FORESIGHT: DESIGNING A RADIO TRANSPONDER MISSION TO NEAR EARTH ASTEROID APOPHIS

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MISSION OVERVIEW

Introduction

The Foresight spacecraft is a concept design for a radio tagging mission to Near Earth Asteroid (NEA) Apophis. This Near Earth Object (NEO), officially referred to as 99942 Apophis (2004 MN4), has a probability of Earth impact of 2.2e-05 (a 1 in 45,456 chance) on Friday, April 13, in the year 2036 [1]. In the year 2029 Apophis will approach the Earth within a distance closer than a geostationary satellite. If Apophis passes through a several-hundred-meter-wide "keyhole" in space during this approach, it will impact the Earth in 2036. If additional Earth-based observations are not sufficient to rule out an impact in 2036, a better determination of the object's orbit is required. Such precision may be obtained by "tagging" the object with a beacon, transponder or reflector positioned on or near the asteroid.

The Foresight spacecraft described here (and shown in Fig. 1a) is a representative design for such an Apophis tagging mission and provides a useful starting point for future designs for other potentially hazardous objects. Primary power is provided by two solar arrays, augmented by batteries. Communications are accomplished with a combination of high and low gain antennas. The spacecraft is maneuvered by a single bi-propellant chemical main engine and a number of small thrusters (using the same propellants). Flight proven equipment was selected for all of the major subsystem components. The scientific instruments are derived from similar missions. The spacecraft uses a Propulsive Transfer Vehicle (PTV) shown in Fig. 1b, a simple bi-propellant chemical stage, to assist in achieving the necessary Earth departure velocity.

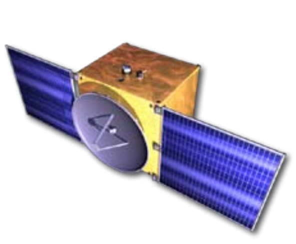

Fig. 1a. Foresight Encounter Spacecraft (ES)

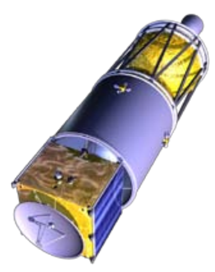

Fig. 1b. Foresight ES and Propulsive Transfer Vehicle (PTV) Stack

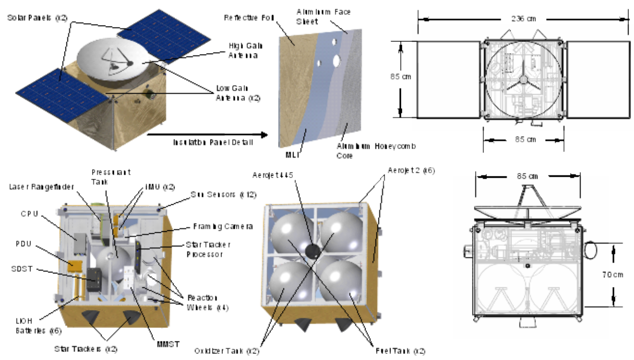

Fig. 1c. Foresight Encounter Spacecraft (ES) Detail

The objective of this mission is to conduct precise in-situ measurements of the position of Apophis in order to enable a very accurate determination of the NEO's heliocentric orbit. According to S. R. Chesley in 2005, the 2029 Apophis-Earth encounter distance is predicted to be 5.89 Earth radii, ± 0.35 Earth radii, (3σ) based on current knowledge [2]. This equates to an approximate largest dimension of the 3σ error ellipse of 4500 km. Our goal is to determine the orbital state with sufficient precision, so that when it is propagated forward to the close approach with Earth in 2029, the long dimension of the +/- 3σ error is reduced to less than 14 km (i.e. 6 times sigma). This translates approximately to a 10% impact probability if the keyhole is right in the middle of the 14-kilometer error ellipse. For the purposes of this study, data must facilitate a deflection mission decision by 01/01/2017. Our measurement approach is two-fold: 1.) Accurately determine the location of the center of mass of Apophis relative to the Foresight spacecraft and 2.) Accurately determine the position of the Foresight spacecraft relative to the Earth and Sun.

The strategy employed by the Foresight mission for improving the knowledge of Apophis' orbital parameters is to send a small tracking spacecraft to Apophis to first determine its center of mass, and then track the asteroid for a specified period of time. The center of mass is determined using an on board Advanced Visual Imagery Mechanism (AIM) and Laser Altimeter Device (LAD), in conjunction range measurements from the Deep Space Network (DSN). Once the center of mass has been determined, the spacecraft will take regular measurements of its position state relative to Apophis while maintaining a constant separation from the asteroid's center of mass in a trailing heliocentric orbit. Simultaneously, the DSN is used to accurately determine the range of the spacecraft from Earth over a period of time. These measurements are combined using an orbit determination algorithm to reduce the uncertainty in Apophis' orbital parameters until the long dimension of the ±3σ of Apophis in 2029 is less than 14 kilometers.

Overall Mission Summary

The Orbital Sciences Corporation (Orbital) Minotaur IV launch vehicle was selected as the primary launch vehicle for the Foresight mission [3]. Launch will take place from the Mid-Atlantic Regional Spaceport (MARS) facility located at the Wallops Flight Facility in Virginia. The combination of a Minotaur IV and a PTV gives significant cost savings as compared to a larger interplanetary launch vehicle with no PTV. The Minotaur IV will deliver the combined Foresight spacecraft to a 185 km (100 nm) circular LEO launched on a due east trajectory from the Mid-Atlantic Regional Spaceport (MARS) facility located at the Wallops Flight Facility in Virginia (i.e. into a 38º inclination departure orbit) [4]. The mass of the combined spacecraft at launch with growth margin is 1,607 kg. The reported payload capability of the Minotaur IV from the MARS is 1,680 kg, yielding a launch vehicle payload margin of 4.4% not including the 20% growth margin included in the vehicle masses themselves. The combined vehicle has a maximum effective diameter of 1.20 m and a length of 4.04 m. This is within the limits of the payload fairing of the Minotaur IV, which are 1.33 m in diameter and 4.62 m in length [4].

Upon delivery, the PTV immediately ignites its main engine and initiates the Earth Departure [PTV Maneuver] phase. A successful Earth Departure maneuver places the spacecraft on a transfer orbit from Earth to Apophis, and the mission enters the Cruise phase, lasting from 115 to 360 days depending on the launch opportunity. Once proper functionality of all instruments and subsystems has been confirmed, the spacecraft performs the Apophis capture maneuver and enters into an orbit about Apophis, beginning the Observation phase. The goal of the Observation phase is to generate accurate mass, bulk density, gravitational, and shape models of the asteroid. These models are generated in a similar fashion to those of Eros by the NEAR mission using a 30 day data arc combining laser rangefinder and imager data[3)]. During this phase, the spacecraft maintains an orientation with the solar arrays facing the Sun and the scientific instruments facing Apophis so as to facilitate continuous fully powered data acquisition. Major data downlinks to the DSN occur once per week: during these periods, the spacecraft is rotated to aim the high gain antenna at the Earth and the batteries augment the non-optimal solar array power production. At the end of the Observation phase, the spacecraft withdraws from the Apophis orbit and enters an Apophis-trailing heliocentric orbit with a 2 km separation from the asteroid's center of mass (the Tracking phase). The position of the spacecraft is measured once a week by the DSN while the spacecraft transmits its range vector from the asteroid. After 300 days of these weekly measurements, the orbit of Apophis will have been determined to sufficient accuracy to meet mission requirements.

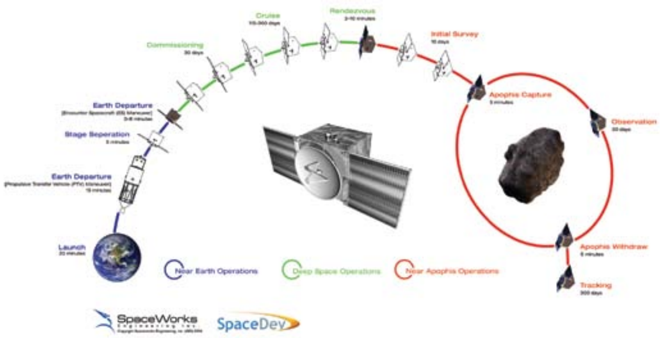

Fig. 2. Foresight Concept of Operations and Mission Schedule for Target Launch Date

The full range of launch windows offers 134 days worth of launch opportunities spanning nearly two and a half years. The target launch date for this mission represents the optimal launch opportunity in the primary launch window and corresponds to the mission scenario in which the PTV is capable of performing the entire Earth departure maneuver without assistance from the ES. The target launch date starts on May 9, 2012 with the transfer trajectory yielding a transfer time of 310 days from Earth to Apophis. Assuming the complete use of the 10 day Initial Survey period and 30 day Observation period, this opportunity yields a 650 day, or roughly 22 month, mission from launch to mission completion.

In order to determine potential mission launch opportunities, a sweep of potential Earth-Apophis transfer trajectories was performed using Bullseye, a Java-based trajectory analysis software package available for purchase by SpaceWorks Software, a division of SpaceWorks Engineering, Inc. (SEI). It was determined that a total delta-V limit of 6,000 m/s represented an ideal compromise between spacecraft size and launch flexibility. Analysis of the five indentified launch windows revealed that the division of delta-V between Earth departure and Apophis arrival was significantly different across the launch windows. The arrival delta-V limit of these five launch windows was set be 2,400 m/s based on preliminary vehicle mass estimates. In addition to the 3,600 m/s delta-V capability of the PTV, an additional 2.5% (90 m/s) of reserves were added to this vehicle to represent unused residual propellant at the end of the burn. For the primary launch opportunity, the spacecraft is not required to burn for the Earth departure maneuver. A station-keeping budget of 20 m/s per year was calculated based on the average perturbing accelerations from these local bodies. To be conservative, the ES carries three years of station-keeping delta-V, or 60 m/s.

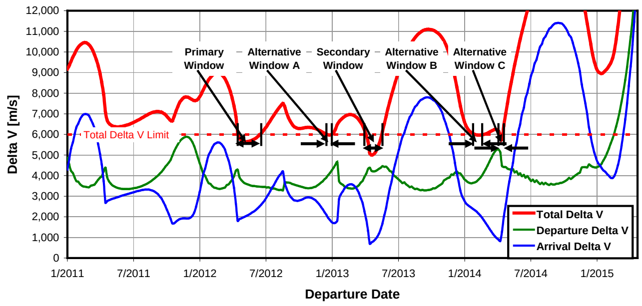

Fig. 3. Departure, Arrival, and Total Delta-V for Minimum Total Delta-V Trajectories from LEO to Apophis

APOPHIS ORBIT DETERMINATION AND 2029 ERROR ELLIPSE PREDICTION

Locating the Center of Mass of the Target Asteroid

Reduction of the error ellipse of Apophis first requires the ability to precisely measure the position of Apophis' center of mass with respect to the Foresight spacecraft. Before these measurements can be made, an accurate shape, rotation, and gravitational model of the asteroid must be developed. These models will be generated during the Observation phase of operations using the PCODP software tool developed by J. K. Miller, et al and used during the Near-Earth Asteroid Rendezvous (NEAR) mission to generate similar models for the asteroid Eros [5]. Optical tracking of landmarks from imaging data generated by the onboard camera along with range data generated by the laser altimeter are used to generate an accurate shape (topography) model and determine rotational details of the asteroid. These data are also used to accurately determine the spacecraft's position with respect to the asteroid's volumetric centroid. While the spacecraft is in orbit about Apophis, continuous prediction of the spacecraft's position with respect to the surface features is combined with the Deep Space Network (DSN) measurements of the spacecraft's position and the spacecraft's own measurements of its position with respect to Apophis to generate a detailed model of Apophis' gravitational field. These three models can then be joined to determine the center of gravity and associated gravity potential parameters. Our assumption is that this process, already established during the NEAR mission, can be relied upon to produce highly accurate measurements from the Foresight spacecraft to Apophis' center of gravity and its center of mass.

Locating Foresight with Respect to the Earth

Once the center of mass and spin axis of the asteroid have been determined, the spacecraft enters the Tracking phase of the mission. From a trailing heliocentric orbit, a series of range measurements from the asteroid to the spacecraft are used in conjunction with the DSN to predict the range from the Earth to the Foresight spacecraft. While DSN can provide reasonably accurate measurements of deep space spacecraft position and velocity, its accuracy based on a single measurement or two is insufficient to determine the orbital parameters to the precision required for this mission. Our analysis indicates that at a given epoch on the 2017 timeframe, the position and velocity of Foresight (and therefore Apophis' center of gravity) *must be known to within tens of meters and thousandths of a millimeter per second in velocity* in order to reduce the error ellipse in 2029 to the required accuracy. A single DSN measurement has 1σ accuracies of roughly 2 meters in the range direction (quite good), but around 2.5 nanoradians in transverse sweep angle [6]. At 1 AU, this transverse sweep angle inaccuracy corresponds to a transverse position inaccuracy on the order of kilometers, orders of magnitude larger than the range inaccuracy. Spacecraft velocity is often estimated over two DSN measurements, differencing position over the time between the measurements. This technique results in velocity accuracies on the order of millimeters per second, but does not produce the velocity accuracies necessary to solve the present problem. The use of two DSN ground stations (the so-called Delta-DORS or Delta Doppler range observation technique) is likewise insufficient to provide the necessary accuracies with a single measurement or two. Our approach is to conduct a series of measurements over a long time arc (many days) in order to produce the necessary accuracy. Since range is the most accurate measurement available from the DSN, our technique only relies on the use of range information (distance along the sight line). A series of range measurements can be used as the basis of a batch filter algorithm (like a one-time Kalman filter) to accurately predict the orbital state at the beginning of the measurement arc. The following sections outline our approach toward modeling the orbital mechanics and establishing both the required frequency of DSN range measurements and the duration of the measurement arc.

Approach to Error Ellipse Reduction

Orbit Propagation and Error Ellipse Estimation

In order to model this problem, internal software tools were developed by the Foresight team to accurately propagate Apophis's orbit state and predict the uncertainty in Apophis' state as a function of number of measurements and time between measurements. An 8th/9th order n-body numerical propagator with a variable step size was used to propagate the actual and dispersed orbits of Apophis forward from a given state and epoch. The Sun, all of the planets and the Earth's moon are considered in the gravitational model. The perturbing effects of the large asteroid-belt asteroids Ceres, Pallas, and Vesta are also included. Solar pressure and the Yarkovsky effect are not modeled, but their associated uncertainties are addressed in the analysis. For a given starting condition, the propagator's step-wise integration tolerances were set so that results for position accuracies were on the order of a few meters in 2029. For the time period considered, Apophis' orbit must be propagated from a starting state for approximately 12 years. This period includes a relatively close approach to Earth in 2021. The propagation eventually reaches the particular close approach in April of 2029 that might result in Apophis passing through a "keyhole" on the b-plane that will change its orbit to a 7:6 resonant return and impact in 2036. The accuracy of this numerical propagator was verified against published data and similar sample cases.

Selected Measurement Frequency and Duration

Based on our simulations (as seen in Fig. 4a), we propose to baseline a weekly measurement frequency for a period of 300 days (approximately 43 DSN range measurements). This solution represents a reasonable compromise between orbit determination accuracy, mission cost, and operational complexity. After taking measurements for 300 days, the error ellipse is reduced to less than 6 km, (+/- 3σ). Adding a full kilometer for Yarkovsky effect and solar pressure acceleration uncertainties, the error ellipse is still well within the desired 14 km target. Note that contribution of error (not just shifting the b-plane mean, but the spread of errors) from these two effects, if not already sufficiently reduced by 2017, can be further reduced from an application of the same batch filtering technique whereby parameters associated with the magnitude and direction of those effects can be added to both the propagation model as well as the optimization variable set. Note also that there is some margin in selecting a 300-day measurement arc. Should the mission be delayed or suffer a failure late in the mission, the goal of reducing the error ellipse +/- 3σ distance (i.e. 6 times sigma) to under 14 km might still be satisfied.

Before and After Mission Distributions on the 2029 Encounter b-plane

Viewed in the 2029 b-plane centered on Earth, the proposed mission has a significant effect on reducing the error ellipse. The new error ellipse prediction appears in Fig. 4b, along with the original error ellipse, for comparison. Axes are square in both the zoomed in and zoomed out views of the error ellipse. The "After Mission" error ellipse is shown in green superimposed on the initial distribution. The “Initial” distribution is obtained by propagating the current state of Apophis (circa 2005) with random normal distributions on the orbital elements taken from JPL’s Sentry database on Apophis using our internally developed orbital propagator. Note that this Initial orbit does not account for any subsequent optimal or radar observations. As discussed previously, the “After Mission” distribution likewise does not account for additional Earth-based observations.

While our goal is strictly to reduce the error ellipse in 2029 and not establish directly whether Apophis will pass though the 7:6 keyhole, it is clear that information such as that calculated for our proof-of-concept would be invaluable to future decision makers. Once a precise orbit is determined in early 2017, future decision makers will be able to plot this “actual” data on a 2029 b-plane to determine if it overlaps the 7:6 keyhole and to what degree. Should a significant danger exist, there would still be sufficient time to mount a deflection mission or take appropriate mitigation steps.

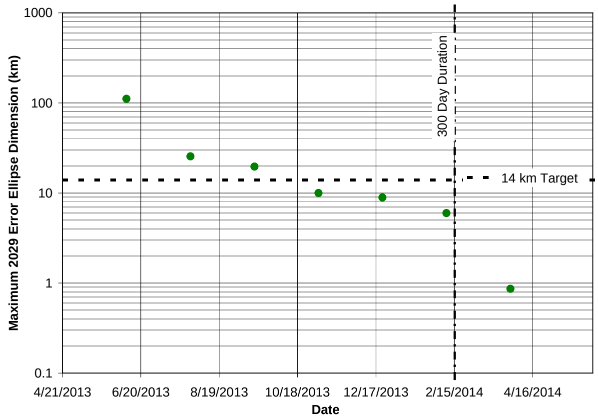

Fig. 4a. Error Ellipse Reduction for Target Mission (With Fine Monte Carlo).

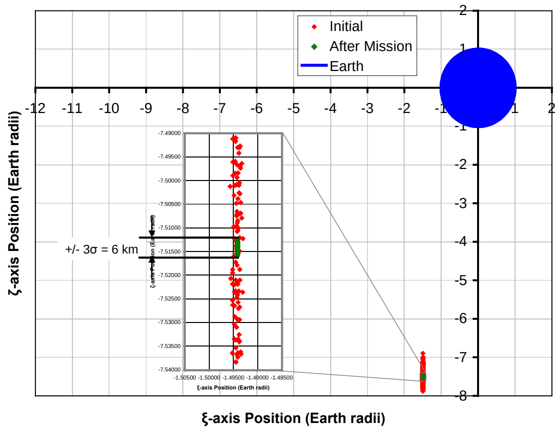

Fig. 4b. Initial and Final B-Plane Error Ellipse Comparison.

SPACECRAFT DESIGN

The overall design philosophy was to achieve the mission goals with a spacecraft designed for minimal life cycle costs, emphasizing a simple, streamlined approach using off-the-shelf components whenever possible. A minimalist approach was employed when selecting scientific payloads and subsystem components for the spacecraft. However, redundancy of critical hardware components was incorporated. A mass breakdown statement for the Foresight Encounter Spacecraft (ES) and Propulsive Transfer Vehicle (PTV) are presented in Tables 1a and 1b.

All of the subsystems inside the ES are enclosed in an 85 cm x 85 cm x 70 cm box wrapped in multilayer insulation (MLI) and a reflective foil to maintain thermal control. In addition, the spacecraft contains several small resistive Kapton heaters to keep component temperatures from dropping too low. Primary power for the spacecraft is provided by two solar arrays. These arrays are folded against the sides of the vehicle during launch. A simple, 1-axis locking hinge mechanism is used to rotate the solar arrays 90º to their fixed body-deployed position once the spacecraft has departed LEO. The solar arrays are augmented by a bank of six Li-Ion batteries. Maximum spacecraft power required during the mission is 280.6 W, with an overall end-to-end solar array efficiency of 14.3% at the end of three years. The spacecraft has a pair of arrays but can accomplish the baseline mission with only one array. Communications to Earth are performed through a fixed high gain antenna and two low gain antennas. The Sun and Earth are always on the same side of the spacecraft, though the Earth’s position varies from that of the Sun by up to 90º. The high gain antenna was therefore positioned to point in the same direction as the solar arrays to allow the solar arrays to continuously provide some amount of power during high data rate transmissions via the high gain antenna, augmented by the batteries.

The Encounter Spacecraft (ES) propulsion system includes a main engine, six maneuvering thrusters, four propellant tanks, a pressurant tank, and associated feed lines and valves. The main engine performs the two main burns for Earth departure and Apophis arrival. On the same spacecraft body face as the main engine are four reaction control system (RCS) thrusters serving to provide attitude control during these maneuvers. The PTV is a simple upper-stage type bi-propellant rocket stage used to propel the Foresight vehicle onto its Trans-Apophis Trajectory. The main engine of the PTV is an Aerojet R-40B bipropellant spacecraft propulsion engine using identical propellants as the ES (MMH/NTO) with four sets of quad Aerojet 21 bipropellant RCS thrusters. During the two main engine burns, attitude control of the

ES is provided by the four RCS thrusters. Otherwise, attitude control is maintained by four reaction wheels. Attitude determination is performed with a combination of measurements from Sun sensors, star trackers, and the fiber-optic gyros (FOGs) in the IMU, integrated by an Extended Kalman Filter (EKF) [7].

There is a minimum suite of science instruments in this mission. Inclusion of any instruments is predicated upon their necessity to the overall science objective, with a view toward limiting mission cost and complexity. Thus, there are two primary instruments on the spacecraft: the Advanced Imagery Mechanism (AIM) and the Laser Altimeter Device (LAD). The primary science goal of the AIM is to develop surface imagery of the asteroid Apophis that will aid in determination of size, shape, and surface/regolith features. Data from AIM will also assist in development of asteroid mass estimates. The AIM will be assumed to have heritage from previous spacecraft missions, specifically based upon the framing camera of the Dawn asteroid mission [8]. The AIM consists of a camera head, CCD, filter wheel, and associated electronics. Radiation hardened optics will pass incoming light through a filter wheel onto a 1024x1024 CCD. Focal length is 150 mm, Field of View (FOV) is 5.5 degrees yielding a resolution of 9.3 m/pixel at a distance of 100 km. A multi-spectral filter wheel will be on the instrument. Anticipated mass for the camera is 5 kg, consuming 12 W. The primary science goal of the LAD is to use reflected laser measurements from asteroid Apophis to determine distance, shape, internal structure (when combined with accurate spacecraft trajectory data), and surface topography. The laser altimeter (or LIDAR) will use a neodymium-yttrium-aluminum-garnet (Nd-YAG) laser. This laser altimeter system will be similar to other instruments used on previous spacecraft such as NEAR Laser Rangefinder (NLR) on NEAR, Light Detection and Ranging Instrument (LIDAR) on Hayabusa, and the Lunar Orbiter Laser Altimeter (LOLA) on the Lunar Reconnaissance Orbiter (LRO) [9, 10, 11]. The range requirement for the LAD will be 50 km and designed for low albedo, diffusely reflecting surfaces present on asteroids [11]. Both the AIM and the LAD will be boresighted such that visual imagery and laser ranging data can complement each other. It is anticipated that the LAD instrument will be able to fire multiple times per minute, returning and firing several million times over the course of the instrument's lifetime. Anticipated mass for the laser altimeter is 5 kg, consuming 20 W (peak) and 15 W (average).

No.	Name		Element Mass [kg]	Subsystem Mass [kg]
1.0	**Structures and Mechanisms**			**26.0**
		Primary Structure	14.8	
		Secondary Structure	5.5	
		Fuel Tank	2.3	
		Oxidizer Tank	2.3	
		Pressurant Tank	0.7	
		Solar Array Support Structure	0.2	
		Solar Array Actuators	0.2	
2.0	**Propulsion**			**7.0**
		Main Engine: Aerojet 445	1.9	
		Main Engine Feed Lines	1.9	
		Maneuvering Engines: Aerojet 2 (x6)	1.6	
		Maneuvering Engine Feed Lines	1.6	
3.0	**Thermal Control**			**5.0**
		Reflective Foil	1.0	
		Multi-Layer Insulation	3.8	
		Heaters	0.2	
4.0	**Power**			**12.1**
		Batteries: Saft VES 180 (x6)	6.7	
		Solar Array: Spectrolab Triple Junction	2.1	
		Power Distribution Unit	1.7	
		Power Cabling	1.6	
5.0	**Command and Data Handling**			**4.9**
		CPU: PowerPC 750FX	0.1	
		Memory: Samsung 64 GB Solid State Drive (x2)	0.1	
		Electronics Module	1.0	
		Wiring	3.7	
6.0	**Attitude Determination and Control**			**12.9**
		Sun Sensors: AeroAstro MSS (x6)	0.4	
		Star Sensors: Terma HE-5AS (x2)	4.4	
		Reaction Wheels: Dynacon MicroWheel 1000 (x4)	6.6	
		Inertial Measurement Unit: LN-200S	1.5	
7.0	**Communications**			**8.9**
		High Gain Antenna	3.0	
		Low Gain Antenna (x2)	0.7	
		Small Deep Space Transponder	2.9	
		Multi-Mode S-Band Transceiver	2.3	
8.0	**Margin (20%)**			**13.6**
9.0	**Dry Mass**			**90.2**
10.0	**Consumables**			**120.1**
		Fuel: MMH	45.1	
		Oxidizer: NTO	74.4	
		Pressurant: He	0.7	
11.0	**Wet Mass**			**210.3**
12.0	**Payload**			**10.0**
		Advanced Imagery Mechanism (AIM)	5.0	
		Laser Altimeter Device (LAD)	5.0	
13.0	**Gross Mass**			**220.3**

Table 1a. Foresight Encounter Spacecraft (ES) Mass Breakdown Statement

No.	Name		Element Mass [kg]	Subsystem Mass [kg]
1.0	**Structures**			**133.6**
		Primary Structure	18.8	
		Secondary Structure	13.9	
		Payload Adapter	5.5	
		Fuel Tank	22.5	
		Oxidizer Tank	22.7	
		Pressurant Tank	50.2	
2.0	**Propulsion**			**31.8**
		Main Engine: Aerojet R-40B	6.8	
		Main Engine Feed Lines	6.8	
		RCS Engines: Aerojet 21 (x16)	9.1	
		RCS Engine Feed Lines	9.1	
3.0	**Thermal Control**			**12.1**
		Reflective Foil	2.4	
		Multi-Layer Insulation	9.7	
4.0	**Power**			**6.2**
		Batteries: Saft VES 180 (x2)	2.2	
		Power Distribution Unit	0.9	
		Wiring	3.1	
5.0	**Command and Data Handling**			**1.6**
		Spacecraft Control Computer	0.1	
		Electronics Module	1.0	
		Wiring	0.5	
6.0	**Attitude Determination and Control**			**1.5**
		Inertial Measurement Unit: LN-200S	1.5	
7.0	**Margin (20%)**			**37.4**
8.0	**Dry Mass**			**224.2**
9.0	**Consumables**			**1,163.1**
		Fuel: MMH	426.1	
		Fuel Reserves / Residuals	10.5	
		Oxidizer: NTO	702.7	
		Oxidizer Reserves / Residuals	17.3	
		Pressurant: He	6.5	
10.0	**Wet Mass**			**1,387.3**
11.0	**Payload**			**220.3**
		Foresight Spacecraft	220.3	
12.0	**Gross Mass**			**1,607.6**

Table 1b. Propulsive Transfer Vehicle (PTV) Mass Breakdown Statement

As seen in Table 2, many of the subsystems will be based upon flight systems and experience of satellite/subsystem developer and project team member SpaceDev, Inc. Where available, flight proven subsystems will be utilized in order to provide higher certainty of mission success. In addition, manufacturers of the relevant subsystems have been identified where possible.

Component	Name	Manufacturer	No. on ES	No. on PTV	Specifications
Propulsion					
S/C Main Engine[22]	Aerojet 445	Aerojet	1	0	Thrust (vac): 445 N, Isp: 309 s, T/W: 24.39
S/C RCS[23]	Aerojet 2	Aerojet	6	0	Thrust (vac): 2 N, Isp: 265 s, T/W: 0.75
PTV Main Engine[24]	R-40B	Aerojet	0	1	Thrust (vac): 4000 N, Isp: 293 s, T/W: 56.4
PTV RCS[25]	Aerojet 21	Aerojet	0	4	Thrust (vac): 21 N, Isp: 285 s, T/W: 3.81, Quad configuration
Thermal Control					
Heaters[26]	Kapton Heaters	Minco	16	0	-200 to 200ºC range, Kapton/FEP material
Power					
Batteries[21]	VES 180	Saft	6	2	Li-Ion space technology, specific energy: 165 Wh/kg, storage: 180 Wh each
Solar Array[22]	Triple Junction	Spectrolab	2	0	GaInP2/GaAs/Ge, BOL power: 289 W/m2, BOL efficiency: 22.5%, EOL power: 256 W/m2, 4% degradation per year
Distribution	PDU	SpaceDev	1	1	16 5-Amp high side relays, Integrated 200-W Li-Ion battery charger, 96 12-bit ADCs, Digital solar array peak power tracking
Command and Data Handling					
CPU[27]	PowerPC 750 FX	IBM	1	0	RISC Microprocessor, 1856 MIPS at 800 MHz with 256 MB RAM, RS-422 / USB / Ethernet compatible
Memory[28]	16 GB SSD	Samsung	2	0	NAND-based SSD, read rate: 57 MBps, write rate: 32 MBps
SCC[29]	8051	Silicon Labs	0	1	1000 MIPS @ 100 MHz, 128 KB Flash, 8448 bytes data RAM, 8 12-bot ADCs, 2 12-bit DACs
Attitude Determination and Control					
Sun Sensor[30]	MSS	AeroAstro	12	0	60º FOV, accuracy ±1º
Star Tracker[31]	HE-5AS	Terma	2	0	22º FOV, <1 arcsec cross-track accuracy, 5 arcsec boresight accuracy
Reaction Wheel[32]	MicroWheel 1000	Dynacon	4	0	Produce 30 mNm torque, hold 1000 mNm angular momentum, mounted with 1 each on XYZ axes and 1 on skew axis
IMU[33]	LN-200S	Northrop Grumman	1	1	Fiber Optic Gyro, silicon accelerometers and electronics
Communications					
Low Gain Antenna[34]	Custom	Ball Aerospace	2	0	S-Band, 50 bps data rate
High Gain Antenna[34]	Custom	Ball Aerospace	1	0	X-Band, 17 kbps data rate at 0.5 AU, SNR: 3, efficiency: 55%
X-Band Transponder[35]	SDST	General Dynamics	1	0	DSN Compatible, X-Band transmit and receive, 2.0 dB Noise Figure, -157.7 dBm Receiver Threshold, 10 ns Ranging Delay Variation, 0.5 ns Carrier Delay Variation
S-Band Transceiver[36]	Multi-Mode S-Band Transceiver	General Dynamics	1	0	DSN Compatible, S-Band transmit and receive, < 2.5 dB Noise Figure, Delay Variation, 0.5 ns Carrier Delay Variation

Table 2. Foresight Subsystem Component Specifications

Reliability and Cost Assessment

A high probability of Foresight mission success is achieved due to simplicity of design, the use of proven components, and design redundancy. Reliability analysis results for the Foresight mission yield a mean mission success probability of 90.2 percent. Stated alternatively, Loss Of Mission (LOM) is expected to take place only once every 10.2 missions. The 90th percentile value for mission success is 89.4 percent. Mission success, for the purposes of this reliability analysis, was defined as proper functioning of the launch vehicle, PTV, spacecraft, and instrument set to determine Apophis' orbit. Fig. 5 shows the resulting probability distribution for mission success. Reliability analysis was performed using a NASA standard fault tree and event sequence diagram approach as described in the NASA Probabilistic Risk Assessment Procedures Guide [12]. Data sources for component failure rate values include commercial manufacturer publications, personal contact with manufacturers, SEI internal models, and technical papers of NASA and other professional organizations.

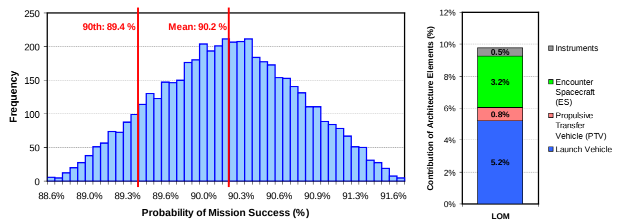

Fig. 5. Histogram of Reliability (20,000 Monte Carlo Trials).and Contribution of Architecture Elements to LOM.

Cost Element Name	DDT&E [FY2007]	Acquisition Cost [FY2007]	Total Cost [FY2007]
Spacecraft Stages	**$57.85 M**	**$23.74 M**	**$81.59 M**
Foresight Encounter Spacecraft	$23.51 M	$14.77 M	$38.28 M
Propulsive Transfer Vehicle	$34.34 M	$9.27 M	$43.61 M
Scientific Instruments (2)*	**-----**	**$6.30 M**	**$6.30 M**
Operations	**-----**	**$20.99 M**	**$20.99 M**
Operations: Flight	-----	$6.80 M	$6.80 M
Operations: Nav&Track (DSN)	-----	$3.09 M	$3.09 M
Operations: Science	-----	$1.70 M	$1.70 M
Operations: On-Station	-----	$9.40 M	$9.40 M
Launch Vehicle: Minotaur IV	**-----**	**$22.00 M**	**$22.00 M**
Total	**$57.85 M**	**$73.03 M**	**$130.88 M**

* Note: AIM cost = $3.71 M and university-developed LAD cost = $2.59 M

Table 3. Life Cycle Cost Statement

A life cycle cost assessment has been performed for the Foresight concept design. The life cycle cost presented here includes Design, Development, Testing and Evaluation (DDT&E), acquisition, operations, and launch vehicle costs. All costs are expressed in FY2007 constant U.S. dollars unless otherwise noted. Launch vehicle costs for the Minotaur IV were assumed to be $22M [13]. A protoflight development plan was assumed, resulting in minimal System Test Hardware (STH). Extensive heritage was assumed for several subsystems including reaction control, attitude control, command and data handling, and thermal control. Existing engines were assumed for the PTV and Foresight spacecraft (resulting in no development cost). The life cycle cost for this mission (including development and 4 years of operations) is estimated to be $130.88 M (FY2007). Operations cost accounts for approximately $21 M, a similar amount to the launch vehicle cost (see Table 3). This spacecraft's cost is similar to the costs for historical NASA Discovery class missions and particularly other previous asteroid missions.

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