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DESIGN OF A CLOSE-PROXIMITY OBSERVATION MISSION FOR 99942/APOPHIS

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ABSTRACT

A concept for a low-cost spacecraft mission to orbit asteroid 99942/Apophis is proposed. The rationale is the potential threat posed by this asteroid which might require active countermeasures in the foreseeable future, to which an orbiter mission could provide essential input. End-to-end mission design is presented, including assessment of launch vehicle, escape stage, transfer propulsion and orbit around the asteroid, where the actual science is performed, including long-term tracking and orbit determination and close-quarter observations, for which windows of opportunity are identified in 2013 and 2021. The calculations are based on thorough numerical analysis. In addition, options for possible high-velocity impactor missions are also regarded.

INTRODUCTION

Asteroid 99942/Apophis, an Aten type Earth-crossing asteroid with a diameter of around 280 m [2], currently is on an orbit with peri- and aphelion radii of 0.75 and 1.1 AU, respectively and a period of 323.5 d. It is foreseen to pass by the Earth at a distance of around 5.94 Earth radii on April 13, 2029 [1]. The perturbations through this encounter will significantly change the orbit, likely increasing its period to over 420 d, changing the asteroid type from Aten to Apollo [13].

Furthermore, if the B-plane parameters of the encounter are within an approximately 600 m wide “keyhole” region [10][13], Apophis will end up in a 7:6 resonant orbit. Then, exactly 7 years and 6 Apophis revolutions later, on April 13, 2036, Apophis will again encounter, and may impact, the Earth.

Based on the current knowledge of Apophis' orbit there is a small (i.e. about 1/50000) but non-zero probability of an Earth impact at that date, which has attracted considerable public interest. As the energy released at impact could be equivalent to the explosion of a 400 MT thermonuclear device [1][3], arguably measures should be taken to determine the orbit and properties of this potentially hazardous object to a high degree of fidelity. This would allow the accurate assessment of the impact risk in 2036 and the following years and also would facilitate the planning of deflection missions, should the need arise.

Since the “keyhole” is only 600 meters wide, a deflection executed before 2029 would require several orders of magnitude less momentum transfer than if the deflection had to be performed after the close encounter. According to [10], prior to 2029, effective deflection could require a ΔV as low as 10^{-6} m/s, as the issue is only to avoid the small “keyhole”. Conversely, after 2029, the minimum ΔV is in the order of 2 cm/s, because then, the task is to avoid the entire Earth impact zone, which is 4 orders of magnitude larger than the “keyhole". As will be shown, a 10^{-6} m/s change in the velocity of Apophis could be easily achieved by means of a high velocity impact even with a small impactor (current estimates of the mass of Apophis are 2.1×10^{10} kg), while imparting as much as 2 cm/s would be extremely demanding and certainly not achievable by means of an impact.

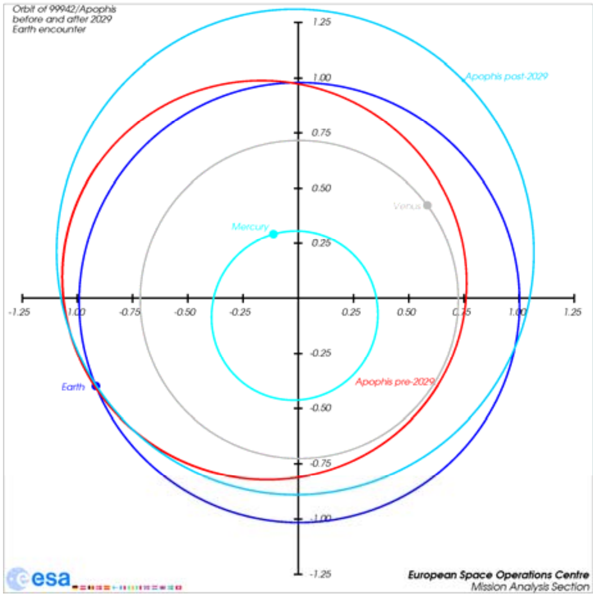

Fig. 1 : Apophis Pre- and Post 2029 Encounter Orbits

The decision whether or not to perform a deflection should be based on reliable knowledge of the asteroid's orbit. However, if the asteroid actually is on a collision course, it might be that we only know for sure when it is too late to do anything about it [10].

As will be shown, the next window for Earth-based observations of this asteroid will open in 2013. However typical Earth based observations via optical telescopes or radar are subject to natural limitations and might not provide the desired level of accuracy or insight. Additionally the effect of force model uncertainties should be considered. Prediction of the Yarkowski effect requires knowledge of the asteroid's shape, mass and mass distribution, rotation, reflectivity distribution and thermal properties, all of which would be difficult to obtain without going there.

These limitations could be overcome by placing a spacecraft in orbit around the asteroid for extended periods, which could be achieved at a low cost and effort. This paper constitutes a proposal for such a mission.

ASTEROID DEFLECTION

The smaller the asteroid, the more feasible deflection becomes. For 99942/Apophis, a small asteroid, deflection is an option. Although the present paper proposes a pure observation mission, the possibilities for deflection are also regarded as an add-on. The most straightforward method (and the only one we regard in detail here) is that of a high-velocity-impact (HVI)[4][5][15][17].

With HVI, a spacecraft is targeted for an impact at a high relative velocity, thus transferring momentum to the asteroid. This results in a small change in the asteroid's orbital velocity, slightly modifying its orbital period. The efficiency of the momentum transfer depends on the asteroid's internal composition: For an elastic impact, 100% of the impactor momentum are transferred to the asteroid. If the impact is partially inelastic, less than 100% are transferred, but if there is a large amount of ejecta, the transferred momentum may significantly exceed that of the spacecraft.

A change in orbital period translates into a difference in position - with respect to where the asteroid would have been without the intervention - that increases with time. If the deflection takes place well in advance of the expected Earth encounter, this can change the trajectory from one that impacts the Earth (or the "keyhole" region) to one that passes at a safe distance.

HVI is not the most efficient momentum transfer concept, but it is certainly the easiest to be implemented with existing technology and it has been seriously taken into consideration for a potential ESA mission [4][5]. Probably, HVI is the only technique available in the foreseeable future.

Other, potentially more effective but also more demanding techniques involve, e.g., the use of thermonuclear devices, changing the asteroid's reflective or thermal properties, positioning propulsion units on the asteroid surface or placing massive, controlled spacecraft in the vicinity to deviate the asteroid through their gravitational attraction. We do not regard such techniques here.

MISSION OBJECTIVES

The primary objectives underlying this mission proposal are:

- Significant reduction of the uncertainty concerning the Apophis orbit
- Improvement in the characterization of the non-gravitational effects that perturb the asteroid orbit

- Observation and characterization of the asteroid rotation, including effects such as YORP

These objectives were defined with the aim of a significant improvement of the predictability of the Apophis orbit beyond what is feasible with Earth-based observations. Reducing the dispersion in the arrival conditions in 2029 will vastly improve the prediction of the 2036 encounter. The knowledge thus obtained would also provide valuable input to support the targeting in any potential deflection mission concerning 99942/Apophis, should the need arise.

The secondary mission objectives lie in the provision of a platform for scientific experiments to study a small asteroid at close quarters. Although these objectives are seen as “secondary” in the context of the present proposal, they are scientific mission goals in their own right.

MISSION DESIGN

The design is consistent with a low-cost mission, using a small launch vehicle and minimizing the development time, cost and risk via the use of off-the-shelf components. The assumptions described here are based on those made in ESA studies [4][5][6].

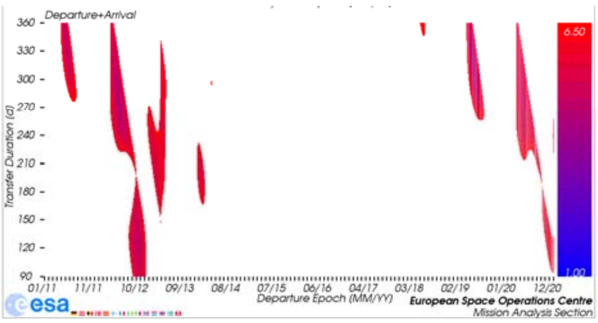

Fig. 2 : Pork-Chop-Plot for Direct Earth-Apophis Transfers, 2011-2020

For a rendezvous with the asteroid, the most appropriate propulsion system is Solar Electric Propulsion (SEP). This is illustrated by the pork-chop-plot in Fig. 2 for the direct Earth to Apophis transfer for the years from 2011 to 2020. The minimum total ΔV for a direct transfer is in the order of 5.6 km/s. This value immediately rules out chemical propulsion for a small-class spacecraft. More complex strategies involving an Earth or Venus flyby, and including deep space manoeuvres (DSM) do not lead to a feasible option for a fully chemical mission.

Conversely, SEP appears appealing and consistent with the objective of using a small spacecraft. The goal is to compute a feasible rendezvous trajectory which minimizes the thrust requirements to reach Apophis. An additional benefit of SEP consists of extending the mission opportunities. While the best launch opportunities for a chemical transfer are in 2012 and 2013 and again in 2020, SEP allows a launch opportunity almost every 18 months, rendering the mission design more flexible.

Launch and Escape Strategy

The European VEGA rocket is assumed as launch vehicle. The payload performance into a 300 km LEO at an inclination of up to 30 deg is stated as 2300 kg [7]. VEGA is not capable of interplanetary launches, therefore an added escape stage is needed, which is assumed to be based on the commercially available Thiokol Star 48 solid rocket motor, which has proven to be the most performing combination to achieve escape conditions [6].

The mass of adapter and spin-up table was taken as 113 kg, that of the stage without the solid propellant but including the engine casing and de-spin mechanism was taken as 164 kg [6]. The mass of the solid propellant mass is chosen to match the required escape. The combined masses must not exceed the launch vehicle performance.

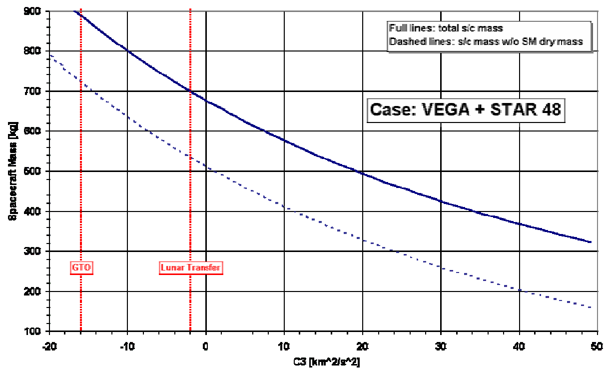

Fig. 3 : Performance Chart for Vega launch with Star-48-Based Escape Stage

Fig. 3 charts the escape performance of with this configuration over the escape energy C3. The through line is the escape mass including the burnt-out solid rocket motor casing and the mechanisms for despin (yo-yo) and spacecraft separation, the dotted line denotes the actual spacecraft mass after separation. For C3 = 9 km^2/s^2, a spacecraft mass of more than 400 kg is achieved.

As an alternative to direct escape, a lunar swing-by could be foreseen, leading to an initial spacecraft mass of 520 kg, albeit at an escape velocity of around 1.1 km/s, imposing a higher load on the spacecraft propulsion system.

Launch Window

The choice of the launch window is driven by numerous considerations. Obviously, several years are needed to design, build and test the spacecraft following formal approval. Then, the relative geometry of Earth and Apophis is such that periods with good observation conditions are few and far between.

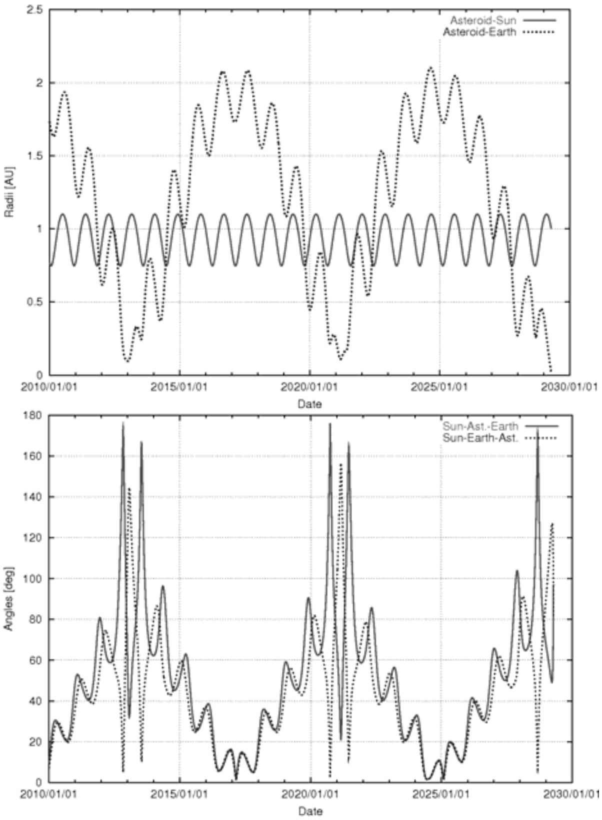

Fig. 5 : Earth-Sun-Apophis Earth-and-Sun-Range and Aspect Angle Evolution

Fig. 5 shows the range and aspect angle evolution. Good viewing conditions exist throughout 2013, when the asteroid approaches the Earth to as close as 0.1 AU. In 2014, the distance again increases rapidly and the asteroid remains behind the sun for years, with a superior conjunction in 2017.

Conditions similar to 2013 recur in 2021, only 8 years before the 2029 encounter. A mission leading to an asteroid arrival between 2015 and 2020 is of limited or no use. Ideally, the mission planning should be such that arrival can be achieved in 2013. The mission opportunity with arrival in 2021 is available in case the earlier opportunity is missed or a second mission is required.

As discussed, SEP allows flexible choice of the launch date to obtain geometrical conditions to best perform RSE.

Optimal SEP Trajectory

The thrust model for the selected SEP unit is based on the Qinetiq T5 Gridded Ion Thrusters (GIT) [6], which is capable of providing a maximum thrust of 20mN.

The transfer optimization goal is to minimize the power requirements while maximizing the spacecraft dry mass (i.e. >330 kg) at arrival. To comply with the requirement of performing radio science in the best observation window, solutions were calculated for a departure ranging from 2011 to 2013.

Baseline: Launch in 2011

For a launch in 2011, which would allow the spacecraft to arrive in the optimal observation window (i.e. 2013) the minimum-power solution requires 700 W at 1AU which corresponds to 16 mN of thrust. This is consistent with the use of one single T5 GIT.

The solution has been baselined for the proposed mission; the main features of the trajectory are reported in Table 1. Fig. 5 shows the nominal transfer to Apophis, including thrust and coast arcs. Launch is at the end of April 2011.The transfer lasts 30 months, leading to arrival in October 2013. The total thrust-on time is 7000 h, broken into 4 arcs, during which the spacecraft consumes 35 kg of Xenon propellant and delivers a delta-v of 2.75 km/s. The mass at arrival is 337 kg.

Launch Vehicle/upper stage	VEGA+SRM
Launch date [Y/M/D]	2011/04/29
Initial Mass [kg]	370
Departure velocity [km/s]	3.5
Arrival date [Y/M/D]	2013/10/04
Final Mass w/o margin [kg]	337
SEP ΔV [km/s]	2.75
Total transfer time [d]	900

Table 1: Summary of Transfer to Apophis in 2011

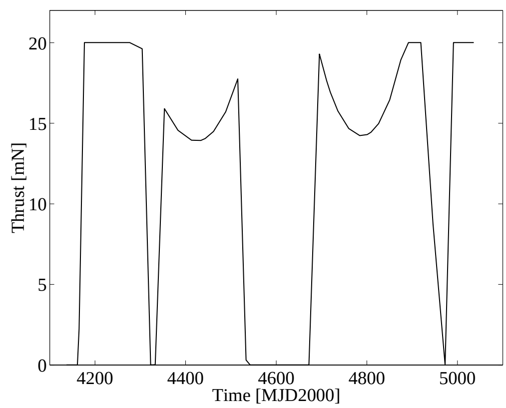

Fig. 3 : Baseline SEP Transfer to Apophis, Optimal Thrust Profile

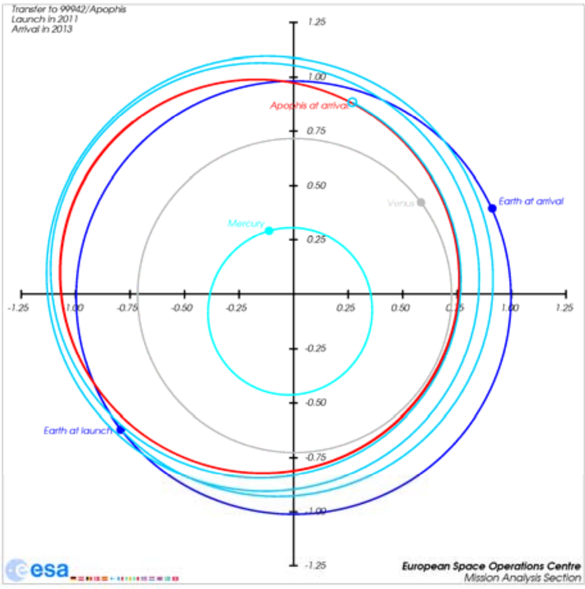

Fig. 4 : Baseline SEP Transfer to Apophis, Launch 2011, Arrival Oct. 2013

Option: Launch in 2013

SEP extends the set of possible solutions for transfer to Apophis, providing opportunities with very similar characteristics almost every two years. This is consistent with a spacecraft design in common with that for the 2011 launch. Solutions were found for launch in April 2013 and also in April 2015.

The 2013 case represents the best transfer opportunity for a rendezvous with Apophis. Again, the criteria of minimizing the power requirement while maximizing the mass at arrival were applied. Compared to 2011, Earth is left at a lower departure velocity, v_{inf}=3.16 km/s (i.e. higher departure mass). The arrival mass at the target is 368 kg.

This case also features minimum power and solar array size requirements. In fact, the SEOP input power at 1 AU is only 340 W, the thrust there 8 mN. Maximum thrust is reached at the orbit of Apophis, as before.

Launch Vehicle/upper stage	VEGA+SRM
Launch date [Y/M/D]	2013/04/01
Initial Mass [kg]	390
Departure velocity [km/s]	3.17
Arrival date [Y/M/D]	2015/05/11
Final Mass w/o margin [kg]	368
SEP ΔV [km/s]	1.7
Total transfer time [d]	769
Total SEP time [d]	???

Table 2: Summary of Transfer to Apophis in 2013

This option represents an interesting alternative, though the 2011 transfer would arrive at a better time for radio science experiments (RSE)

Similar transfer solutions exist in 2015. We do not take them into account as the arrival would occur during the worst geometrical condition for RSE.

Final Approach and Orbit Insertion

Unlike high-thrust transfers using chemical propellant, final approach with SEP will always be at a low relative velocity and a phase angle of 90 deg, i.e, the onboard science camera also used for optical navigation will “see” the asteroid half-illuminated throughout the far approach phase.

Orbiting The Asteroid

For close orbits around medium-sized asteroids such as 433/Eros, the major source of orbital perturbations is the strong inhomogeneity of the typically very irregularly shaped body. Stable spacecraft orbits can mostly be achieved there for a retrograde orbit near the asteroid’s equator plane.

For a small asteroid such as 99942/Apophis, a stable orbit of the retrograde class would be too close to the asteroid and therefore not feasible. Larger relative orbital radii, with respect to the mean body radius, are required. The gravitational inhomogeneity becomes less important; solar radiation pressure forces (SRPF) are the most significant perturbation source.

A different class of solutions is available for stable orbits around small asteroids in the presence of SRPF: Terminator orbits, which by definition are near-perpendicular to the current sun direction [4][5][6][9][16]. Numerical simulation (Fig. 7) shows that such orbits are not only stable, but also self-stabilizing, i.e, SRPF will rotate the spacecraft orbit plane such that it remains oriented with its normal roughly pointing towards the sun, within a margin of several degrees, even as the asteroid pursues its trajectory around the sun.

On such an orbit, the altitude variations are small, there is no eccentricity build-up and no stationkeeping is required to prevent surface impact or escape. Stability is available for a range of altitudes and cross-sections.

Table 3 shows the results for a numerical simulation of a wide range of scenarii for terminator orbits around 99942/Apophis. The nominal orbit radius was varied between 250 and 750 m. The spacecraft mass at arrival was assumed as 400 kg, the effective cross section was varied between 5 and 15 sqm. Within the chosen range of cross sections, which should cover all thinkable spacecraft configurations, self-stabilizing orbits are feasible. Below 250 m radius, the risk of impact grows, above 750 m, a growing tendency towards escape is observed. For small cross sections, stationkeeping is required already at radii near 250 m.

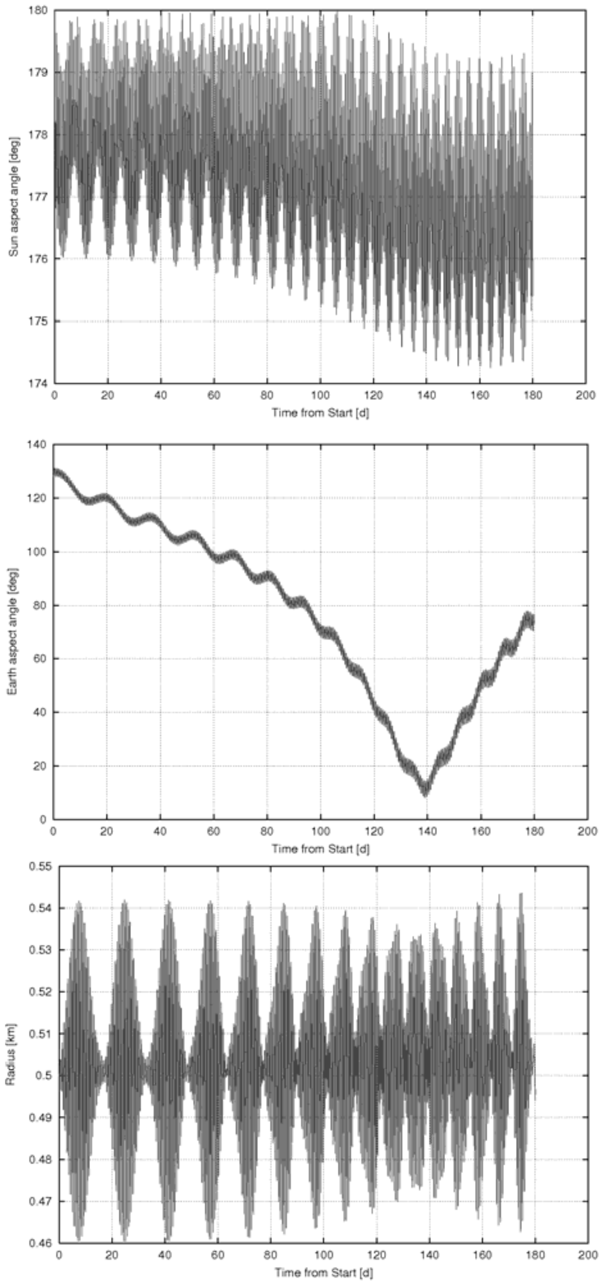

Fig. 3 : Sun- and Earth Aspect Angles and Radius 6 Month Evolution on Sample Terminator Orbit

The advantage of self-stabilizing orbits is their safety, their predictability and the associated reduced operational effort. The fact that such a wide range of radii and reflective cross sections is available indicates that there is considerable freedom in the choice of the orbit, e.g., to avoid resonance with the asteroid body rotation, and also in the spacecraft design. The absence of thruster manoeuvres improves the quality of the tracking data and thus benefits the radio science experiment.

Alt. [m]	Mean vel. [cm/s]	Orb. Per. [≈ h]
250	7.5	6
500	5.3	16.5
750	4.3	30

Table 3: Characteristics of Self-Stabilizing Terminator Orbits Around Apophis for a Range of Altitudes

The disadvantages are that the orientation of the orbital plane is essentially fixed, this might not be the optimal geometry for radio science, and it also might not offer the most favourable scenario for surface observations, as the spacecraft will be permanently located above the terminator.

On any other orbital orientation, stationkeeping will be required, It could be shown [4] that a very simple control algorithm is sufficient, requiring only knowledge of the current altitude (obtainable e.g., via Radar or LIDAR altimetry) and involving only radial manoeuvres. When the lower (or upper) altitude deadband is transgressed, a small outward (or inward) manoeuvre is implemented. This strategy, while not optimal, is easy to implement, lends itself to autonomous operations, works for any kind of orbit and incurs a cost of only a few m/s per month.

In summary, terminator orbits are applicable within a certain range of orbital radii and appear to be the only class of long-term stable orbits around a small asteroid such as 99942/Apophis. In addition, a variety of other solutions exist, from low hovering to long-distance co-orbiting heliocentric trajectories, but these require stationkeeping and are less appropriate for the given mission objectives.

RADIO SCIENCE EXPERIMENT

RSE refers to the exact determination of the asteroid orbit using radiofrequency tracking, specifically two-way-ranging and two-way-Doppler measurements (relying mostly on the latter) via a transponder on or near the asteroid.

The possibility of a lander equipped with a transponder is sometimes proposed, but the technological challenges associated with the thermal conditions prevailing on the surface and the likely degradation of the signal quality resulting from this and the issue of occultation due to the asteroid rotation and shape cast some doubt on the usefulness of landing a beacon.

Instead, we propose using a spacecraft in a stable orbit around the asteroid [12] for RSE. If the orbit is viewed edge-on, i.e., the Earth Aspect Angle not too far from 90 deg (as is the case throughout most of the 6-month study period in 2013, Fig. 7), the ripple in the measurements due to the spacecraft motion on its orbit at a relative velocity of 4 to 7 cm/s (see Table 3) will be clearly discernible.

Another important input parameter for the RSE assessment is the achievable measurement accuracy. For Doppler, the 1σ-uncertainty is assumed as less than 1 mm/s in X-band. For ranging, the assumed 1σ-uncertainty is less than 2 m, the measurement bias 20 m. The stochastic noise of the spacecraft accelerations is not known at this point. In the proposed orbital configuration the effect of the inhomogeneous gravity field should be limited, and the absence of control manoeuvres on the self-stabilizing orbit is a bonus, helping to keep the noise down.

Actual assessment of the achievable accuracy, based on this input and following multi-month measurement campaigns, is a very complex task, the results of which depend on a variety of parameters. Operational experience for RSE using an orbiter around a small asteroid is limited. The Japanese Hayabusa mission had different objectives and a different mission profile [18]. The experience obtained by NEAR-Shoemaker around the much larger 433/Eros could be, to some extent, applicable [12][16].

A full study still needs be undertaken, results therefore cannot be given in this paper, which is intended as a summary of the mission analysis inputs that lead to the RSE study.

FURTHER SCIENCE

Having a spacecraft in a close orbit around the asteroid constitutes a unique opportunity for further science in addition to the RSE, e.g., observations of the asteroid surface and interior and its interaction with the ambient environment. The mission design proposed here foresees SEP, so the spacecraft will be equipped with solar arrays of significant size, allowing for even relatively power-hungry experiments such as subsurface radar. Additionally, one or several surface packages could be deployed for in-situ surface science, the mission mass budget allowing.

IMPACTOR TRANSFERS

This section provides a brief overview of HVI transfers. 99942/Apophis prior to its Earth flyby in 2029 is an Aten type asteroid. For Atens it is rather easy to design effective Impactor trajectories. This is a consequence of the relative geometry between the orbits of Earth and asteroid.

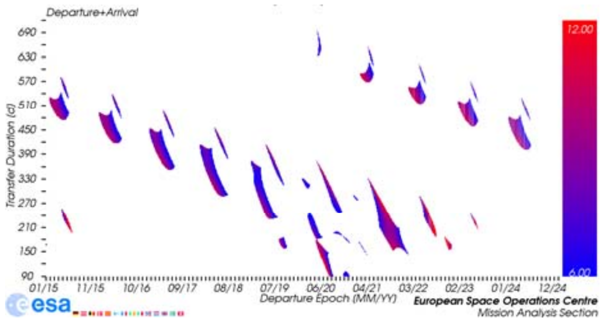

Fig. 4 : Pork-chop plot for direct HVI transfers

Since Atens cross the Earth orbit an effective HVI can often be achieved simply by sending out a spacecraft at a low departure velocity; and letting the asteroid run into it. Venus or Earth swingbys are mostly not necessary and only add complexity to the mission.

Fig. 8 shows the pork-chop plot for direct HVI transfers to Apophis. Unlike the rendezvous case (Fig 2), there are HVI transfer options every year. However the pork-chop plot only considers ballistic transfers without DSM.

HVI design requires a trade-off between maximizing either the imparted momentum or the induced change in semi-major axis (Δs.m.a.). Also of relevance is the asteroid illumination which affects the navigation during the terminal approach.

When designing the HVI trajectory to maximize Δs.m.a., a near-tangential impact is obtained, which tends to limit the impact velocity. An additional issue is the illumination. While approaching the asteroid tangentially, the spacecraft "sees" the asteroid at around 90 deg phase angle (as a half-moon).

When maximizing the imparted momentum instead, transfers with approach in the radial outbound direction can be found. Here, the asteroid is viewed fully illuminated at approach. Compared to a "half-moon" at the same distance, the fully illuminated body would appear brighter by a factor of π.

So for this class of transfers, the asteroid's fully illuminated face could be detected from a more than three times larger distance, which would significantly facilitate the autonomous on-board navigation and thus enhance the chance of mission success, i.e., scoring a hit.

Table 4 compares the two different design approaches. The launcher is again VEGA. Launch is assumed to take place in 2015-2017, making use of the precise orbit determination performed in 2013-2014 by an orbiter and RSE.

The variation in semi-major axis is hardly better in the maximum Δs.m.a. case. This difference corresponds to difference in the change in the period of Apophis of 0.01s.

However, the maximum Δs.m.a. solution leads to an asteroid viewed during final approach at a phase angle of more than 90°: the s/c sees the asteroid as less than a half -moon.

Conversely, the maximum-imparted-momentum transfer approaches the asteroid under much better illumination conditions, though at higher speed. Clearly, here the navigation and GNC aspects should be studied carefully and a trade-off made.

	Max Δa	Max Final Momentum
Departure	7 May 2015	12 Aug 2015
Arrival	1 Nov. 2016	24 Dec 2016
Final mass	355 kg	430 kg
ΔV imparted	9.2e-5 m/s	2.4e-4m/s
ΔV_t imparted	8.6e-5 m/s	6.9e-5 m/s
Δ s.m.a.	680 m	647 m
Period Variation	0.2065 s	0.1966 s
Delta Position per year	7.00 km	6.66 km

Table 4 : Optimal Impact Trajectories for Apophis

Table 4 shows that a deflection of around 7 km per year (of time difference between HVI deflection and Earth encounter) is achievable with a small impactor. Even with just a few years between the deflection HVI and the 2029 Earth encounter, the accumulated deflection should suffice to guarantee safe avoidance of the 600 m "keyhole". An additional interesting feature of the HVI options is that launch windows are available every year and in all cases they trajectory can lead to a high velocity impact.

On the other hand, these results confirm that a post-2029 HVI deflection mission, if it is found that an Earth impact is likely in 2036, is impossible, even with a much larger spacecraft.

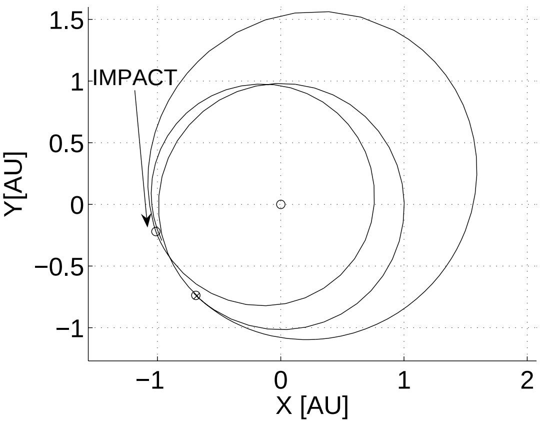

Fig. 5 : Maximum Δs.m.a . Change HVI trajectory

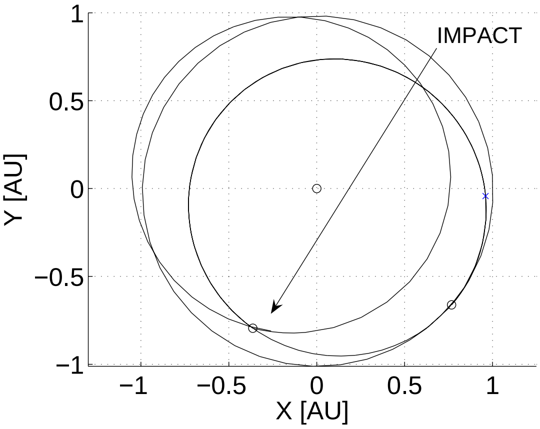

Fig. 6 : Maximum Momentum Transfer HVI trajectory

CONCLUSIONS

This paper presents end to end mission analysis results for a low-cost orbiter mission to asteroid 99942/Apophis. Based on launch in 2011 with the small European launcher Vega and a solid propellant upper stage, the transfer lasts 30 months with arrival in 2013, after which the spacecraft would be placed into a self-stabilizing terminator orbit with a radius of less than 750 meters.

One gridded ion thruster is used as solar electric propulsion unit. The arrival mass is around 337 kg, which would allow a sizeable added scientific payload. The prime mission objective is to provide characterization of the asteroid orbit and its perturbations via precise long-term tracking of the orbiting spacecraft.

Two windows for RSE exist, one in 2013, one in 2021. In 2029, the asteroid will have a close encounter with the Earth. There is a strong interest in being able to predict its post-2029 orbit to assess the risk of collision in 2036. If counter-measures are necessary, they should be taken well before the 2029 encounter. To this end, the precise determination and characterization are vital. After 2029, the required effort rises by 4 orders of magnitude and deflection might be impossible.

In addition to the detailed description of the orbiter mission design, the options for complementary missions aiming at deflection via a high velocity impact are assessed and presented.

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