THE NEPTUNE / TRITON EXPLORER MISSION: A CONCEPT FEASIBILITY STUDY

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ABSTRACT

Technological advances over the next 10 to 15 years promise to enable a number of smaller, more capable science missions to the outer planets. With the inception of miniaturized spacecraft for a wide range of applications, both in large clusters around Earth, and for deep space missions, NASA is currently in the process of redefining the way science is being gathered. Technologies such as 3-Dimensional Multi-Chip Modules, Micro-machined Electromechanical Devices, Multi Functional Structures, miniaturized transponders, miniaturized propulsion systems, variable emissivity thermal coatings, and artificial intelligence systems are currently in research and development, and are scheduled to fly (or have flown) in a number of missions. This study will leverage on these and other technologies in the design of a lightweight Neptune orbiter unlike any other that has been proposed to date. The Neptune/Triton Explorer (NExTEP) spacecraft uses solar electric earth gravity assist and aero capture maneuvers to achieve its intended target orbit. Either a Taurus or Delta-class launch vehicle may be used to accomplish the mission.

1. INTRODUCTION

A mission to Neptune presents a daunting challenge. We are just at the beginning of robotic and human exploration, and so there is much to discover. In order to routinely visit the outer limits of our solar system, there is a need to develop tools, technologies, and methods, none of which are to date satisfactory for efficient, cost-effective, and rapid implementations. The present paper highlights work performed as part of Thesis research at the George Washington University [1]. It is high-level in its need to constrain the treatise to 8 pages, but provides a glimpse into the complexities of designing a mission to the Neptune/Triton system.

2. SCIENTIFIC OBJECTIVES AND SCIENCE MEASUREMENTS

Two spacecraft have explored planets of the outer solar system. Only one (Voyager 2) has ever visited Neptune in August 1989. Although the Voyager 2 flyby of Neptune yielded a wealth of information, an orbiter is needed to expand our understanding of Neptune and its moons (in particular Triton) beyond what can be achieved by a single flyby. The importance of a visit to Neptune and Triton has been recognized in a number of studies, including the COMPLEX (Committee on Planetary and Lunar Exploration) report published in 1998 [2]. For instance, the study of Triton's thin atmosphere and its changes over time can provide insight into the Earth's own global warming. In addition, Triton is expected to provide clues as to the characteristics of Kuiper-belt objects, a cloud of material found beyond the orbit of Pluto and believed to be pristine remnants of the early solar system. The major scientific objectives for a Neptune/Triton mission can be summarized as follows:

Neptune

1) Study the plasma density and flow velocity. Determine whether plasma originates from the solar wind, Neptune's atmosphere, or Triton's upper atmosphere, and in what proportion does each contribute.
2) Measure Neptune's magnetosphere and determine whether it shows any activity.
3) Determine the processes involved in auroral emissions.
4) Study the wind structure and temperature profile of Neptune's atmosphere. In spite of the large difference in distances from the Sun between Uranus and Neptune, both wind and thermal characteristics of their atmospheres are similar.
5) Study the atmospheric composition, including the aerosol-forming species and their physical properties.
6) Determine the surface magnetic field of Neptune, and help explain its asymmetry as compared to Jupiter and Saturn.
7) Study the dynamics of Neptune's rings, and determine their composition and age. Study the rings' high dust-fraction.

Triton

1) Measure the magnetic and gravity fields to help define models of the internal structure.
2) Image the remaining 70% of Triton surface not covered during the Voyager flyby.

3) Conduct stellar occultation measurements to probe Triton's atmosphere.
4) Measure molecular species in Triton's atmosphere via infrared and millimeter-wave spectroscopy. In particular, determine the abundance of CO, and the existence of hydrocarbons, nitriles, and noble gases. Measure the distribution and source of aerosols in the atmosphere. Measure wind-speed and determine how it relates to the atmospheric thermal structure.
5) Determine the composition of Triton's ionosphere.
6) Measure the distribution of ices on Triton's surface, including N_2, CO, CO_2, CH_4, and H_2O. Determine the composition of dark matter on its surface, and find out its origin.
7) Measure the composition of materials on the surface of Triton, and observe how seasons affect the distribution and nature of surface condensates.
8) Determine the process and mechanisms driving Triton's plumes.
9) Measure the plasma environment and its interaction with Triton's atmosphere and surface. Measure the magnetic field to less than 1 nT.
10) Study Triton's impact history to help constrain the number of cometary objects believed to exist in the outer solar system.
11) Determine the processes involved in auroral emissions.

Neptune's Satellites
1) Measure the density and composition of Neptune's small satellites.
2) Determine the collisional history of Neptune's satellites, and determine their interaction with Neptune's rings.

3. SCIENCE INSTRUMENTS

The development of payloads that are optimized for fast, lightweight spacecraft suitable for outer planetary missions is a challenge that just recently has been undertaken. The Deep Space 1 (DS1) spacecraft has flown an integrated camera/spectrograph design to help to this end [3]. The Miniature Integrated Camera and Spectrometer (MICAS) is a 12-kg, 8-watt instrument designed to cover four channels from 80 nm in the ultraviolet, to 2.4 μm in the near infrared. Although it suffered a number of problems (UV spectrometer does not work, and sensitivity is not as expected), the Neptune Integrated Camera and Spectrometer (NICAS) on-board the Neptune / Triton Explorer (NExTEP) will baseline a similar design, as it is quite reasonable to assume that MICAS' problems would be resolved by the time this mission comes to fruition. The NICAS instrument consists of a panchromatic camera (referred as camera henceforth), and Ultra-Violet (UV), Visible (VIS), and Near Infrared (NIR) spectrometers. Each element will be used sequentially, allowing for a single telescope to be shared between the several focal planes. The camera is also used for optical navigation.

The second highly integrated instrument package, the Neptune Energetic Particle Detector (NEPD) will be based on a design developed for the Messenger mission. NEPD consists of an energetic particle detector, which integrates electron and ion detectors into a single 1-kg, 1-watt package. Its energy coverage is expected to be in the range of 20 to greater than 500 keV. The second particle instrument, the Neptune Plasma Analyzer (NEPA), is to be based on the current DS1 Plasma Experiment for Planetary Exploration (PEPE), a plasma ion mass and energy spectrometer, and a plasma electron energy spectrometer integrated into one package. Although PEPE has a mass of 5.6-kg and consumes 9.6-watts, preliminary studies carried out for the Space Technology 5 (ST5) mission indicate that it is reasonable to expect an instrument package mass of 1.5-kg, and a power of 1.5-watts, within the time frame of NExTEP. The energy range for this instrument would be between 3 eV and 30 keV. Finally, the Neptune Fields Instrument (NEFI) will also be based on the baseline design expected from ST5. This 1-kg, 1-watt magnetometer is expected to have a dynamic range of 0.1 to 100 nT, and a sensitivity of 0.01 nT for an ambient magnetic field of 100 nT.

Finally, although not traditionally considered a "science instrument", NExTEP will carry a monolithic silicon micro-machined accelerometer (MACCEL). Its design will allow it to provide accurate 3-axis vector measurements during the aerocapture portion of the flight, and possibly during close flybys of Triton (through its rarified atmosphere). The design approach will provide a sensitivity of better than 10^{-9} g at 1 Hz, a bandwidth of at least 0.01 and 20 Hz and an extremely large dynamic range from 10 to 10^{-11} g [4].

The highly integrated instrument packages presented above provide science measurements commensurate with the requirements of NExTEP. The properties of each instrument are summarized in Table 1.

Table 1: NExTEP Instrument Characteristics

Instrument	Measurement Range	Mass (kg)	Power (W)	Data Rate	Image Size	Size (cm)
NICAS	80 nm – 2.4 μm	10.2	6.8	2 Mbps per detector	600 Mbits (UV, VIS, and IR spectrometers) 5 Mbits (pan camera)	18 x 32 (mounting interface) x 28
NEPD	20 to >500 keV	1	1	1 Kbps	N/A	9 x 9 x 13
NEPA	3 eV to 30 keV	1.5	1	2 Kbps	N/A	13 (diameter) x 15 (length)
NEFI	0.1 to 100 nT	1	1	1 Kbps	N/A	4 x 4 x 6 (sensor) 10 x 20 x 5 (electronics)
MACCEL	10 to 10^{-11} g	0.1	0.1	1.2 Kbps	N/A	3 x 3 x 3
Total		*13.8*	*9.9*			

4. MISSION DESIGN

The concept of "Design-to-Cost" (DTC) previously applied to military operations is now routinely implemented in NASA missions at the very early stages of design. The design process, although initially influenced by scientific goals, must also iterate its results with the expected mission cost; hence the successful mission design would have achieved meaningful science within the programmatic constraints of cost and schedule. Another important driver influencing cost and schedule is the availability of technologies, and it is of particular importance here since a cost-effective mission to Neptune must rely heavily on advancements that are yet to be proven. The assumption here is that technologies would have been developed and become available in time for this mission implementation.

4.1 Trajectory Design and Propulsion Options

The trajectory design for a mission to Neptune is inexorably attached to the propulsion technology proposed. The great distance to Neptune dictates the use of either gravity assist maneuvers, advanced propulsion options such as Electric Propulsion (EP), advanced chemical propulsion, or a combination of all of them. The current NExTEP baseline trajectory design uses all these options, in what is known as a Solar Electric Earth Gravity Assist (SEEGA) trajectory. Here, Solar Electric Propulsion (SEP) is used until the Earth swing-by at which time the spacecraft accumulates enough energy to reach Neptune in a hyperbolic trajectory. Minor trajectory adjustments en route are performed by the advanced chemical propulsion system.

A rough idea for the magnitude of the problem may be obtained by looking at the classic minimum-energy, co-planar, direct-transfer trajectory (Hohman Transfer). It can be easily shown that a direct transfer of this nature would take about 30.7 years, and require a C_3 of 135.9 km^2/s^2. Although the injection ΔV may be reduced about 2.5 km/s by assuming departure from a GTO orbit rather than from a circular orbit at 185 km, it would still require 5.8 km/s. The corresponding Neptune Orbit Insertion (NOI) ΔV is about 0.4 km/s, a number applicable to the minimum energy transfer case, and assuming an initial Neptune orbit with a period of 109 days. For a faster transfer, the ΔV required for NOI is considerably higher than the minimum energy case (~ 7 km/s), and aerocapture appears to be the only cost-effective choice in achieving it. Additional on-board ΔV requirements at Neptune will depend on the final orbit. Finally, since Triton orbits Neptune at a 157-degree inclination orbit (retrograde), the arrival trajectory will be adjusted so as to match Triton's orbit plane, and occur at the leading end of Neptune's orbit.

Trade studies were carried out in order to resolve the high-energy and flight-time issues associated with a Neptune mission. A gravity-assist trajectory similar to the one used by Voyager 2 was considered, which could reduce the flight time to about 12 years. However, gravity assist trajectories as complex as Voyager 2 place important constraints on the launch window and still require an Expendable launch Vehicle (ELV) capable of achieving high-energy planetary injection performances. A number of alternative trajectories were considered that utilized more advanced technologies, including chemical and electric propulsion options [5]. Among the chemical (CHEM) options investigated were solid (injection and arrival stages), hybrid (arrival stage, and trajectory ΔV corrections), and liquid (injection and arrival, and trajectory ΔV corrections). Among the EP options investigated were ion propulsion (Electrostatic), and Magneto-Plasma-Dynamic (MPD) arcjets (Electro-thermal), for injection and trajectory ΔV corrections out to 2.3 AU, and Pulsed Plasma Thrusters (Electromagnetic) for trajectory ΔV corrections. From the EP options, the Xenon ion propulsion engine showed the greatest promise in terms of efficiency in the specific impulse (Isp) range between 3500 and 5000 seconds. Given the state of development of the Xe-ion engine, this may be the most mature choice (e.g., DS1), and was the baseline for NExTEP. From the CHEM options, a new chemical monopropellant based on Hydroxyl-Ammonium Nitrate (HAN) shows the most promise in terms of specific impulse, packing density, and operating temperatures for a deep space mission such as NExTEP.

A search of existing literature provided a reference on upper and lower bounds for travel times, and applicability of ion propulsion for a mission to the outer planets [6]. Depending on variables such as gravity assist maneuvers, arrival hyperbolic excess velocity, and spacecraft mass, "fast" electric-propulsion travel times to Neptune may vary from about 8 to 12 years. The advantage of a SEP system increases for small delivered mass, high transfer energies and high (~ 17 km /s) arrival hyperbolic excess velocities. The baseline trajectory developed here uses a SEP low-thrust trajectory in the inner solar system, and a chemical mission for the remainder of the voyage. The trajectory calls for using SEP within a distance of 2.3 AU from the Sun, combined with a two-year Earth gravity assist (SEEGA). After the Earth swing-by, the SEP propulsion stage is dropped (together with the solar arrays), and the spacecraft then continues with the CHEM system to Neptune using power from an on-board thermoelectric source. At

Neptune, the spacecraft performs an aerocapture maneuver, and is placed on a highly elliptical retrograde orbit. Figure 1 illustrates the baseline trajectory design.

The complete trajectory was calculated using a unique analytical approximation method developed without the need to resource to numerical integration or calculus of variations. Although this technique is only good for conceptual design, it is capable of highlighting the main mission design drivers and showing implementation feasibility. The computation steps are shown next:

1) The spacecraft is injected into a two-year Hohman Transfer orbit by the launch vehicle.
2) The SEP system provides a spacecraft ΔV such that the energy at Earth swing-by is equivalent to that of a four-year keplerian orbit.

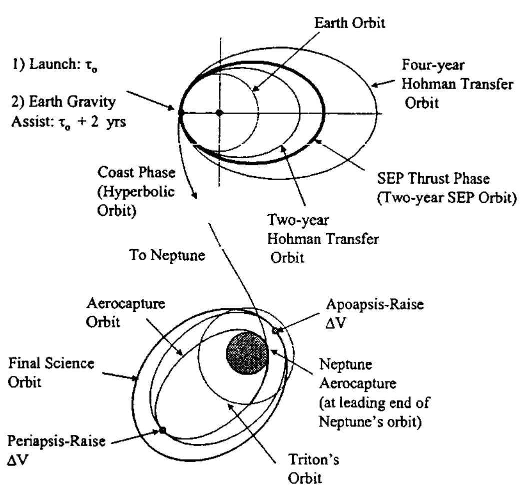

Figure 1: NExTEP Trajectory

3) The approach velocity is equivalent to that of a Hohman transfer for the four-year trajectory.
4) The flight-path angle is adjusted to allow for energy gain after the Earth flyby. If this were not done, a pure Hohman transfer from a high-energy orbit to a lower-energy orbit would incur (as expected) an energy loss.
5) The SEP module is ejected after the SEEGA is complete, after which the spacecraft is in a hyperbolic (coast) trajectory to Neptune.
6) A skip-entry analysis [7] is performed at Neptune, where the velocity change before and after the atmospheric entry is adjusted (via Lift/Drag and flight-path angle changes) such that the resulting ΔV places the spacecraft in an initial orbit around Neptune. The semimajor axis of this orbit (and hence its period), will be determined based on results from the aerocapture analysis, for reasonable g-loads and aerodynamic heating.
7) A periapsis-raise maneuver is performed at apogee of the aerocapture orbit to avoid a second atmospheric entry. The periapsis will be adjusted such that it lies within the inner boundary of the innermost ring. Given that the orbit inclination is made to match that of Triton (157-degrees), the choice of periapsis helps ensure the orbit geometry does not intersect Neptune's rings.
8) The apoapsis is adjusted, such that the final orbit period is a multiple of Triton's orbit.

Detailed computations may be found in Reference [1]; results are summarized in Table 2. The two-year Hohman transfer orbit requires a launch energy (C_3) of 25.7 km^2/sec^2, as opposed to 62.78 km^2/sec^2 for a four-year orbit. The difference is then made up by the SEP system, which must provide an on-board ΔV of 2.9 km/sec. This is a very modest value, and is well within the performance reach of current SEP systems. The difference in C_3 is quite considerable though, and enables the utilization of a Taurus-class launch vehicle for the projected vehicle injection mass of 77 kg. This assumes an "enhanced" standard Taurus 2230 launch vehicle with a STAR 37FM third stage motor, which provides a performance of about 115-kg for a C_3 of 26 km^2/sec^2. Utilization of a Lockheed-Martin Delta 7325, would increase the injected mass capability to 400 kg, providing a relatively more expensive but still reasonably cost-effective alternative.

Table 2: Results of Trajectory Computation and Atmospheric Entry Aerothermodynamics

SEEGA		**Aerocapture**	
Launch Energy, C_3 (km^2/sec^2)	25.7	Neptune-Centered Arrival Velocity (km/sec)	30.1
SEP on-board ΔV (km/sec)	2.9	B-Plane Offset (km)	25264
Vehicle Injected Mass (kg)	77	Entry Interface Altitude (km)	500
Taurus Performance for C_3 (kg)	115	Desired ΔV at Aerocapture (km/sec)	7.1
First Leg Duration (years)	2	Entry Flight Path Angle (degrees)	-1
Flight Path Angle at Earth Swing-by (degrees)	9.2	Lift / Drag	0.1295
Heliocentric ΔV gained (km/sec)	7.8	Atmospheric Scale Height (km)	39.6
		Maximum Aerodynamic Load (g)	2
Hyperbolic Trajectory		Pull-up Altitude (km)	200–400
Flight Time (years)	6.7	Total Heat Load (Joules)	1.7×10^8
Excess Hyperbolic Velocity at Neptune (km/sec)	19.2	Maximum Body Average Heating Rate (watts/m^2)	9.4×10^6

5. SPACECRAFT DESIGN

Spacecraft mass is a critical parameter influencing length of travel to Neptune and the outer planets in particular insofar as SEP performance is concerned [8]. The design of Micro/Nano-spacecraft (1 to 100 kg-class) requires the consideration of a number of elements, ranging from miniaturization to system-level and component-level integration. The approach selected here is to seek subsystem and system integration into a spacecraft that shares resources to the maximum extent possible.

Contributing factors to the overall level of systems integration include advances in the areas of electronics packaging, micro-machining, multi-functional structures, and integrated power and thermal systems. Electronic packaging technologies include "system-on-a-chip" approaches, multi-chip module (MCM) electronic stacks, and Application Specific Integrated Circuits (ASIC). Micro-electromechanical Systems (MEMS) and Application Specific Integrated Micro-instruments (ASIM) constitute the latest efforts in micro-machining. Multi-Functional Structure (MFS) technologies which combine the spacecraft structure with electronic and power components while also providing spacecraft thermal control were partly demonstrated in DS1, and will be also demonstrated in ST5. Technological advances that facilitate a maximum level of spacecraft integration within the 2015 timeframe are utilized throughout this study.

5.1 Spacecraft Architecture

The central element in the architecture is a 3D Redundant Multi-Function Module (RMFM) which provides both for simplicity and for integration of diverse functions. This packaging technology consists of seven layers or "slices", each specializing in a particular spacecraft function, but cross-strapped to allow for redundancy of operation. The level of redundancy will depend on the work-sharing architecture, and represents a balance between complexity and operational safety. The spacecraft architecture main elements are illustrated in Figure 2. In the following sections, key spacecraft subsystems will be addressed in some detail. Reference [1] provides information on *all* spacecraft subsystems, and includes detailed sizing computations.

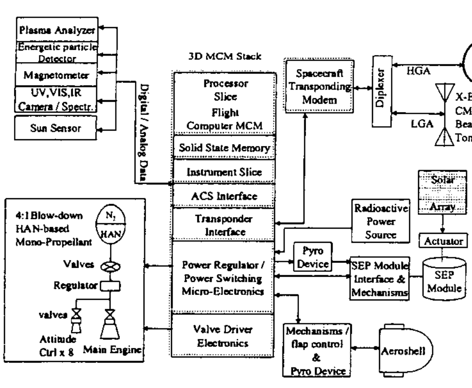

Figure 2: NExTEP Physical Block Diagram

5.2 Main Propulsion System

Two propulsion systems are used in NExTEP: electric (SEP), and chemical (CHEM). Given that the CHEM propulsion system will be part of the payload to be carried along by the SEP system, it is necessary to size it first. In addition, CHEM system mass and size results will impose by necessity some constraints on the final science orbit.

Chemical Propulsion System

The CHEM system will be used for trajectory corrections en-route to Neptune after the Earth gravity assist, and for final orbit adjustments after Neptune's aerocapture. The monopropellant choice for NExTEP is based on Hydroxylammonium Nitrate (HAN). This selection is based on across-the-board advantages in the areas of safety, performance, density, and thermal management requirements. Compared to Hydrazine, HAN-based monopropellants are relatively easier to handle (N_2H_4 is toxic and flammable), can reach specific impulse values of 270 seconds (N_2H_4 can deliver up to about 230 seconds), densities are about 40% higher (1.4 g/cm^3, versus 1.0 for N_2H_4), and operating temperatures range from –33 to +65 °C (N_2H_4 is maintained at greater than 7 °C, and freezes at 0 °C). All these properties are ideal for an environment such as Neptune's where low temperatures (~ 50 K), and severe mass and size limitations benefit greatly from the HAN-based monopropellant properties listed above. It should be noted that a number of monopropellant formulations have been tried with HAN, generally differing from the amount of carbon content. General characteristics for the NExTEP advanced monopropellant thruster are as follows:

Operating temperature: -33° C to 65° C
Operating Power Requirement: ~ 1 watt
Propellant Base: Hydroxylammonium Nitrate (HAN)
Operating Modes: Continuous and Pulsed
Specific Impulse: 260 seconds

The ability to operate the thrusters on a continuous or pulsed mode makes this system suitable for attitude control, just as for ΔV maneuvers. Minimum impulse bit achievable is close to a cold gas system (~45 mN-s). CHEM system and science orbit data are shown in Table 3.

Solar Electric Propulsion System

The main difference in sizing an electric propulsion system from a chemical system is that for an electric system the major contribution to the mass comes from the power subsystem, rather than fuel. This can be easily understood if one considers that fuel performance (measured by specific impulse) is several orders of magnitude greater for electric engines (between 1500 to 10000 seconds). On the other hand, electric engine performance is in turn proportional to power input, which tends to drive the power system mass. The approach used here to size the electric propulsion system follows the "classic" approach [5], where the propulsion system is important enough to drive the design of the power system itself. This mainly refers to the solar arrays, whose sole purpose is to feed the SEP system, and will be jettisoned along with the SEP module after the SEEGA maneuver is complete.

Table 3: Science Orbit and CHEM Propulsion System Data

Science Orbit Parameters	
Neptune Equatorial Radius - Rn (km)	24764
Semimajor Axis (in Rn)	22.7
Periapsis Radius (in Rn)	1.4
Apoapsis Radius (in Rn)	44.0
Eccentricity	0.94
Inclination (deg)	157
Orbit Period (d)	11.8
CHEM Propulsion system sizing input	
Periapsis Raise ΔV (m/sec)	81
Apoapsis Lower ΔV (m/sec)	31
Trajectory Corrections and Orbital Maneuvering (m/sec)	223
Total CHEM on-board ΔV (m/sec)	335
CHEM System Parameters	
Monopropellant I_{sp} (sec)	260
Propellant Mass (kg)	5
Tank Mass (kg)	1
Main Engine Mass (kg)	0.6
Main Engine Thrust (Newton)	80

A payload mass-fraction optimization technique was used to size the ion propulsion system (IPS). Figure 3 shows the payload mass-fraction as a function of specific impulse, obtained from properties applicable to an NSTAR-derivative (NASA Solar Electric Propulsion Technology Applications Readiness) thrusters [9], and NExTEP mission requirements. As can be seen, the optimum specific impulse (I_{sp}) for this thruster is found at values greater than 3000 seconds. Taking advantage of the NSTAR ion thruster heritage, we set the I_{sp} at 3,300 seconds, with a corresponding exhaust speed of 32.4 km/sec. This shows that the SEP system as defined can be optimized to yield the greatest possible mass fraction. Reference [1] shows detailed sizing computations for the SEP system; results are summarized in Table 4.

Since the solar array is part of the SEP system exclusively, it will be briefly addressed here. The SEP trajectory, as well as the allowable SEP thrusting time defines the panel maximum operating distances. The assumption here is that the ion engine would be run for a total of 8000 hours or about 334 days (IPS allowable on-time). However, since the available thrust time is 730 days (two years), the IPS will not be needed about 46% of the time. It is postulated that this "dead time" can be chosen to be when the spacecraft is farthest from the Sun (and the solar array is less efficient). For an assumed SEP trajectory with a heliocentric apoapsis at about 2.9 AU, SEP thrusting times may be limited to a distance of approximately 2.3 AU on either side of apoapsis. It is then assumed here that the array is to operate at a distance of 2.3 AU (worst case) and provide 197 watts EOL, and sized accordingly. Using quad-junction GaAs cells with 35% efficiency and a packing factor of 95%, the solar array area comes out to 3.6 m^2. The mass allocation for a 305-watt array BOL comes out to 3.3 kg for a specific mass of 10.7 W/kg.

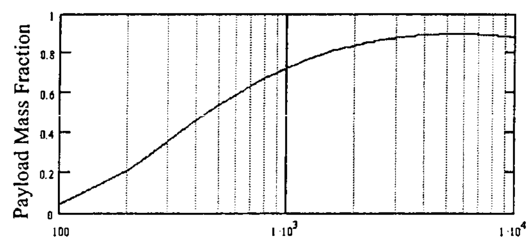

Figure 3: Finding the Optimum Specific Impulse

Table 4: SEP System Performance Parameters

Specific Impulse (sec)	3,300
Payload Mass Fraction	0.882
Propellant Mass (kg)	6.4
Propellant Flow Rate (kg/sec)	2.2×10^{-7}
Electric Power Source (watts)	197
Thruster and PPU Mass (kg)	1.5
Tank and Feed System Mass (kg)	0.96
Solar Array Mass (kg)	2
SEP System Wet Mass (kg)	11

5.3 NExTEP Communications

Given the amount of science data collection and the corresponding need for fast data rates, it is expected that ultimately optical communications will become the standard form of data transmission from outer planet spacecraft. However, within the 2015 timeframe, Ka-band appears the most promising, having been already validated during the DS1 flight. The Spacecraft Transponding Modem (STM), being developed by NASA JPL is a miniaturized next generation X-band

and Ka-band transponder which uses heritage from the DS1 Small Deep Space Transponder (SDST). It is expected to become available in 2003, and includes uplink carrier acquisition, command and telemetry frame level interface, downlink telemetry data rate profiling, Reed-Solomon encoding, turbo coding, QPSK downlink telemetry modulation, regenerative PN code ranging, and modem timekeeping. The STM will provide a modem-like interface with the spacecraft, and allow X-band uplink and X-band and Ka-band downlink. Amplifiers and antennas need to be added to complete NExTEP's communication system.

Computations for the up/down link budgets via an on-board High Gain Antenna (HGA) and Ka-band (35 GHz), assumed the use of the 70-meter DSN antenna. Results of the downlink analysis show that at Neptune distances (4.6 billion km), a HGA 2-meters in diameter and a radio frequency power output of 5 watts will be able to communicate with a receiver of 4000 Hz bandwidth, and end up with a signal-to-noise ratio (SNR) of 3.8 dB. Assuming that the acceptable SNR at the receiver is 0 dB, 3.8 dB also represents the link margin. Since the bandwidth is proportional to the channel capacity, increasing the bandwidth will also increase the data rate capability, at a penalty of increasing the link (receiver) noise. The Shannon error-free limit gives an idea for "error-free" communications in the presence of noise. It allows a maximum data rate communication of 7 kbps for a bandwidth of 4000 Hz, assuming coding techniques (e.g. Reed-Solomon and/or convolutional code) are used that can push the data rate capacity to this level.

Based on this downlink budget, a two-year orbital tour of the Neptunian system would yield 446 Gbits of science data, assuming a 3:1 lossless data compression ratio. Note however, that this situation may be improved as it is based on a DSN utilization rate of only 33% per 11.8-day orbit. Higher data compression rates may also be used, and are currently being investigated.

5.4 Power Subsystem

NExTEP will use cutting-edge technological advancements across-the-board, including ultra-low power electronics. A Radioactive Power Source (RPS) will be used to generate electricity. The RPS is based on a JPL planned development of an advanced source capable of delivering about 9.4 watts/kg. With the environmental concerns attached to radioactive sources, the goal has been to reduce the amount of plutonium necessary to obtain a given power output. Several thermal-to-electric conversion technologies are under study, with RTPV (Radioisotope Thermoelectric Photovoltaic generator) and AMTEC (Alkali Metal Thermal to Electric Converter) being the leading candidates. Although both technologies are currently comparable in their efficiency, AMTEC has shown the potential of achieving heat-to-electric power conversion efficiencies in excess of 30 % [10], and is therefore the baseline for NExTEP. Assuming a specific power of 9.4 watts/kg, the required RPS mass for NExTEP is estimated at 4.0 kg. Based on straightforward scaling from the current JPL design, NExTEP's RPS may only require roughly 1 kg of PuO_2.

5.5 Aerocapture System

NExTEP uses recent developments in aeroshell ablative materials. These new generation materials center around lightweight ceramic ablators invented and developed at NASA Ames for atmospheric entry. Given the heat loads encountered during atmospheric flight, an ablative material capable of withstanding loads of approximately 1000 watts/m^2 is needed. Of the family of ceramic ablators PICA (Phenolic Impregnated Carbon Ablator), with densities in the range of 0.25 to 0.60 gr/cm^3 is baseline. Effective heat of ablation range from about $6x10^4$ to $2x10^5$ Joules/gr of material. A ballistic coefficient of 17.3 kg/m^2 and L/D of 0.12949 are the two defining parameters for the aeroshell. The assumption made here is that these properties can be built into the aeroshell configuration, and that changes can be modulated real-time via flaps. Finally, it is worth mentioning that although the term aeroshell is being used loosely to refer to the spacecraft enclosure (forebody), strictly speaking it also includes the backside of the HGA.

6. SPACECRAFT LAYOUT

The spacecraft configuration proved to be a most challenging endeavor as it varies depending on the mission phase. There are four such phases and associated spacecraft configurations: SEP, open aeroshell cruise, aerocapture, and science operations. Figure 4 shows a layout for each phase/configuration. The SEP module and arrays are jettisoned at the end of phase 1, whereas the aeroshell is jettisoned at the end of phase 3, prior to science operations.

Mass and power allocations are summarized in Table 5. Once again, technology advances were assumed across the board for most subsystems, including those not summarized in this paper, such as structure and mechanisms, thermal, Guidance, Navigation and Control, and Flight Software.

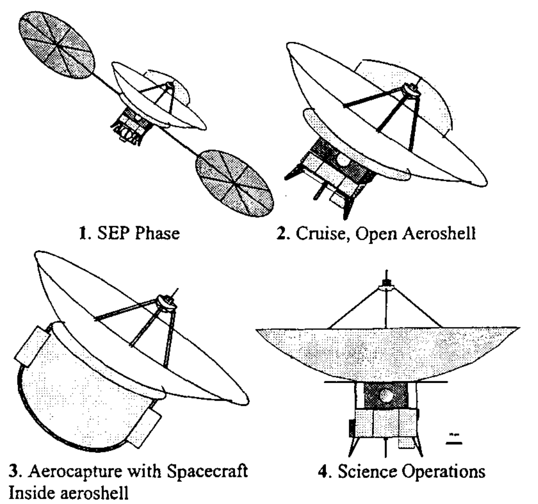

Figure 4: Spacecraft Layout is Different Depending on Mission Phase

Table 5: NExTEP Mass and Power Budgets

SUBSYSTEM	MASS (KG)	Peak POWER (Watt)	Low/Standby POWER (Watt)
Science Instruments	13.7	13.9	6.9
Spacecraft Structure and Mechanisms	5.7		
Command & Data Handling (3D RMCM)	0.1	0.5	0.5
Communication	1.0	10.0	1.0
Power Systems	4.0		
Thermal Control	0.7	1.3	0.1
Flex Harness	1.0		
Radiation Shielding	3.0		
Attitude Control Sensors	1.0	2.3	2.3
Chemical Propulsion	6.9	3.0	0.0
Aeroshell	9.8		
Total Power Requirements (Excluding SEP)		31.0	10.8
Spacecraft Wet Mass (w/o SEP Module)	46.8		
Add 20% contingency	**56.2**	**37.2**	**13.0**
SEP Module Wet Mass and Power Requirements	17.2	**197**	
Add 20% Contingency	20.7		
Total Spacecraft Injected Mass (Incl. Contingency)	76.9		

7. CONCLUSIONS

The work summarized here ascertains the feasibility of a Neptune orbital mission within tight budgetary constraints, by using advanced technologies expected to be developed and available within the next 10 to 15 years. The use of electric propulsion, aerocapture, advanced spacecraft technologies, and an overall system and subsystem integration approach, reduces the total vehicle injected mass (77 kg) which in turn enables the utilization of more cost-effective launch vehicle options. At about 44 kg total, the NExTEP Neptune orbiter is optimized to carry 37% of its mass in scientific payloads. With a two-year SEEGA, the spacecraft reaches Neptune 8.7 years after launch. The use of aerocapture also enables this fast transfer, as it provides a ΔV in the order of 7 km/sec to attain an initial 13.6-day period orbit. Communications via Ka-band allows reasonable amounts of data to be transmitted, with about 446 Gbits of scientific data returned over a two-year orbital tour of Neptune and Triton. All these features combined (SEEGA, aerocapture, Ka-band, advanced technologies) suggest a means for making the outer solar system closer to our reach on a routine basis.

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