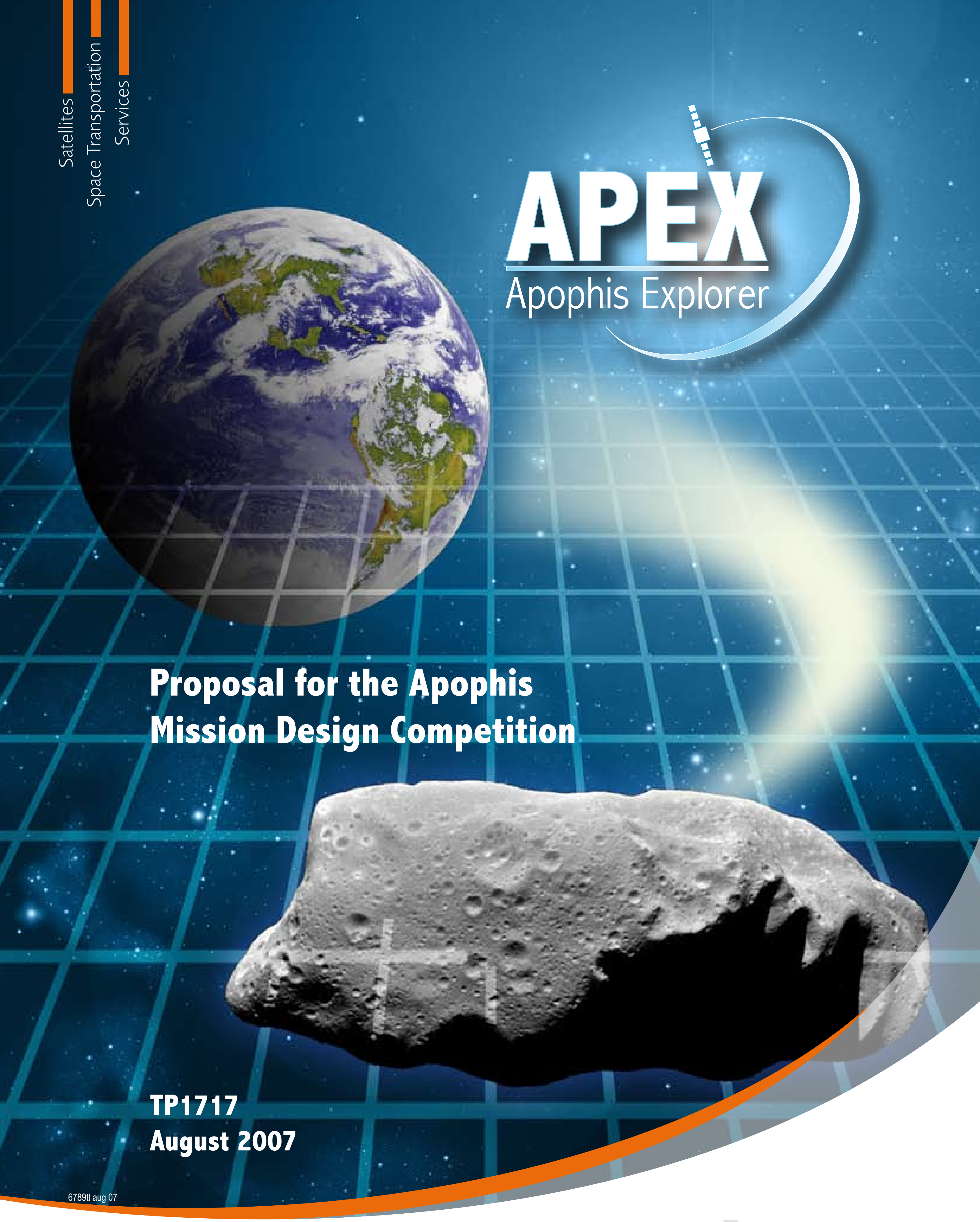

APEX

Apophis Explorer

Proposal for the Apophis Mission Design Competition

TP1717

August 2007

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All the space you need

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Recipient of Prize Money

If this proposal is successful the team has agreed the prize money should be donated to the Annual Scholarship Programme of the International Space University (ISU). Of the over 2400 people who have graduated from ISU, the vast majority have received some form of financial assistance to make their experience possible. Last year's Annual Scholarship Campaign provided over $126,000 in student aid funds. This money was used to help Summer School students from Brazil, India, Japan, Mexico, Poland, Turkey and the United States, and it aided Masters Students from China, Indonesia, Colombia and Nigeria. The funds can be donated on-line at the ISU Website: www.isunet.edu

TABLE OF CONTENTS

EXECUTIVE SUMMARY

On April 13th 2029 asteroid Apophis will have a close encounter with Earth which could send it on a collision course with our planet. Unfortunately, the orbit of Apophis is subject to a number of gravitational and non-gravitational perturbations, some of which are currently known or modelled with limited accuracy. These uncertainties are then amplified by the orbit propagation process and result in significant errors in position determination at the close passage in 2029. The Yarkowski Effect (YE) alone can shift the position of Apophis by over 100 km between 2017 and 2029, which means it cannot be ignored in the orbit propagation models.

Since a 10% error in modelling non-gravitational perturbations like the YE can translate into more than 30 km error (3σ) in the position of Apophis in 2029, these perturbations need to be modelled with accuracies of a few percent, more than can be achieved by radio tracking alone. The thermal properties of Apophis in particular are the dominant error contributor in YE estimations, so that a dedicated thermal mapping campaign is necessary. Therefore, tracking the orbital evolution of Apophis with very high accuracy needs to be complemented by a determination of the physical properties of Apophis with unprecedented precision.

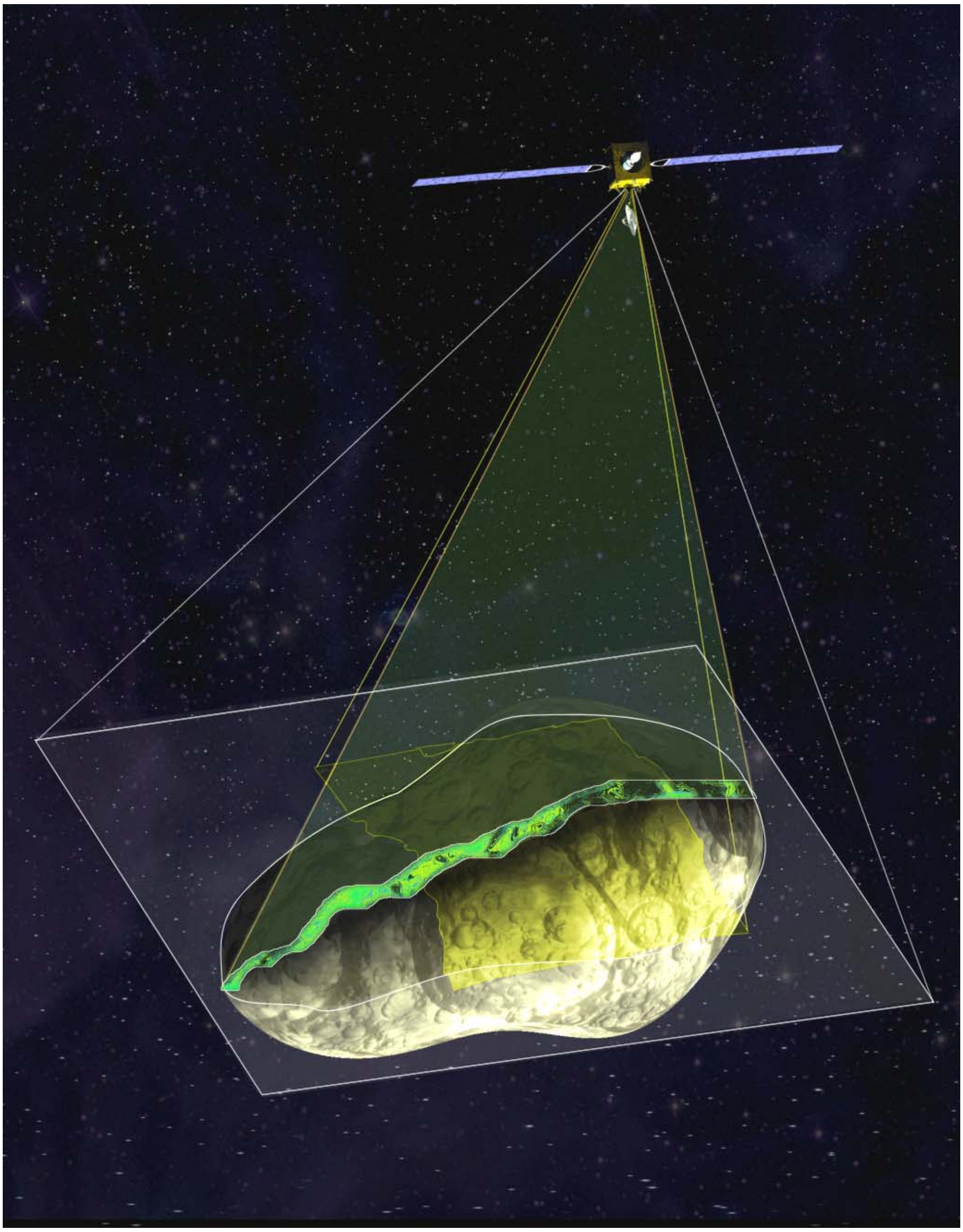
The Apophis Explorer (APEX) mission achieves this through the most detailed and extensive investigation of an asteroid ever performed. An orbiter, instrumented with a sophisticated remote sensing payload, will study the asteroid for several years, tracking Apophis' motion along its orbit with a Radio Science Experiment and determining the orbital elements to parts per billion. At the same time APEX will also study all of Apophis' key physical characteristics and in particular its thermal properties with high spatial and temporal resolution.

The high area to mass ratio of Apophis (compared e.g. to Eros), represents a double challenge in this respect:

- Firstly, its own low gravity leads to the orbiter motion being influenced, to an unusually high degree, by non-gravitational forces acting on the spacecraft. As a result, these forces have to be modelled, or directly measured, to very high accuracy.
- Secondly, the low gravity of Apophis means that, for remote sensing, ground track speeds tend to be very low (cm/s) and orbits decay over time, making global mapping a slow process with standard orbits.

To meet the first challenge, during radio science the APEX spacecraft is placed in the most (and only) stable low-altitude orbit around Apophis: a terminator orbit with the spacecraft keeping a constant cross section to solar radiation. This allows long manoeuvre-free periods when the spacecraft orbit can be tracked to the ground with great accuracy. An accelerometer on board is also necessary to measure non-gravitational perturbations on the spacecraft.

To meet the second, the APEX mission is designed to map the asteroid surface at a broad range of wavelengths simultaneously with high spatial and temporal resolution. This is achieved relatively quickly thanks to a "forced" fast orbit around Apophis, using the spacecraft's own solar electric propulsion system to generate "artificial gravity".



APEX Orbiter Gathering Data on Apophis

1 MISSION REQUIREMENTS

1.1 MISSION OBJECTIVES

The Apophis Mission Design competition calls for the design of a mission whose primary requirement is to reduce the long dimension of Apophis' 3σ error ellipse to 14km, 12years before the April 2029 close encounter. Therefore knowledge of all relevant parameters and effects affecting Apophis' orbit, obtained by April 2017, has to be sufficient to propagate the orbit 12 years forward in time, within the specified error ellipse.

In this proposal, the error ellipse requirements are interpreted as applicable in the b-plane, which is the standard tool to describe Near Earth Object (NEO) encounters with Earth. A comprehensive review of Apophis and its threat to Earth can be found in [Ref 1]. Keyholes are defined as regions in the b-plane with the following property: if a NEO passes through them, it will definitely hit Earth at some time in the future [Ref 2,Ref 3]. The ratio between the Earth disk at the impact encounter and the size of the keyhole at the pre-impact encounter is also a measure of the increase in energy that is required to successfully deflect the NEO after the pre-impact encounter. In the case of the 2029 encounter of Apophis, this ratio is of 4-5 orders of magnitude, and therefore any deflection needs to be accomplished well before then. This is illustrated in Figure 1-1 [Ref 4]. The final position of Apophis on the 2029 b-plane, and also the uncertainty in this position, are influenced by two types of contributions:

1. **Initial conditions:** any uncertainty in the initial state of Apophis (Ephemerides) will propagate to the final error ellipse.
2. **Forces perturbing Apophis' orbit:** uncertainty in the models or measurements of forces acting on the asteroid throughout the propagation period.

The effect of all these contributions has to be implemented in an orbital propagator, so that the absolute magnitude at the end of the propagation period, as well as the sensitivity to parameter uncertainties, can be estimated.

1.2 APOPHIS ORBIT PERTURBATIONS

1.2.1 Gravitational Perturbations

All Solar System bodies exert gravitational forces on Apophis. These forces can be calculated with an uncertainty small compared to the requirements, and are routinely included in orbital propagators. An exception to this are perturbations from the smaller main belt asteroids, which are not well known. However Apophis' orbit is largely inside the Earth's orbit, and perturbations from these asteroids can be treated as negligible.

1.2.2 Non-Gravitational Perturbations

There are two separate forces associated with the solar radiation incident on Apophis:

- the Solar Radiation Pressure, which is a result of photon momentum exchange between the sunlight and the asteroid
- the Yarkovsky Effect, which is a result of asymmetric thermal re-emission of absorbed Solar Radiation by a rotating body

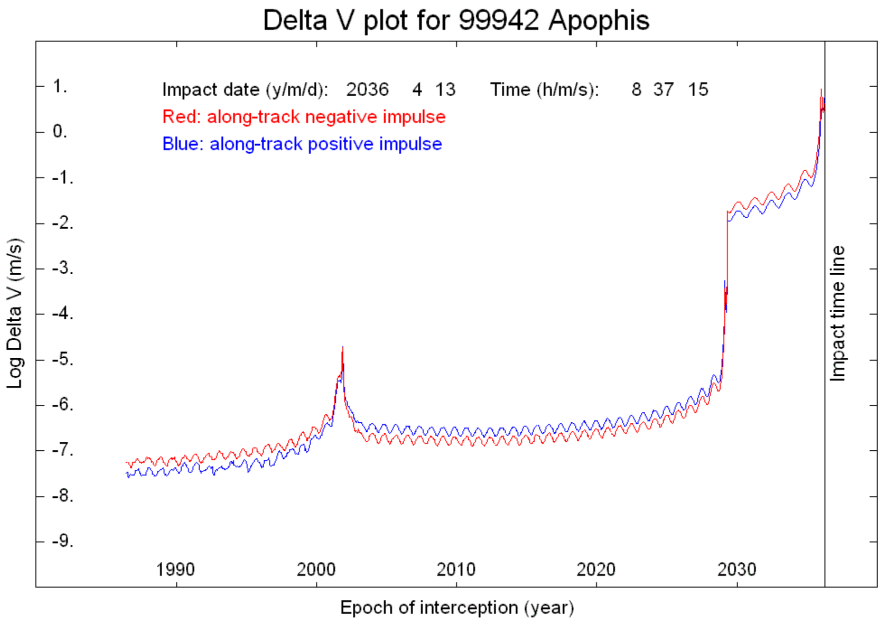


Figure 1-1 Along track ΔV requirements for Apophis deflection [Ref 4]

Solar Radiation Pressure

The magnitude of the Solar Radiation Pressure (SRP) depends on the albedo of the object, its reflectivity and shape, and in particular on the angle of the surface with respect to the incident radiation. The resulting SRP acceleration is proportional to the area to mass ratio, which for a uniform sphere is inversely proportional to the radius. All other parameters being equal, the SRP effect is thus more important for smaller asteroids than for larger ones. The net effect of the SRP on the orbit is a superposition of a cyclical contribution, with a period equal to the sidereal period, and a secular contribution.

Yarkovsky Effect

Solar radiation that is absorbed by a body in a heliocentric orbit is subsequently re-emitted as thermal radiation. Rotation of the body can lead to an asymmetric along-track thermal emission, and the resulting acceleration can modify the orbit of the body significantly, particularly over long periods of time [Ref 7, Ref 8].Depending on the orientation of the spin axis relative to the orbital plane, two separate effects can be distinguished: a spin axis perpendicular to the orbital plane results in the diurnal Yarkovsky Effect (YE), whereas a spin axis lying within the orbital plane will result in the seasonal YE. The diurnal effect depends on the rotation period of the object itself, whereas the seasonal effect depends on the orbital period.

In practice, the YE on any real object will be a combination of diurnal and seasonal effects. For Apophis, the orientation of the spin axis is essentially unknown. The YE has been computed for the two extremes – normal to orbital plane and lying in it – and the diurnal YE is found to be significantly larger than the seasonal one. An accurate long term prediction of the YE requires knowledge of the spin axis orientation evolution over time.

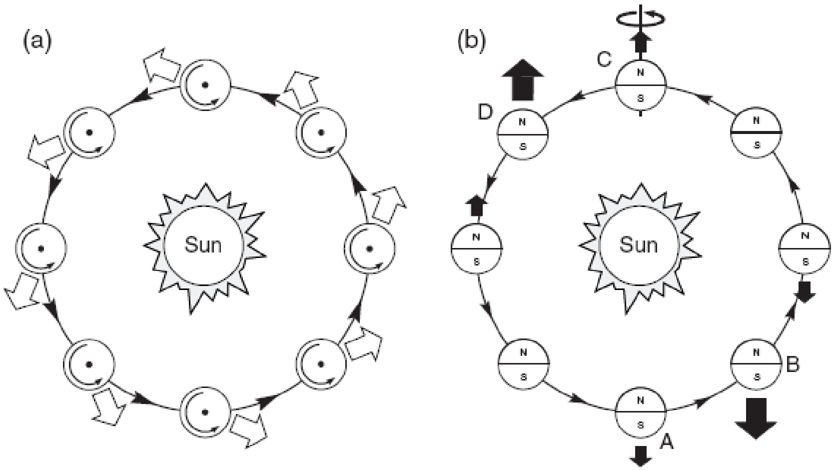


Figure 1-2 Diurnal and seasonal Yarkovsky effects in heliocentric orbits [Ref 8]

It is evident that the YE is a function of many parameters:

- Incoming solar flux and asteroid surface absorptivity – this determines the input energy received by the object
- Thermal inertia (a combination of thermal conductivity, surface layer density and specific heat capacity) – this determines the response, spatial and temporal, to the heat input
- Rotation state of Asteroid – the orientation of the spin axis determines diurnal and seasonal effects, and rotation period sets the frequency at which input energy is applied
- Thermal emissivity – this determines the balance between input energy and emitted radiation, and therefore affects the surface temperature and thermal radiation force

An analytical YE model exists for the case of a spherical object, of uniform thermal parameters, based on the assumption that surface temperature variations are small compared to the average [Ref 7]. Numerical models have been developed to significantly generalise this case [Ref 9].

1.2.3 Magnitude of Non-Gravitational Forces

The YE and the SRP effect have to be quantified in order to assess their significance. We have developed an orbital propagator to estimate the position of Apophis in April 2029, starting from initial conditions in April 2017. The propagator includes:

- Gravitational effects of the Sun, Moon, Earth and all planets except Pluto
- Analytical model for both diurnal and seasonal YEs on spherical uniform objects
- Analytical model for the SRP

A large number of Apophis physical parameters enter the estimation, in particular, of the YE. The current knowledge of Apophis is summarised in Table 1-1, and shows the nominal values used in the propagation. Some of the values are derived from ground-based observations, others are estimates based on experience with other asteroids. The orientation of the spin axis is essentially unknown; in order to be conservative, an orientation normal to the orbital plane has been assumed, in order to maximise the diurnal YE. The resulting displacements that were computed due to non-gravitational forces, compared to the nominal, gravitational only position, are shown in Table 1-2.

Compared to the mission requirements of 14km as 3σ error ellipse long dimension, equivalent to a 1σ requirement of 4.7km, one immediate conclusion can be drawn: for Apophis, it will be necessary to determine, model or measure the YE with a relative accuracy of the order of 1-2%, in order to meet the mission requirements. Apophis' thermo-optical parameters are the ones that are least constrained (see Table 1-1). Gaining knowledge of the YE on Apophis at the 1-2% level is thus identified as one of the key challenges of this mission.

Parameter	Value	Unit	Uncertainty	Source / Comment
Semimajor axis	0.9222614	AU	2.4E-8	JPL Small Body database
Eccentricity	0.1910594	-	7.6E-8	"
Inclination	3.3313146	deg	2.0E-6	"
Pericentre argument	126.38557	deg	1.1E-4	"
Longitude of node	204.45915	deg	1.1E-4	"
Mean anomaly	307.36307	deg	3.2E-5	"
Perihelion	0.75	AU		Derived from parameters above
Aphelion	1.1	AU		"
Sidereal Period	0.89	yr		"
Asteroid radius	135	m	30	[Ref 5]
Bulk Density	2700	kg/m^3	1300-2700	Typical asteroid range
Asteroid mass	2.78E+10	kg	1.3E+10 to 2.8E+10	Derived from radius and density
Rotation period	30.6	h	0.01	[Ref 6]
Emissivity	0.9	-		
Visible albedo	0.33	-	0.08	[Ref 5]
Surface Thermal Inertia	116.62	$J/s^{1/2}/m^2/K$		Derived from thermal parameters below
Surface density	2000	kg/m^3		Typical value
Surface conductivity	1.00E-02	W/m/K	1.0E-3 to 1.0E-1	Large variations possible
Heat capacity	680	J/kg/K		Typical value

Table 1-1 Apophis parameters and uncertainties (current knowledge)

Effect	Displacement on 2029 b-plane [km]
Yarkovsky Effect	110.8
Solar Radiation Pressure	2.8

Table 1-2 b-plane displacement of Apophis due to non-gravitational effects

1.2.4 Impact of Yarkovsky Effect on Apophis Orbit Determination

The standard way of determining the unknown orbit of a target body is by means of a Radio Science Experiment (RSE), in which an orbiter in a gravitationally bound orbit is tracked from Earth. The motion of the object's CoM is then derived from that of the tracked object. The success of the orbit determination depends on including all relevant forces acting on both the orbiter and the body.

For Apophis, the Yarkovsky effect (YE) needs to be known with an accuracy of about 1-2%. This force depends on a large number of currently only poorly constrained thermo-optical parameters. Moreover, the functional dependence on these parameters is only known for relatively simple models. The conclusion is that standard RSE alone will not be sufficient to meet the mission requirements.

Instead, it is essential to measure directly all relevant physical parameters entering the YE. This requires a very detailed characterisation of Apophis in the early mission phase, comprising:

- Gravitational characterisation
- High resolution visual imaging campaign
- Thermal mapping campaign

With these measurements, a thermo-optical Finite Element Model of Apophis will be developed and refined during the initial mission phase. A functional dependence of the YE on all relevant parameters will then be derived from this model and used in the RSE orbit determination process.

1.3 SENSITIVITY ANALYSIS

1.3.1 Methodology

We need to estimate the accuracy to which all relevant parameters affecting the final error ellipse have to be measured: Apophis Ephemerides, YE and SRP parameters. Our propagator is also used for the sensitivity analysis. Nominal estimates are used for every parameter that enters the propagator, i.e. for the initial conditions as well as for all variables used in force calculations. The sensitivity is then estimated by varying one parameter at a time, propagating Apophis between 2017 and 2029, and computing the shift in the b-plane position at the end of this period.

1.3.2 Error Budgets

The total 3σ error budget of 14km for the long axis of the Apophis error ellipse needs to be apportioned to the many parameters entering the budget. Uncertainty requirements have been derived on the basis that the contribution of the uncertainty in Ephemerides is of the same order of magnitude as that due to the Yarkovsky effect (as will be seen, the contribution from the SRP is negligible). Table 1-3 to Table 1-5 present the various error budgets, based on 1σ errors in order to ensure consistency between measurements. The requirement for the long error ellipse axis is then 4.7km. To be conservative, we add the errors linearly within each contribution, assuming worst-case correlation between the parameters, then add the three contributions in quadrature, since correlations across the contributions are small. The resulting total 1σ error budget is then equal to $(3.28^2+0.04^2+2.55^2)^{1/2} \approx 4.2$km, which is equivalent to a 3σ error of 12.6km. This provides a 10% margin with respect to the requirements.

Parameter	Value	Unit	Error (1σ)		Sensitivity		B-plane error (1σ)	
Solar Flux @1AU	1363	W/m^2	0.25	%	1	%/%	0.277	km
Spin axis inclination	90	deg	0.1	%	0.18	%/%	0.020	km
Asteroid radius	135	m	0.5	%	1.111	%/%	0.615	km
Asteroid mass	2.78E+10	kg	0.05	%	1.111	%/%	0.062	km
Rotation period	30.62	h	0.25	%	0.402	%/%	0.111	km
Emissivity	0.9	-	2	%	0.199	%/%	0.441	km
Visible albedo (surface absorptivity)	0.33	-	1	%	0.43	%/%	0.476	km
Surface Thermal Inertia	116.62	$J/s^{1/2}/m^2/K$	1.5	%	0.772	%/%	1.283	km
Surface density*	2000	kg/m^3	1	%	0.386	%/%	0.428	km
Surface conductivity*	1.00E-02	W/m/K	1	%	0.362	%/%	0.401	km
Heat capacity*	680	J/kg/K	1	%	0.362	%/%	0.401	km
Total (linear sum)							3.28	km
Total (quadratic sum)							1.59	km

* included in Thermal Inertia

Table 1-3 Error Budget for Yarkovsky Effect Parameters

Parameter	Value	Unit	Error (1σ)		Sensitivity		B-plane error (1σ)	
Solar Flux @1AU	1363	W/m^2	0.25	%	1	%/%	0.007	km
Visible albedo (surface absorptivity)	0.1	-	1	%	0.3	%/%	0.008	km
Asteroid radius	135	m	0.5	%	2	%/%	0.028	km
Asteroid mass	2.78E+10	kg	0.05	%	1	%/%	0.001	km
Total (linear sum)							0.04	km
Total (quadratic sum)							0.03	km

Table 1-4 Error Budget for Solar Radiation Pressure Parameters

Parameter	Value	Unit	Error (1σ)		Sensitivity		B-plane error (1σ)	
Semimajor axis	0.922261415	AU	5	m	0.111	km/m	0.555	km
Eccentricity	0.191059415	-	2.0E-9		9.42E+7	km	0.188	km
Inclination	3.331314642	deg	2.0E-8	deg	2.79E+6	km/deg	0.056	km
Pericentre argument	126.3855713	deg	2.0E-9	deg	2.84E+8	km/deg	0.568	km
Longitude of node	204.4591523	deg	2.0E-9	deg	4.34E+8	km/deg	0.868	km
Mean anomaly	307.3630785	deg	2.0E-9	deg	1.59E+8	km/deg	0.318	km
Total (linear sum)							2.55	km
Total (quadratic sum)							1.23	km

Table 1-5 Error Budget for Ephemerides Parameters

2 MEASUREMENT PRINCIPLE AND PAYLOAD

2.1 APOPHIS ORBIT DETERMINATION

We propose to implement orbit determination from a closed orbit around Apophis. It offers a relatively direct determination of the CoM motion and also a low risk in terms of required technology development (as opposed to using a lander).

2.1.1 Radio Science Experiment Principle

A Radio Science Experiment (RSE) will be used to determine the motion of Apophis' CoM from ranging and Doppler tracking of the orbiting spacecraft from the ground. The basic principle of the experiment is to extract the desired, unknown parameters from a sophisticated fit of the spacecraft ranging and Doppler tracking measurements, to a model – incorporated within an extended Kalman filter – that includes all forces acting on the spacecraft. At any instant in time, the total force acting on the spacecraft, $F_{S/C}$, will be given by:

$$F_{S/C} = F_{Grav,SS} + F_{Non-Grav,SS} + F_{Grav,Apophis} + F_{Non-Grav,Apophis} + F_{Non-Grav,S/C}$$

Each of these force terms will in general depend on various parameters, which themselves may be a function of others. The force terms are described, and the parameter dependencies summarised, in Table 2-1. The Kalman filter will solve for all the above parameters simultaneously, based on the input provided by the spacecraft ranging and Doppler tracking measurements. This applies also to the non-gravitational forces acting on Apophis, namely the SRP and the Yarkovsky effect. It will only be possible to develop detailed models of these effects once the thermo-optical properties of Apophis have been determined in situ.

RSE benefits from continuous ranging and Doppler tracking for periods of time that are as long as possible. The length periods are mainly limited by the orbit stability and the need to offload momentum wheels. Stable terminator orbits at an altitude of 500m have been identified that offer continuous periods of up to 50 days, and the momentum wheels have been sized such that offloading will not be required during these 50 days. The orbital period of the spacecraft is ≈21 hours at this altitude. A continuous RSE measurement campaign including more than 50 complete orbits is therefore possible.

2.1.2 Spacecraft Ranging and Doppler Tracking

The main input to the orbit determination procedure will be the ranging and Doppler tracking measurements carried out between the Ground Station (GS) and the spacecraft. Single frequency X-band systems achieve ranging accuracies of 5m and are limited by GS biases. A dual Ka/X band system, as is being developed for BepiColombo, improves this to the level of <1m. Standard Doppler tracking routinely achieves accuracies of better than 0.1mm/s, using integration times of the order of 60s. These "instantaneous" accuracies are achieved along the line of sight.

We will carry out individual ranging measurements at a frequency of once every 10 minutes, and Doppler measurements at a frequency of once every 25 minutes, for continuous periods between correction manoeuvres. Given the long orbital period of the spacecraft around the asteroid, many data points will be available for each orbit. The use of a single GS results in 8 hour windows once a day.

In parallel with the RSE measurements, visual imaging of the asteroid using the Wide Angle Camera and laser altimeter ranging will be conducted to verify the position of the spacecraft relative to the asteroid.

Force	Parameters	Derived Parameters
$F_{Grav,SS}$ Solar System Gravitational	- S/C position - Solar System bodies (excl. Apophis) positions	
$F_{Non-Grav,SS}$ Solar System Non-Gravitational: - SRP - Solar Wind	- S/C position - S/C attitude - S/C surface properties	
$F_{Grav,Apophis}$ Apophis Gravitational	- S/C position - Apophis position - Apophis rotation state - Apophis Mass & Gravitational harmonics	Apophis position function of: - Gravitational Forces acting on Apophis - Non-Gravitational Forces acting on Apophis (SRP and Yarkovsky Effect)
$F_{Non-Grav,Apophis}$ Apophis Non-Gravitational: - Reflected solar flux - Thermal radiation	- S/C position - Apophis position - Apophis rotation state - Apophis thermooptical properties	Apophis position function of: - Gravitational Forces acting on Apophis - Non-Gravitational Forces acting on Apophis (SRP and Yarkovsky Effect)
$F_{Non-Grav,S/C}$ Non-Gravitational Forces from S/C itself: - RCS noise - Propulsive Fuel Leakage	- S/C design / operations - RCS / Propulsion properties	

Table 2-1 Forces on Spacecraft and Parameters for Orbit Determination

2.1.3 Non-gravitational Spacecraft Orbit Perturbations

The following non-gravitational forces will perturb the orbit of the spacecraft around Apophis:

- Solar Radiation Pressure
- Reflected Sunlight from Apophis (albedo)
- Thermal radiation from Apophis
- Solar Wind Momentum exchange
- Propulsive / Non-propulsive noise from Reaction Control System / Propulsion

Accelerations due to the last two forces have been estimated to be $<10^{-9}ms^{-2}$, and will not introduce significant uncertainty as will be derived below. For the other three forces, typical spacecraft accelerations due to these effects have been compared to the gravitational effects from Apophis. In order to be conservative, non-gravitational effects have been computed at perihelion. At the nominal altitude of 500m, two bulk densities have been considered, 2700kg/m^3 and 1300kg/m^3. The following, additional assumptions about the spacecraft have been used in estimating the non-gravitational contributions:

- Mass 700kg
- Surface area 15m^2 and 50% reflectivity for SRP estimate (solar panels, terminator orbit)
- Surface area 5m^2 for Apophis albedo, thermal radiation and Lambert reflection from Apophis

The results are shown in Table 2-2. As can be seen, by far the largest non-gravitational force on the spacecraft is the SRP.

Effect	Density 2700 kg/m^3		Density 1300kg/m^3	
	Acceleration [ms^{-2}]	Fraction of Gravity [%]	Acceleration [ms^{-2}]	Fraction of Gravity [%]
Solar Radiation Pressure	2.6x10^{-7}	5.9	2.6x10^{-7}	12.4
Apophis Albedo Radiation Pressure	2.0x10^{-8}	0.5	2.0x10^{-8}	1.0
Apophis Thermal Radiation Pressure	1.5x10^{-9}	0.03	1.5x10^{-9}	0.07
Central Gravity	4.4x10^{-6}	99.3	2.1x10^{-6}	99.3
Higher Order Gravity	3.0x10^{-8}	0.7	1.4x10^{-8}	0.7

Table 2-2 Magnitude of non-gravitational perturbations of spacecraft orbit

2.1.4 Accuracy of Apophis Orbit Determination

The accuracy of the orbit determination will be limited by systematic effects, introduced through incomplete knowledge of the non-gravitational forces acting on the spacecraft. These systematic effects are likely to dominate the statistical uncertainties.

SRP is the most significant of these non-gravitational forces. In a terminator orbit, the orbital focus will be offset from the CoM of Apophis by a distance ε, given by the balance between SRP and gravitational pull from the asteroid (see Figure 2-1). Provided that ε is small compared to the orbital radius R, it will approximately be given by

$$\varepsilon \approx \frac{a_{SRP}}{a_{Grav}} R$$

where a_{SRP} is the expected acceleration due to the SRP, and a_{Grav} is the gravitational acceleration. If this ratio is 12.3% then at an altitude of 500m the resulting offset is 78m.

The accuracy requirements on the semi-major axis for Apophis' CoM are 5m, and therefore the equivalent accuracy on the position of the spacecraft will be also of the order of a few meters. If the contribution of the SRP is required to be <2m, then the SRP acceleration needs to be modelled in the orbital determination, and determined by it, to an accuracy of <2.5%. In terms of absolute acceleration, this is equivalent to about $6x10^{-9}ms^{-2}$. Modelling of SRP effects on the spacecraft needs to include a detailed surface model, including surface properties such as reflectivity, and their degradation over time. From experience, only an accuracy of 10% is considered to be achievable this way.

Therefore it is concluded that an on-board accelerometer, of sensitivity better than $5x10^{-9}ms^{-2}$ in all directions, is required to ensure that the Apophis' CoM orbit determination can be achieved with the necessary accuracy.

2.2 MEASUREMENT REQUIREMENTS

The derived measurement requirements are summarised in Table 2-3 and are described below.

2.2.1 Radius, Shape and Volume

Estimates of both Yarkovsky and solar radiation pressure effects require measurement of the asteroid radius. For a non-spherical object, a shape model describing an irregular surface needs to be developed. The required accuracy of 0.5m as derived from the Yarkovsky effect sensitivity analysis is used to set the spatial resolution requirement for visual imaging. A margin of 40% is applied and thus a spatial resolution requirement of 0.3m for the visible imager is obtained.

Absolute sizing of the asteroid requires knowledge of the distance to the asteroid. A laser altimeter is used for this, aligned with the optical imager for correlation purposes. The accuracy requirement on the distance measurement requirement is 0.5% as derived from the sensitivity analysis. For visual correlation, a laser footprint comparable to the image resolution would be desirable.

2.2.2 Mass, Density and Mass Distribution

The mass and higher order gravitational multipoles of Apophis are best determined by tracking the spacecraft in a gravitationally bound orbit, carrying out a Radio Science Experiment (RSE). The desired parameters are obtained by fitting a model containing these and other parameters, representing non-gravitational effects, to the observed Doppler measurements. Doppler tracking is used for this rather than the ranging data, due to its higher sensitivity to small accelerations. The accuracy requirement on the determination of the main mass (monopole) term is 0.05%. Requirements on spacecraft ranging and Doppler tracking are 1m and 0.1mm/s, respectively. Gravitational parameters and shape model can be combined to obtain the mean bulk density of the asteroid, as well as information on the density homogeneity. This will finally result in a relation between Apophis' CoM and its centre of figure.

2.2.3 Rotation State

The rotation state of Apophis is defined by its rotation period, and the position and the orientation of the spin axis in inertial space. The requirement on the accuracy of the rotation period is 0.25%, or around 4.6 minutes assuming the nominal period. This is equivalent to an angular resolution of 0.9° and to a spatial resolution of 2m. The requirement on the accuracy of the spin axis orientation is set to 0.1°. External forces and torques acting on Apophis can lead to a precession of the spin axis over time. The Yarkovsky effect in particular is strongly dependent on the spin axis orientation. It is therefore envisaged to repeat the measurement of the rotation state at regular intervals throughout the mission.

2.2.4 Incident Solar Flux and Albedo

The incident solar flux and the visual albedo of Apophis directly determine the magnitude of non-gravitational forces on the asteroid. The accuracy requirements are derived directly from the sensitivity analysis, being 0.3% for the solar flux and 1% for the visual albedo.

2.2.5 Surface Temperature and Emissivity

The thermal inertia of Apophis, a crucial parameter affecting the Yarkovsky effect, will be estimated from measuring the surface temperature and its emissivity, using a Thermal Infra-Red Spectrometer and mapping the whole surface of Apophis. A numerical thermal analysis shows that the required accuracy of 1.5% for the thermal inertia requires an absolute temperature sensitivity of 0.6K. Combined with a ground track speed of ≈3cm/s this sensitivity has to be achieved after averaging for not more than 20s. The accuracy of the emissivity determination is derived from the sensitivity analysis as 2%.

Additional requirements result from the need to construct an accurate thermo-optical model of the entire asteroid, in order to derive functional dependencies for the orbit determination process. These requirements are:

- Global surface coverage
- Spatial resolution 10m
- Global mapping at 20 different Local Solar Times (LST)
- Correlation with optical images and shape model
- Thermal mapping campaign duration short compared to Apophis season

The last requirement is necessary to ensure a clear separation of diurnal and seasonal Yarkovsky effects.

2.2.6 Spacecraft Ranging and Doppler Tracking

The requirements on the accuracy of spacecraft ranging and Doppler tracking for the orbit determination process, are set to <1m and 0.1mm/s. In addition, continuous measurement periods spanning many spacecraft orbits around Apophis, uninterrupted by any station keeping or wheel-offloading manoeuvres, are required.

2.2.7 Non-gravitational Forces acting on Spacecraft

As described, the orbit determination requires knowledge of non-gravitational spacecraft accelerations at the level of $<6 \times 10^{-9} ms^{-2}$. Given the uncertainty in these forces, a considerable margin is applied and a requirement on the measurement of these forces at the level of $10^{-9} ms^{-2}$ is set.

2.2.8 Asteroid Composition

For the interpretation of both the thermal and albedo measurements, understanding the mineralogy of the asteroid is of great help. For this reason, a simple Near-IR spectrometer co-aligned with the main imager has been added to the payload complement.

Measurement Phase	Parameter	Requirement
Visual Imaging	Spatial resolution shape model	0.3m
	S/C – Apophis Distance	0.5%
	Visual Albedo	1%
	Rotation Period	0.25%
	Rotation Axis	0.1°
	Surface Coverage	Global
Gravitational Characterisation	Apophis Mass	0.05%
	Spacecraft ranging	1m
	Spacecraft Doppler tracking	0.1mm/s
	Non-gravitational force measurement	$10^{-9}ms^{-2}$
Thermal Mapping	Temperature sensitivity	0.6K in 20s
	Emissivity	2%
	Spatial resolution	10m
	Surface Coverage	Global
	Local Solar Times	20
	Campaign duration	<< Apophis year (324 days)
Orbit Determination Phase	Spacecraft ranging	1m
	Spacecraft Doppler tracking	0.1mm/s
	Non-gravitational acceleration measurement	$10^{-9}ms^{-2}$
	Uninterrupted periods between manoeuvres	>> one orbital period

Table 2-3 Measurement Requirements

2.3 MEASUREMENT CAMPAIGN

2.3.1 High Resolution Optical Imaging Campaign

The standard procedure to construct an asteroid shape model is based on high resolution optical images of the asteroid [Ref 10]. Images are taken from various positions, for example from within the equatorial plane and also from above the poles, and then combined to produce the final shape model. Simultaneous altimeter ranging is used to provide the absolute scale factor. The spacecraft will be placed in well-defined stand-off positions for this, at different latitudes above the equatorial plane of the asteroid and taking images for at least one whole rotation of the asteroid, with images separated by angles of 20°. Periodic station-keeping will be required to maintain the spacecraft position. Spacecraft pointing accuracy equivalent to one imager pixel needs to be maintained. The ideal distance to the asteroid will be chosen such that it completely fills the field of view of the imager, while still providing sufficient spatial resolution. For the nominal Apophis size of 270m diameter and the chosen FOV, this means that the optical imager should have a 2048x2048 pixel CCD.

The rotation period can be determined to the required accuracy of approximately 4.5mins or better during the imaging campaign. The location and orientation of the spin axis can also be determined by parameter estimation based on optical imaging. Angular uncertainties of 0.001° have been achieved previously [Ref 10], exceeding the requirements of 0.1°. The absolute solar flux also needs to be determined, and the known heliocentric distance is sufficient to predict the absolute solar flux at the required level of approximately 0.1% [Ref 12].

The visual albedo is determined from brightness measurements using the optical imager. With knowledge of the incoming solar flux, the albedo can be derived from the intensity of the reflected light. Photometric measurements of global asteroid albedos with an accuracy of 10% have been achieved [Ref 13], the limitation being the uncertainty on local surface viewing angles rather than photometric sensitivity. The high resolution imaging campaign, with spatial resolution of <0.5m, will be sufficient to measure the albedo, locally and at well known viewing angles, to about 1% as required.

The spin axis orientation can have a significant impact on the visual imaging campaign, as the asteroid illumination clearly depends on it. The larger the spin axis projection onto the orbital plane, the more likely it is that parts of Apophis are not illuminated until significantly later in the Apophis year. If this is the case, the mission offers sufficient flexibility to complete the shape model later on.

2.3.2 Gravitational Characterisation Campaign

It is proposed to orbit the asteroid in a terminator orbit, starting with a relatively high altitude (1500m) and then lowering the orbit as more information about the gravity field is gathered. A coarse measurement of the gravity field is done at the high altitudes, requiring only a small number of orbits, and then more orbits are spent at lower altitude to further refine the measurements. In principle, as low an altitude as is considered safe from a stability point of view is ideal. This altitude depends amongst other things on the exact asteroid shape, but will generally lie between 400m and 900m. Terminator orbits are relatively stable and therefore free of orbital or momentum wheel desaturation manoeuvres for significant periods of time. Using this approach, a mass determination accuracy to better than 0.01% has been demonstrated [Ref 10], meeting the 0.05% requirement for Apophis.

2.3.3 Thermal Mapping Campaign

A polar orbit is used to ensure full global coverage. The orbit altitude and the orbital period depend on the properties of the thermal spectrometer that will be used. Here, we have investigated the possibility of using 2 MERTIS spectrometers, such as developed for BepiColombo [Ref 16].

In order to clearly separate diurnal and seasonal effects, the duration of a full thermal mapping campaign has to be short compared to one Apophis year (324 days). The campaign is then repeated at different times during one Apophis year. As the gravitational field of Apophis is very low, we will use a 7hr powered orbit at 1000m to provide quicker global coverage. This approach has a significant impact on the propulsion system design but is essential to reduce the campaign duration.

An orbital period of 7 hours creates a repeat cycle of 91 hours. Each pixel will be imaged a minimum of 2 times at a given local time; this is achieved as the adjacent swaths overlap each other by 50% at the equator. The coverage is such that the maximum incidence angle between the MERTIS line-of-sight and the normal to the surface will be no more than 15° assuming a spherical body.

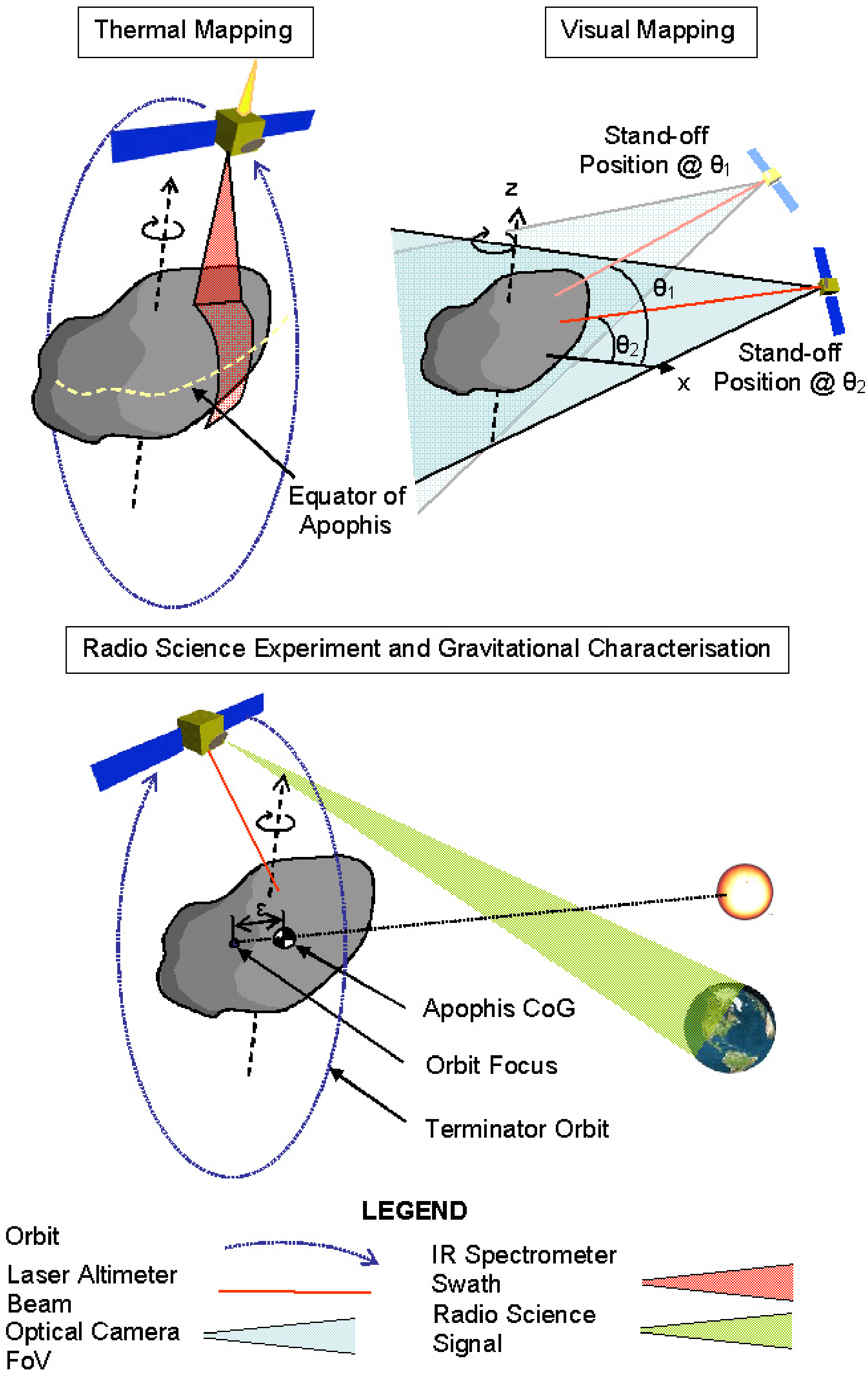


Figure 2-1 Measurement Phases of the APEX mission

A non-spherical body will also give minimum incidence angles of 15° assuming a local surface gradient of no more than 15°, any additional gradient will add to this observed angle.

In order to image the surface at varying Local Solar Times (LST), the orbital plane is precessed by a given angle and the repeat cycle repeated. Precessing the plane through 180° in equal steps of 18 degrees will ensure that every pixel is imaged at least at 20 different LSTs. The 10 precessions of the repeat cycle will take just under 40 days, instead of >280days using an unpowered orbit.

Continuous radial thrust of the powered orbit effectively provides the spacecraft with 'artificial gravity', significantly stronger than Apophis' natural gravity. This has the side benefit of providing the in-built safety of the spacecraft velocity exceeding the escape velocity, so that any thruster failure results in the spacecraft escaping from Apophis, rather than impacting it.

Near-IR spectra of the asteroid will also be taken in conjunction with the thermal mapping.

2.3.4 Radio Science Experiment Campaign

It is proposed to orbit the asteroid in a terminator orbit, in as low an orbit as is possible while maximising the periods between station keeping and wheel desaturations. The optimum altitude will depend, amongst other variables, on the gravity field of the asteroid. A strongly non-spherical shape will result in a higher altitude. A worst-case ellipsoid of 2:1 aspect ratio has been assumed to derive a baseline altitude of 500m. At this altitude, continuous measurement periods of 50 days are possible (see section 3.1.1).

2.3.5 Science Phase Timeline

Within the overall mission timeline, measurements will be carried out during the science phase, which is nominally defined as starting following orbit insertion. An arrival time of January 2014 leaves 3.2 years to achieve the mission objectives. This is equivalent to approximately 3 ½ full Apophis years.

The science phase can roughly be split up into three separate phases:

- Initial phase, starting with the end of the transfer to Apophis, including detailed visual imaging and gravitational characterisation
- One complete Apophis year, during which visual imaging, thermal mapping and radio science experiments will be carried out periodically, and the detailed asteroid model will be completed iteratively on the ground
- Two further Apophis years, during which mainly radio science experiments will be carried out. The aim is to compare these measurements with orbital predictions based on the model developed during the first Apophis year. Visual imaging and thermal mapping will be carried out as required for verification and model consolidation purposes.

The detailed measurement timeline is shown in Figure 2-2. As has been indicated before, both the high resolution visual imaging and the thermal mapping campaigns can be optimised as a function of the spin axis orientation, once this is known. A spin axis with a significant projection onto the orbital plane means that seasonal effects may in fact dominate diurnal effects. Whatever the spin axis orientation, one Apophis year will be sufficient to complete the asteroid model.

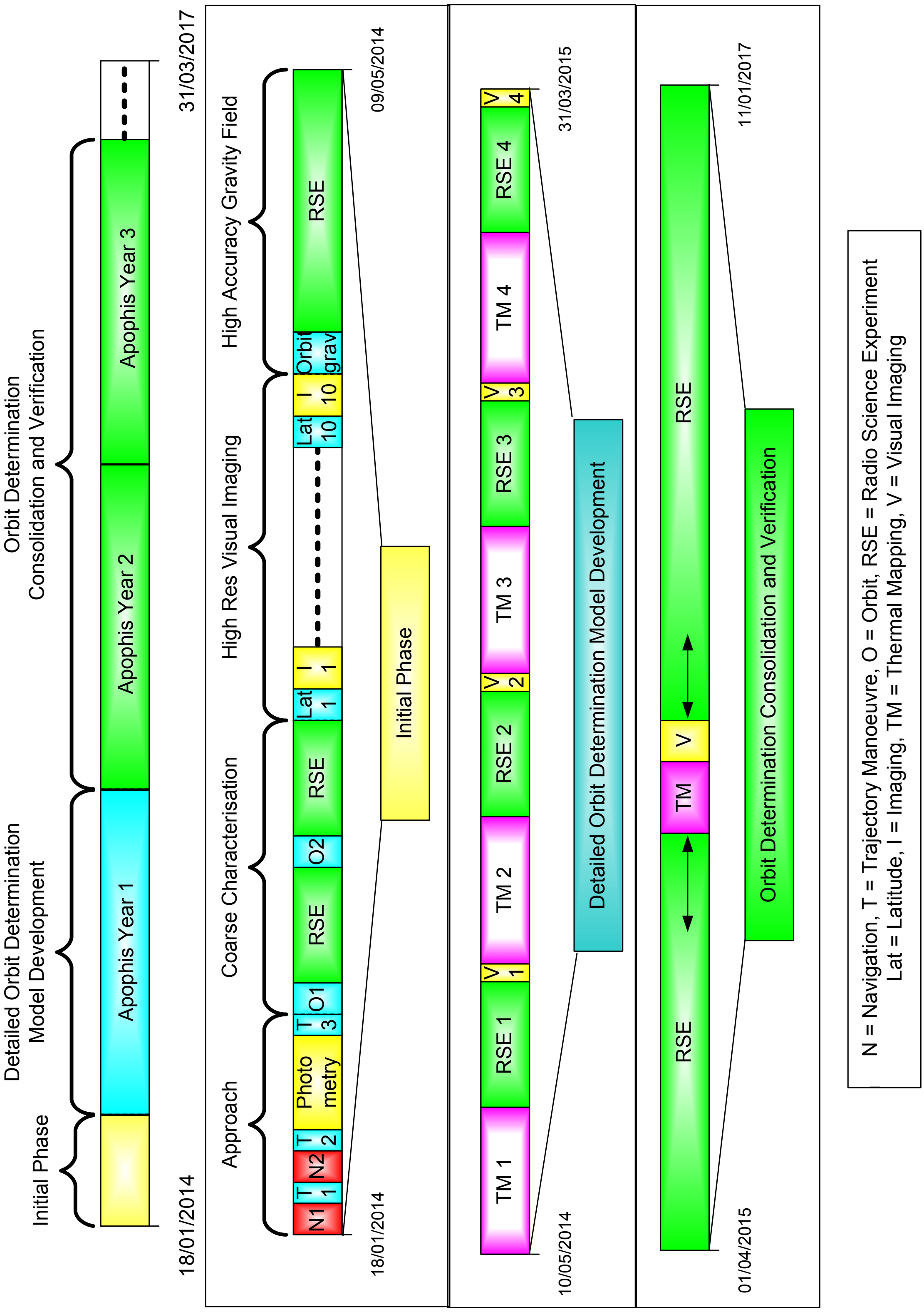


Figure 2-2 Science Phase Measurement Timeline

2.4 PAYLOAD INSTRUMENTS

A complement of payload instruments has been identified, which, combined with the measurement strategies described above, meets all mission requirements. In order to minimise the duration of the mission implementation phase, emphasis has been placed on instruments that either exist already or are being developed for approved flight programmes.

2.4.1 Optical Imagers

The spatial resolution of the optical imager will be given by its angular resolution multiplied by the distance to the asteroid. A camera based on the Dawn framing camera [Ref 14], for example, could be fitted with a 2048x2048 pixel CCD, with individual pixel angular size of 50μrad and a camera Field-Of-View (FOV) of 5.8°. With the nominal Apophis diameter of 270m, at a distance of 2.7km the asteroid would be filling the camera FOV, and an individual pixel would correspond to a spatial resolution of 0.13m. This resolution would be adequate to meet the requirements, with sufficient margin to account for variations in the overall asteroid size.

In addition to this high resolution camera, it is proposed to use a simple Wide Angle Camera (WAC) for basic visual imaging, to be used as verification / monitoring tool during the thermal mapping and radio science campaigns. The main requirement is that its FOV should be large enough to cover the whole asteroid from an altitude of 500m, i.e. 27°. A WAC such as that used for Philae on Rosetta meets this requirement [Ref 15].

2.4.2 Thermal IR Spectrometer

A suitable candidate IR spectrometer is the MERTIS spectrometer [Ref 16]. The expected temperature range on Apophis is between 200K and 400K and falls within the sensitive band of MERTIS (7-14μm). Two MERTIS IR spectrometers with 4° across-track FOV can be combined to give a total FoV of 8° and a swath width on the ground of 140m, from an altitude of 1000m. The spatial resolution of an individual pixel is then 0.7 x 0.7m meeting the 10m spatial resolution requirement. A grey body spectrum will be fitted to the output from each pixel, providing a measurement of local surface temperature and emissivity. Exposure times of a few seconds are sufficient to achieve the required temperature sensitivity of 0.6K. At a ground track speed of 3.3cm/s, this does not compromise the spatial resolution of 10m.

2.4.3 Laser Altimeter

It is proposed to use a laser altimeter such as the Hayabusa LIDAR [Ref 11]. This altimeter provides a vertical resolution of better than 1m, and has a beam divergence of 0.5mrad, resulting in a footprint of around 1m diameter from an altitude of 1km. This performance is sufficient to meet all requirements.

2.4.4 Radio Science Experiment

The high performance required by the Radio Science Experiment (RSE) demands utilization of a dual Ka- and X-band link system. Such a system is currently under development for the BepiColombo Mission. The target ranging accuracy is 0.15m, and the target Doppler tracking performance is approximately 0.0015mm/s using an integration times of 1000s [Ref 17]. This performance meets the requirements.

2.4.5 Accelerometer

It is proposed to use an accelerometer such as SuperSTAR on GRACE [Ref 18, Ref 19]. This accelerometer offers a sensitivity of better than $10^{-9}ms^{-2}$ in one direction, and better than $10^{-10}ms^{-2}$ in the other two directions, thus meeting the requirements.

2.4.6 Near-IR Spectrometer

A small, compact NIR spectrometer, based on extensive space heritage is included.

2.4.7 Summary

The main performance characteristics of the payload instruments are summarised in Table 2-4.

Payload Instrument	Characteristics	Use	Heritage
Wide Angle Camera	- FOV 53° - Angular resolution 0.9mrad / pixel	- Coarse Imaging - Position monitoring during Thermal Mapping and Radio Science	ROLIS, Rosetta
High Resolution Imager	- FOV 5.5° - Angular resolution 47μrad / pixel - 2048x2048 pixel CCD	- High resolution shape model and asteroid sizing	Dawn FC (modified)
Thermal IR Spectrometer	- Wavelength 7-14μm - FOV 4° - Spectral resolution 0.1μm - Angular resolution 0.7mrad / pixel	- Global Thermal Mapping	MERTIS Spectrometer
Laser Altimeter	- Vertical resolution <1m - Beamwidth 0.5mrad	- Absolute distance measurement for asteroid sizing - Correlation with visual and thermal imaging - Position monitoring during Radio Science	LIDAR-C
Accelerometer	- Sensitivity $<10^{-9}ms^{-2}$ in one axis - Sensitivity $<10^{-10}ms^{-2}$ in two axes	- Measurement of non-gravitational forces on spacecraft	GRACE Accelerometer
NIR Spectrometer	- Wavelength 0.9-2.5μm - FOV 64 x 0.25 mrad - Spectral resolution 3-30 nm	- Measurement of surface mineralogy	VIR/VIRTIS SIR – SMART-1 NIS-NEAR

Table 2-4 Performance Characteristics of Payload Instruments

3 MISSION ANALYSIS

3.1 ORBITING APOPHIS

Detailed formulation of the proximity operations at the asteroid requires knowledge of the physical parameters of the asteroid, in particular the mass, its distribution and the position of the spin axis. In principle, 4 different strategies are possible for performing the proximity operations:

1. Orbiting the asteroid within the classical Laplace Sphere Of Influence (SOI)
2. Flying an orbit in a region of space between asteroid's SOI and Hill's sphere, where the motion is strongly dependant on both solar and asteroid gravity
3. Flying a non Keplerian orbit, actively compensating (with thrust) for some disturbances
4. Flying a heliocentric orbit, in formation with the asteroid

3.1.1 Orbit for the Radio Science Experiment (RSE)

Option 1 is the preferred option for the RSE, as explained in section 2.1; however there are constraints in this option due to the balance between solar & asteroidal gravity and SRP. SRP is particularly harmful when acting in the orbital plane of the spacecraft, as it produces an acceleration modifying the semi-major axis and eccentricity of the orbit. Figure 3-4 shows an example of instability: after a few orbits the s/c crashes on the asteroid. The effect of SRP can be minimised by choosing an orbital plane normal to the Sun direction, i.e. by choosing a terminator orbit. In this case the effect of SRP on terminator orbits is that the focus of the orbit is not any longer the centre of mass of the asteroid, but rather offset from it. This is because the trajectory is determined by the equilibrium condition between the SRP and the gravity of the central body along the sun direction (Figure 3-1). The other benefit of terminator orbits is that they are eclipse-free.

In theory it is possible to find such equilibrium orbits at any altitude. In practice because of the presence of harmonic terms and Sun gravity, there are only limited ranges of radii at which stability is possible. Assuming a reasonable area/mass ratio of 0.025 for the spacecraft and an asteroid shaped as an ellipsoid with 2:1 ratio between major axis and minor axis, the result is that there is in fact a range of radii allowing this equilibrium or quasi-equilibrium condition.

- For an asteroid density of 1300 kg/m^3 this region extends from 400m to 800m. Orbital radii between 500 and 600m give excellent stability
- For an asteroid density of 2700 kg/m^3 this region extends from 500m to 1100m. The best choice is to stay in the 600-900m range

Figure 3-1 and Figure 3-2 show an example of achievable orbits. At these altitudes, SRP is much stronger and more relevant than the harmonic terms, i.e. the assumptions made on the shape have little effect on the orbital stability. It can be seen that the line of nodes rotates over the 50 days of the simulation: this is because the sun direction changes in time, so the equilibrium plane rotates as well.

These orbits are stable for long periods. Intervals of 50 days or more can be accommodated without any need for manoeuvres (Figure 3-3). This is exactly what is needed for the RSE. The analysis performed has shown that these orbits can be found, regardless of the actual mass and mass distribution of Apophis.

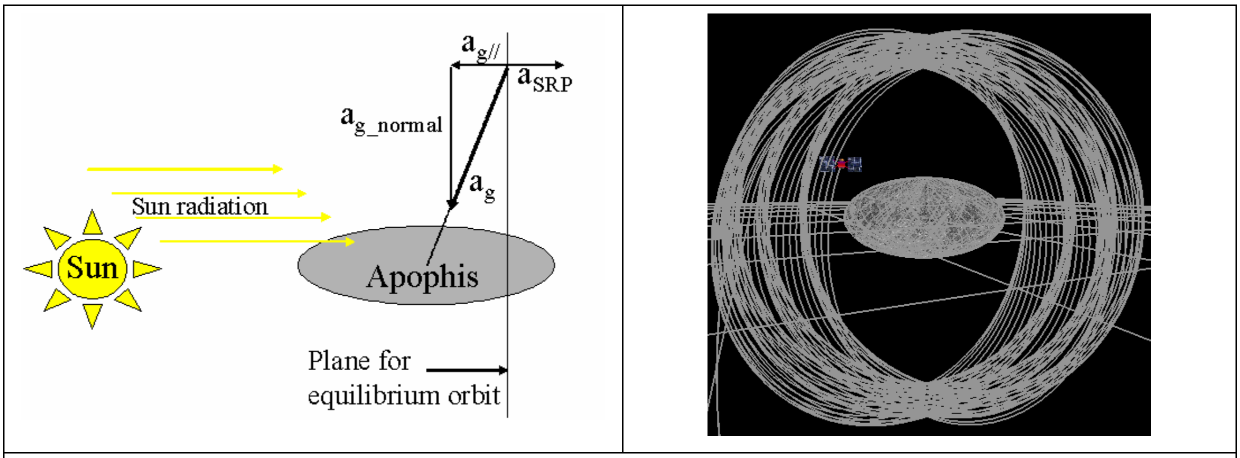


Figure 3-1: Orbit for RSE: terminator orbit. Left) Principle of equilibrium for terminator orbits. Right) Ex: r=600m. Asteroid density=2.7t/m3. Front View. 50 days simulation

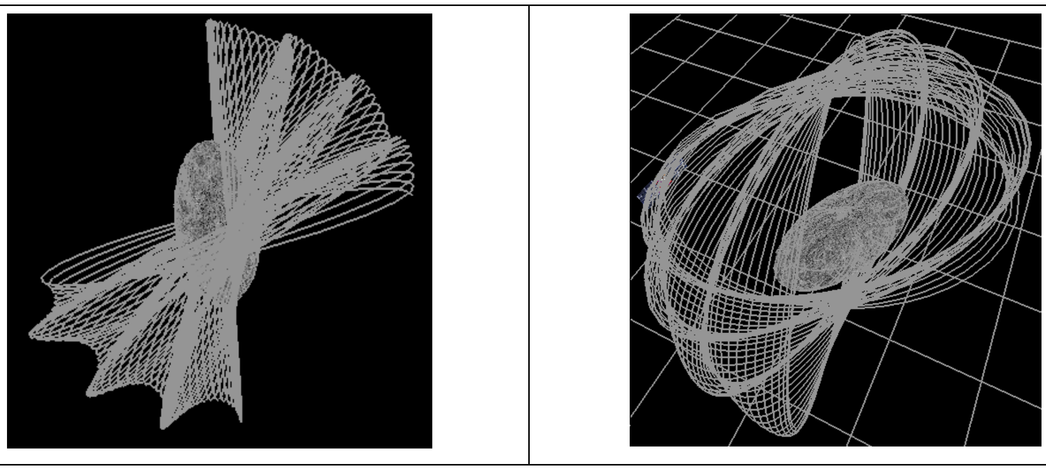


Figure 3-2: Orbit for RSE: terminator orbit, r=600m. Asteroid density=2.7t/m^3. Top View and side view. 50 days simulation

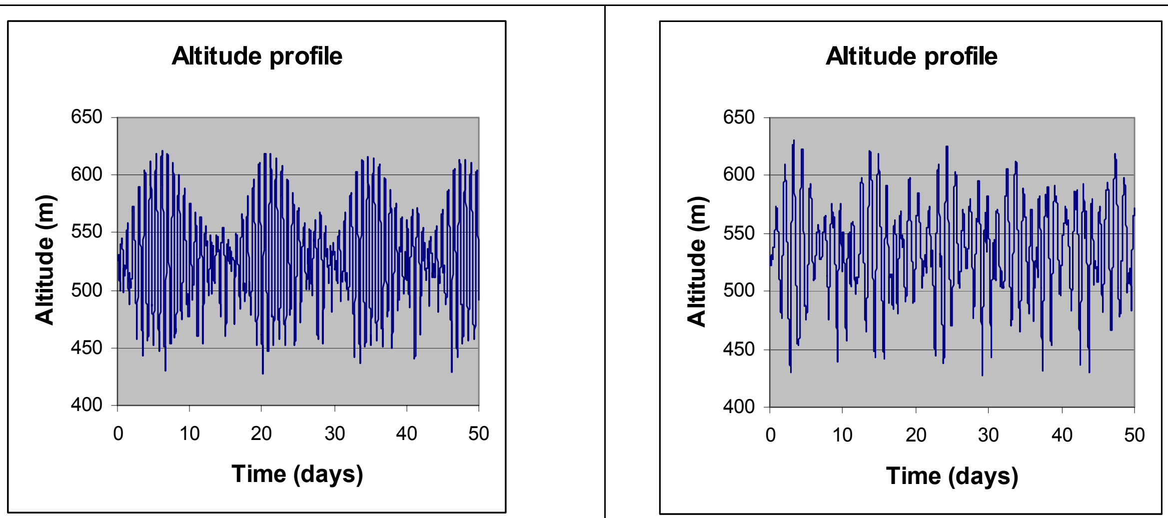


Figure 3-3: Altitude profiles for RSE orbit, r=600m. Asteroid density=2.7t/m^3 (left) and 1.3t/m^3 (right).

3.1.2 Orbit for the thermal measurements

The objective of the thermal measurements is to collect data for modelling the Yarkowsky effect: the whole surface of the asteroid has to be covered, at different Local Solar Times (LST). A further requirement is that the mapping has to be quick in order to complete the LST map at various heliocentric radii and in order to allow as much time as possible for the RSE phase. Quick mapping can only be achieved by orbiting the asteroid or by means of non Keplerian orbits. Polar circular orbits provide the best coverage of all latitudes. A single terminator orbit (such as the one used for RSE) is not enough for the necessary thermal measurements, because it is a polar orbit only if the spin axis of Apophis is normal to the orbital plane of Apophis and such an orbit only covers two LST (6am-6pm).

Rather than accept the instability and orbit the asteroid in a 500m polar orbit manoeuvring twice per orbit, a better, safer and more robust solution is to fly a non-Keplerian orbit. Continuous thrust is applied to the s/c in order to compensate the effects of SRP and the harmonic terms and to generate an acceleration towards the asteroid, artificially increasing the central force term of the orbit. The centripetal force requirement dominates, so the thrust will be essentially directed towards the asteroid. The desired altitude and period of the powered orbit determine the acceleration to be artificially provided: this can be translated into thrust/mass (T/m) and ∆V requirement (Figure 3-6).

The orbital period choice is based on the requirements on coverage time for the whole surface and on the number of crossing points at the equator (equispaced over 360º longitude). The baseline scenario asks for 13 crossing points and surface coverage in about 4 days: the solution is to orbit Apophis in 13:3 resonance with its period (3 Apophis days equal 3.81 Earth days). This gives an orbital period of 7.04 hours. Figure 3-7shows the resulting ground track.

The radius is chosen according to mapping resolution and required ∆V and thrust for flying the powered orbit: 1km radius meets all the requirements. The necessary T/m is then 59.4mN/tonne: this exceeds the natural gravity of Apophis by a factor 30 to 60 (with assumed densities of the asteroid). The orbital velocity is 24.7cm/s, high enough to escape from the SOI of Apophis in case of failure of the propulsion system (Figure 3-8); this guarantees safety of the mission.

3.2 MISSION TRANSFER

Two burn Hohmann transfers between Earth and asteroids are strongly dependent on the phasing of the Earth and NEO. This strategy can be improved by using a 3(or more)-impulse transfer. This effectively allows an intermediate transfer orbit to improve phasing. Table 3-1 identifies the locally optimal transfer opportunities in order to reach Apophis sufficiently ahead of 2017. A high thrust system has been assumed.

We have chosen to use Solar Electric Propulsion (SEP) to perform the deep space manoeuvres (DSM) and the rendezvous manoeuvres around the asteroid (the forced orbit in particular) and a simple chemical propulsion module to provide the impulse for Earth escape, as this is the most mass-efficient solution overall. The April 2013 short transfer in Table 3-1 has been selected as mission baseline. It is possible to trade off the arrival time at the asteroid against ∆V (Table 3-2). The extent of mass penalty for faster transfers is limited due to the high Isp of the SEP system. Soyuz has an adequate mass capability to meet the needs of APEX.

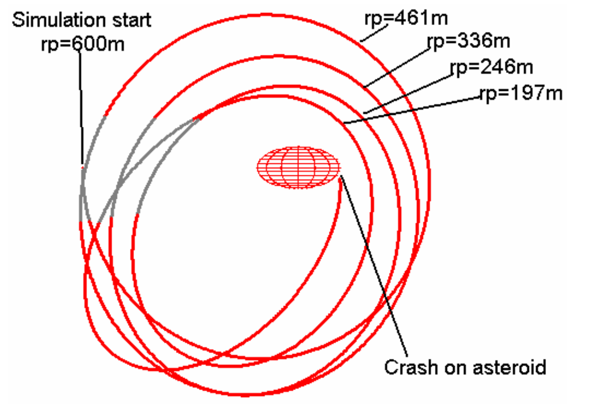


Figure 3-4: Instability on uncontrolled orbit. Ex: polar orbit; node LST 12am-12pm.

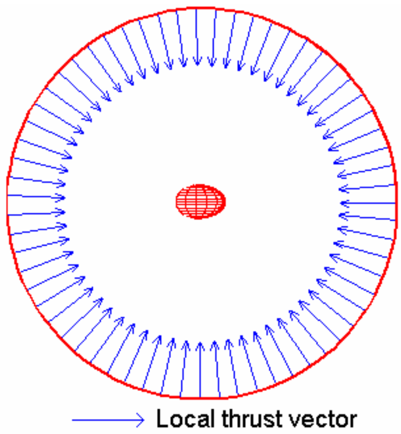


Figure 3-5: powered orbit

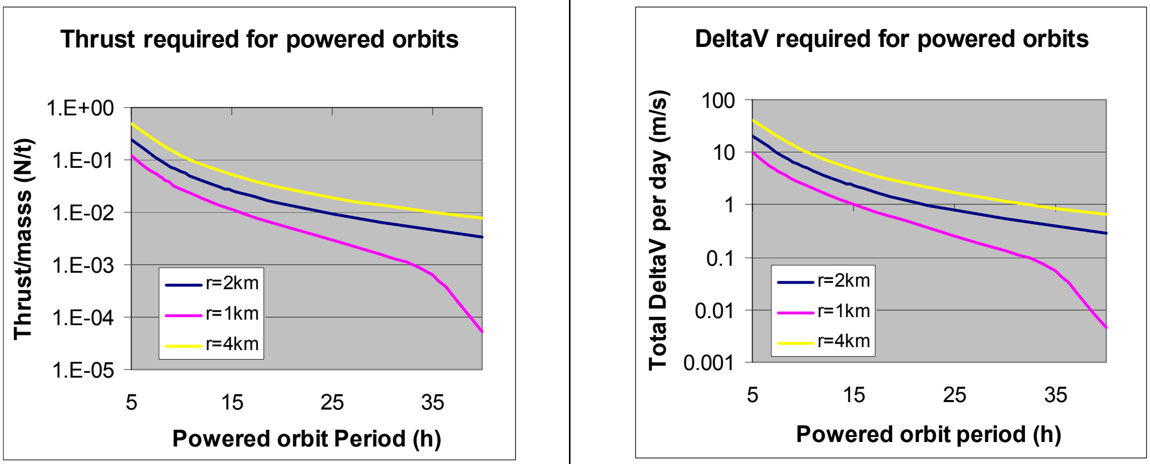


Figure 3-6: Thrust and DV per day required for powered orbit as function of orbital period, for different orbital radii

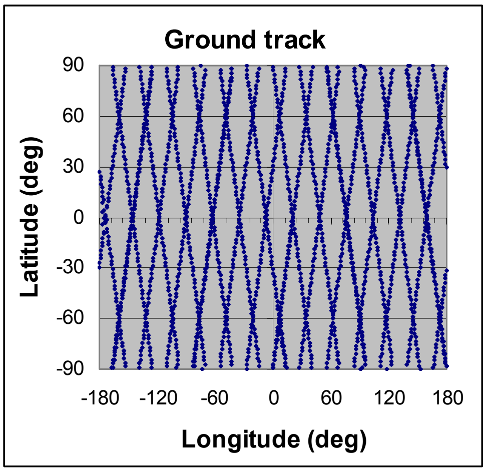


Figure 3-7: Ground track for thermal measurements: 13 equi-spaced crossing points at the equator

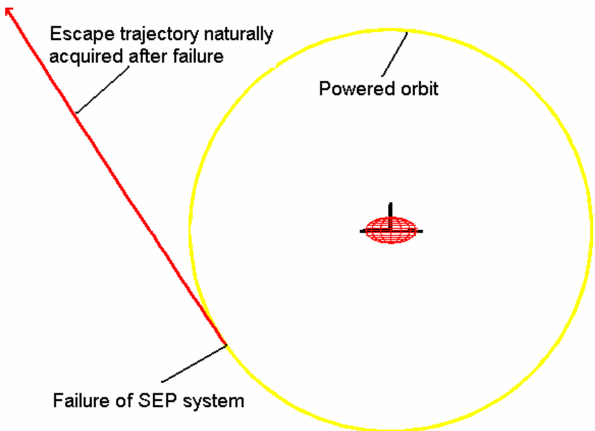


Figure 3-8: Mission safety in case of failure of SEP system during thermal measurement phase

3.3 MISSION OVERVIEW AND OPERATIONS

The proposed mission concept assumes a Soyuz Fregat launch from Kourou. At launch the spacecraft stack consists of a chemical propulsion stage and a science spacecraft. Soyuz Fregat injects the spacecraft stack into Geostationary Transfer Orbit (GTO), the chemical propulsion stage then performs an apogee raising sequence boosting the apogee radius to a sub lunar orbit before a final perigee burn provides the escape ΔV. Following escape the chemical propulsion module is jettisoned and the spacecraft continues the transfer to Apophis using the solar electric propulsion system. This system comprises one prime and one redundant flight proven PPS1350 Plasma thruster with a specific impulse of 1510s and a maximum thrust of 90mN. After visual acquisition of the target the approach trajectory is assessed and final course corrections are planned and performed.

A very efficient way to handle this (Figure 3-9) is to split the final manoeuvre of the mission into a series of smaller, thrust and coast arcs to incrementally approach the target. Prior to each arc, an orbit determination campaign is performed to provide the best estimate of the spacecraft position and velocity. The durations required for the tracking campaigns are likely to be a few days. During this approach sequence, the asteroid will be acquired visually by the spacecraft, at a range of several hundred thousand km. The precision in the target location improves after each approach arc. Each thrust arc can be adapted to compensate for any error in the target location in the previous arcs.

After some time in preliminary orbit, where improved estimates of the target gravity field are obtained, the S/C moves to operational orbit. The spacecraft is inserted into a terminator orbit at a safe altitude (at least 1.5km) for an initial 1 month science campaign. The objective of this phase is to measure the asteroid's physical properties sufficiently to allow the selection of the optimum obits for the mapping and radio science phases.

Science Phase

After completion of the initial assessment the detailed science phases begin, the objective of the first science phase is to provide global high resolution visual imaging and high resolution mass and gravity field determination. The global mapping is carried out from an altitude where the asteroid fills the narrow angle camera field of view (2.7km). The gravitational characterisation is achieved by orbiting around the poles of the asteroid at the lowest stable altitude (~500m), and tracking the spacecraft using the radio science experiment. This phase of the mission will take 18 days.

The radio science phase aims to determine the Apophis orbit parameters to a high level of accuracy, during this phase the spacecraft operates in a terminator orbit which maximises the stability. Wheel off-loading operations are required every 45 days. The thermal mapping phase uses a polar orbit at 1km altitude with a constant thrust applied to reduce the period to ~7 hours. An observation period of 40 days is required to provide measurement of each area on the surface at 20 local times, this observation sequence is repeated another 3 times to measure seasonal variations (see Figure 2-2).

The spacecraft will be controlled by a single mission operations centre throughout the mission. A nominal contact strategy of 8 hours per day to a single 35m ground station has been assumed for all mission phases. The spacecraft operations will be based on an onboard master schedule with onboard control procedures. The master schedule will be updated on a regular basis from the ground. The onboard software will ensure that in case of prolonged communications blackout the spacecraft will stay safe and operational until the communications link is re-established.

Launch date	Arrival date	Transfer time (days)	Vinf Earth (m/s)	Departure Dv (m/s)	launch incl. (deg)	DV dsm (m/s)	Vinf arrival (m/s)	**total DV from GTO (m/s)**
26/04/2012	25/01/2014	629	4269.31	1599.05	34.68	0.00	1123.16	**2722.41**
08/05/2012	15/03/2013	307	2930.24	1175.08	38.55	0.00	2031.40	**3206.48**
13/04/2013	23/11/2014	580	4778.95	1796.45	34.75	208.00	370.94	**2375.58**
21/04/2013	22/01/2014	271	4802.16	1805.89	33.33	19.00	721.28	**2546.97**
02/02/2015	24/05/2016	472	3269.07	1268.82	17.68	24.00	2248.05	**3541.54**

Table 3-1: Summary of high thrust opportunities to Apophis

T/m (mN/t)	Launch	Arrival	Vinf	inclination	**CP dv**	**SEP dv**
100	17/04/2013	09/03/2014	4158.793	37.592339	**1558.771**	**1255.598**
100	22/04/2013	18/01/2014	3336.0793	49.881779	**1288.472**	**2286.97**

Table 3-2: Example of trade-off between arrival time and DeltaV for SEP-rendezvous transfer

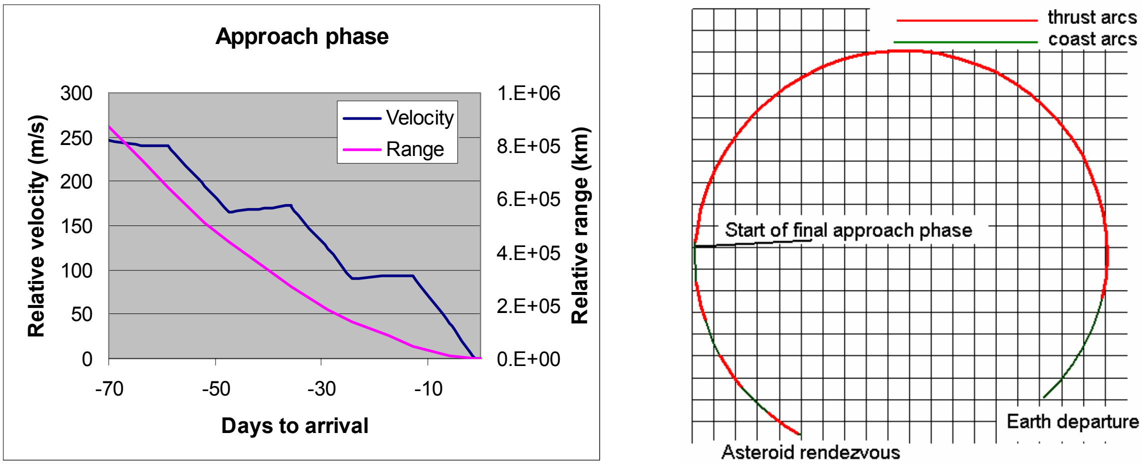


Figure 3-9: *left)* relative velocity and range between s/c and asteroid in the approach phase; *right)* Baseline transfer

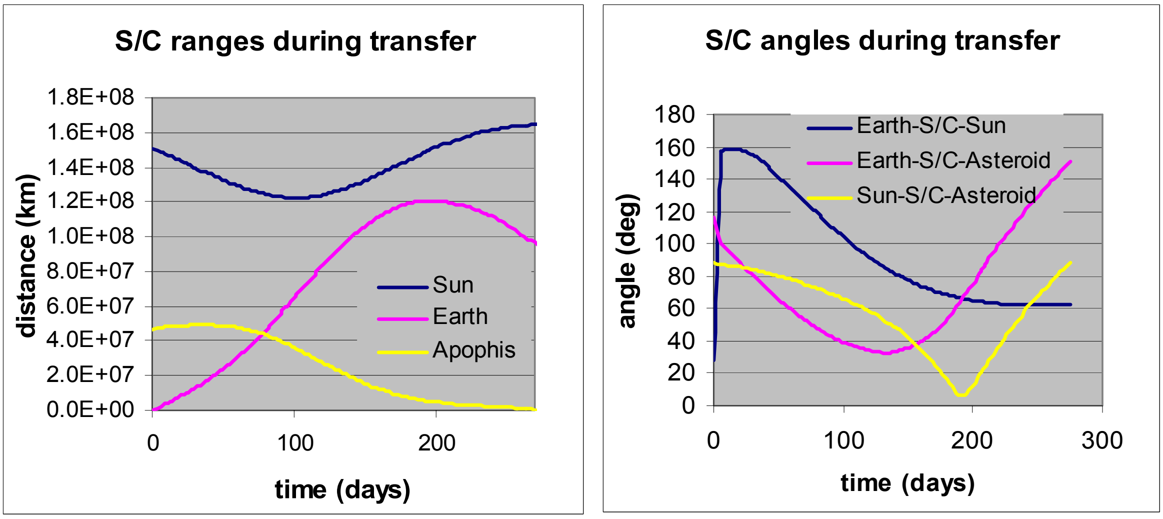


Figure 3-10: Ranges and angles of interest for the s/c during transfer

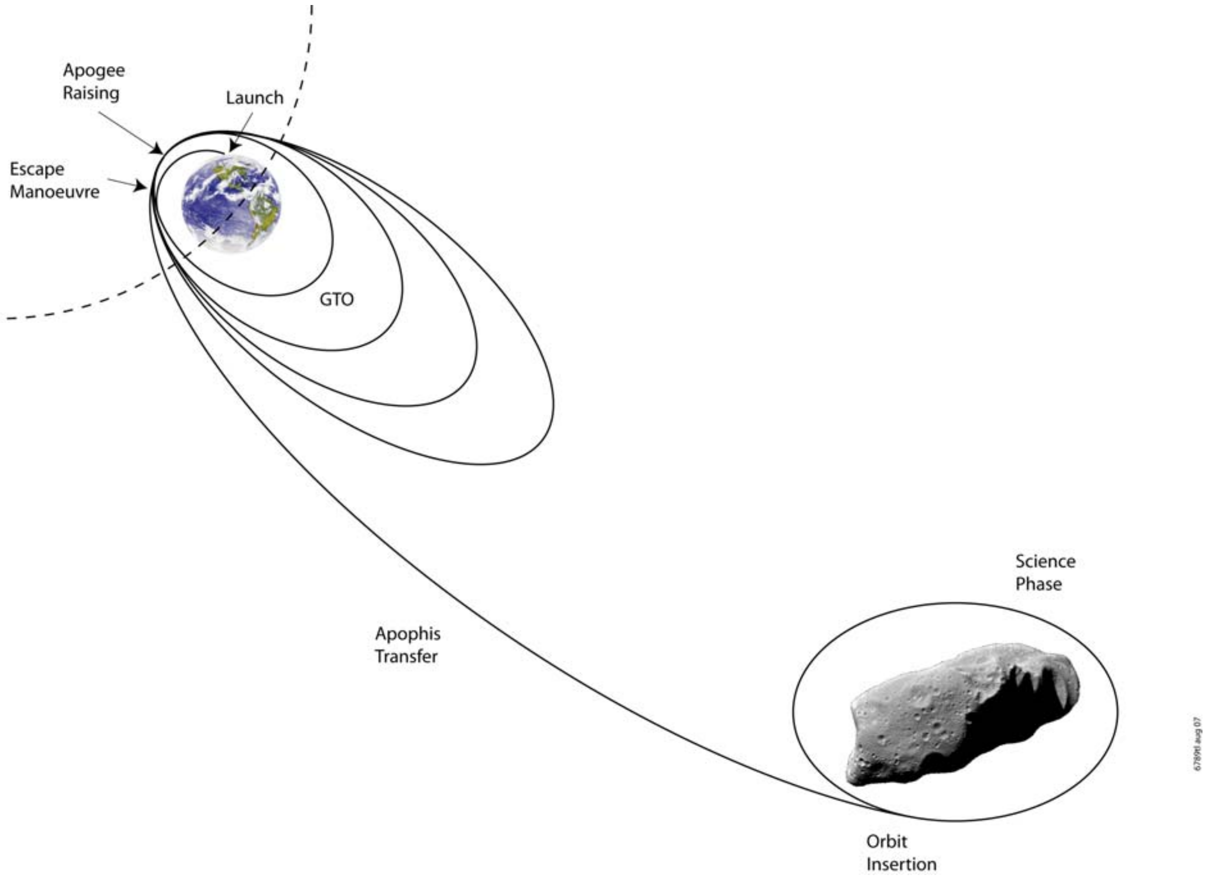


Figure 3-11: Mission Scenario

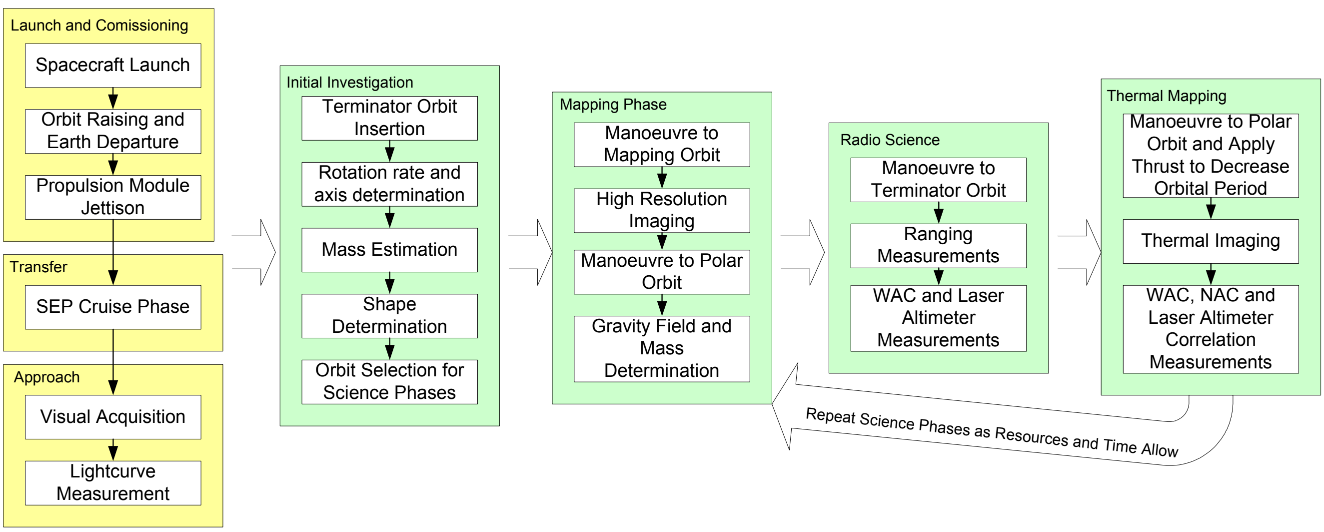


Figure 3-12: Mission Phases and Operations Flow

4 BASELINE MISSION DESIGN

4.1 SPACECRAFT ARCHITECTURE

The space segment to be launched includes two main modules: at the lower position attached to the Fregat upper stage is the chemical propulsion module, in the upper the main spacecraft. Figure 4-1 shows the spacecraft composite in launch configuration and accommodated in the Soyuz Fregat ST fairing. A robust spacecraft design is required to ensure the mission goals are fully met by April 2017, so the spacecraft and propulsion module design includes a high level of redundancy and uses equipments and payloads with flight heritage. For instance, the chemical propulsion module is a rebuild of the system developed for the LISA Pathfinder mission.

The configuration of the spacecraft is heavily driven by the physical and operational requirements of the subsystems and payload over the course of the mission. The overall spacecraft configuration is based on a cuboid comprising a central cylinder, an internal shelf, and side panels. The size of the structure itself is driven by the size of the Xenon tank for the central cylinder, and the required radiator area for the side panels. The resulting body configuration fits in a $1.3m^3$ cube with a 0.85m diameter central cylinder. An adapter ring interfaces the central cylinder to the propulsion module during launch and cruise prior to SEP ignition.

Propulsion System

The majority of the ΔV for the transfer and the operational orbit is provided by the solar electric propulsion system, the system is designed to provide a variable thrust throughout the transfer phase to maintain the thrust to mass ratio at 100mN per tonne. During the forced mapping orbits a thrust level of 59mN per tonne is required. These requirements result in thrust levels between 90mN and 38mN over the mission lifetime. This variable thrust requirement is met by using the PPS1350 thruster previously qualified on the SMART-1 mission which has a thrust range from 30-90mN and has been life tested for a total impulse in excess of the requirement for this mission. The Xenon and pressurant tanks are housed within the central cylinder. The PPS thrusters are mounted on a face supported by the central cylinder and close to the main axis of the spacecraft. The associated electronics, the SEP relay box and power unit will dissipate significant heat and are therefore located on a radiator face.

AOCS/RCS system

The AOCS system is standard for a 3-axis stabilised spacecraft comprising reaction wheels, inertial measurement sensors, star trackers, and sun sensors for initial acquisition and safe mode. Four (3+1 spare) momentum wheels are collocated on the bottom face of the spacecraft and arranged in a pyramidal configuration. The relatively large 70Nms reaction wheels have been selected to ensure the stable environment required by the RSE phase in which the thrusters remain inactive. The wheels will be gradually spun up during the RSE as they absorb the disturbance torques until they reach their 6000rpm limit after a minimum of 50 days, at which point the RCS thrusters will perform a momentum dump. The worst-case gravity gradient contribution during RSE is small at ~4% of the overall RSE disturbance torque of 1.78E-5Nm. When in non-RSE phases, the maximum possible disturbance torque of ~8E-5Nm is well within the 0.3 Nm max torque limit of the wheels, Figure 4-2 shows the result of the SRP analysis of the S/C. Although the selected wheels are of high performance, the manoeuvres required during the mission are relatively benign, so the power consumption is low.

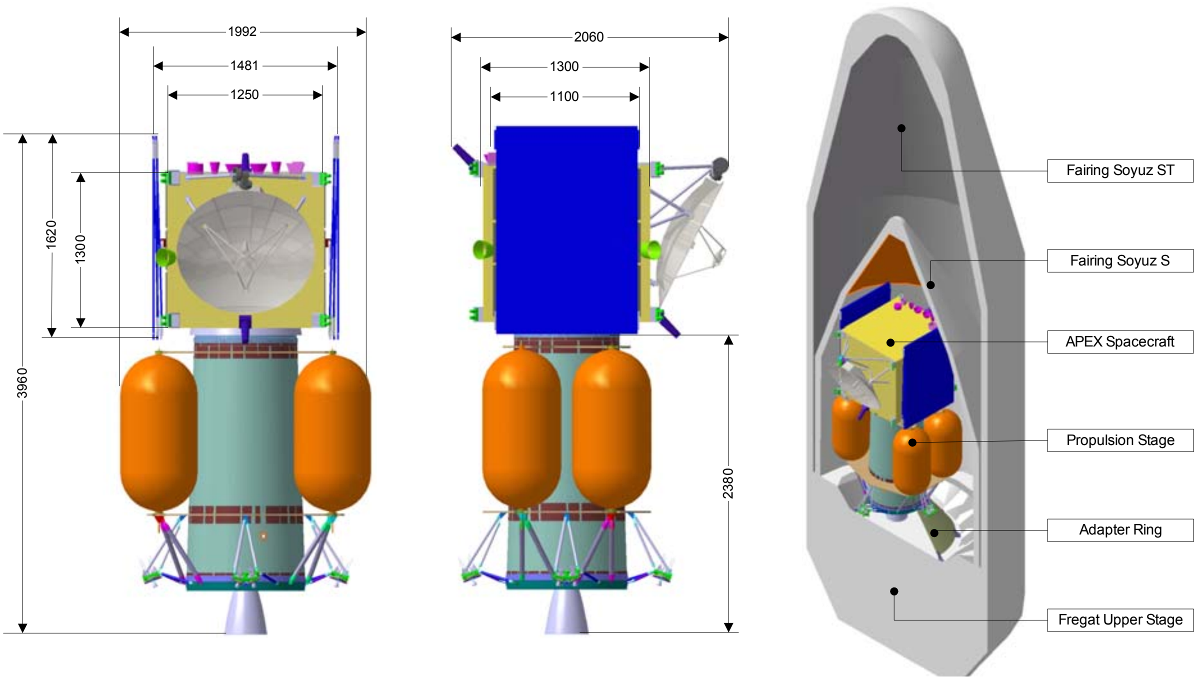


Figure 4-1: APEX Composite Configuration (left), Soyuz Launch Configuration: Orbiter, Propulsion Module and Two fairing Options S and ST (right)

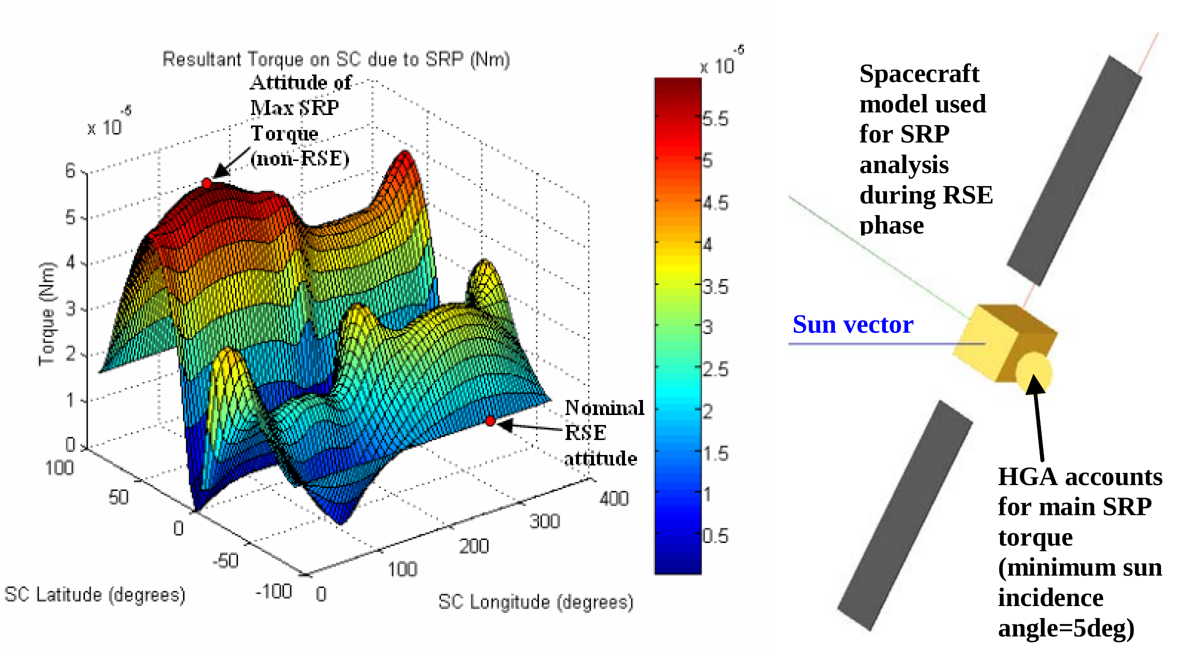


Figure 4-2: Solar Radiation Pressure Assessment for Wheel Sizing

The momentum capacity of the wheels has been sized specifically to avoid firing of thrusters during the RSE phase.

To complement the wheels, a small monopropellant RCS system is also included for attitude control manoeuvres, wheel-offloading and operation orbit changes around Apophis. 16 RCS thrusters (8+8 redundant) are located on each corner of the spacecraft and the hydrazine tank housed in the central cylinder. Two star trackers are located on one of the SA-mounting faces pointing 45 degrees from the array to avoid blinding from the array, sun and asteroid during the various phases of the mission. An accurate 7” star tracker has been selected to ensure compatibility with the 10” pointing requirement during the mapping campaigns. Finally, a sun sensor is used on the top and bottom faces of the spacecraft to help with the steering of the arrays.

Power and Thermal System

The electrical power is provided by two 6m^2 steerable solar arrays (SA) using triple junction GaAs cells. Each wing is divided into 4 panels to facilitate stowing of the arrays during the launch phase. The SAs are sized to fulfil the power requirements needed to operate the solar electric propulsion system during the transfer phase. The thermal control of the spacecraft uses two 1.1m^2 radiators located on the faces of the orbiter where the SAs are mounted to provide a guaranteed view to deep space. The Power Conditioning Distribution Unit (PCDU) and battery are both located on the internal shelf, with the PCDU benefiting from a direct access to a radiator face.

OBDH System

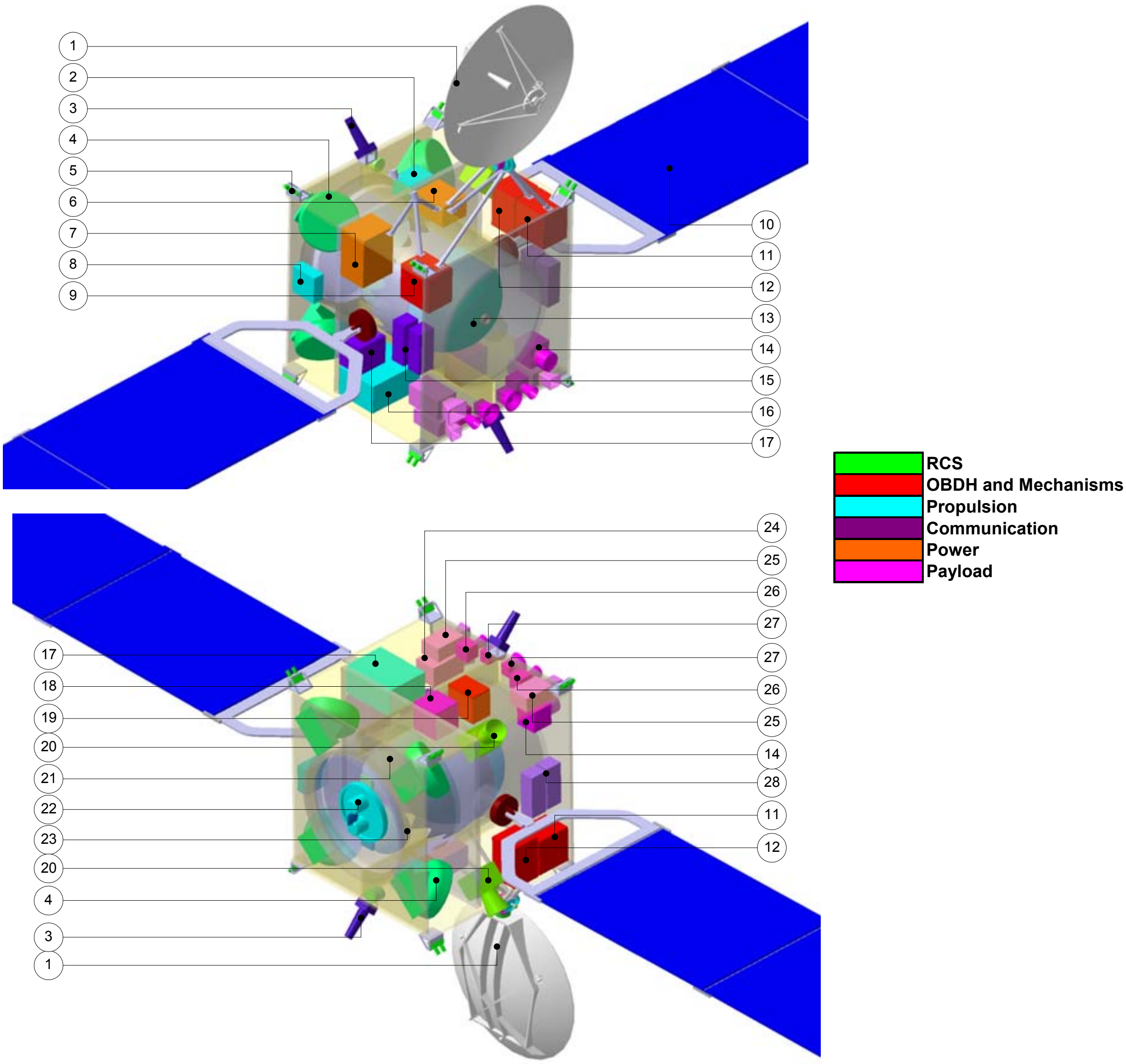
The On-Board Data Handling (OBDH) system includes a fully redundant computer and 20Gbits of mass memory to store the data generated during the 13 day high resolution mapping campaign. In all other phases of the mission the data collected can be transmitted to ground during the standard 8 hour daily contact period. The OBDH subsystems are located on a radiator face as are the Solar Array Drive Electronics (SADE) while the Pyro Release Mechanism is located on the vertical inside panel.

Communication System

The spacecraft is equipped with a dual band communications system. For communication with Earth the X-band system is used to receive telecommands and transmit spacecraft housekeeping and science data, the Ka-band system however is used for accurate ranging. The nominal 17kbps link will be via a steerable high gain antenna (HGA) mounted on a 2 degrees of freedom support on the top face of the orbiter. Two low gain antennas (LGA) located on opposite faces of the spacecraft will provide omni-directional coverage for the communication during the initial apogee raising, escape sequence and as a backup. The HGA is baselined to be deployed after propulsion module separation and used during the rest of the transfer and the operational orbits at the asteroid. The transponder and associated electronics are located on both radiator faces of the spacecraft to ensure a good thermal control.

Payload system

To meet all the mission operational requirements, the payload is located on an optical bench on the face opposite the main propulsion. It is composed of two wide angle cameras, two narrow field cameras, two thermal IR spectrometers, a near IR spectrometer, and a LIDAR. In addition, an accelerometer is located on the vertical inside panel on the bottom face of the orbiter.



Part	System	Part	System
1	High Gain Antenna (HGA)	15	Ka-Band Transponder & TWTA
2	Xenon Flow Control Unit (FCU)	16	Propulsion Power Unit (PPU)
3	Low Gain Antenna (LGA)	17	Ka-Band Translator
4	Momentum Wheel	18	Accelerometer
5	Reaction Control System Thruster	19	Pyro Release Mechanisms
6	Battery	20	Star Tracker
7	Power Conditioning Unit (PCDU)	21	Pressurant Tank
8	SEP Relay Box	22	PPS Thruster
9	RTU	23	Hydrazine Tank
10	Solar Panel	24	NIR Spectrometer
11	Solar Array Drive Electronics (SADE)	25	Thermal IR Radiometer
12	Command and Data Management Unit (CDMU)	26	Dawn Camera
13	Xenon Tank	27	Wide Angle Camera
14	LIDAR	28	X-Band Transponder & TWTA

Figure 4-3 – APEX Orbiter Subsystems Definition

4.2 SYSTEM BUDGETS

Subsystem	Basic Mass [kg]	Margin [kg]	Total Mass [kg]	TRL
Payload	28.4	3.1	31.5	9
Power	88.2	7.9	96.1	9
Electric Propulsion	93.5	11.7	105.2	9
Chemical Propulsion	14.5	1.0	15.5	9
AOCS	52.4	2.6	55.1	9
Data Handling	29.0	1.5	30.5	9
Communications	59.0	7.7	66.7	9
Harness	27.1	5.4	32.5	5
Structure	88.0	17.6	105.6	5
Thermal Control	8.0	1.4	9.4	9
Spacecraft Dry Mass	**488.2**	**59.8**	**548.0**	
Spacecraft Dry Mass incl. 20% System Margin			**657.6**	
Propellant			207.1	9
Spacecraft Wet Mass			**864.7**	
Chemical Propulsion System	97.4	4.9	102.3	9
Thermal control system	9.3	0.5	9.8	9
Structure	75.2	3.8	79.0	6
Harness	7.8	0.4	8.2	6
Separation system	8.5	0.4	8.9	6
Propulsion Module Dry Mass	**198.2**	**9.9**	**208.1**	
Propulsion Module Dry Mass incl. 20% System Margin			**249.7**	
Propellant			1107.9	9
Propulsion Module Wet Mass			**1357.6**	
Stack Mass			**2222.3**	
Launch Capability			**3023.0**	
Launch Margin			**800.7**	

Table 4-1: System Mass Budget

	Chemical Propulsion	Solar Electric Propulsion
Deterministic ΔV	1401.0	2287.0
Dispersion correction	50.0	
Launch window	23.0	
Inclination	167.0	
Loss at departure	280.3	
Remote sensing		15.0
Forced orbits		1030.0
Total	**1921.3**	**3332.0**
Margin	96.1	133.3
Total incl. margin	**2017.3**	**3465.2**

Table 4-2: Mission ΔV Budget

5% margin included on chemical delta-v, 4% margin on solar electric delta-v

Subsystem	Launch mode	Sun Acquisition Phase 1AU	Transfer 0.82-1.1 AU	Composite Safe Mode	Transfer (firing chemical stage) 0.82-1.1 AU	Transfer firing SEP 0.82-1.1 AU	Radio Science Mode 0.75-1.1 AU	Thermal Mapping Mode 0.75-1.1 AU	Mapping Eclipse Mode 0.75-1.1 AU	Mode/ Manoeuvring/Wheel offloading
Spacecraft										
Payload	0	0	1	1	1	1	38	55	39	23
Power	22	149	22	22	22	22	75	165	22	112
Electric Propulsion	0	0	0	0	0	2091	0	1272	1272	0
Chemical Propulsion	0	0	0	0	0	0	0	0	0	111
AOCS	0	33	33	33	33	33	33	33	33	33
Data Handling	47	47	47	47	47	47	47	47	47	47
Communications	74	74	115	74	115	115	188	188	115	147
Thermal Control	36	55	106	116	96	36	36	36	24	36
Spacecraft Total	179	358	324	293	314	2345	417	1796	1551	510
Propulsion Module										
Chemical Propulsion	2.1	23.1	2.1	23.1	23.1	0	0	0	0	0
Thermal Control	0	168	168	168	168	0	0	0	0	0
Propulsion Module Total	2.1	191.1	170.1	191.1	191.1	0	0	0	0	0
TOTAL POWER [W]	181	549	494	484	505	2345	417	1796	1551	510

Table 4-3: Power Budgets for Main Mission Modes

The power demand equates to an array area of ~10m^2. A 20% area margin is applied therefore the arrays are 12m^2.

Mission Phase	Science data rate (kbps)	Spacecraft housekeeping (kbps)	Total collection rate (kbps)
Asteroid Approach	0.261	1	1.261
Initial Investigation	0.54	1	1.54
Global Mapping	15.4	1	16.4
Radio Science	0	1	1
Thermal Mapping	4.077	1	5.077

Table 4-4: Data Collection Rates for Each Science Phase

X-band Downlink Antenna	Range [AU]	Margin [dB]	Data Rate [kbps]
1m HGA	2.07	3	9.7
LGA boresight	2.07	3	0.01
LGA 50 deg off-pointing	2.07	3	0.005
X-band Uplink Antenna	**Range [AU]**	**Margin [dB]**	**Data Rate [kbps]**
1m HGA	2.07	6	49.1
LGA boresight	2.07	6	0.04
LGA 50 deg off-pointing	2.07	6	0.016

Table 4-5: Data Transmission and Reception Rates

5 PROGRAMMATICS AND COSTING

Schedule Assessment

The key programmatic driver for the APEX mission is the short timescale available for the completion of the scientific observations, there must be sufficient data available to inform a decision on a deflection mission by 2017. A first assessment of the scientific measurements required to meet the mission requirements has concluded that a minimum of three years of observation are required. The mission design must therefore enable spacecraft arrival at Apophis by early 2014.

The design presented in the previous chapters is based on a launch window between 16th and 28th April 2013 with an asteroid arrival in January 2014. The schedule shown opposite is based on the Mars Express programme which completed the implementation activity and launch campaign in 4.5 years by using only existing equipments from current suppliers. With this development timescale and a launch date in January 2013 the decision to implement such a mission must be taken urgently.

As a result of the restricted schedule duration the equipment selected for the spacecraft and scientific measurement is all either flight qualified or under development for existing missions planned for launch by 2013. The rebuild equipment must be purchased from the existing suppliers early in the programme to mitigate obsolescence.

Mission Costing

The price estimate provided for the mission is based on a detailed product tree for the spacecraft and experience in launch and operations costing. The table opposite shows an initial assessment of the mission price including, launch, payloads, spacecraft, operations and ground segment. This is an enveloping cost based on flight standard equipment, no cost minimisation activity has been carried out. It is assumed that: the mission is launched on Soyuz from Kourou; operations and ground segment costs are ~3M$ per year; all other costs were based on previous missions with similar elements and the Rough Order of Magnitude (ROM) price is based on mid 2007 Economic Conditions. The ROM price is presented for information only; it is not committing and not open to acceptance or negotiation.

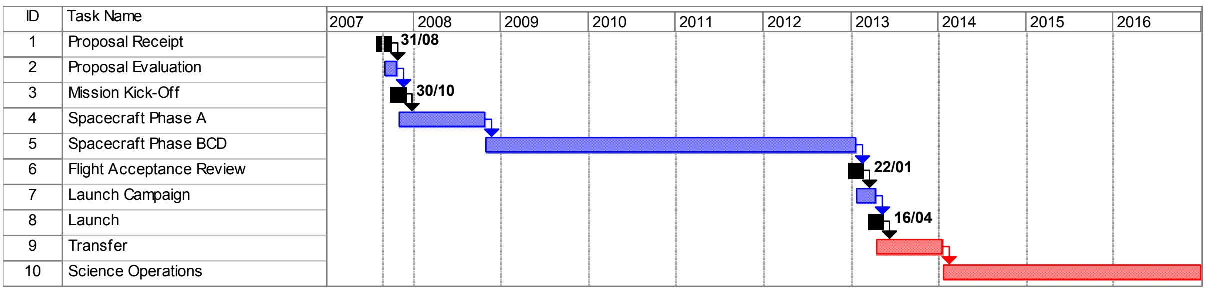


Figure 5-1: APEX Mission Overall Development Schedule

APEX - Apophis Explorer	
Ec. Cond. Mid-2007	Aug-07
Currency k$	ROM Price
Astrium Ltd	Estimate
PHASE A	4,000
PHASE B	22,970
SPACECRAFT PHASE C/D	
Phase C/D Flight Equipment	
AOCS	13,200
Chemical Propulsion	6,600
Electric Propulsion	26,400
Electrical Power	22,440
Harness	1,320
Communcations	19,800
Data Handling and Software	19,800
Structure	5,940
Mechanisms	1,320
Thermal Control	1,980
GSE	19,800
Assembly, Integration and Test / Verification	26,400
System Level Project Office	
Management & Control	15,840
Product Assurance	7,920
Engineering	40,920
Contingency	64,200
Total Phase C/D	**293,830**
PAYLOADS PHASE C/D	
Dawn Camera	10,560
NIR IR Spectrometer	7,920
Thermal IR Mapper	9,240
Wide Angle Camera	6,600
Laser Altimeter	5,280
Accelerometer	2,640
Contingency	15,000
Total Phase C/D	**57,290**
PROPULSION MODULE PHASE C/D	
Phase C/D Flight Equipment	10,560
GSE	1,980
Assembly, Integration and Test / Verification	5,280
System Level Project Office	5,680
Contingency	2,400
Total Phase C/D	**25,900**
Total Phase C/D	**376,990**
Phase A, B, C/D	**403,960**
Phase E	
Launch (Soyuz), 2013	**60,000**
Ground Segment	**6,600**
LEOP, including Industrial Support	**13,200**
Operations, 4 years	**10,000**
Total Launch and Phase E	**89,800**
Total Mission	**493,760**

Table 5-1 APEX Cost table

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