Deep Space Mission Applications for NEXT: NASA's Evolutionary Xenon Thruster

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NASA's Evolutionary Xenon Thruster (NEXT) is designed to address a need for advanced ion propulsion systems on certain future NASA deep space missions. This paper surveys seven potential missions that have been identified as being able to take advantage of the unique capabilities of NEXT. Two conceptual missions to Titan and Neptune are analyzed, and it is shown that ion thrusters could decrease launch mass and shorten trip time to Titan compared to chemical propulsion. A potential Mars Sample return mission is described, and comparison made between a chemical mission and a NEXT based mission. Four possible near term applications to New Frontiers and Discovery class missions are described, and comparisons are made to chemical systems or existing NSTAR ion propulsion system performance. The results show that NEXT has potential performance and cost benefits for missions in the Discovery, New Frontiers, and larger mission classes.

Nomenclature

$k_{tankage}$ = tankage mass fraction (includes structure) (%)
$k_{payload}$ = payload mass fraction (%)
$\dot{m}$ = mass flow rate (mg/s)
P = power (kW)
T = thrust (mN)
α_{power} = power specific mass (kg/kW)
$M_{propellant}$ = Xenon propellant mass (kg)
$M_{SEP\text{-}Fixed}$ = fixed system mass (kg)
$M_{payload}$ = Mass of payload (kg)
$M_{payload+chemical}$ = Mass of the payload and chemical system (kg)
$M_{SEP\text{-}Dry}$ = Dry mass of SEP (kg)
M_{launch} = Launch mass ($M_{propellant} + M_{payload+chemical} + M_{SEP\text{-}Dry}$) (kg)
R_J = Radius of Jupiter
V_∞ = Velocity at the sphere of influence with respect to the target planet (km/s)

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I. Introduction

The NASA's Evolutionary Xenon Thruster (NEXT) was selected in 2002 for technology development by the NASA Headquarters Office of Space Science, Solar System Exploration Division under the Next Generation Ion (NGI) Engine Technology NASA Research Announcement (NRA).[1] NEXT is part of the Next Generation Electric Propulsion (NGEP) project within the In-Space Propulsion Technology Program managed by the NASA Marshall Space Flight Center (MSFC). The primary objective of NGEP is to significantly increase performance for primary propulsion for deep space missions by leveraging NASA's very successful ion propulsion program for low-thrust applications. As part of NGEP, a significant effort has been made to identify and analyze deep space missions that might benefit from the use of NEXT technologies. This paper provides an overview of systems analysis conducted on seven potential mission applications for NEXT. The missions span a range of destinations and classes.

- Outer Planet Missions: Titan Explorer, Neptune Orbiter
- Mars Missions: Mars Sample Return
- New Frontiers Missions: Comet Sample Return, Jupiter Polar Orbiter
- Discovery Missions: Multiple Asteroid Rendezvous, Near Earth Asteroid

All of the missions considered use solar arrays as the power source for electric propulsion (SEP). Conceptual missions using Nuclear Electric Propulsion (NEP) are not addressed by this study.

This paper begins with a brief overview of the NEXT subsystem and describes performance assumptions made for the mission analysis. Then for each mission concept we describe the objectives of the analysis, provide a brief description of the proposed mission's goals and architecture, list major assumptions, explain the analysis methodology, and summarize results. This is a survey paper, intended to provide an overview of analysis conducted to date, and the amount of detail provided is limited in some cases. Much of the analysis represents ongoing work, and further details are available from other sources (as referenced) or can be obtained by contacting the authors directly.

II. NEXT Subsystem Overview

The NEXT project is developing ion thrusters, advanced power processing, xenon propellant management and gimbal technologies that will advance the state-of-art to meet the needs of certain future deep space missions. The NEXT project is intended to advance the technology to NASA Technology Readiness Level (TRL) 6, with the exception of full thruster life demonstration. TRL 6 requires system/subsystem model or prototype demonstration in a relevant environment. As a key element of advancing NEXT technology readiness, critical component tests will be performed to qualification levels, with subsequent design/analysis updates to support transition to flight development. The effort will provide sufficient maturity and risk reduction to enable prudent selection of the technologies for a space mission in 2006. The development is being conducted in two phases, with breadboard level development and integration already completed in the one-year Phase 1, and engineering model development and integration of a multi-thruster system in the 2.5-year Phase 2 that was initiated in October 2003.

The technology elements of the NEXT ion propulsion subsystem are illustrated in Figure 1. These include the High Pressure Assembly (HPA) and Low Pressure Assembly (LPA) of the Propellant Management System (PMS), the Power Processing Unit (PPU), thruster and gimbal. Other elements of ion propulsion flight systems, including the xenon tank and system control unit, are not within the scope of NEXT technology development.

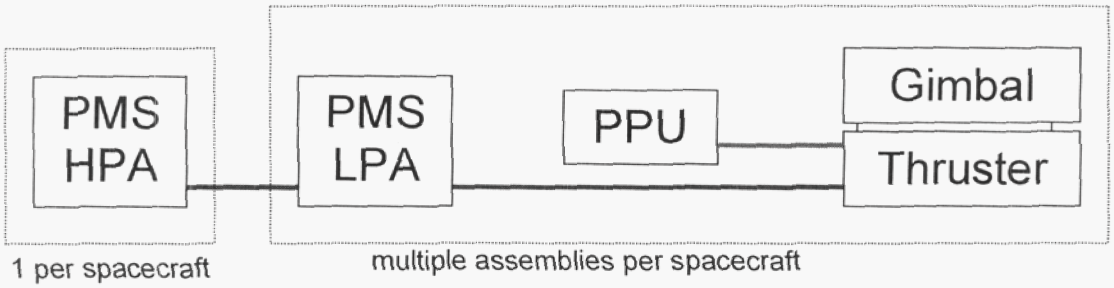


Figure 1: Elements of the NEXT Ion Propulsion Subsystem

The NEXT 40-cm thruster incorporates design improvements beyond NSTAR (the 30 cm diameter thruster for Deep-Space 1).[2] These improvements include: a magnetic circuit designed for improved beam flatness and reduced discharge losses; high-temperature stabilized rare-Earth magnets; a compact propellant isolator, and advanced ion optics and discharge cathode designs for longer life. Figure 2 illustrates the features of the NEXT thruster, with an image of the engineering model thruster developed in Phase 1 of the NEXT Project.

NEXT Thruster Characteristics

- 0.6 – 6.9 kW input power
- Ring-cusp electron bombardment discharge chamber
- 40 cm beam diameter
- 2-grid ion optics
- Beam current at 7 kW: 3.52 A
- Maximum specific impulse > 4100 sec
- Maximum thrust > 237 mN
- Peak efficiency > 70%
- Xenon throughput > 270 kg, 405 kg qualification level
- Mass target: 12 kg

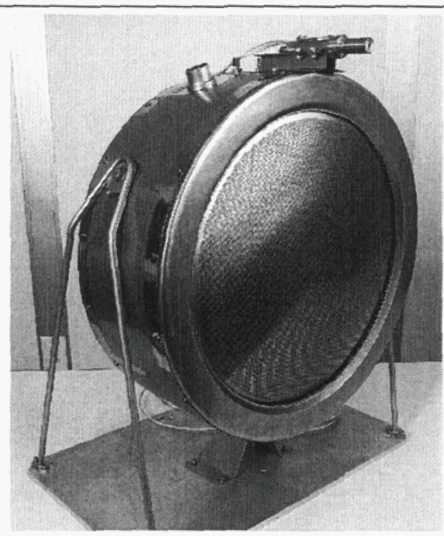


Figure 2: NEXT Thruster Technology Approach

The Power Processing Unit combines a technical approach previously developed by Boeing Electron Dynamic Devices and NASA GRC[3,4] with NSTAR-heritage approaches. A new modular supply approach provides high efficiency for the beam supply across the throttle range. Other supplies, including discharge, accelerator, neutralizer and heater supplies, are based on NSTAR designs, providing low development costs and risks. PPU characteristics include:

- 0.6 – 7.2 kW Input Power
- Peak efficiency > 95%
- Primary input power voltage range 80 – 160 V
- Mass target < 26 kg

The Propellant Management System represents a significant departure from the NSTAR technical approach.[5] The PMS LPA is built around a flow control kernel consisting of a Moog Proportional Flow Control Valve (PFCV) and a new Aerojet-designed thermal throttle for each of the three xenon feeds to a thruster. The thermal throttle consists of heaters and temperature sensors integrated onto a Mott sintered-plug flow control device. The flow control kernel has both a pressure control loop and temperature control loop to precisely provide the xenon flow rates within ± 3% of the appropriate thruster throttle setting. Upstream of each flow control kernel is the PMS HPA, which provides first stage pressure regulation. This overall approach is expected to significantly reduce PMS mass and volume over the NSTAR SOA approach, while improving aspects of the system performance.

NEXT system performance goals have evolved since initiation of the project.[6,7] Originally designed as a 1 – 10 kW thruster, the power range was re-scoped to 1 - 6 kW in response to the design requirements provided to the project. With thruster performance demonstrated in Phase 1, the upper power range was expanded to 7 kW. This allowed performance improvements for the design reference missions with one fewer thruster string. For the higher power missions, polynomial fits generated from the phase 1 throttle table were used for trajectory analysis and optimization. The polynomial fits and the resulting system efficiency and specific impulse (I_{sp}) are shown in Figure 3 below.

- NEXT Phase 1 Performance Polynomials, High Thrust (Low I_{sp}) Curve:

$$\dot{m} = 4.481 - 3.977P + 2.177P^2 - 0.364P^3 + 0.0197P^4 \text{ (mg/s)}$$
$$T = 46.850 - 27.944P + 29.436P^2 - 4.963P^3 + 0.265P^4 \text{ (mN)}$$
$$P_{min} = 1.252 \text{ kW}, P_{max} = 7.455 \text{ kW}$$

- NEXT Phase 1 Performance Polynomials, High Specific Impulse (Low Thrust) Curve:

$$\dot{m} = 3.630 - 1.7266P + 0.6470P^2 - 0.07184P^3 + 0.002892P^4 \text{ (mg/s)}$$
$$T = 36.467 + 3.7746P + 6.882P^2 - 0.6815P^3 + 0.02334P^4 \text{ (mN)}$$
$$P_{min} = 1.252 \text{ kW}, P_{max} = 7.455 \text{ kW}$$

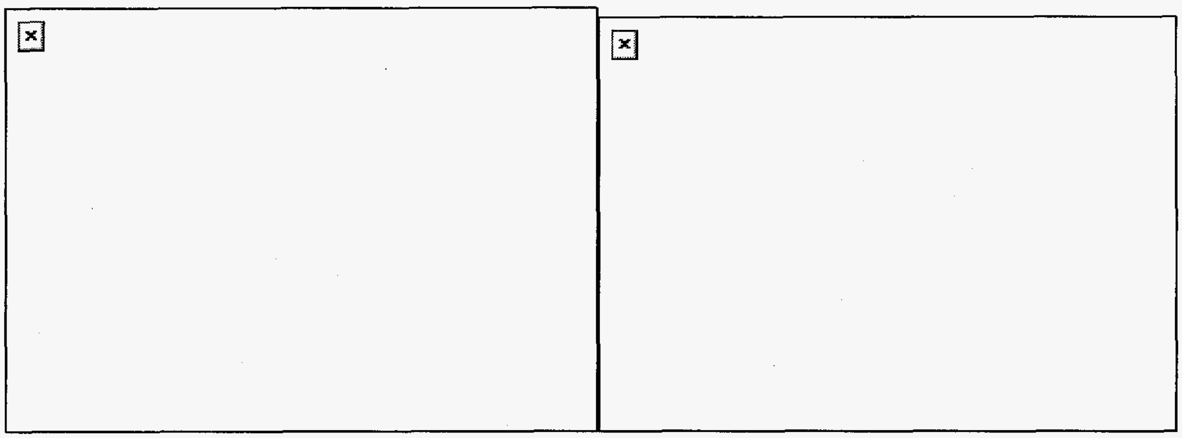


Figure 3: NEXT Phase 1 Throttle Table, Specific Impulse and System Efficiency

As the project conducts Phase 2, with the reduced emphasis on the high power missions that the original requirements were directed to, the project has begun looking towards lower power missions. In response to early analyses of Discovery- and New Frontiers-class missions, emphasis has been on increasing low power performance parameters. Using test data from the Phase 1 testing, current baseline NEXT thruster performance has been revised to increase thruster efficiency across the throttle table and to add throttle points as low as 500 W thruster input power. The revised thruster throttle table performance points have been demonstrated in Phase 2 tests. System performance below the peak power points has also been improved by adding PPU beam module addressing so that fewer modules can be operated at more optimal efficiencies during low power operations. This table, referred to as the Phase 2 throttle table, ranges from 0.74 to 6.9 kW thruster input power. Finally, an additional, more aggressive thruster throttle table, referred to as NEXT Table 9A, has been developed to determine if further performance improvements at low power are important technology objectives. For lower power missions, this throttle table was used for mission analysis. The polynomial fits and the resulting system efficiency and specific impulse are shown in Figure 4 below.

- NEXT Table 9A Performance Polynomials, High Thrust (Low Isp) Curve:

$$\dot{m} = 3.131 - 2.621P + 1.660P^2 - 0.2832P^3 + 0.01531P^4 \text{ (mg/s)}$$
$$T = 16.80 + 11.88P + 12.930P^2 - 2.234P^3 + 0.1102P^4 \text{ (mN)}$$
$$P_{min} = 0.616 \text{ kW}, P_{max} = 7.252 \text{ kW}$$

• NEXT Table 9A Performance Polynomials, High Specific Impulse (Low Thrust) Curve:

$$\dot{m} = 2.2087 - 0.1757P - 0.041P^2 + 0.05205P^3 - 0.00466P^4 \text{ (mg/s)}$$

$$T = -0.7077 + 51.75P - 13.510P^2 + 2.835P^3 - 0.1843P^4 \text{ (mN)}$$

P_{min} = 0.616 kW, P_{max} = 7.252 kW

Figure 4: NEXT Throttle Table 9A, Specific Impulse and System Efficiency

Table 1 indicates which throttle tables were used for analysis of the each of the candidate missions.

Mission	NEXT Throttle Table
Titan Explorer	Phase 1
Neptune	Phase 2
Mars Sample Return	Phase 1
Comet Sample Return	Table 9A
Jupiter Polar Orbiter	Phase 2
Multiple Asteroid Rendezvous	Table 9A
Near Earth Asteroid	Table 9A

Table 1: NEXT Throttle Tables used for Mission Analysis

III. Beyond New Frontier Class Outer Planet Applications

This section describes outer planet mission applications for NEXT, matching the original Deep Space Design Reference Missions (DSDRM's). The project investigated both Titan Explorer and Neptune Orbiter missions during Phase 1 of the project to establish a requirement baseline and evaluate system performance metrics. At the beginning of Phase 2, additional activities were conducted to perform a more detailed investigation of the Titan Explorer mission, including definition of a SEP module. The SEP module concept is useful in understanding both mission performance and ion propulsion system interface conditions. This section describes the overall mission architecture of the DSDRM's including the proposed trajectory, thrust profile, and overall spacecraft configuration.

A. Mission Analysis and Performance, Titan

1. Mission Description

Titan Explorer's mission objective is to deliver a 34 kg. science payload to a 1700 km circular orbit around Titan and a 365 kg. in-situ probe to the surface of Titan. The baseline SEP mission architecture is shown in Figure 5.

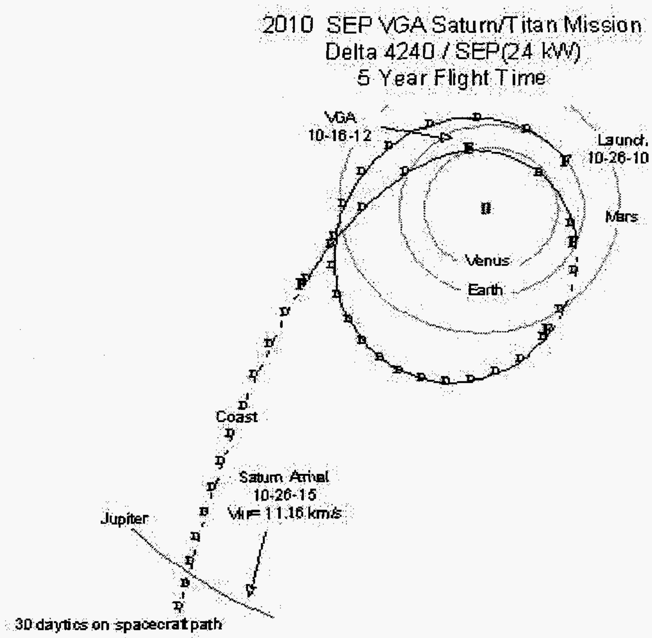


Figure 5: Titan Explorer Mission Overview
(Composite Profile: Does not show the Trajectory used in this Particular Study)

The spacecraft consists of an orbiter, SEP module, and in situ-probe which are launched together on an Earth escape trajectory. The SEP module uses multiple NEXT thrusters to provide primary propulsion as the spacecraft orbits the sun and performs a Venus Gravity Assist (VGA) approximately two years after launch. The module is jettisoned as the spacecraft moves away from the sun (beyond 3 Astronomical Units) and orbit insertion at Saturn is conducted using either chemical propulsion or an aerocapture system. After insertion, the science mission is conducted by an orbiter and an in-situ probe. The in-situ probe is released prior to Titan Orbit Insertion (TOI). This is a class A mission requiring full redundancy and 30% mass and power margins are maintained for all elements of the spacecraft except the SEP system. A 30% dry mass margin, 8% propellant mass margin, and 5% power margin are maintained for the SEP system.

2. Objective of Analysis

The analysis objective is to define the basic configuration of the SEP module and to compare the performance of missions using aerocapture and chemical propulsion for Titan Orbit Insertion (TOI). Four different SEP-aerocapture configurations are studied to determine the optimum power level and number of operational NEXT thrusters for the module. A SEP-chemical configuration is also studied to look at the relative costs and benefits of this architecture.

A separate analysis conducted for the 2004 In-space Integrated Space Transportation refocused studies (IISTP II) compared the performance of these SEP based architectures to an all-chemical Titan Explorer mission. A brief performance comparison, derived from this work, is also presented in this section.

3. Spacecraft Analysis and Design

The spacecraft was designed by JPL's Advanced Projects Design Team, ("Team X"), which uses a real time collaborative engineering team to quickly develop new mission and spacecraft concepts and designs. Trajectory and mission analysis support was provided by NASA Glenn Research center. The lander, aerocapture system, and orbiter science payloads were derived from previous work by Noca et al.[8] In the aerocapture missions, the orbiter contains the science instruments, aeroshell, attitude control sensors and avionics, radioisotope thermoelectric generators (RTG's) and a small monopropellant chemical propulsion system for attitude control and orbit circularization after the aerocapture maneuver. In the chemical insertion scenarios, the orbiter's aeroshell is replaced by a large bipropellant chemical propulsion system used for Saturn Orbit Insertion and Titan Orbit

Insertion (TOI). The required ΔV is substantial, almost 3000 m/s plus multiple gravity assists, resulting in a much heavier orbiter than in the Aerocapture cases. The mass of the lander and orbiter for both scenarios are shown in Table 2 below.

Element	SEP + Aerocapture	SEP + Chemical Insertion
In-Situ Probe Mass	365	365
Orbiter Wet Mass	1159	2355
Payload Total	1524	2720

Table 2: Titan Explorer Orbiter/In-Situ Probe Mass Summary

The SEP module contains the electric propulsion system, solar arrays, and thermal control system required to operate three or four NEXT thrusters simultaneously. The module contains no communications or attitude control system, as these functions are provided by the orbiter. Power is provided by 4 or 6 ultraflex solar arrays providing 24 or 36 kW of power respectively, at beginning of life (BOL) at a distance of 1 astronomical unit (A.U.) from the sun. A single redundant thruster and power supply are included in all configurations.

The orbiter, in-situ probe, and SEP module are shown in their stowed configuration in Figure 6 and in their deployed configuration in Figure 7. The orbiter is shown with the aeroshell required for aerocapture. The SEP module is separated from the orbiter prior at approximately 3 A.U. and the in-situ probe is separated from the orbiter prior to TOI. On aerocapture missions, this means that the lander is separated prior to the aerocapture maneuver. On chemical missions, this means that the lander is separated after Saturn Orbit Insertion and before TOI.

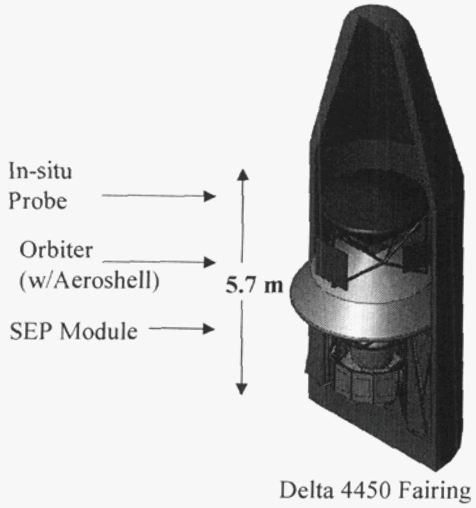


Figure 6: Titan Explorer in Stowed Configuration

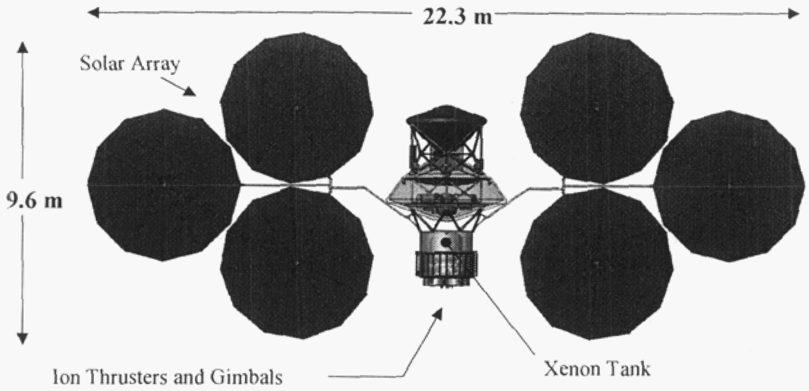


Figure 7: Titan Explorer in Deployed Configuration

4. Results

Five options were studied to determine the best SEP module configuration for the mission. The objective was to provide similar launch mass margins for each design and then calculate the relative cost of the options. Table 3 summarizes both the assumptions and results of the four SEP-aerocapture and one SEP-chemical architectures considered. The high specific impulse throttle profile is used in all of these cases because it provided better performance than the high thrust profile. The costs given are in $FY2004 and are calculated relative to the baseline case. No total costs were generated in this analysis.

	Baseline	Option 1	Option 2	Option 3	Option 4
Thruster Configuration	4 + 1	3 + 1	3 + 1	3 + 1	4 + 1
Cruise Time	5.9 yrs.	5.9 yrs.	5.65 yrs.	5.9 yrs.	8.0 yrs.
BOL Solar Power	24 kW	24 kW	24 kW	23.15 kW	35 kW
Saturn Insertion	Aerocapture	Aerocapture	Aerocapture	Aerocapture	Chemical
Launch Vehicle	Delta 4450	Delta 4450	Delta 4450	Delta 4450	Atlas 551
Mass Margin	68 kg.	108 kg.	72 kg.	77 kg.	-32 kg.

Table 3: Titan Explorer Mission SEP Architecture Options

Note: Option 4 cost assumes an Atlas 551 launch vehicle. The next larger launch vehicle, a Delta IV 4050H, adds several tens of millions of dollars to the total mission cost.

Comparing the baseline to option 1 shows that the use of three operational NEXT thrusters rather than four increases overall mass margin on the launch vehicle. This occurs because there is not enough array power available to operate the fourth thruster for much of the mission. As a result, the marginal mass benefit from adding the fourth thruster is more than offset by the additional mass of the thruster, PPU, and associated structure and thermal control hardware. Options 2 and 3 are variations on option 1 that lower the cruise time and array power respectively until the launch vehicle mass margin is approximately equal to the margin in the baseline case. Option 2 has the lowest overall cost of all the options considered, showing that lowering cruise time is more economical than lowering power level. Option 2 also defines the optimum module configuration for SEP-aerocapture. This option has 3 operational + 1 redundant NEXT thruster (3+1) and a total power of 24 kW.

Option 4 uses chemical propulsion for orbit insertion at Saturn/Titan and has considerable mass and power penalties when compared to option 2. The SEP-chemical case requires a much heavier orbiter, much larger solar array, an extra thruster and PPU, and a larger launch vehicle as well as requiring more time to reach its destination. In addition, the mass margin is slightly negative on the selected launch vehicle, indicating a risk that the mission will be forced onto the next largest vehicle. The next vehicle, a Delta 4 Heavy, costs several tens of millions of dollars more than the Atlas. Therefore, while this option reduces technology risks associated with aerocapture, it adds considerable cost and mass risk to the program.

	Option 2	Option 4	Chemical Option A	Chemical Option B	Chemical Option C
Thruster Configuration	3 + 1 NEXT	4 + 1 NEXT	All Chemical	All Chemical	All Chemical
Gravity Assist	VGA	VGA	EJGA*	VEEGA	VVVGA
Launch C3 (km^2/s^2)	14.3	14.3	25.6	12.0	9.7
Cruise Time	5.65 yrs.	8.0 yrs.	~8.15 yrs.	~10.4 yrs.	~9.8 yrs.
Saturn Insertion	Aerocapture	Chemical	Chemical	Chemical	Chemical
Launch Vehicle	Delta 4450	Atlas V 551	Delta 4 Heavy	Atlas V 531	Delta 4 Heavy
Mass Margin	+72 kg.	-32 kg.	+970 kg.	+500 kg.	+650 kg.

* The earliest usable launch opportunity for this option is 2015

Table 4: Titan Explorer SEP missions compared to Chemical Mission Options

Table 4 compares the performance of SEP-aerocapture and SEP-chemical architectures to three all-chemical mission options studied as part of IISTP II. The SEP-aerocapture architecture provides superior performance to the all-chemical options both in terms of cruise time and launch vehicle. Chemical option A is competitive with the SEP-chemical option in terms of trip time, but requires a larger launch vehicle. In addition, the Earth-Jupiter gravity assist (EJGA) opportunity selected for this option occurs infrequently, and the first usable opportunity available is in 2015. Both the SEP assisted VGA opportunity and the chemical Venus-Earth-Earth Gravity Assist (VEEGA) and Venus-Venus-Venus Gravity Assist (VVVGA) opportunities occur much more frequently. Chemical option B has a longer flight time than the SEP-chemical option, but uses a smaller launch vehicle. However, the use of an Earth gravity assist for a mission may add cost and complexity to mission operations related to planetary protection. Chemical mission option C is the best ballistic option that occurs frequently and does not requires an Earth gravity assist. It has a longer trip time and uses a larger launch vehicle than the SEP-chemical option.

It should be noted that the SEP-aerocapture and SEP-chemical missions may also benefit from the use of multiple gravity assists. Additional work is needed to consider this option for improving SEP mission performance.

5. Summary

Overall, from a performance viewpoint, NEXT is well suited to the Titan Explorer mission, particularly when combined with aerocapture for Titan orbit insertion. The SEP-aerocapture architecture performs well, both in terms of mass and cruise time, and the use of NEXT simplifies the design of the SEP module by limiting the number of operational thrusters required to accomplish the mission.

B. Mission Analysis and Performance, Neptune

In the 2002 solar system exploration decadal survey, the Neptune Orbiter mission was listed as the second highest priority mission (after the Jupiter Polar Orbit mission) for giant planet research.[9] Because less is known about Neptune's atmosphere, magnetosphere, and rings than Jupiter's, a comprehensive mission is required and trade-offs between the orbit, payload and power, and other mission parameters are necessary. Multiple probes are expected with at least one probe's orbit having a low periapse altitude. To address some Neptune mission parameters, the analyses for the Neptune Deep Space Design Reference Mission study (DSDRM), as defined in the NEXT NRA, are discussed.

1. Transportation Approach

To be consistent with the DSDRM requirement to use the smallest enabling launch vehicle, the NEXT SEP module is to be launched on a Delta IV 4240. The SEP module is injected onto a high-energy hyperbolic orbit to Neptune at perigee. The trajectory is such that the SEP module builds up the required energy in the inner solar system with the continuous low thrust provided by NEXT. It then uses a Venus flyby to redirect its energy to Neptune. Figure 8 shows the SEP portion of the Neptune trajectory. The bold curve depicts the thrusting portion of the trajectory.

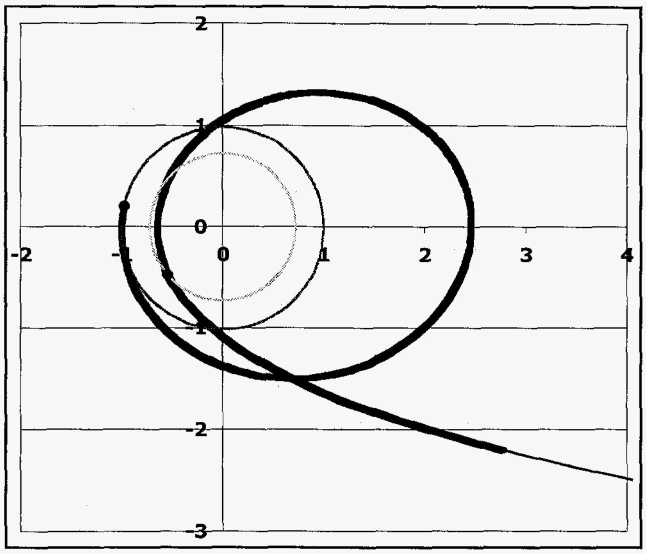


Figure 8: NEXT portion of Neptune trajectory.

The SEP module is jettisoned after thrusting is complete, near 3 A.U. The spacecraft continues for an additional 27 A.U. on its transfer to Neptune and is captured with an aerocapture system. A small SOA chemical system performs a small ΔV to raise periapse to avoid atmospheric re-entry and to circularize the final parking orbit.

2. Mission Assumptions

The payload and power requirements for a Neptune mission were defined in the DSDRM. The payload was assumed to be 850 kg, including the mass of the aerocapture system. In the previous Neptune DSDRM analysis the aerocapture system was assumed to be an advanced aeroshell.[10] During the IISTP 2004 Focused Studies, ballutes (an inflatable drag device) were assumed to perform the capture.[11] The payload requirement for a Neptune orbiter mission during the IISTP Focused Studies was 1,070 kg. It is not clear whether this difference in payload requirement is due to different aerocapture systems or some other factor(s). Regardless, both systems assumed the same entry velocity constraints.[11] This analysis did not focus on the method of aerocapture, but instead on performance of the NEXT system in order to meet the DSDRM performance requirements. Solar array power was defined to be 30 kW at 1 A.U. and it was capped at 25 kW maximum. It was assumed in the DSDRM that the solar arrays could be manipulated in such a way as to not exceed this power limit during the Venus flyby. For this analysis the solar array power was not capped at 25 kW; all 30 kW were available to the propulsion system with the exception of 250 Watts reserved for housekeeping activities. The reason for this is twofold: (1) to take advantage of the possible performance benefits associated with an additional 5 kW available to the propulsion system (with no mass penalty), and (2) NEXT's dry mass has grown at least 10% since the last published Neptune DSDRM analysis.[10] A complete list of assumptions is shown below.

- Mission Assumptions:
  - Launch date: 2010
  - Capture orbit parameters not investigated
  - Maximum Entry Velocity Constraints:
    - 28 – 30 km/s => 16 – 20 km/s arrival V_∞
- Launch Vehicle Assumptions:
  - Delta IV 4240
  - 10% launch vehicle contingency
- SEP Stage Assumptions:
  - 4+1 thrusters/PPUs
  - High I_{sp} throttle curve
  - Two, three, or four thrusters operating
  - P_{max}: 7.255 kW
  - P_{min}: 0.817 kW

5% Xenon propellant contingency for flight errors
3% Xenon propellant contingency for reserves
90% propulsion duty cycle
174 W/kg solar arrays (150 W/kg with structure)
$1/R^2$ array power
2% per year array degradation
250 Watts supplied for housekeeping activities
Power System Specific Mass: 9.7 kg/kW
Xenon tank mass fraction: 3.4% of total propellant
Power and Thermal Structure: 16%
Stage cabling (not PPU): 6% of power mass
Tankage Mass Fraction (includes structure [26% dry] and 3% Xenon propellant contingency): 13.7%
Payload Structure Mass Fraction: 8.5%
Spacecraft Adapter: 42 kg
30% dry mass contingency

3. *Mission Analyses*

For this analysis the calculus-of-variations trajectory optimization code, SEPTOP, developed by Carl Sauer at JPL, was used. It models the interplanetary transfer as a two-body problem and optimizes the trajectory. In order to assess the trade-off between payload and trip time a top-level equation for the launch mass, M_{Launch}, was previously developed for input into SEPTOP.[12]

$$M_{\text{Launch}} = \alpha_{\text{power}} \times P + M_{\text{SEP-Fixed}} + M_{\text{payload}} \times (1 + k_{\text{structure}}) + M_{\text{propellant}} \times (1 + k_{\text{tankage}})$$

The values for the above parameters (inputs) were:

α_{power} = 9.7 kg/kW
$M_{SEP\text{-}Fixed}$ = 742 kg
$k_{structure}$ = 8.5%
$k_{tankage}$ = 13.7%
P = 30 kW

4. *Results*

Trade studies were performed on the throttle curve [high thrust vs. high specific impulse (I_{sp})], thruster count (4+1 vs. 3+1), and minimum number of operating thrusters (1 vs. 2). The 4+1 thruster/PPU configuration outperformed the 3+1 thruster/PPU configuration. Figure 9 shows the results of the payload and trip time trade for the 4+1 thruster/PPU configuration along with the arrival V_∞.

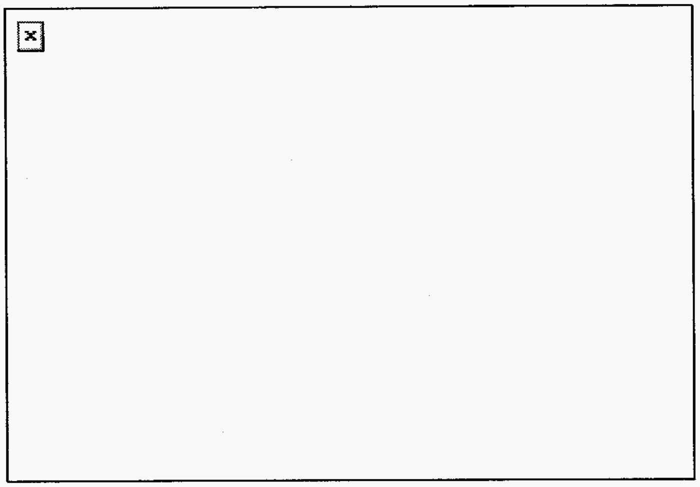


Figure 9: Payload to Neptune vs. trip time for the high thrust and high I_{sp} throttle curves.

As Figure 9 shows, the top two NEXT payload curves permit single thruster operation. Although the high thrust option delivers the required payload in slightly less time than the high I_{sp} option, payload requirements greater than 900 kg would mean that neither the high thrust nor high I_{sp} option offers a reduced trip time. When the minimum number of operating thrusters is increased to two, the high thrust and high I_{sp} options meet the payload requirement in nearly 10.5 years. However, the high I_{sp} option's payload curve actually crosses over the high thrust option's payload curve at 10.5 years and offers better performance for payloads greater than 850 kg. Out of these four NEXT payload curves, the high I_{sp} option payload curve with a minimum of two thrusters operating was selected over the other three for further analyses for the following two reasons: (1) to be consistent with previous Neptune DSDRM analyses that assumed a minimum of two operating thrusters, and (2) because payload requirements tend to increase; therefore, the high I_{sp} option is chosen over the high thrust option.

Figure 9 also depicts the arrival V_∞. Only one curve is shown--the other three curves are nearly identical. It shows indirectly (through the arrival V_∞) that the entry velocity constraint for the aerocapture system is not violated. The power and thruster variation histories are shown in Figure 10.

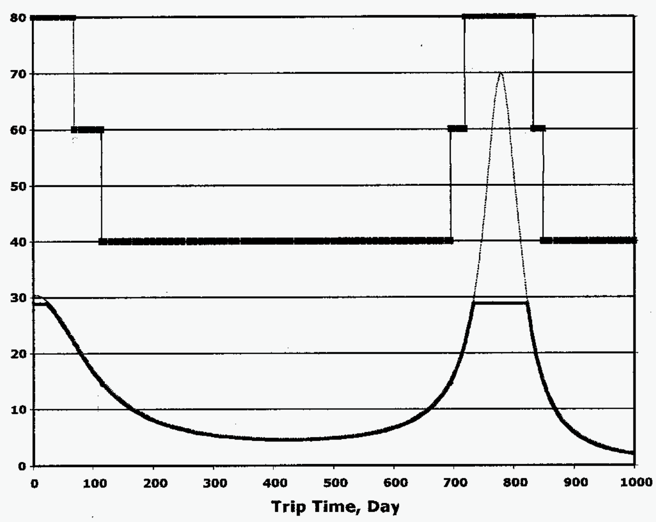


Figure 10: Power and thruster history variation.

The optimal thruster variation is 4-3-2-3-4-3-4-3-2 thrusters as power varies with distance from the sun. Excess array power enables maximum thruster power twice during the transfer. Also note that during lowest power operations, two thrusters are operating. Clearly, it would be more efficient to have one thruster operating at this 4 – 5 kW range as opposed to two thrusters splitting this power. Operating at higher efficiency results in increased performance as Figure 9 shows.

5. Summary

The 4+1, high I_{sp} option with a 30 kW BOL solar array meets the payload requirements as stated in the DSDRM when launched on a Delta IV 4240. Single thruster operation offers a slight increase in delivered payload mass for a given trip time. A payload increase greater than 10% of the current DSDRM requirement will require a larger launch vehicle. This was the case for the IISTP 2004 Focused Studies, which used an Atlas V 551 to meet a payload requirement of 1,070 kg with a similar transfer time.[11]

IV. Mars Mission Applications

1. Analysis Objective

A Mars Sample Return (MSR) mission is of great interest for Mars Exploration and is a relatively challenging mission from a ΔV point of view. The overall objective of this analysis is to make a direct comparison between an MSR mission utilizing chemical propulsion (supplemented by aerobraking) and a SEP based mission utilizing NEXT. The immediate objective is to look at the feasibility of taking a chemical MSR mission requiring two launch vehicles and accomplishing the same objectives with a SEP MSR mission requiring a single launch vehicle. This comparison illustrates in general terms the benefits and limitations of SEP for unmanned Mars missions.

2. Mission Description

The objective of the MSR mission is to land on the surface of Mars, retrieve a sample of the surface, and return it to Earth for scientific analysis. There are many chemical propulsion based mission architectures that can used for the MSR mission using different combinations of single or multiple orbiters and landers launched on one or more launch vehicles. A representative chemical MSR mission utilizing two launch vehicles, one combined orbiter/earth return vehicle (ERV), a primary lander, and a backup lander is used as the baseline mission in this analysis. This architecture is one of several options under consideration and has following sequence of events.

1) The primary lander and Orbiter/ERV are launched on a type II ballistic trajectory to Mars using a Delta 4050 Heavy launch vehicle. The spacecraft are launched directly to an Earth escape trajectory and have a nominal flight time of 10 months.
2) A backup lander, supported by a cruise stage, is launched on a type II ballistic trajectory to Mars using an Atlas V 511 launch vehicle. This spacecraft is also launched on an Earth escape trajectory and is targeted to arrive at about the same time as the primary lander.
3) After the orbiter and cruise stage reach Mars vicinity, the landers are released and enter the Martian atmosphere using direct entry trajectories.
4) The orbiter conducts a Mars Orbit Insertion burn that places the spacecraft in an elliptical orbit. Aerobraking is used to circularize the orbit over a 6 month period.
5) The landers collect samples and the Mars Ascent Vehicles (MAV) carry the sample into orbit.
6) The MAV and the Orbiter rendezvous, transferring the sample to Earth Return Vehicle (ERV)
7) The Orbiter/ERV use chemical propulsion to escape Martian orbit and enter an Earth return trajectory.
8) Once in Earth vicinity, the ERV is released for direct entry into the Earth's atmosphere.

Elements of the chemical architecture were sized using pre-existing mass and power budgets for the Orbiter/ERV, lander, MAV, and cruise stage and the overall mass of each element is shown in Table 5.

The use of a second launch vehicle for the backup lander considerably increases the cost of the mission and only addresses risks associated with the lander itself. A launch failure for the first launch vehicle still results in loss of mission. The SEP mission architecture eliminates the second launch vehicle by placing the primary lander, backup lander, and orbiter/ERV on a single launch vehicle. It has the following sequence of events.

1) The primary lander, backup lander, and Orbiter/ERV are launched directly to escape velocity.
2) The vehicles proceed together to Mars using a powered SEP trajectory.
3) The landers are released by the Orbiter/ERV and enter the Martian atmosphere using direct entry trajectories.
4) The orbiter/ERV uses SEP to spiral down to low Mars Orbit
5) The landers collect samples and the Mars Ascent Vehicles (MAV) carry the sample into orbit.
6) The MAV and the orbiter rendezvous, transferring the sample to the Earth Return Vehicle (ERV)

7) The Orbiter/ERV uses SEP to spiral up and reach escape velocity.
8) A powered SEP trajectory is used to return to Earth vicinity
9) Once in Earth vicinity, the ERV is released for direct entry into the Earth's atmosphere.

This architecture is notional and there are several issues related to landing and rendezvous that are not addressed. In particular, the release point for the landers has not been defined, so they may require additional ΔV to reach a Mars direct entry trajectory. In addition, the low entry speed may affect guidance and navigation accuracy during landing and the relatively large solar array may effect rendezvous guidance and control. These issues and others need to be addressed before this architecture can be used for an MSR mission.

	Chemical MSR Mission		
	Chemical Mission w/Aerobraking	**Optional Backup Lander**	**SEP/NEXT MSR Mission**
Launch Vehicle	Delta 4050H	Atlas V 511	Delta 4050 H
Launch Date	Nov 2013	Nov-Dec 2013	July 2013
Mars Arrival Date	Sept 2014	Sept 2004	Jan 2016
Mars Departure Date	Nov 2015		Jan 2018
Earth Return Date	July 2016		Jan 2019
Mission Duration	**2 yrs, 8 mon**		**5 yrs, 6 mon**
LV C3 (km^2/s^2)	9.3	10	1.21
On-Board Delta-V total [km/sec]	3.9	0	TBD
Launch Mass (30% dry mass margin)	6882	2794	9092.8
Launch Vehicle Capability	7710	3105	9120
Launch Vehicle Margin	828	311	27.2
Mass Summary (kg)	**Chemical Launch 1**	**Chemical Launch 2**	**Single Launch**
1) Orbiter/ERV DRY mass	1029	0	858
2) SEP Propulsion Module	0	0	1486
3) Orbiter/ERV Propellant	3215	0	1625
4) Mars Lander 1 (Wet)	2480	0	2480
5) Sample Capture Hardware	112	0	112
6) Earth Entry Vehicle	43	0	43
7) Cruise Stage (Dry)	0	225	0
8) Cruise Stage Propellant	0	89	0
9) Mars Lander 2 (Wet)	0	2480	2480
Total Launch Mass (kg)	**6879**	**2794**	**9093**

Table 5: Mars Sample Return Mission Summary Table

3. Analysis and Results

The spacecraft was sized by modifying elements of the chemical architecture to accommodate the electric propulsion system. Elements of the orbiter related to the chemical propulsion system and aerobraking were removed and a SEP propulsion module was added to the mass equipment list (MEL). A solar array providing 30 kW of power at beginning of life (1 AU) with a specific mass of 140 W/kg. was incorporated into the SEP module to provide power for six NEXT thrusters, four of which can be operated simultaneously. Two extra thrusters are required to meet throughput requirements and no redundancy was provided for the EP system. Elements of the cruise stage were incorporated into the orbiter to provide power and structural support for the backup lander. The landers and ERV were not changed. A 30% dry mass margin was maintained on all elements of the system, though no margin was applied to propellant mass or launch vehicle capability. The resulting mass of each element is shown in Table 5. No verification has been made that all elements of the spacecraft fit within the launch vehicle faring.

The Earth-Mars transfer trajectory was optimized using MALTO, a low thrust trajectory optimization tool developed at JPL, and the resulting inbound and outbound trajectories are shown in Figure 11 and Figure 12 below. Table 6 shows the duration of each mission phase.

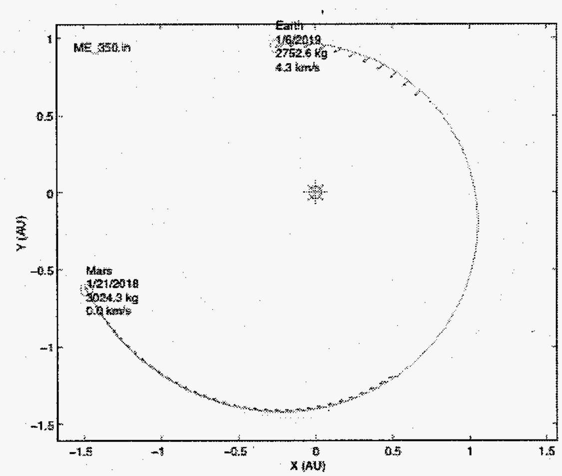


Figure 11: Outbound Trajectory (Earth to Mars, 1.5 years)

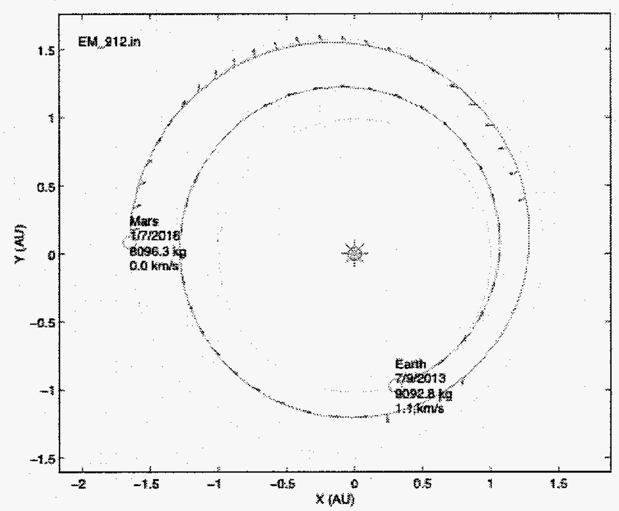


Figure 12: Return Trajectory (Mars to Earth, 1 year)

Mission Phase	Transit Time
Earth to Mars Transit	1.5 year
Spiral Down to Low Mars Orbit	267 days
Rendezvous and Stay Time	300 days
Spiral Up to Mars Escape Trajectory	178 days
Mars to Earth Transit	1 year
Total Mission Time	5.5 years

Table 6: Mission Phase Duration

The SEP system has a much higher launch mass than the chemical system, and therefore has a relatively low launch C_3. The low C_3 and low thrust to weight ratio result in a relatively long Mars transit time. Once the landers are released, the system's thrust to weight ratio increases. As a result, the return trajectory requires significantly less time than the outbound trajectory. Significant time is also required to spiral down to and up from low Mars Orbit. In addition, a relatively long stay time of 300 days is required before the window opens for the return trajectory.

4. Summary

Overall, use of NEXT on the orbiter/ERV increases the mass delivered to Mars by over 2400 kg. when compared to a type II chemical trajectory and allows a single launch vehicle to carry both the primary and backup lander for this mission. This is a significant benefit that simplifies the mission and saves the cost of the extra launch vehicle. However, use of SEP also increases the mission's duration by 1.8 years and requires a 30 kW power system. These significant penalties may outweigh the mass advantage in this particular mission architecture. Overall, this mission illustrates well both the advantages and disadvantages of SEP for Mars missions. Though the use of SEP greatly improves mass capability, it also substantially lengthens the trip time.

For future work, there are several things that might improve the performance of SEP for this mission. One possibility is to use a combination of SEP and aerobraking for orbit lowering at Mars. The solar arrays on the SEP module have a large surface area that can be used for aerobraking during orbit lowering at Mars. This has the potential both to increase mass margin and to lower overall trip time. It is also desirable to look at how sensitive the departure time is to changes in mass or stay time. The current stay time of 300 days is governed by orbital dynamics, not by mission requirements, and it would be interesting to determine if a small change in mass or shortened spiral time might allow a significantly earlier departure date.

V. New Frontiers Mission Applications

A. Comet Surface Sample Return

1. Mission Description

A Comet Surface Sample Return mission is one of the recommended solar system flight missions identified in the 2002 solar system exploration decadal survey.[9] The objective of this mission is to rendezvous with a comet, collect a sample of material and return it to the surface of the Earth for scientific analysis. For this study, comet Tempel 1 was chosen as a representative destination. Tempel 1 is a fairly challenging target that can probably be reached within the New Frontiers cost cap.

The description presented here is a summary of a detailed study of the use of SEP on a Comet Sample Return mission. A detailed description of the study's assumptions and results is available from Ref. 13.

2. Objective of Analysis

The objective of this study is to compare the performance of the current state of the art NSTAR ion thruster to the NEXT system. This analysis investigates the performance of both propulsion systems over a range of power levels, number of operational thrusters, and thruster throttling modes. For the NSTAR system, we look at 4 and 5 operational thrusters with 1 spare. This variation in thruster number allows a determination of best performance based on a baseline array power of 12 kW. For this same baseline power, it is obvious that 2 NEXT thrusters is a sensible choice with three being having too great of maximum power level above the baseline and one thruster having too small a maximum power below the baseline. The NEXT thruster is evaluated using a high thrust throttling profile and a high specific impulse throttling profile to provide an evaluation of performance over the operational throttling range for this mission.

3. Mission Analysis Assumptions and Methodology

Figure 13 depicts the orbits of the Earth and comet Tempel 1 about the sun. The mission objective is for the spacecraft to rendezvous with Tempel 1, gather a sample, and return this sample to Earth. The spacecraft's position and velocity must match Tempel 1's position and velocity during rendezvous.

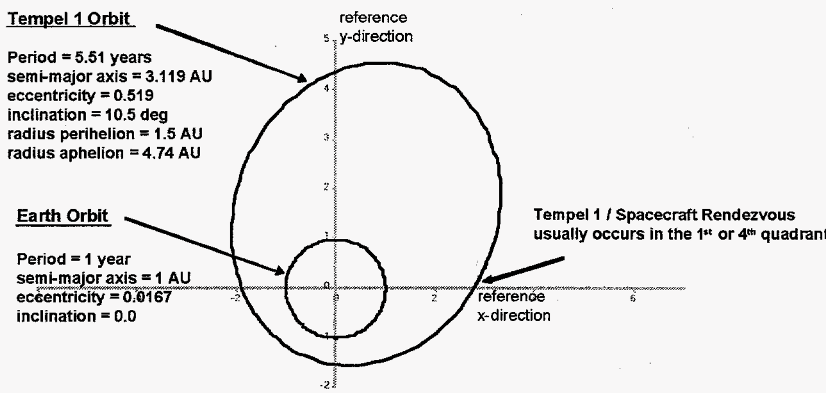


Figure 13: Earth and Tempel 1 Orbital Descriptions

The payload is modeled as 50 kg of mass that remains at the comet and a net non-propulsive mass that is returned to Earth. This net returned mass consists of the sample, sample return system, and remaining spacecraft mass. The launch vehicle chosen for this investigation was the Delta IV 4040 which is the smallest Delta medium class launch vehicle. The SEPTOP low thrust optimization program was used for this analysis. A more detailed description of the analysis assumptions is available in reference 13.

4. Results

Figure 14 shows a typical Tempel 1 roundtrip comet sample return trajectory. The salient feature consist of the following:

1) Launch vehicle assisted Earth departure to a C_3 of ~ 17
2) SEPS assisted transfer to rendezvous with Tempel 1
3) 60 day stay at Tempel 1, during which a sample is taken from the comet and returned to the spacecraft
4) SEPS assisted return to an Earth flyby to allow sample return to Earth, perhaps through a direct entry capsule

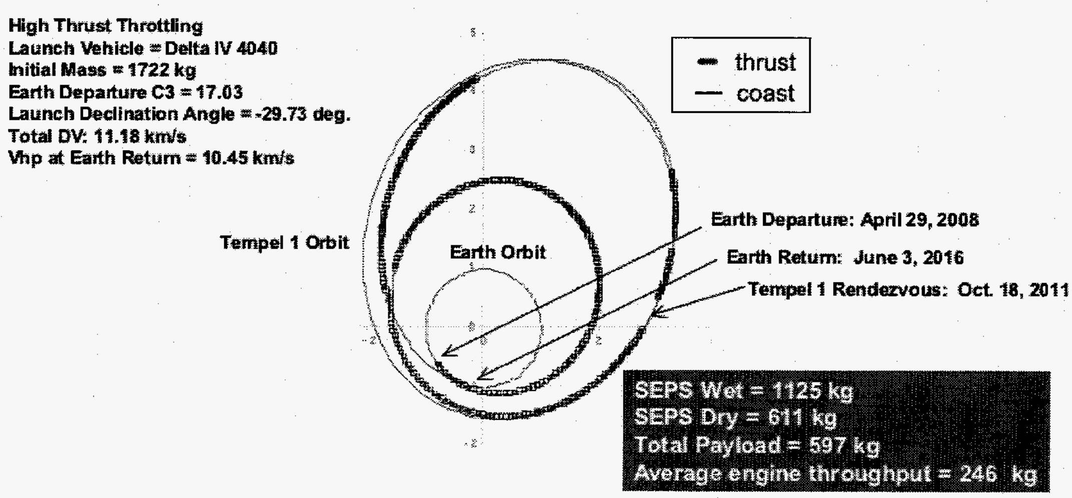


Figure 14: Typical Tempel 1 Roundtrip Comet Sample Return Trajectory

The following depiction of payload vs. array power, shown in Figure 15, shows systems payload delivery performance as a function of solar array power level. First, comparing NEXT high thrust throttling and high Isp throttling, the high thrust throttling mode of NEXT provides a small increase in payload over the high Isp throttling mode. Likewise, over the investigated range of array power, the high thrust throttling mode provided a small decrease in required power for a given set payload.

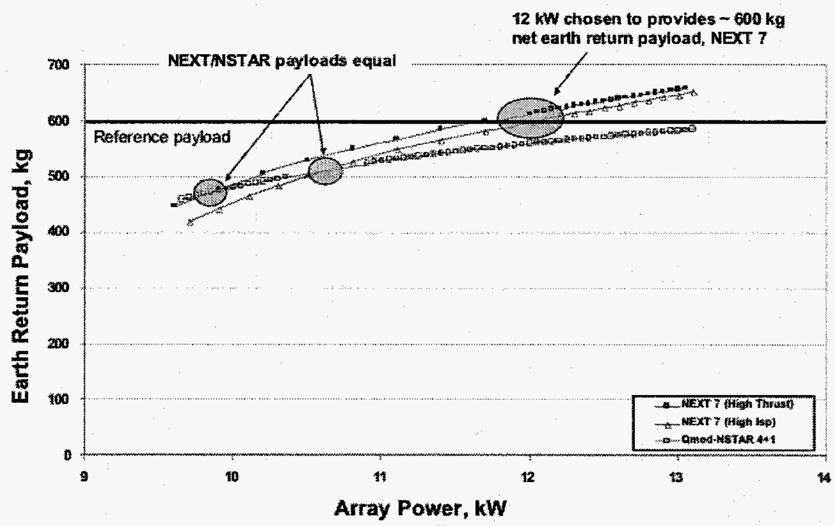
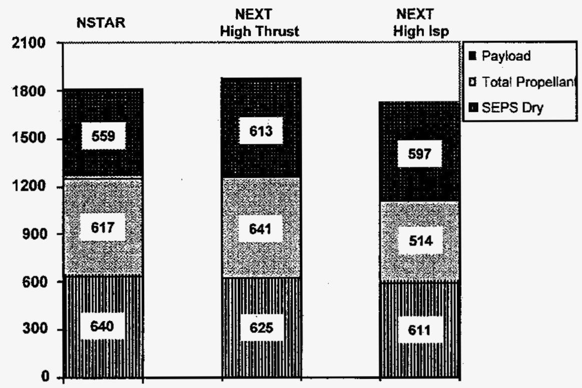


Figure 15: Comet Sample Return Payload Performance vs. Array Power

A second comparison of NEXT with NSTAR shows a cross-over in payload mass occurs for power levels between 9 kW and 11 kW. Also, note that for array power above ~9 kW, NEXT provides an increasing advantage in payload mass as the power level increases.

Figure 16 summarizes the relative performance of the NSTAR and NEXT based systems for the comet sample return mission at an array power of 12 kW (BOL, 1 AU). 12 kW was chosen as a baseline power to perform this analysis for two reasons: the first is that this power level provides a sizable 600 kg of Earth return payload for NEXT; the second reasons is that a NEXT phase 1 study was performed at 12 kW as a baseline array power.

Figure 16 shows that NSTAR with a 4+1 engine configuration delivers over ~ 543 kg back to earth. High thrust throttling mode of NEXT delivers about 613 kg and the high Isp throttling mode delivers approximately 597 kg to Earth return. Thus, for the Tempel 1 CSSR mission, the NEXT thruster provided modest improvement in payload over NSTAR. Also this NEXT payload improvement was realized with a 2+1 engine configuration that may imply significant cost and complexity benefits over the 4+1 NSTAR configuration.



	NSTAR	NEXT High Thrust	NEXT High Isp
Isp Max, sec	3070	4180	4190
Power Max, kWe	2.57	7.23	7.23
Power Min, kWe	0.58	0.82	0.82
Avg. Throughput, kg	146.1	308.0	234.7
Engine Configuration	4+1	2+1	2+1

Figure 16: Comet Sample Return Mass Delivery Summary (12 kW array)

B. Jupiter Polar Orbiter

1. Mission Description

The Jupiter Polar Orbiter mission is cited in the decadal survey as the highest priority mission for giant planet research.[9] This mission would enable a better understanding of Jupiter's strong magnetic and gravity fields and its deep atmosphere. Determining the composition of Jupiter's core and atmosphere is key. To avoid the highest-flux parts of the Jovian radiation field, a very low perijove, $< 1.1\ R_J$, is necessary. At least three atmospheric entry probes that can penetrate to the 100 bar pressure level and that can sample a range of latitudes within 30 degrees of the equator are required. The orbiter would be expected to remain in orbit for at least one year.

2. Objective of Analyses

The Jupiter Polar Orbiter study had the following objectives: (1) identify a desired capture orbit, (2) identify a SOA chemical system to enable capture, (3) understand the trade-offs between such parameters as thruster count, throttling curve, power, payload, and trip time, (4) understand the trade-off between two different mission scenarios: (a) jettisoning the entire SEP module after the thrust phase and then relying on RTG's for spacecraft power and (b) jettisoning the propulsion system after the thrust phase and retaining the solar arrays for spacecraft power, and finally, (5) compare the performances of NEXT and ballistic trajectories.

3. Transportation Approach

Two transportation approaches to achieve the science mission were investigated. Both used launch vehicles compatible with the New Frontiers mission class, an Atlas V or a Delta IV, to achieve Earth escape. The first Jovian approach, herein referred to as NEXT/Chemical, utilizes a SEP module with a NEXT subsystem to perform the heliocentric transfer to Jupiter with the aid of a single VGA. A SOA chemical system provides the necessary ΔV to capture into Jupiter's orbit and is jettisoned after capture. This approach has a nuclear and non-nuclear option. In the nuclear option the SEP module is jettisoned after use, near 2 A.U., and the orbiter and chemical system are transferred to Jupiter. Radioisotope Thermoelectric Generators (RTG's) power the orbiter after the SEP module is jettisoned. The non-nuclear option retains the solar arrays associated with the SEP module and jettisons the NEXT propulsion system along with the xenon tank, structure and cabling. The jettisoned mass totals 530 kg for a 2+1 thruster/PPU configuration. Solar arrays would provide power past 2 A.U. and at Jupiter, possibly eliminating the need for RTG's. Cost, performance, and to a lesser degree, complexity, are the important trade-offs between the nuclear and non-nuclear option. The second Jovian approach, herein referred to as SOA chemical, utilizes ballistic trajectories aided by multiple gravity assists or no gravity assists (direct) to transfer and capture into Jupiter, with mission power provided by RPS.

4. Mission Assumptions

With several key mission parameters undefined at the time of this study, parametric analyses were performed for a range of possible mission scenarios. The required payload to Jupiter, consisting of the mass of the science payload, spacecraft bus and associated systems and any necessary propulsion, was undefined. In order to analyze the performance of NEXT/Chemical and SOA chemical options, a wide range of delivered payload masses were investigated. The effects of not knowing the required payload necessitated top-level trades such as launch vehicle selection and array power. The capture orbit period and required ΔV were also unspecified. Therefore, a capture orbit period ΔV trade was performed as shown in Figure 17. This figure also illustrates the mass that the SOA chemical system had to accelerate to enable it to be captured in the targeted orbit. One power level is shown; the capture orbit ΔV trade is the same regardless of the power level, but the masses change. In order to meet science objectives and minimize ΔV, a 30-day capture orbit period was chosen. This would allow time for data transmission, processing, and any subsequent decision-making. A 30-day capture orbit period results in a ΔV of roughly 800-1200 m/s when the maneuver performed during a 3.3 – 4.0 year trip time range. The SOA chemical system that would perform this capture was sized only for this single, large maneuver; any necessary post-insertion ΔV was not accounted for in this study.

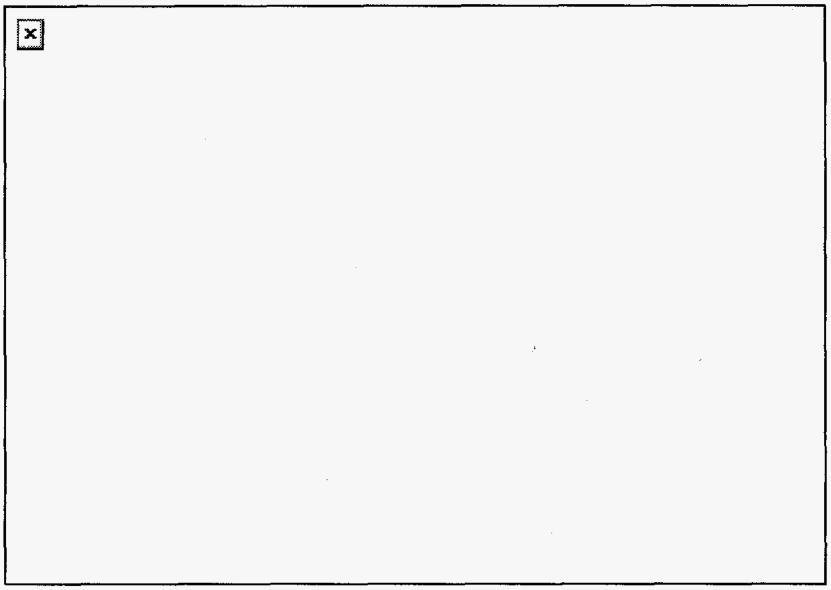


Figure 17: Jupiter Polar Orbit Capture orbit ΔV trade.

The ballistic trajectory database that was available at the time of this analyses had some launch dates that were several years before the end of this decade. It was assumed that there exist ballistic trajectories that have similar performance in a later, more realistic timeframe for this mission. This should not provide any difficulty since one of the objectives of this analysis was to access the performance of NEXT/Chemical and compare it to nominal SOA ballistic trajectories. For the non-nuclear option as described above, the jettisoned mass for a 2+1 thruster/PPU configuration was 530 kg. This included thrusters, PPUs, the Xenon tank, structure and cabling plus 30% contingency. A complete list of assumptions follows:

- Mission Assumptions:
  - Launch dates for NEXT/Chemical: 2009-2010
  - Launch dates for the direct ballistic trajectories: 2005-2011
  - Launch dates for the DVEGA ballistic trajectories: 2003-2008
  - Launch dates for the VEGA ballistic trajectories: 2003-2008
  - Perijove altitude: 7,149 km
  - 30 day capture orbit period
  - 2% ΔV margin
  - 3% gravity losses
- Launch Vehicle Assumptions:
  - Atlas V 551 (except where noted)
  - Delta IV 4050H launch vehicle performance curves extended beyond C_3 of 60 km^2/s^2 by extrapolation (effected the direct ballistic trajectories which had C_3 in the 70 and 80 km^2/s^2 range)
  - 10% launch mass contingency
- Chemical System Assumptions:
  - N_2O_4/N_2H_4 propellant

I_{sp} 328 seconds
140 psi chamber pressure
SOA composite tank strength, 30 mil linear thickness
5% propellant contingency
0.5% residuals
5% ullage
30% dry mass contingency

- SEP Stage Assumptions:
  2+1 thrusters/PPUs
  High thrust throttling curve
  One or two thrusters operating
  P_{max}: 7.255 kW
  P_{min}: 0.817 kW
  5% Xenon propellant contingency for flight errors
  3% Xenon propellant contingency for reserves
  90% propulsion duty cycle
  174 W/kg solar arrays (150 W/kg with structure)
  $1/R^2$ array power
  2% per year array degradation
  250 Watts supplied for housekeeping activities
  Power System Specific Mass: 9.7 kg/kW
  Xenon tank mass fraction: 3.4% of total propellant
  Power and Thermal Structure: 16%
  Stage cabling (not PPU): 6% of power mass
  Tankage Mass Fraction (includes structure [26% dry] and 3% Xenon propellant contingency): 13.7%
  Payload Structure Mass Fraction: 8.5%
  Spacecraft Adapter: 42 kg
  30% dry mass contingency
- Jettisoned Mass Assumptions for Non-Nuclear Option:
  530 kg (3 thrusters, 3 PPUs, Xenon tank, structure, cabling, + 30% contingency)

5. *Mission Analyses*

The two-body trajectory optimization code SEPTOP was used for this analysis. In order to examine the trade off between mass prior to capture at Jupiter, $M_{payload+chemical}$ and trip time, NEXT stage system inputs similar to those described in the Neptune mission analysis section were calculated. The SEPTOP equation modeling launch mass, M_{Launch}, is repeated here.[10]

$$M_{\text{Launch}} = \alpha_{\text{power}} \times P + M_{\text{SEP-Fixed}} + M_{\text{payload+chemical}} \times (1 + k_{\text{structure}}) + M_{\text{propellant}} \times (1 + k_{\text{tankage}})$$

The values for the above parameters (inputs) were:

α_{power}	= 9.7 kg/kW
$M_{SEP\text{-}Fixed}$	= 510.5 kg
$k_{structure}$	= 8.5%
$k_{tankage}$	= 13.7%
P	= Varied between 13 – 25 kW

α_{power}, $k_{structure}$, and $k_{tankage}$ are linear scaling factors that model the rate at which their respective systems grow. $M_{SEP\text{-}Fixed}$ is the fixed mass of the systems and remains fixed as the other parameters change. After solving for $M_{payload+chemical}$, for a given trip time, the chemical capture portion of the mission was performed parametrically. Chemical capture analysis was performed using the Advanced Chemical Propulsion System (ACPS) model created by SAIC that uses physics and experiential based models to find the overall chemical system mass.

6. Mission Trades

To enable a large performance range, three different launch vehicles were initially chosen. The Delta IV 4240 encapsulated the low end of delivered payload capability. Two large launch vehicles, the Atlas V 551 and the Delta IV 4050H, provided the high end capability. Initial analyses focused on the low thrust-throttling curve with a 3+1 thruster/PPU configuration. Solar array power was set to 25 kW. Figure 18 and Figure 19 show these results with a comparison to ballistic trajectories.

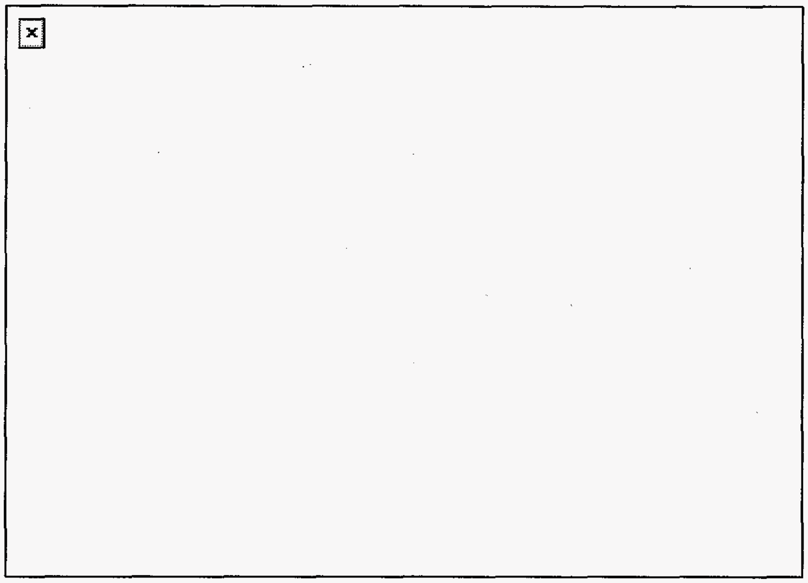


Figure 18: Low end performance comparisons.

As Figure 18 shows, the non-nuclear option's performance is reduced approximately 15% compared to the RTG option. Both outperform most ballistic trajectories. Figure 19 shows similar results for the larger launch vehicles.

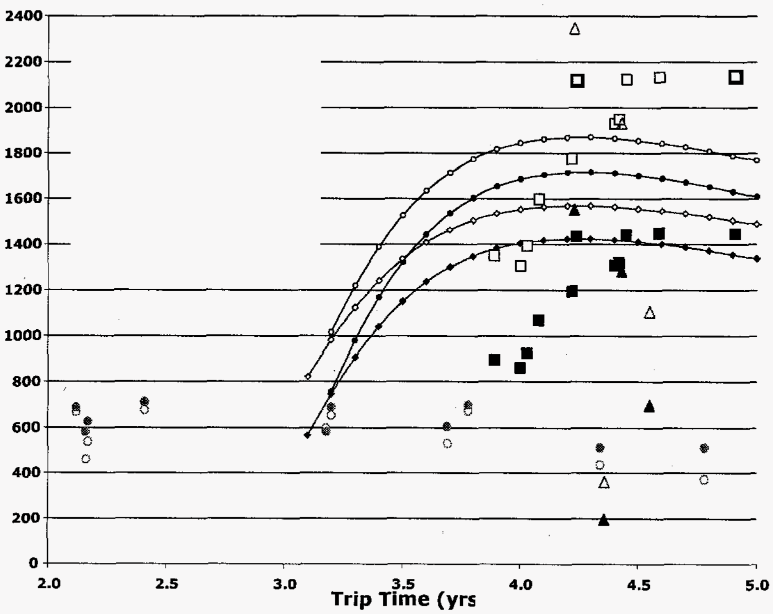


Figure 19: High end performance comparisons.

Because the New Frontiers mission cost cap is $700M, further analyses emphasized reducing solar array power and launch vehicle cost. Therefore, solar array power, thruster count, and throttling curves (high vs. low thrust) were traded on an Atlas V 551. As Figure 20 shows, the trade studies performed on thruster count and throttling curve indicated that the high thrust 2+1 configuration performed the best and was selected for further analyses

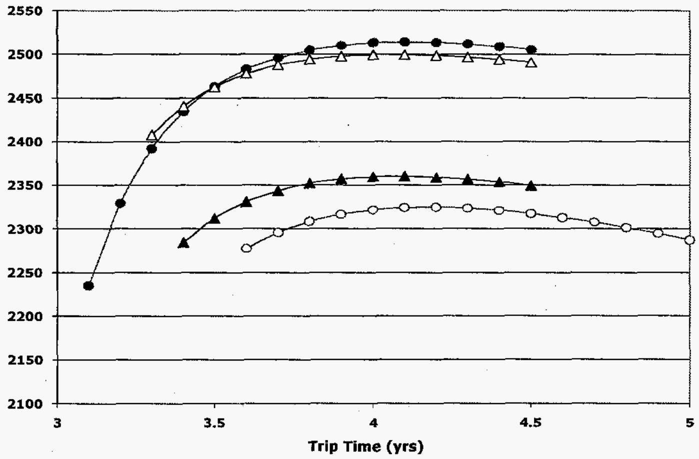


Figure 20: JPO thruster count and throttling curve trade.

7. Results

A performance comparison between NEXT/Chemical and SOA ballistic trajectories is shown in Figure 20 for various power levels for the nuclear option (jettisoning the SEP module near 2 A.U. and relying on RTG's for power).

Figure 21: Jupiter payload comparison with power trade (RTG power option).

As the power trade shows, adding 1 kW of array power increased payload mass by roughly 80 kg at the lower power levels and roughly 50 kg at the mid power levels. The optimal transfer time was near four years. Figure 21 also shows that no known SOA ballistic trajectories exist that can deliver a payload greater than 900 kg in less than 4 years. Figure 22 shows the same trade on the second, non-nuclear, option with similar results. The propulsion system, propellant tank, structure and cabling, totaling 530 kg, were jettisoned, leaving the solar arrays to supply power.

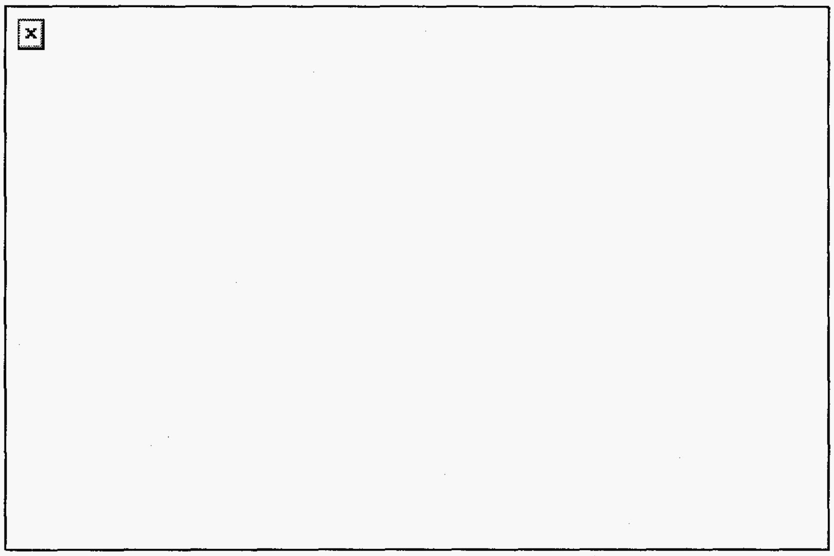


Figure 22: Jupiter payload comparison with power trade (no RTG's).

Trades on power also revealed that the declination of the departure asymptote (DLA) increased as solar array power decreased for a given trip. Furthermore, the DLAs generally exceeded 28.5 degrees for power levels below 20 kW. This is illustrated in Figure 23. Because of the geometry of the interplanetary transfer, the parking orbit inclination must be greater than or equal to the DLA prior to injection. Further analysis showed that delaying the Venus flyby a couple of days or more reduced the launch declination below 28.5 degrees while only reducing performance by 1-2% for one selected power level. Further analysis needs to be performed to verify that this marginal performance reduction is true across all power levels. It is well known that launch vehicle performance will be reduced for declinations above 28.5 degrees. For this analysis the launch vehicle performance curves reported by Kennedy Space Center were reduced 10%.[14]

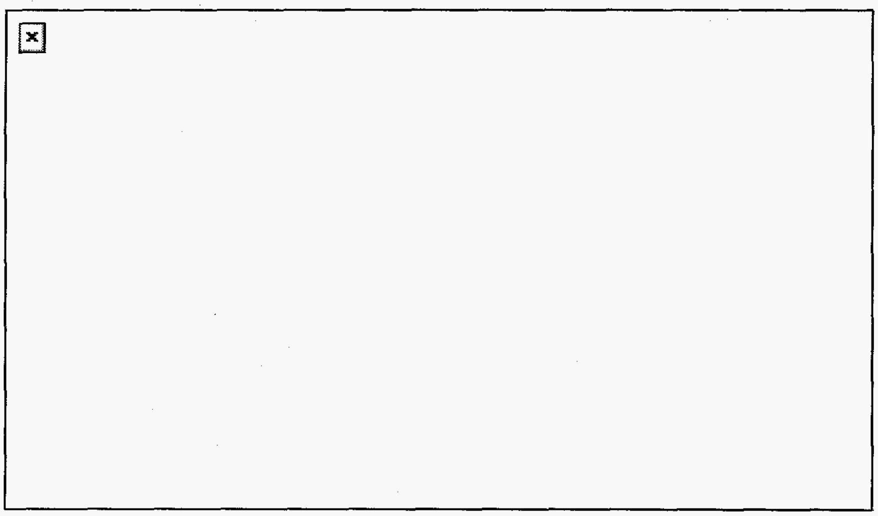


Figure 23: Declination of the departure asymptote for various power levels

A typical power and thruster history along with a trajectory profile are shown in Figure 24 and Figure 25. The second thruster was turned off for roughly 200 days during the high A.U. part of the mission indicating an efficient use of the thrusters. Also, the power level never reached the lower limit--avoiding thruster operation at the least efficient part of the throttle curve. Throughput was not considered to be an issue for any of the performance curves shown; NEXT throughput limits are expected to exceed 270 kg.

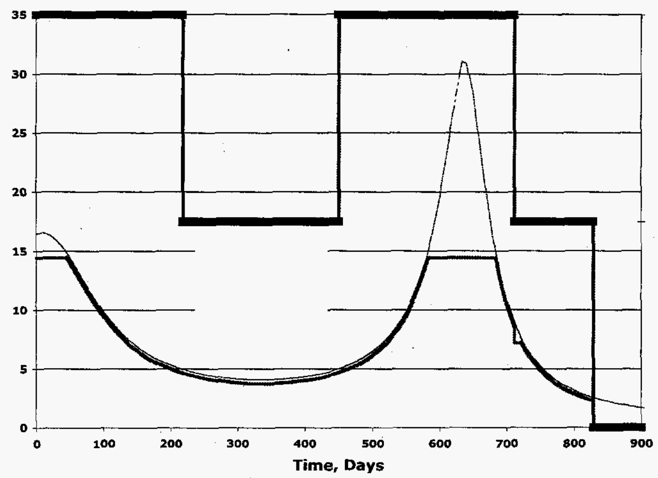


Figure 24: Jupiter Polar Orbiter Power and thruster history

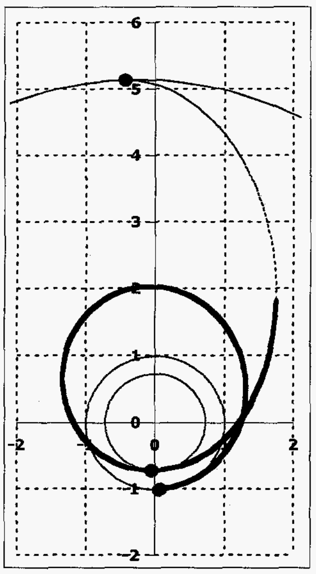


Figure 25: Jupiter Trajectory Profile

8. Summary

Because the required payload to Jupiter for this mission was undefined, the benefits of a NEXT/Chemical transportation approach can be stated in terms of delivered payload ranges. As Figure 19 shows, for payloads less than 800 kg the direct ballistic trajectories reach Jupiter in just over two years on the Atlas V 551 or Delta IV 4050H. For larger payloads, a NEXT/Chemical system can arrive at Jupiter in roughly 3.5 – 4.0 years, depending on the power level, on an Atlas V 551 with a 2+1 thruster/PPU configuration. In general, NEXT seems to offer better performance than ballistic trajectories, even for the non-nuclear option. The few ballistic trajectories that do show similar performance do so with two planetary flybys including Earth. NEXT uses a VGA, thus avoiding any environmental and safety concerns associated with an EGA when RTG's are flown. The high end of the NEXT/Chemical performance is more difficult to define. Assuming that the highest power level that reasonably can be expected to be within the cost caps of a New Frontiers missions is 17 kW, then the high end is nearly 1,300 kg with a near four-year transfer time. Payloads in excess of this will require some of the following: (a) more array power, (b) the Delta IV 4050H launcher, (c) a ballistic trajectory with multiple gravity assists, possibly including Earth, with transfer times likely greater than four years, (d) a greater capture orbit period, (e) any combination of (a) – (d). To increase the performance of a NEXT/Chemical transportation option and make a true apples-to-apples comparison with ballistic trajectories that use multiple gravity assists, an additional Venus gravity assist could be modeled (along with updating the ballistic trajectory database to reflect more realistic launch dates). However, this is difficult to model and would likely increase the transfer time and therefore was not done in this study.

NEXT also offers a non-nuclear transportation approach by dropping (staging) the propulsion system and keeping the solar arrays for power. This staging approach is similar to the approach that the European Space Agency seems to be taking with the BepiColombo mission to Mercury in 2012.[15] On the Jupiter mission, this approach reduces performance by roughly 10%, but this may be offset by the potential cost savings and procurement issues. At Jupiter, a 13 kW array that has not degraded due to Jupiter's radiation environment might produce roughly 500 Watts of power in a sun-normal configuration, more than most currently existing RTG's.[16] The amount of power actually available from the array depends on many factors, including orientation and radiation exposure, and much more analysis is needed to determine the feasibility of this architecture. The performance of NEXT also suggests that a tighter capture orbit is possible depending on the desired payload and power level.

VI. Discovery Class Missions

1. Mission Description

Discovery class missions are of particular interest because they provide numerous near-term, recurring flight opportunities for electric propulsion systems. One mission, Dawn, is currently in development and several other missions using NSTAR propulsion systems have been or will be proposed in the near future. Unlike the missions considered above, Discovery missions vary widely in scope with varying destinations and science goals. For purposes of comparison, two representative discovery missions were selected for study. The first is a "Dawn-like" multiple asteroid rendezvous mission delivering a spacecraft to the asteroids Vesta and Ceres using a low thrust trajectory with a Mars gravity assist. The second is a Near Earth Asteroid mission similar a class of Discovery missions currently in development.

2. Objective of Analysis

The objective of this analysis is to compare the performance of NEXT to state of the art electric propulsion systems currently used for Discovery class missions. The first mission, a dual asteroid rendezvous, is similar to the Dawn mission. The second mission, a near Earth asteroid mission, is typical of a class of Discovery missions that favor the use of electric propulsion for primary propulsion.

3. Dual Asteroid Rendezvous Analysis Assumptions and Results.

The Dual Asteroid Rendezvous is a "Dawn-like" mission that targets two asteroids for Vesta and Ceres, for close range observation. These asteroids are the same targets selected for the Dawn mission, and the launch date and launch vehicle are the same ones currently selected for Dawn. The mission was analyzed with the following assumptions.

- Launch Vehicle: Delta II 2925H
- Launch date: July 2006
- On board ΔV: ~10.5 km/s (NSTAR nominal mission)
- Vesta Rendezvous date: September 2011
- Vesta Departure date: April 2012
- Propellant used during Vesta Stay: 10 kg.
- Ceres Rendezvous date: July 2015
- Solar array power: 8.6 kW (1 A.U. BOL)
- Mars Gravity Assist trajectory
- Redundant thruster is required for the first target (Vesta)
- No thruster redundancy required for the second target (Ceres)
- NSTAR xenon throughput capability: 150 kg.
- NEXT xenon throughput capability: 300 kg.

The trajectory is constrained by a combination of targets rendezvous and launch dates and requires extended periods of operation at low power levels. Trajectory optimization was conducted using Mystic, a low thrust optimizing tool that is able to optimize trajectories in the presence of multiple external constraints. The baseline NSTAR mission uses one operating thruster at a time, but because of the high xenon throughput required, two thrusters are need to reach Vesta and three to reach Ceres. NEXT is able to reach the first target with one thruster and the second with two. The results of the analysis are summarized in Figure 26.

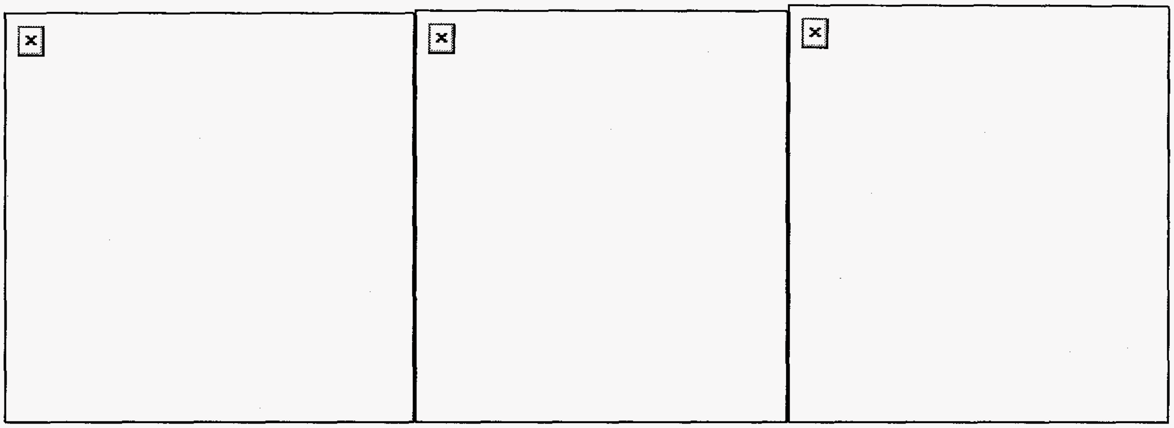


Figure 26: "Dawn-like" Mission Performance Characteristics

Figure 26 shows that using NEXT moderately improves the net mass delivered, yielding a performance increase of about 60 kg. In addition, NEXT's nominal throughput capability exceeds 270 kg/thruster, allowing it to meet mission throughput requirements with only two thrusters. This simplifies the electric propulsion system. A cost benefit analysis will be conducted in the future to determine if there is a cost benefit for this configuration.

4. Near Earth Asteroid Mission Analysis Assumptions and Results

The near-Earth asteroid mission targets an asteroid relatively close to Earth orbit for observation and sampling and has characteristics typical of a class of discovery missions that use electric propulsion. The mission uses a smaller solar array than Dawn, but is still relatively power rich because the target is relatively close to the sun. A typical power vs. time profile for this mission is shown in Figure 27. This mission was analyzed with the following assumptions:

- Launch Vehicle: Delta 2925
- Launch year: 2011
- On-board ΔV: between 4.5 and 6.5 km/s
- Mission duration: 3.2 years
- Solar array power: 6 kW (1 A.U. BOL)
- No gravity assist
- Redundant thruster required
- NSTAR xenon throughput capability: 150 kg.
- NEXT xenon throughput capability: 300 kg.

Trajectory optimization was conducted using VERITOP, a well-known low thrust optimization tool. The optimizer selects the launch date and rendezvous date that maximizes mass delivered. Results of the analysis are shown in Figure 28 and Figure 29 below.

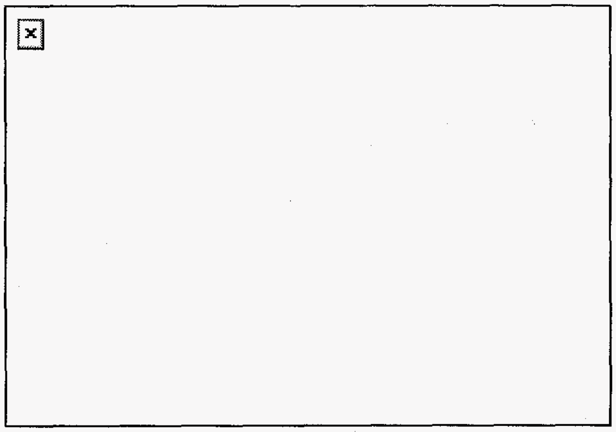


Figure 27: Near Earth Asteroid Mission Power Profile

(Composite Profile: does not show a Trajectory used in this Particular Study)

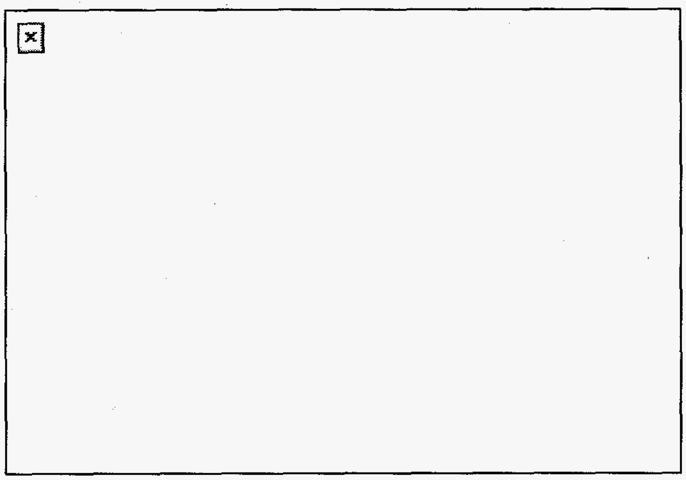


Figure 28: Near Earth Asteroid Delivered Mass Performance

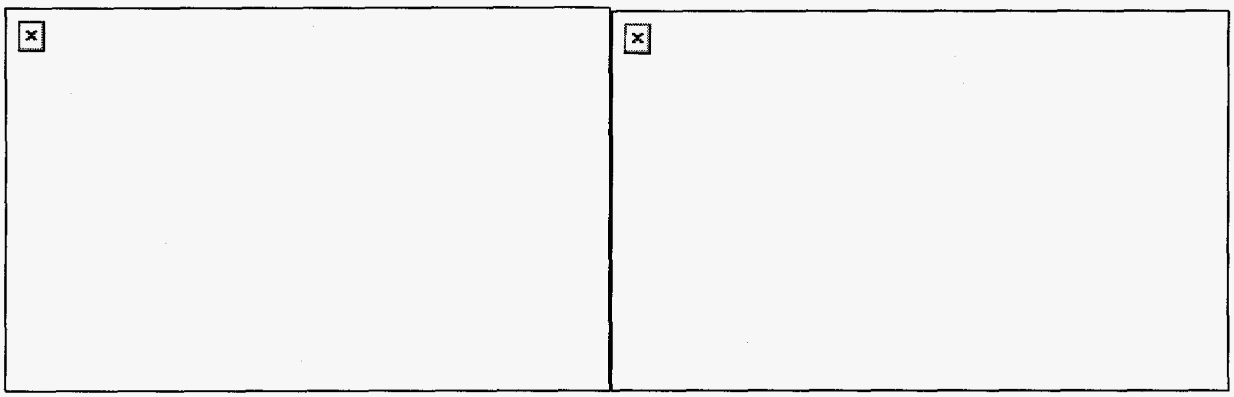


Figure 29: Near Earth Asteroid Mission Throughput and Thruster Count

Figure 28 compares the net mass performance of the single NSTAR, dual NSTAR, and NEXT systems on this Discovery mission. NEXT performs well overall, delivering over 200 kg. more payload than the single NSTAR system and requiring one fewer thruster to meet xenon throughput requirements. The number of PPU's required is the same for both configurations since only one NSTAR thruster operates at a time. NEXT's mass advantage over the dual NSTAR system is smaller, about 50 kg, and the main benefit of NEXT is that is achieves a moderate mass advantage with fewer thrusters and fewer PPU's. This simplifies the subsystem and results in some cost savings to the mission. A cost benefit analysis will be conducted in the future to quantify the cost benefits of this configuration.

VII. Conclusions

In support of the Next Generation Electric Propulsion (NGEP) program and NASA's Evolutionary Xenon Thruster development (NEXT), a significant effort has been undertaken to identify and analyze potential deep space mission applications for NEXT. An overview has been presented of seven concept missions that could potentially use NEXT for primary propulsion. The following conclusions have been reached.

• Titan Explorer – From a performance point of view, NEXT is well suited for this mission, particularly when combined with aerocapture for Titan orbit insertion. The combination of SEP and aerocapture significantly reduces trip time compared to all other options, delivering a 1,525 kg. spacecraft with a 35 kg. orbital science payload and a 365 kg. In-Situ probe in 5.7 years when launched on a Delta 4450. A SEP-chemical architecture delivers the required payload in 8.0 years on an Atlas V 551. Fully chemical propulsion architectures can be competitive with the SEP-chemical architecture in terms of trip time, but only when using a relatively rare Jupiter gravity assist opportunity. Non-Jupiter gravity assist opportunities require a longer, 10 year trip time.

• Neptune Orbiter – NEXT continues to meet the payload requirements of the Deep Space Design Reference Mission (DSDRM) to Neptune. A 4+1 configuration with a 30 kW solar array delivers a 850 kg. spacecraft with aerocapture system to Neptune when launched on a Delta IV 4240. The 850 kg. includes all elements of the spacecraft and aerocapture system. The science payload is unspecified, but is a small fraction of this mass. A payload increase greater than 10% of the current DSDRM requirement will require a larger launch vehicle. This was the case for the IISTP 2004 Focused Studies, which used an Atlas V 551 to meet a payload requirement of 1,070 kg with a similar transfer time.

• Mars Sample Return – Use of NEXT as primary propulsion for the orbiter and Earth Return Vehicle (ERV) substantially increases the mass can be delivered to Mars, enhancing the delivered mass capability of a Delta IV Heavy launch vehicle by over 2400 kg. when compared to a Type II chemical trajectory. However, use of NEXT also results in a significant time penalty, increasing mission duration by 1.8 years and requiring a 30 kW power system. These penalties may outweigh the mass advantage provided by NEXT for this particular mission architecture.

• Comet Surface Sample Return (CSSR) – NEXT performs well on this mission and provides the same mass performance as a much more complex and expensive NSTAR system. Analysis of a spacecraft launched on a Delta IV 4040 with a 12 kW solar array showed that a SEP system using 2 active NEXT engines and one spare (2+1) is able to provide a modest improvement in payload sent to the selected target and returned to Earth when compared to an NSTAR system using 4+1 engines. Further work is needed to compare the SEP architecture to missions using chemical systems for primary propulsion and to determine the optimum power level.

• Jupiter Polar Orbiter – A parametric analysis has been conducted on two different mission architectures using NEXT for heliocentric propulsion and chemical propulsion for orbit insertion at Jupiter. One architecture option would use RTG's for spacecraft power at Jupiter, the other, non-nuclear option would retain the solar arrays for spacecraft power at Jupiter. In both architectures, NEXT uses a Venus gravity assist, avoiding any environmental safety concerns that would be associated with an Earth gravity assist. In contrast, ballistic trajectories would likely require an Earth fly-by. The analysis spans multiple power levels, indicating that adding 1 kW of array power increases payload by roughly 5%, and launch vehicles and suggests that there are some regimes where a combination of 2+1 NEXT engines provide a mass performance advantage over all-chemical trajectories. In particular, there are no known SOA ballistic trajectories that would deliver a spacecraft greater than 900 kg in less than 4 years; a mission scenario that NEXT can meet. The 900 kg. includes all elements of the spacecraft. The science payload is unspecified and is a small fraction of this mass. Further work is needed to determine if this SEP architecture is cost effective when compared to all-chemical mission options.

The non-nuclear option is potentially applicable only for spacecraft requiring relatively low power. It does not apply if the payload requires high power for instruments or for data transmission. Further analysis is needed to determine if this non-nuclear option is actually feasible when spacecraft orientation, radiation degradation, and other environmental effects are considered.

• Discovery Missions – on a multiple-asteroid rendezvous mission, NEXT has a moderate mass advantage over NSTAR and requires one fewer thruster to meet mission throughput requirements. On a Near Earth Asteroid mission, NEXT significantly outperforms an NSTAR system with a single operating thruster, delivering more than 200 kg. additional payload to the final destination. NEXT system moderately outperforms a dual operating NSTAR system, but because NEXT has fewer thrusters and fewer PPU's, it has a simpler subsystem that results in some cost savings with no loss of mission performance. Further work is needed to quantify the cost benefits of NEXT for this mission.

This is a survey paper, intended to provide an overview of analysis conducted to date, and the amount of detail provided is limited in some cases. Much of the analysis represents ongoing work, and further details are available from other sources (as referenced) or can be obtained by contacting the authors directly.

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