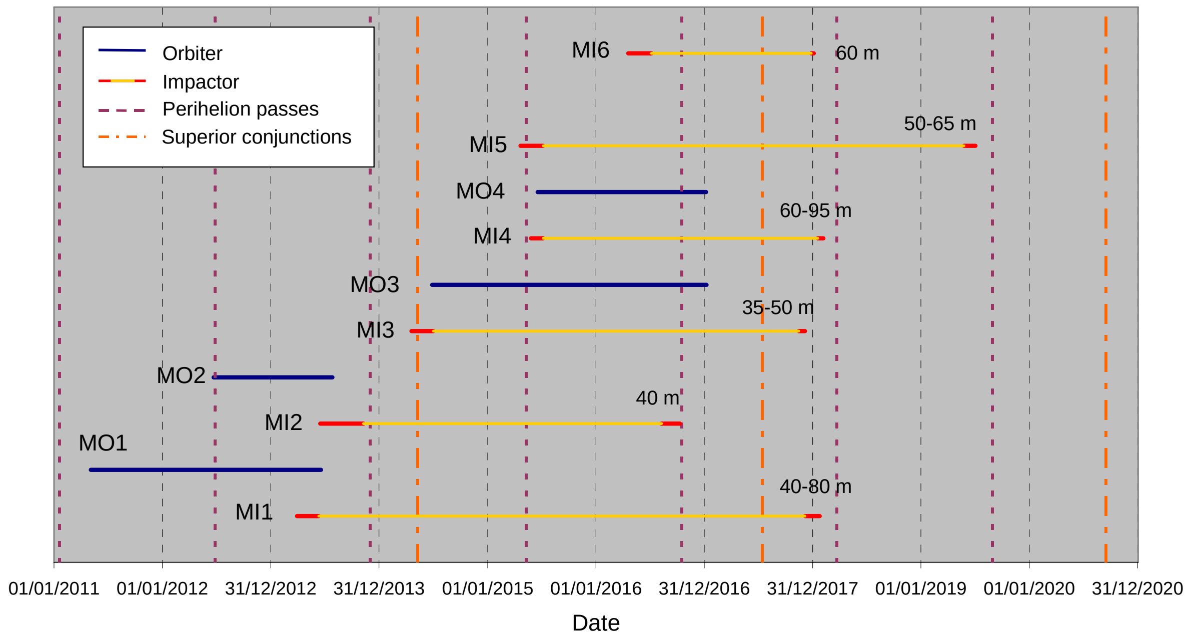

Figure 10: Orbiter and Impactor missions to asteroid 1989 ML in the timeframe of application

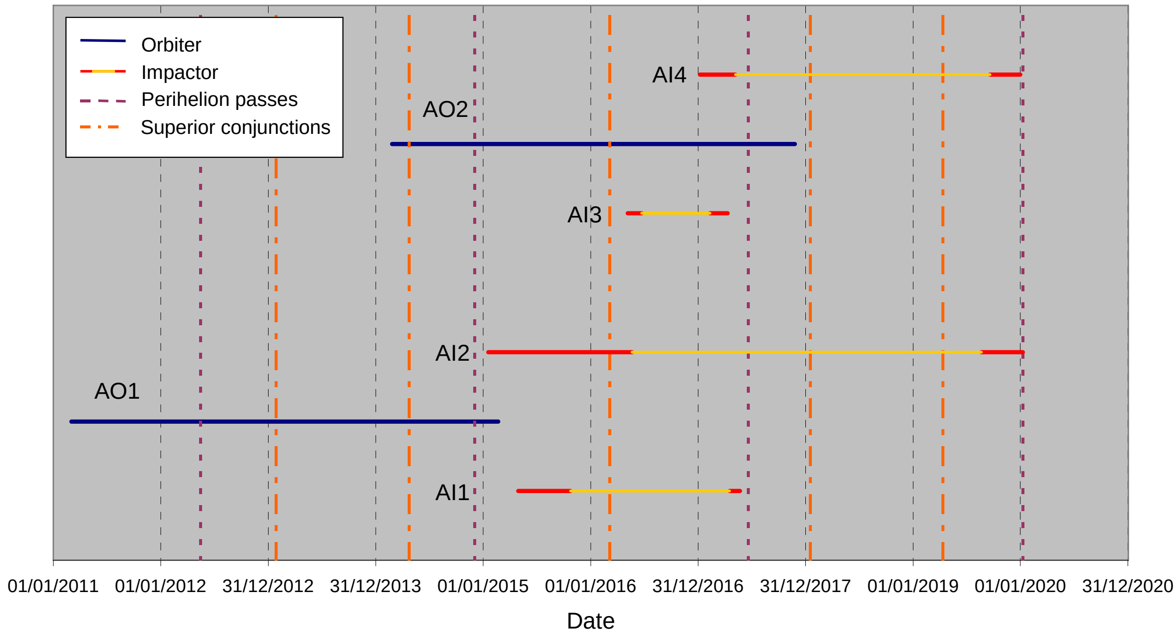

Figure 11: Orbiter and Impactor missions to asteroid 2002 AT$_4$ in the timeframe of application

ORBITER ARRIVAL STRATEGY TO THE ASTEROID

The Orbiter arrival to the asteroid was analyzed in the study taking into account the following aspects:

- The spacecraft arrives with a small relative velocity to the asteroid due to the decelerating effect achieved with the low-thrust engines and will have a residual drifting velocity for the approach
- The Orbiter will need finding the asteroid by making use of its own camera through a sky scanning process in the area covered by the asteroid dispersion ellipsoid
- The scanning is not only in space but also in time, due to the dependence of the detection time on the along-track projection of the asteroid dispersion ellipsoid

As a consequence of the analysis it was determined that 2 months are required in average to carry out the approach to the asteroid in case of the faintest object (2002 AT$_4$) with the detection limit of the selected spacecraft camera (a Dawn type camera) and the existing knowledge of the asteroid ephemerides.

ORBITER STABILITY AT THE ASTEROID

The main function of the Orbiter will be that of measuring the deviation imparted on the asteroid orbit by the Impactor collision. In order to perform such goal the spacecraft will carry a Radio Science Experiment that will allow inferring such deviation in the asteroid orbit. However, to achieve the expected estimation performances it is extremely important that the spacecraft is subjected to a very stable environment with respect to the unknown perturbations over its trajectory about the asteroid. It is planned to achieve such goal by making use of the following processes:

- Detailed characterization of the effect of the solar radiation over the spacecraft by a dedicated RSE campaign during cruise (in coast phases)
- Detailed determination of the asteroid gravity field and other asteroid features during the Asteroid Characterization Phase
- Spacecraft system design based on a low level of parasitic forces to spoil as little as possible the orbit knowledge. This is materialized by developing an AOCS system that has a balanced wheel off-loading (WOL) system with scarce WOL maneuvers (nominally one every two weeks)
- Finally, locating the spacecraft in an orbit which is self-stabilized for long time periods that do not require of correction burns and presents a stable dynamic environment

Of the above elements, the last one has an impact on the mission design, as there is the need to find long duration self-stabilized orbits. Such orbits exist and make use of a particular balance that can be achieved between the solar radiation pressure over the spacecraft and the asteroid gravity field. The photo-gravitational orbits (see [5] and [6]) make use of such balance to enable a manifold of solutions which are stable during long periods of time in a band of altitudes above the asteroid.

Such types of orbits are terminator orbits with an orbital plane slightly behind the asteroid in the Sun-asteroid line and perpendicular to that line (to compensate the asteroid gravity with the solar radiation pressure, see Figure 12). They are quasi-circular and have the property that the orbital plane rotates at the same rate as the asteroid about the Sun, thus preserving the orientation of the orbit with respect to the Sun-asteroid line.

In fact, an interval of stable orbits can be found for the selected asteroids to carry out the RSE experiment. The inferior stability limit is posed by the irregular shape and gravity field of the asteroid which will make the spacecraft impact on the surface. The upper limit in altitude is conditioned by the solar gravity, which can put the spacecraft in an escape trajectory from the asteroid.

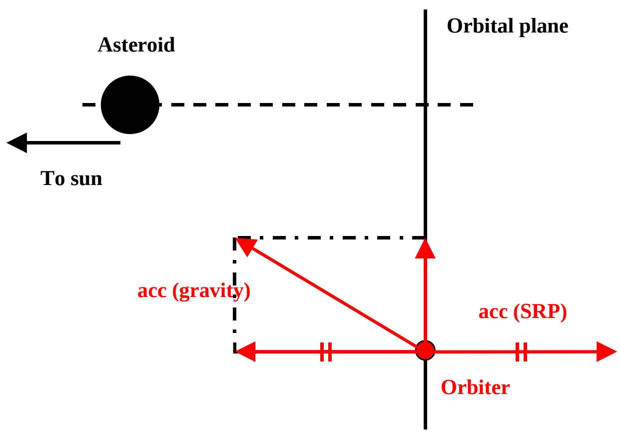

Figure 12: Simplified photo-gravitational problem

A dedicated analysis on each of the asteroids allowed defining the abovementioned stability regions. A Monte Carlo tool was developed for such purpose, which propagated numerous orbits at different orbit heights for long periods to discriminate among the cases: stable and unstable. Different sizes, masses, rotation directions and gravity field shapes were also assumed in the target asteroids to arrive to the following stability limits over 100-day propagations:

Table 2: Nominal stability boundaries for terminator orbits about 1989 ML and 2002 AT$_4$

Asteroid mass	Asteroid shape	Altitude above 1989 ML		Altitude above 2002 AT$_4$	
		Minimum	Maximum	Minimum	Maximum
Minimum	Sphere	0.3	3.6	0.2	2.4
	Prolate spheroid	1.9	3.6	0.2	2.4
	Oblate spheroid	1.5	2.9	N/A	N/A
Maximum	Sphere	1.7	7.1	0.3	3.6
	Prolate spheroid	3.7	7.2	1.4	3.6
	Oblate spheroid	2.7	6.0	N/A	N/A

As it can be observed, the larger the mass, the larger the stable orbit altitude above the asteroid and the larger the stability limits. The shape of the asteroid conditions the minimum limit of the stability region. The more irregular the body the higher the minimum limit is to avoid instabilities in the orbit. An oblate shape is not considered possible for 2002 AT$_4$ and as such the case was not computed. The effect of the rotation axis orientation and its rate was observed of second order and thus was not included in the above table.

The resulting stable orbits about the asteroids are as the ones represented in Figure 13. The plots present the three projections of the stable trajectory in a rotating reference system with the X-axis oriented along the Sun-asteroid line, the Z-axis along the asteroid orbit pole and the Y-axis forming a right-handed reference system with the other two. As observed in the plots, although it is not an exactly repeating figure, the shape of the orbit is quite stable in time (100 days in this case).

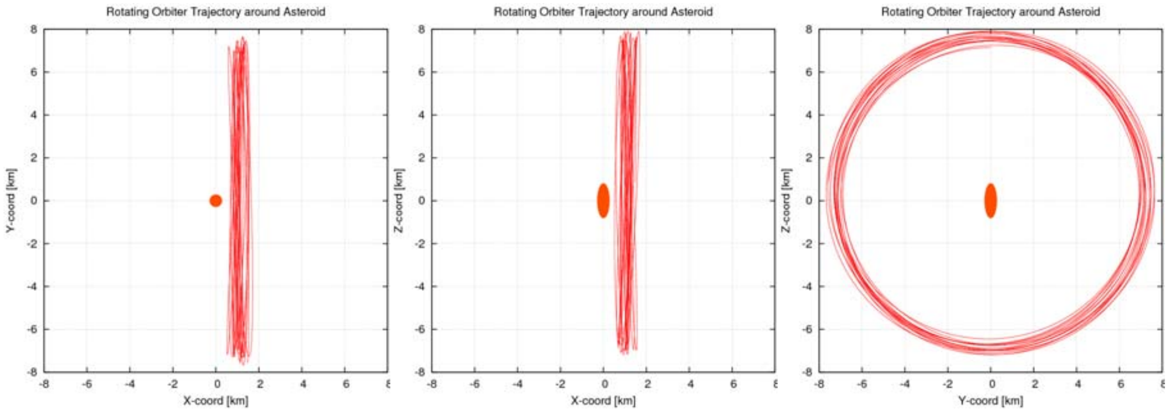

Figure 13: Orbiter stable photo-gravitational orbit about 1989 ML in rotating reference system

RADIO SCIENCE EXPERIMENT (RSE)

The RSE is one of the key experiments in the mission. It will allow determining the imparted deflection to the asteroid and it will need matching the 10% accuracy imposed by ESA on the achieved deflection. The proposed experiment is based on a dual frequency link communication channel (Ka band and X band) that will allow achieving a high degree of radiotracking performances.

It is not the purpose of this paper to present the results of this assessment in detail. However, it was demonstrated during the study that, after a campaign to determine the gravity field and rotation state of the asteroid, the key parameter in achieving the requested deflection accuracy relies on an accurate characterization of the solar radiation pressure over both the spacecraft and the asteroid (including the effect of the Yarkovsky faint forces). For such purposes the spacecraft will need to be equipped with a thermal radiometer to enable determining the radiative properties of the minor body.

MISSION TIMELINES

With the above selected mission scenarios it was possible to define the full operational timelines. These take also into account for the following Orbiter operations:

- A 30-day coast phase at start of the Orbiter transfer shall be present to characterize and test the SEP engine
- The asteroid approach phase will consist of at least two months of operations used to find and approach the Orbiter to the asteroid
- A 3-month asteroid characterization phase will begin after insertion in asteroid orbit
- An Orbiter hibernation phase will start after the previous phase until seven moths previous to the impact
- During the first 6 months of those last 7 months, the first RSE campaign will take place
- The last month prior to impact and the first after impact will be devoted to the preparation and realization of the impact operations. This includes the re-location of the Orbiter at a safe position to observe the collision
- After the collision observation phase, another 6-month period will be allocated for the second RSE campaign that will allow determining the change in the semi-major axis
- Finally in the Don Quijote “Plus” mission scenario, a surface package will be delivered on the asteroid for analysis of the impact area

For the Impactor, the following operational conditions were considered:

- 1-month period to escape from LEO by using up to 10 burns up to escape
- Up to one month for spacecraft commissioning
- Venus swingby in the transfer to 1989 ML
- 2-month terminal approach phase where the spacecraft tracking is reinforced with the use of delta-DOR to improve as much as possible the impact success. This phase includes the final autonomous impact operations which are carried out the last day to go to impact

Proposed mission timelines to 1989 ML and 2002 AT_4 are respectively given in Figure 14 and Figure 15. The description of all the phases correspond to the given mission profiles in the previous sections and the operational conditions reported in this section.

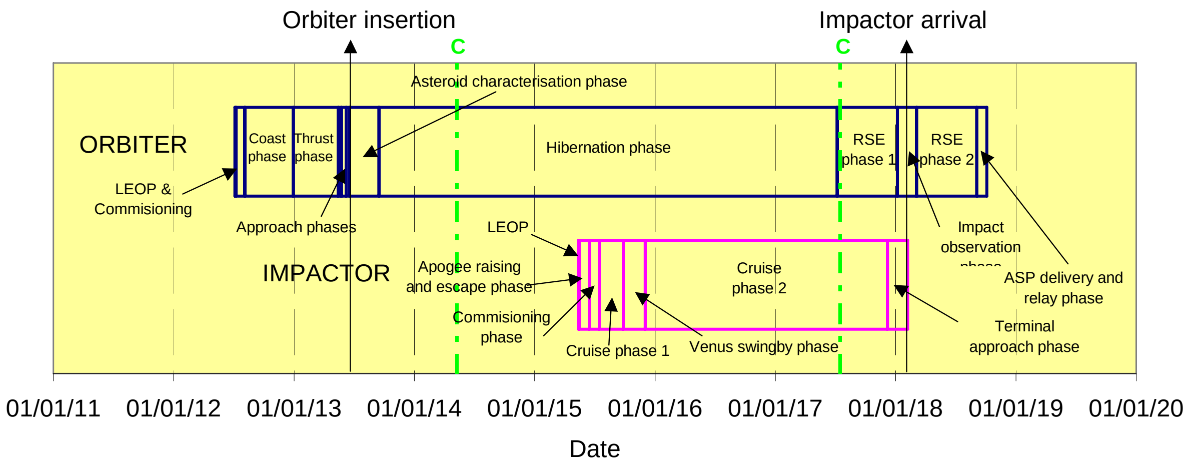

Figure 14: Operational timeline of the proposed mission to 1989 ML. Green lines marked with "C" signal superior conjunctions

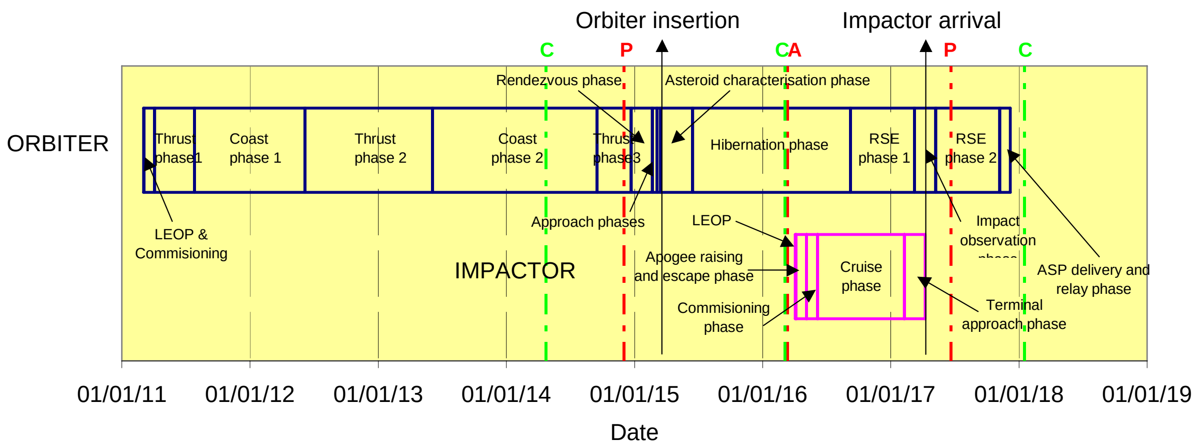

Figure 15: Operational timeline of the proposed mission to 2002 AT_4. Green lines marked with "C" signal superior conjunctions and red lines with "A" and "P" signal asteroid passes by its aphelion and perihelion

CONCLUSIONS

The mission analysis activities carried out for the Don Quijote mission within the EADS Astrium – DEIMOS Space consortium have rendered the following conclusions:

- Orbiter low-thrust transfer trajectories to the selected asteroids compatible with the imposed mission constraints were found and fully described. Proposed launcher is Soyuz 2-1B launched from Kourou (French Guyana)

- Impactor ballistic transfer trajectories to 1989 ML and 2002 AT_4 compatible with the imposed mission constraints were found and fully described. Proposed launchers are Dnepr/Vega
- Full mission scenarios were derived from the obtained Orbiter and Impactor transfer profiles compatible with the imposed mission constraints
- Orbiter arrival and in-orbit operations were made compatible with the required asteroid trajectory deflection experiment. In particular, a photo-gravitationally stable orbit is proposed as the nominal orbit to perform the Radio Science Experiment
- The simulations of the RSE demonstrated that the physical characterization of the asteroid surface and thermal properties is a key element in achieving the requested 10% accuracy in the deflection determination
- Full mission timelines were proposed for the impacting missions to 1989 ML and 2002 AT_4

The results of the DEIMOS analysis contributed very significantly to the definition of requirements and constraints for a minisatellite project, currently being considered by ESA as possible in-orbit technology demonstrator.

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