AAS 11-189

INTERPLANETARY ELECTRIC PROPULSION URANUS MISSION TRADES SUPPORTING THE DECADAL SURVEY

John W. Dankanich* and James McAdams†

The Decadal Survey Committee was tasked to develop a comprehensive science and mission strategy for planetary science that updates and extends the National Academies Space Studies Board's current solar system exploration decadal survey. A Uranus orbiter mission has been evaluated as a part of this 2013-2022 Planetary Science Decadal Survey. A comprehensive Uranus orbiter mission design was completed, including a broad search of interplanetary electric propulsion transfer options. The scope of interplanetary trades was limited to electric propulsion concepts, both solar and radioisotope powered. Solar electric propulsion offers significant payloads to Uranus. Inserted mass into the initial science orbit due is highly sensitive to transfer time due to arrival velocities. The recommended baseline trajectory is a 13 year transfer with an Atlas 551, a 1+1 NEXT stage with 15 kW of power using an EEJU trajectory and a 1,000km EGA flyby altitude constraint. This baseline delivers over 2,000kg into the initial science orbit. Interplanetary trajectory trades and sensitivity analyses are presented herein.

INTRODUCTION

Mission design trades were carried out in support of the decadal mission survey for a Uranus System mission.[1] The scope of trades presented is limited to interplanetary electric propulsion trajectories, both solar and radioisotope powered, to Uranus. Trades originally included broad search of Neptune orbiter solutions, but the detailed results were only requested for Uranus. Several guidelines were provided for the Uranus mission trade space. The solutions must have a reasonable backup solution. The timeline for investigation is from 2018 to 2023 launch dates. The total transfer times considered ranged from 10 – 13 years. The final mission design selected was provided to the APL ACE team for a preferred point design mission study. The mission study was a Concept Maturity Level 4; a point design to subsystem level mass, power, performance, cost, and risk.[2]

The interplanetary trajectory is influenced by the arrival conditions at Uranus and minimizing the excess velocity. The final result is from an iterative process with the science orbit specified as shown in Figure 1 with a periapsis of 1.3 R_{Uranus} by 20.08 days. Following the primary science campaign, the spacecraft must have enough mass for a 730 m/s chemical ΔV to complete the satellite tour shown in Figure 2. The orbit insertion and tour details are provided by McAdams et al.[3]

* Aerospace Engineer, NASA's In-Space Propulsion Technology Project, 21000 Brookpark Rd., M/S 77-4, Cleveland, OH, 44135.
† Mission Design Lead Engineer, The Johns Hopkins University Applied Physics Laboratory, 11100 Johns Hopkins Rd, Laurel MD, 20723.

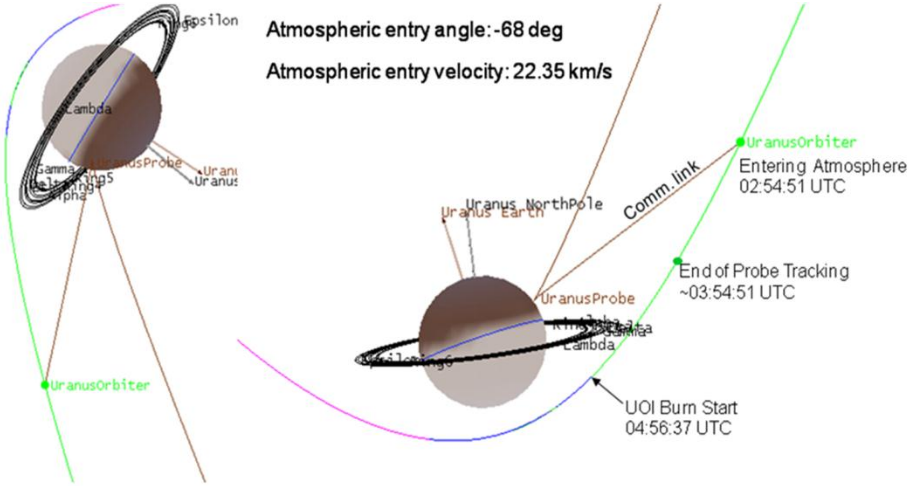

Figure 1. Uranus orbiter and prove arrival trajectories.

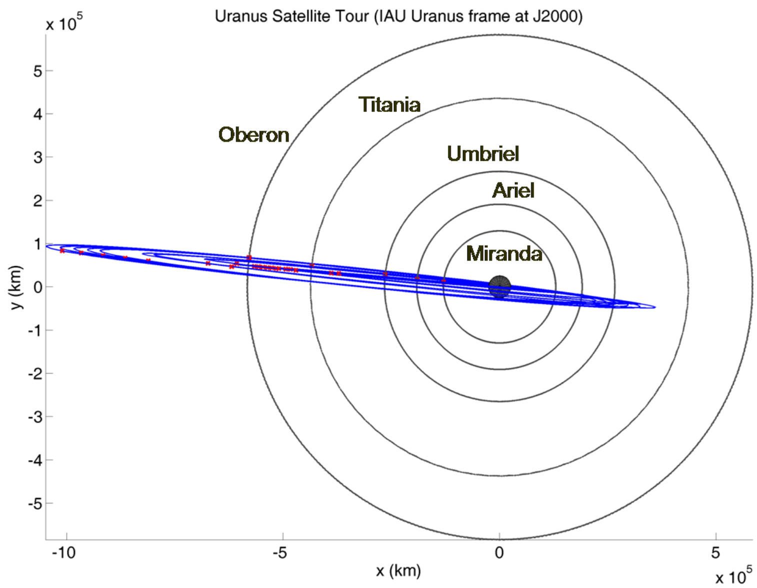

Figure 2. Uranus satellite tour from initial science orbit.

BASELINE TRAJECTORY ASSUMPTIONS

The electric propulsion mission trades were conducted using the Mission Analysis Low-Thrust Optimization (MALTO)[4] Tool. The electric propulsion margins and assumptions can have significant effect on the results. The baseline assumptions are provided in Table 1 unless otherwise stated. Also, the solar electric propulsion (SEP) thrusters assume a thruster model that can throttle from minimum to maximum power while changing thrust and specific impulse to coincide with the demonstrated performance of the thruster. For radioisotope powered electric propulsion trades, the thruster is allowed to operate at the optimal specific impulse for the specified thruster efficiency. This was chosen because the SEP thrusters are at Technology Readiness Level (TRL) 6 while the REP thruster is at TRL 3.[5]

Table 1. Thruster and optimization assumptions

	Solar Electric Propulsion	Radioisotope Electric Propulsion
Power, kW*	12	0.7
Housekeeping Power, kW	0.0	0.0
Thruster Efficiency, %	NEXT[6]	55%
Specific Impulse, s	NEXT	Optimized
Duty Cycle, %	90%	90%
Solar Array Model	Ultraflex	NA
Number of Thrusters	2	1
Launch Vehicle	Atlas 551	Atlas 551 w/ Star 48

* Solar power specified at 1 AU, radioisotope power is constant throughout the mission

SOLAR ELECTRIC PROPULSION RESULTS

As was shown by Landau, Lam and Strange,[7] the combination of solar electric propulsion and gravity assists enable missions with larger payloads than with chemical propulsion over a broad range of flight times and power levels. Several of the highest performance trajectory solutions are provided in Table 2. The broad search included launch opportunities between 2018 and 2030. The baseline trajectories from Table 2 are shown in the appendix. These solutions are for 10 year transfer times using the NEXT 2+1 SEP stage with 15 kW of power as proposed for the Titan Saturn System Mission.[8] Inserted mass estimated in tables 3-8 used a SEP stage mass with a CBE of 625kg that is dropped prior to the chemical orbit insertion.

Table 2. Highest performing SEP trajectory sequences to Uranus.

Trajectory Sequence	Launch Date	Delivered Mass, kg	Propellant Mass, kg	Arrival V_∞, km/s
EEJU	7/14/2020	3756	411	8.67
EEJU	5/24/2019	4238	330	9.90
EVVEJU	8/31/2018	4941	426	10.96
EVEJU	1/29/2020	4020	440	9.41
EMEJU	2/5/1019	4245	311	10.51
EESU	8/4/2026	3378	426	8.97
EEEJU	3/10/2018	5627	639	13.16

Overall, performance to Uranus is greatly increased with the use of a Jupiter Gravity Assist (JGA). Because of this, and the existence of primary and backup solutions in 2019 and 2020, they were the original the baseline solar electric propulsion options within the 2018 – 2023 timeframe. For planning missions of the next decade, it is important to note that the phasing of Jupiter and Uranus will make the Jupiter gravity assist to Uranus only available for launch opportunities before 2020. After Jupiter gravity assists become unavailable, Saturn flybys will provide the greatest delivered mass capabilities.

One of the largest challenges to a Uranus mission is the distance that must be traveled to reach Uranus. Because of the distance that must be traveled, the spacecraft must either have a very long transfer time or arrive with a very high V_∞. The high arrival velocities require a very large chemical Uranus Orbit Insertion (UOI) maneuver to enter the science orbit. The UOI ΔV as a function of V_∞ was provided by APL and is shown below in Figure 3. Aerocapture can enable a shorter mission time with greater delivered mass, but the risk associated with avoiding the rings eliminated Aerocapture from the trades.

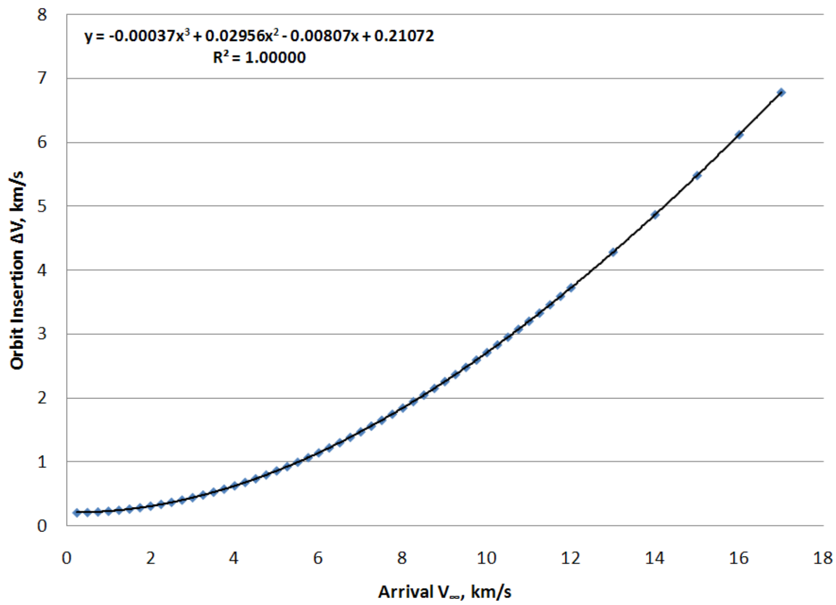

Figure 3. Uranus orbit insertion ΔV as a function of V_∞.

Recommended EEJU Solutions and Trades

The EEJU trajectory provides the higher delivered mass capability with a viable backup opportunity. The 2019 and 2020 launch EEJU trajectories for 10 year solutions are shown in Figures 4 and 5 respectively. The mission must be designed for the lower performance solution with the worst case launch energy and propellant load. Due to its lower performance, trades are conducted on the 2019 launch solution.

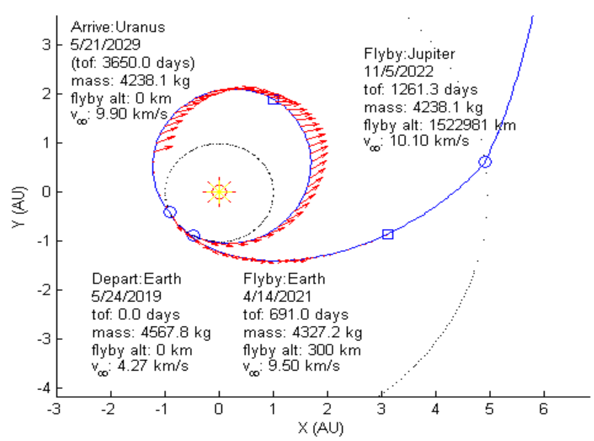

Figure 4. EEJU 10 year trajectory for 2019 launch opportunity.

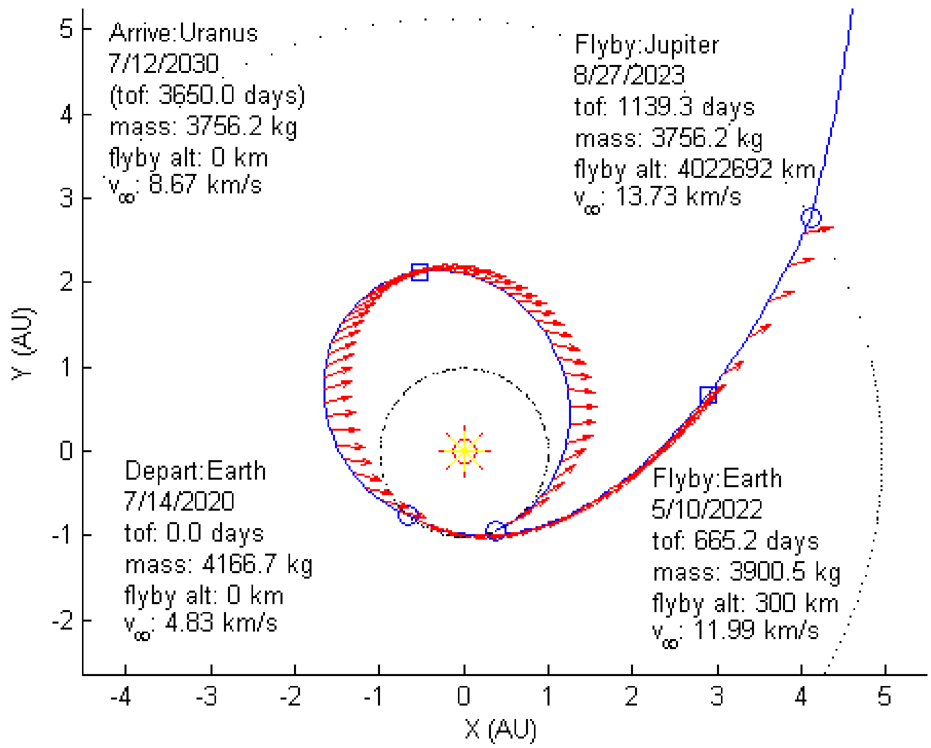

Figure 5. EEJU 10 year trajectory for 2020 launch opportunity.

Trip Time Sensitivity

As shown in Table 3 and Figure 6, increasing the trip time from a 10 year transfer to 13 years can significantly reduce the arrival velocity and therefore increases the overall delivered mass. The transfer time has negligible effect on the EP propellant for delivered mass capability.

Table 3. Transfer time effect on inserted mass.

Tranfer Time, years	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	Chemical Propellant, kg	Inserted Mass, kg
10.0	4568	625	330	9.9	2036	1577
10.5	4571	625	325	9.0	1827	1794
11.0	4574	625	324	8.2	1638	1987
11.5	4567	625	314	7.5	1466	2162
12.0	4578	625	322	6.9	1314	2317
12.5	4576	625	318	6.4	1183	2450
13.0	4573	625	314	5.9	1066	2568

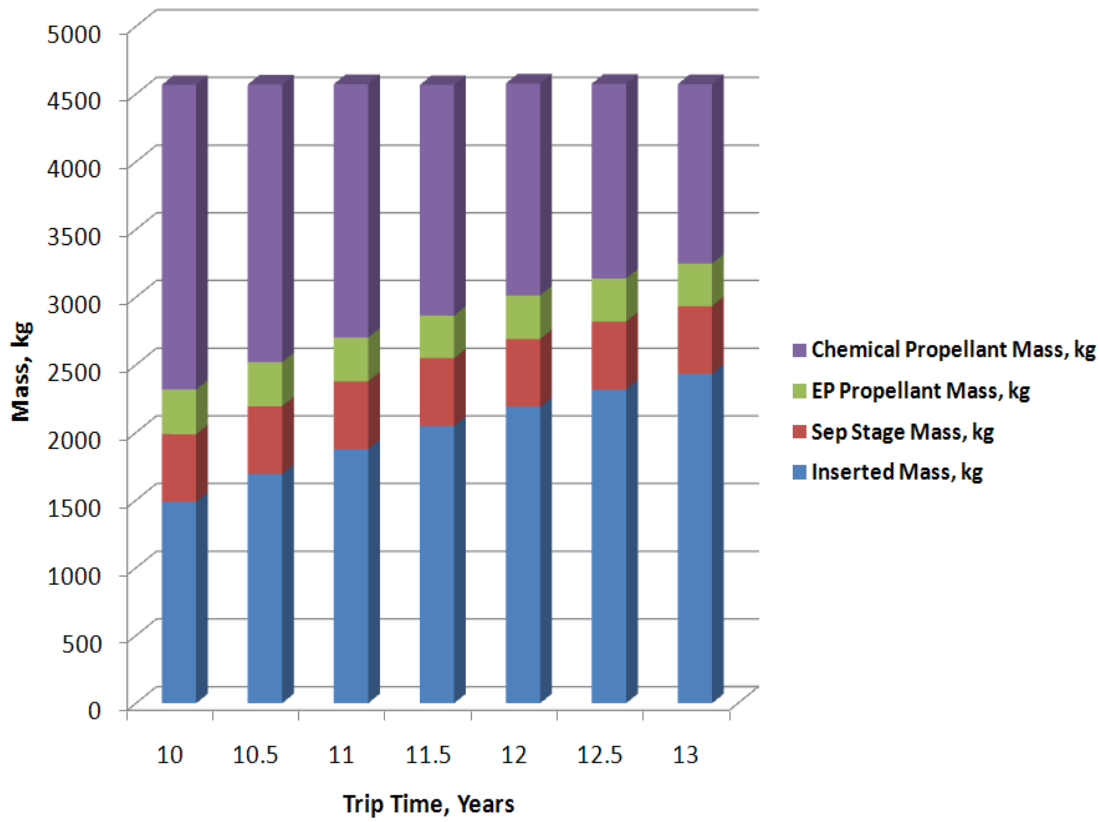

Figure 6. Mission performance versus transfer time.

Earth Flyby Distance Sensitivity

The highest performing trajectory optimizes to a very low flyby altitude for the Earth Gravity Assist (EGA). Because the Uranus mission will likely require the use of a radioisotope power, it is desired to have a high flyby altitude constraint. The mission performance as a function of flyby altitude is shown in Table 4 and Figure 7. The inserted mass is not very sensitive to the flyby attitude from 300km to 1,000km, but the performance drop off becomes steep at several thousand kilometer altitudes. For safety considerations, it was determined to use 1,000km as the minimum EGA flyby altitude.

Table 4. EGA flyby altitude effect on inserted mass.

EGA Flyby Altitude, km	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	Chemical Propellant, kg	Inserted Mass, kg
300	4568	625	330	9.9	2036	1577
400	4565	625	330	9.9	2034	1576
500	4561	625	328	9.9	2033	1575
600	4559	625	329	9.9	2031	1574
700	4556	625	329	9.9	2029	1572
800	4553	625	331	9.9	2027	1571
900	4551	625	332	9.9	2025	1569
1000	4548	625	334	9.9	2024	1565
5000	4402	625	429	9.9	1889	1460
10000	3624	625	446	10.0	1448	1105

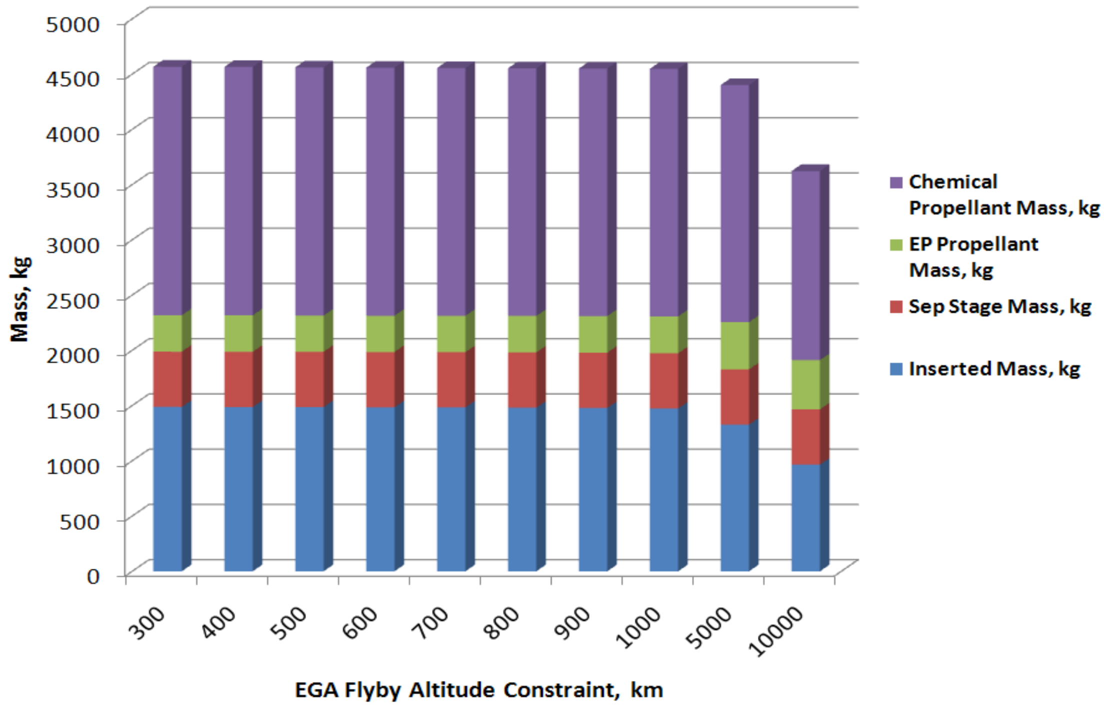

Figure 7. Mission performance versus EGA flyby altitude.

Launch Vehicle Performance

After iterating with the overall spacecraft design, the baseline mission was established to be a 12 year transfer to reduce the UOI ΔV. This new baseline is shown in Figure 8. Using this 12 year transfer time baseline, the performance capability was assessed for smaller launch vehicles.

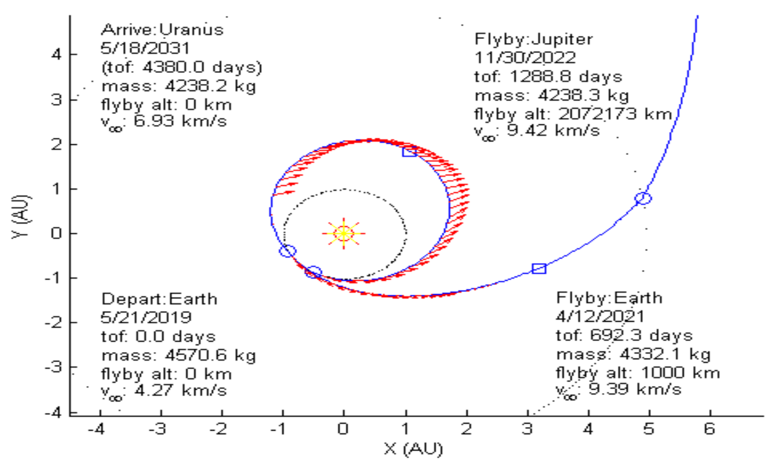

Figure 8. Performance for a 12 year transfer and a 1,000km EGA flyby altitude.

Launch vehicle step down decreases the inserted mass by approximately 200 kg per step down. The mission performance is shown in Table 5 and Figure 9.

Table 5. Performance for the Atlas V family of launch vehicles.

Launch Vehicle	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	Chemical Propellant, kg	Inserted Mass, kg
Atlas 551	4571	625	332	6.93	1310	2303
Atlas 541	4264	625	334	6.94	1201	2104
Atlas 531	3879	625	332	6.95	1064	1859
Atlas 521	3435	625	311	6.96	911	1587
Atlas 511	2921	625	281	6.96	735	1280
Altas 501	2167	625	239	6.96	475	828
Altas 401	2713	625	268	6.96	664	1156

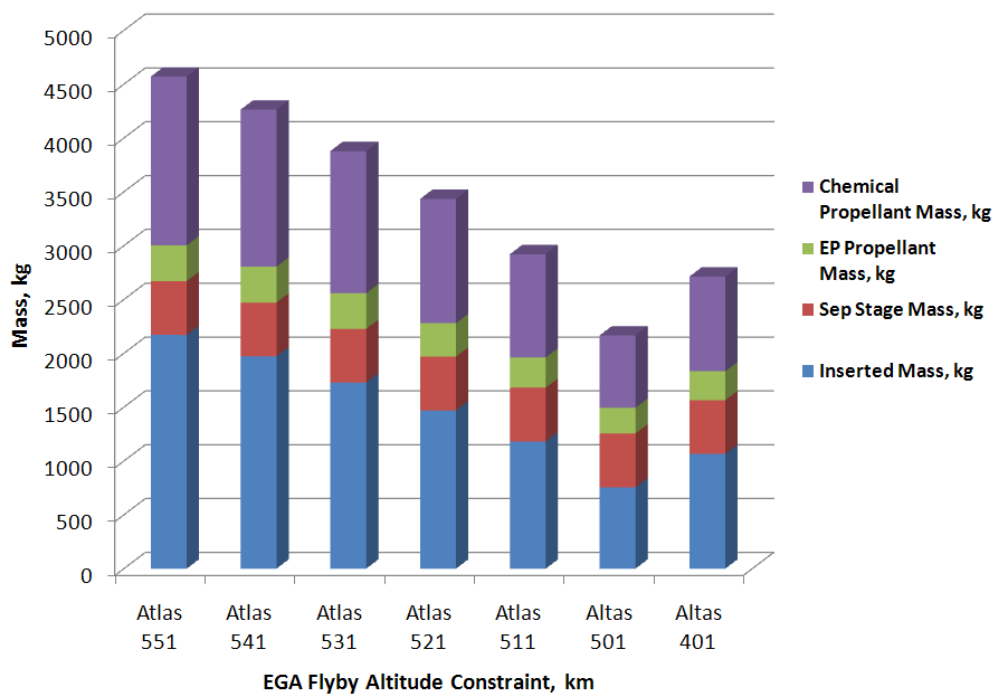

Figure 9. Mission performance versus launch vehicle.

Power Level Sensitivity and Single Thruster Performance

One concern of using electric propulsion is the cost of the SEP stage. To minimize the cost, a smaller solar array, fewer thrusters, and/or a commercial Hall thruster can be considered. The performance of lower power and only one thruster is shown in Table 6 and Figure 10. The results show that there is very little performance drop-off as the power level is decreased, only one thruster is used, or a Hall thruster is employed. The Hall thruster shown below is a 2+1 BPT-4000 system and provides the greatest decrease in C3, but also required twice the propellant.

Table 6. Performance of lower cost SEP stage options.

SEP System	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	Chemical Propellant, kg	Inserted Mass, kg
15 kW NEXT	4571	625	332	6.93	1310	2303
14 kW NEXT	4518	610	312	6.92	1301	2294
13 kW NEXT	4470	595	300	6.92	1294	2282
12 kW NEXT	4419	580	287	6.91	1283	2268
11 kW NEXT	4375	565	285	6.91	1273	2252
10 kW NEXT	4309	550	266	6.91	1262	2231
Single NEXT	4384	500	288	6.92	1301	2294
BPT-4000	4735	625	685	6.93	1242	2183

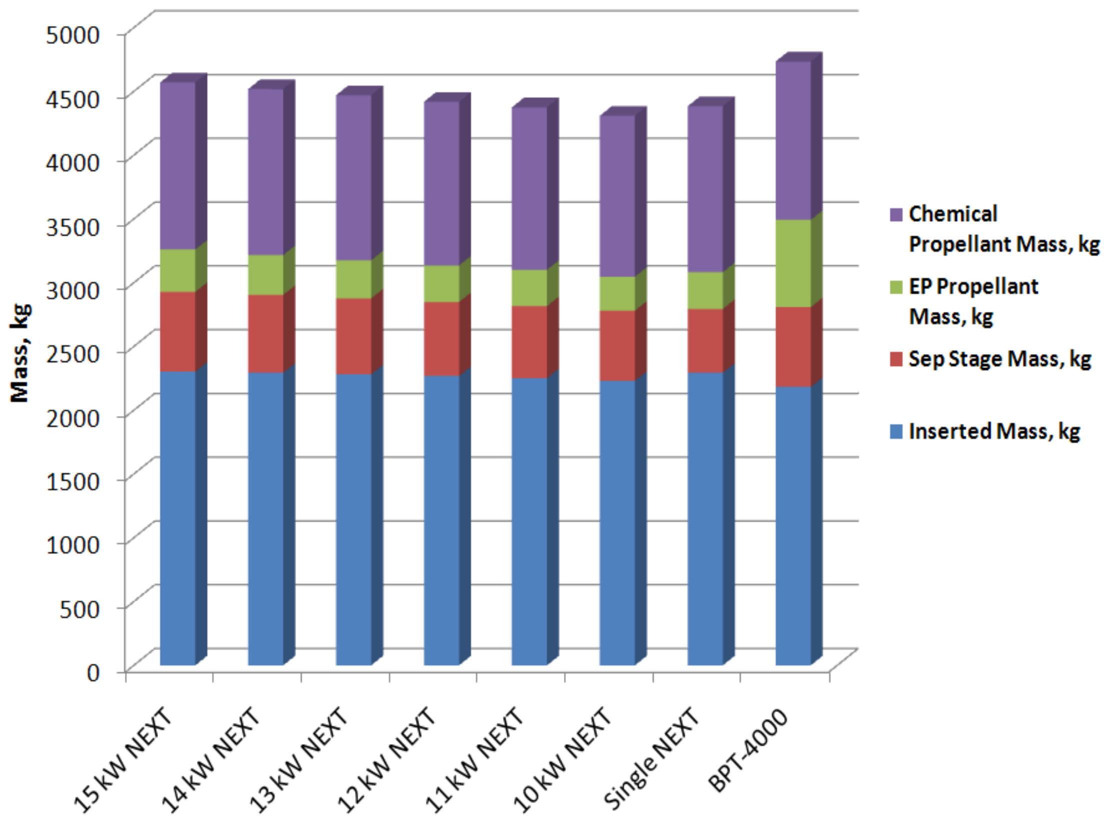

Figure 10: Mission performance of lower cost SEP stage options.

All Solar Power

Another consideration is the performance of Solar Electric Propulsion if radioisotope is unavailable for spacecraft power. As was shown in Figure 10, the performance is not very sensitive to available power. A 60 kW solar array is assumed for the all solar powered case. The solar array performance is modeling using the DARPA FAST array. The resulting trajectory is shown in Figure 11. The overall delivered mass is not greatly enhanced using the additional power. A comparison of using the FAST array and the radioisotope powered 15 kW SEP stage is shown in table 7.

Table 7. Performance of all solar powered system versus RPS powered SEP stage concept.

SEP System	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	Chemical Propellant, kg	Inserted Mass, kg
15 kW NEXT	4571	625	332	6.93	1310	2303
60 kW NEXT	4851	415	450	6.92	1443	2543*

* Inserted Mass is higher because the solar arrays (CBE ≈510 kg) not staged before the UOI

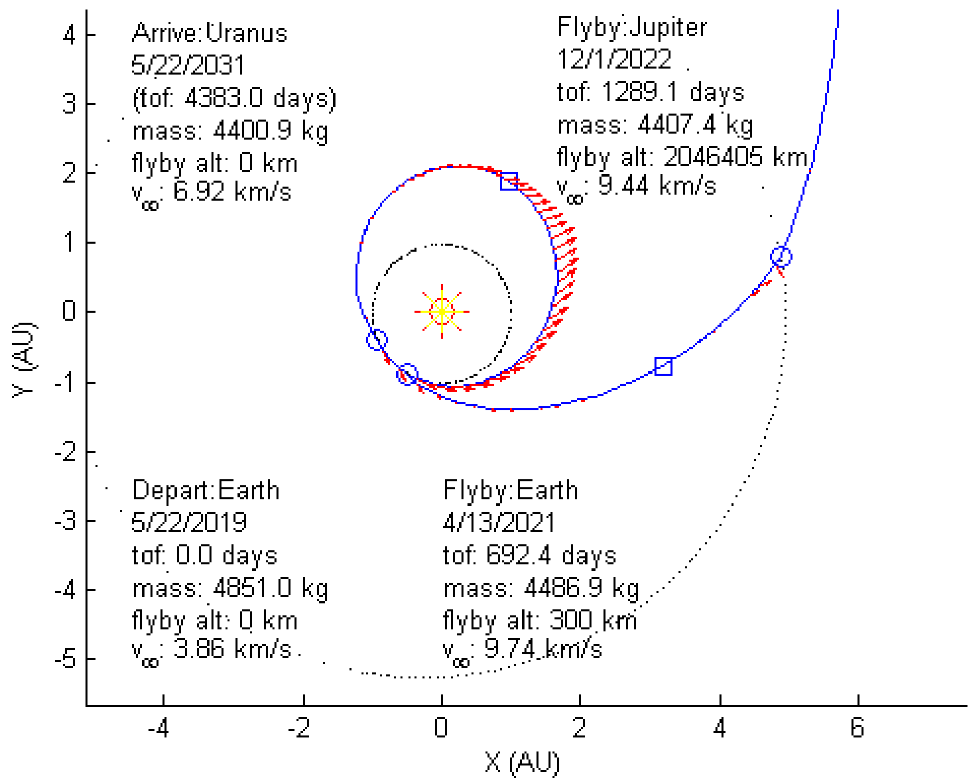

Figure 11. Mission performance with 60 kW a solar array.

SEP Options without Jupiter Gravity Assist

Low-thrust trajectories using solar electric propulsion were found to deliver significant payloads to Uranus insertion with reasonable cruise durations of 10-12 years and with a minimal EP system configuration; one or two operating thrusters and less than 15kW of solar power. Because the JGA opportunities are not available within the limits of the decadal study beyond a 2020 launch, an additional constraint eliminating the option of a JGA was imposed to make the study more broadly applicable. Unfortunately, as was shown in table 2, all of the highest performing SEP solutions within the launch years benefited significantly from a JGA. Within the 2018 – 2023 launch opportunities, two options were found for consideration; a long duration multi-gravity assist sequence or only an Earth Gravity Assist (EGA).

Long Duration Interplanetary Transfer

The solution shown in figure 12 is for an E-E-V-E-E-S gravity assist solution to Uranus. The challenge with this trajectory option is that the arrival velocity is very high. As shown in table 8, the 13 year transfer option has an arrival V_∞ approaching 12 km/s, which is prohibitive for a chemical insertion. A 15 year transfer does lower in the insertion ΔV to 2 km/s, but this is likely still prohibitive for the follow-on satellite tour. This option may be viable with aerocapture.

Table 8. Performance of a long-duration option without a JGA.

Tranfer Time, years	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	UOI ΔV, km/s	Chemical Propellant, kg	Estimated Mass after UOI, kg
15.0	6251	625	776	8.4	2.02	2257	2593
13.0	6271	625	403	12.0	3.72	3591	1651

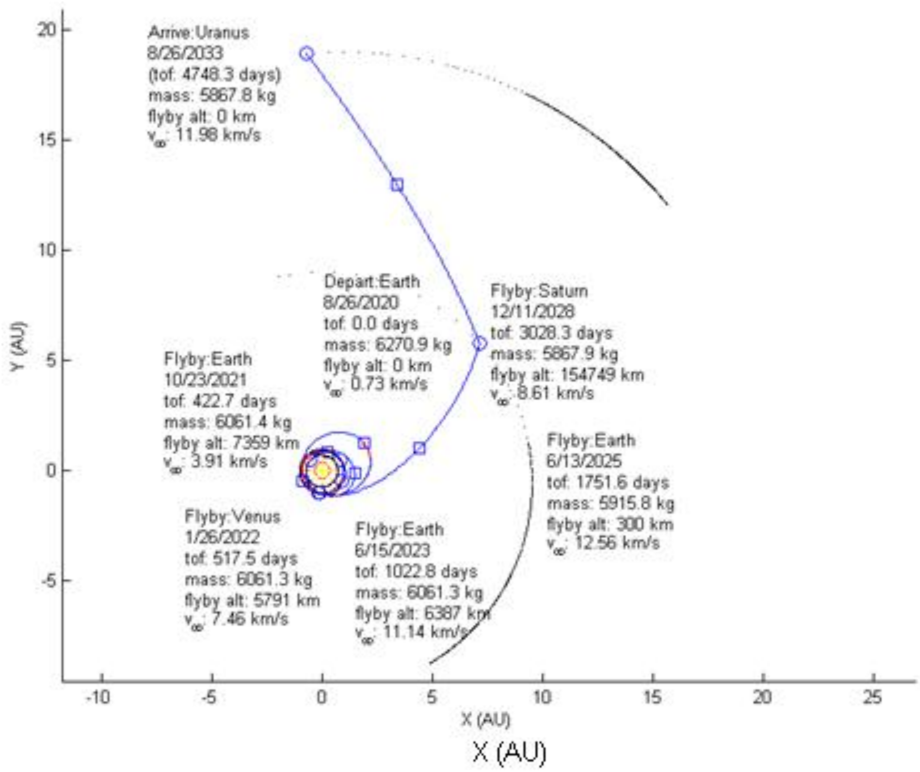

Figure 12. Long duration transfer to Uranus without a JGA.

Earth Gravity Assist

A solution to Uranus with a lower arrival velocity is to leverage only a single Earth Gravity Assist. Because of the propulsive capability of the Solar Electric Propulsion system, the launch vehicle can propel the vehicle with a launch energy of only 11.8 km^2/s^2 and then SEP system will raise the aphelion of the spacecraft orbit and after two heliocentric revolutions over 4 years, the spacecraft will perform an Earth gravity assist towards Uranus. After the EGA, the SEP system can be staged and the spacecraft will cruise another 9 years before performing the Uranus orbit insertion. The 9 year transit from Earth to Uranus allows the spacecraft to arrive with a V_∞ of 7.21 km/s. This option delivers sufficient payload to Uranus with a feasibly chemical orbit insertion maneuver. This option also allows for repeatable launch opportunities because it only requires an Earth Gravity Assist. The trajectory shown in figure 13 is this recommended solution at 20 days prior to the optimal launch date, for launch window requirements, and with a 30 day forced coast period after launch and before the EGA. This solution is based on 20 kW of solar array power, 335W of housekeeping power assumed, and two operating NEXT thrusters with a 90% duty cycle. Because of the study constraints, this option was carried for the complete spacecraft design. The estimated performance is shown in table 9, though the detailed higher fidelity results are available in the final decadal report. [1]

Table 9. Performance of an EGA option without a JGA.

System	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	UOI ΔV, km/s	Chemical Propellant, kg	Estimated Mass after UOI, kg
2 NEXT, 20 kW	4200	1000	788	7.21	1.55	966	1491

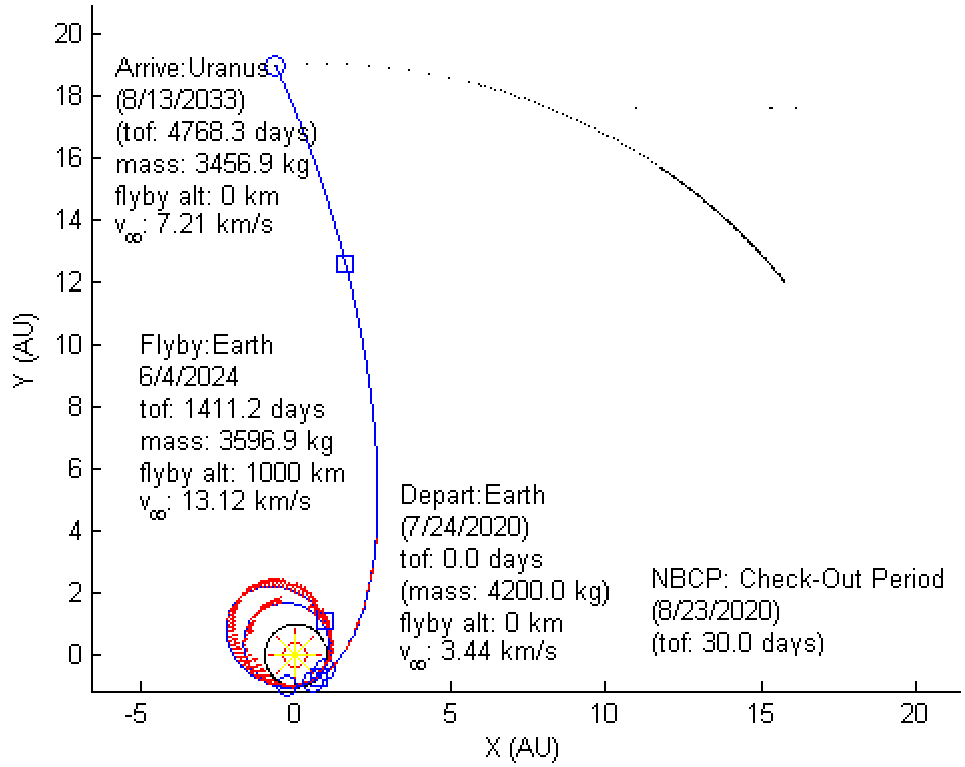

Figure 13. EGA option to Uranus with a JGA.

RADIOISOTOPE ELECTRIC PROPULSION RESULTS

Options were also evaluated assuming the use of radioisotope powered electric propulsion (REP). For the radioisotope powered electric propulsion trades, the baseline is the Atlas 551 with a Star 48 upper stage. The baseline power is 700 Watts and a baseline trip time of 13 years.

Chemical Insertion or EP Rendezvous

With the use of radioisotope power for the electric propulsion system, the EP can be leveraged to lower the arrival velocity and decrease the chemical insertion ΔV. The REP can also reduce the launch requirements for the chemical UOI, but because the power is very low compared to the SEP capability, the improved launch performance is minimal. A comparison of the performance of SEP vs. REP is shown in Table 10. An REP trajectory with chemical UOI is shown in Figure 14. The performance of the SEP solution is approximately 1,000kg greater than the any of the REP options. The highest performance REP solution leveraged the REP to decrease the arrival velocity but not rendezvous. Figure 15 compares various REP options with the SEP solution. The EP rendezvous is not recommended for this mission because the chemical propulsion system can leverage the large gravity well for an efficient UOI.

Table 10. REP JGA performance with comparison to SEP and direct options.

Propulsion System	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	Chemical Propellant, kg	Inserted Mass, kg
SEP	4573	625	314	5.9	1066	2568
REP Free V_∞	2200	0	154	5.1	493	1554
REP 4 km/s	2095	0	161	4.0	343	1590
REP 3 km/s	2091	0	274	3.0	234	1582
REP 2 km/s	1921	0	274	2.0	152	1495
REP 1 km/s	1730	0	274	1.0	101	1355
REP 0.1 km/s	1563	0	274	0.1	80	1209
REP w/o JGA	1299	0	384	0.1	56	858

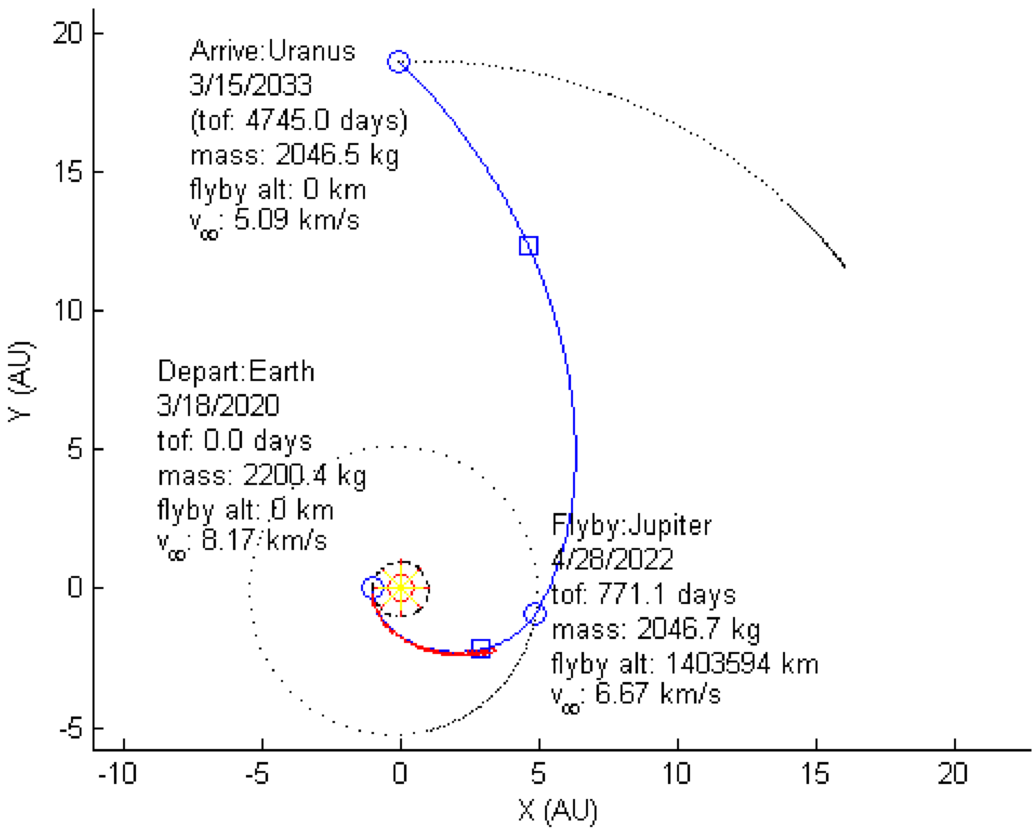

Figure 14. REP trajectory with chemical UOI.

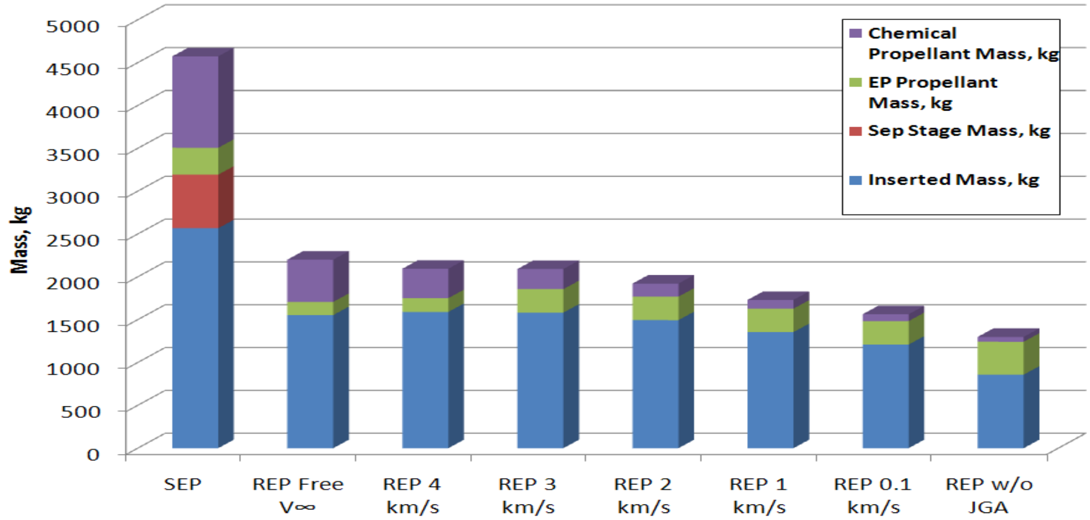

Figure 15. Performance comparison of various REP options and SEP comparable solution.

SUMMARY

Based on the trades conducted, the recommended starting point for a higher fidelity mission and spacecraft point design is a 1+1 NEXT propulsion system with a 15 kW solar array assuming the JGA is allowed. The recommended trajectory is shown in Figure 16, and the trajectory characteristics are shown in Figures 18 – 20. Without a JGA option, the recommended starting point for a higher fidelity mission and spacecraft point design is a 2+1 NEXT propulsion system with a 20 kW solar array. The recommended trajectory is shown in figure 17, and the characteristics are shown in figures 21-23. Because of the increase in mission risk, new technology requirements, and decreased performance, no REP solutions are recommended for a Uranus orbiter mission.

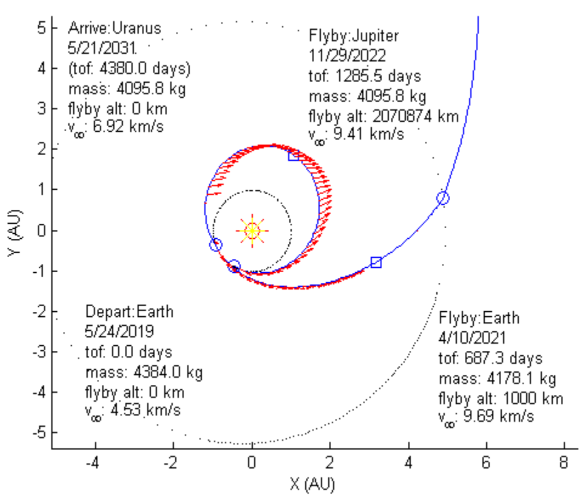

Figure 16. Recommended EEJU solution with a JGA.

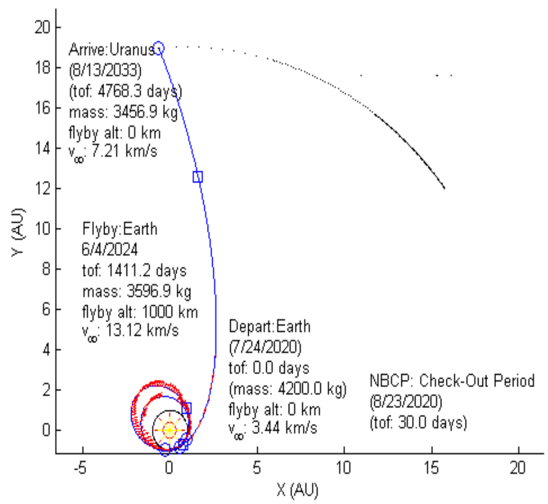

Figure 17. Recommended solution without a JGA.

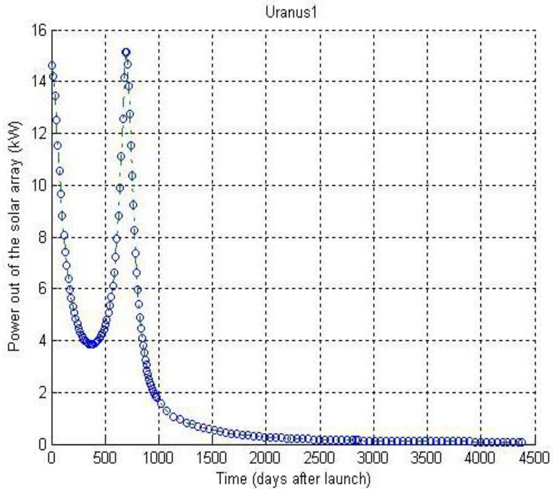

Figure 18. Solar array power over EEJU mission time.

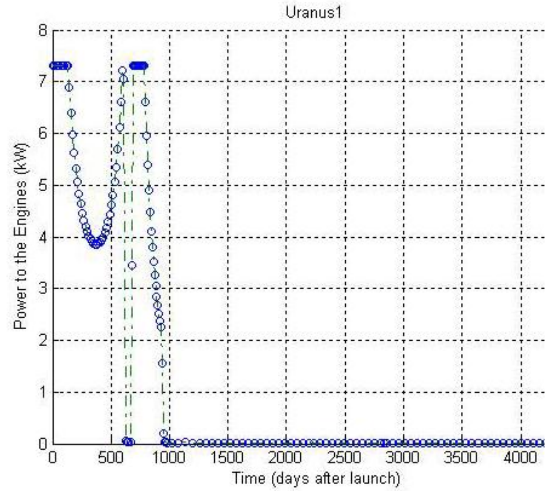

Figure 19. PPU input power over EEJU mission time.

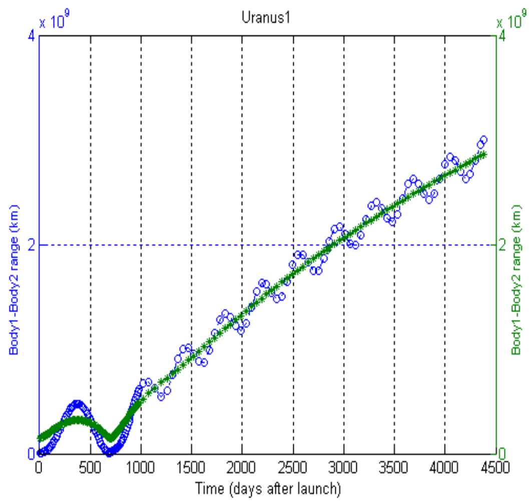

Figure 20. Distance from the Earth and Sun from the spacecraft over the EEJU mission.

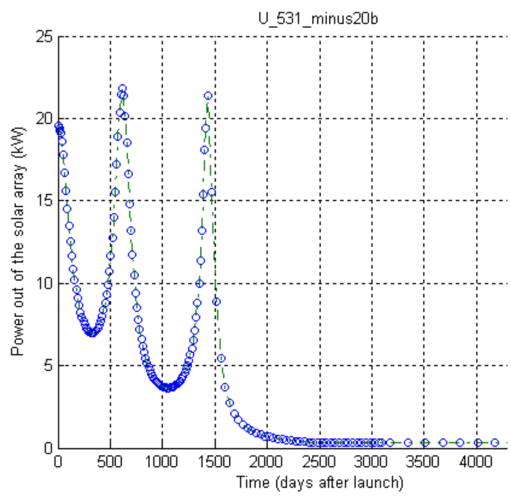

Figure 21. Solar array power over EEU mission time.

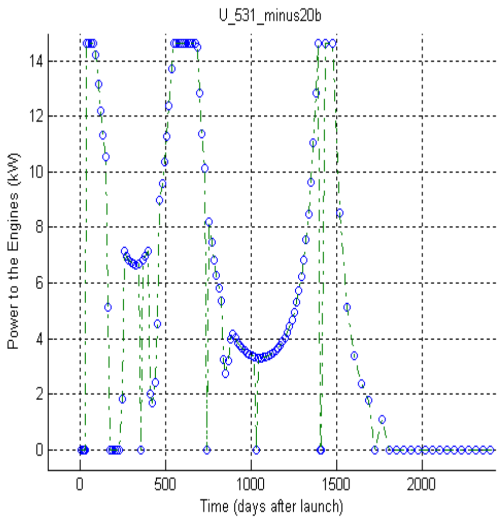

Figure 22. PPU input power over EEU mission time.

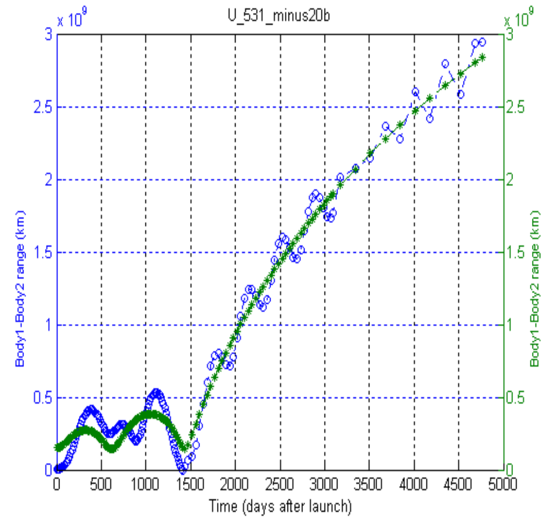

Figure 23. Distance from the Earth and Sun from the spacecraft over the EEU mission

ACKNOWLEDGMENTS

This work was funded by NASA supporting the Ice Giant Orbiter / Probe Decadal Study led by The Johns Hopkins University Applied Physics Laboratory (JHU/APL), with significant input from Yanping Guo, Steve Oleson, Rob Gold, Doug Eng, and Helmut Seifert.

APPENDIX

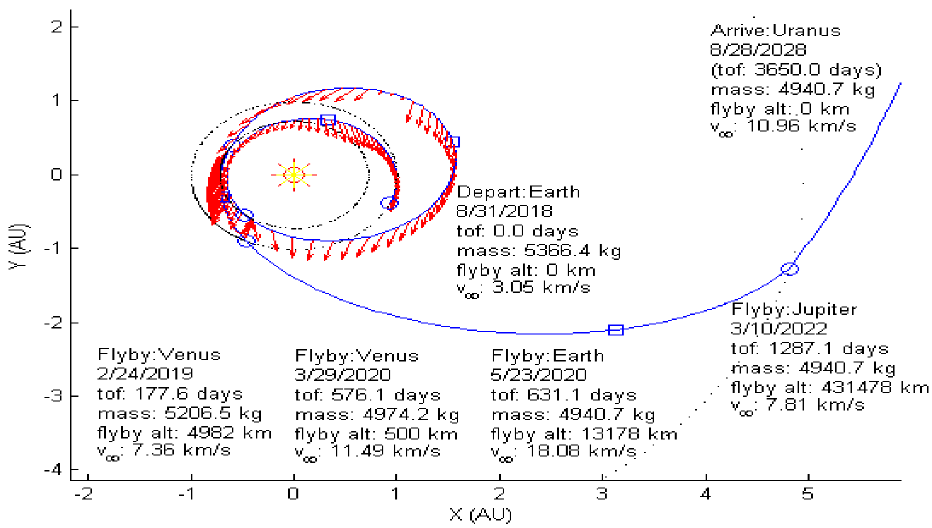

Figure A-1 Example EVVEJU 10 year trajectory.

.

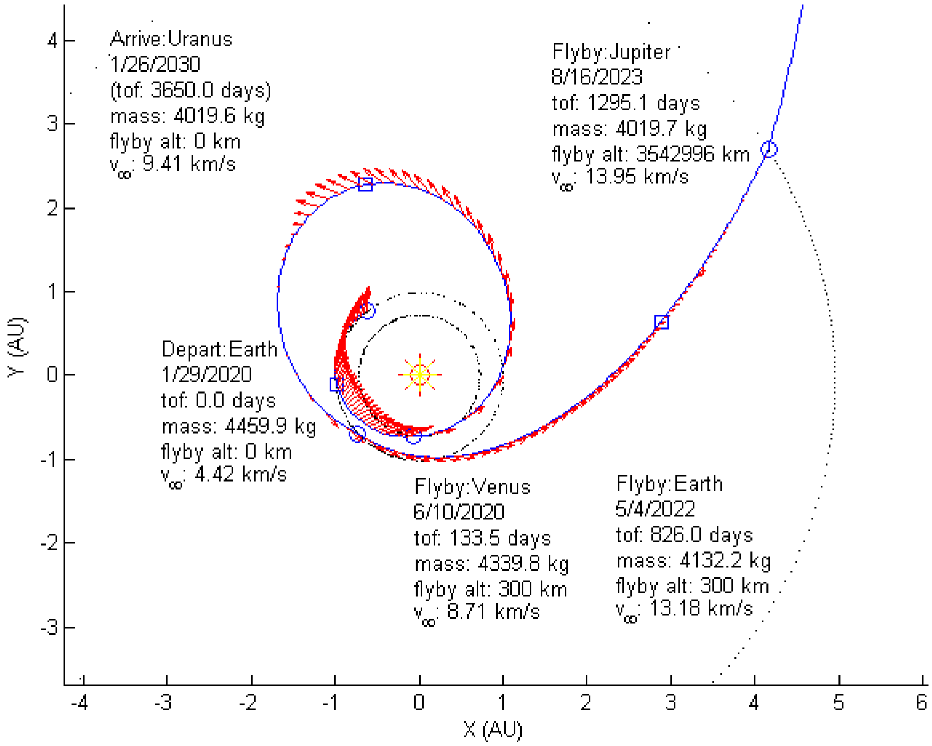

Figure A-2. Example EVEJU 10 year trajectory.

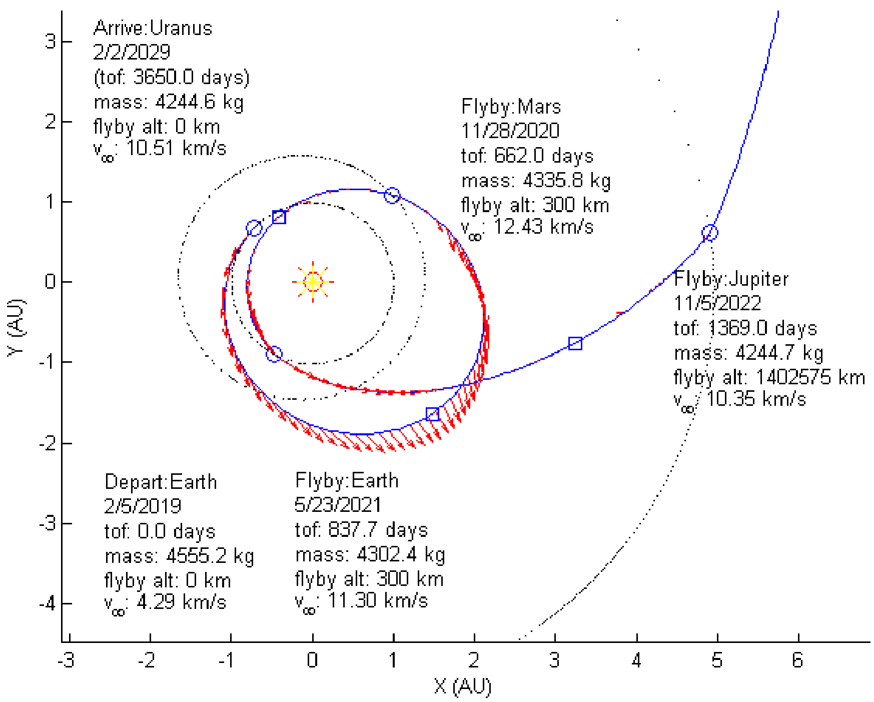

Figure A-3. Example EMEJU 10 year trajectory

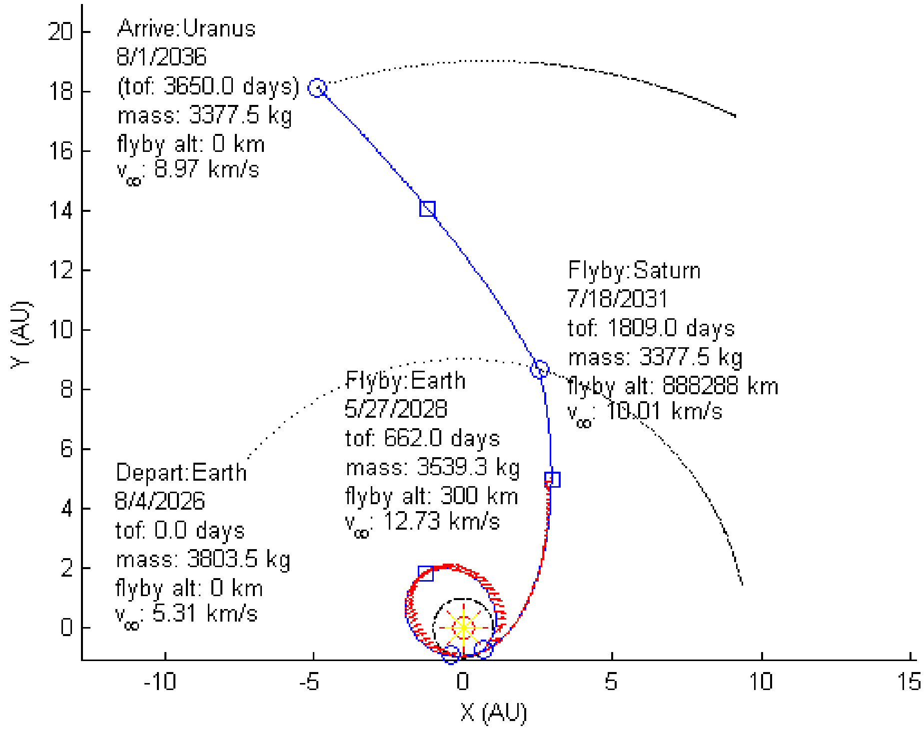

Figure A-4. Example EESU 10 year trajectory

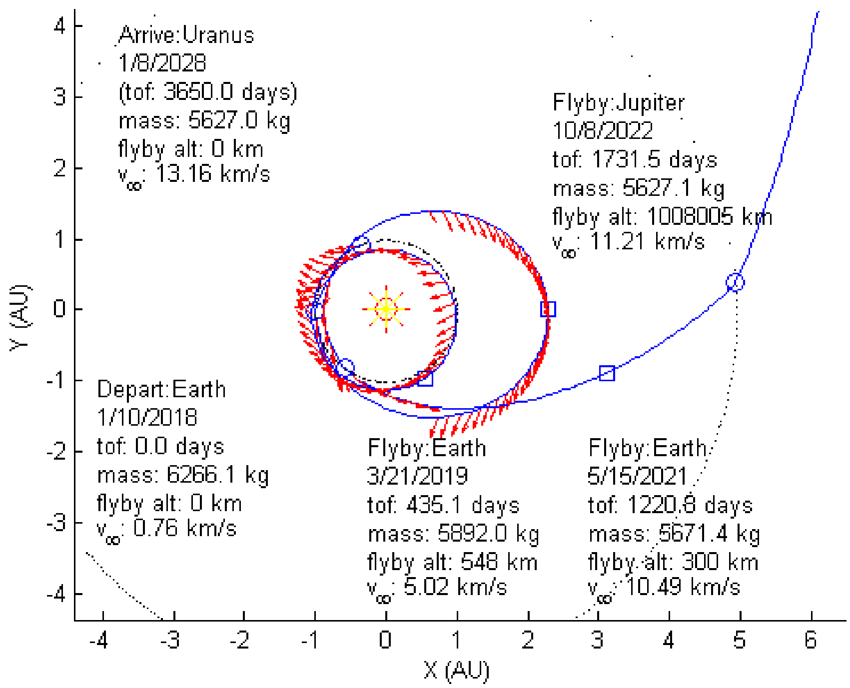

Figure A-5. Example EEEJU 10 year trajectory.

REFERENCES

[1] Hubbard, W. B., et al., "Ice Giants Decadal Study," SDO-12345, Planetary Science Decadal Survey Mission Concept Study Final Report, May, 2010.

[2] Vane, G., "Concept Maturity Level: Establishing s Shared Language to Articulate Maturity of a Space-Science Mission Concept and Cost Uncertainty in the Early Formulation Phase," Presented to the Planetary Science Decadal Survey Steering Group, Washington, D.C., July 6, 2009.

[3] McAdams, J., Scott, C., Guo, Y., Dankanich, J., and Russell, R., "Conceptual Mission Design of a Polar Uranus Orbiter and Satellite Tour," AAS 11-188, 21st AAS/AIAA Space Flight Mechanics Meeting, New Orleans, LA, February, 13-17, 2011.

[4] Sims, J. A., Finlayson, P. A., Rinderle, E. A., Vavrina, M. A., and Kawalkowski, T. D., "Implementation of a Low-Thrust Trajectory Optimization Algorithm for Preliminary Design," AIAA 2006-6746, AIAA/AAS Astrodynamics Specialist Conference, Keystone, CO, August 21-24, 2006.

[5] Mankins, J. C., "Technology Readiness Levels: A White Paper," NASA, Ofice of Space Access and Technology, Advanced Concepts Office, April 6, 1995.

[6] Herman, D., "Status of the NASA's Evolutionary Xenon Thruster (NEXT) Long-Duration Test after 30,352 Hours of Operation," AIAA 2010-7112, 46th Joint Propulsion Conference, Nashville, TN, July 25-28, 2010.

[7] Landau, D., Lam, T., and Strange, N., "Broad Search and Optimization of Solar Electric Propulsion Trajectories to Uranus and Neptune," AAS 09-428, AAS/AIAA Astrodynamics Specialist Conference, Pittsburgh, PA, August 9-13, 2009.

[8] Titan Saturn System Mission Final Report (on the NASA Contribution to a Joint Mission with ESA)," January 30, 2009.