# Reaching the Outer Planets with Nuclear Electric Propulsion: Trades, Sensitivities, and the Case for a Neptune System Explorer

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**Abstract.** Over the last year, a large effort that involved several NASA agencies and DOE was initiated to evaluate the mission benefits and understand the sensitivities of Nuclear Electric Propulsion (NEP). This paper first describes the sensitivities of the mission design parameter space (i.e. the trades between propulsion system characteristics, power level, system efficiencies, and flight times). It also illustrates the findings for a conceptual Neptune System Explorer mission. A point design for this mission is presented, using a 100-kWe Power and Propulsion Module (designed in parallel by a NASA MSFC led effort) along with a representative science payload. This mission features a fly-by of Nereid, a capture and 10-month stay around Triton, a transfer to an elliptical polar orbit around Neptune and science in this orbit for 12 months. The system features a very high downlink data rate from Neptune (several 10s of Mbps), and a comprehensive suite of science instruments. Variations in power levels around the design point are investigated. This analysis shows where technology development should be directed to fully take advantage of the NEP capabilities.

## INTRODUCTION

Early 2002, MSFC led a NASA agency wide team under the In-Space Propulsion Program to define a conceptual design of a 100-kWe Nuclear Electric Propulsion (NEP) vehicle. The design of the vehicle was generic enough to fit various missions going to various destinations. A parallel study led by JPL was undertaken and had for objectives to design a representative "sciencecraft" that would be propelled by the Nuclear Electric vehicle and would take advantage of the high power once at destination. To fully understand the system and science impacts of NEP, it was important to identify a science mission and a science payload that took advantage of the NEP vehicle and that would be scientifically attractive. It was also of interest to identify the design requirements, interfaces and constraints that influence the design of the NEP vehicle and sciencecraft. Key issues were the radiation environment, science instrument requirements, telecommunications requirements, thermal issues, attitude control issues and configuration constraints.

During five concurrent engineering design sessions, a single study option was considered for a Neptune System Explorer (NSE) mission. Significant trajectory design, science objective definition, and payload design work occurred pre-session to seed the Sciencecraft design effort. The convened JPL team designed the "Sciencecraft" portion of the integrated spacecraft and referred to the MSFC-led "Tiger Team" design for the baseline NEP vehicle. Liens were held against the NEP vehicle design to meet the requirements of the NSE mission. Subsequent work done by the Nuclear Systems initiative (NSI) Science Definition Team (SDT) rated the Neptune/Triton mission as one of highest interest for the exploration of the Outer Planets.

Since the flight times to Neptune for this design point are considerably long, sensitivities around that point were investigated. This paper starts by setting up the grounds for the sensitivity analysis. It lays out the NEP parameters that directly influence the mission design and the governing equations. It then describes the NSE mission, science, instruments, Sciencecraft design and NEP vehicle. Finally it combines both the basic NEP equations derived in the first part and the results of the NSE design to explore the trade space and draw conclusions for the technology.

## NEP MISSION DESIGN PARAMETER SPACE

We develop here simple relationships to help understand the parameter space of a NEP mission design. These relationships are of first order in this very complex parameter space. They should not be used as absolute, but rather as a first estimate. Their intent is to show how the variables influence each other. They will help understanding where the technology should be headed to benefit the missions. The following simple relationships can also be used when access to trajectory codes is limited. It establishes a simple way for approximating flight time and performance of an NEP mission once a few higher fidelity trajectories have been run.

### The five principal independent parameters

The mission design and trades of an NEP system depends fundamentally on five independent parameters: NEP system electric power $P_o$, launch injected mass $M_o$, electric thruster characteristics (specific impulse Isp and total efficiency $\eta$), and mission $\Delta V$. The NEP system electric power depends on the size of the nuclear reactor and efficiency of the power conversion system. This power is assumed to remain constant over the trajectory. The launch injected mass can vary significantly depending on the launch vehicle chosen and the altitude of launch. Figure 1 provides delivery capability as a function of altitude for the Delta IV Heavy launch vehicle (NASA KSC Database). The electric thruster characteristics will also vary as a function of the thruster technology and design. This analysis should help guide the development of the appropriate thruster technology. The mission $\Delta V$ is function of the destination, and Figure 2 provides ranges of $\Delta V$ for various potential NEP missions.

There is an additional variable, which depends on the specific design of all the NEP vehicle's subsystem (and technologies): the NEP spacecraft dry mass. A description of an NEP vehicle is given in several references (Lipinski, 2002, Elliott, 2003). The mission design will rely heavily on this value. Each reactor, power conversion system, radiator system technology, and electric propulsion system will imply a different system implementation and dry mass. The aim of this section is not to go into details about the technologies but to grasp their effect on the mission design.

The mission flight time is a resultant of all these parameters and of a trajectory optimization.

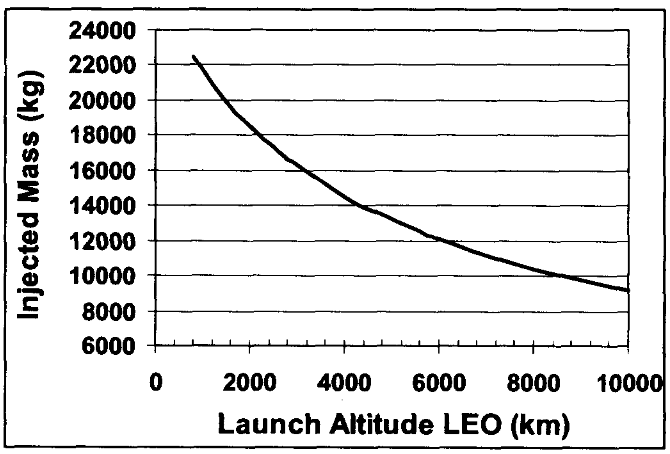


**FIGURE 1.** Delta IV Heavy injected mass as a function of altitude (LEO).

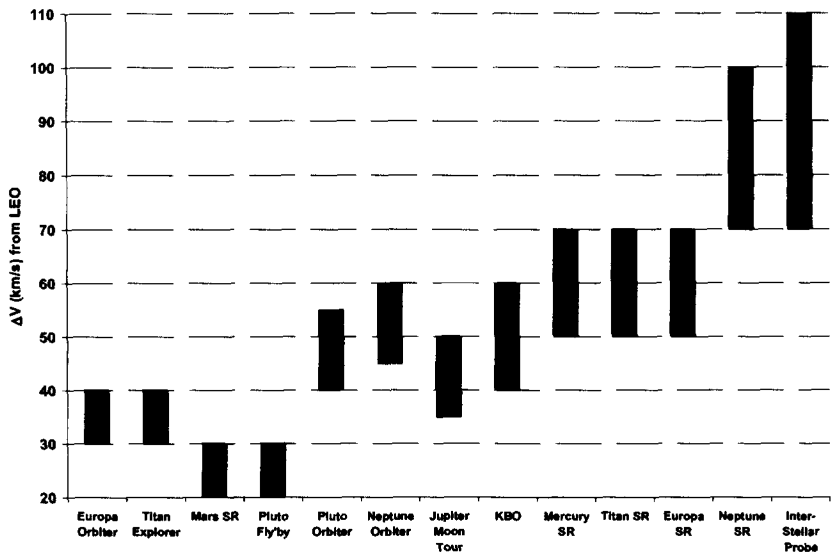


**FIGURE 2.** Ranges of ΔV for various potential NEP missions

## Relating the parameters

Two equations govern the parameters and the mission trade space:

$$\text{The Rocket Equation: } \Delta V = g_o \times Isp \times \ln\left(\frac{M_o}{M_o - M_{prop}}\right) \qquad [1]$$

$$\text{The Mass Flow Rate Equation: } \frac{M_{prop}}{T_{burn}} = \frac{2 \times \eta_{sys} \times P_o}{(g_o \times Isp)^2} \qquad [2]$$

where ΔV is the magnitude of the effective velocity change, $g_o$ is the acceleration constant (9.8 $km/s^2$), Isp the thruster specific impulse, $M_o$ launch injected mass, $M_{prop}$ is the propellant mass used, $T_{burn}$ the burn time of the thrusters, or in other words the total duration for which the thrusters are on to provide the required ΔV, $\eta_{sys}$ the total system efficiency, and $P_o$ the NEP system electric power. $T_{burn}$ is related to the total flight time.

In addition, the total dry mass of the NEP vehicle can be seen as mainly varying as a function of the Power level $P_o$, and as a function of the propellant mass $M_{prop}$. We can thus decompose the total dry mass of the NEP vehicle as such:

$$M_{tot\_dry} = \alpha_{power} \times P_o + K_t \times M_{prop} + M_{NEP\_fixed} + M_{payload} \qquad [3]$$

Where $\alpha_{power}$ is the power generation, conversion, and heat dissipation specific mass, $K_t$ is a propellant "tankage" factor, $M_{NEP\_fixed}$ is the mass of the NEP vehicle that does not vary, or only varies weakly as a function of propellant mass or power level, and the payload mass is the sciencecraft with all its subsystems (that is integrated with the NEP vehicle). The Neptune System Explorer description given below will clarify what is included in each mass category. Table 1 gives details of what is assumed in the "Tankage" factor $K_t$.

**TABLE 1.** Tankage factor description

| **The tankage factor includes:** | |
|---|---|
| Propellant $\Delta V$ margin | 2 % |
| Engine restarts, Leakage, Testing | 0.1 % |
| Navigation errors, Thrust misalignment | 1.6 % |
| Modeling errors | 1.5 % |
| Fill errors | 0.6 % |
| Flow uncertainty, FCD characterization | 2 % |
| Residuals | 2.5 % |
| Tanks | 2.5 % |
| Tank structure | 4 % |
| Thermal | 0.1 % |
| Additional dry propulsion mass for added propellant | 2 % |
| **Total** | **~19 %** |

At the same time,

$$M_{tot\_dry} = M_o - M_{prop} \tag{4}$$

Thus we can rewrite [3] and [4] simply as:

$$M_{prop} = \frac{M_o - \alpha_{power} \times P_o - M_{NEP\_fixed} - M_{payload}}{1 + K_t} \tag{5}$$

This equation shows that ***for a given Power level and system specific mass (alpha), the propellant mass to the first order is only dependent on the initial mass and fixed masses. It does not vary with Isp or mission $\Delta V$.*** The Isp will tune to the mission $\Delta V$ to keep the propellant mass fixed. In other words, if one assumes the maximum launch capability at a launch altitude, say the Delta IV Heavy at 2000 km altitude (18450 kg), and say a 100-kW power system, ***a mission to Mars or a mission to Neptune will use the same amount of propellant (for the same science payload mass)***. What will vary between these two missions are the Isp and the trip time. Let's establish now how they vary.

The Isp, if freely optimized, will tune to the $\Delta V$ to give the required $M_{prop}$ of equation [5]. This Isp is the minimum trip time Isp for that $\Delta V$. Increasing the Isp from its optimal point for the given conditions will decrease the amount of propellant (thus not filling the full capability of the launch vehicle), and will increase the trip time. Decreasing the Isp will imply higher propellant mass and won't fit within the capability of the launch vehicle.

Since the propellant mass is independent of Isp and $\Delta V$ to the first order, the Rocket Equation can then be written as:

$$Isp = \frac{\Delta V}{g_o \times K_o} \quad \text{where } K_o = \ln\left(\frac{M_o}{M_o - M_{prop}}\right) = \text{constant} \tag{6}$$

The Isp is then a ***linear function*** of $\Delta V$.

Finally, the other parameter impacted by [5] is the flight time. The only way to accurately calculate the flight time to a destination is to numerically integrate a trajectory. Equation [7] permits the calculation of the burn time, which is the duration the thrusters are on. To first order, there is a linear relationship between the burn time and the flight time. We will use this relationship to infer trip time impacts of other parameters. The burn time can be inferred from the mass flow rate equation and depends on the mission $\Delta V$.

$$Tburn = \frac{M_{prop} \times (g_o \times Isp)^2}{2 \times \eta_{sys} \times P_o} \quad [7]$$

In reality, the mission might be constrained by the availability of the electric thrusters and thus by some Isp range. If the Isp becomes a fixed parameter and not a variable anymore, this analysis will be only true for one ΔV and launch mass. The trip time won't be minimized for other ΔVs.

We will now apply these results to the example of a Neptune System Explorer. This study actually had other objectives, which were to conceptually design a "payload" (science and spacecraft) in the context of a potential NEP mission. We will show how the relationships derived above provide a very good first understanding of the parameter space.

## NEP NEPTUNE SYSTEM EXPLORER SPACECRAFT DESIGN

For clarity and consistency in the terminology, we shall call the NEP Neptune System Explorer a "Spacecraft". This spacecraft consists of two parts: the "NEP vehicle", which consists of all the elements related to the Nuclear Electric Propulsion System (power generation and conversion system, thermal system, attitude control, structures, electric propulsion...) and a "Sciencecraft", which consists of the science payload and all subsystems that drive the whole spacecraft (avionics, telecom, sciencecraft thermal and structures, attitude guidance and navigation, etc...). The sciencecraft could be seen as a distinct element "bolted onto" the NEP vehicle.

### Science Mission at Neptune/Triton

#### *Science Objectives*

The over-arching Science Objectives of a Neptune System Explorer are the following:

1. Determine the past evolution and present-day nature of Neptune and Triton, prime examples of an "Enriched" giant planet and an icy "Plutino", large Kuiper belt-like object. Neptune and Triton are two examples of diverse planetary formation and evolution in the outer solar system, in the frigid regime beyond Jupiter/Saturn.

2. Investigate/determine the formation mechanisms and evolutionary histories for each object.

3. Investigate/determine present-day processes. These include dynamics, meteorology at Neptune, photochemistry, thermochemistry at Neptune and Triton, geology (including active cryovolcanos) on Triton and magnetospheric processes (interaction with solar wind).

4. Investigate Neptune ring system and shepherding satellites.

5. Investigate Neptune System small satellites.

One additional mission constraint was to arrive as soon as possible in the Neptune System, since Neptune/Triton is currently passing through Southern Summer Solstice and thus enhanced cryovolcanic and atmospheric activity on Triton is expected for the next 2-3 decades.

#### *Science Instruments*

The science payload consists of 3 major subsystems: an observational system on the main spacecraft, 3 atmospheric entry probes for Neptune, and 3 surface probes for Titan. The spacecraft will provide complete global coverage at a variety of altitudes for both Triton and Neptune. This implies Nadir viewing over both poles and equators, variable distances to accomplish diverse science (~700 to 2000 km altitude for Triton, ~1000 to 100000 km altitude for

Neptune and Rings), orbits about both Triton and Neptune for approximately 1 year each, and variable inclinations during the mission. The 3 Neptune atmospheric probes will sample the composition of the atmosphere all the way down to the 100 bar pressure level at a variety of latitudes (over 60 degrees span of latitudes between the 3 vehicles).

The suite of instruments for the spacecraft is summarized in Table 2, 3 and 4. Instruments for the atmospheric and surface probes were defined and used to estimate the probe masses, but the probes were not designed in detail. They were carried simply as mass allocations in the spacecraft payload. In addition to the science payload, a high-rate optical telecommunication back-up system was included. This system uses the high-resolution telescope for data transmission.

**TABLE 2.** Neptune System Explorer Spacecraft Science Instruments mounted on a Scan-Platform.

| Instrument | Mass (kg) | Power (W) | Data rate (Mbps) | Heritage/comments |
|---|---|---|---|---|
| High Resolution Telescope | 100 | 30 | 20 | 1-m aperture |
| Wide-Angle Camera | 3 | 5 | 5 | |
| Context Camera | 5 | 5 | 5 | |
| Optical Com. System | 60 | 10000 | 50 | Backup to RF comm. |
| UV Hyperspectral Mapper | 15 | 25 | 5 | |
| NIR Mapping Spectrometer | 30 | 35 | 40 | |
| Thermal imager | 10 | 8 | 0.25 | Includes radiator |
| **TOTAL** | **223** | | | |

**TABLE 3.** Neptune System Explorer Spacecraft Science Instruments mounted on the spacecraft.

| Instrument | Mass (kg) | Power (W) | Data rate (Mbps) | Heritage/comments |
|---|---|---|---|---|
| RADAR/SAR | 100 | 600 | 1 | 6-m reflector antenna |
| Active Magnetospheric Sounder | 2 | 10000 | | Peak pulse (W) |
| Microwave Radiometer | 2 | 50 | | |
| Ion Neutral Mass Spectrometer | 5 | 30 | | |
| Surface Sublimation Experiment | 55 | 10000 | | Peak pulse (W) |
| **TOTAL** | **164** | | | |

**TABLE 4.** Neptune System Explorer Spacecraft Science Instruments mounted on a 10 m Boom that allows 360 deg. scans.

| Instrument | Mass (kg) | Power (W) | Data rate (Mbps) | Heritage/comments |
|---|---|---|---|---|
| Magnetometer | 2 | 1 | | |
| Charged Particle Spectrometer | 5 | 10 | | |
| Dust Detector | 2 | 3 | | |
| Plasma Wave Spectrometer | 5 | 10 | | |
| **TOTAL** | **14** | | | |

*Mission Scenario*

This mission proposes launching a 100 kWe nuclear electric propulsion (NEP) integrated spacecraft into Earth orbit. The spacecraft would spiral out from Earth and cruise to the Neptune system, where it would spiral in. After a fly-by of Nereid, the spacecraft would spiral into circular orbit about Triton where it would release 3 in-situ lander probes and take its full suite of science measurements over an extended period. Triton's orbit around Neptune is approximately circular at 14.5 Neptune Radii (Rnept = 24,670 km) and is retrograde (inclination = 157°). The spacecraft would then escape Triton and use both the electric propulsion (EP) thrusters and Triton gravitational assists to change the orbit plane and drop the periapsis down lower at Neptune. Since the desired final orbit is polar elliptical with respect to Neptune and with a periapsis inside the rings, a ring jump maneuver is required and can be performed by the EP system. The rings were assumed to cover a region between 1.6 Rnept and 2.6 Rnept. Once in this orbit, the 3 in-situ atmospheric probes could be released. Observations of the rings could be done in the different phases and orientations of the mission. The final Neptune science orbit is a 15-day polar elliptical orbit with periapsis at 1.5 Rnept.

One mission design option looked at early on was to add a phase at the end of this mission scenario that would bring the spacecraft to a low circular equatorial orbit around Neptune. The spacecraft would then be able to maintain a non-Keplerian orbit hovering above the rings of Neptune for rings observations. However the transfer from elliptical polar orbit to low circular equatorial orbit was too demanding ($\Delta V \sim 9.4$ km/s without gravity assists) to fit within the allowable $\Delta V$ budget. The $\Delta V$ for hovering above the rings for one month was estimated at about 0.7 km/s.

The scenario for this mission is quite complex since it requires orbiting different bodies, in different inclinations and orbit shapes. Table 5 summarizes mission phases and duration.

**TABLE 5.** NEP Neptune System Explorer Mission Scenario

| Phase | Starting Orbit | Final Orbit | Duration |
|---|---|---|---|
| Earth spiral out | LEO 2000 km | Earth escape | 2.4 years |
| Heliocentric trajectory | Earth escape | Neptune capture | 17.2 years |
| Nereid Fly-by and spiral down | Neptune capture | Triton orbit, i=157° | 8 months |
| Triton spiral in | Triton escape | Triton polar 700 km altitude | 1.8 months |
| Triton science | LTO 700 km | LTO 700 km | 10 months |
| Triton spiral out | Triton polar 700 km | Triton escape | 1 month |
| Transfer to Neptune Polar | Triton's orbit | Polar elliptical 31 day | 8 months |
| Transfer to Neptune Science | Polar elliptical 31 day | Polar elliptical 15 day | 3 months |
| Neptune science | | | 12 months + |
| | **FIRST SCIENCE RETURN** | | **20.4 yrs (!)** |
| | **TOTAL MISSION DURATION** | | **23 + yrs** |

## Sciencecraft Design

*Systems Overview*

The following system requirements were established early on in the study to scope the design: single Delta IV-Heavy launch, spacecraft 3-axis stabilization, full redundancy (with selected waivers on elements such as the high-gain antenna), and technology cutoff year of 2009-2010. Figure 3 provides a conceptual configuration drawing of the sciencecraft.

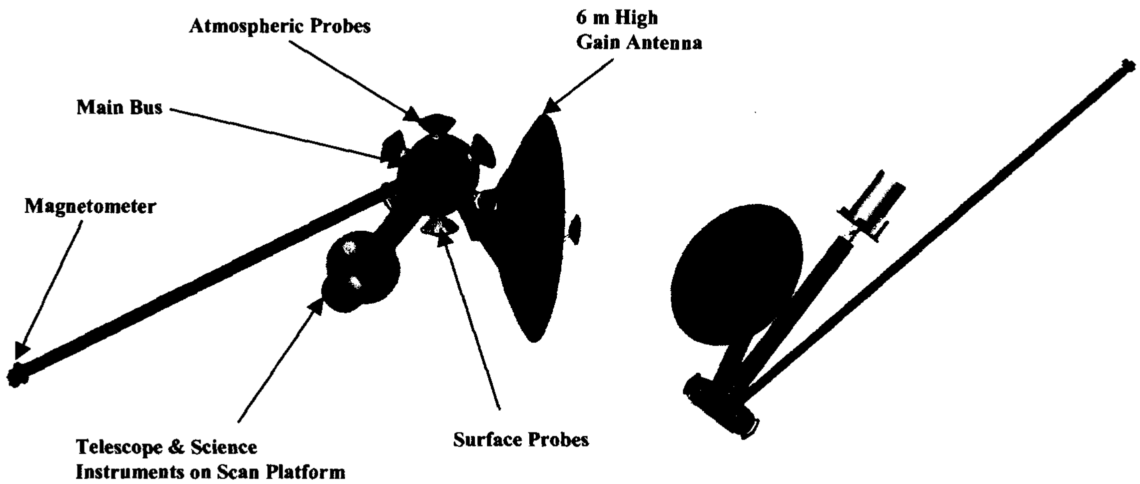


**FIGURE 3.** Conceptual Configuration of the Sciencecraft (NEP vehicle not shown)

The sciencecraft primary power would be provided by the 100 kWe nuclear reactor on the NEP vehicle and distributed by the power electronics at the sciencecraft. Reactor startup power would be provided by a small solar array and batteries. Three-axis stabilization would be maintained/corrected via the gimbaled EP thruster clusters, reaction wheels, and RCS thrusters (out on booms extended from the sciencecraft for momentum unloading). The sciencecraft and instruments would be nadir-pointed during observations.

The science data rate would be ~76 Mbps during peak observations. A simplified, decoupled scenario was used, composed of science data-taking 1/3 of the time, high-rate telecom 1/3 of the time, and 1/3 of the time as margin. After compression and coding, the high-rate downlink was sized for up to 50 Mbps from the spacecraft to Earth. The high-rate downlink would be implemented with a ~7.5 kW RF Klystron design for Ka-band communication through a 6 m gimbaled and deployed mesh HGA to the 70 m Deep Space Network (DSN). Both Medium Gain Antennas (MGAs) and Low Gain Antennas (LGAs) are also provided for command, telemetry, and as a low-rate communication backup. By demanding a total of 20.7 kW (27.0 kW if 30% contingency is assumed), the telecom high-rate downlink mode would be the driving mode in sizing the power subsystem.

For mass and power contingencies, the design for the spacecraft followed the JPL's design guidelines, and assumed 30% system-level growth contingency. Table 6 provides a mass breakdown (without contingency) of the sciencecraft.

*Radiation Environment*

The radiation environment is the combination of the natural effects as well as the environment created by the nuclear reactor. The various radiation components are shown in Table 7. They assume a 400 kWt or 100 kWe reactor, an 18-yr mission (slightly under the final converged mission design point's lifetime), a 0.9-m Be-LiH reactor shield (everything shielded by Be-LiH, that is, thrusters, radiators, and ducting are within the shadow of the shield), 1000 kg of Xe propellant, a payload at 25-m away from the reactor core, 2500 km/28.5 degree spiral-out and 100-mil aluminum shielding. Zero radiation design margin (RDM) is included in the results of table 7, but an RDM factor of 2 was used for the estimation of total radiation dose during the mission and sizing of the sciencecraft subsystems.

**TABLE 6.** Sciencecraft mass and power breakdown (no contingency)

| Subsystem | Mass (kg) | Power (@ highest demanding mode, W) |
|---|---|---|
| Instruments | 411 | 809 |
| Neptune Atmospheric Probes (3) | 240 | |
| Titan Surface Probes (3) | 285 | |
| Command and Data | 36 | 84 |
| Attitude Control | 99 | 467 |
| Telecommunications | 201 | 18811 |
| Power | 187 | 1263 |
| Thermal | 96 | 115 |
| Struct., Booms, Adapt., Mech. & Cabling | 491 | |
| **TOTAL** | **2046** | **(2660 kg with 30% contingency)** |

**TABLE 7.** Neptune System Explorer mission representative radiation environment, no RDM; 4-pi steradians spherical shell.

| | | Total Dose, krad [Si] | Neutron Fluence, $cm^{-2}$ |
|---|---|---|---|
| Natural Environment | | | |
| | Earth Spiral-out | 135 | $0.5 \times 10^{11}$ (note c) |
| | Solar Protons | ~ 2 (note a) | Negligible |
| | Neptune | ~ 5 (note b) | Negligible |
| Reactor | | 38 | $1.5 \times 10^{11}$ |
| **TOTAL** | | **180** | $\mathbf{2.0 \times 10^{11}}$ |

(a) Estimated based on previous Pluto study. Subject to actual trajectory definition and mission time frame (solar maximum or solar minimum)
(b) Based on Voyager data, the total dose level should not exceed 5 krads for a couple of years at Neptune.
(c) The level quoted is the one expected for 1-year at GEO due to trapped electrons. Note that the number quoted is the equivalent 1-MeV neutron fluence that would produce the same displacement damage as the trapped electrons actually present in the natural environment.

### *Spacecraft Avionics*

The spacecraft Command and Data System (CDS) assumed a technology cutoff of 2010 and design life of 25 years (to include appropriate design life margin). CDS would be cross-strapped, dual string redundant. ACS, Telecom, Propulsion, Power Distribution and Payload hard devices would use multi-drop, high-speed serial data and control interfaces. All electronic parts would be rad-hard to 300 krads total incident dose (TID) and a single event upset (SEU) threshold Linear Energy Transfer (LET) > 37 Mev/mg/$cm^2$. Each electronic assembly should have its own 100 mil or greater aluminum enclosure meeting the mission shock and vibration requirement. Enclosure thickness could be increased to meet TID radiation shielding of radiation-tolerant parts in ACS and Science subsystems. Localized spot shielding is an option ("dog house" cover over electronic parts). The mass memory requirement is 2.2 Terabits for 8 hours of data collections (76 Mbits/sec of science data would be collected for 8 hours). A serial port would be provided to the NEP controller/computer (26 meters away). The spacecraft CDS would control and monitor the NEP controller.

The technologies required for CDS include a processor capable of greater than 400 MIPs for science and engineering. A next generation compiler would be used similar to ANSI C++, and a next generation real-time operating system (RTOS) would be used similar to products from LINUX or VxWorks. Baselined 200 Gbit 4" harddisk drives may be available in 2010 and are enabling technology. Thirty of these disk drives would be flown. The 6-terabit unit would provide redundant storage of the 2.2 terabits of data between high-rate downlinks. Two disk drives would be powered on at any give time, and they are inherently rad-hard.

### *Attitude Control System*

The main Attitude Control System (ACS) design requirements are driven by the telecommunication and science subsystems. The High Gain Antenna (HGA) pointing accuracy should be less than 1 mrad in Neptune Orbit. The science instrument pointing accuracy shall be less than 1 mrad, and pointing stability shall be less than 1 μrad/sec in Neptune Orbit. The safe mode pointing accuracy should be less than 8.5 mrad in Neptune Orbit. A scan platform will be used for the science instruments, and the HGA will be gimbaled. To get sufficiently good pointing, the inertial reference units and trackers would be mounted to a thermally stable optical bench.

The ACS system would be fully redundant. It would include 2 star trackers, 2 inertial reference units each with 4 accelerometers and 4 gyros, 6 two axis digital Sun sensors providing 4 pi steradian coverage, 4 reaction wheels mounted in a 5 deg. pyramid configuration, two sets of 8 five N RCS hydrazine thrusters mounted in a cross pattern, and hydrazine tanks. Half of the RCS system would be mounted on the sciencecraft and the other half close to the Power Conversion System. A preliminary estimate for the propellant load is 200 kg and is sized to perform 2 start/stop maneuvers per day for 2 years. The Xenon propellant tank would be placed at the Center of Mass (CM) and have 2 booms extended from it: one to science, the other to the reactor and associated power distribution. The ion engines would thrust at the CM orthogonal to the spacecraft long axis. Thus, as the propellant is used, the CM shift is minimized. However, no calculation was done of the ion engine cluster gimbal angles.

During cruise, attitude sensing would be provided by a single star tracker and a single inertial reference unit (IRU). Control would be provided by gimbaling the ion thrusters. This control was assumed to be adequate to compensate for solar pressure induced torques, which dominate on the trip to Neptune.

The nadir science mode control would be provided by a star tracker, an inertial reference unit and reaction wheels. The primary disturbance torque is gravity gradient, and reaction wheels are sufficient to absorb the cyclic momentum. Aerodynamic and solar pressure torques are negligible in this mode. Secular momentum would be unloaded with thrusters. If inertial science mode control (for gravity mapping) is required, control will be more difficult. The gravity gradient torque is quite high as the spacecraft passes 45 deg. to nadir. It would be necessary to add more reaction wheels or a CMG to cover this mode. It was assumed that gravity mapping would be done when the spacecraft is nadir pointed with communication through the gimbaled HGA.

### *Telecommunications & Ground Systems*

The high data rate science requirement drives the design of the telecom and ground systems. The science instruments uncompressed data rate at Neptune and Triton would be ~76 Mbps, which translates into 50 Mbps compressed data rate downlink (including coding). A simplified, decoupled mission operations scenario was used, which sliced the day as science data-taking 1/3 of the time, high-rate telecom 1/3 of the time, and 1/3 of the time as margin. The communication with the Deep Space Network (DSN) thus assumed 8 hours per day of downlink.

The design of the Telecom system is based on an X-band frequency uplink operation of the DSN at an uplink bit rate of 2 kbps, and a Ka-band frequency downlink operation. At maximum range (31 AU), a 6-m deployable antenna would be used for high data downlink operation at Ka-band. The efficiency of this 6-m antenna was assumed to be 50%. This antenna would be mounted on a short boom and gimbaled. Extra fine pointing of the Ka-band antenna (0.1 mrad for pointing losses of 1 dB) would be achieved with a Ka-band monopulse system.

During Science operation, high-gain telecom is "off". Science data would be stored and transmitted later to Earth through the HGA. To provide a data rate of 50 Mbps during science downlink, the RF transmitted power needs to be 7.5 kW. This assumes a Bit Error-Rate of $10^{-5}$ (deemed acceptable for the enormous data volume), coding of rate 0.5 convolutional with constraint length of 7, with Reed-Soloman Coding added, and a data margin of 3 dB.

The power amplifier technology would be based on a new Klystron design rather than multiple traveling wave tubes (TWTs). The efficiency of the Klystron power amplifier is about 40%, which yields a DC input power to the Telecom system of about 18.75 kW. Besides the power amplifier, the system would use current designs and available transponders. Redundant systems and dual string of all telecom components was assumed.

The baseline design also assumes Ka band operation for all the MGAs and the LGAs, although it is possible to close the link using X-band antennas. During launch and cruise modes, Ka-band medium gain antennas of gain 15 dB can close the communication link with Earth. A 15 W RF transmit TWT was added as a low power provider in case the high power tube (Klystron based) fails to operate.

A quick assessment of a high-rate optical communication backup system was made. Although not discussed in detail here, a mass allocation was made for the optical system (planning to use the high-resolution telescope) and bookkept with the instruments/payload.

The ground system requires 2.2 Terabits of raw data storage per day (without margin and redundancy). This is equivalent to 250 Gigabytes of data storage per day, or about 2,400 Terabits of data storage for the life of the mission. This is for the raw science data only. Each day of the science mission, this mission will generate 85% of the total Cassini mission science data volume (measured in bits of raw instrument data), or about 938 times more data (measured in bits) than the Cassini mission. Ground data storage and on-board science data storage would be significant issues. The onboard storage will need to store enough data to handle the science data (including retransmission buffers) even if a tracking pass for one day has been missed. Onboard and ground temporary data storage (retransmission buffers) have not been sized to take into account the loss of one day of tracking.

The bundling protocols that would be used for this mission are presently in design. These protocols would provide automatic retransmission of all mission data. There is a low risk for assuming these protocols since it can be expected that they will be commonly used for deep space missions well before this mission's technology cut-off date of 2010.

The following new technologies would be required for the Telecommunication and Ground Systems:

- Design and construction of deployable large diameters (6-7) m mesh reflectors operating at Ka-band
- Design and construction of high power amplifiers with high efficiency(requires developing new cathodes and anodes to withstand the long-life time operation without failure or reduced efficiency)
- Manage high thermal power without degrading telecom
- Design and architecture of small deep space transponders with both X- and Ka- band operation with lower mass, power, volume, cost, and higher flexibility to be able to handle high power and high data rate availability for future deep space communication
- Develop DSN 70 m Ka- band capability.

*Power*

Only the power system that resides on the sciencecraft is described here. The Nuclear Reactor power system will be presented in the next section "NEP Vehicle Design". The sciencecraft power system consists of solar array and secondary battery, which were sized for launch and initial reactor deployment modes. The reactor would take four days to start-up (of the order of 100 hours), and would need about 800 W continuous load. In a 2000 km circular Earth orbit (~ 30-40 min eclipse), the combination of two 10-AH $NiH_2$ batteries based on CPV technology and a two-wing fixed 4.4 $m^2$ solar array based on Triple Junction Solar cells would provide power during the start-up mode. The batteries assumed a 20% DOD during the 'start-up' eclipses.

The power system also assumes a 120 V bus and will down-convert to provide 28 V to subsystems. Redundant power management and distribution (PMAD) electronics will be provided for each payload instrument and for flight system avionics.

*Thermal*

The basic thermal design uses a passive thermal control system with additional heaters, a pumped loop system to transfer telecom thermal dissipation and other spacecraft thermal dissipation, and interface isolation from the NEP vehicle. The thermal isolation from the NEP system would be with thermal conduction isolation and MLI to minimize thermal radiation interchange from the NEP stage.

The main heat rejection system for the spacecraft and propulsion system electronics is the payload/electronics radiator mounted on the NEP vehicle. Initial design concept is that a series of thermal transfer plates, heat pipes, and pumped loop system would be used to transfer the thermal energy from the spacecraft to the payload/electronics radiator.

In general, with the exception of the telecom system and the radar, the powers of flight elements are within the nominal dissipation, and standard thermal control techniques are used. The thermal control system design would use technology concepts that have been previously used, but new technology would be incorporated if developed. The high power dissipation of the telecom system and radar may require more efficient thermal energy transfer technology development.

*Structures/Configuration*

There were some needed configuration changes to the MSFC-led "Tiger Team" NEP vehicle design accommodate this mission. Locating the electric thrusters outside of the radiation shield's shadow cone would cause unmanageable radiation scatter back onto the science module, so all thrusters (and front corners of radiators) must be inside the shadow cone. But there isn't sufficient room inside the cone without either widening the shadow angle by increasing the size and mass of the shield or lengthening the boom substantially, which also increases mass. Even then, there is no good way to have the thrusters tucked in close and thrusting lengthwise without impinging on the structure, even if they are angled outwards, which causes cosine losses and increased propellant use. So the most feasible configuration seemed to be having the thrusters pushing sideways – but that will cause a major center of gravity (c.g.) shift as the fuel is depleted, unless the tank is co-located with the thrusters. Hence, the NEP vehicle configuration was revised to split the radiator/boom at the c.g. found without the tanks and thrusters, and the EP module was inserted at this point. Also, the thrust vector was oriented perpendicular to the lengthwise direction.

The Xenon tank was re-sized from 1.4 m diameter at 4500 kg Xe, by volume ratio, to 8200 kg Xe to yield approximately a 1.7 m diameter. The structure mass of the sciencecraft bus was estimated parametrically, based on the masses of the other subsystems and components which it supports, using established JPL methodologies.

Many factors went into determining the sizing and placement of numerous components. The radiators were shaped to fall within the shadow cone of the reactor and have a total area (on one side) of 168 $m^2$. The probes would be located so that they would have an open FOV to space even if the HGA fails in deployment or operation. Scan platform instruments would be located on a boom so that their FOV is not obstructed by the HGA. However, the fields and particles boom would obstruct the view of the scan platform instruments slightly. If the scan platform boom does not deploy, this location will allow data to be taken by turning the spacecraft (instead of rotating the platform). Fields and particles instruments are located on a 20 m boom to get them about 10 m from the scan platform instruments. This length is easily modified.

For the stowed launch configuration, it appeared that height within the launch vehicle was more of an issue than the diameter. The configuration assumes the sciencecraft on top. The electric thruster clusters need to be folded back to fit in the fairing. Folded radiator panel stacks fit, but their structural integration with the extendable main booms would require some clever engineering. The height of the c.g. above the adapter interface may be of concern.

## NEP vehicle Design

*TigerTeam 100-kW Vehicle Conceptual Design*

The NEP vehicle design is based on the MSFC "Tiger Team 100-kW Vehicle" design (Bonometti, 2003). Please consult the reference for details on that design. The objectives of this study were to design a sciencecraft and a mission that utilizes the Tiger Team NEP design. However, two major changes were done to that design during the course of this study. First, the configuration was modified to accommodate the ion engines and tank in the middle of the vehicle. The rationale for that changes is explained in the sciencecraft Attitude Control and Structures/Configuration sections above. The second change was to use ion engines with a higher power per engine and Isp than the Tiger Team design.

In summary, this 100-kW NEP vehicle is based on a Heat Pipe Cooled reactor. The shield is sized to provide less than 100 krads at the payload (20 m away) and less than $10^{11}$ fast n/cm$^2$. The half cone angle of the shield is 7.5°, which provides a shadow cone of about 3.3 m radius at 25 m away from the reactor. The 3 Brayton cycle power conversion system assumes a turbine inlet temperature of 1144 K. The Brayton converters are 40 kW each and there is an additional unit for redundancy. A sodium heatpipe cooled reactor with Nb1%Zr clad UN fuel was chosen for the point design and interfaces with the Brayton systems through three separate heat exchangers. The radiators are configured in 2 stepped triangular patterns. The working fluid is a HeXe mixture that dumps 256 kWt of waste heat to water heat pipe radiators via a interchange with a pumped liquid water loop. Radiator inlet temperatures of 491 K and a 330 K outlet result in approximately 240 m$^2$ of radiating area being required.

*System Design*

Figure 4 shows a conceptual configuration for the overall spacecraft. The design assumed that the Xe tank and thrusters would be located in the middle of 2 booms, not necessarily equally spaced, but rather sized such that the tank is located at the Center of Mass (CM) of the spacecraft. The reactor and shield are placed on one side of the boom while the payload occupies the other side, mainly to reduce the radiation environment. The power conversion system is located close to the reactor, although most of its power management system will be located on the sciencecraft side. The whole spacecraft, once stowed, will fit within the Delta IV Heavy shroud (~ 5 m x 19 m). The interface between the spacecraft and the launch vehicle is located on the nuclear reactor and power conversion side.

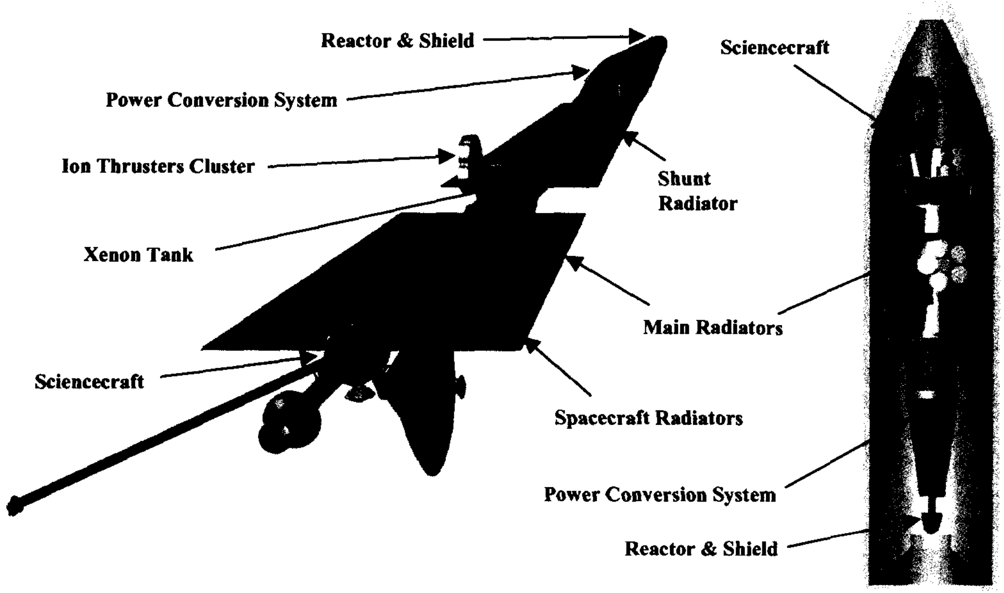


**FIGURE 4.** Overall spacecraft conceptual configuration (deployed and stowed)

Table 8 provides the spacecraft equipment list and also identifiess which component is included in which element of equation [3]. The mission design assumes this mass equipment list. Here again, 30% contingency was applied to all elements of the NEP vehicle.

Using Table 8, for this 100 kW system, the parameters of equation [3]:

$$M_{tot\_dry} = \alpha_{power} \times P_o + K_t \times M_{prop} + M_{NEP\_fixed} + M_{payload} \quad [3]$$

have the following values: $\alpha_{power}$ = 55.4 kg/kW, $P_o$ = 100 kW, $K_t$ = 19%, $M_{NEP\_fixed}$ = 1385 kg, $M_{payload}$ = 2660 kg. This equation equals the launch vehicle capability for 7450 kg of propellant.

*Electric Propulsion System*

The ion propulsion system (IPS) is based on the NEXIS (Nuclear Electric Xenon Ion System) ion thruster technology, currently being funded by the NASA In-Space Propulsion Technology Program. This technology was chosen for its high suitability (high power, high Isp) for this mission. The NEXIS engine is a 65-cm diameter ion engine that can process 20-50 kW of electric power and use Xenon as propellant. The thruster predicted characteristics at different power level are summarized in Table 9. The thrusters are powered by the Power Processing Units (PPUs), which convert the power from the transformers to the voltages and currents required by the engine. The transformers convert the output voltage of the turbo-alternators (and cabling) to a higher operating voltage for the PPUs. The efficiency of the transformers and PPUs was estimated at 0.95 each. Each PPU provides power to one thruster at a time, but is connected to 2 thrusters total in a crossed-trapped architecture.

**TABLE 9.** NEXIS ion thruster assumed characteristics

| **Power (kWe)** | **20** | **28** | **50** |
|---|---|---|---|
| Isp (s) | 7500 | 9000 | 12000 |
| Thruster Efficiency | 0.78 | 0.80 | 0.81 |
| Throughput (kg) | 1000 | 1250 | 2500 |

The feed system and PPUs are controlled by the Digital Control Interface Unit (DCIU), which accepts and executes high-level commands from the spacecraft computer and provides propulsion subsystem telemetry to the spacecraft data system. Each thruster was estimated at 20 kg, PPUs and transformers at 3 kg/kW, feed system at 2 kg per thruster plus 10 kg of fixed mass, and DCIUs at 3 kg each.

For the NSE mission, the optimized specific impulse is around 9650 sec, which implies a power per thruster of approximately 33 kW and a thruster efficiency of approximately 0.8. The thruster throughput is 1400 kg per engine. The total propellant mass would be about 8232 kg including 10% contingency. The EP system would be composed of 7 thrusters, 3 operating, 3 more for throughput and 1 redundant. Three PPUs would be needed for power requirements, plus one PPU, one thruster, and one DCIU added for single-fault tolerance. The cluster of thrusters is gimbaled (2 DOF).

**TABLE 8.** NEP Neptune System Explorer total mass breakdown (30% contingency)

| Subsystem | Mass (kg) | Mass + contingency (kg) | Mass Model |
|---|---|---|---|
| **Sciencecraft/Spacecraft** | **2046** | **2660** | $M_{payload}$ |
| **Power Generation & Conversion** | **4260** | **5537** | $\alpha_{power}$ |
| Reactor | 650 | 845 | |
| Shield | 362 | 471 | |
| Primary Heat Transport | 213 | 277 | |
| Reactor I & C | 125 | 162 | |
| Reactor system superstructure | 60 | 78 | |
| Reentry system | 50 | 65 | |
| Brayton Power Conversion | 1224 | 1591 | |
| Turboalternator units (3) | 321 | | |
| Recuperator (3) | 402 | | |
| Gas & Bleed Cooler (3) | 351 | | |
| Gas Ducting (3) | 150 | | |
| Heat Rejection System (HRS) | 823 | 1070 | |
| Power Management & Distribution | 210 | 273 | |
| Power Conversion and HRS structure | 75 | 98 | |
| Power Conversion and HRS cabling | 110 | 143 | |
| RCS Propulsion System + structure | 58 | 74 | |
| Trusses, Deployment Systems | 300 | 390 | |
| **Electric Propulsion System** | **1124** | **1461** | |
| Ion thrusters | 140 | 182 | $M_{NEP\_fixed}$, $K_t$ (2%) |
| PPUs, Transformers, DCIUs | 412 | 536 | $M_{NEP\_fixed}$ |
| Feed System | 29 | 38 | $M_{NEP\_fixed}$, $K_t$ (2%) |
| Tanks & Structure | 399 | 519 | $K_t$ (6.6%) |
| Structures, Mech. & Cabling | 144 | 187 | $M_{NEP\_fixed}$ |
| **Propellants** | | | |
| Xenon (10% contingency, Table xx) | | 8232 | $K_t$ (10.3%) |
| Hydrazine | | 200 | $M_{NEP\_fixed}$ |
| **Launch Vehicle Adapter** | | 344 | $M_{NEP\_fixed}$ |
| **TOTAL** | | **18434 kg** | **(LV capability: 18450 kg)** |

## NEPTUNE SYSTEM EXPLORER MISSION DESIGN

### Baseline Design

Given the 100-kW system, sciencecraft, and mission objectives described above, Carl Sauer, Greg Whiffen, and Anastassios Petropoulos (JPL) were able to compute trajectories that fit within the launch vehicle constraints. Table 10 summarizes the trajectory phases and expenditures. As this table shows, the flight times to reach Neptune with such a system are extremely high. The next section discusses how to reduce these times.

The starting Earth orbit is 2000 km circular. At that altitude the Delta IV Heavy launch vehicle capability is 18450 kg. After a long escape, due to the rather high optimized Isp, the spacecraft starts its interplanetary flight. Neptune is reached 19.6 yrs after launch, and the spacecraft is captured by Neptune. A fly-by of Nereid is planned before encountering and spiraling in to Triton. Once Triton's science is over, the spacecraft spirals out again, and after several Triton gravity assists reaches a Neptune polar elliptical orbit ($67^{\circ}$ plan change) with periapsis inside the rings at 1.5 Rnept. The final Neptune science orbit is a 15-day polar elliptical orbit with periapsis at 1.5 Rnept. One of the major questions was to check if a ring jump could be done in a single pass with the NEP system. It was found that it is in fact possible. Figure 5 shows all the features of the transfer between Triton's orbit and Neptune's polar orbit and of the ring jump. This trajectory was optimized using static dynamic control for a slightly lower Isp (9000 s) but also for a slightly lower initial mass (9500 kg instead of 11400 kg). It was assumed however that the $\Delta V$ would be very similar for the actual case. The minimum trip time Isp for the Earth to Neptune trajectory is 9652 seconds. The trajectories were run with a total efficiency of 70%. Increasing this efficiency could reduce the trip time by several months (~18 months for an efficiency of 76%), and it would not impact the propellant mass.

**TABLE 10.** NEP Neptune System Explorer mission timeline and $\Delta V$

| Phase | Phase duration (yrs/*months*) | $\Delta V$ (km/s) | Initial phase mass (kg) | Propellant (kg) |
|---|---|---|---|---|
| Earth escape | 2.4 | 6.14 | 18450 | 1163 |
| Heliocentric trajectory | 17.2 | 36.2 | 17290 | 5513 |
| Neptune's capture, Nereid Fly-by | 0.7 | 2.72 | 11792 | 335 |
| Triton's capture to 700 km altitude | 0.2 | 0.63 | 11458 | 76 |
| Triton escape | *1.1* | 0.50 | 11382 | 59 |
| Triton Fly'by 1 | *0.02* | 0.001 | 11323 | 0 |
| Triton Fly'by 2 | *2.0* | 0.69 | 11323 | 82 |
| Triton Fly'by 3 | *0.6* | 0.05 | 11241 | 5 |
| Triton Fly'by 4 | *1.0* | 0.02 | 11235 | 2 |
| Polar orbit insertion | *3.5* | 0.84 | 11233 | 107 |
| Ring jump | *1.0* | 0.105 | 11126 | 13 |
| Final Science orbit | *3.0* | 0.97 | 11113 | 113 |
| **TOTAL** | **21.5 yrs** | **48.9 km/s** | **11000 kg** | **7450 kg** |

### Sensitivities from Baseline Design

Using Equations [5], [6], [7] and the values found for equation [3], and also optimized trajectories, a parametric analysis was done, especially to investigate ways to reduce the flight time. When optimized trajectory data was applied to the equations, it was found that: **Flight Time ~ 1.5 $T_{burn}$** (for Neptune). So this approximation was used for the cases where no trajectory was available. Also, this analysis assumes that scaling of the NEP vehicle dry mass to higher power is done following a square root law. The term in equation [3] representing $\alpha_{power} \times P_o$ was scaled with the square root of the power ratios. The $M_{NEP\_fixed}$ was scaled as the 1/3 of the power ratio (might be optimistic),

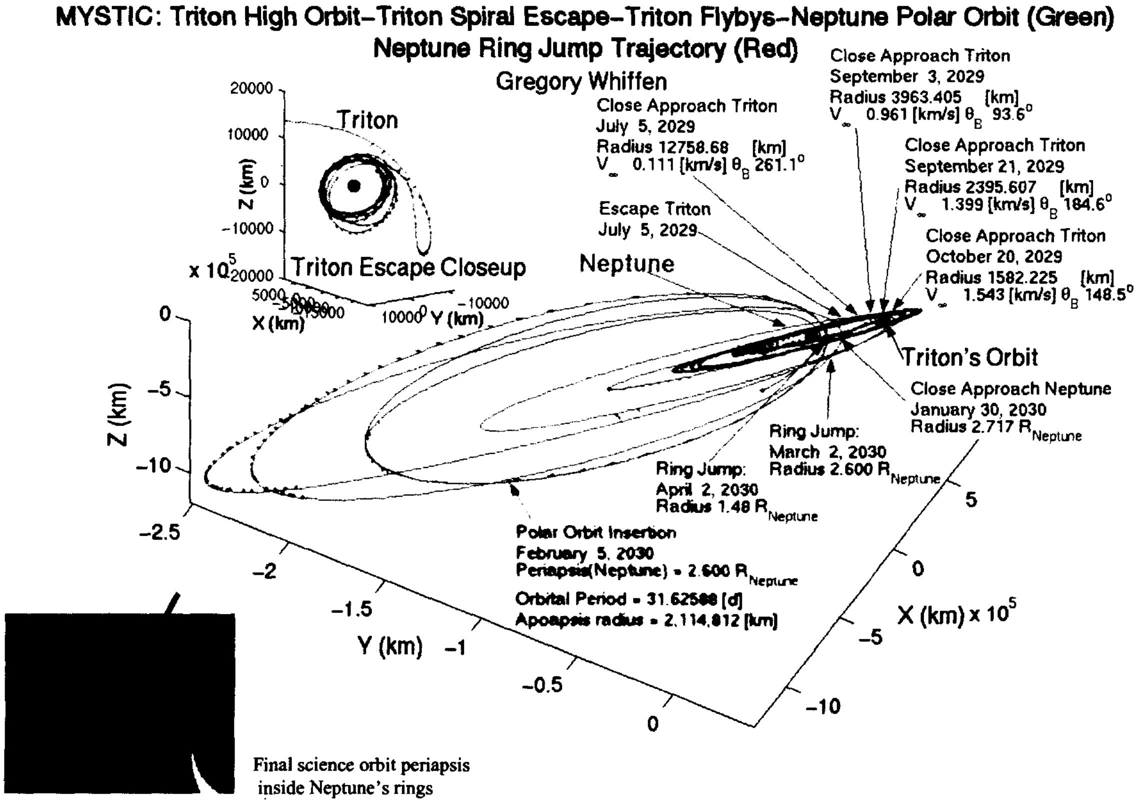


**FIGURE 5.** Representative trajectory transfer between Triton's low orbit and Neptune elliptical polar orbit.

and the $M_{payload}$ was kept constant. The scaling of the power system is based on historical trends, and might need to be reviewed.

Figure 6 shows the variation in flight time when Power level $P_o$ and launch mass $M_o$ are varied. Clearly, flight time is going to be reduced by increasing the power level and also increasing the launch mass. There is an asymptote though for which increasing the launch mass does not decrease much the flight time. And the benefit of increasing the power level is lower as power increases. This plot also shows that **significantly** reducing the flight time will require a significant reduction in the NEP dry mass (impacts $\alpha_{power}$ as well as payload and other fixed masses). The curves on the lower left of the plot are based on a 100 kW design with $\alpha_{power}$ = 35 kg/kW, $M_{NEP\_fixed}$ = 1000 kg, $M_{payload}$ = 1000 kg. ***The greatest impact on the NEP mission design to reach lower flight times and launch masses is the NEP spacecraft dry mass.*** Reducing the NEP spacecraft dry mass by a factor of 2 decreased the flight time by a factor of 1/3 and launch mass by a factor of 2.

Figure 7 is a snapshot of Figure 6 for fixed launch masses (18500 kg at 2000 km altitude, 21500 kg at 1000 km altitude, and 40000 kg for approximately 2 launches). The power level impact is captured in this view, and it shows an optimum for various launch masses. This optimum see to lay around 200-300 kW for the technology described in this paper and a single Delta IV Heavy launch. The use of multiple launch vehicles or reducing the NEP spacecraft dry mass will move that optimum to higher powers (> 500 kW), although the curves are flatter there and thus the benefit will be less as power increases.

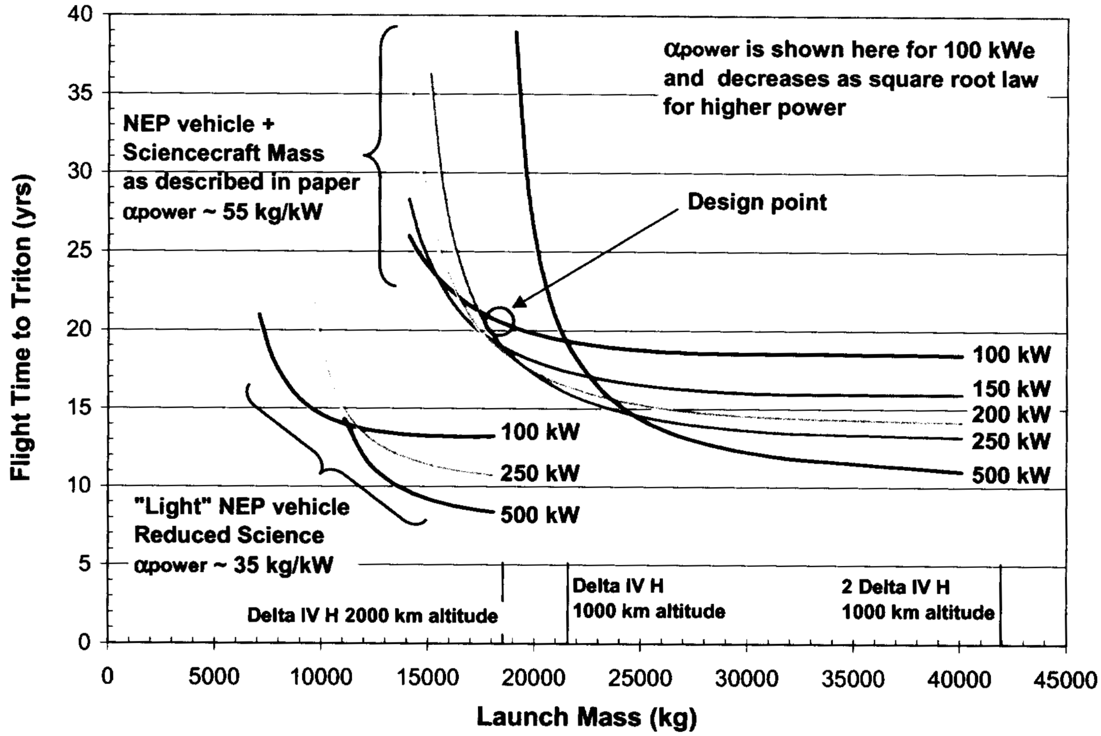


**FIGURE 6.** Launch mass and power level impact on Flight Time for NSE.

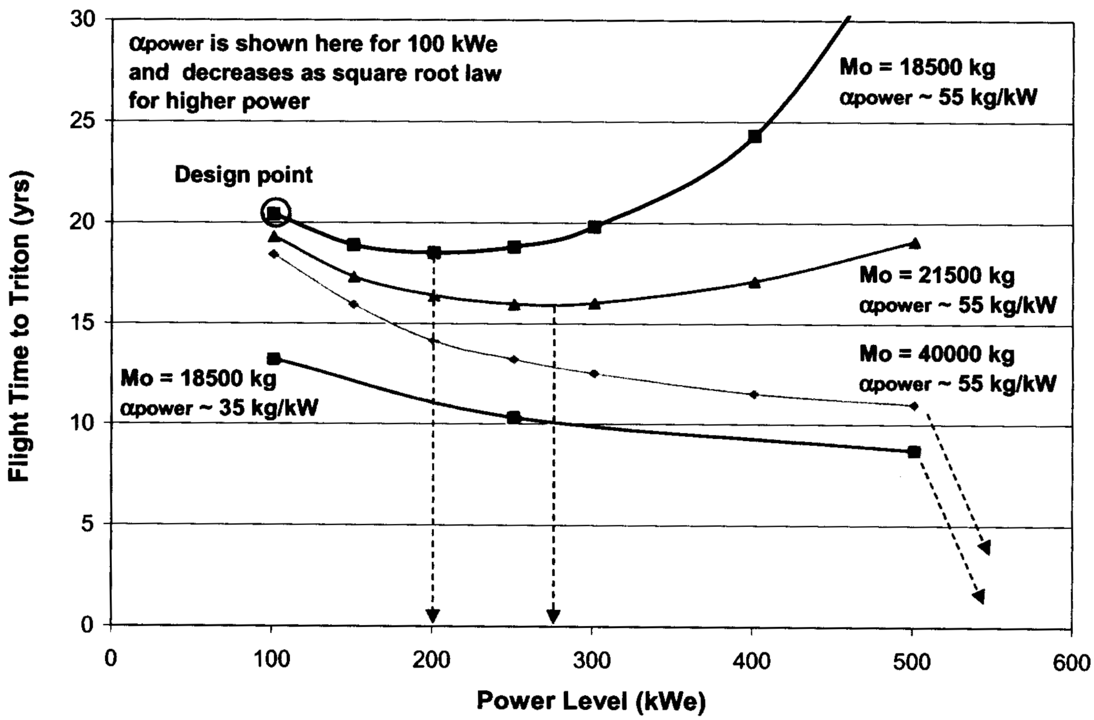


**FIGURE 7.** Launch mass and power level impact on Flight Time for NSE.

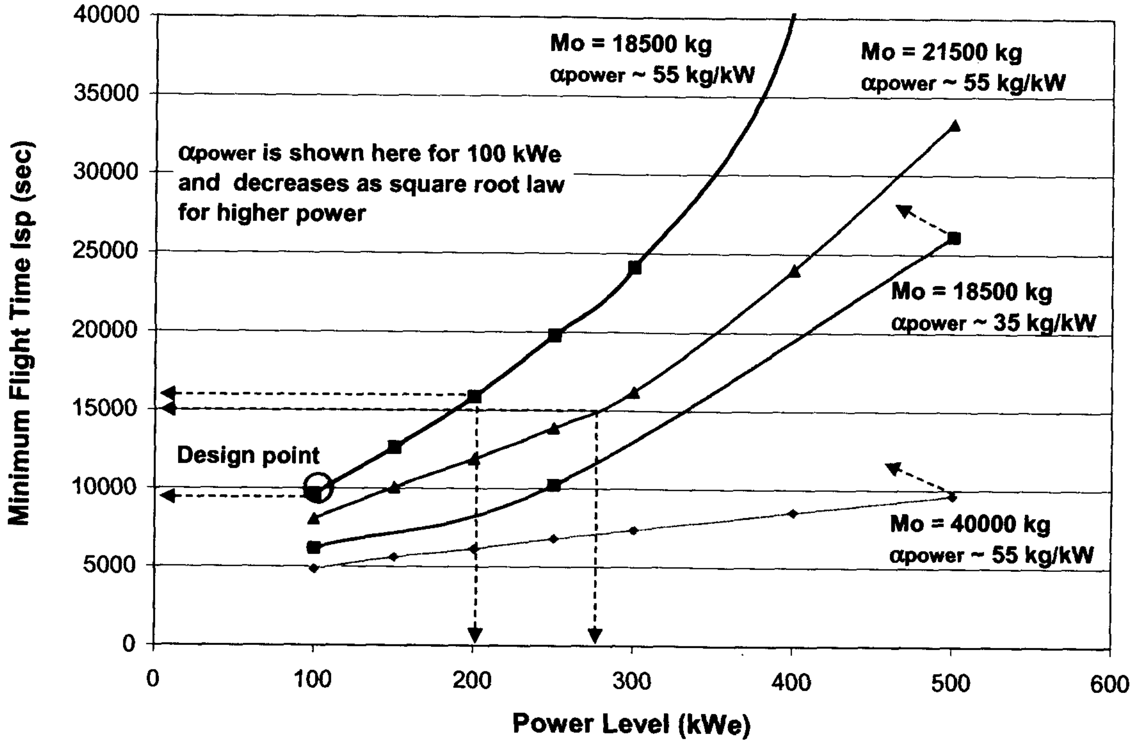


**FIGURE 8.** Launch mass and power level impact on thruster specific impulse for NSE.

Figure 8 shows the impact on the thruster specific impulse as power level increases. Here again, the Isp shown on this plot are for minimum flight times. There is an other Isp that will enable the mission, and that Isp is the minimum propellant mass one. This latter Isp will increase the trip time for the same payload/sciencecraft mass. For a given mission the minimum propellant mass Isp will be **higher** than the minimum trip time Isp. This figure shows that Isp in the range of 10000 s to 20000 s are to be expected for optimum powers above 200 kW. For multiple launches, the Isp will be reduced to a 5000 – 10000 s range.

And finally Figure 9 is for those who would like to reproduce previous Figures with the equations derived in the first section. It shows the wide range of variation in $\Delta V$ and how it is impacted by launch altitude, NEP vehicle mass and power level. Interestingly, the $\Delta V$ is not intuitive as its behavior as a function of power level will vary depending on the launch mass.

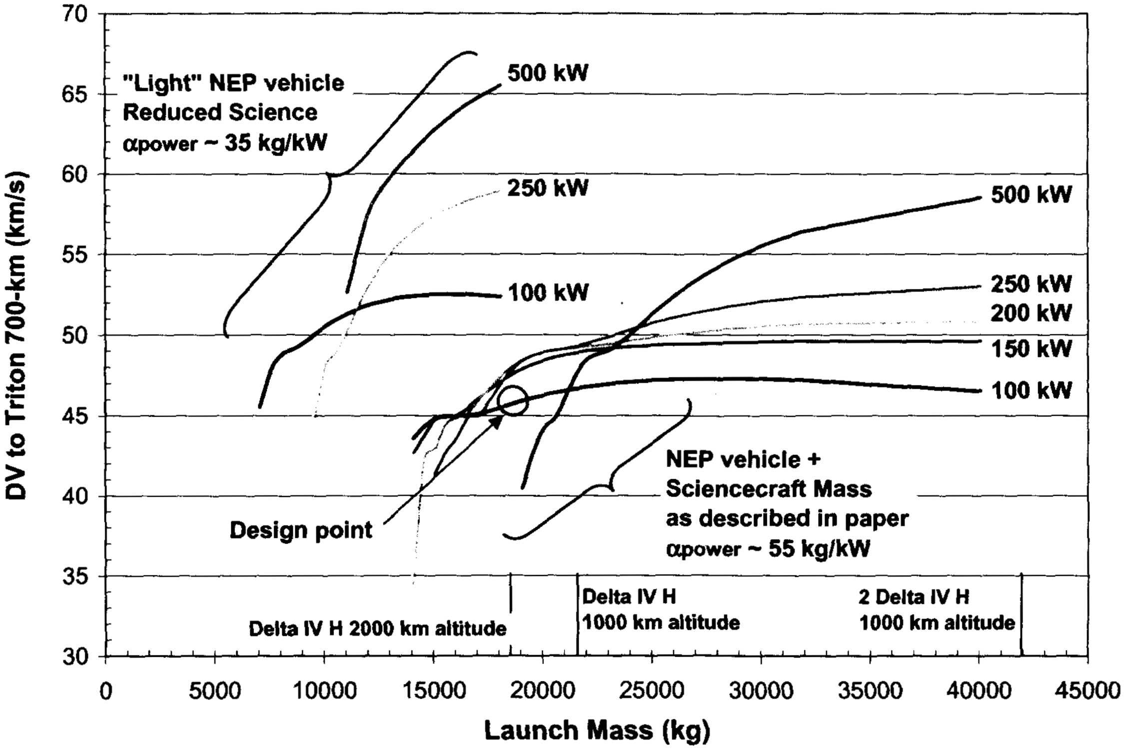


**FIGURE 9.** Trajectory ΔV as a function of launch mass and power level.

## CONCLUSIONS

This paper presents a set of relations that help the investigation and understanding of the NEP mission design trade space. These relations are used in the specific case of a Neptune System Explorer. The design of the "sciencecraft" part of the spacecraft is presented and details are given for all subsystems. Science objectives and instruments are also considered for this potential NEP mission. The design of the "NEP vehicle" part of the spacecraft was provided by MSFC and is summarized. The first result of this paper is that this scientifically attractive mission for an NEP application features a sciencecraft on the order of 2000-3000 kg dry, which is rather large (the Cassini spacecraft dry mass was approximately 2700 kg). The implications of the sciencecraft design on the NEP vehicle part of the spacecraft were also captured for the various subsystems. The high power potential at destination that NEP enables implies relatively heavy payload masses. However, the mission presented in this paper is highly capable, with 50 Mbps science data downlink from Neptune, or about 1000 times more data (measured in bits) than the Cassini mission.

The combination of the sciencecraft design and the NEP vehicle design done by MSFC is then the basis for the mission and sensitivity analysis. Given the current assumptions, the flight time to reach Triton is rather long (20 years), which is programmatically unacceptable. Ways to reduce that flight time are shown. The two parameters of the design space that have the most impact on flight time are the total spacecraft dry mass and the NEP electrical power level. To reach Neptune in 10 years will require an NEP spacecraft of about half the current mass and a power level of around 300 kWe, or an NEP system dry mass as described here but a launch mass probably equivalent to 3 Delta IV Heavy and above 500 kWe. The thruster specific impulse for the 20-year mission is around 9600 s, and the Isp for the cases that would reach Neptune/Triton in 10 years would be in the range of 8000 - 15000 s.

To refine all the results presented here, additional work should be done in defining more realistic NEP vehicle and sciencecraft masses (which means designs) for higher power levels (300 kW, 500 kW), since the results are very sensitive to that parameter.

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