BENEFITS OF APPLICATION OF ADVANCED TECHNOLOGIES FOR A NEPTUNE ORBITER, ATMOSPHERIC PROBES, AND TRITON LANDER

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ABSTRACT

Missions with planned launch dates several years from today pose significant design challenges in properly accounting for technology advances that may occur in the time leading up to actual spacecraft design, build, test and launch. Conceptual mission and spacecraft designs that rely solely on off the shelf technology will result in conservative estimates that may not be attractive or truly representative of the mission as it actually will be designed and built. This past summer, as part of one of NASA's Vision Mission Studies, a group of students at the Laboratory for Spacecraft and Mission Design (LSMD) have developed and analyzed different Neptune mission baselines, and determined the benefits of various assumed technology improvements. The baseline mission uses either a chemical propulsion system or a solar-electric system. Insertion into orbit around Neptune is achieved by means of aerocapture. Neptune's large moon Triton is used as a tour engine. With these technologies a comprehensive Cassini-class investigation of the Neptune system is possible. Technologies under investigation include the aerocapture heat shield and thermal protection system, both chemical and solar electric propulsion systems, spacecraft power, and energy storage systems.

Key words: Neptune; Triton; Aerocapture; Technology.

1. INTRODUCTION

The aim of this study is to quantify the benefits of advanced technology for a Cassini-class mission to Neptune. This data would be useful in guiding a technology development program leading up to such a mission. Given a price tag of almost \$3B and a launch date beginning in 2017, this mission would likely have its own technology development program.

The requirements for this mission are that it cost no more than \$5B, that a nuclear reactor not be used, that it must be launchable by a Delta IV Heavy or lesser rocket, and that the trip time not exceed 12 years.

Our methodology was to construct a baseline mission satisfying these requirements. For each technology area under study, the baseline was reevaluated assuming a reasonable range of improvement in the technology. The benefits were quantified in terms of launch mass, and where possible, cost. The cost estimates do not, however, include the cost of developing the technology to the specified level. This was beyond the scope of our study. Cost estimates were made using the 2003 JPL (Jet Propulsion Laboratory) Cost Model.

1.1. Science Goals

Current models suggest that Uranus and Neptune have similar compositions and histories, and that exploring either one will yield useful information about the other, and about the primordial solar system. Triton is believed to be a Kuiper belt object captured by Neptune. Thus a mission to Neptune would also gather information about the Kuiper belt. For this reason Neptune is considered a more desirable target than Uranus, despite the distance.

The Neptune system has 4 major targets of investigation: the planet, the rings, the magnetosphere, and Triton. The

I. Neptune	1.	Measure the composition of Neptune's deep atmosphere
	2.	Measure the thermal structure of Neptune's deep atmosphere
	3.	Measure the winds of Neptune's deep atmosphere
	4.	Image the entire planet at various spatial locations and times
	5.	Spectrally image the planet in UV to far-IR at various locations and times
	6.	Measure the three-dimensional structure of the magnetic field
	7.	Measure the three-dimensional structure of the gravitational field
	8.	Measure atmospheric properties of upper atmosphere
II. Triton	1.	Image Triton globally at high resolution (100 m)
	2.	Image areas of Triton surface at very high resolution (10 m)
	3.	Spectrally image surface in UV to far-IR for surface composition (100 m)
	4.	Measure magnetic field of Triton
	5.	Measure gravitational field of Triton
	6.	Measure atmospheric properties of Triton
	7.	Examine relevant geologic properties, including plumes and surface features
	8.	Map surface temperatures of Triton
III. Rings	1.	Image rings at high resolution (100 m) and determine orbital characteristics of rings
	2.	Image minor satellites and determine orbital characteristics of satellites
	3.	Image ring arcs in UV to far-IR at high resolution (100 m)
	4.	Image Proteus, Larissa, and Nereid in UV to far-IR at high resolution (100 m)
	5.	Determine composition of large ring bodies and minor satellites
	6.	Determine ring particle size and composition
	7.	Determine composition and mass of Proteus, Larissa, and Nereid
	8.	Measure magnetic fields produced by ring bodies or minor satellites, if any
IV. Magnetosphere	1.	Observe magnetosphere at various spatial locations and times
	2.	Determine composition, energy, temperature, and distribution of particles trapped in magnetosphere

Table 1. Measurement objectives

measurement objectives of the mission are presented in Table 1.

1.2. Model Overview

The mission is modeled using ICEmaker (Integrated Concurrent Engineering), a software tool developed at the LSMD. It is a medium fidelity model. The spacecraft is modeled at the component level, with components inherited or extrapolated for predicted technology advances. Components are sized according to first principles subject to reasonable approximations. For example, the structural bus is modeled with rules of thumb based on continuous mechanics, not finite element analysis. The thermal balance is based only on radiative calculations, with margins to accommodate conduction through the bus. Orbital mechanics are modeled as a series of two-body problems, but the SEP trajectory is selected from a set of trajectories developed at the Jet Propulsion Laboratory (JPL) for NASA's In Space Propulsion (ISP) program. Contingency is applied at the system level, based on standard AIAA mission classes. Aerocapture is not modeled computationally. Aerocapture parameters were estimated based mainly on [3], [4], and [15].

2. BASELINE OVERVIEW

The baseline mission consists of four modules: a Neptune orbiter, an atmospheric probe, a Triton lander, and a SEP (Solar Electric Propulsion) carrier. The total wet mass without contingency is 4224 kg. This is launched into a 10.26 year trajectory with a 4 year science tour at Neptune and Triton. Insertion is accomplished by aerocapture. Further details of the baseline are covered by subsystem below.

2.1. Mission Design

A Boeing Delta IV launch vehicle lifts the spacecraft to a C3 of 18436000 m^2/s^2. The mission then uses solar electric propulsion with a VJGA (Venus Jupiter Gravity Assist) to reach Neptune in 10.26 years. The SEP engines are shut off at 3 AU (Astronomical Units), but the SEP module is retained until just prior to insertion, to carry the probe and a downlink antenna. 5 months prior to insertion, the probe is released from the carrier. The probe enters Neptune's atmosphere and relays its data to Earth through the carrier just before aerocapture.

Aerocapture takes place with an entry velocity of 22 km/s and a Δv of 6667 m/s . Peak deceleration is 22 g . The design uses a slender body ellipsled aeroshell. The mass fraction of the aeroshell was assumed to be 28% of the entry mass [3]. In light of more recent studies such as [4], a mass fraction of 44% would be more realistic for current TPS (Thermal Protective System) technology. However, materials advancements could reasonably lower this to 36% and current estimates of trailing ballute aeroshells are much lower.

The science phase of the mission lasts for 4 years, during which the orbiter will make a flyby of Triton once every 12 Earth days. It will use Triton as an engine to increase the inclination of its orbit from near 0° to ∼75°.

The orbiter releases the Triton lander prior to one of the flybys. The lander uses chemical rockets to guide its descent with a Δv of 1125m/s. No aeroshell is used for the lander. The lander will survive on the surface for 8 hours while relaying its data to the orbiter.

2.2. Thermal

Because of the wide range of thermal environments, from Venus to Neptune, the spacecraft was designed for a slight cold bias at Neptune where the thermal environment is the most stable, and the orbiter is operating at its highest power levels. The craft is designed for a target operating temperature between 285 K and 308 K.

The resulting configuration uses a moderate heater array with a total of 145 RHUs at 1 W each, supplemented with 75-100 W of cartridge heaters for colder areas of the spacecraft.

The orbiter uses a thermal coating with emissivity in the range of 0.1 - 0.07 (anodized titanium, some vapor deposited metals). The solar absorptivity is not a driving factor in Neptune orbit. For transit, the aeroshell and SEP carrier stage use a Ag-AlO overcoat ($A = 0.08, E = 0.19$) due to its low dependence on solar and IR radiation to maintain temperature while still keeping the spacecraft warm enough during eclipses and ballistic cruise beyond Jupiter.

The orbiter and SEP stage are also equipped with deployable heat pipe radiators totaling 9 m^2 coated with MgO/AlO white paint ($A = 0.09, E = 0.92$). Additionally, the power processors are mounted on the outer surface SEP stage with 0.4 m^2 of fixed radiator area per unit.

A Freon-12 pumped fluid loop is used to transport heat from the RTGs to either the interior of the spacecraft or to the radiators (modeled after the system used on MER). Internal orbiter and SEP components and tanks are wrapped with up to 7 kg of multi-layer mylar insulation (MLI).

The atmospheric probe uses MLI on the body, as well as a blunt conical heat shield with backshell for Neptune entry, with 28 passive RHUs for internal heating.

The Triton lander, because of the extreme cold environment of Triton (34 K) uses a 1 cm layer of aerogel where possible (weight < 0.001 kg).

2.3. Propulsion

The orbiter propulsion system serves mainly to provide trajectory corrections and manuevers throughout the mission. Upon reaching Neptune, the orbiter propulsion system puts the spacecraft into the proper entry trajectory and performs the periapsis raise maneuver. Within the Neptune system, it changes the orbit's plane from equatorial to polar by using Triton as a cranking engine. Combined, these manuevers require a Δv of 1770 m/s. This is provided with a dual mode, N_2O_4/Hydrazine propsulsion system. The orbiter has a single 5 kg thruster capable of 445 N thrust, an analog to a TRW DMLAE (Dual Mode Liquid Apogee Engine). The propulsion requirements are met with 277 kg of hydrazine and 363 kg of N_2O_4. Two tanks are used to store the main and ADACS (Attitude Determination and Control) propellant assuming a PV/W figure of 10,000 m. To maintain proper pressure levels in these tanks, roughly 3.2 kg of pressurant and an 18.7 kg pressurant tank are also present. In addition, the propulsion system uses another 32 kg of support components (plumbing, pressure transducers, etc.).

Because the Neptune probe uses a passive attitude control system, it has no need for a propulsion system.

The Triton lander is ejected from the spacecraft on a Triton approach while within the Neptune system. Its propulsion system slows the lander to a point several meters above the Triton surface, at which point the lander will drop and soft-land on its compressible landing pads. This sequence of manuevers requires a Δv of 1125 m/s. The lander uses a monopropellant system with a single 5 kg thruster capable of 44.5 N of thrust. Assuming an ISP of 285 s, 56.5 kg of propellant (hydrazine) is needed to meet this requirement. Also, roughly 0.3 kg of pressurant are used to maintain proper storage of the hydrazine in its tank. Using the same PV/W as before of 10,000 m, two 1.7 kg tanks are used to store the hydrazine and pressurant. Propellant lines, propulsion system management, and other support components add 2.8 kg of mass to the system.

2.4. Telecom

The science instruments included in the baseline require an average transmission rate of 164 kbps from Neptune, assuming 8 hours per day of downlink time is available. To meet this goal the orbiter carries a 3.6 m Ka-band dish antenna. It broadcasts with 98 W RF (Radio Frequency) power and an antenna efficiency of 65%. The data is encoded with a rate $1/2$ turbo code requiring $E_b/N_o = 0.8$ for a BER (Bit Error Rate) of 10^{-5} . The ground station was assumed to be a 70 m DSN (Deep Space Network) antenna with 70% efficiency, but an additional 3 dB increase in gain was assumed to account for planned upgrades scheduled to be complete well before Neptune insertion [12]. An omnidirectional emergency antenna is not included because even at Ka band with 98 W RF power (a dubiously possible power level), the maximum achievable data rate is approximately 1 bps.

The orbiter includes a smaller 1.2 m S band antenna for communication with the Triton lander. The lander uses a wide-angle antenna with 1.4 W RF power for uplink and no downlink. It has no active pointing system, and the design assumption is that it can passively point the antenna to within 45° of the orbiter during its short life.

The atmospheric probe uses a similar S band wide angle antenna with 45° pointing accuracy, but it broadcasts at 3.5 W RF power. Since the orbiter is still within its aeroshell at this point, the carrier stage includes a 2 m S band antenna to relay the probe data. S band is used as opposed to a higher frequency to reduce atmospheric losses.

The link between the carrier stage and earth is accomplished by means of a 1.3 m X band antenna with 2.5 W RF power for downlink.

2.5. C&DH

The C&DH (Control and Data Handling) system was modeled in low fidelity. No improvements in this area were considered, since it is believed that the private sector will substantially develop C&DH technologies without NASA's help. The orbiter used 2 Harris RH-3000 computers for redundancy and 24GB of flash memory from SEAKR. The carrier module shared the orbiter's C&DH subsystem. The probe used 1 Harris RH-3000 computer and needed no external storage. The lander used 1 Harris RH-3000 computer with 768 Mb of external flash memory.

2.6. Power

The spacecraft has been designated both an average and peak power during each of 8 mission phases. The primary driver for the power system is the last phase, Science, both due to larger peak power requirements, and power decay associated with radioisotope power sources.

The science phase is tabulated with a peak power of 895 W and an average power of 685 W (including battery charge), both including a 40% contingency factor. Secondary batteries reduce the maximum power load by up to 82 W with contingency, bringing the power supplied by the RTGs to 813 W end-of-life.

The beginning-of-life (BOL) power requirement, given ~14 years of mission time, is met by 8 advanced stirling RTGs, producing a total power of 992 W, with a total weight of 128 kg (124 W and 16 kg each). The orbiter carries a 15 kg secondary lithium-ion battery for load distribution in Neptune orbit.

The SEP stage is equipped with 77.5 m^2 of quad-junction solar arrays to meet the specified trajectory maximum power of 31 kW BOL at 1 AU, with a weight of 240 kg. Additionally, the SEP stage carries 21 kg of primary lithium thionyl-chloride batteries to power both the orbiter and SEP stage during launch, until the RTGs are brought online.

The atmospheric probe is powered during its descent by 14 kg of lithium thionyl chloride batteries. During cruise, the probe is connected to the orbiter power system via an umbilical connection. The Triton lander carries 17 kg of batteries, and is also powered by the orbiter during transit.

2.7. ADACS

The pointing control requirement during cruise is driven by the pointing requirements of the SEP stage antenna. Near Neptune, the pointing accuracy needed is ~.2°. During this phase of the mission, attitude control is provided by a set of twelve .22 N hydrazine thrusters on the SEP stage. The SEP stage also carries a full complement of attitude control sensors, including 3 sun sensors and 3 star trackers. Inertial measurements are provided by gyroscopes and accelerometers (in an IMU) within the orbiter. Major trajectory control maneuvers can be accomplished by altering the thrust direction of the gimbaled NEXT ion engines.

After the SEP stage disengages, the orbiter performs aerocapture. Altitude control is necessary during aerocapture to compensate for uncertainties in atmospheric density and to maintain an acceptable aerocapture corridor (i.e., to not go so low into the atmosphere that the spacecraft burns up, or so high in the atmosphere that the spacecraft does not successfully capture into an appropriate Neptune orbit). Control is provided by six 70 N thrusters piercing the cooler side of the elipsled aeroshell. Venting hydrazine from these thrusters should reduce the heat conducted to the spacecraft through the metal plumbing. The expected heating rates were not calculated, however.

After aerocapture, the aeroshell is shed, exposing a set of sixteen .22 N thrusters that provide full 3-axis control in a perfect couples configuration (the thrusters fire in pairs, such that there is no net translatory motion of the spacecraft, only a net torque). The imager becomes the driver for the pointing control of the spacecraft during this phase of the mission. To satisfy this finer requirement as well as to improve pointing stability, the orbiter also carries a set of 4 reaction wheels. To maximize the duration of both science and telecom operations, the science payload is divided between two scan platforms. The power budget was sized to allow simultaneous data collection and telecom transmission, thus greatly increasing the total quantity of data taken in the mission.

2.8. Science & Instruments

The nominal science tour of Neptune is 4 years long. No science observations are made before arrival at Neptune because all instruments are enclosed within the aeroshell. All instruments are heritage or extrapolated from other missions. The total science return is 21 Tb.

The orbiter baseline includes the following instruments: Radar altimeter (Cassini), USO (Ultra Stable Oscillator) (Cassini), wide and narrow angle imager (Mars Observer Camera (MOC)), IR (InfraRed) spectrometer (Cassini Composite InfraRed Spectrometer (CIRS)), visible/near IR mapping spectrometer (Cassini Visible and Infrared Mapping Spectrometer (VIMS)), UV (UltraViolet) spectrometer (Galileo UltraViolet Spectrometer (UVS) and Cassini UltraViolet Imaging Spectrograph (UVIS)), magnetometer (Galileo and Cassini), dust instrument (Galileo Dust Detector System (DDS) and Cassini Cosmic Dust Analyzer (CDA)), plasma subsystem (Galileo and Cassini Plasma Spectrometer (CAPS)), ion detector (Galileo Energetic Particle Detector (EPD)), cosmic ray detector (Voyager Cosmic Ray System (CRS)), ion & neutral mass spectrometer (Cassini Ion and Neutral Mass Spectrometer (INMS)), plasma wave instrument (Cassini Radio and Plasma Wave Science instrument (RPWS)), energetic neutral atom instrument (Cassini Ion and Neutral Camera (INCA), gamma ray spectrometer (Near Earth Asteroid Rendezvous (NEAR) Gamma Ray Spectrometer (GRS)), and microwave radiometer (NPOESS (National Polar-orbiting Operational Environmental Satellite System) Preparatory Project (NPP) Advanced Technology Microwave Sounder (ATMS)). The average data rate for the orbiter is 167 kbps and the maximum rate is 342 kbps.

The atmosphere probe carries: Doppler wind instrument (Cassini USO), atmospheric structure package (Huygens Atmospheric Structure Instrument (HASI) and Galileo Atmospheric Structure Instrument (ASI)), net-flux radiometer (Galileo Net Flux Radiometer (NFR) and Cassini Descent Imager Spectral Radiometer (DISR)), neutral mass spectrometer (Galileo Probe Mass Spectrometer (GPMS)), nephelometer (Galileo), and radio emission detectors (Galileo Lightning and Radio emission Detectors (LRD)). The maximum data rate is 132 bps, and the total return is 570kb.

The lander carries: atmospheric structure package (Huygens HASI and Galileo ASI), mass spectrometer (Galileo GPMS), imagers (DISR), APXS (Mars Pathfinder Alpha Proton X-ray Spectrometer). The maximum data rate is 2492 bps, and the total return is 68.5Mb.

3. RESULTS

Aerocapture is an enabling technology for this mission. Using SEP injection and chemical insertion, it was necessary to eliminate the probe, orbiter, radar altimeter, dust instrument, cosmic ray detector, and energetic neutral atom instrument. The launch margin was just 17 kg with contingency. Using chemical injection and chemical insertion, we made the aforementioned sacrifices and also lengthened the cruise time to 15.84 years. We concluded that without aerocapture, we could not meet the science objectives.

Fig. 1 shows the trade space between aeroshell mass fraction and payload. Here the payload is defined as the mass of the orbiter's instruments and the entire lander, since the lander is carried until after aerocapture but the probe is not. The baseline has an aeroshell mass fraction of 28% and a payload of 472 kg. We now believe that contemporary technology is capable of no better then 40–44%. An improvement to 36% , just an 8% improvement in technology, would allow for 26kg additional payload.

Of all technology areas, the instruments have the greatest marginal mass yield. That is, a 1 kg change in instrument mass on the orbiter yields a 5.7 kg change on the mission mass, assuming that the instruments' power consumption scales with their mass. Most important are the instruments carried on the lander. There is a small-scale delta of 2.8 kg lander wet mass for every 1 kg of instruments added to the lander. This yields a 16× rollup from the lander's instruments to the mission's total wet mass.

Fig. 2 shows the effects of increased RTG energy density on the wet mass of the spacecraft. The baseline used Stirling 2.0 generators, with 7.75 W/kg. For comparison, Cassini used solid state SiGe thermoelectric generators

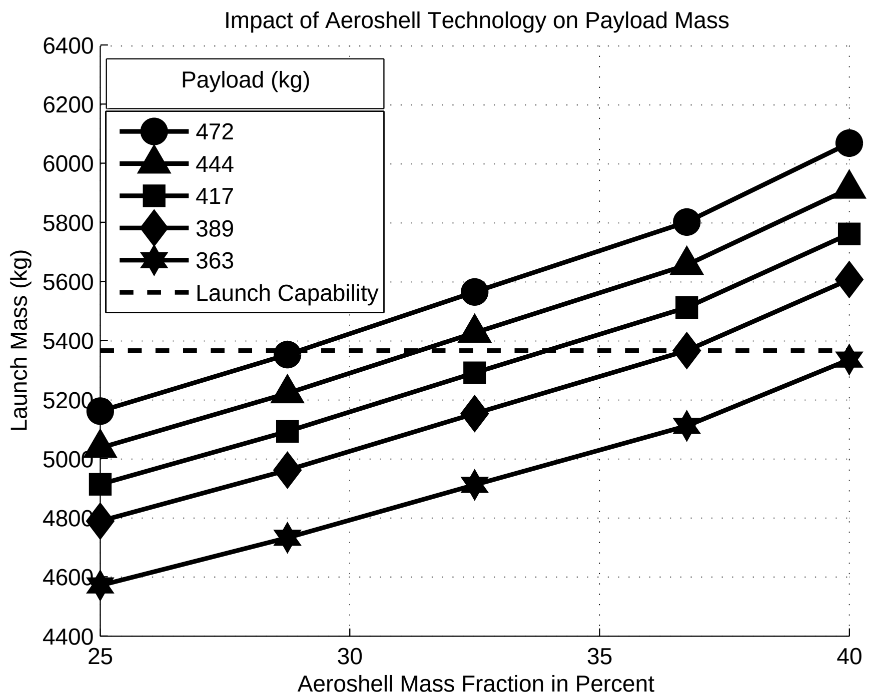

Figure 1. Allowable payload by aeroshell mass fraction.

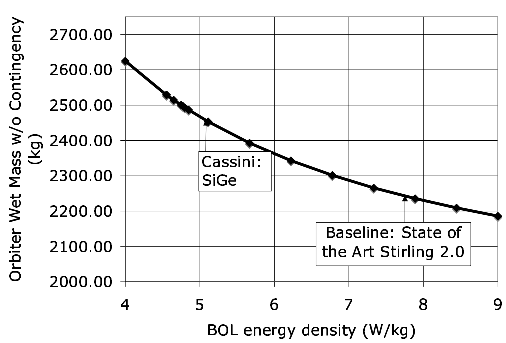

Figure 2. Effects of RTG technology on orbiter mass.

that achieved 5.07 W/kg. RTGs show a large range of potential improvement, corresponding to a mission wet mass change of ∼200kg.

The structural material was also a high yield area. 11% of the orbiter's wet mass is composed of the structural material, so this should not be surprising. The baseline uses an aluminum bus, but a graphite/epoxy composite, if manufactured properly, has the potential to reduce spacecraft structure mass by up to 66% due to its high tensile strength, high modulus of elasticity (stiffness) and low density - as demonstrated by state-of-the-art composite propellant tanks. See Fig. 3.

However, replacing the baseline's propellant tanks with advanced composites has only moderate yield. The baseline assumed a slightly conservative PV/W of 10,000 m for the tanks. State of the art composite overwrap tanks can have PV/W as high as 21,600 m, and corporations claim that they can develop PV/W to as high as 100,000 m. However, as Fig. 4 shows, there are diminishing returns in developing the tanks past 30,000 m.

Solar cell efficiency is a moderate yield area. Since the solar cells are not inserted into Neptune orbit, there is a smaller roll-up to the mission mass. Also, deficiencies in the power of the SEP stage can be accomodated by tradeoffs in mission design. For example, a VEEJSGA (Venus-Earth-Earth-Jupiter-Saturn) trajectory was found

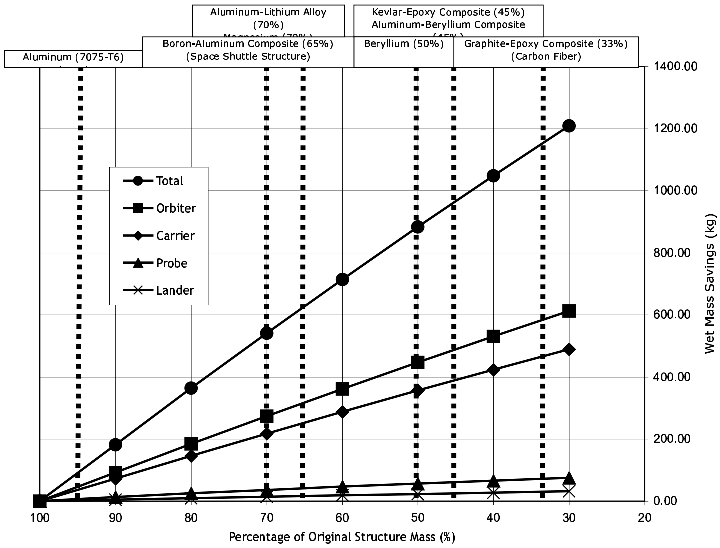

Figure 3. Advanced structural materials.

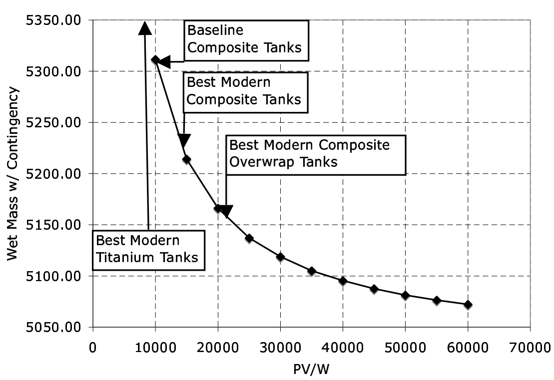

Figure 4. Effects of tank material on launch mass.

that reaches Neptune in the same time as the baseline. Using this trajectory, the same payload could be sent to Neptune with no SEP at all.

Advanced telecom technologies are also modeled. Inflatable and mesh antennas are unable to directly reduce mission mass. However, increasing antenna size allows for marked reduction in transmission power, allowing the orbiter to use 1 fewer RTG. See Table 2 for the summary. Antenna mass estimates are based on [9]. Inflatable antennas have low aperture efficiencies, complex and heavy deployment mechanisms, and little mass savings over fixed antennas for small and moderate sized antennas. Mesh antennas simply cannot achieve low areal densities at high frequencies. These limitations, coupled with the modest bandwidth requirements of the mission, lead us to believe that advanced antenna design will have minimal payoff.

However, improvements to the DSN are extremely beneficial. An increase in link time from 8–20 hours per day saves 225 kg total on the mission. This is at a nominal cost of $20M for the added time [22].

Antenna Type	Fixed	Inflatable	Mesh	Fixed
Diameter (m)	3.60	5.05	4.50	3.62
Antenna System Mass (kg)	45.4	52.3	59.3	46.7
Aperture Efficiency (%)	65	40	65	65
RF Power (W)	97.5	68.0	69.0	32.0
DSN time (h/day)	8	8	8	20
Launch Mass (kg)	5312	5234	5265	5087
Savings (kg)	0	78	47	225

Table 2. Advanced antenna technologies

4. CONCLUSIONS

We conclude that a decade from now, for less than Cassini's cost, a deep-space mission could answer key questions about Neptune, Triton, and by extension the Kuiper belt. Aerocapture and RTGs are key enabling technologies for this mission. Aerocapture in particular is in need of significant development to support this mission. TPS material, aeroshell design, and aerocapture guidance algorithms all require work. Batteries, structural composites, DSN upgrades, and RTG energy densitites are also enhancing technologies. We conclude that development in these technology areas will yield the greatest benefits to a Neptune mission in the next 13 years.

5. FURTHER STUDY

The obvious and necessary extension of this study is to estimate the R&D cost required for these advancements. This was beyond the scope of the present study, but necessary to make the results truly useful. Only in the case of DSN time was even an estimate possible. Though we did estimate mission cost with the 2003 JPL Cost Model, the model is not calibrated for nonexistent technology.

Additionally, we would like to consider RTG-powered probes and landers. An RTG-powered lander would be able to collect useful data for weeks or months. From a science standpoint, this would enable seismic studies, shedding light on Triton's geysers and internal composition. An RTG-powered atmospheric probe could float around the planet on a balloon, measuring temporal changes in the atmosphere.

Inflatable ballutes are also a tempting option for study. They have the potential to greatly reduce the mass of the aerocapture system.

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