Proceedings of Nuclear and Emerging Technologies for Space 2011
Albuquerque, NM, February 7-10, 2011
Paper xxxx; NETS Track 1/Session 1

Joint Radioisotope Electric Propulsion Studies – Neptune System Explorer

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Abstract. The Neptune System Explorer (NSE) mission concept study assessed opportunities to conduct Cassini-like science at Neptune with a radioisotope electric propulsion (REP) based spacecraft. REP is based on powering an electric propulsion (EP) engine with a radioisotope power source (RPS). The NSE study was commissioned under the Joint Radioisotope Electric Propulsion Studies (JREPS) project, which sought to determine the technical feasibility of flagship class REP applications. Within JREPS, special emphasis was given toward identifying tall technology tent poles, as well as recommending any new RPS technology developments that would be required for complicated REP missions. Based on the goals of JREPS, multiple RPS (e.g. thermoelectric and Stirling based RPS) and EP (e.g. Hall and ion engines) technology combinations were traded during the NSE study to determine the most favorable REP design architecture. Among the findings from the study was the need for >400W_e RPS systems, which was driven by EP operating powers and the requirement for a long-lived mission in the deep solar system. Additionally multiple development and implementation risks were identified for the NSE concept, as well as REP missions in general. Among the strengths of the NSE mission would be the benefits associated with RPS and EP use, such as long-term power (~2-3kW) at Neptune and flexible trajectory options for achieving orbit or tours of the Neptune system. Although there are still multiple issues to mitigate, the NSE concept demonstrated distinct advantages associated with using REP for deep space flagship-class missions.

Keywords: RPS, EP, REP, JREPS, NSE

INTRODUCTION

The JREPS project was commissioned by the NASA RPS program office in an effort to analyze flagship applications of radioisotope electric propulsion. The task was commissioned as a joint effort between the Jet Propulsion Laboratory (JPL) and the Glenn Research Center (GRC). After surveying science community interests and scoping out missions to targets that would test the limits of REP technology, it was decided to focus on NSE, which would be a flagship level mission concept. After initial trajectory, power, and propulsion trades by the project team, the concept design was fully analyzed with the assistance of JPL's concurrent engineering team, Team X. The goal of this report is to discuss the findings of this case study.

Though NSE is an early concept study (e.g. pre-phase A by NASA standards), much attention was given to the mission trade space with emphasis on REP enabling aspects of the mission. These initial trades and Team X assessment led to two viable architecture options based on the use of NASA's Evolutionary Xenon Thruster (NEXT) and either the 420W Advanced Radioisotope Thermoelectric Generator (ARTG) or 550W Stirling Radioisotope Generator (SRG). The baseline NSE mission would utilize either architecture and consist of a 2030 launch aboard a Delta IV Heavy launch vehicle for a 15-year transfer to Neptune and three years of science operations. The likely opportunity for the proposed NSE mission would be the 2030 NASA Outer Planets Flagship Mission.

The proposed mission would provide detailed measurements of the Neptune-Triton system from Neptunian orbit. The instrument payload is simulated as a black box, which would consist of a suite of in-situ and remote sensing instruments adapted from current deep space missions, to remain open to any future changes in science interests. The study examined the potential science return associated with transit and tour options about the Neptune-Triton system.

Although the baseline mission would not fit on the Delta IV Heavy launch vehicle from a mass perspective, it is within acceptable margin during this early stage of concept formulation. It is assumed that increases in launch vehicle capabilities would be available prior to a 2030 mission launch. The required power for the mission would be approximately 3 kW and a total of ~ 740 Gbits of science data would be downlinked over a nominal 37 orbits about the Neptune-Triton system spanning three years.

MISSION PURPOSE

Science

Exploration of the Neptune system would be premised about five key science questions, which form the basis of the proposed science observation requirements, orbital tour planning, and instrument selection:

- How and from what material did Neptune and its system form?
- How did Neptune's formation influence the development of the rest of the solar system?
- How have Neptune and its system evolved since its formation?
- What processes give Neptune the characteristics we observe today?
- How could knowledge of Neptune and the Neptunian system help us to understand exoplanets and exoplanetary systems?

A dedicated Triton orbit would likely not be feasible after Neptune orbit insertion due to significant increases in propellant mass. To accommodate for this, the baseline baseline orbital tour is designed to provide numerous Triton flybys during which measurements could be made at various distances and inclinations. Remote sensing instruments designed for Neptune science would also be capable of capturing Triton data upon approach; in situ measurements of dust and volatiles could be made during low altitude Triton flybys. While providing a complete description of the Neptunian system is likely beyond the bounds of technical feasibility, the proposed science requirements provide a reasonable framework for conducting a flagship scale Neptune mission.

Instruments

Due to mass constraints posed by the spacecraft system design, mass would be limited to 150 kg for the instrument suite, which could consist of multi-spectral imagers and spectrometers. Ample power would be provided by the RPS system following cruise, which would limit the available power at 1 kW - generally more than a 150 kg instrument suite would consume. Volume constraints were not taken into account though configuration for deployables, such as a magnetometer boom and radar antennae, were taken into account during spacecraft design. Attitude control pointing requirements are based on the operating parameters of a representative narrow field imager. During future iterations more detailed examination would be required to downselect specific instruments.

In addition to remote sensing instruments, probes would be deployed to yield Neptunian atmospheric data. The probe instrument package and its support mechanisms would utilize several instruments for in situ and bolometric measurements during atmospheric descent. Each probe would be allotted 150 kg (separate from the 150kg instrument suite allotment), including support equipment on the spacecraft. It is otherwise treated as a black box, capable of providing its own power and communication to the NSE orbiter through its descent to ~ 20 bar.

MISSION OVERVIEW

Mission Design

The mission architecture is based on a 15-year transfer consisting of directing the spacecraft towards Neptune through Jupiter and Earth flybys, and slowing it en route through the use of EP thrusters. Neptune orbit insertion (NOI) would be achieved through the use of a 2 km/s chemical propulsion maneuver, with a bi-propellant propulsion system. A trajectory plot with pertinent encounter parameters and thrust profile is provided in figure 1, where approximately 4 years of coast would occur in the middle of the trajectory. The strategy for combining chemical propulsion with EP is to use chemical propulsion for large maneuvers such as NOI and to use EP for small maneuvers between Triton flybys. The chemical and EP-based trajectory was found to be the most mass-optimal solution for the NSE concept.

A Neptune orbit of 2000 km altitude with a six-month period was selected for this study to maximize science return. The REP system could be used for orbital maintenance and to assist with orbital tour maneuvers, while the chemical propellant systems would perform NOI and also assist with tour maneuvers. Inclination changes during the orbital tour would allow coverage of a wide range of Neptune latitudes, and due to Neptune's 16-hour rotation period, sufficient longitudinal coverage would be assured. There would be a total of 37 Triton flybys in the orbital tour, which would offer a wide range of Triton ground tracks as well as radio science occultations of Neptune and Triton.

Two Neptune atmospheric probes would be released two months prior and arrive during NOI. The spacecraft would support a communications relay with the probes during NOI. The probes would have a 1.5 hour descent with relay ranges of 50,000 km or less to the orbiter and elevation angles as low as 2° above the local horizontal. This probe delivery scenario would add significant operational complexity since multiple critical events would occur simultaneously. Alternate strategies such as probe arrival on the periapsis following NOI could be investigated as a way to reduce this complexity.

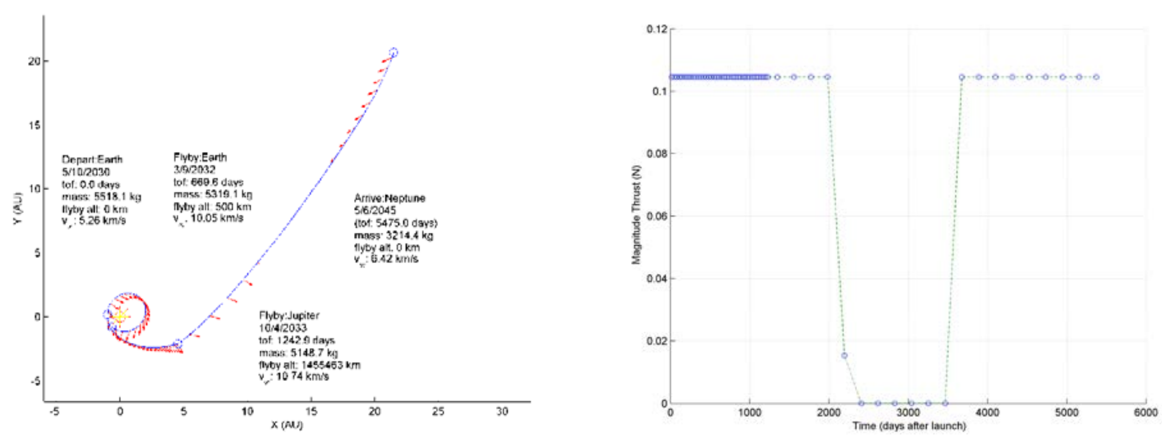

FIGURE 1. (Left) Baseline NSE trajectory plot and (Right) EP thrust profile.

System Overview

The proposed NSE mission would likely launch in 2030 and achieve Neptune orbit in 2045, followed by 3 years of science operations. The length and distance of this mission would be among the strongest requirement drivers of the mission. Being a flagship class mission, spacecraft subsystems would also require redundancy. As is standard with NASA missions, all required technologies for the mission would need to be at a minimum of technology readiness level six at least five years prior to launch. All spacecraft subsystems would also need adequate testing to ensure an 18-year operating lifetime. Much of the spacecraft and mission design would share heritage with Cassini and the Mars Reconnaissance Orbiter.

The proposed spacecraft would be launched aboard a Delta IV Heavy launch vehicle and utilize Earth and Jupiter flybys, as well as EP thrusting and chemical propulsion, to achieve the required trajectory and Neptune orbit. Communications would be maintained during transfer and science operations through the use of NASA's Deep Space Network (DSN). The attitude control system (ACS) would provide control for all coasting and gimbaling of the ion thrusters and high gain antenna. The thermal and structural subsystems would work together to ensure that all subcomponents have adequate support throughout the various phases of the mission and could withstand the extreme cold temperatures of the deep solar system. The science payload is currently being treated as a black box, based on a 150 kg and 380 W payload suite as well as two 150 kg atmospheric probes.

The spacecraft architecture would be based around either the 420W ARTG or the 550W SRG. The choice of either RPS would make minor mass and power differences, but would carry bigger risk implications in comparison to one another. The EP system would use NEXT in both architectures. The detailed similarities and differences between spacecraft architectures are shown in table 1. The combinations of RPS, EP engine, and mission trajectory determined the biggest driving factors of the mission during this study.

The system's driving power mode would occur during inner solar system cruise, at which time the EP system would operate at peak power to take advantage of the peak RPS power output. During the two Earth flybys, the spacecraft would need to utilize EP to meet trajectory requirements, while constantly communicating with Earth to ensure safe operations during the flyby maneuver. During deep solar system cruise, these constraints would be lifted because the spacecraft would not operate the EP and telecom systems simultaneously. The system power modes were designed to comply with the RPS power decay, which would decay based on the Plutonium-238 heat source.

The baseline mission trajectory has the capability of delivering 3214 kg to Neptune, and either RPS based option would satisfy this requirement. There would be insufficient mass margins for the current Delta IV Heavy launch vehicle capabilities, but these are acceptable based on the design accuracy at this early stage of formulation.

TABLE 1. Overview of NSE concept design characteristics

Subsystem	Details	420W ARTG based architecture mass	550W SRG based architecture mass
Top level system design	18 year Neptune mission: 15 year transfer and 3 years science operations	781.5 kg (mass at Neptune)	826 kg (mass at Neptune)
Launch vehicle	Delta IV Heavy with launch C_3 = 27.7 km^2/s^2. Performance capability = 5518 kg. Payload fairing diameter = 5m (static envelope diameter = 4.6m).	Launch mass: 5637 kg Comparison to launch vehicle: -119 kg	Launch mass: 5746 kg Comparison to launch vehicle: -228 kg
Science and instruments	Three-year Neptune/Triton orbital tour. Instrument suite and two atmospheric probes.	450 kg	450 kg
Power	Trade between 420W ARTG and 550W SRG. Dual centralized power electronics with full cross strapping.	416.1 kg Ten 420W ARTG units (no spares)	504.1 kg Nine 550W SRG units (one spare)
Propulsion	3 NEXT ion thrusters (1 spare) - assuming 74% efficiency and 3100 second Isp. Ultra light tanks, six 22N and nine 1N thrusters, two 623N AMBR engines with 333 second Isp (1 spare).	EP - hardware: 175.4 kg, propellant: 1466.7 kg Chemical propulsion - hardware: 73.9 kg, propellant: 1271.6 kg	EP - hardware: 173 kg, propellant: 1446.2 kg Chemical propulsion - hardware: 70.5 kg, propellant: 1252.9 kg
Attitude control system	Inertial measurement unit, reaction wheels, star trackers, star camera, accelerometers. Gimbal drive electronics for antennas and electric thrusters.	43 kg	43 kg
Structure and mechanisms	Probe spin ejection mechanisms, high gain antenna and UHF antenna gimbals, and ion engine support structure.	Structure: 533.4 kg Cabling: 129.7 kg	Structure: 547.4 kg Cabling: 141.7 kg
Thermal	IR waste heat thermal control (internal thermal louver), 40 RHUs	194.8 kg	190.5 kg
Computer data system	4 Gbyte NAND Flash (science data), 128 Mbyte NOR Flash (Flight Software), 512 Mbyte NOR Flash SDRAM. Supports 12 instrument interfaces.	28.3 kg	28.3 kg
Telecom	Two UHF 1.5m MGA, one X/Ka band 3m HGA, two X-band LGA, and one X-band MGA	72.7 kg	72.7 kg

SPACECRAFT DETAILS

Power

The power subsystem would be responsible for supplying and conditioning electrical power to all power consuming components throughout the spacecraft. These components range from the avionics of the spacecraft to the power-processing unit (PPU) of the electric propulsion system. The EP system would rely on the power subsystem to provide it with sufficient energy at the required voltage.

The mission trajectory assumed that high power EP thrusting would be done predominantly at the beginning of the mission when RPS output is at a maximum. However the spacecraft power system would still be required to output sufficient power to operate the EP system for orbital maintenance and tour assistance about Neptune, which would occur 15 years after launch.

Spares would be carried for the Stirling RPS units due to the fact that if a converter fails, the unit would output significantly less power than when fully operational. The 420W ARTG design is more robust than the Stirling options since individual thermoelectrics fail with only a small reduction in power output from the unit. As with the 550W SRG architecture, margin was included in the power requirements for the 420W ARTG case; however a full flight spare was not. The electrical power design remained constant regardless of the type of RPS assumed.

Inner solar system cruise would be the driving power mode for the proposed NSE orbiter due to the increased power draw from the electric propulsion system and simultaneous communications. Based on the power budget for this operating mode, the spacecraft configuration could be made up of either ten 420W ARTG (no spares) or nine 550W SRG (one spare).

The 420W ARTG is projected to have 14% conversion efficiency with an output power of 420 W_e and a system mass of 40 kg resulting in a specific power of 10.5W_e/kg at the start of the mission. It would achieve these parameters based on 12 General Purpose Heat Source (GPHS) modules, which contain a total of 5.28 kg Plutonium-238. The system's power output would decay at 1.6% per year, as is typical for RTG devices, which means that the output power would decrease to 330 W_e at 14 years after launch. The development of the 250W ARTG would be vital to the design of the 420W ARTG because they would share an almost identical design, except for the thermoelectric (TE) couple material. The material selection is based on developments in the Advanced Thermoelectric Convertor (ATEC) program, which is currently being researched at JPL for the 250W ARTG. Work on the 420W ARTG is being carried out indirectly, based on advances in the ATEC program.

The 550W SRG is projected to have 35% conversion efficiency with an output power of 550 W_e and a system mass of 54 kg resulting in a specific power of 10.2W_e/kg at the start of the mission. It would achieve these parameters based on 6 GPHS modules, which contain a total of 2.64 kg Plutonium-238. The system's power output would decay at 0.8% per year, which means that the output power would decrease to 488 W_e at 14 years after launch. The 550W SRG would leverage many of the design principles from the current Advanced Stirling Radioisotope Generator (ASRG) development. Although the 550W SRG would be larger and contain different structural materials than the ASRG, the functioning principles and controller design are assumed to be the same. There is not any current work being carried out on the 550W SRG by NASA GRC.

The NSE mission concept has made fairly aggressive assumptions for the design of the 420W ARTG and 550W SRG. Since neither of these devices is currently available, the specific power of these units is based on speculation. If these capabilities are not met in actual development, the overall power system mass would increase significantly.

Propulsion

The propulsion system must meet the trajectory and operations requirements set on launch, cruise, orbit insertion, and science orbits. The EP system would need to be able to provide enough thrust to support the trajectory, but also must be light enough to meet mass restrictions. It was determined through mission design trades that the best trajectory to follow would be to have the EP engines slow down the spacecraft along its route to Neptune following a Jovian gravity assist. Upon arrival at Neptune, the spacecraft would undergo a high-impulse insertion maneuver that could only be accomplished with a chemical bi-propellant rocket engine.

The baseline launch vehicle (Delta IV Heavy) must attain a C_3 of 27.7 km^2/s^2 for the mission trajectory. The biggest chemical propellant driver would be the Neptune orbit insertion. The ΔV budget for the chemical system would be 1080 m/s, which would be driven by NOI and also aids in orbital tour maneuvers. The biggest EP propellant driver would be the orbital tour of Neptune. The ΔV budget for the EP system would be 1280 m/s, which is driven by orbital maneuvers and the transfer cruise to Neptune.

The chemical propulsion system would consist of two AMBR 623 Newton bi-propellant engines with serving as a spare. This engine has been developed and flight proven to 140 lb_f thrust and 333 sec I_{sp} (35). The system also would consist of six 22 Newton and nine 1 Newton monopropellant thrusters for attitude control considerations. The EP system would consist of 3 NEXT thrusters with one serving as a spare. There is a spare because each NEXT thruster was assumed to be able to handle ~ 600 kg of Xenon before it starts to fail, thus, the two required thrusters would handle the ~ 1200 kg of Xenon propellant while the third thruster will serve as backup. The NEXT engine is able to provide the low thrust (e.g. mN levels) and high Isp (e.g. ~3000 sec) required by the trajectory. The EP system would have an average 3 kW power level available to it throughout the mission, and although not optimized to operate in the 3 - 3.5 kW power range, the baseline NEXT engine is projected to be fully capable of operating at this range. Its efficiency would slightly drop from optimal conditions and the thruster may be susceptible to failure modes, although more research is needed to verify this. The EP engines would be operated on a duty cycle so that all of the engines would survive until the end of the mission, rather than operating each one to failure.

The use of either pyrolytic graphite (PG) or carbon-carbon (CC) grids could enhance the throughput of NEXT thrusters to well beyond 1400kg. This result is unprecedented because it means that NSE could complete the mission with only one NEXT engine (with the mandatory spare for safety assurance). It quickly becomes apparent that having a high throughput EP engine would lower the complexity and mass of the spacecraft because fewer units would be required. This would prove crucial since both REP case studies were right at the limits of the current Delta IV Heavy launch vehicle capability. NASA GRC heads PG research, and JPL heads CC research.

Attitude Control System

The more typical constraints come from the precision in pointing knowledge and control required to accurately use remote sensing instruments. Pointing requirements are partially modeled off of MRO's HiRISE and Cassini's ISS, which has a maximum pointing knowledge of <0.001 degrees with stability and control of 0.02 degrees/second at 1000 km from Triton for imaging at a maximum resolution of >10 m/px.

Star trackers and sun sensors in conjunction with inertial rate sensors and accelerometers would provide attitude determination for the spacecraft. HGA pointing would be determined using star cameras and attitude control would be accomplished using a reaction wheel assembly (RWA), which would be desaturated using a monopropellant propulsion system.

Special care would need to be taken for the 550W SRG based spacecraft architecture to mitigate any complications that might arise from vibration. Slightly different operational frequencies among the SRG units could result in a beat phenomenon. These beats, like any vibration, would impose a stress on the subsystems of the spacecraft, especially pointing control. There are two methods to mitigate this risk, aside from damping and isolating the amplitude of vibrations. The first would be to stagger the operating frequencies of the Stirling engines such that they are several Hertz apart. While this would still create beats, the frequency would be too high to interfere with the spacecraft controller's bandwidth. In addition to this option, orienting the Stirling RPS along different axes would decrease the beat amplitude. Vibrations from Stirling engines propagate in the axis of the unit, thus placing them such that no two RPS lie on the same axis would reduce the amplitude of the vibration. There is no clear mitigation for the beat frequency issue at the moment and, assuming all other things equal, makes the 550W SRG option higher risk than the 420W ARTG option.

Structure

The spacecraft structure would be required to interface with and provide support for all other subsystems. The driving factor for the mechanical structure would be mounting and configuring the RPS such that they would not interfere thermally with each other or interfere with the science measurements of the instruments. Although the number and size of the RPS units would drive the configuration of the orbiter, the propulsion system would determine the size of the support structure. The sizable propulsion tanks must be supported to withstand the launch loads experienced when the tanks are full.

In addition to the RPS and propulsion support structure, the mechanical subsystem would also be required to support and articulate a large HGA for Earth communications. The HGA must be configured in such a way that it would allow for Earth communication without blocking the field of view of the science instrumentation. Its support must be strong enough to handle the large impulsive orbit insertion maneuver while simultaneously pointing to Earth for communication.

Though the number of proposed RPS units would not significantly impact the mass of the spacecraft structure, it does severely affect the spacecraft configuration. Trades determined the configuration of the RPS units based on the thermal and structural properties of the system. In order to keep the spacecraft center of gravity as low as possible, the RPS units would be mounted externally to the bus in two rings. The RPS units would all be cantilevered off the bus structure and located 60° from each other around the circumference of the bus. Additionally, the upper ring of RPS units would be shifted 30° from the lower ring to ensure that the units would not thermally interfere with each other. The design configurations for the 420W ARTG and 550W SRG based architectures are shown in figure 2.

The instrument package would occupy the remaining surface atop the proposed NSE orbiter in order to reduce exposure to the radiation effects caused by the RPS units at the bottom of the spacecraft. The radar box would be located adjacent to the HGA gimbal to allow for easy deployment. The magnetometer boom would be opposite of the HGA gimbal and would be deployed after the NOI maneuver.

A major risk associated with the baseline configuration of the spacecraft is the number of access doors that would be required to mount the RPS units prior to launch. Roughly 10 RPS units would be required to power the spacecraft and each might require a separate access door for installation on the pad. Based on current NASA practices of installing RPS units on a spacecraft integrated with the launch vehicle would make NSE spacecraft integration very challenging

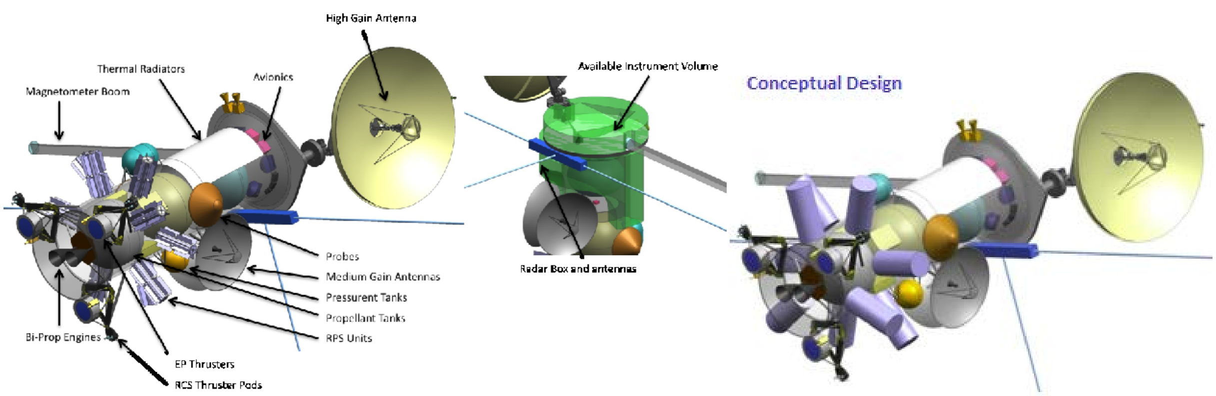

FIGURE 1. (Left, Middle) Conceptual design for NSE deployed configuration using 420W ARTG based architecture, with emphasis on dedicated instrument volume. (Right) Conceptual design for NSE deployed configuration using 550W SRG based architecture. Because the 550W SRG has minimal design heritage, the radiator surface was approximated as a cylinder.

Thermal

The thermal subsystem would be required to ensure that waste heat from the RPS units is properly rejected and that the spacecraft would be able to properly stay warm in the deep solar system. Instrument thermal control is assumed to be included in the allocated mass for the science payload. Due to this, the driving requirement for the thermal system would be temperature control of the propulsion system. Two separate propulsion systems and their propellants would be required to be kept at their operational temperatures in order to ensure mission success.

Although the propulsion system would drive the requirements for the thermal subsystem, the actual design is highly sensitive to the type of RPS used for the NSE orbiter design because thermal output from the RPS units could be harnessed and distributed across the spacecraft in place of heaters or other heat sources. The thermal design for the 420WARTG option of the proposed NSE orbiter would utilize a significant amount of Cassini heritage by harnessing the residual heat with the use of thermal louvers. Heat would likely be transported to keep the large propulsion tanks at room temperature. On the other hand, the high efficiency 550W SRG means there would be less residual heat to recycle into the spacecraft. In this case, additional thermal devices would supplement the thermal design. Electric heaters, radiators, multi layer insulation, thermostats and temperature sensors would also be used throughout the spacecraft for thermal control regardless of the type of RPS used.

Command and Data System

The NSE command and data system (CDS) would be responsible for the hardware to support storage, management, and commanding of science data, spacecraft telemetry data, and flight software. The CDS system must interface with the telecom, payload, power, and ACS subsystems for commanding and control. CDS must support up to 12 instrument interfaces, provide appropriate storage of science data that would be accumulated at a rate of 2 Gbits/day during the science phase of the mission, and withstand a total ionizing radiation dose of 50 krad (including radiation from space and radiation effects from the RPS).

The majority of the components that would be used are based on hardware developed for the Mars Science Laboratory (MSL) mission. JPL has developed this hardware specifically so that it could be a starting point for future missions. The design also assumed the use of software derived from MSL's architecture.

Telecommunications

The NSE telecommunications (telecom) system would be responsible for all spacecraft communications hardware and links. More specifically, the telecom system design must support science data and telemetry return, as well as command uplinks. General requirements include supporting a 2 way communications links between the spacecraft and Earth and supporting a one-way UHF link with the two science probes.

The telecom system design would be a fully redundant Ka, X, and UHF-band system. The Ka-band was assumed in this study for transmitting the science data at 100 kbps. The long range of this mission and the moderate data rate drove the use of Ka-band communications. The X-band system would be used for long range, low data rate transmissions (1kbps), near earth communications, and all uplinks (2 kbps). The UHF-band system would be used for acquiring the probe data at 2 kbps. A gimbaled HGA would be used to simplify spacecraft operations, especially critical events like Neptune orbit insertion and science data downlink. It was assumed that NSE would communicate to the DSN's 34m and 70m receiver stations.

Ground Data System

The ground system would primarily supports requirements relating to science data return, spacecraft health monitoring, and spacecraft tracking. As a result, the ground systems element design is a collaborative design between the telecommunications, computer data systems, and mission design subsystems. The ground system would support tracking for critical events, which include an Earth flyby, Jupiter flyby, and Neptune orbit insertion. Ground system tracks would also return the biweekly accrual of 20 Gbits science data biweekly with a latency of less than 10 days. The communications link would lasts at least 6 hours, including no less than 4 hours of Doppler tracking during spacecraft cruise. Additionally, 10 Mbits of probe data gathered during Neptune orbit insertion and spacecraft health updates would be supported.

The NSE ground system design would be driven by these requirements as well as mission round-trip light-time delay of 8 hours, ground station visibility, and ground station capability. The ground system design would use the Deep Space Network (DSN) for telecommunications, where both the 34-meter and 70-meter antennas were assumed to be operational throughout the duration of the mission. Prospectively, the largest ground systems risk is that a 70-meter antenna (or equivalent) would not be available for use during the mission. If so, the mission could utilize a single 34-meter antenna and make one of the following alterations: lengthen the pass each day from 5.6 to 8 hours, provide more power to the telecom system, or enlarge the spacecraft telecom dish to 4 meters. All mitigations would ensure that the mission would still be above the science floor.

RISK ANALYSIS

The purpose of the NSE concept study was to not only investigate a specific REP mission, but also make an assessment of difficult requirements for REP missions in general. Although the NSE concept had some unique mission risks, it also gave insight into multiple risks that would affect multiple REP missions.

The major NSE concept specific risks consist of simultaneous critical events during NOI, induced beat effect from multiple SRG units, assumed availability of the 70m DSN ground station, and negative mass margin on the launch vehicle. As mitigations ideas for each of these issues have already been discussed, the highest priority should be given to lowering the risk associated with the NOI scheme since this could potentially endanger the entire mission. It is assumed that these issues could be solved in further iterations of the NSE concept.

The NSE concept risks that also represent general REP risks would be multiple RPS integration at the launch vehicle and plutonium-238 availability. Large-scale REP missions would likely require multiple RPS units, which would need to be integrated with the spacecraft once it is already integrated with the launch vehicle. In recent history, Cassini has integrated the most RPS with 3 GPHS-RTGs, and the NSE concept would require three times as many RPS with the use of either nine 550W SRG or ten 420W ARTG units. This would add a much greater degree of difficulty due to the number of access doors needed on the launch vehicle shroud. In comparison, the availability of Plutonium-238 fuel for the RPS would be a much bigger hurdle to overcome for REP missions. REP missions of any class will not get past the concept phase, if this issue is not dealt with in the near future. As it currently stands, NASA's available Plutonium-238 resources are already dedicated to other missions. REP advocates should ensure that any future supplies include a dedicated supply for a REP mission, such as NSE.

Among the most important considerations for future iterations of this study are the projected availability of Plutonium-238 and the integration of multiple RPS on the launch pad because these considerations would set limits on the type of REP applications that could be accomplished. Specific to NSE, the probe implementation strategy must be studied in detail because the current baseline of simultaneously retrieving probe data during orbit insertion would be highly risky and alternative strategies must be implemented.

CONCLUSION

Of all the various trades made in the NSE study, the 420W ARTG and 550W SRG based architectures proved to be the only worthwhile options. Options involving current RPS technology, such as the ASRG, proved to add too much complexity and mass to the spacecraft design because more RPS units would be required to achieve the kW level power demand. In the end it was found that a bigger RPS building block would be vital to doing a flagship-class REP mission, like NSE.

The NSE study has touched on multiple areas of interest to both the science and technology communities. It was found that there could be a feasible option for orbiting the Neptune system with a REP-based spacecraft. It is important to note that this feasibility is established with respect to current and near term technology. More important than the current design are certain requirements that the spacecraft must fulfill, which remain constant regardless of technology. This distinction is crucial because these requirements should be carried through for any future technology considerations, of which there should be many since this mission would not launch until 2030. In addition, multiple issues and risks have been identified that should be better understood and mitigated in future iterations of this study. Among the most important results of this initial look at NSE is that it has expanded the scope of REP to giant planets and shown that such a mission could be feasible.

NOMENCLATURE

ACS = attitude control system
ARTG = advanced radioisotope thermoelectric generator

SRG	=	Stirling radioisotope generator
CDS	=	command and data system
DSN	=	Deep Space Network
EP	=	electric propulsion
GPHS	=	General Purpose Heat Source
GRC	=	Glenn Research Center
JPL	=	Jet Propulsion Laboratory
JREPS	=	joint radioisotope electric propulsion studies
NEXT	=	NASA's Evolutionary Xenon Thruster
NSE	=	Neptune System Explorer
NOI	=	Neptune orbit insertion
PPU	=	power-processing unit
REP	=	radioisotope electric propulsion
RPS	=	radioisotope power source
TE	=	thermoelectric
W_e	=	Watts electric

ACKNOWLEDGMENTS

The research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

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