

Nuclear Electric Propulsion for Uranus Missions Parametric Assessment

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Nuclear Electric Propulsion (NEP) is an exciting technology that has potential to enable low-thrust missions to Uranus that are otherwise impossible using traditional chemical propulsion. NEP provides enhanced mission flexibility by opening launch windows for such missions, potentially improves the payload delivery capabilities, and increases the power available upon arrival for use in propulsion and communications. This paper investigates the mission trade space, illustrates the tradeoff between time of flight from Earth to Uranus and the usable mass at destination, and considers the effect of different NEP system mass targets on mission performance and required power.

I. Introduction

NEP potentially enables novel missions to the outer solar system. NEP has a low technology readiness level, though it is currently under development by NASA's Space Nuclear Propulsion Program [1], [2]. Fission surface power (FSP) developments and additional technology maturation have advanced kW-class reactors to the point where they could be used for power anywhere in the solar system. The NEP system in this paper leverages FSP developments, assuming the same high-assay, low enriched Uranium (HALEU) as the FSP project, and assumes some improvement in mass through continued technology development. A range of mass per power values for the NEP vehicle are considered given the uncertainty in the result of future technology development. Similar power levels to the FSP project were also chosen: 10 – 40 kW_e. At this range of power, this paper will show that NEP systems have the potential to reduce transfer time and increase payload delivery, both while also providing power for science upon arrival. This presents an exciting opportunity for new science missions to these planets considering that current thermoelectric generators are limited to hundreds of watts.

NEP systems remain flexible upon arrival and can support landers, moon tours, and even sample returns. This paper primarily focuses on the outbound trajectory selection and resulting spacecraft/payload masses. Recent discussions with planetary scientists revealed that, relative to previous missions, increased payload capacity is not a primary driver. Instead, reduced transit times and more flexible launch windows are key priorities [3]. For this reason, faster than normal transit times were investigated. A variety of launch vehicles were investigated to further demonstrate mission flexibility, though to avoid any concerns of launch vehicle availability only Falcon Heavy Expendable results are presented in this paper. The methodology allows post-processing the trajectory results for a variety of NEP system mass assumptions as well as different EP thruster options, both of which are discussed in the methodology. Similar analysis has been performed for missions to Neptune using similar methodology [4]. This paper focuses on Uranus and considers a wider range of possible NEP system mass per power. The results illustrate the optimal power to maximize payload for a given NEP power system mass assumption, vehicle mass model and launch vehicle selection. The tradeoff between time of flight to Uranus and the usable payload and propellant mass at destination shows times of flight between 9.5 and 17 years yield 150 to 2000 kg of usable mass depending on the NEP system mass per power and the power level chosen.

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II. Methodology

The first step in analyzing NEP missions to Uranus was selecting what trajectories were of interest. Because launch flexibility was a priority of this study, complicated trajectories with multiple flybys were avoided. Though these trajectories may be more optimal than ones investigated, they occur too infrequently to rely on. Instead, this paper focuses on trajectories utilizing only Earth flybys since they can be utilized regardless of year or positioning of other planets. Several of these trajectories to Uranus were considered, as described in Table 1.

Table 1. Destination and Trajectories

Destination	Flyby Sequence	Strengths	Weaknesses
Uranus	Direct	Fastest	No added C_3
	EEU (1-year coast before EGA)		
	EEU (2-year coast before EGA)		
	EEEU (1-year, then 2-year coast)	Slowest	Most added C_3 from EGA

Each flyby sequence comes with its own strengths and weaknesses in terms of time of flight (TOF) and extra energy gained from the flybys. The direct trajectory doesn't use any Earth Gravity Assists (EGAs) so it is theoretically able to reach Uranus the soonest, however it is 100% reliant on the launch vehicle's C_3 to put it in a high enough Earth orbit at the beginning of the mission. The double EGA shown on the bottom of the table is the slowest with three years of coast combined between the EGAs, however it does fly by Earth twice, gaining lots of additional velocity on its way to Uranus.

Although the combination of NEP and these simplified trajectories can enable extremely flexible missions, there are still optimal launch dates to minimize ΔV . Porkchop plots were generated for an Earth – Uranus trajectory to find approximately when the optimal Earth departure was using the minimum realistic C_3 and TOF values. A heavy NEP system reduces C_3 , but EP makes up for the lower C_3 and enables lower $V_{\infty, arrival}$, leading to a smaller capture burn for UOI. These launch date guesses were then used to create reference decks for each flyby sequence with Copernicus, a NASA trajectory optimization software [5]. The launch date was left free so that Copernicus could vary the launch date for the minimum ΔV . Initially these reference decks were just going to be created for the 2040 launch opportunity, however research was done to ensure this was a representative launch year. Copernicus decks were built for every year from 2035 to 2048 and the resulting ΔV was recorded. The differences in ΔV between launch years are shown below for a EEU trajectory with a two-year coast before the EGA, which will be referred to as the E(2y)EU trajectory.

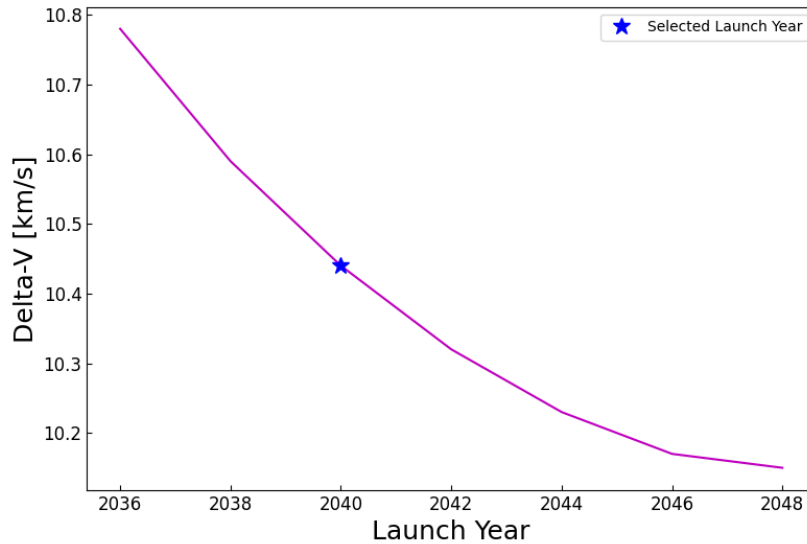


Figure 1. E(2y)EU ΔV vs Launch Year

With a representative launch year chosen, the final versions of the reference decks were created. Shown below are the EU and E(2y)EU reference decks.

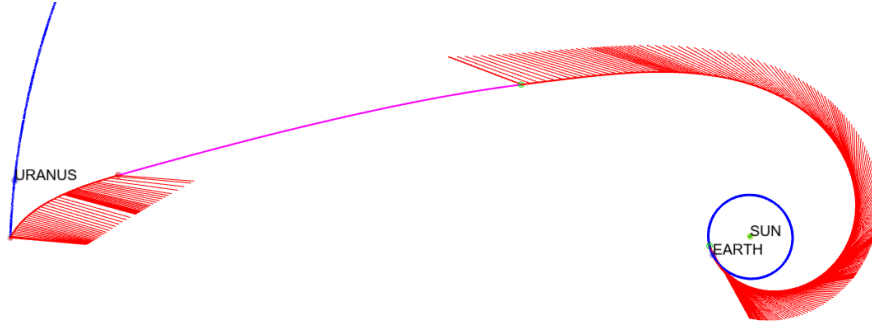


Figure 2. EU Copernicus Reference Deck (C_3 20, TOF 2920 days, I_{sp} 4000s, $V_{\infty, arrival}$ 3 km/s)

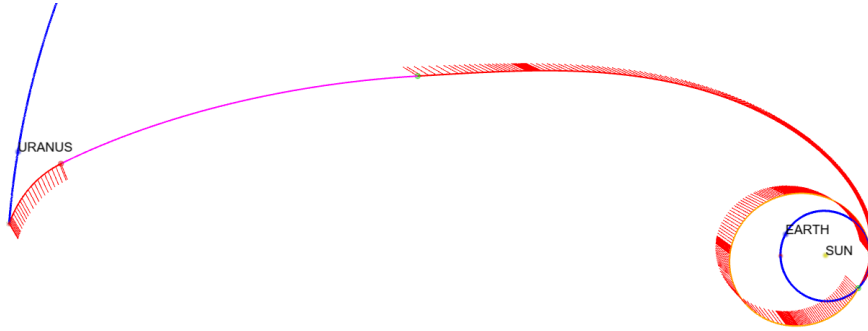


Figure 3. E(2y)EU Copernicus Reference Deck (C_3 20, TOF 4927 days, I_{sp} 4000s, $V_{\infty, arrival}$ 2.5 km/s)

There are many differences between these trajectories, however the most obvious one is the significantly higher thrust level on the direct trajectory. This is because the direct trajectory is not gaining any additional C_3 from an Earth flyby so it must have a higher thrust level to compensate. The E(2y)EU trajectory gains C_3 from the Earth flyby shown in the picture, however this does limit the range of C_3 s this mission can be launched with because above a certain launch energy you actually burn too far from the Earth to do a two-year EGA and must waste fuel shrinking that orbit. The two-year mission above also shows that the spacecraft will have a nearly 100% duty cycle during the EGA, however this drops to about 70% during the interplanetary leg.

Initial sweeps in Copernicus were done using an in-house python-based wrapper called CAPYBARA (Copernicus Analysis & Processing for Yielding Batch Astrodynamics Results & Assessments) that iterated each mission's parameters across areas of interest. Initially the range was wide but sparse, but density was added to the trade space parameters as areas of interest were identified. The total range of inputs swept across is shown in the tables below.

Table 2. EU Trade Space

	C_3 (km ² /s ²)	TOF (years)	I_{sp}	$V_{\infty, arrival}$
Minimum Value	20	7	4000	0
Maximum Value	45	17	8000	3

Table 3. E(2y)EU Trade Space

	C_3 (km ² /s ²)	TOF (years)	I_{sp}	$V_{\infty, arrival}$
Minimum Value	0	9	4000	0
Maximum Value	30	17	8000	3

The direct trajectory was run to a higher C_3 value and shorter TOF to account for the lack of any EGA. Cases with an EGA require the spacecraft to return to Earth; higher launch C_3 's eventually result in diminishing returns as the spacecraft needs to slow for its return. The EEU trade space was narrowed down after some cases were observed to be burning back towards Earth, wasting ΔV in the process. The direct trajectories were also investigated down to 7 years as opposed to 9 years with an EGA due to the EGA adding 2 years to the TOF. Both trajectories were run at 4000 and 8000 s I_{sp} representing the NEXT Ion Thruster [6] and HiPEP Ion Thruster [7], respectively. Additionally,

a trade study was done to identify the optimal arrival v_{∞} . This was done by balancing the propellant needed for the chemical UOI burn with the NEP propellant used to slow the spacecraft prior to UOI. A Copernicus deck was created to solve for the needed chemical burn dependent on arrival velocity; a $V_{\infty, \text{arrival}}$ of 2.5 km/s requires a chemical UOI of 210 m/s is used. For both trajectories the target orbit at Uranus is the same as UOP's arrival orbit: r_a of 5.04E6 km, and r_p of 28,400 km.

The Copernicus deck optimizes the trajectory to minimize power for each launch vehicle C_3 , TOF, and I_{sp} of interest. The final mass is fixed at 1000 kg, while the initial mass is left free. This solves for the maximum spacecraft α_{sc} . Here, α_{sc} is defined as the final mass in kilograms per kW jet power. This is not to be confused with α_{ps} , defined above as the mass per power of the NEP power system in kg per kW_e delivered to the thruster PPUs. This approach solves for the maximum delivered mass to Uranus for a given power. However, the maximum delivered mass does not necessarily mean the largest payload. The raw Copernicus data must be post-processed to account for tankage mass of the NEP and chemical propellant and other subsystems that are sized as a function of the spacecraft mass. The lowest power that closes a mission typically will have a higher duty cycle than a mission optimized for a given trajectory to minimize ΔV . Figure 4 is a demonstration; for a given TOF, C_3 , and I_{sp} the trajectory was solved for a range of possible α_{sc} values, and then post-processed with a given dry mass build up and an assumed α_{ps} . This shows that the minimum power solution – the maximum α_{sc} – is not the highest payload but is close enough to be representative of a possible payload for those parameters, in this case falling within 4%.

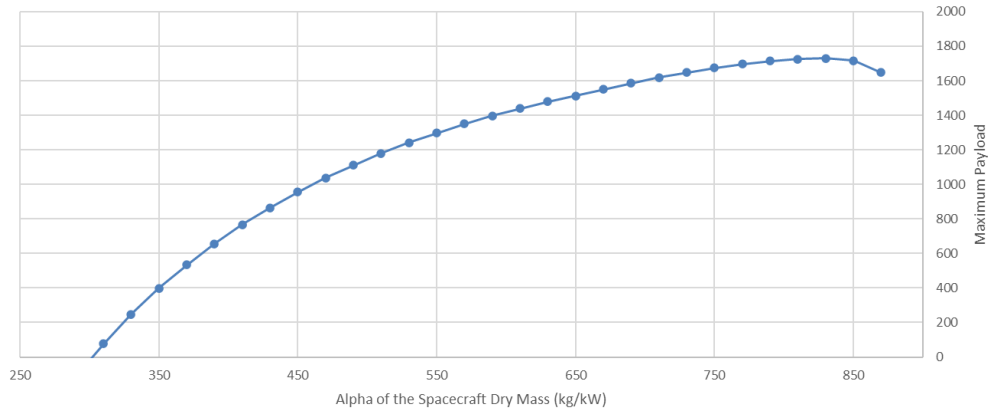


Figure 4. Example Maximum Payload Delivered for an EEU Trajectory with an initial C_3 of 21, TOF of 12 years, and I_{sp} of 4000s

The trajectory was optimized across a sparse sampling of the trade space defined in Tables 2 and 3 above. The results were post-processed to determine which scenarios were worthwhile for further analysis. E(2y)EU – an EGA occurring two years after launch – was selected to run in more detail. The direct EU case, E(1y)EU, and EEEU performed worse than E(2y)EU in the initial results when post-processed with the mass model using any of the three α_{ps} curves from Figure 5. EU and E(1y)EU did not deliver as large of a payload as E(2y)EU. Shorter TOF was of interest, so the EEEU case spending three years to perform Earth flyby sequences did not allow short times of flight.

Once the initial data is output from Copernicus, it is post-processed utilizing an in-house python code OCELOT (Object-oriented CONOPS Evaluation and Logistics Operations Toolkit) designed to scale the results and estimate the masses of each subsystem. The NEP power system mass is estimated as a function of power using one of the curves in Figure 5. Each curve represents the specific mass of NEP power systems with different technology assumptions as a function of electrical power. The ‘Gen 0’ and ‘Gen 1’ curves represent systems directly derived from current technology developments, such as NASA’s fission surface power project, and from systems with lower mass that would require further technology investment, respectively. Investments that may enable such mass reductions include lighter reactor designs that scale better to lower power, reliable Stirling or higher-temperature Brayton power conversion, and novel shielding materials. The ‘Estimate’ line represents the mass estimate output of a system sizing model of the NEP power system with a specific set of technology assumptions (e.g., HALEU sodium heat pipe cooled

reactor with a specific cross-section design, Brayton power conversion operating at 1200 K power conversion inlet temperature).

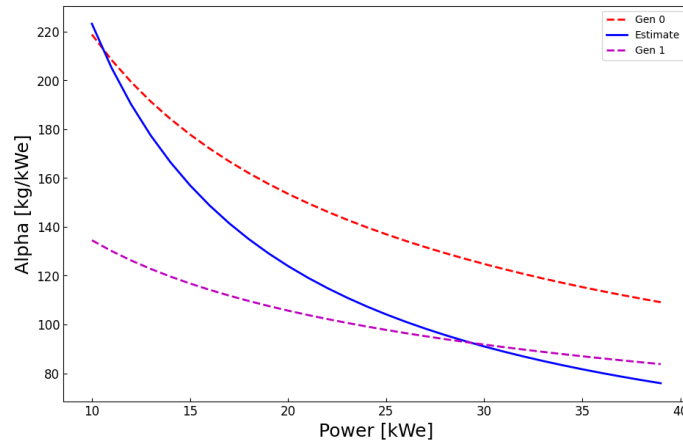


Figure 5. Power system specific mass (α_{ps}) as a function of power for three sets of technology choices: derived from current investments, scaled around a specific reactor and power conversion point estimate, and the result of additional technology investments.

The post-processor scales up the initial mass, and power until the initial mass matches the launch vehicle capability. This maintains the thrust-to-weight ratio and the validity of the trajectory solution. Subsystem masses are estimated using physics-based sizing for some and component selection for others. Specific hardware was selected for the electric thrusters: NEXT-C Ion thrusters for 4000 s I_{sp} and Hi-PEP Ion thrusters for 8000 s I_{sp} . These are existing thrusters developed by NASA for EP, and the selections represent the most efficient thrusters for their specific impulse. Propellant mass is estimated based on a user-defined delta-V schedule as in Table 4.

Table 4. E(2y)EU Delta-V Schedule Example

Maneuver	Delta-V (m/s)
Earth Departure Burn	3,727
Uranus Injection Burn	6,094
Uranus Slowdown Burn	3,921
Uranus Orbit Insertion Burn	210
Moon Tour at Uranus	150
RCS Reserve	75
Total	14,177

The power system mass is estimated using the Figure 5 curves after determining the required electric power by applying thruster efficiency, PPU efficiency, and a performance margin to the scaled jet power. The maximum available payload is determined after a dry mass margin is applied by comparing the calculated mass values to the scaled fixed final mass from Copernicus. The difference between the mass values is interpreted as the available payload. If the available payload is negative (the dry mass exceeds the scaled final mass) then the mission does not close for that launch vehicle at that C3 for the assumptions made in the spacecraft mass buildup. This means that the assumptions on the NEP system and its mass are key in determining the payload mass that can be delivered. A sample MEL is shown below.

Table 5. Sample MEL

	Basic Mass (kg)	MGA (%)	Predicted Mass (kg)
Power System	2469.3	15	2839.7
Auxiliary Power	234	20	280.8
EP Thrusters and PPU	278.6	10	306.5
EP Tanks and Piping	343.6	20	412.3
RCS Thrusters	5.4	10	5.9
RCS Tanks and Piping	59.7	20	71.6
Structure	650.8	20	780.9

Communications	54	20	64.8
CDH	35	20	42
GNC	97	20	116.4
Thermal	62	20	74.4
Payload	700	0	700
Dry Mass	4,989.4	18	5,695.3
RCS Propellant	651.2	4	677.3
EP Propellant	3,196.7	4	3,324.5
Dry Mass Margin		18	1,025.2
Wet Mass			10,722.3
LV Capability		20	10,937.1
Total Dry Mass Margin	34.7% (705.9 MGA + 1,025.2 margin)		

III. Results

Post-processing each combination of NEP generation and I_{sp} reveals the trajectories that close with a positive payload delivered to Uranus. However, sweeping over three dimensions leads to a difficult to visualize trade space. For simplification only one I_{sp} is plotted at a time. To simplify the data further, a launch vehicle curve can be applied to show only trajectories enabled by a specific launch vehicle. Falcon Heavy Expendable is the focus of this paper due to a combination of high availability and low cost. The maximum payload for each trajectory is solved by comparing the wet mass of the spacecraft from the MEL to the available C3 of the spacecraft. The images below compare the possible payload delivered to Uranus orbit on a direct trajectory for a 4000 s NEXT Ion thruster and an 8000 s HiPEP Ion thruster, both launched on a Falcon Heavy Expendable.

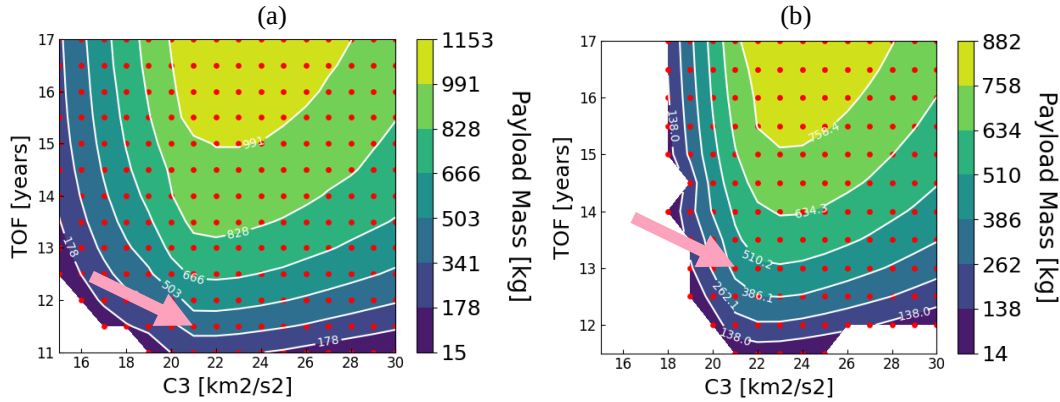


Figure 6. Maximum Payload with Gen 0 NEP System at (a) 4000 s and (b) 8000 s I_{sp}

The maximum payload delivered shown on the color bar is the key takeaway from these plots. A generation 0 NEP system using thrusters with an I_{sp} of 4000 s delivers almost 300 kg more than a system that uses 8000 s thrusters. These plots show that 4000 s thrusters could deliver the same payload of 400 kg as 8000 s thrusters 1.5 years faster with the same launch energy, signified with the pink circles on each plot. Moving to the AMA estimated NEP vehicle does affect this comparison as can be seen in the following figures.

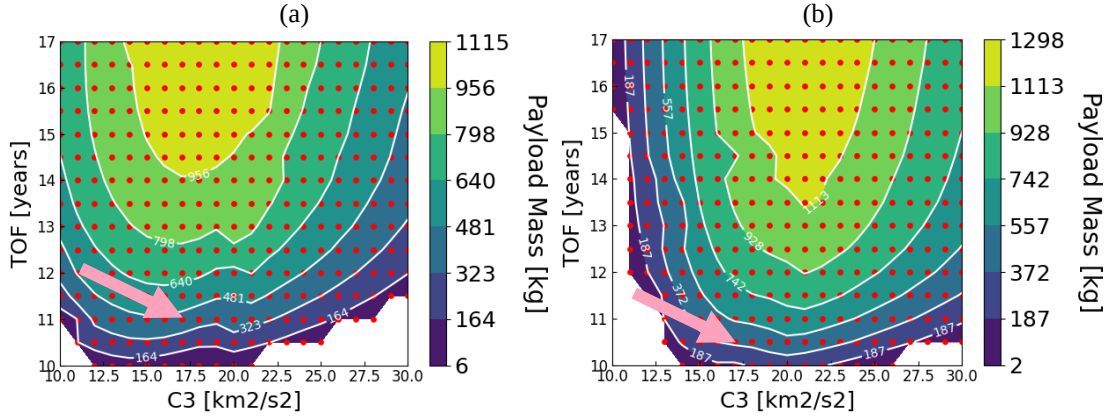


Figure 7. Maximum Payload with Estimated NEP System at (a) 4000 s and (b) 8000 s I_{sp}

The estimated generation results are slightly different than generation 0, with the 8000 s thrusters converging on higher payloads but requiring higher launch energies. Another significant difference is that the ‘estimated’ system enables positive payloads with TOF as short as 10 years, whereas the fastest missions with ‘gen 0’ had TOF of 11 years. In these plots we see the same 400 kg payload pointed out in Figure 6 is delivered 6 months faster with 8000 s than with 4000 s with the ‘estimated’ NEP system. This shows that the estimated generation NEP vehicle would likely benefit from operating at a higher I_{sp} by using HiPEP Ion thrusters as opposed to NEXT. Figure 7 also shows that the 400 kg payload discussed can be delivered using the ‘estimated’ NEP system as much as 2.5 years faster than generation 0 at 8000 s, demonstrating the significance of technology maturation to these missions.

The corresponding powers to these maximum payloads are plotted in the same manner below, again for generation 0 NEP at 4000 and 8000 s I_{sp} .

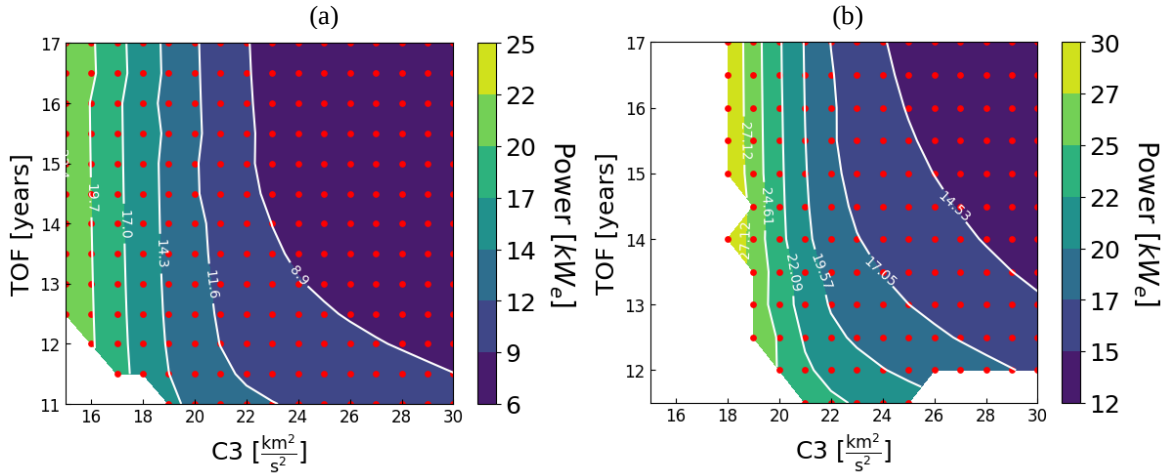


Figure 8. Minimum Power of Gen 0 NEP System at (a) 4000 s and (b) 8000 s I_{sp}

Figure 8 shows the minimum power required to deliver the payloads shown in Figure 6. As C3 and TOF increase, required power goes down. There is more time for thruster operation, enabling a lower thrust level. The higher I_{sp} 8000 s system requires higher power to optimize the same trajectories due to its lower thrust for a given power level and I_{sp} . The same plots are also shown for a generation 1 NEP system below.

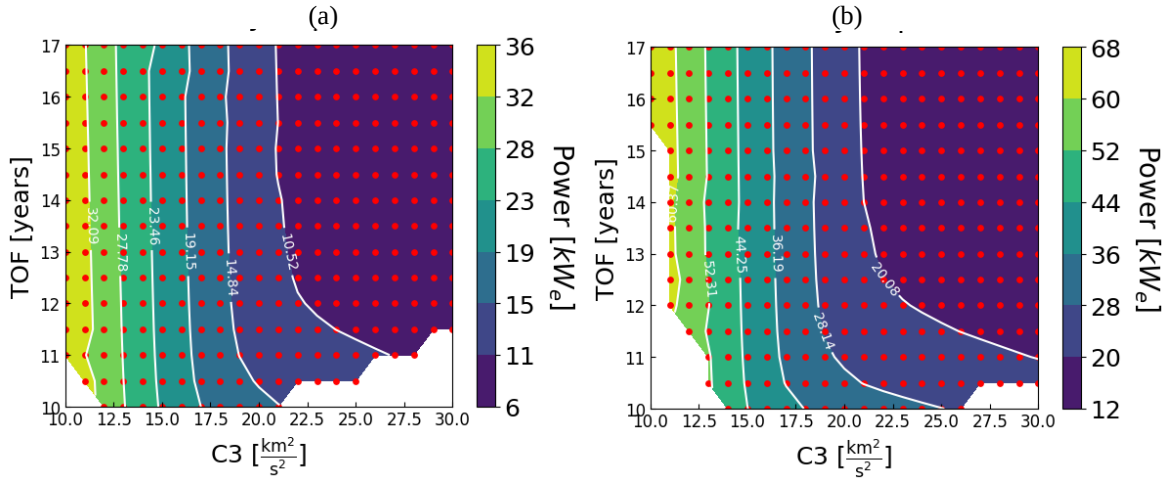


Figure 9. Minimum Power of Estimated NEP System at (a) 4000 s and (b) 8000 s I_{sp}

Figure 9 shows the minimum power required to deliver the payloads shown in Figures 7. Earlier it was shown that an estimated generation NEP vehicle operating at 8000 s delivers more payload than a NEP vehicle operating at 4000 s. However, these contours show how that would require a much higher power reactor. At 4000 s the highest converged power was 36 kW_e, however at 8000 s that increases to 57 kW_e. Building a reactor at nearly 20 kW_e higher power may well not be worth it for a less than 200 kg increase in payload mass, but again the mission architecture affects all these trades.

To simplify even further, the maximum payload for every TOF is pulled from those contours and plotted against TOF (the blue line in Figure 10 below). AMA's best estimates for generation 0 and generation 1 NEP systems are shown with the dotted lines to see where each I_{sp} fits between the generations. The two bottom points are two trajectories from the Uranus Orbiter Probe study for comparison [8].

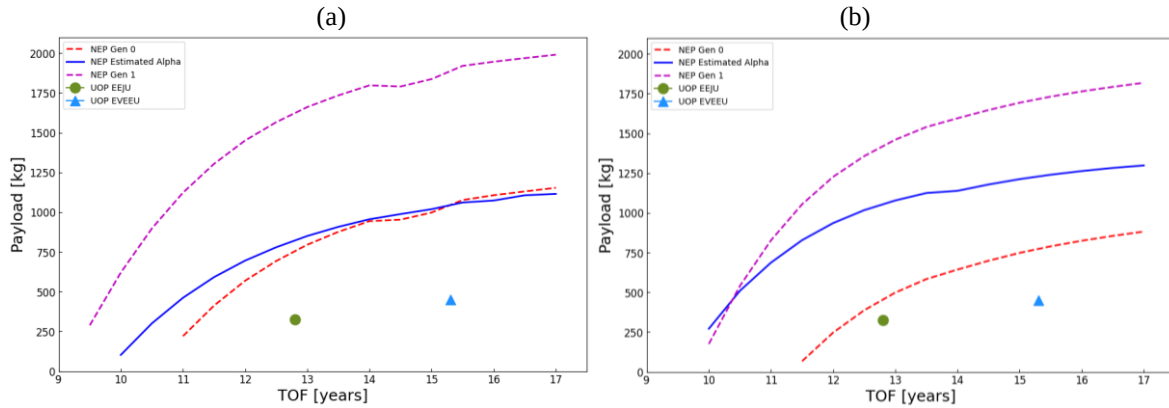


Figure 10. Maximum Payload of Estimated NEP System at (a) 4000 s and (b) 8000 s I_{sp} with bounding lines showing Generation 0 and Generation 1 performance

The figures above show that regardless of TOF, I_{sp} , and even NEP generation, NEP can deliver more payload than UOP. The exciting takeaway is that even at shorter TOFs than UOP's trajectory, NEP could still deliver up to 3 times the payload mass. The UOP payload utilizing an EEJU trajectory could be delivered by the estimated NEP system in about 10.5 years at 4000 s I_{sp} and 10 years at 8000 s. For the estimated alpha system, an I_{sp} of 8000 s delivers more payload than 4000 s at every TOF, however that is associated with higher power levels shown below.

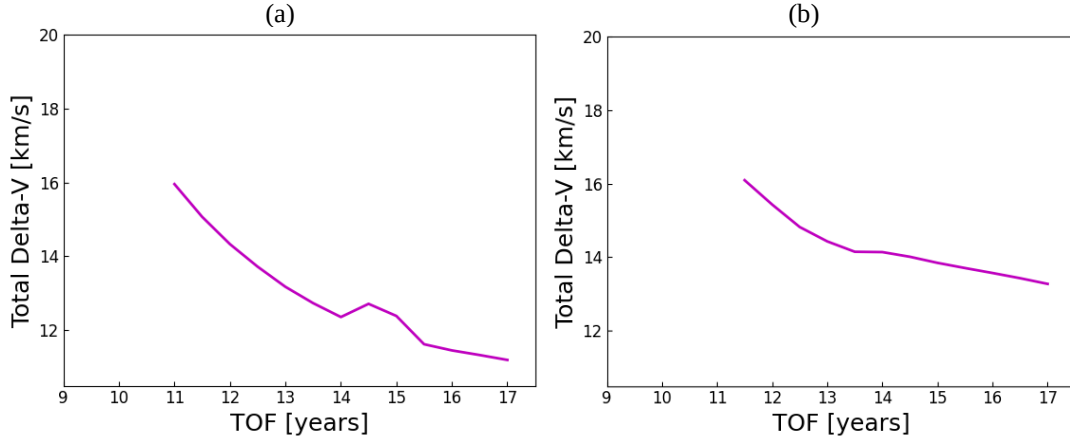


Figure 11. Delta-V for Maximum Payload of Generation 0 NEP System at (a) 4000 s and (b) 8000 s I_{sp}

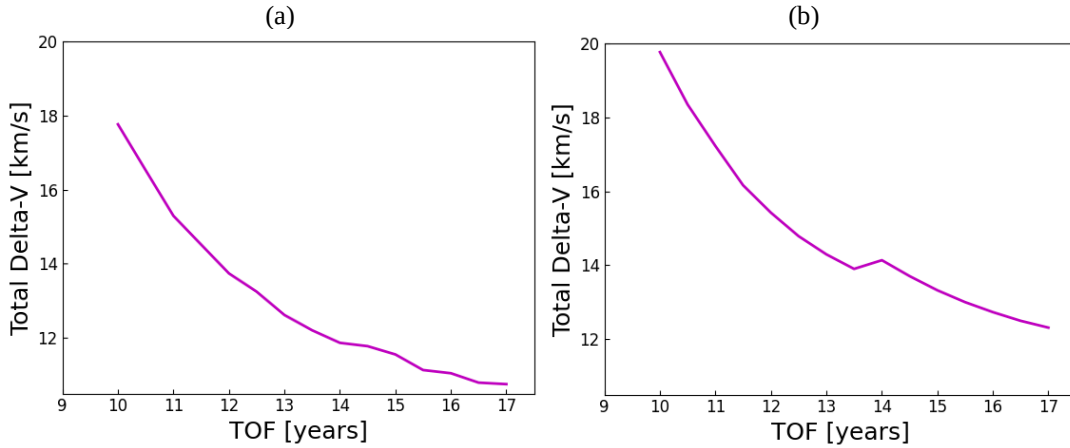


Figure 12. ΔV for Maximum Payload of Estimated NEP System at (a) 4000 s and (b) 8000 s I_{sp}

All four plots show the same trend of ΔV decreasing as TOF increases. A shorter TOF allows the trajectory to be more optimal as it becomes closer to a Hohmann transfer versus more of a shortcut trajectory. 8000 s is shown to require more ΔV than 4000 s which tracks with its associated higher power shown before. Moving from Generation 0 to the Estimated NEP system sees roughly a 1 km/s ΔV decrease across all TOFs at 4000 s. At 8000 s the differences are larger at the extremes of TOF, but the ΔV requirements are approximately equal from 12 to 14 years.

IV. Discussion and Conclusion

This paper demonstrates potential missions to Uranus across a range of NEP systems. Due to the uncertain nature of the performance of these systems, a range of power, α_{ps} , and I_{sp} values were investigated. To investigate this the large trade space, a parametric approach to the mission analysis was used to allow determination of delivered usable mass as a function of TOF for a range of chosen power system and spacecraft assumptions.

At the low-performance end of the range considered, the delivered mass is comparable to that of the chemical Uranus Orbiter and Probe decadal survey mission study. This system may still enable new science by providing kilowatts of power and thus data rate from the Uranus system. However, the benefits of EP in the Uranus system are limited, with little available mass for propellant to perform maneuvers in the Uranus sphere of influence.

At the higher-performance end of the power system specific mass considered, there is ample delivered mass available to enable maneuvers with significant ΔV requirements, such as plane changes and spiraling in/out of moon orbits. This capability would enable new mission designs as well as high communication data rates for a different instrument suite selection. A 20 kW_e system with modest improvement over an FSP-derived system, provides enough delivered mass that a compelling mission may be designed.

For any of the power system performance levels considered, there remain significant challenges to be overcome in the technology development. These include long-life power conversion that must operate for the full 12- to 18-year

mission. It is worth noting that a lower-mass power system enables a shorter Earth-Uranus transit time that could relax the life requirement by two to three years compared to the heaviest power system considered. An 11- to 15-year mission may be significantly more attractive than a 14 to 18-year mission.

This analysis should inform NASA's plans for technology development of NEP for science mission applications. A Uranus mission places even more emphasis on low mass and long operational life than do current developments for fission surface power. Additionally, the analysis here illustrates that the optimal power level for an ice giant science mission NEP is likely between seven and 25 kW_e, depending on how the reactor concept scales with thermal power and the achieved efficiency of the power conversion system. Off-optimal higher power levels can close the mission with much less penalty than the difference in power system mass due to the lower ΔV possible with higher thrust-to-weight ratio, but there is still significant penalty going to off-nominal power. Different launch and Earth-departure capabilities do have an impact on the optimal power, with more-powerful launch vehicles favoring larger, higher-power NEP systems. These considerations may drive additional or different technology choices to meet the more demanding mission requirements and may provide justification for a different-scale power system.

Based on the parametric analysis completed, further study and development of a conceptual point design including selection of a science package that takes advantage of the available power at destination is warranted as follow-on work. This is needed to evaluate the degree to which new science is enabled and to more specifically quantify the amount of propulsion needed at destination and the resulting requirement on power system specific mass.

Acknowledgments

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V. References

- [1] K. A. Polzin, A. K. Martin, F. Curran, R. Myers and M. Rodriguez, "Strategy for Developing Technologies for Megawatt-class Nuclear Electric Propulsion Systems," in *International Electric Propulsion Conference*, Boston, MA, 2022.
- [2] M. E. Duchek, W. Machemer, C. Harnack, M. Clark, A. Pensado, K. Polzin, A. Martin, F. Curran, R. Myers, M. Rodriguez, D. V. Rao, R. Dyson and R. Scheidegger, "Key Performance Parameters for MW-Class," in *AIAA ASCEND*, Las Vegas, NV, 2022.
- [3] T. Reuter, R. Myers, P. Christensen, L. Dudzinski and K. Polzin, "Accelerating Space Science with Nuclear Technology: The Tempe Workshop," Institute for Space Science and Development, Hastings-on-Hudson, NY, 2023.
- [4] T. Kokan, C. R. Joyner, D. J. H. Levack, B. J. Muzek, R. W. Noble and C. B. Reynolds, "Nuclear Electric Propulsion Outer Planetary Science Mission Concepts," in *AIAA SciTech*, Orlando, FL, 2025.
- [5] J. Williams, *Copernicus Version 5.2.0*, NASA Johnson Space Center, 2022.
- [6] G. C. Soulas, M. T. Domonkos and M. J. Patterson, "Performance Evaluation of the NEXT Ion Engine," in *39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*, Huntsville, 2003.
- [7] J. E. Foster, T. Haag, M. Patterson, G. J. Williams Jr., J. S. Sovey, C. Carpenter, H. Kamhawi, S. Malone and F. Elliot, "The High Power Electric Propulsion (HiPEP) Ion Thruster," in *40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*, Fort Lauderdale, 2004.
- [8] A. Simon, F. Nimmo and R. C. Anderson, "Uranus Orbiter & Probe," National Aeronautics and Space Administration, 2021.
- [9] M. Duchek, A. Boylston, D. Langford, S. Greenhalge, K. Polzin and R. Myers, "Nuclear Electric Propulsion for Outer Planet Science Missions," in *International Electric Propulsion Conference*, Toulouse, France, 2024.