Apophis Mission Design Competition

MISSION PROPOSAL:

ORACLE

***OR**bit determination of **A**pophis for*
***CL**ose encounters with **E**arth*

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MONASH University
Engineering

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NOTE: Any awarded prize money should be sent to student team members Dilani Kahawala and Hemant Chaurasia.

EXECUTIVE SUMMARY

This proposal presents an outline of the *Oracle* mission, a pioneering endeavour to precisely and efficiently determine the orbit of a near-Earth asteroid and assess its potential for a future impact with Earth. The target of this mission is Apophis, a 270-metre-wide asteroid with an orbit that will pass uncomfortably close to Earth in April 2029. There is a danger that if Apophis passes through a particular region of space called a "keyhole" during this close encounter, it will impact the Earth in 2036.

In providing a solution for the problem of determining Apophis' orbit, *Oracle* follows a design philosophy characterised by simplicity, cost-efficiency and singularity of purpose. The usual suite of scientific instruments is replaced with just two: a LIDAR rangefinder and an AMIE imager. With a total predicted mission cost of $226 million, few competing solutions could claim to be more cost-efficient.

The *Oracle* mission design calls for a 651 kg spacecraft to be launched atop a Delta II 7925 booster on a direct transfer orbit to Apophis. After a 10-month coasting phase to Apophis, *Oracle* will fire its main engine and come to a hover position 1 km from the surface of Apophis. From here, *Oracle* will employ a LIDAR laser rangefinder alongside its Attitude Determination and Control System, to measure the position of Apophis relative to the spacecraft to within 1 metre. This data will be transmitted back to Earth, while the spacecraft itself is simultaneously tracked by ground stations. This tracking shall follow the well-established method of sending tracking signals to the spacecraft which are returned immediately to the ground station. From observing the time delay and frequency change of the returned signal, very accurate radar ranging and radar Doppler measurements can be made of the spacecraft. Combining this data with the known position of Apophis relative to the spacecraft and past astronomical measurements of the asteroid, a Kalman filtering process can be used to progressively refine the determination of Apophis' orbit. Continuing this process for 14 months is predicted to refine Apophis' orbit to such an extent that the threat of an Earth impact can be known to 10% confidence – a sufficiently high probability to motivate immediate preparations for a deflection mission.

Oracle will also carry an AMIE Imager instrument onboard to analyse the size, geometry and surface composition of Apophis. At the conclusion of *Oracle*'s primary mission, an extended operations phase may be undertaken which will involve landing on the surface of Apophis by freefall. Should this be successful, Apophis' orbit determination may be refined even further over time, providing invaluable insights into the factors which influence the orbits of near-Earth asteroids.

A catastrophic asteroid impact on Earth is not a question of if, but when – and like its namesake, *Oracle* will help humanity peer into the future to find an answer.

GLOSSARY OF TERMS

ACS	Attitude Control System
AGS	Addition Ground Stations
AIU	Attitude Interface Unit
AKM	Apogee Kick Motor
AMIE	Advanced Moon micro-Imager Experiment
CDH	Command & Data Handling
CER	Cost Estimating Relationship
EPS	Electrical Power System
ET	Earth Terminals
FGS	First Ground Station
GNC	Guidance, Navigation and Control
GSE	Ground Support Equipment
GSOS	Ground System Operations & Support
HGA	High-Gain Antenna
IA&T	Integration, Assembly & Test
IMU	Inertial Measurement Unit
LGA	Low-Gain Antenna
LIDAR	Light Detection and Ranging
LOOS	Launch and Orbital Operations Support
MGA	Medium-Gain Antenna
NEAR	Near Earth Asteroid Rendezvous
Ops/Maint.	Operations and Maintenance
PL	Payload
RCS	Reaction Control System
RDT&E	Research, Development, Testing & Evaluation
RTG	Radioisotope Thermoelectric Generator
S/C	Spacecraft
SMAD	Wertz & Larson, "Space Mission Analysis and Design" [8]
SW	Software
TCS	Thermal Control System
TFU	Theoretical First Unit
TT&C/DH	Telemetry, Tracking & Command / Data Handling

1.0 – MISSION OVERVIEW

1.1 – INTRODUCTION AND CONTEXT

This proposal presents an outline of the *Oracle* mission, which aims to precisely and efficiently determine the orbit of the near-Earth asteroid Apophis and thereby assess its potential for a future impact with Earth.

The asteroid Apophis was discovered in June 2004 and is estimated to be 270 metres in diameter (see Tables 1 and 2 below). Soon after its discovery, Apophis was found to have an orbit which passes dangerously close to the Earth in April 2029 – in fact, for a short time the calculated probability of impact with the Earth rose to be as high as 1 in 37. Subsequent observations were able to refine the estimate of Apophis' orbit, yielding a much lower impact probability. While an impact with the Earth in April 2029 has now been essentially ruled out, there remains a slim but non-zero probability (~0.0022% [1]) that Apophis will pass through a region of space called a "keyhole" during its 2029 close encounter – and if Apophis passes through this keyhole, it will guarantee an impact with the Earth on April 13, 2036. According to NASA estimates [1], such an impact would be equivalent to over 400 megatons of TNT, or roughly twice as powerful as the eruption of Krakatoa in 1883.

TABLE 1 – PHYSICAL ATTRIBUTES OF APOPHIS [2]

Parameter	Value
Orbit type	Aten (Earth-crossing)
Albedo	0.33
Diameter (m)	270
Rotation period (hr)	30.5376
Mass (kg)	2×10^{10}

TABLE 2 – ORBITAL CHARACTERISTICS OF APOPHIS [3]

Orbital Element	Value	1-sigma Variation
Semi-major axis (AU)	0.922281	2.369e-08
Eccentricity	0.191074	7.407e-08
Inclination (deg)	3.331	2.021e-06
Ascending node (deg)	204.457	0.0001068
Argument of the perihelion (deg)	126.395	0.0001059
M (deg)	169.914	3.928e-05

The *Oracle* mission derives its motivation from this lingering uncertainty surrounding Apophis' potential to impact the Earth in 2036. While the probability of an Earth impact in 2036 is small, it is considered significant enough to warrant a dedicated space mission for the purpose of more precisely defining Apophis' orbit and characterising the threat it poses to the Earth. If this threat is found to be more serious than previously thought, the findings of *Oracle* will be invaluable in planning and implementing any emergency mission to deflect the asteroid.

1.2 – CORE MISSION OBJECTIVES

The *Oracle* mission has been designed to satisfy core requirements as specified by The Planetary Society, which are listed in Table 3 below. If Apophis is in fact on a course to impact the Earth in 2036, the requirements listed in Table 3 will ensure that *Oracle* is able to identify the threat accurately enough and soon enough to enable a successful deflection mission.

TABLE 3 – CORE MISSION REQUIREMENTS

REQUIREMENT	SPECIFICATION
Functional	
• Tracking Accuracy	Must reduce the long dimension of Apophis' 3σ error ellipse (during its 2029 close encounter with Earth) to 14 km
Constraints	
• Schedule	Must complete primary mission by 2017 at the latest

In translating these core requirements into a complete mission design, *Oracle* follows a design philosophy characterised by simplicity, cost-efficiency and singularity of purpose. While the current impact probability of Apophis (0.0022%) is considered significant enough to justify the conduct of *Oracle*, the threat of impact is not considered severe enough to justify a very large expense on this mission. Thus, rather than design a large and costly mission involving multiple scientific instruments and associated infrastructure, *Oracle* has been designed as a streamlined and focussed solution to the specific problem of determining Apophis' orbit. This, *Oracle*'s primary objective, is accomplished with less mass, less power, fewer instruments and less monetary cost than alternative mission concepts.

In the unlikely event that Apophis is found to be on an impact trajectory with the Earth, *Oracle* will be the best opportunity to characterise the geometry and surface composition of Apophis. Such information, although basic and easy to attain, will no doubt prove invaluable in the planning of any subsequent deflection mission to the asteroid. As such, this has been accommodated as a secondary objective in *Oracle*'s mission design.

1.3 – MISSION PROFILE ALTERNATIVES AND SELECTION

Three major mission profiles were considered as solutions to the design problem. These were:

1. **Orbiter with lander:** An orbiter spacecraft sent to Apophis carrying one or more lander units, which would land on the surface of Apophis and act as radio beacons or otherwise assist in tracking operations.

2. **Combined orbiter/lander:** A spacecraft sent to Apophis which would itself land on the surface of Apophis and communicate with Earth as a means of tracking.

3. **Orbiter only:** A spacecraft sent to Apophis which would remain orbiting or hovering at a safe distance from Apophis' surface, and conduct tracking operations from this position.

In choosing among these three candidate mission architectures, core objectives and design priorities outlined in Section 1.2 were followed closely. Landing on the surface of the asteroid brings about a large number of unknowns, as very little is known about the precise surface environment of Apophis. The gravitational field around Apophis, which is almost certainly non-spherical, will also be increasingly irregular towards the surface and will thus cause complications in designing any landing mission. In terms of tracking Apophis through the spacecraft, such a task can be accomplished quite easily without landing on the surface at all (using a rangefinder instrument and standard radar ranging and range-rate tracking of the spacecraft itself).

Owing to Apophis' low mass, its gravitational field is extremely weak. The thrust required to hover in a position 1 km above the surface of Apophis is thus sufficiently miniscule that a spacecraft could quite feasibly maintain such a position for the entire duration of the primary science phase at Apophis, without incurring much cost in propellant.

Landing on Apophis would present unique scientific opportunities for characterising its physical attributes – however, this is not the primary mission of *Oracle* as specified by the Planetary Society. Designing and operating lander units, or a landing function for the whole spacecraft, would add a significant overall expense to the mission and unnecessarily complicate mission operations. All critical mission objectives as outlined in Section 1.2 can be achieved through the use of an orbiter-only mission profile.

Thus, following *Oracle*'s philosophy of simplicity, cost-efficiency and singularity of purpose, an **orbiter-only** profile was chosen for the *Oracle* mission.

1.4 – OVERVIEW OF CHOSEN MISSION PROFILE AND TRACKING METHOD

The *Oracle* mission will involve launching a small spacecraft on a Delta II 7925 launch vehicle into a direct transfer orbit to Apophis (see Fig. 1). This trajectory and the associated launch date have been calculated by JAQAR's Swing-by Calculator, an orbit optimisation program. The trajectory and launch date presented here represent the optimum solution, maximising the allowable spacecraft mass while still allowing for mission completion by 2017.

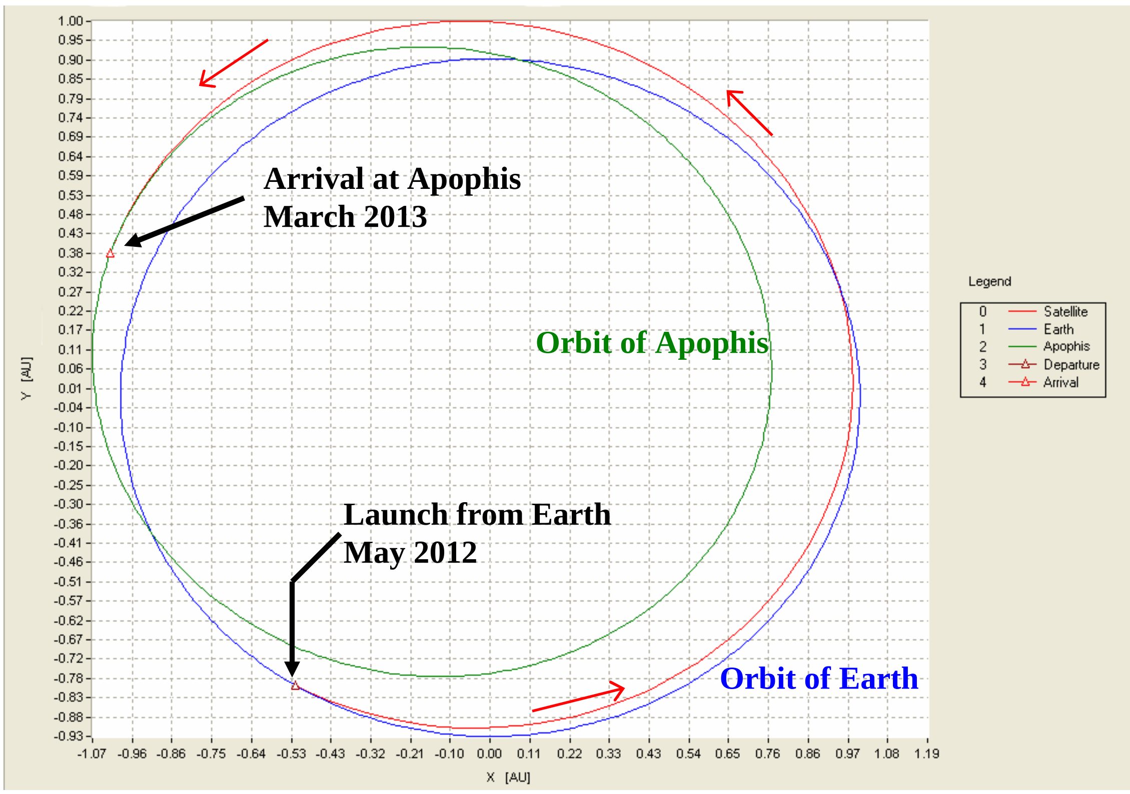

FIGURE 1 – Planned mission trajectory (in red) between Earth and Apophis

Upon arrival, an onboard propulsion system will slow *Oracle* to a state of rest with respect to Apophis. The spacecraft will then "hover" in a position at a fixed distance from the surface (1 km), using a very small amount of thrust to counteract the miniscule gravitational pull of Apophis.

While in this hover position, *Oracle* will use its onboard attitude determination systems (star cameras and sun sensors) combined with a LIDAR rangefinder instrument to measure the precise position of Apophis relative to the spacecraft (to an accuracy within 1 metre). This measurement will be repeated and recorded at regular intervals over a period of approximately 14 months (see Section 2.1 for a justification of this timeframe). As a secondary objective, images of Apophis will be recorded using the onboard AMIE imager and relayed to Earth at regular intervals, for the purposes of characterising Apophis' size, geometry and surface composition.

With regards to tracking operations, ground stations on Earth will regularly send tracking signals to *Oracle*, which will be returned to Earth as soon as they are received. By observing the time delay between sending and receiving the tracking signal, computers at the ground station can very accurately determine the range to *Oracle* (to within 1 metre), and through observing the Doppler shift of the returned signal, they can also determine the range-rate (to within 1 mm/s).

Combining these range and range-rate measurements of *Oracle* with the relative position of Apophis (from data returned by the rangefinder instrument and ADCS), very accurate range and range-rate measurements of Apophis can be derived. As *Oracle*'s mission at Apophis progresses, a growing set of such measurements will accumulate. Through a Kalman filtering process of these measurements, the orbit of Apophis can be ultimately determined to a very high accuracy, fulfilling *Oracle*'s primary mission.

At the conclusion of *Oracle*'s primary mission, an optional extended phase of operations may be undertaken to further refine the accuracy of Apophis' orbit determination, in an effort to better understand influences on the orbits of near-Earth asteroids (including phenomena such as the Yarkovsky effect). This extended phase will include the option of attempting a landing on Apophis via a simple freefall to the surface. In Apophis' weak gravitational field, such a freefall will result in very minimal impact to the spacecraft – specifically, an impact velocity of only 4 mm/s after a 26-minute freefall from the nominal hovering distance of 1 km. This will enable close-up images of Apophis to be taken, potentially bringing about a better understanding of its surface and composition. After landing, the spacecraft may continue to operate until it is no longer functioning, as any additional tracking data will always be useful in refining predictions of Apophis' future orbit.

2.0 – OPERATIONS OVERVIEW

2.1 – TOP-LEVEL MISSION PARAMETERS AND BUDGETS

Top level mission parameters and budgets are the major factors that constrain *Oracle*'s overall and subsystem design. The main budgets are Financial, Mass, Power and Schedule. Other top level mission parameters are Launch Vehicle and instruments to achieve mission objectives.

2.1.1 – Mission Duration

We have determined that to sufficiently constrain the orbit of Apophis, 14 months of regularly-spaced orbital measurements will be more than sufficient to achieve the core requirements set out in Section 1.2. This follows analysis by Chesley [4] of the effect of a 365-day transponder-based tracking mission of Apophis, which is predicted to reduce the error ellipse dimensions to a much greater degree than is actually required to complete this mission (see Fig. 2 below). The flight time to reach Apophis from launch is 10 months. The mission duration has been designed with a contingency of at least 2 months (i.e. 12 months of orbital measurements are predicted to be sufficient), to facilitate any further collection of orbital or image data.

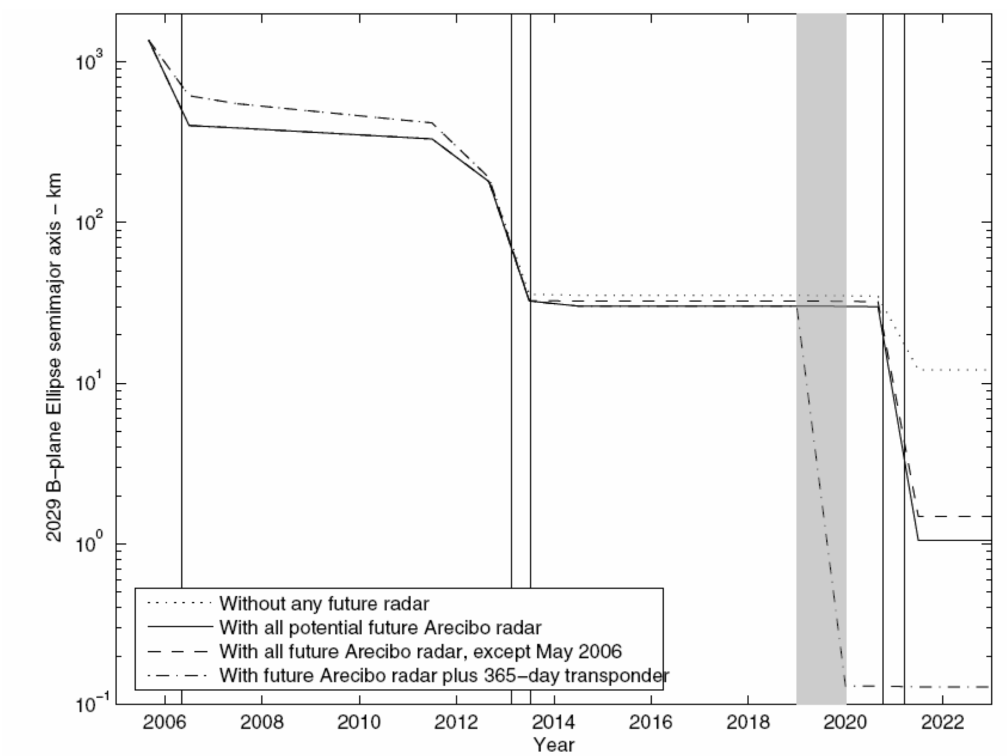

FIGURE 2 – Analysis by Chesley [4] illustrating the effect of a 365-day transponder-based tracking mission on Apophis' 2029 error ellipse dimensions over time.

2.1.2 – Financial Budget

Oracle is a low cost mission in accordance with the NASA Discovery Program objectives [5]. The low cost is achieved through simplicity of the spacecraft design, low mass and by using less expensive ground station resources. While there is no externally imposed cost cap for this mission, we have decided to set $300 million (FY2007) as an upper limit – a cap which is well within the bounds of the NASA Discovery Program and easily accommodated by our design (refer to Section 3).

2.1.3 – ΔV Budget

Based on simulations of *Oracle*'s transfer orbit carried out using JAQAR's Swing-by Calculator, the total ΔV requirement has been calculated to be approximately 1520 m/s. Adding a 30 m/s ΔV margin for trajectory correction manoeuvres and a further 38 m/s for the 14-month hover at Apophis, this yields a total ΔV budget of **1588 m/s**, as presented in Table 4 below.

TABLE 4 – ΔV BUDGET SUMMARY

Component	ΔV (m/s)
Main burn on arrival at Apophis	1520
Trajectory correction manoeuvres	30
14-month hover at Apophis	38
TOTAL	**1588**

2.1.4 – Mass Budget

Oracle's chosen launch vehicle (Delta II 7925) has a launch capability of approximately **937 kg** (boosted mass) on a direct transfer orbit to Apophis – this defines the upper limit of *Oracle*'s mass budget. This boosted mass capability is a conservative estimate derived from mission planning data pertaining to launch vehicles, as provided by the NASA Cost Estimating Website [6]. Simulations of *Oracle*'s transfer orbit, made using JAQAR's Swing-by Calculator, indicate that a launch C3 of approximately 7 km^2/s^2 is required. Taking a conservative C3 = 10 km^2/s^2 and including a further 10% margin in the payload capacities provided in [6], this yields the quoted boosted mass capability of 937 kg.

While 937 kg is an acceptable upper limit to the mass budget, *Oracle* is intended to be a much more lightweight mission than this. To provide a realistic estimate of the true mass budget of the *Oracle* spacecraft, we choose to draw from the past experience of NASA's NEAR mission [7]. With a similar destination but somewhat more ambitious in scope and utilising technology now more than 10 years old, the mass budget of the NEAR mission serves as an ideal upper bound to that of *Oracle*.

Hence we examine the mass budget summary presented in [7] and modify this to better fit the description of *Oracle*:

- NEAR's instruments are replaced with our own chosen AMIE imager and LIDAR rangefinder instruments (with two LIDAR rangefinders for redundancy of this critical function)
- Masses of solar panels, batteries, transponders, processors and star-trackers are updated to suit *Oracle*'s specific requirements
- Total propellant mass required is calculated based on *Oracle*'s ΔV requirements, overall mass and main engine specific impulse (an iterative calculation)

The resulting conservative mass budget for *Oracle* is included in Table 5 below, with a budgeted dry mass and loaded mass of **335.1 kg** and **651.2 kg** respectively. It should be noted that this experience-based estimate still leaves a 26% margin in terms of the launch vehicle's actual launch capability, allowing ample room for further inflation of the mass budget should it prove inadequate. However, considering the relative simplicity of *Oracle*'s mission compared to the most similar precedent, JAXA's Hayabusa mission (510 kg loaded mass), *Oracle*'s loaded mass of 651.2 kg is very unlikely to be inadequate.

2.1.5 – Power Budget

For the purposes of this preliminary mission design, *Oracle*'s power budget is estimated using a method similar to that employed for the mass budget (Section 2.1.4). The NEAR mission is taken as an ideal upper bound to the power budget requirements of *Oracle*, owing to NEAR's similar destination, more ambitious scope, and technology that is now more than 10 years old. Hence, the stated power budget of NEAR [7] is used as a realistic estimate of Oracle's power budget, with the modifications as per Section 2.1.4.

The resulting conservative power budget for *Oracle* is **365.9 W**. Details of the power budget breakdown are included in Table 5 on the following page.

TABLE 5 – MASS AND POWER BUDGET SUMMARY

Subsystem	Component	Mass (kg)	Power (W)
Instruments	AMIE Imager*	1.8	9.0
	LIDAR Rangefinder (2)*	7.4	17.0
Propulsion	Propulsion Structure	33.1	
	Propulsion System	85.1	
	Propellant - main burn*	306.1	
	Propellant - hover and extra manoeuvering*	10.0	
Power	Ultraflex Solar Panels*	4.0	
	Battery*	5.0	4.3
	Power System Electronics	6.1	2.5
Telecomm	HGA	5.0	
	M/LGA	0.7	
	Solid State Amplifiers (2)	4.1	38.7
	Transponders (2)*	6.0	12.9
	Command Detector Units	0.7	
	Telemetry Conditioner Units	1.7	3.8
	RF Switches, Coax cables	3.0	
GNC	Reaction Wheels	12.9	20.0
	Star Tracker*	2.0	7.5
	IMU	5.3	21.4
	Digital Sun Sensors (5)	1.9	0.3
	Attitude Interface Unit*	2.0	10.0
	Flight Computers*	2.4	8.0
CDH	Command and Telemetry Processors (2)*	4.8	11.9
	Solid State Recorders (2)	3.0	6.4
	Power Switching Unit	5.9	0.7
Mechanical	Spacecraft Primary Structure	78.0	
	Spacecraft Secondary Structure	18.0	
	Despin Mass and Balance Mass	6.1	
Thermal	Thermal Blankets, Heaters, Heliostats	11.0	
	Propulsion survival heaters		75.8
	Spacecraft and instrument survival heaters		71.0
	Instrument operations heaters		40.2
Harness	Harness and Terminal Boards	18.1	4.5
TOTALS	**Total Power Budget**		**365.9**
	Spacecraft Dry Mass	**335.1**	
	Spacecraft Loaded Mass	**651.2**	
	Payload adaptor	45.0	
	Boosted Mass	**696.2**	
LV	Total Margin	240.8 (26%)	
	Launch Capability of Launch Vehicle	937.0	

* Starred items have been modified from NEAR to suit *Oracle*. Other items have been given masses equal to those of NEAR subsystems [7], as a conservative approximation.

2.1.6 – Instrument Selection

A key aspect of *Oracle*'s primary mission is carried out by the LIDAR Rangefinder instrument, which is essential for the accurate determination of Apophis' position with respect to the spacecraft. Owing to the critical nature of its function, two identical LIDAR instruments are carried aboard *Oracle*, providing complete redundancy in case of failure.

ESA's AMIE Imager is also included to contribute towards the secondary objectives of determining the size, geometry and composition of Apophis.

There are a whole plethora of instruments developed for the study of asteroids such as Impactors, Infrared Imagers, Spectrographs and Magnetometers – however, the inclusion of these incur additional weight and power costs without contributing to *Oracle*'s primary mission. In keeping with the desire to develop a streamlined mission at low cost, we have selected the two instruments that provide the most scientific benefit without incurring a significant weight or power penalty.

2.2 – MISSION SUBSYSTEMS

2.2.1 – Launch Vehicle

FIGURE 3 – Delta II 7925 launch vehicle [17]

The launch vehicle chosen for the *Oracle* mission is the Delta II 7925 (Fig. 3), a time-honoured and highly reliable launch system which has been used for several of NASA's deep space missions including NEAR. This launch vehicle has the capability to lift a boosted mass of 937 kg on a direct transfer trajectory to Apophis, allowing a very generous mass budget at an affordable cost (see Section 2.1.3 and Table 5).

At an early stage of the mission design process, cheaper and smaller-capacity launch vehicles such as Taurus 3113 and Cosmos were considered. The Taurus 3113 was simulated to have a launch capacity of 368 kg (boosted mass) to Apophis. While this was initially thought an achievable mass constraint, subsequent analysis of spacecraft bus requirements revealed that a 368 kg boosted mass spacecraft would be a very difficult mass constraint and possibly unattainable, given our operational requirements.

Thus, a larger launch vehicle was chosen, with the Delta II 7925 proving to be the ideal choice in terms of launch capability, payload volume, launch cost and reliability. Choosing the Delta II 7925 allows the mass budget of *Oracle* to be much more relaxed, and also provides a more than ample amount of payload space to accommodate the *Oracle* spacecraft.

2.2.2 – Structure

The *Oracle* spacecraft features a simple and lightweight cubic structure which encloses and supports all subsystems and instruments, following the precedent of JAXA's Hayabusa mission. The high-gain antenna is mounted on the top face of the structure, with the main engine nozzle mounted on the opposite face. The two LIDAR rangefinders and the AMIE imager are all located on the same top face as the high-gain antenna, and the two Ultraflex deployable solar panels are mounted on the sides of the cubic structure. Wherever possible, RCS thrusters are placed to avoid interfering with the solar panels or external instruments. The flight computer and other sensitive electronics will be located as near to the centre of the spacecraft as possible, to maximise shielding from radiation.

The structure itself will be constructed from aluminium, chosen for its high strength-to-weight ratio and ease of machining. The total mass of the primary and secondary structure is 96 kg, based on the NEAR mission's mass breakdown (Table 5) – this agrees with the rule of thumb estimate for structure mass as cited by Wertz and Larson [8].

2.2.3 – Spacecraft Propulsion

The propulsion system onboard *Oracle* will be a dual mode N_2O_4/N_2H_4 system. There will be one main engine, running on both N_2O_4 and N_2H_4 as a bipropellant engine with a specific impulse of 330 s [8]. As the main engine will be required for large velocity changes (ΔV = 1520 m/s), it is advantageous to choose this engine to be a bipropellant engine rather than a monopropellant engine – which, while cheaper, simpler and more reliable, has a lower performance than a bipropellant engine. The mass of propellant required for the main engine burn has been calculated to be 316.5 kg, found by a conservative and iterative calculation method as described in Section 2.1.4.

Together with the main engine, there will also be RCS thrusters located around the spacecraft which run solely on N_2H_4 (monopropellant thrusters). These thrusters, approximately 12 in number, will be used for spacecraft attitude control in tandem with the reaction wheel assembly during all phases of the mission.

Combining the RCS thrusters and main engine into one dual-mode package allows them to share a common fuel tank, producing valuable weight savings for *Oracle* and ensuring that the spacecraft remains overall lightweight and simple. Ion engines may be considered to replace the proposed dual-mode propulsion system – however, ion engines tend to have large power demands (hundreds of Watts) which bring about an associated net increase in the amount and complexity of hardware.

2.2.4 – Attitude Control (ACS)

Oracle cannot complete its mission if it has no means of controlling the spacecraft's orientation in space. *Oracle* must be 3-axis stabilised, to a pointing accuracy of roughly 0.1° (based on rangefinder, imager and communications antenna pointing requirements). As mentioned in the previous section, the ACS will consist of approximately 12 hydrazine monopropellant thrusters in a dual-mode arrangement with the main engine, acting together with a reaction wheel assembly and de-spin mass. Momentum dumping from the reaction wheels can be facilitated by the hydrazine RCS thrusters throughout the mission.

The ACS will have many important jobs to do, including:

- Removing the spin of the spacecraft after release from the Delta II 7925 launch vehicle's upper stage (which will be spinning for stability during the orbital insertion burn).
- Maintaining 3-axis stability of the spacecraft during the coast phase, and pointing the spacecraft in the right direction when high-gain antenna communications with Earth are required.
- Orient the spacecraft so that the LIDAR rangefinder and AMIE imager are pointed towards Apophis when necessary.
- Assist in maintaining a stable hover position 1 km above Apophis' surface.

The de-spin mass plays an important role in the first of these listed tasks. The RCS thruster system plays a vital role in all tasks, although wherever possible, it will be best to rely on the reaction wheels for small manoeuvres rather than waste progressively more and more propellant on the task.

2.2.5 – Thermal Control (TCS)

The interplanetary trajectory of the *Oracle* mission is a thermally benign one with little variation in incident solar flux. Albedo loads after arrival at Apophis are not anticipated to be a large source of heat input to the spacecraft. Thus, a traditional combined passive/active thermal control approach of multi-layer insulation, radiator panels and heaters can easily be employed to ensure that all of *Oracle*'s components remain within their acceptable temperature ranges.

Heaters employed within the spacecraft will be fully redundant and will be controlled electronically rather than via mechanical thermostats, as solid-state electronic control of the heaters is found to be more reliable than the alternative. The components most in need of heaters will be the propellant tanks. Multi-layer insulation blankets will cover the majority of the surface of the spacecraft, with some panels cut out to make room for radiator panels to reject excess heat from within the spacecraft. The radiator panel surfaces can be finished with high-emissivity materials such as silvered Teflon, a popular choice for spacecraft radiators. Components known to generate large amounts of heat, such as power amplifiers, will be strategically placed near these radiator areas of the spacecraft to ensure efficient management of the spacecraft's thermal state.

2.2.6 – Power

Main power to the spacecraft will be supplied by deployable solar arrays. The particular brand chosen is UltraFlex 175 manufactured by ABLE Engineering (Fig. 4). This is a light weight, low volume, triple junction solar array which provides 27% efficiency compared with less than 22% for traditional triple junction GaAs cells. UltraFlex arrays weigh less than 25% of the weight of an equivalent traditional array. The panels and the deployment mechanism have been extensively tested and a full test of the system will be carried out on the Space Technology 8 mission due to launch in 2009. [9]

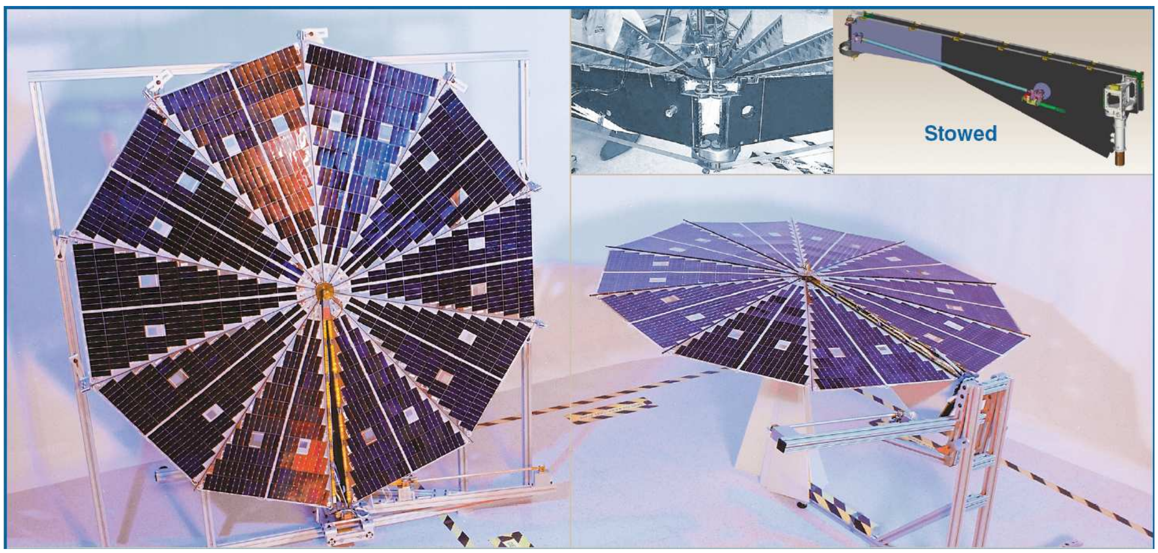

FIGURE 4 – UltraFlex deployable solar arrays [9]

The other possibilities for power generation include primary batteries, fuel cells and a Radioisotope Thermoelectric Generator (RTG). Primary batteries were immediately discounted due the length of the mission. While fuel cells provide a high specific power, the disadvantage is due to the fact that the fuel has to be carried as part of the payload. Thus fuel cells are not ideal for a small spacecraft with a limited mass budget such as *Oracle*. RTGs are generally employed when a spacecraft is too far from the sun to generate sufficient power. It is very costly and provides little added benefit in terms of power generation capabilities compared to solar arrays. [8]

The arrays will be arranged on the spacecraft as two separate deployable units with a total area of 2.1 m^2 and weigh 4.3 kg. They are capable of producing the total power requirement of the spacecraft of 350W even when the angle of incidence of sunlight deviates from the normal by 45°. This allows greater flexibility in attitude control for imaging the asteroid. For the mission duration of 2 years, the degradation of the array will be negligible. However, a further redundancy of 30% was added to the solar array area for any unexpected increases in power consumption or in case of failure of one panel. Since efficiency and simplicity are paramount to the success of the mission, a

Direct Energy Transfer system is used for power regulation. A Peak-Power Tracker method was also considered. However, as the dc-dc converter used in this method operates in series with the solar array it uses 4 – 7% of total power. [8]

The method of energy storage for the spacecraft will be a rechargeable Lithium Ion battery. This has to be capable of supplying the full spacecraft power load when the solar arrays are in shadow. This is most likely to occur during the daily transmission of data where the HGA is required to point towards Earth for 30 minutes. There may also be occasions where imaging the asteroid will place the solar arrays in shadow. Only 10 images and 60 LIDAR measurements are taken per day and each measurement is very short duration. Thus the battery was designed to be able to supply the spacecraft load for a maximum of 1 hour at a time.

A Li-ion battery was chosen due to its high energy density, longer life cycle and low depth of discharge compared to NiCd batteries. SAFT VES 180 was the best choice with a specific energy of 165 Wh/kg. With each cell weighing 1.1 kg, four cells will be used with more than 50% redundancy. [10]

2.2.7 – Communications

Oracle will have daily downlink and uplink communications with ground stations in order to transmit telemetry, health and science data and to receive ranging tones and command data. The communications subsystem needs to be highly reliable as the success of the entire mission rests on the spacecraft's ability to regularly communicate its position. The following table gives a summary of the key components of the subsystem:

TABLE 6 – KEY COMMUNICATIONS SUBSYSTEM COMPONENTS

Component		**Characteristics**	**Redundancy**
Spacecraft antennas	High Gain (HGA)	1.5m, parabolic, fixed	1
	Medium gain (MGA)	0.3m, parabolic, fixed	1
	Low gain (LGA)	fixed	2
Amplifier	Solid state	40 W output power	2
Transponder	Small Deep Space Transponder (SDST) [11]	Ka band	2
Antenna efficiency	55%		
Communications band	Uplink	Ka, 34 GHz	
	Downlink	Ka, 32 GHz	
Signal to noise ratio (SNR)	9.6 dB		
Bit Error Rate (BER)	10^{-5}		
Modulation scheme	BPSK		

Parabolic antennas were chosen due to their proven space heritage and high performance for a given weight. Other types of antennas such as phased array antennas were considered. However, in these alternatives the cost and weight penalty outweigh the performance benefits. All antennas are fixed for higher reliability and so communications with Earth requires the attitude control system to orient the entire spacecraft. Low gain antennas are fixed on either side of the spacecraft providing omnidirectional coverage for use during emergencies and for locating Earth.

Ka band was chosen for both uplink and downlink communications due to the low power requirements and to keep the antenna size as compact as possible. This means that solid state amplifiers can be used for transmission instead of the high power travelling wave tube amplifiers (TWTA), which incur a weight penalty. The SDST in design has its heritage in the Deep Space 1 mission with benefits of being very light and low in power consumption. Finally, the BPSK modulation scheme was chosen as it makes good use of the spectrum and is least susceptible to errors.

On average, with the LIDAR taking 50 measurements per day and AMIE taking 10 images, the total data gathered per day is 9 Mbits. All TT&C and scientific data will be processed and stored onboard. Once a day, the accumulated data will be transmitted to Earth. The communications subsystem was sized to be capable of transmitting and receiving the required amounts of data at the worst case distance of 1.9 AU from Earth. The following table gives the data rates achievable with various combinations of spacecraft and ground station antennas at a transmitted power of 40W. For most of the mission, the spacecraft will be much closer to Earth and higher data rates will be possible. Normal daily downlink communications will be carried out using the HGA at a data rate of 4.7 kbps transmitting to a 15m ground station antenna. The total tracking time required will be 30 minutes.

TABLE 7 – ACHIEVABLE DATA RATES

Antenna	Data rates at worst case distance 1.9 AU			
	2m ground station	15m ground station	34m ground station	70m ground station
High gain	84 bps	4.7 kbps	24 kbps	0.1 Mbps
Medium gain antenna	negligible	190 bps	975 bps	4.1 kbps
Low gain x 2	negligible	negligible	negligible	16 bps

Table 8 gives worst case uplink data rates for ground stations using 100W transmitters transmitting to the HGA.

TABLE 8 – WORST-CASE UPLINK DATA RATES

Ground station antenna size (m)	Data rate
2	180 bps
15	10.6 kbps
34	54.6 kbps
70	0.2 Mbps

Generally a 15m ground station will be sufficient for uplink communications. If a particularly high data rate is required such as during orbit manoeuvres, the larger antennae can be used to achieve significantly higher data rates. Daily uplink communication will be used in tracking the spacecraft, which will in turn be used in constraining the orbit of Apophis through Kalman filtering.

Emergency

In the event that the HGA cannot be used for communications, the medium gain antenna can be used to transmit to 15m, 34m or 70m antennas and still salvage the primary mission data and some images. The LGAs provide effective omnidirectional coverage and can be used for locating Earth.

Ground stations

The ESA ESTRACK ground stations will be used for main operations, as they have 15m antennas. Not all of these stations have Ka band reception and transmission capabilities at present and will need to be upgraded before they can be used. If more scientific data needs to be transmitted, then it will be more economical to use a larger ground station which will provide a higher downlink data rate. The 34m and 70m NASA DSN antennas will generally be used in emergencies. The cost of using a 15m ground station is much less than using a larger antenna for a shorter period of time.

2.2.8 – Guidance, Navigation and Control (GNC)

The GNC subsystem controls the attitude of the spacecraft. During normal operation, it is responsible for controlling the thrusters for ΔV manoeuvres, ensuring pointing accuracy for imaging, measurement taking and communications with Earth. When these activities are not taking place, it is responsible for maintaining the spacecraft attitude such that the solar panels can provide the maximum power. During an emergency GNC is responsible for ensuring that the spacecraft carries out the necessary procedures to remain safe and operational. This includes attempting to place the Earth within the antenna pattern of the LGA or the MGA in order establish ground communications.

The subsystem has its heritage in the NEAR mission and uses the same suite of sensors. Attitude determination uses star trackers, sun sensors and an Inertial Measurement Unit (IMU) containing gyros. Attitude corrections are carried out using actuators and reaction wheels. The flight computer and the Attitude Interface Unit (AIU) maintain closed loop attitude control.

Gyros and accelerometers in the IMU are used for three-axis rate measurement. While three are sufficient, four are included for redundancy. The sun sensors are positioned so that the spacecraft can recover its attitude from an inadvertent tumble. The Star Tracker used in *Oracle* is from the PROBA mission and enables us to acquire a very high accuracy inertial reference [12]. It has the added benefit of being much lighter than its predecessors. In case of Star Tracker failure, AMIE can be used for capturing star images which can then be processed on the ground to provide a sufficient reference frame.

Reaction wheels alone are sufficient for normal attitude control. While three reaction wheels are sufficient to provide 3-axis control, four are included for redundancy. Hydrazine monopropellant RCS thrusters are used to assist in ΔV manoeuvres and to balance external torques such as radiation torque.

The flight computer continuously estimates the spacecraft state and compares this with the state required for a particular operation. Control outputs are then generated to correct any deviations. The controls generated by the flight computer are implemented by the AIU which controls the reaction wheels and thrusters. The most stringent requirement on the attitude control system is imposed by the HGA which has a beam width of 7.6 milliradians. In order to maintain an effective and steady communications link with the ground, we impose a 2 milliradian pointing accuracy requirement. The GNC suite of instruments is capable of achieving a pointing accuracy of 1.7 milliradians. Both the flight computer and the AIU are completely redundant and fully cross-strapped. In the case of flight computer failure, the AIU is capable of putting the spacecraft in a safe state. The two AIUs are BAE Systems RAD6000 space computers while the flight computer is a Honeywell dependable microprocessor developed for NASA's Millennium Program and to be flown on Space Technology 8 in 2009. [13]

2.2.9 – Command and Data Handling (CDH)

The Command and Data Handling subsystem comprises of redundant command and telemetry processors, redundant solid state recorders (SSR) and an interface to a redundant 1553 standard bus for communicating with other subsystems. The redundant components are cross-strapped among themselves and among the telecommunications subsystem. The CDH subsystem is responsible for command execution, telemetry management and autonomous control of spacecraft. The architecture for this subsystem is based on the corresponding subsystem on the NEAR mission.

The command and telemetry processors are BAE Systems RAD6000 space computers. The FPGA based RAD6000 was successfully used as the flight computer on the Mars Rover. The FPGA based implementation allows for convenient reconfiguration of the processor and the system is available as a double redundant computer subsystem. [14] The solid state recorders are those used on NEAR supplied by SEAKR.

The data generated by TT&C and the instruments are transmitted to a ground station on a daily basis. Per day, the storage requirement is less than 10 Mbits. Taking into consideration the fact that we may wish to take more than 10 photos a day or store the image data for longer periods and transmit together at a later time, the storage requirement will be upgraded to be able to store up to 100 photos a day for up to one week. This corresponds to 0.7 Gbits of storage. Leaving a margin for reliability, the storage capacity for a single SSR is 1 Gb. Another identical SSR is available for redundancy in case of failure of the first or if data exceeds the capacity of the first SSR.

The CDH subsystem is capable of storing commands for execution at a later time, either at a specified Mission Elapsed Time (MET) or when autonomous action is required. Autonomous action is based on evaluating combinations of rules to determine if action needs to be taken to remedy the situation.

2.2.10 – Instruments

With the intention of designing a streamlined mission, *Oracle* is equipped with two instruments for orbit determination and gathering scientific data.

2.2.10.1 – LIDAR (Light Detection and Ranging)

This instrument is used to find the distance between Apophis and the spacecraft very accurately. It does so by reflecting a pulse of laser light from the asteroid surface and looking at the scattered light and the delay. The narrow width of the laser beam allows *Oracle* to map the physical features of Apophis to a much higher resolution than would be possible with radar. The LIDAR will be boresighted with the imaging camera.

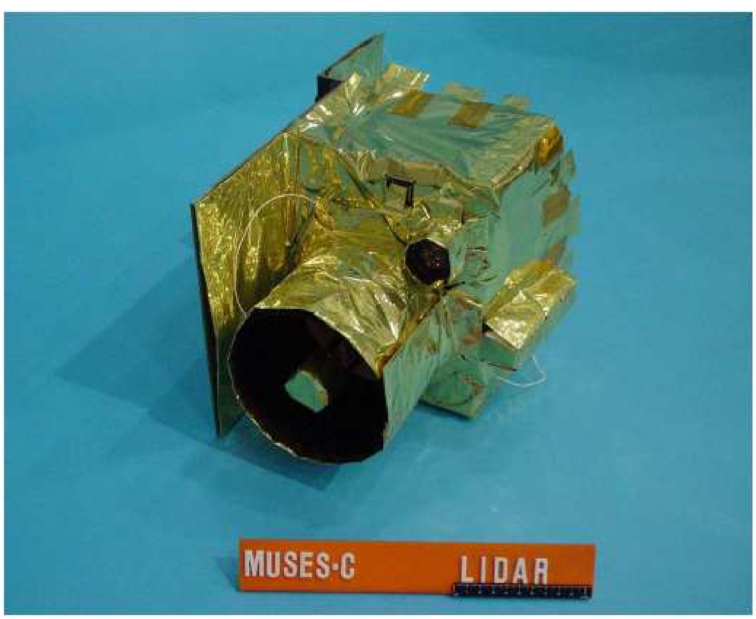

FIGURE 5 – LIDAR rangefinder instrument [15]

Specifications [15]

Range: 50m ~ 50km
Accuracy: ±1m (@ 50m)
Laser wavelength: 1064nm
Repetition rate: 1 Hz

With a weight of only 3.7 kg, which includes power conversion electronics and a radiator, the LIDAR used in the Hayabusa mission (Fig. 5) is lighter, more compact and uses less power than the equivalent instrument used in the NEAR mission. [7]

This instrument is critical to the accurate orbit determination of Apophis and central to the success of the mission. Consequently, two identical instruments will be placed on the spacecraft for redundancy. The LIDAR will take 50 measurements per day accumulating 0.15 Mbits of data which will be compressed, stored and transmitted to a ground station on a daily basis.

2.2.10.2 – AMIE Imager

AMIE (Fig. 6) is an imaging system from the ESA mission to the Moon, SMART-1. The instrument was developed by SPACE-X and includes a miniaturised micro-camera and micro-processor electronics. The camera successfully provided high resolution CCD images of lunar areas with colour imaging at three filters (750 nm, 915 nm and 960 nm).

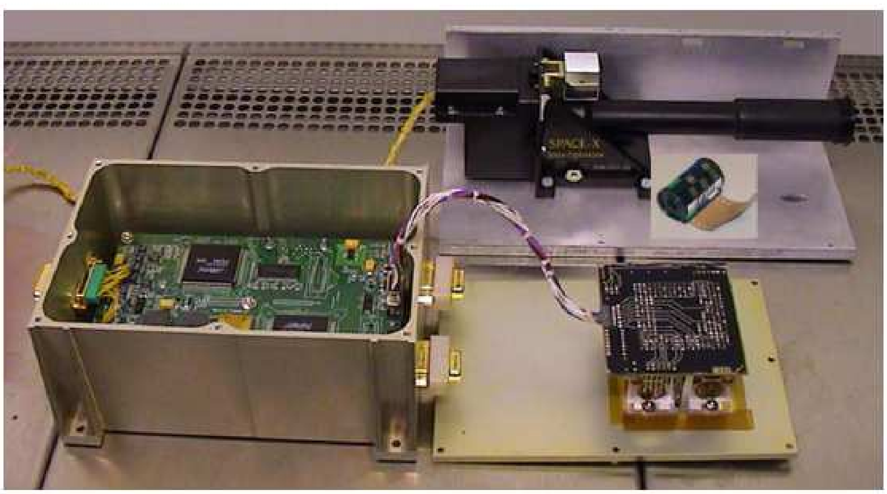

FIGURE 6 – AMIE Imager [16]

Specifications [16]

FOV: 5.3°×5.3°
CCD: 1024×1024 with 10 bits per pixel
Image compression unit with high data compression rate
Total mass: 2kg (including radiation shielding)

The imaging wavelengths chosen for the SMART-1 mission allowed us to discern geological properties specific to the moon. The filter wavelengths will be adjusted to obtain the maximum information when imaging Apophis. With a compression rate of 12:1, each image is 0.88 Mbits. The camera will take 10 images every day as a worst case scenario and these will be transmitted to Earth on a daily basis. The limit of 10 is purely to restrict the transmission time required when the Apophis is at its maximum distance from Earth. More images can be taken at a closer distance to Earth or if multiple transmission sessions are used or a larger ground station antenna is used.

2.2.11 – Redundancy and Reliability

Oracle's mission design philosophy is to create a streamlined mission targeted at achieving the core goal of determining Apophis' orbit more accurately. Due to the urgency and importance of the mission, where there will be no second opportunity to obtain this valuable data, reliability becomes critical. Throughout the design, we have placed great emphasis on reliability by striving to design a spacecraft that is simple with fewer potential weak points. Bottlenecks were avoided as much as possible, so that failure at one point does not compromise the entire mission. The following steps were taken to achieve this:

- A launch vehicle with a long track-record of reliability was chosen
- All technologies used in *Oracle* have been or will be spaceflight-tested on actual missions
- All design was done with generous margins to allow for operation in unexpected situations
- Solar panels and antennas are fixed to minimise failure in moving parts. (*Oracle*'s solar panels are deployable, but this is a once-off deployment.)
- For mission critical components such as the transponder, LIDAR, processors, heaters and antennae, there is at least one redundant backup
- The mission operations are kept simple, with no risky manoeuvres (such as landing on Apophis) required by the spacecraft

To map out the issues for consideration in terms of reliability, a failure mode effects and criticality analysis is provided in Table 9 on the following page.

TABLE 9 – FAILURE MODE, EFFECTS AND CRITICALITY ANALYSIS

Failure mode	Effect	Criticality	Remedy
Launch vehicle failure	Failure to lift off will mean missing the ideal launch window.	Minor	Launch at next available opportunity
	Failure to achieve orbit	Fatal	Loss of spacecraft
Solar panel failure	Provided only one panel fails, spacecraft will be able to generate more than half the power requirement due to the design margin.	Fatal if both panels fail. Manageable with reduced capability if one panel fails	Turn off unnecessary equipment. This may mean that secondary scientific objectives are not achieved
Antenna failure	Disables high speed communications. Unless rectified, communications data rate will be significantly reduced. The number of AMIE images capable of being transmitted will be significantly reduced. Central mission objective can still be achieved.	Medium	Use medium gain antenna.
LIDAR failure	Accurate information on the position of Apophis with respect to spacecraft will be lost. Spacecraft tracking should still give sufficient accuracy to achieve key mission objective	Major if both LIDARs fail. No effect if only one fails.	Use images from AMIE to judge distance.
ACS failure	Stability of spacecraft will be lost; unable to point HGA at Earth, or LIDAR at Apophis	Major – Apophis tracking operations severely compromised	Make do with remaining functional RCS thrusters and momentum wheels; otherwise loss of spacecraft
Thermal system heater failure	Propellant tanks, electronics or other sensitive components could go to dangerous temperatures and cause them to fail	Major – loss of propulsion, computers, communication, etc	Loss of spacecraft
AMIE failure	Images for Apophis will not be obtained. This has no impact on the key mission objective	Minor	Rely on LIDAR mapping for imagery of Apophis

From the failure modes investigated in Table 9, we can come to some conclusions about the necessary reliability of each of *Oracle*'s subsystems. It is evident that LIDAR, ACS and Thermal System heaters are very crucial to the mission, whereas the loss of AMIE or the HGA will not be quite so catastrophic. This illustrates the reasoning behind our decision to design redundancy into the LIDAR, ACS and Thermal System heaters, to ensure the best possible chance of success for *Oracle*.

3.0 – PROJECT COST

3.1 – COST-ESTIMATING METHODOLOGY

A preliminary estimate of the total project cost of our Apophis mission has been obtained by a parametric cost estimation method following Section 20.3 of Wertz & Larson [8]. Cost estimating relationships (CERs) from this source have been used to derive an estimate of total project cost based on various top-level parameters of our mission, including:

- Spacecraft total dry weight
- Communications subsystem weight
- Structure weight
- Thermal subsystem weight
- Electrical power system weight
- Telemetry, Tracking & Command / Data Handling subsystem weight
- Attitude determination and control subsystem weight
- Apogee kick motor weight
- Number of lines of code of flight software
- Number of lines of code of ground software
- Launcher type
- Frequency band
- Duration of ground station operations
- Amount of contractor labour (staff-years)
- Amount of government labour (staff-years)

Where more than one CER has been provided to estimate a given cost component, the CER yielding the higher estimate has been chosen. To ensure a conservative overall result, the quoted standard errors for each CER have been added to the mean values.

3.2 – PROJECT COST ESTIMATE

Appendix A contains results of parametric cost modelling following the methodology outlined in Section 3.2. Adding standard errors to mean estimated values gives a conservative estimated project cost of **$226 million** (FY2007 US dollars). This proves the *Oracle* mission to be an outstandingly cost-efficient solution to the problem of reliably refining the orbit determination of asteroid Apophis.

4.0 – CONCLUDING REMARKS

A preliminary design for the *Oracle* mission to asteroid Apophis has been presented, revealing this to be an outstandingly cost-efficient, reliable and focussed solution to the problem of more accurately determining Apophis' orbit. The mission design is not distracted by extraneous scientific objectives – rather, the primary objective of determining Apophis' orbit is considered supreme. With a conservatively estimated price tag of just $226 million, there is no doubt that *Oracle* is cost-effective by any measure – and the sheer simplicity of *Oracle*'s plan for tracking Apophis ensures it a very high probability of success.

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APPENDIX A: PROJECT COST ESTIMATION

Monash University Apophis Mission Design Team

Mission Cost Estimation

Segment	Category	Area	Cost Component	Parameter X (unit)	X Value	Cost (FY00$K)	Error (FY00$K)
Space	RDT&E	Payload	Communications	comm subsystem weight (kg)	21.2	7,490	3,820
Space	RDT&E	Spacecraft	Spacecraft (whole)	spacecraft dry weight (kg)	335.1	33,845	11,169
Space	RDT&E	Spacecraft	Structure	structure weight (kg)	96.1	6,943	2,638
Space	RDT&E	Spacecraft	Thermal	thermal system weight (kg)	11	1,806	813
Space	RDT&E	Spacecraft	EPS	EPS weight (kg)	15.1	947	540
Space	RDT&E	Spacecraft	TT&C/DH	TT&C/DH weight (kg)	13.7	3,994	2,277
Space	RDT&E	Spacecraft	ADCS	ADCS weight (kg)	26.5	7,952	3,817
Space	RDT&E	Spacecraft	Apogee Kick Motor	AKM weight (kg)	0	-	-
Space	RDT&E	IA&T	Integration, assembly & test	S/C + PL RDT&E cost (FY00$K)	29,132	7,252	3,336
Space	RDT&E	Program	Program level cost	S/C + PL RDT&E cost (FY00$K)	29,132	11,155	4,016
Space	RDT&E	GSE	Ground support & equipment	S/C + PL RDT&E cost (FY00$K)	29,132	6,805	2,314
Space	SW	-	Flight software	Flight SW lines of code (thousands)	10	4,350	-
Space	TFU	Payload	Communications	comm subsystem weight (kg)	21.2	2,968	1,276
Space	TFU	Spacecraft	Spacecraft (whole)	spacecraft dry weight (kg)	335.1	14,409	5,187
Space	TFU	Spacecraft	Structure	structure weight (kg)	96.1	1,259	491
Space	TFU	Spacecraft	Thermal	thermal system weight (kg)	11	276	168
Space	TFU	Spacecraft	EPS	EPS weight (kg)	15.1	889	391
Space	TFU	Spacecraft	TT&C/DH	TT&C/DH weight (kg)	13.7	2,808	1,151
Space	TFU	Spacecraft	ADCS	ADCS weight (kg)	26.5	3,739	1,271
Space	TFU	Spacecraft	Apogee Kick Motor	AKM weight (kg)	0	-	-
Space	TFU	IA&T	Integration, assembly & test	S/C + PL total weight (kg)	335.1	3,485	1,533
Space	TFU	Program	Program level cost	S/C + PL total recurring cost (FY00$K)	11,938	4,071	1,588
Space	TFU	LOOS	LOOS	S/C + PL total weight (kg)	335.1	1,642	690
Launch	-	-	Launch vehicle unit cost	Launcher type	Delta II 7925	58,362	-
Ground	FGS	Devel.	Facilities	(based on ground SW cost)	-	792	-
Ground	FGS	Devel.	Equipment	(based on ground SW cost)	-	3,564	-
Ground	FGS	Devel.	Ground software	Ground SW lines of code (thousands)	20	4,400	-

APPENDIX A: PROJECT COST ESTIMATION

Segment	Category	Area	Cost Component	Parameter X (unit)	X Value	Cost (FY00$K)	Error (FY00$K)
Ground	FGS	Devel.	Logistics	(based on ground SW cost)	-	660	-
Ground	FGS	Devel.	Management	(based on ground SW cost)	-	792	-
Ground	FGS	Devel.	Systems Engineering	(based on ground SW cost)	-	1,320	-
Ground	FGS	Devel.	Product Assurance	(based on ground SW cost)	-	660	-
Ground	FGS	Devel.	Integration & Test	(based on ground SW cost)	-	1,056	-
Ground	ET	Hardware	Ground comms electronics	Frequency band	Ka	750	-
Ops/Maint.	GSOS	-	Maintenance	Duration of ground station ops (years)	2	1,751	-
Ops/Maint.	GSOS	-	Contractor labour	Amount of labour (staff-years)	40	6,400	-
Ops/Maint.	GSOS	-	Government labour	Amount of labour (staff-years)	40	4,400	-
TOTAL	**LIFE**		**TOTAL LIFE CYCLE COST**			**164,738**	**32,130**
				Conservative Total Project Cost (FY07$M) :			**226.0**

Notes:

1. The above parametric cost estimation follows Section 20.3 of Wertz & Larson, "Space Mission Analysis & Design" [8]
2. Where two CERs have been provided by SMAD, the CER giving the higher cost estimate has been chosen.
3. To ensure a conservative total project cost, the quoted standard errors for each CER have been added to the mean values.