Uranus Orbiter and Probe: Mission Challenges and Concept Updates Since the Origins, Worlds, and Life Decadal Survey

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Abstract

Origins, Worlds, and Life: Planetary Science and Astrobiology in the Next Decade identified a Uranus Orbiter and Probe as the highest-priority strategic mission for the decade 2023-2032, as it enables broad cross-disciplinary science in the largely unexplored Uranian system. The mission architecture evaluated by the Decadal Survey was a singular proof of concept demonstrating that a moderately instrumented mission could deliver Decadal-priority science with a reduced cost and risk posture by leveraging existing technologies to the maximum extent possible. With revised assumptions since the Decadal, we have explored a large trade space including launch vehicles, propulsion options, cruise trajectories, available power sources, viable concept of operations, and science data return for later launch dates without a Jupiter Gravity Assist. The most repeatable trajectory solutions employ either a commercially derived solar electric propulsion (SEP) transfer stage or the availability of a more capable launch vehicle under development, such as the SpaceX Starship. Orbit insertion has been moved further from Uranus to acknowledge the remaining uncertainty in Uranian ring structure. A streamlined, SEP-adaptable, orbiter design was developed using 2 Next Gen RTGs, and the probe design was matured, reducing the entry gravitational acceleration, and assuming the largest Decadal-recommended payload to provide margin for future instrument selections. With this updated design, we also constructed a detailed concept of operations for three representative science cases, returning 13-15 Gbits of science data and spacecraft telemetry per ~34-day orbit.

1. Introduction:

Origins, Worlds, and Life: Planetary Science and Astrobiology in the Next Decade (*OWL*; NASEM, 2023) endorsed a Uranus Orbiter and Probe (UOP) as the highest priority new strategic (or Flagship) mission for the decade 2023-2032. This mission would provide in-depth study of an Ice Giant planetary system, a class of giant planets smaller and colder than Jupiter or Saturn, but nonetheless pivotal for understanding solar system formation, planetary migration, the evolution of ocean worlds, and even exoplanets. The Decadal study presented a mission concept capable of meeting broad cross-disciplinary science goals and the desired measurements to meet them with a suite of capable instruments and an atmospheric entry probe (A. Simon et al. 2021).

However, missions to the outer solar system are challenging to implement; vast distances and limited resources (e.g., mass, power, data rate) require that any mission design be carefully optimized to meet the highest priority science goals. The UOP mission architecture presented to the Decadal Survey was a proof of concept that a moderately instrumented mission could deliver Decadal-priority science with a reduced cost and risk posture by leveraging existing technologies to the maximum extent possible (A. Simon et al. 2021). To properly scope the mission, the Decadal Survey mission concept team made several key assumptions:

- Launch toward the end of the decadal period (2031-2032);
- Launch performance of a Falcon Heavy Expendable (FHE), coupled with a Jupiter gravity assist (JGA), allowing flight times to Uranus of ~13 years;
- Availability of three Mod 1 Next Generation Radioisotope Thermoelectric Generators (Next Gen RTGs), each providing up to 250 W_e at fueling (C. Sandifer 2024);
- Uranus orbit insertion performed inside of the rings, to optimize propellant usage;
- Release of the atmospheric probe after orbit insertion; and
- Use of Ka-band to maximize science downlink data rates and return volume.

However, since the Decadal Survey was released, updated information that challenges some of these initial assumptions has become available. First, it was determined that a 2031-2032 launch date would be precluded by current funding profiles, and thus, the Jupiter gravity assist would not be available; alternative launch periods without Jupiter were presented in the Decadal Survey study but not further optimized, as each incurred longer flight times and/or reduced deliverable mass. Second, the Mod 1 Next Gen RTG production outlook (T. Tofil et al 2023, C. Sandifer 2024), with the first unit expected to be manufactured in 2030, does not allow for the delivery of three fueled units in time for a UOP launch in the early to mid 2030s. Lastly, 2023 James Webb Space Telescope (JWST) images and subsequent analysis of the Uranus ring system indicate a highly variable, and semi-continuous ring/dust system, which has not mitigated the likelihood that a low-altitude orbit insertion could be unsafe (I. de Pater et al. 2013; M. Hedman et al. 2025). The combined effects of this updated information on interplanetary trajectory, orbit insertion, power optimization, and orbital operations must be considered in tandem with the UOP Decadal mission objectives to determine if a successful mission is still possible.

This paper presents updated analyses of the UOP mission concept design that have been completed since the Decadal Survey. Section 2 summarizes the challenges in designing and realizing any mission to Uranus. Section 3 presents the available trajectory trade space. Section 4 discusses the architecture trade space for these challenges, based on current technology availability and updated performance information. Section 5 details concepts of operations that fit within new power and data rate assumptions. The paper concludes with next steps and future studies that should be undertaken.

2. Uranus Orbiter Mission Challenges

The overarching challenge in delivering a spacecraft into Uranus orbit is doing so within an acceptable timeframe, while retaining sufficient post-capture capability for accomplishing mission objectives. This challenge is heightened by Uranus's solar distance (up to 20 AU) and the unique characteristics of the Uranian system, including its obliquity (97.77°), potential ring and dust hazards, and lower planetary and satellite masses relative to the gas giant systems where previous analogous missions have been conducted. Thus, numerous constraints and multiple competing or conflicting objectives must be reconciled to develop an optimal system solution that minimizes complexity and maximizes overall science return. For the initial mission phases from launch through Uranus capture, key mission design priorities and objectives for maximizing mission value include:

- Minimizing time of flight to Uranus;
- Limiting relative arrival velocity and required insertion propellant burn, Δv;
- Delivering sufficient mass into orbit;
- Minimizing exposure to potentially hazardous environments;
- Achieving a suitable orbit to satisfy probe delivery and system tour requirements; and
- Maximizing programmatic flexibility with highly repeatable launch opportunities.

2.1 Reaching the Uranus system

The most propellant-efficient direct-to-Uranus trajectory (with a realistic operational duration) would be accomplished via a Hohmann transfer, where the spacecraft approach vector is tangential to the planet's velocity (i.e., at low angle), so the Δv required for orbit insertion is minimized. Orbital transfers to Jupiter, for example, are typically designed so that at least the final leg is roughly Hohmann (2.7 years), with total cruise durations on the order of 5-6 years (D. Ellison et al. 2025). Examples of this trajectory characteristic are the interplanetary cruises flown by Galileo (L. D'Amario et al. 1992) and Juno (T. Kowalkowski et al. 2008) and planned for Europa Clipper (L. Cangahuala et al. 2025). For Uranus, however, a direct Hohmann transfer would require a ~16 to 17-year flight time, leading to total mission durations of more than twenty years.

In general, trajectories with reduced flight times also involve higher-energy approaches, with spacecraft velocity vectors at larger angles relative to the planet's velocity vector, that correspondingly increase the Δv required for capture, see Figure 1. Thus, flight time and the Uranus orbital insertion burn (UOI) magnitude must be carefully traded within acceptable resource constraints. In the 2021 Decadal study, assuming the use of a Falcon Heavy Expendable (FHE) launch vehicle and launch readiness dates in 2031-2032, planetary alignments allowed for a Jupiter gravity assist (JGA) to accelerate the spacecraft toward Uranus and reduce the transit time to a more acceptable 13 years, while retaining the capability to deliver up to a 4,000 kg (wet mass, post orbit insertion) spacecraft. However, Jupiter will not be in a suitable alignment for launches after

2033 and through 2044. We have investigated alternative mission architectures to mitigate the loss of the JGA, including: 1) inner solar system gravity assists, 2) a more efficient continuous thrust system, such as solar or nuclear electric propulsion, and/or 3) a more capable launch vehicle. These are each discussed in Section 3.

2.2 System arrival and orbit insertion

Upon arrival, capturing into Uranus orbit presents additional challenges. First, Uranus's comparatively low mass (least of the giant planets), combined with the higher arrival velocities associated with non-Hohmann final leg trajectories, necessitate Δv magnitudes that are generally larger than at other planets (D. Ellison et al. 2025). The large angles between the respective spacecraft and planet velocity vectors on arrival also place the periapse of the capture orbit, and at least a portion of the capture burn arc, on the night side of the planet, and out of the view of Earth, see Figure 1. Figure 2 depicts a Uranus orbit insertion (UOI) case that occurs in full view of Earth; however, it is not uncommon for a portion of the UOI burn to be occulted from the perspective of the Deep Space Network (A. Simon et al. 2021).

Another factor complicating spacecraft arrival and capture is the presence of the Uranian ring system, which extends from a radius of ~98,000 km to 38,000 km (R. French et al. 1986; I de Pater et al. 2013). In the 2021 Decadal Study, it was assumed that the insertion burn could be performed inside of Uranus's rings to minimize propellant requirements. Although the ring environment is not well constrained, it was assumed that sufficient low-altitude ring gaps could be identified from Voyager, ground-based observations, and JWST/Hubble data sets. More recent JWST data show a complex and dusty environment throughout ring system, though scattered light from Uranus does not allow for clean characterization of the environment down to the top of the atmosphere. Several studies also note that the ring structure appears to change over time (e.g., I. de Pater et al. 2006, 2013; M. Hedman et al. 2025). Unless better constrained in the future, orbit insertion should be moved to a higher altitude ring gap, completely outside of the rings, or away from the ring plane to reduce mission risk. Note that each of these alternative insertion location options incurs different propellant penalties as a function of altitude. This is exacerbated by the fact that the velocity boost from Uranus's gravity (Oberth effect) is much less than at Jupiter or Saturn, given Uranus's significantly lower mass. Finally, the high obliquity of the Uranian system also affects spacecraft arrival and capture design, as the ring system orientation varies considerably relative to the incoming spacecraft trajectory over the course of a Uranian year. For arrival dates near the upcoming equinox in early 2050, the spacecraft would be approaching Uranus roughly perpendicular to the ring/moon plane.

2.3 Uranus system tour and resource limitations

Once in the system, spacecraft operations are dictated by science requirements, as well as by power management and data volume return capabilities. After orbit insertion, the 2021 Decadal orbiter begins on a highly inclined orbit relative to the planet and delivers an atmospheric probe during the first orbit (after first apoapse); placing probe release after orbit insertion reduces the risk from competing critical probe relay and orbit insertion burn activities (A. Simon et al. 2021). The probe operations are conducted as early as possible due to its relative high science priority and to take advantage of the orbiter's lower altitude post-capture periapse which must be raised prior to beginning the moon tour; delays impose a mass/propellant penalty. After the probe data relay is complete, the prime system science tour begins.

To meet science requirements, the 2021 Decadal study assumed a robust suite of orbiter instruments and a full tour of the Uranus system with flybys of the five largest moons. Repeated flybys of Titania were used to lower the orbital inclination and enable an in-plane satellite tour; other satellites can be used for this purpose, though Titania, as the largest moon, is the most efficient (A. Simon et al. 2021). Near Uranus's equinox, typical arrival geometries allow for accelerated equatorialization due to the low arrival asymptote declination relative to Uranus's equator during that time. This low declination results in capture orbits that have a low periapse latitude and which therefore require fewer satellite flybys to adjust the inclination for an equatorial system tour (D. Ellison et al. 2025). As Uranus moves away from equinox, these lower declination captures become increasingly difficult, and polar tours become a more favorable option when considering overall flight time.

In the 2021 Decadal concept, three Mod 1 Next Gen RTG units provided ~708 W_e of power at launch, ~553 W_e at Uranus arrival, and ~530 W_e at end of mission (EOM), with a supplemental battery used to handle peak power draw during the most energy-intensive events, such as orbit insertion. Even at this reasonable power level, instruments could not be powered on continuously during the tour science phase, with periodic interruptions required when the spacecraft was in downlink or propulsive maneuver mode, as discussed later in Section 4.1.

Maximizing science data return capability from 20 AU, within available power and feasible pointing constraints, is a considerable challenge and was a significant trade in the 2021 Decadal study. It was determined that Ka-band is necessary to enable sufficient data return to meet UOP Decadal science objectives and to allow the return of all data collected within a ~34-day orbit. The best overall design solution, pairing a 3.1-m high gain antenna (HGA) with a 58 W_{RF} (100 W_e) traveling wave tube amplifier (TWTA), would achieve a maximum downlink rate of 19.5 kbps using a single 34-m antenna through the Deep Space Network (DSN).

Although the use of Ka-band (26-40 GHz) does require 3-axis fine pointing control of the spacecraft using reaction wheels, X-band (8-12 GHz) solutions were found to be insufficient at Uranus (beyond supporting navigation related measurements and activities), providing only ~6-7 kbps at equivalent transmit power. Note that the best achievable Ka-band rate of 19.5 kbps from Uranus is roughly comparable to the average 22.5 kbps X-band rate realized by Cassini (4-m HGA, 20-W_{RF} TWTA) when using a single, 34-m High Efficiency DSN antenna (S. Asmar et al. 2018). The main differentiator is that Cassini operated at roughly half the Earth range (10 AU) as Uranus, and signal strength diminishes proportional to the inverse square of the distance. Higher data rates on either Ka- or X-band could be achieved at Uranus by arraying multiple DSN 34-m stations, subject to availability and increased operational cost.

3. Updated Trajectory Trade Space

In this study, we have updated the trajectory work from the 2021 Decadal concept to address the following areas:

- investigate later launch dates with a variety of launch vehicles and propulsion types;
- study the feasibility of aerocapture for orbit insertion;
- improve the probe delivery conditions; and
- evaluate different system tour options based on new arrival dates.

3.1. Launch through Uranus capture

Once it became clear that the NASA budget profiles would delay a UOP project start beyond what was assumed in the Decadal Survey, a study was initiated to search for opportunities with launch readiness dates (LRDs) in the 2035-2040 timeframe, reflecting a project start of no earlier than 2027. Jupiter and Saturn remain unavailable for gravity assists during this window but achieving Uranus arrival before or close to the upcoming 2050 equinox – a clear desire among the science community, but not a firm requirement – could still be possible. The revised launch date assumptions also allow for the consideration of launch vehicles currently in development, and a re-examination of state-of-the art solar electric propulsion (SEP) technologies, that could be used to offset the loss of JGAs. Potential options as reevaluated for 2035 to 2040 LRDs are presented in Table 1 and discussed below.

Table 1. Summary of cruise trajectory options investigated for 2035-2040 launch dates.

	Jupiter Assist	Falcon Heavy Direct	Inner Solar System Assist	Solar Electric Propulsion	Starship Direct
Launch Vehicle	Falcon Heavy	Falcon Heavy + upper stage	Falcon Heavy	Falcon Heavy	Starship
Example Trajectory[a]	E-(ΔV)EJ-U	E-U	E-VEE-U	E-E-U or E-EE-U	E-U
Minimum Perihelion	1 AU	~1AU	≥0.5 AU	≥0.8 AU	~1 AU
Cruise	13 years	10-13 years	13-14 years	12-14 years	~10 years
Arrival velocity (v_∞)	6.3 km/s	5 km/s	6-8 km/s	6-8 km/s	6 km/s
Deliverable mass to Uranus orbit	4000 kg	~700 kg (STAR-48b, 13.5 years) ~1275 kg (STAR-75, 10 years) ~1375 kg (STAR-75, 13.5 years)	3000-4000 kg	3000 kg (12 years) 3700 kg (13 years) 4200 kg (14 years)	>> 4000 kg
Repeatability	Only within narrow windows	Repeatable any year	Max mass not available yearly	Repeatable any year	Repeatable any year
Primary Implementation Challenges	Unavailable 2033-2044	~700 kg dry mass to orbit, requires large upper stage	Challenging thermal environments	Increased complexity relative to chemical propulsion	Requires on-orbit refueling capability

Notes: a. E = Earth, V = Venus, J = Jupiter, U = Uranus, ΔV = large engine burn

The 2021 Decadal study established that a super heavy lift launch vehicle would be required to deliver a spacecraft in excess of 4,000 kg (wet mass post-UOI) to Uranus within an acceptable cruise duration of ~13 years (Appendix C launch curves in A. Simon et al. 2021). Of the vehicles anticipated to be available for 2031-2032 launches, the Falcon Heavy Expendable (FHE) was baselined, as the Space Launch System (SLS) was projected to incur significantly higher launch costs and induce far more challenging launch environments. However, when considering an updated assumption of a launch no earlier than 2035, the trade space can be widened to include additional vehicles nearing maturity. Of those, the SpaceX Starship is projected to provide suitable performance and has been added to the trade space in Table 1. Other options, such as New Glenn, are not predicted to have sufficient performance for a Uranus mission, while the availability of SLS for an interplanetary science mission is uncertain at this time (T. Haws et al. 2024, R. McNutt et al. 2024).

Without Jupiter, direct (no gravity assist) Uranus trajectories were considered using FHE or Starship. As indicated in Table 1, a direct FHE trajectory is infeasible for UOP. Even the addition of a relatively large solid rocket upper stage, such as a STAR-75, can inject only ~1275 kg into the interplanetary transfer orbit, a substantial proportion of which would then be needed for capture at Uranus. We note that to compare between various trajectory studies, one must carefully assess what is included in “deliverable mass.” D. Landau et al. (2025) lay out the definition of a “core mass,” which includes the payloads, probe, and spacecraft subsystems minus the propulsion system, and show that the additional mass of the structure and tanks will scale with the propellant load needed for orbit insertion and for completing the system tour. The deliverable core mass is referenced in some studies, while we include the full propulsion system and propellant in our deliverable mass calculations. In addition, best practices dictate that NASA Flagship-class mission trajectories should have a minimum 21-day launch period and a backup launch opportunity within approximately one year of the primary opportunity; the propulsion system must be sized for the worst-case performance across both the prime and back-up periods.

Studies of FHE+upper stage trajectories can find paths to Uranus if they include Earth or Venus gravity assists. However, these trajectories either: 1) have unacceptably long flight times (>14 years), 2) do not repeat yearly with sufficient deliverable mass, or 3) require a staged bipropellant (biprop) propulsion system (first stage ejected after orbit insertion) and a much-reduced core mass to close (A. Simon et al. 2021, D. Landau et al. 2025, R. Karimi et al. 2025). Venus-enabled trajectories also require managing challenging thermal accommodations with minimum perihelia of ~0.6 AU or lower (D. Landau et al. 2025); lower perihelia enable larger deliverable mass. In these cases, the spacecraft thermal design must accommodate a wide range of conditions throughout the mission lifetime with adequate margins, from the high heat loads close to the Sun to the cryogenic temperatures experienced at 20 AU. The OSIRIS-REx spacecraft has demonstrated that, while survivable, subsystems designed for 0.8 AU operations will suffer some degradation at 0.5 AU, such as paint blistering (D. DellaGiustina et al. 2023) and increased radiation effects. For comparison, both Cassini and Galileo had minimum perihelion distances of about 0.7 AU, though many of their spacecraft components are now obsolete. Designing an outer solar system spacecraft with modern components and instruments for 0.6 AU operations would levy significant qualification and testing requirements not yet fully considered, and we did not pursue these options further.

Another alternative that could make up for the loss of the JGA involves revisiting the potential use of solar electric propulsion (SEP) during interplanetary cruise. A SEP stage was baselined in the prior UOP Decadal study (W. Hubbard 2010), as Jupiter was not available during its 2020–2023 targeted launch window. A SEP stage concept was also included in three of the six mission architecture options considered in the 2017 NASA Ice Giants Pre-Decadal Study (M. Hofstadter et al. 2017), and in a 2019 European Space Agency (ESA) Ice Giants Study (ESA 2019). A common thread through these studies is the use of a separable SEP stage as opposed to integrating a SEP system into the Uranus orbiter spacecraft. While a SEP stage adds an additional complexity, we note that the BepiColombo mission uses a SEP transfer module to deliver two stand-alone orbiters to Mercury (J. Benkhoff et al. 2021). The BepiColombo SEP stage will separate after an 8-year cruise which is significantly longer than a UOP stack would fly before separation (5 years) (J. Benkhoff et al. 2021; J. Englander et al. 2026).

A detachable SEP transfer stage design for UOP is driven by the fact that the power required for operating existing SEP engines can only be feasibly generated by current solar array technology out to approximately 4 AU due to diminishing light levels. Although SEP is currently infeasible at Uranus's heliocentric distance, a SEP stage is highly enabling in that it would allow UOP to launch with the same flight time, propellant mass, and perihelion distance in any year and deliver equivalent spacecraft mass to Uranus without dependence on Jupiter or Venus gravity assists, Figure 3. SEP-enabled spacecraft are now widely used for operating in Earth orbit with thrusters being produced for commercial customers that are also suitable for interplanetary applications such as UOP (J. Englander et al. 2026). It should also be noted that nuclear electric propulsion (NEP), featuring a nuclear reactor, was considered but not included in the trade space for 2025 to 2040 LRDs due to the projected unavailability of that technology within this timeframe.

Lastly, preliminary analysis based on currently available performance information indicates that a Starship-based solution could be launched in any year into a direct trajectory, thus delivering more than sufficient mass to Uranus orbit within as few as 10 years with feasible arrival velocities (e.g., D. Gochenaur et al. 2025). However, there are several key outstanding challenges with this implementation, including the need for low Earth orbit refueling before departure for Uranus, with a campaign of many launches to fill the fueling depot. Spacecraft-specific fairing accommodations will also need to be addressed, such as the pre-launch integration of RTGs, thermal waste heat management, and contamination concerns throughout the launch and refueling phase until the spacecraft is injected onto its interplanetary trajectory.

3.2 Arrival and potential for aerocapture

Aerocapture provides a potential alternative approach to executing a large chemical insertion burn on planetary arrival, where the spacecraft, enclosed within a protective aerocapture vehicle, would skim the upper atmosphere to rapidly dissipate the approach velocity and achieve orbit. This approach has the potential to reduce the UOP orbiter mass and volume by reducing overall chemical propellant requirements and shortening transit times by accommodating higher arrival velocities. Aerocapture could potentially be combined with any of the launch/transfer options in Table 1 but only offers an advantage in the case of very high velocity, short flight time, trajectories, such as on a Starship or SLS direct trajectory (D. Mages et al. 2025).

For example, the 2010 UOP Decadal study (W. Hubbard 2010) concluded that aerocapture begins to offer mass advantages when capture Δv exceeds 2 km/s, which is appreciably higher than what was required in the 2010 and 2021 Decadal reference mission designs (1.66 km/s and 1.09 km/s, respectively). The 2010 UOP study also observed that spacecraft packaging and probe accommodations are much more challenging with aerocapture and acknowledged the need for substantial technology investment to be ready for flight, including a successful prior flight demonstration under relevant test conditions. The 2017 Ice Giants Pre-Decadal Study (M. Hofstadter et al. 2017) agreed that aerocapture technology is best suited for solutions where orbit capture requirements exceed that which is achievable with conventional bipropellant systems (<3 km/s) and is thus unnecessary for a Uranus mission but may yet be enabling for a Neptune mission.

Much work has been done to assess aerocapture system risks and mature aerocapture technology (e.g., ADRAT 2023, ANDRAT 2024, S. Dutta et al. 2024, 2025, 2026; A. Gomez-Delrio et al. 2024, 2025, R. Deshmuhk et al. 2025). In our study we again considered whether aerocapture could enhance a Uranus orbiter mission without introducing undesirable levels of complexity and risk. Our analysis identified several spacecraft design challenges and risks imposed by the accommodation of a blunt body aerocapture vehicle. In addition to general packaging concerns to fitting the flight system within an aeroshell, integrating the spacecraft, and multiple RTGs, within the aeroshell inside a launch vehicle fairing will be complex. The compact packaging will need to ensure that any spacecraft, probe, and instrument radiator fields of view are clear of hot components, including RTGs, which is a challenge in a tightly packed aeroshell.

We also find that aerocapture poses a significant thermal management challenge that affects flight system design. The two Next Gen RTGs required for UOP generate four times the waste heat of the single Multi-Mission RTG (MMRTG) carried by Mars Science Lab (MSL) or Dragonfly (e.g., G. Holtzman et al. 2020, K. Gonter et al 2025), leading to a far larger and more complex heat rejection system (e.g., A. Gomez-Delrio et al. 2025). The system must perform over a wide range of environmental extremes including the period around the aerocapture event when external heating is at a maximum and the ability to reject RTG waste heat is likely compromised. While thermal configuration is a recognized concern, a potentially risky and time-critical reconfiguration of the thermal system to thermally stabilize the RTGs after aeroshell separation on orbit presents an additional challenge.

The use of aerocapture for UOP would also introduce numerous operational challenges and potential risks. Uncertainty about Uranus's upper atmospheric conditions is likely to remain prior to the ability to conduct an in-situ investigation (i.e., prior to the UOP mission) increasing the risk that aerocapture may not perform as required (ADRAT 2023). This risk is exacerbated by the need to tightly control the post-capture orbit to avoid potential ring hazards. This scenario is further complicated by the fact that several critical spacecraft subsystems, including propulsion, communications, and navigation would not be fully commissioned, characterized, and properly calibrated until after delivery into Uranus orbit and separation from the encapsulating aeroshell vehicle. The resulting schedule compression limits and constrains the on-orbit commissioning period prior to execution of time-critical, post-capture activities such as initial orbit determination and hazard assessment, contingency maneuver planning, and preparations for probe deployment. Similarly, encapsulated science instruments could not deploy covers, heat shields, booms, or antennae and would likely lose crucial cruise performance trending and calibration periods. After investigating currently available technologies and potential performance, and given the increased challenges and operational complexity, we concluded that aerocapture was too risky and not needed for our UOP design.

3.3 Probe delivery and Uranus system tour

A novel aspect of the 2021 Decadal concept was an end-to-end tour design that featured probe delivery after orbit insertion to remove critical event conflicts between orbit insertion and probe data relay (A. Simon et al. 2021). Consistent with that approach, additional investigations have since been conducted that are focused on 1) further reducing maximum entry accelerations to allow for the delivery of modern instrumentation capable of making higher quality measurements and minimizing requirements imposed on vehicle components, and 2) accommodating increased orbiter flyover altitudes and probe-orbiter distances that are necessitated by the redesigned, higher-altitude capture as described previously.

Recent investigations have focused on limiting the probe peak entry acceleration to 50 g (compared with 110 g in the 2021 Decadal study), which requires reducing the entry vehicle's horizontal flight path angle compared to the -30.7° used in the 2021 Decadal reference design (A. Simon et al. 2021). As shown in (D. Ellison et al. 2025), Uranus's ring system presents a physical obstacle that prevents reduction of the flight path angle from any arbitrary capture orbit. The primary mechanism for achieving shallow entry flight path angles (e.g. approximately -16° to -18°), while avoiding probe delivery trajectories that intersect the inner rings, is to reduce the capture orbit latitude of periapse. Lower periapse latitudes are most easily achieved, with commensurately low times of flight, with low interplanetary arrival declinations relative to Uranus's equator. Low arrival declinations are more easily accessed around Uranus's equinox, that is, towards the end of the launch dates considered here.

The second major focus of recent studies surrounding atmospheric probe delivery is the accommodation of higher orbiter flyover altitudes, where the spacecraft passes through the $\epsilon - \nu$ ring gap during probe descent. Analysis presented in (D. Landau et al. 2025) and (D. Ellison et al. 2025) demonstrates that adequate probe data relay is achievable with a higher flyover while not fundamentally altering probe operations or propagating additional constraints into the design of the subsequent tour phase (M. Amato et al. 2024).

The 2021 Decadal concept featured a full Uranus system tour after the completion of the probe's mission. Given the challenges of power generation at 20 AU, and the uncertainty surrounding the availability of fissile material for future outer planets scientific missions, recent satellite tour designs have focused on achieving the baseline requirements identified by the 2021 Decadal under a conservative set of operational assumptions. Example tour designs presented in (D. Landau et al. 2025) and (D. Ellison et al. 2025) demonstrate that a baseline tour is possible within four years of probe delivery. Furthermore, it is possible to achieve the baseline tour requirements of three flybys per major satellite (Miranda, Ariel, Umbriel, Titania, and Oberon) in that timespan, while maintaining a minimum of 30 days between encounters (D. Ellison et al. 2025), which accommodates the science data and telemetry downlink volume requirements identified by the 2021 Decadal subject to the restrictive data rates that are expected while operating in the Uranian system (see Section 5).

4. Spacecraft Design Updates

The spacecraft (orbiter and probe) design concept has been updated since the 2021 Decadal study to satisfy the following primary objectives:

- investigate alternatives for lowering power generation requirements without sacrificing mission capability and Decadal science return;
- consider development of alternative power generation sources;
- reduce mass to the extent possible;
- maximize compatibility across the updated set of viable architecture alternatives, including those leveraging Starship and an SEP stage with an FHE; and
- update the probe design with latest knowledge.

4.1. Reduced power generation

The 2021 Decadal study baselined three Mod 1 Next Gen RTGs for power generation, with each unit providing up to 245 W_e at fueling and a combined output of ~530 W_e by EOM. This solution struck a balance between conserving plutonium (Pu) resources and enabling robust science return with minimal operational compromises. With three Next Gen RTGs, the highest load cases, involving the firing of the main engines, could be supported with a small secondary battery; with this solution the instruments would be fully operable during the mission science phase outside of main engine operations and downlink periods. A fourth Next Gen RTG was initially considered to allow for continuous instrument operation, but would have resulted in higher spacecraft dry mass, greater cost, and a far more complex nuclear launch integration process. The use of multiple MMRTGs was also considered but determined to be infeasible because as many as seven of them, requiring more than 2.5 times the net mass accommodation, would be required to generate the equivalent EOM power provided by three Mod 1 Next Gen RTGs.

After the 2021 Decadal study was completed, the updated production outlook for the Mod 1 Next Gen RTG (C. Sandifer 2024) suggested the possibility that three fueled units might not be ready in time for 2035 UOP launch. To mitigate that risk, we investigated whether the Decadal UOP mission could be accomplished with only one or two Next Gen RTGs. An initial energy balance analysis was conducted to examine the relationship between the number of RTGs and the amount of science data that could be returned per reference orbit (R. Anderson 2024). Given that a higher capacity battery would be required to support cases with fewer than three Next Gen RTGs, the analysis is based on three primary activities occurring during each orbit: science collection, downlink, and battery charging. The relative time (i.e., fraction of an orbit) spent on each of these charging- and discharging-intensive activities is not independent but rather is constrained by the requirement to achieve battery energy balance. This approach allows for a rough-order feasibility assessment without the need to assign a specific battery capacity or a particular concept of operations (ConOps). The orbiter power equipment used for this analysis (see Appendix I) was retained from the 2021 UOP Decadal, but with the following updates:

- increased total payload power allocation to 50 W with added contingency;
- removal of all spacecraft bus electrical heaters in favor of a pumped thermal fluid loop (instruments would retain survival heaters);

- assumed use of the GS Yuasa Generation 3 LSE134 Li-ion battery cells (155 Wh/kg, flown on the International Space Station and selected for the Dragonfly lander), with known charge/discharge characteristics and demonstrated low-rate charge acceptance;
- introduction of a reduced power 3-axis mode for the 2-Next Gen RTG case that allows for recharge without having to toggle spin-stabilized deep hibernation mode; and
- addition of a spin-stabilized hibernation mode and transition modes for the 1-Next Gen RTG case (insufficient energy available to recharge the battery while maintaining 3-axis stabilization).

The spacecraft communications architecture and ~34-day reference orbit duration were retained from the 2021 Decadal study. The analysis calculated the net time per orbit that is available for science collection, given that one third of the time is spent in downlink (assuming daily 8-hr DSN contacts at 19.5 kbps), and with the required battery recharge time determined by applying the energy balance constraint. The results are presented in Table 2.

Table 2. Analysis of Science Return for Different Power Scenarios

	3 Next Gen RTGs	2 Next Gen RTGs	1 Next Gen RTG
End of Mission Power	520 W_e	350 W_e	175 W_e
Battery Recharge	0% of orbit	28% of orbit	76% of orbit
Data Downlink	33% (8 hr/day)	33% (8 hr/day)	17% (4 hr/day)
Science Collection	67% of orbit	39% of orbit	7% of orbit
Data Volume	2.3 GB per orbit	2.3 GB per orbit	0.4 GB (X-band)
Instrument Duty Cycle	Off: downlinks	Off: battery recharge and downlinks	Off: most of orbit
Payload Enhancement	Possible (>30% power margin)	Unlikely	No

For the 2 Next Gen RTG case, the 2021 Decadal study reference data volume can still be returned, albeit with a nearly 42% reduction in per orbit net science collection availability due to battery recharging requirements. With only 1 Next Gen RTG, there is insufficient power to support downlink while in 3-axis mode, precluding the use of higher-rate Ka-band communications that require fine pointing. Achievable data rates are significantly lower (~6 kbps) for an X-band, spin-stabilized spacecraft at the same transmit power. Moreover, a 1 Next Gen RTG spacecraft has only enough power to maintain 4-hr daily downlinks instead of 8-hr. These factors combine to provide a mere fraction (17%) of the data return capability of either the 2 or 3 Next Gen RTG configurations, with a spacecraft that is only available for science data collection for 7% of any given science orbit.

The initial energy balance analysis proved sufficient to conclude that a 1 Next Gen RTG spacecraft cannot achieve the Decadal-prioritized science collection and data return. However, a 2 Next Gen RTG implementation is potentially viable assuming constrained operations. In that case, the potential effects of spacecraft operational unavailability during both downlink and recharge periods can likely be mitigated with careful science orbit activity planning and the use of an appropriately sized battery. Given these findings, a 2 Next Gen RTG system has been baselined for additional concept development. A more detailed, follow-up analysis of the two-Next Gen RTG case from a ConOps perspective is presented below in Section 5.

4.2. Power generation alternatives

Given the general uncertainty over the availability of ^{238}Pu and the future of the NASA RPS program, the potential utility of alternative power generation sources for the orbiter were investigated. Any feasible replacement for the two currently baselined Next Gen RTGs would need to: be capable of generating ~350 W_e of power by EOM; be available within LRDs from 2035-2040; be roughly equivalent in total mass to the Next Gen RTG (~56 kg); and require ≤ 4 total units to be integrated to the spacecraft, per established nuclear launch integration practice (R. McNutt et al. 2015). The use of MMRTGs, which were already ruled out as infeasible for UOP per the discussion in Section 4.1, would also be unavailable in a scenario with limited ^{238}Pu fuel availability. Other alternatives we considered include the use of Americium (^{241}Am)-based RPS devices, solar arrays, and/or possible fuel cell technology.

An ongoing series of studies and technology development programs for Americium (^{241}Am) fueled RPS generators (Am-RTG) is intended to provide alternative power solutions for space applications (R. Ambrosi et al. 2025). While ^{241}Am is more readily available and more cost-effectively produced than ^{238}Pu fuel (R. Mesalam et al. 2025a), and its half-life is considerably longer at 432 years, its thermal output relative to unit mass (*i.e.*, energy density) is ~5 times less than ^{238}Pu (H. Sargeant et. al. 2025), therefore providing only a fraction of the energy available for the generation of electrical power and the potential harvesting of waste heat for use within the spacecraft. Two versions of ^{241}Am fueled generators are being developed: one using thermoelectric power conversion technology (Am-RTG), and another with higher efficiency dynamic Stirling convertors (Am-RSG). The Am-RTG, based on the latest qualification of an engineering model (R. Mesalam et al. 2025a), is projected to weigh approximately 50 kg and supply up to 50 W_e (DC) at beginning of life (BOL), dropping to ~40 W_e after 20 years of operation, given an expected degradation rate of ~1% per year (R. Ambrosi et al. 2019). As many as nine of these devices would be required to provide ~350 W_e at EOM, even more than would be required for MMRTGs (Section 4.1).

Alternatively, the Am-RSG (R. Mesalam et al. 2025b, H. Sargeant et. al. 2025) has the potential to offer increased electrical power efficiency. Experimental results obtained from operating a half-sized practical demonstrator and test bed (H. Sargeant et. al. 2025; R. Ambrosi et al. 2025) indicate that the Am-RSG's Stirling engines could potentially generate ~88 W_e (AC) from 400 W_{th} at fueling, with the implication that a full-scale Am-RSG generator (800 W_{th} heat source) may be able to achieve up to ~180 W_e (AC) at BOL. A practical application of this generator technology would also require the operation of electronics to actively control the Stirling convertors and convert their AC output to DC, further reducing the net power available to the spacecraft bus. The total mass of an Am-RSG, including controllers, has been estimated at 124 kg (MEV) (R. Mesalam et al. 2025b). Based on these most recently published estimates, it is likely that at least two Am-RSGs would be required to replace the generated power of two Next Gen RTGs for UOP, but at more than double the mass per device and with approximately 20% of the generated waste heat available to the spacecraft for facilitating thermal management. The lifetime of a Stirling engine's moving parts is also a concern for long deep space missions. Previous testing on plutonium-based dynamic RPSs indicated that the Stirling engines could be robust for a 17-year lifetime (S. Oriti et al. 2023). Further studies are also needed to address potential vibration concerns from unbalanced operational scenarios should any of the individual Stirling engines within a generator fail or need to be shut down. In principle, the Am-RSG could offer a feasible alternative for UOP if the required technology development (currently estimated as TRL ~3.5) is completed and there is sufficient production of ^{241}Am fuel; two to three flight qualified and fueled operational units would need to be ready for launch by 2035 under our currently assumed launch readiness dates.

An additional consequence of the possible shut down of U.S. ^{238}Pu production would be the loss of the Light-Weight Radioisotope Heater Units (LWRHUs) that would provide spot heating sources on the UOP probe to enable it to survive the ~20 to 30-day coast period between orbiter separation on-orbit and Uranus atmospheric entry without increasing battery mass and volume. While some thermal trades are possible to reduce the number of RHUs, heat from RHUs are still needed to minimize high heater power utilization during the coast period. An Americium-based RHU is also being explored as a potential replacement that would provide 3 W_{th} with a specific thermal power of 15 W_{th}/kg, but the expected timeline for availability of flight-ready units is unknown. (R. Ambrosi et al. 2025). If Pu-based RHUs are not available for UOP probe use, either the Americium RHUs or a larger battery capacity would be required, with adjustments needed to the probe volume and size to accommodate them.

The potential use of solar power has also been considered as an alternative for the UOP orbiter should the Next Gen RTG or Am-RSG be unavailable. Using currently available, and near-future, technologies, a solar array area on the order of 570 m^2 would be required to supply ~350 W_e to the spacecraft at 20 AU when also accounting for the losses associated with transferring power across such an extremely large array. The required 570 m^2 array area, roughly five times that of Europa Clipper's solar arrays (100 m^2), and ~60% larger than the JWST sunshade (312 m^2), would be well beyond anything assembled to date, and is likely to require revolutionary, rather than evolutionary, design advancements. Significant technology development would also be required to produce solar cells and cover glass capable of operating at the extreme low-intensity, low-temperature (LILT) environment at Uranus. Moreover, even if these technology challenges are overcome, such a large array would create significant challenges for spacecraft attitude control, which would likely compromise slew rates, pointing accuracy for science observations and Ka-band data return, and science viewing windows if the spacecraft must maintain a fixed attitude throughout the mission. Significant instrument accommodation challenges, including the provision of required fields of view and mitigating glint, are also likely. Given these and other likely associated challenges, the use of solar arrays to generate power for UOP does not appear to be feasible for LRDs in the 2035-2040 timeframe, even with significant technology investment.

Lastly, advanced fuel cell technology, in which chemical energy of a fuel is converted into electricity, is another potential option for power generation. Early versions of these systems were developed for the Gemini and Apollo missions, but the historical focus has been on high power needs at approximately room temperature (e.g., K. Burke 2003). These systems have high power output, but at high mass; for reference, the Apollo 1.5-kW cell weighed 111 kg (M. Warshay & P. Prokopius 1990). With future developments, mass efficient, cryogenic, cells may yet have potential applicability and feasibility for UOP, but further consideration of that option is left to future work.

4.3. Orbiter mass optimization

Reducing the number of Next Gen RTGs from three to two also provided an opportunity to reconfigure the spacecraft concept to be more structurally and thermally efficient (Figure 4). First, the propulsion system was rearranged with the fuel tanks placed outside of a now shorter central cylinder, allowing for a more compact spacecraft layout that is also better suited for a pumped fluid loop thermal system, with the external fuel tanks tented under multi-layer insulation (MLI). The propellant type and corresponding engine/thruster selections were also updated from a hydrazine-based system to monomethyl hydrazine (MMH), allowing for lower-temperature storage and less stringent thermal requirements, as has also been baselined in other recent studies (S. Weinstein-Wiess et al. 2025).

Other noticeable differences from the 2021 Decadal orbiter design include a reconfigured UHF-band antenna for supporting probe communications from higher flyover altitudes, and an upgrade of the UT700 processor-based avionics to a higher performing GR740 processor-based design, which allows the orbiter to potentially control an added SEP stage as a remote terminal and enable advanced onboard data processing and compression options. Radiation shielding could also be reduced across the spacecraft without a JGA, affording additional mass savings opportunities. Together, these orbiter design updates have resulted in a net 17.5% (338 kg) reduction in maximum expected value (MEV) orbiter dry mass, as outlined in Table 3 and shown in greater detail in Appendix I, Table A1. Note that the updated mass total also includes an increased allocation for the orbiter instrument payload (78 kg) to incorporate higher contingency factors.

The above-described enhancements to the orbiter proof of concept design are consistent with the stated approach of the 2021 Decadal study, in leveraging existing technologies and industry state of practice to the maximum extent possible to consciously maintain a reduced cost and risk posture. Other potentially enabling state-of-the-art or in-development technologies and implementation approaches have also been identified (*e.g.*, S. Weinstein-Wiess et al. 2025). This is consistent with the stated purpose of conducting advanced concept studies to produce and assess a broad spectrum of ideas and alternatives in exploring the mission trade space (S. Hirshorn 2016). In any case, it is important to note that the emphasis of these early concept studies should continue to be on establishing feasibility with a focus on maximizing overall mission value, while maintaining flexibility and not prematurely locking into any particularly optimized solution prior to entering Phase A. For the time being, our approach is to maintain a design reference that maximizes compatibility with multiple candidate mission architectures and that can serve as a feasible baseline against which potential enhancements can be evaluated.

Table 3. Mass summary comparison between 2021 UOP Decadal reference orbiter and current design concept.

UOP Orbiter Mass Summary	MEV Mass (kg)		Change	Comments
	2021	Current		
Total MEV Dry Mass	1999	1674	-16.3%	
Orbiter Bus	1933	1595	-17.5%	
Structures	812	576	-29.0%	Central cylinder & upper structure optimization
Propulsion	436	264	-39.5%	Updates for MMH-based system, reduced Δv
Avionics	31	32	2.5%	Updated architecture, allows for SEP control
Power	267	221	-17.4%	Removed an RTG, increased battery size
AD&C	77	79	2.4%	Minor changes
Thermal Control	75	193	158.9%	Pumped fluid loop, updated S/C configuration

Communications	104	121	16.0%	Reconfigured UHF antenna (for probe relay)
Harness	131	109	-16.3%	Wrap factor
Instrument Payload	67	78	17.8%	Higher contingency added

4.4. Launch vehicle and propulsion system analysis

The updated orbiter design was also assessed for potential compatibility as paired with an SEP stage launching on an FHE, and for a Starship direct launch, with the intent of developing a common spacecraft design that is generally applicable across these leading architectural alternatives. A proof-of-concept SEP stage architecture for UOP that is compatible with multiple, commercially available solar array and SEP component alternatives has been developed as a part of the trade study detailed in Englander *et al.* 2026. An example mass summary and detailed Master Equipment List (MEL) are provided in Appendix I, illustrating the performance of one of the many SEP stage design configurations considered, based on commercially available PPS5000 Hall effect thrusters and a 35 kW (end of life) solar array rated at 1 AU. The PPS5000 configuration is considered as a potential design reference because it can be assembled from commercial off-the-shelf (COTS) SEP system components that are routinely operating in Earth orbit today. Several other SEP engine alternatives are also considered in Englander *et al.* 2026, including the commercially available, gridded ion RIT-2X thruster, and the H-10 high-power density Hall thruster currently in development. Figure 5 illustrates the SEP stage as configured with Airbus rigid panel solar arrays as baselined for the ESA Mars Earth Return Orbiter (ERO). Other alternatives identified with similar capabilities include the Airbus NL Flexible Compact Array and the Lockheed Martin Multi-Mission Array.

Starship compatibility has also been initially assessed using the latest version of the User's Guide (SpaceX 2020) and through an ongoing dialogue with SpaceX. The orbiter as currently designed would have no issues fitting within Starship's large (8-m diameter) fairing, but as discussed in Section 3.1, thermal management of the orbiter's Next Gen RTGs while encapsulated and the mitigation of potential contamination concerns during on-orbit refueling operations would need to be resolved. The current spacecraft design would also have to be further modified to some extent to accommodate the more challenging thermal environments of an inner solar system tour trajectory, but that remains as future work.

4.5. Probe design maturation

The probe design has also evolved since the 2021 Decadal study to incorporate lessons learned and development updates from maturation and risk reduction efforts for the DAVINCI Venus probe (R. Ripley & C. Goodloe 2025), see Figure 6. In addition, the full Decadal instrument suite was incorporated into the descent vehicle, and a set of updated analyses were performed for thermal, structural, data return, and battery sizing to ensure maximum flexibility in future payload selection. The 2021 Decadal probe design included three notional instruments which has been expanded to six in this updated study (mass spectrometer, atmospheric structure sensors, ortho-para hydrogen detector, plus nephelometer, helium abundance detector, and net flux radiometer). The size of the descent vehicle was therefore increased, providing additional flexibility in accommodating future instruments and supporting a larger battery to provide ample power margin. The entry system scale increased to accommodate the larger diameter descent vehicle, leading to overall probe mass and size growth (Table 4) relative to that in 2021.

The probe telecom system was also updated to include the use of adaptive data rate (ADR) with fixed step rates as backup. The switch to ADR maximizes data return from the descent vehicle after the backshell is jettisoned. Additional trades were performed to optimize size and mass of the descent vehicle. Some of these trades involved looking at a vented probe design vs sealed and an alternate RF antenna to reduce structural mass; these trades do not produce large changes in the probe mass or size. The final design would be optimized once payload selections and accommodations are finalized, and some savings could be realized at that time.

Table 4. Probe Mass summary comparison between 2021 Decadal UOP probe design and the updated design concept.

Probe Mass Summary	MEV Mass (kg)		Change	Comments
	2021	Current		
Probe Total MEV Dry Mass	268	402	50.0%	
Descent Vehicle	124	186	50.0%	Structure updated, battery size and instrument volume increased
Entry System	122	179	46.5%	Scaled up for increased descent vehicle size
Probe Instruments	22	37	70.0%	Added 3 additional instruments

5. Updated Concept of Operations

This updated design study also investigated how a mission might operate and return high priority science with reduced spacecraft power levels. From the science perspective, *in-situ* sampling and proximity-based science that cannot be recovered from future Earth-based assets remain the highest priority. To that end, this study retains the probe requirements (i.e., delivery after orbit insertion and operation to 10 bars to deliver critical *in situ* data) untouched, changing only the entry conditions as described in Section 3.3.

For orbital science, a recent community science poll confirms that while a range of geometries are desired for each science objective, the close passes of the planet and major moons are a high priority (A. Simon et al. 2025). For each science objective, the poll found that a range of viewing/sampling geometries is desired to best achieve the science at each system target. To analyze how this might be achieved, a notional 33.3-day orbit was examined to assess the power, data, and operations management that maximized the science return for a scenario with two Mod 1 Next Gen RTGs. Rather than attempting to optimize every orbit of a tour to meet the requirements of every objective, observations were instead interleaved to fill the orbit.

To fill an orbit, a simplified instrument payload was assumed to scope the power level and data rates, see Table 5. For these instruments, the power and data rates were roughly derived from the 2021 Decadal study with no specific heritage assumptions, nor are they meant to be inclusive of all instrument characteristics. For remote sensing data volume estimates we included two spectrometers and a single camera, consistent with the Decadal study threshold payload (A. Simon et al. 2021). All instruments were assumed as fixed-mounted, with any scanning requirements exceeding feasible spacecraft slewing capability assumed to be included within individual instrument power and mass accommodations. Instrument survival heater power (when turned off) was assumed to be 30% of operating power, and 2:1 compression was applied for all data types. Three bounding science cases were explored: 1) a satellite flyby-focused orbit, 2) an atmosphere and rings-focused orbit, and 3) a fields and particles-focused orbit. For each case, observations were added ranging from brief, data-intensive sets (e.g., remote sensing), to long duration but lower data rate sets (e.g., field and particles system mapping).

Table 5. Simplified Payload Assumptions

Instrument Type	Data Rate[a]	Power[b]
1024x1024 Framing Camera	12.6 Mbit per frame	10 W
512x1024 Pixel Spectrometer	6.3 Mbits per frame	12 W
1024-Channel Point Spectrometer	12.3 kbits per spectrum	10 W
Magnetometer	11 kbps	5W
Fields and Particles Suite	10 kbps	13W

Notes: a. Remote Sensing data rates assume 12 bits per pixel. b. All instruments are assumed to have a 1-hour warm up before science acquisitions begin.

A higher-fidelity version of the energy balance analysis described in Section 4.1 was conducted, incorporating selectable battery capacities and more realistic charging and discharging scenarios based on relevant modes captured in the spacecraft Power Profiles (Appendix I). As a starting point, daily downlinks and battery recharging periods were included. For each case, necessary targeting maneuvers and navigation activities were also added along with typical spacecraft housekeeping data allocations and downlink rates. A two-hour time step was selected as a reasonable cadence that would limit modeling complexity, yet allow for consideration of either four-, six- or eight-hour DSN contacts. The analysis approach allows for various spacecraft base power configurations to be selected per any given two-hour interval throughout the orbit:

- DSN Tracking (2-way X-band communications);
- DSN Science Downlink (X-band uplink, Ka-band downlink);
- Science Imaging (assuming greater use of reaction wheel at higher power); or
- Science Fixed Attitude (lower reaction wheel power).

Along with a selection of primary power configuration, the analysis also allows for the inclusion of an appropriate transient event for each interval: e.g., reaction wheel desaturation (DSAT), Orbital Trim Maneuver (OTM) engine burn, or optical navigation (OpNav) imaging. Power for each of the notional science instruments can be toggled on or off for each interval. For the fields and particles suite, the model assumes the power expenditures and data rates are steady state per the values in Table 5 when powered, and only survival power is used when turned off. The same on/off behavior is modeled for the two imagers and the point spectrometer, but the model also allows an entry for the number of frames or spectra to be taken within any interval when they are turned on, allowing for a more realistic calculation of data collection rates. When none of the above power consuming activities are selected for a given interval, the model defaults to a lower power 3-axis mode that allows for battery recharging. The model generates plots of the battery state of charge and cumulative data volume throughout the modeled orbit, allowing for analyses of any selected spacecraft and payload configuration.

Using ConOps planning like that of Cassini (B. Paczkowski & T. Ray 2004) or Europa Clipper (L. Cangahuala et al. 2025), and varying with the science use case, downlink periods in the model were removed as necessary to accommodate required long-duration science observations, and notional instrument observation sets were added. To ensure that a given scenario closes in term of the power and data downlink availability, data acquisitions were limited by the battery depth of discharge to no more than 50% (40% preferred), and data carryover to minimize the data retained onboard at the end of the orbit. Additional observing periods were then added to fill the data acquisition and return capability of the orbit. An example for Science Case 1 (satellite flyby) is shown in Figure 7, with the power and data volume profiles shown in Figure 8. Of course, data volume and instrument cycling can be adjusted for a realistic future mission profile, but this proof-of-concept analysis shows that significant science can still be returned using only two Mod 1 Next Gen RTGs when paired with a reasonably sized, 120 Amp-hour battery, enabling up to 8-hour daily Ka-band DSN passes. Corresponding plots for Science Cases 2 and 3 are shown in Figures 9 through 12.

6. Conclusion and Next Steps

Using updated information, the design of the UOP Decadal mission concept has been revisited, resulting in a more streamlined design than was possible during the rapid Decadal study. We find that it is possible to reduce orbiter mass and volume and modify operation solutions to close with 2 Next Gen RTGs, in agreement with past lower fidelity studies (R. Anderson 2024, S. Weinstein-Weiss et al. 2025). The approach trajectory was also modified to move the orbit insertion farther from Uranus using extra propellant accommodated in this design. The probe delivery and design were also matured, reducing gravitational acceleration loads and increasing the payload accommodation volume. High fidelity concepts of operations were constructed across a variety of science scenarios, including all the necessary spacecraft maneuvers and battery charging periods, to show that 12-14 Gbits of science (plus ~1 Gbit of spacecraft telemetry) can be returned across each 33.3-day orbit.

Our study has also found a robust suite of trajectory and propulsion solutions to reach Uranus without using a Jupiter gravity assist. The best solutions, with reasonable flight times and yearly launch windows, employ either a commercially derived SEP transfer stage or a next generation super heavy lift vehicle; our current point design is compatible with either option and the SEP stage can deliver the streamlined spacecraft (Table 3) to Uranus in ~13 years (Table 1). BepiColombo is expected to demonstrate the use of a separable interplanetary SEP stage in late 2026; further, multiple options for SEP system components and solar arrays appropriately sized for a UOP-specific SEP stage are commercially available today. For next generation launch vehicles, updated information on lift capability, launch loads, and spacecraft accommodation constraints will need to be resolved. For example, further study is needed to understand accommodation of UOP in a Starship fairing, including the complexities of refueling and maintaining spacecraft thermal control while stowed for an extended period.

Although not required, future technologies can be leveraged by UOP as they continue to mature and reach flight-ready levels. For example, advanced fuel cells could help with the power constraints of this mission, particularly with a limited availability of traditional RTG power systems. Data return can also be enhanced with advanced compression techniques and/or machine learning that can process data onboard, as many missions and instruments are actively investigating. Optical communication is also promising, improving data return rates from 20 AU to an estimated 50 to 500 kbps based on Psyche mission technology demonstrations (A. Biswas et al. 2025). However, the power implications for transmitters and pointing control from such great distances, as well as the cost and schedule to develop ground stations, are currently poorly constrained.

Ultimately the final mission design parameters for UOP will depend on launch and arrival date constraints, propulsion solutions, partnership agreements, and final payload selections. Our current design maintains robust margins and accommodation flexibility, such as the SEP control system in the avionics, and extra mass and volume margins for instrument selections on the orbiter and in the probe. However, our spacecraft design concept can be further optimized and streamlined once launch readiness dates and launch vehicle availability are determined, along with constraints on power systems, data volume, and cost profile. Most importantly, this design study demonstrates that without requiring a Jupiter gravity assist, and in a power-constrained environment, important cross-disciplinary science can be returned without requiring major technology developments. These results suggest that high-fidelity point design studies of SEP-driven missions, as well as super heavy lift options, should be initiated as soon as is feasible to prepare for exploring the exciting and enigmatic Uranian system.

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Appendix I: Mission Design Tables

Table A1. Mass summary for the UOP design reference orbiter, as configured with a SEP stage using PPS5000 engines and a 35-kW Airbus rigid panel solar array. Maximum Possible Values (MPV) reflect the addition of 30% unallocated margin to the MEVs as shown.

UOP Flight System Mass Summary	CBE	Cont.	MEV	MPV
Orbiter Total Dry Mass	1500.2 kg	12%	1673.7 kg	2175.8 kg
Orbiter Bus	1439.7 kg	11%	1595.3 kg	
Structures	501.0 kg	15%	576.1 kg	
Propulsion	250.3 kg	6%	264.1 kg	
Avionics	27.7 kg	15%	31.7 kg	
Electrical Power	203.8 kg	8%	220.8 kg	
Attitude Determination and Control	75.2 kg	5%	78.8 kg	
Thermal Control	176.5 kg	9%	193.2 kg	
RF Communications	107.2 kg	13%	121.0 kg	

Harness	98.1 kg	12%	109.5 kg	
Orbiter Instrument Payload Total	60.5 kg	30%	78.4 kg	
Probe Total Mass	337.7 kg	19%	402.1 kg	482.5 kg
Descent Vehicle	148.9 kg	25%	186.0 kg	
Entry System	158.6 kg	13%	178.7 kg	
Probe Instruments	30.2 kg	24%	37.4 kg	
Orbiter+Probe System Dry Mass	1837.7 kg	13%	2075.8 kg	2658.3 kg
MMH Residuals at 20°C				25.1 kg
Oxidizer (NTO-MON3) Residuals at 20°C				38.0 kg
GHe Pressurant				11.9 kg
Usable MMH Propellant and Oxidizer (2315 m/s ΔV)				2717.5 kg
Orbiter-Probe System Launch Mass (Wet)				5450.9 kg
Point Design UOI Injection Mass				6314.0 kg
Unused Point Design Injection Mass				863.1 kg
SEP Stage Dry Mass	1417.3 kg	13%	1601.5 kg	2082.0 kg
SEP Stage Xenon				1648.0 kg
SEP Stage RCS Propellant				45.0 kg
SEP Stage Launch Mass (Wet)				3775.0 kg
Total Flight System (Orbiter+Probe+SEP) Dry Mass (Tanks Empty)				4740.3 kg
Total Flight System MPV Launch Mass (Wet)				9225.8 kg
Launch Vehicle Capability (FHE, C_3=20.3 km²/s²)				10055.0 kg
Unused Launch Vehicle Capability				829.2 kg
Propellant Mass Fraction (Orbiter+Probe)				49.9%

Table A2. Master Equipment List for a SEP-compatible orbiter bus design

Subsystem/Component	Unit Mass CBE (kg)	No. of Units	Total Mass CBE (kg)	Contin-gency	Total MEV Mass (kg)	Heritage Basis
Structures			500.97	15%	576.12	
Primary Structure - Cylinder	199.90	1	199.90	15%	229.89	Europa Clipper
Primary Structure - Electronics Vault	92.80	1	92.80	15%	106.72	Europa Clipper
RTG Bracket	20.00	2	40.00	15%	46.00	New Horizons
1-lb Thruster Supports (total)	25.00	1	25.00	15%	28.75	Europa Clipper, scaled
Probe Release System	22.00	1	22.00	15%	25.30	Estimated, similar to Cassini
MMH Fuel Tank Support Bracket	9.92	4	39.68	15%	45.63	From CAD
Pressurant Tank Support Bracket	6.88	4	27.52	15%	31.65	From CAD
Miscellaneous Secondary Structure	29.27	1	29.27	15%	33.66	Typical
Launch Vehicle Interface HW (S/C side)	24.80	1	24.80	15%	28.52	Europa Clipper
Propulsion			250.27	6%	264.09	
Oxidizer Tank	57.25	1	57.25	3%	58.97	Northrop (NGIS) 80517
MMH Fuel Tank	29.83	2	59.65	3%	61.44	Northrop (NGIS) 80425
GHe Pressurant Tank	18.50	2	36.99	3%	38.10	ARDE D4619
Main Engine (L3Harris R-45-15 HiPAT, 445 N)	5.05	4	20.19	3%	20.80	Re-qualify with latching valves
Steering/RCS Thruster (Moog DST-12, 22 N)	0.66	16	10.56	3%	10.88	Europa Clipper
Fuel/Ox Service Valve	0.20	2	0.41	3%	0.42	Typical, many missions
Helium Service Valve	0.17	8	1.40	3%	1.44	Typical, many missions
Low-Pressure Latch Valve	0.85	12	10.25	3%	10.56	Typical, many missions

High-Pressure Latch Valve	0.71	4	2.83	3%	2.92	Typical, many missions
Dual Seat Pressure Control Valves	0.36	4	1.45	3%	1.49	Typical, many missions
Pressure Transducer	0.78	12	9.32	3%	9.60	Typical, many missions
Filter	0.70	4	2.80	3%	2.88	Typical, many missions
Tubing / Fasteners / Tube Clamps / Etc.	16.54	1	16.54	20%	19.85	Typical, many missions
Misc. Thermal Hardware	6.34	1	6.34	20%	7.61	Typical, many missions
Cabling (Wire, Harness, Supports)	14.28	1	14.28	20%	17.14	Typical, many missions
Avionics			27.66	15%	31.74	
Integrated Electronics Module (IEM)	19.22	1	19.22	15%	22.10	Parker Solar Probe, Van Allen Probes
Propulsion Diode Box (PDB)	1.75	4	7.00	15%	8.05	Parker Solar Probe
Remote Interface Unit (RIU)	0.09	16	1.44	10%	1.58	Parker Solar Probe
Electrical Power			203.80	8%	220.83	
Shunt Regulator Unit (SRU)	19.00	1	19.00	10%	20.90	New Horizons, Van Allen Probes
Power Switching Unit (PSU)	13.90	2	27.80	10%	30.58	Derived from Dragonfly
Battery	45.00	1	45.00	15%	51.75	Based on GSYUASA LSE 120 Cell (8x)
NG-RTG (Mod 1)	56.00	2	112.00	5%	117.60	Estimated as minor mod of GPHS-RTG
Attitude Determination and Control			75.15	5%	78.80	
Sodern Hydra Star Tracker - Sensor Head	1.40	3	4.20	3%	4.33	IMAP
Sodern Hydra Star Tracker - Electronics Unit	1.80	2	3.60	3%	3.71	IMAP
Northrop Grumman LR-450 mHRG IMU	4.55	2	9.10	3%	9.37	New NG model replacing Scalable SIRU
Adcole Digital Sun Sensor - Sensor Head	0.25	5	1.25	3%	1.29	PSP, DART, IMAP, New Horizons, Juno, etc
Adcole Digital Sun Sensor - Electronics Unit	1.50	2	3.00	3%	3.09	PSP, DART, IMAP, New Horizons, Juno, etc
Collins Aerospace RSI 68-75/60 Reaction Wheel Assembly	8.50	4	34.00	3%	35.02	PSP, MESSENGER (similar)
Radiation Shielding Allocation	20.00	1	20.00	10%	22.00	Estimate
Thermal Control			176.50	9%	193.25	
MLI Blankets (66.1 m^2))	41.40	1	41.40	20%	49.68	Typical, many missions
Heaters	0.10	10	1.00	20%	1.20	Typical, many missions
PRT	0.01	200	2.00	20%	2.40	Typical, many missions
Louvers	6.60	1	6.60	20%	7.92	Typical, many missions
ChoSeal	0.10	8	0.80	20%	0.96	Typical, many missions
Paint/Coatings	1.00	1	1.00	20%	1.20	Typical, many missions
Pumped Fluid Loop System		1	123.70	5%	129.89	PSP, Dragonfly, Europa Clipper
CFL CS 3/8" Lines	8.30	1	8.30	5%	8.72	Dragonfly
CFL CS 1/2" Lines	24.90	1	24.90	5%	26.15	Dragonfly
CFL Fittings, Transitions, etc.	1.90	1	1.90	5%	2.00	Dragonfly
CFL Saddle Flanges (radiators, PCA, PIA)	25.00	1	25.00	5%	26.25	Dragonfly
CFL Mixing Valve	1.00	2	2.00	5%	2.10	Dragonfly
Coolant Trim Radiator	5.00	1	5.00	5%	5.25	Dragonfly
Pumps and Check Valve Assy (2 pumps)	4.00	1	4.00	5%	4.20	Parker Solar Probe
Pump Electronics Stack (2 boxes)	4.00	1	4.00	5%	4.20	Parker Solar Probe
Pump Cntrl to Pump Harness Assmbly	1.00	1	1.00	5%	1.05	Parker Solar Probe
Differential Pressure Sensors (1)	0.60	1	0.60	5%	0.63	Parker Solar Probe
Differential Pressure Sensors (2)	0.60	1	0.60	5%	0.63	Parker Solar Probe
Total Pressure Sensors (1)	0.60	1	0.60	5%	0.63	Parker Solar Probe
Total Pressure Sensors (2)	0.60	1	0.60	5%	0.63	Parker Solar Probe
Total Pressure Sensors (3)	0.60	1	0.60	5%	0.63	Parker Solar Probe
Fill and Drain Valves	0.60	1	0.60	5%	0.63	Parker Solar Probe
NG-RTG Fluid Interface Manifolds (4)	1.00	4	4.00	5%	4.20	Estimated
Coolant	40.00	1	40.00	5%	42.00	Dragonfly
RF Communications			107.23	13%	120.95	
Radio	3.10	2	6.20	10%	6.82	APL Frontier Radio, many missions
X-Band EPC	1.42	2	2.84	5%	2.98	Europa Clipper
Ka-Band EPC	1.42	2	2.84	5%	2.98	Europa Clipper

Low Noise Amplifiers (LNAs)	0.01	4	0.04	10%	0.04	Europa Clipper
USO	0.50	2	1.00	15%	1.15	Many missions
Fanbeam Antenna	1.00	2	2.00	10%	2.20	Parker Solar Probe
Medium Gain Antenna- X-band	1.20	1	1.20	5%	1.26	New Horizons
Medium Gain Antenna, 2x2 Helix Array (UHF Probe Link)	21.00	1	21.00	15%	24.15	TIMED, Van Allen Probes
Low Gain Antenna	0.43	2	0.86	5%	0.90	Europa Clipper
HGA Assembly	31.51	1	31.51	15%	36.24	Europa Clipper (3.1-m diameter)
HGA Radome	0.62	1	0.62	15%	0.71	Europa Clipper (3.1-m diameter)
RF Mini Vault (under HGA)	16.00	1	16.00	15%	18.40	Europa Clipper
Ka-Band Hybrid	0.05	1	0.05	5%	0.05	Europa Clipper
X-Band Hybrid	0.03	1	0.03	5%	0.03	Europa Clipper
UHF Hybrid	0.05	1	0.05	5%	0.05	Europa Clipper
Ka-Band TWTA	1.05	2	2.10	10%	2.31	Europa Clipper
X-Band TWTA	1.20	2	2.40	10%	2.64	Europa Clipper
UHF SSPA	1.20	2	2.40	10%	2.64	DAVINCI
X-Band Isolator	0.53	2	1.06	5%	1.11	Europa Clipper
X-Band Diplexer	1.00	2	2.00	5%	2.10	Europa Clipper
UHF Diplexer	1.00	2	2.00	20%	2.40	Europa Clipper
X-Band Coaxial Transfer Switch	0.13	2	0.26	10%	0.29	Europa Clipper
UHF Coaxial Transfer Switch	0.13	1	0.13	10%	0.14	Europa Clipper
X-Band SP3T Switch	1.10	4	4.40	10%	4.84	Europa Clipper
UHF SP2T Switch	0.50	1	0.50	10%	0.55	Europa Clipper
X-band semi-rigid Coax	0.12	8	0.96	5%	1.01	Europa Clipper
X-Band Waveguide	0.18	1	0.18	10%	0.20	Europa Clipper
Ka-band semi-rigid Coax	0.12	2	0.24	5%	0.25	Europa Clipper
Ka-Band Waveguide	0.18	2	0.36	10%	0.40	Europa Clipper
Ka-Band Isolator	0.10	2	0.20	5%	0.21	Europa Clipper
UHF semi-rigid Coax	0.12	15	1.80	5%	1.89	Europa Clipper
Orbiter Subsystems Subtotal Dry Mass (Excluding Harness)			1341.58	--	1485.78	
Harness (SC Subsystems Subtotal Dry + Payload Dry) * 7%			98.14	12%	109.49	Average percentage based on prior mission actuals
Orbiter Bus Total Dry Mass			1439.7	11%	1595.3	

Table A3. Orbital Operation Power Profiles. This spacecraft bus power mode list was used for energy balance analyses (exclusive of payload power values listed in Table 5), corresponding to components in the MEL. Some components are only used in cruise or for orbit insertion and are shown for completeness.

Orbiter Bus Components (not including instrument power)		Hardware Parameters		Primary Operating Modes - Uranus Tour Orbits					Transient Mode Changes		
				DSN Tracking	Science Downlink	Science Imaging	Science Fixed Attitude	Battery Charge	OTM	SLEW	DSAT
Subsystem	Component	Unit Power (W)	Contin-gency	# in use	# in use	# in use	# in use	# in use	# in use	# in use	# in use
Prop.	Main Engines	79.4	10%	0	0	0	0	0			
	RCS Thrusters (7W per fuel/ox valve pair)	7.0	10%	0	0	0	0	0	4		4
	ACS Engine Valve Heaters	1.5	30%	4	4	4	4	4			

	HiPat heaters (flange, injector, etc)	0.0	30%	0	0	0	0	0			
	Latch Valve (1/4" Vacco)	18.0	3%	0	0	0	0	0			
	High-Pressure Latch Valve	45.8	3%	0	0	0	0	0			
	Pressure Transducer	0.9	3%	0	0	0	0	0	6		6
Avionics	IEM	29.8	10%	1	1	1	1	1			
	RIU	0.0	10%	8	8	8	8	8			
Electrical Power	Shunt Regulator Unit (SRU)	15.6	10%	1	1	1	1	1			
	Power Switching Unit (PSU)	19.0	10%	1	1	1	1	1			
Attitude Control	Star Tracker - Sensor Head	0.8	3%	2	2	2	2	2			
	Star Tracker - Electronics Unit	6.0	3%	1	1	1	1	1			
	Inertial Measurement Unit	15.0	3%	1	1	1	1	0			
	Sun Sensor - Electronics Unit	0.4	3%	1	1	1	1	1			
	Reaction Wheel Assembly	20.0	3%	1.6	1.6	3.2	1.6	1.6	4	4	4
Thermal Control	Thermal Pump and Controller	23.0	20%	1	1	1	1	1			
	Radiator Heater	5.0	20%	1	1	1	1	1			
RF Comm.	Frontier Radio - X-Band Rx	6.3	10%	0	0	1	1	1			
	Frontier Radio - X-Band Tx/Rx	10.9	10%	1	0	0	0	0			
	Frontier Radio - Ka-Band Tx, X-band Rx	11.5	10%	0	1	0	0	0			
	Frontier Radio - UHF Rx Only (Probe Only)	6.3	10%	0	0	0	0	0			
	Frontier Radio - UHF Tx/Rx (Probe Only)	10.9	10%	0	0	0	0	0			
	USO 1	0.6	15%	1	1	1	1	1			
	USO 2	0.6	15%	0.27	0.27	0.27	0.27	0.27			
	X-Band TWTA - Tx	100.0	10%	1	0	0	0	0			
	X-Band TWTA - Standby	50.0	10%	0	0	0	0	0			
	Ka-Band TWTA - Tx	100.0	10%	0	1	0	0	0			
	Ka-Band TWTA - Standby	50.0	0%	0	0	0	0	0			
	UHF SSPA - Tx (Probe Only)	50	10%	0	0	0	0	0			
	UHF SSPA - Standby (Probe Only)	10	10%	0	0	0	0	0			
	X-Band Coaxial Transfer Switch	0.1	10%	0	0	0	0	0			
	X-Band SP3T Switch	1.1	10%	0	0	0	0	0			
	X-Band SP3T Switch	1.1	10%	0	0	0	0	0			
	UHF SP2T Switch		10%	0	0	0	0	0			

CBE Load Power (W)			273	274	199	166	150	117	82	117
Contingency Power (W)			28	28	18	17	16	6	2	6
MEV Load Power (W)			301	302	216	182	166	122	85	122
MPV (Margined) Load Power (W)			391	392	281	237	216	159	110	159

Available RTG Power (W) (includes distribution losses)			320	320	320	320	320			

Power Margin (W)			-71	-72	39	83	104			

References

Report of the Aerocapture Demonstration Relevance Assessment Team (ADRAT) 2023. https://smd-cms.nasa.gov/wp-content/uploads/2023/07/adrat-final-report.pdf.

Report of the Autonomous Navigation Demonstration Relevance Assessment Team (ANDRAT) 2024. https://science.nasa.gov/wp-content/uploads/2024/09/andrat-final-report.pdf.

Amato, M., Haas, T., Simon, A. et al. 2024, Uranus Orbiter Probe Drivers and Trades. 21st International Planetary Probe Workshop. https://ntrs.nasa.gov/citations/20240007545.

Ambrosi, R. Williams, H., Watkinson, E., et. al. 2019, European Radioisotope Thermoelectric Generators RTGs, and Radioisotope Heater Units RHUs, for Space Science and Exploration. Space Sci. Rev. 215, 55. https://doi.org/10.1007/s11214-019-0623-9.

Ambrosi, R., Barco, A., Mesalam, R. et al. 2025, European Americium Radioisotope Power Systems, Missions and Applications. Proceedings of the American Nuclear Society. https://doi.org/10.13182/xyz-47347.

Anderson, R. 2024, Uranus Orbiter and Probe (UOP): Mission Architecting Challenges. Uranus Flagship Workshop: Investigating New Paradigms for Outer Planet Exploration. https://icegiants.jhuapl.edu/uploadedDocs/events/presentations/Day2_Session4_2_Anderson%20Uranus%20Workshop%20Final%20for%20posting.pdf

Asmar, S.W., French, R.G., Marouf, E.A. et al. 2018, PDS3 Cassini Radio Science User's Guide. https://doi.org/10.17189/1522471.

Benkhoff, J., Murakami, G., Baumjohann, W. et al. 2021, BepiColombo - Mission Overview and Science Goals. Space Sci. Rev. 217, 90. https://doi.org/10.1007/s11214-021-00861-4

Biswas, A., Srinivasan, M., Andrews, K., et al. 2025, Overview of the deep space optical communications DSOC, technology demonstration. Proc. SPIE 13355, Free-Space Laser Communications XXXVII, 133550J. https://doi.org/10.1117/12.3043966.

Burke, K.A. 2003, Fuel Cells for Space Science Applications. AIAA First International Energy Conversion Engineering Conference. https://doi.org/10.2514/6.2003-5938.

Cangahuala, L., Campagnola, S., Bradley, B. et al. 2025, Europa Clipper Mission Design, Mission Plan, and Navigation. Space Sci. Rev. 221, 22. https://doi.org/10.1007/s11214-025-01140-2.

D'Amario, L., Bright, L., Wolf, A. 1992, Galileo Trajectory Design. Space Sci. Rev. 60, 23-78. https://doi.org/10.1007/BF00216849.

de Pater, I., Gibbard, S., Hammel, H. 2006, Evolution of the dusty rings of Uranus. Icarus 180, 186-200. https://doi.org/10.1016/j.icarus.2005.08.011.

de Pater, I., Dunn, D.E., Stam, D.M. et al. 2013, Keck and VLT AO observations and models of the Uranian rings during the 2007 ring plane crossings. Icarus 226, 1399-1424. https://doi.org/10.1016/j.icarus.2013.08.001.

DellaGiustina, D.N., Nolan, M.C., Polit, A.T., et al. 2023, OSIRIS-APEX: An OSIRIS-REx Extended Mission to Asteroid Apophis. Planet. Sci. J. 4, 198. https://doi.org/10.3847/PSJ/acf75e.

Deshmuhk, R., Chadalavada, P., Dutta, S. Lugo, R., 2025, 6-DoF Uranus Aerocapture Trajectory Analysis. AIAA SCITECH 2025 Forum. https://doi.org/10.2514/6.2025-1510.

Dutta, S., Davis, W., Deshmukh, R., et. al. 2025, Aerocapture Solutions for Flagship-class Uranus Orbiter and Probe. AIAA SciTech Forum. https://doi.org/10.2514/6.2025-1508.

Dutta, S., Deshmukh, R., Hinkle, A., et al. 2026, Aerocapture Earth Demonstration Design to Support Future Planetary Mission Needs. SCITECH 2026 Forum. https://doi.org/10.2514/6.2026-0535.

Dutta, S., Shellabarger, E., Scoggis, J., et. al. 2024, Uranus Flagship-class Orbiter and Probe Using Aerocapture. AIAA SciTech Forum. https://doi.org/10.2514/6.2024-0714.

Ellison D. et al. 2025, Uranus Orbiter and Probe: System Capture and Orbital Operations, AAS/AIAA Astrodynamics Specialist Conference, Boston, MA, 2025. Paper AAS 25-669. https://www.researchgate.net/publication/397756025_Uranus_Orbiter_and_Probe_System_Capture_and_Orbital_Operations.

Englander, J., Ellison, D., Hatten, N., et al. 2025, Robust Access to Uranus via Solar Electric Propulsion. AIAA Journal of Spacecraft and Rockets, submitted

French, R., Elliot, J., Levine, S. 1986, Structure of the Uranian Rings: II. Ring orbits and widths. Icarus 67, 134-163. https://doi.org/10.1016/0019-1035(86)90181-8.

European Space Agency (ESA), 2019. CDF Study Report: CDF-187(C), Ice Giants: A Mission to the Ice Giants – Neptune and Uranus. https://sci.esa.int/web/future-missions-department/-/61307-cdf-study-report-ice-giants.

Gochenaur, D., Gentgen, C., de Weck O. 2025, Starship as an Enabling Option for a Uranus Flagship Mission. 2025 IEEE Aerospace Conference. https://doi.org/10.1109/AERO63441.2025.11068722.

Gomez-Delrio, A. and Dutta, S. 2024, Design Implications for Aerocapture Systems Placing Flagship-class Uranus Orbiters. AIAA SciTech Forum. https://doi.org/10.2514/6.2024-0953.

Gomez-Delrio, A., Davis, W., Deshmukh, R., Dutta, S. 2025, Design Considerations for Aerocapture Delivery of Uranus Orbiter and Probe. AIAA SciTech Forum. https://doi.org/10.2514/6.2025-1704.

Gonter, K., He, X., Williams, B. et al. 2025, Dragonfly: Detailed Thermal Control System Design. International Conference on Environmental Systems 2025. https://ttu-ir.tdl.org/items/f72cb81e-9a60-4dfd-aeba-cce91064c665.

Haws, T., Sunkara, D., Pepper, B., Holt, L. 2024, Space Launch System (SLS): Uniquely Enabling for Science Missions to the Outer Planets and Beyond. AIAA AVIATION FORUM and ASCEND 2024. https://doi.org/10.2514/6.2024-4811.

Hedman, M., de Pater, I., Cartwright, R., et al. 2025, Spectral trends across the rings and inner moons of Uranus and Neptune from JWST NIRCam images. Planet. Sci. J. 6, 204. https://doi.org/10.3847/PSJ/adf325.

Hirshorn, S. 2016, Expanded Guidance for NASA Systems Engineering. Volume 2: Crosscutting Topics, Special Topics, and Appendices (NASA/SP-2016-6105-SUPPL). https://ntrs.nasa.gov/citations/20170007239.

Hofstadter, M., Simon, A., Reh, K., et al. 2017, Ice Giants Pre-Decadal Survey Mission Study Report, JPL D-100520. http://arxiv.org/abs/2511.13946.

Holtzman, G., Coker, R., Ercol, C. et al. 2020, Dragonfly: Thermal Design Overview. International Conference on Environmental Systems 2020. https://ttu-ir.tdl.org/items/79364f8b-9721-4562-a029-3dee16adab8d.

Hubbard, W. B. 2010, Mission Concept Study: Ice Giants Decadal Study. https://sites.nationalacademies.org/cs/groups/ssbsite/documents/webpage/ssb_059323.pdf.

Karimi, R., Landau, D., Weinstein-Weiss, S., Elliott, J. 2025, Destination Uranus: Interplanetary and Capture Trajectory Architecture Analysis for a Flagship-Class Orbiter and Probe. AIAA SCITECH 2025 Forum. https://doi.org/10.2514/6.2025-2737.

Kowalkowski, T., Johannsen, J., Lam, T. 2008, Launch Period Development for the Juno Mission to Jupiter. AIAA 2008-7369. https://doi.org/10.2514/6.2008-7369.

Landau, D., Persinger, R., Karimi, R., Hofstadter, M., Castillo-Rogez, J., Mitchell, K. 2025, Uranus Cruise and Tour Design Impacts on Science, Cost, and Risk. 2025 IEEE Aerospace Conference. https://doi.org/10.1109/AERO63441.2025.11068400.

Mages, D., Restrepo, R., Chadalavada, P., et al. 2025, Mission Design and Navigation for Uranus Aerocapture Leveraging Super Heavy-Lift Launch Vehicles. AIAA SCITECH 2025 Forum. https://doi.org/10.2514/6.2025-1509.

McAdams, J., Scott, C., Guo, Y., Dankanich, J., Russell, R. 2011, Conceptual mission design of a polar Uranus orbiter and satellite tour. Spaceflight Mechanics 2011: Proceedings of the AAS/AIAA Space Flight Mechanics Meeting. Advances in Astronautical Sciences, vol 140.

McNutt, R., Ostdiek, P., et al. 2015, Nuclear Power Assessment Study Final Report. https://science.nasa.gov/wp-content/uploads/2024/02/npas-final-report.pdf.

McNutt, R., Vernon, S., Brandt, P., Paul, M., Lusthaus, R. 2024, High-speed scientific spacecraft launches with commercial launch vehicles. Acta Astronautica 217, 18-26. https://doi.org/10.1016/j.actaastro.2024.01.024.

Mesalam, R., Barco, A., Tute, R., et al. 2025a, Qualification Testing of the ESA Radioisotope Thermoelectric Generator. Proceedings of the American Nuclear Society. https://www.ans.org/pubs/proceedings/article-57948/.

Mesalam, R., Schmitz, P., Sargeant, H. et. al. 2025b, Americium Fuelled Radioisotope Stirling Generator for Lunar Surface Mobility Systems. Acta Astronautica 228, 331–345. https://doi.org/10.1016/j.actaastro.2024.12.001.

National Academies of Sciences, Engineering, and Medicine (NASEM), 2023, Origins, Worlds, and Life: A Decadal Strategy for Planetary Science and Astrobiology 2023-2032. Washington, DC: The National Academies Press. https://doi.org/10.17226/26522.

Oriti, S., Wozniak, E., Yang, M. 2023, Status of Dynamic Radioisotope Power System Development for NASA Missions. Proceedings of the American Nuclear Society. https://doi.org/10.13182/nets23-41855.

Paczkowski, B. and Ray, T. 2004, Cassini science planning process. Space Ops 2004 https://hdl.handle.net/2014/38415.

Ripley, R. and Goodloe, C. 2025, Venus Probe Architecture. 2025 IEEE Aerospace Conference. https://doi.org/10.1109/AERO63441.2025.11068583.

Sandifer, C. 2024, Outer Planets Assessment Group 2024: RPS Program Status Updates. https://ntrs.nasa.gov/citations/20240007816.

Sargeant, H., Oriti, S., Bell, K., et al. 2025, Technology Demonstrator of an Americium-Fuelled Radioisotope Stirling Generator. Proceedings of the American Nuclear Society. https://doi.org/10.13182/xyz-47395.

Simon, A., Cohen, I., Hedman, M., Hofstadter, M., Mandt, M., Nimmo, F. 2025, Uranus Flagship Science-Driven Tour Design: Community Input Poll. https://arxiv.org/abs/2505.05514.

Simon, A., Nimmo, F., Anderson, R. et al. 2021, Planetary Mission Concept Study for the 2023-2032 Decadal Survey: Uranus Orbiter & Probe UOP, Mission Concept Design Study Final Report. https://tinyurl.com/2p88fx4f.

SpaceX 2020, Starship Users Guide, Revision 1.0. https://spacex.relayto.com/e/starship-users-guide-37uiuuepbks0x.

Tofil, T., DeMicheal, T., Woytach, J, Fleurial, J.-P., Birch, J. Davis, S. 2023, The Next Generation Radioisotope Thermoelectric Generator Project -- Overview and Progress Status. Proceedings of the American Nuclear Society. https://doi.org/10.13182/nets23-41913.

Warshay, M. and Prokopius, P.R. 1990, The fuel cell in space: yesterday, today and tomorrow. Journal of Power Sources 29, 193-200. https://doi.org/10.1016/0378-7753(90)80019-A.

Weinstein-Wiess, S., Elliott, J., Freeman, A. et al. 2025, Uranus Orbiter and Probe: A Novel Approach to Meet the Challenges. 2025 IEEE Aerospace Conference. https://doi.org/10.1109/AERO63441.2025.11068554.

Figures

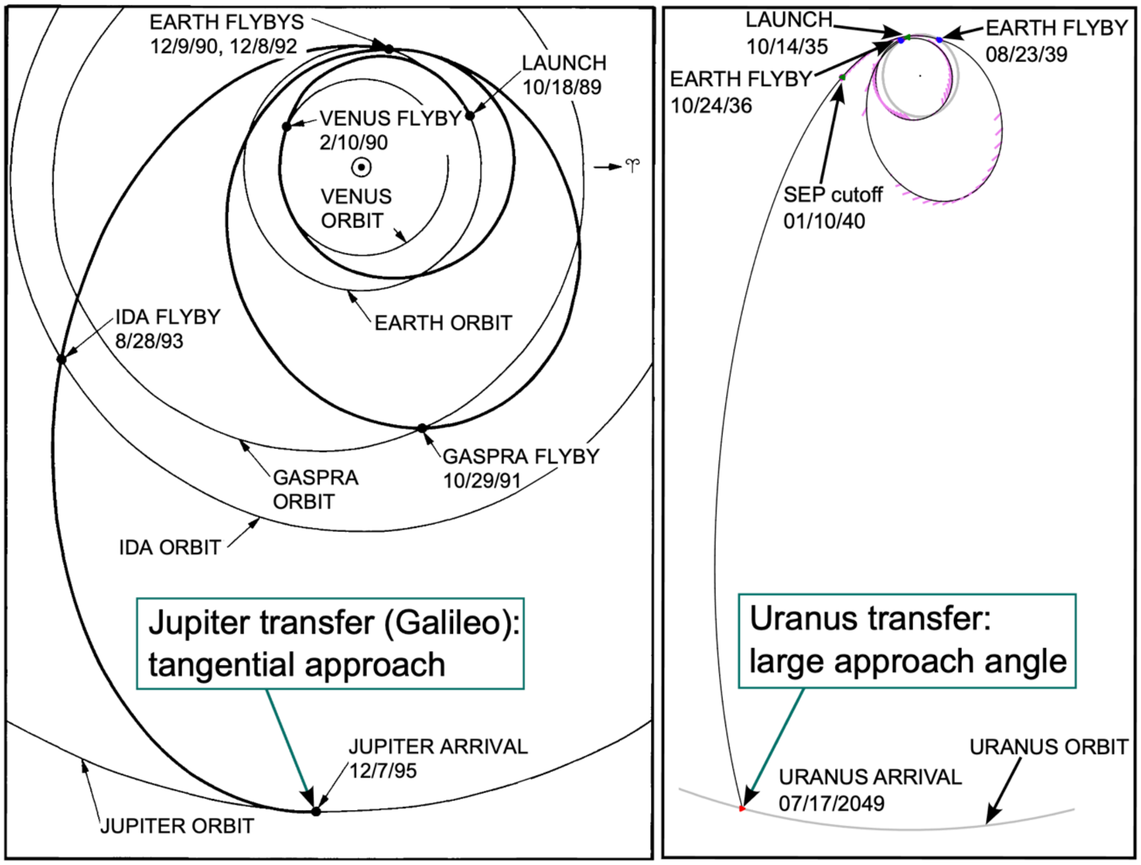

Figure 1: Orbit insertion approach angle comparisons. Unlike typical Jupiter arrival geometries (left) (Galileo, reproduced from L. D'Amario et al. 1992), Uranus approach (right) involves a large approach angle, increasing the magnitude of the required orbit insertion burn.

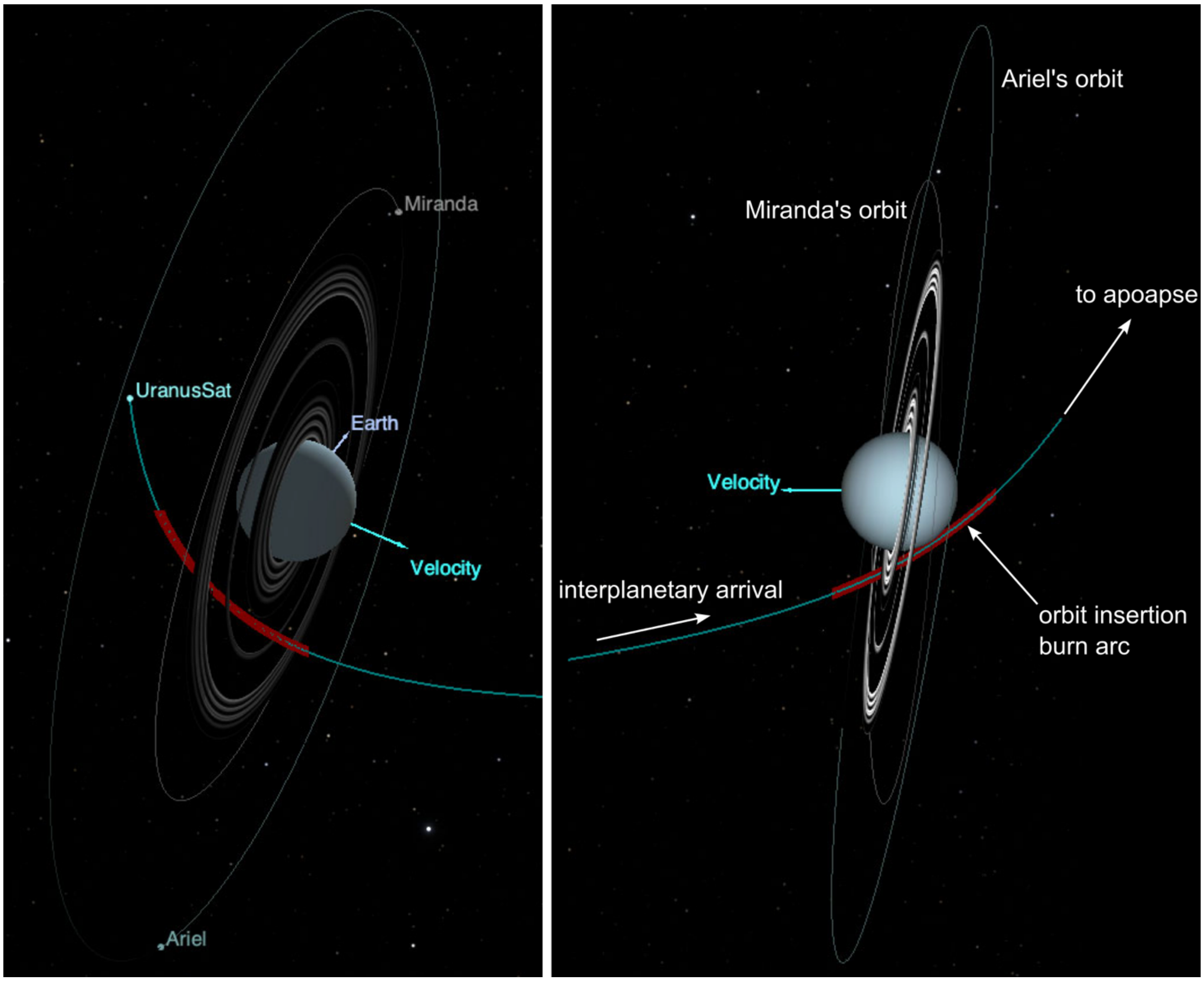

Figure 1. Example Uranus orbit insertion outside the zeta ring. Uranus orbit insertion maneuver traverses Uranus's ring plane in the ϵ-ν ring gap (left). Uranus orbit insertion (UOI) burn (1.62-hour duration, 1331 m/s) as viewed from Earth (right). While this example occurs in full view of ground stations, this is not a feature of all UOI geometries, where some portion of the burn is typically occulted with respect to Earth. Uranus's orbital velocity vector is indicated for reference. Reproduced from (D. Ellison et al. 2025).

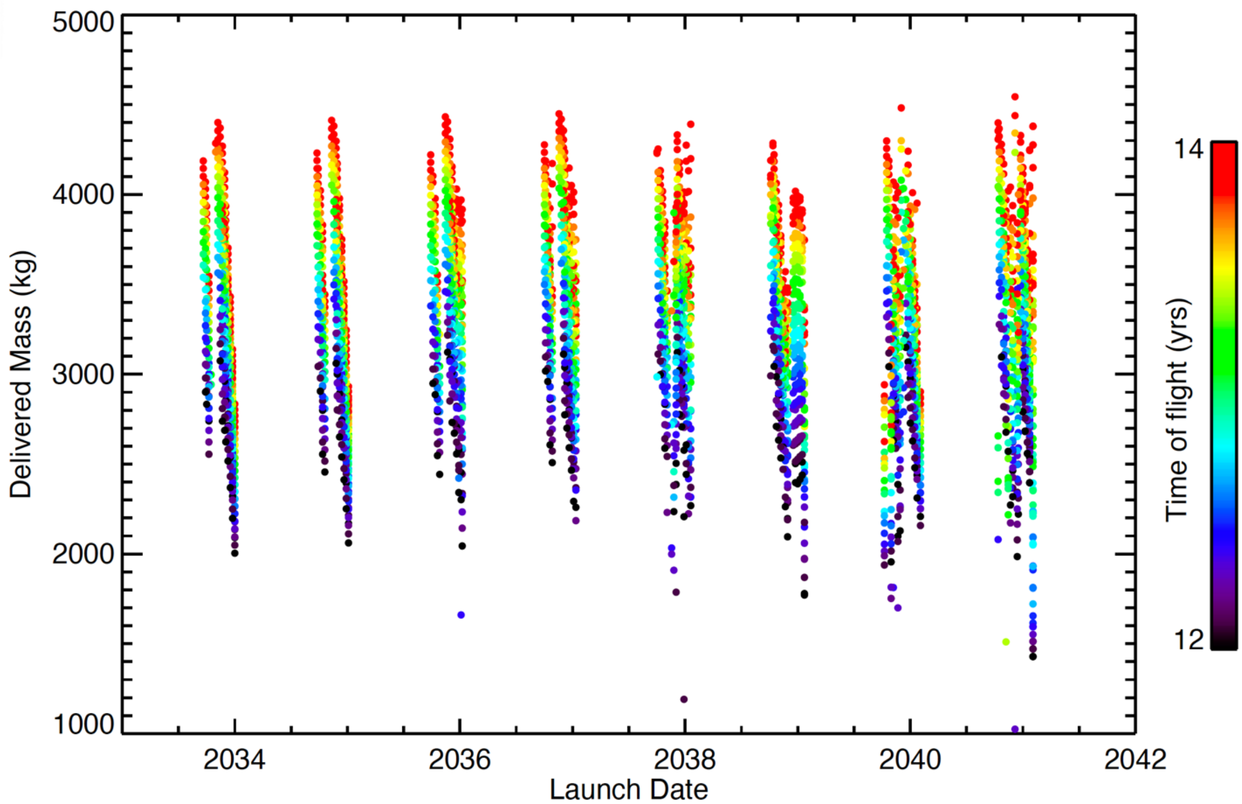

Figure 3. Deliverable mass for example SEP trajectories. For flight times of 12 to 14 years, sufficient mass can be inserted in Uranus orbit with repeatable launch windows; if less delivered mass is required, flight times can be reduced further. This example uses the Hall PPS500 thruster performance (Englander et al. 2026)

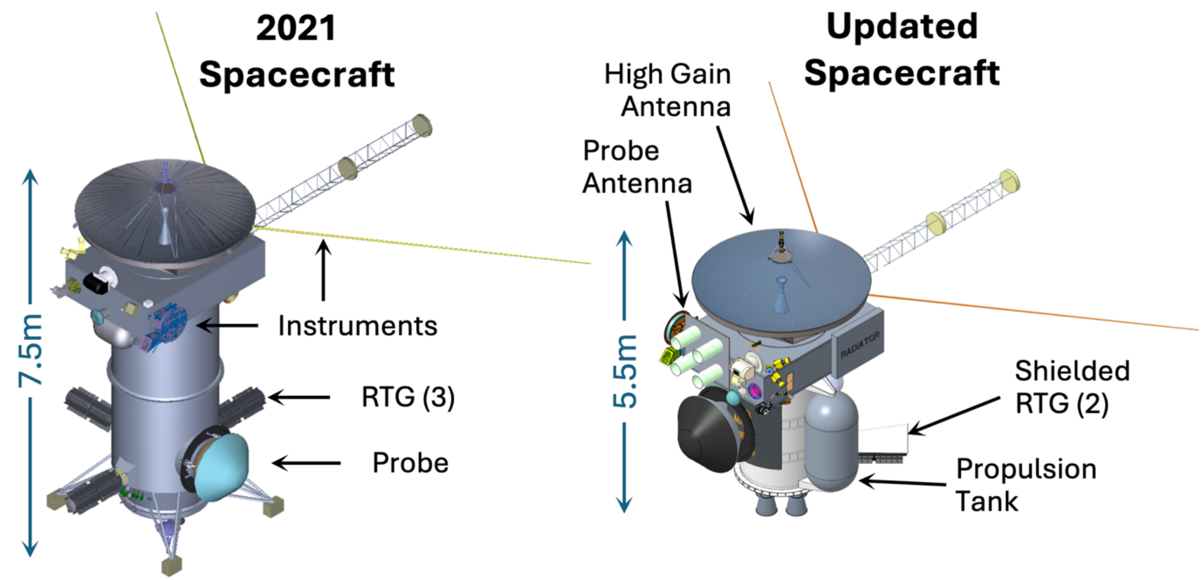

Figure 4. UOP spacecraft configurations in the 2021 Decadal (left) and updated (right) designs. The updated UOP spacecraft layout is significantly more compact and features two Next Gen RTGs rather than three.

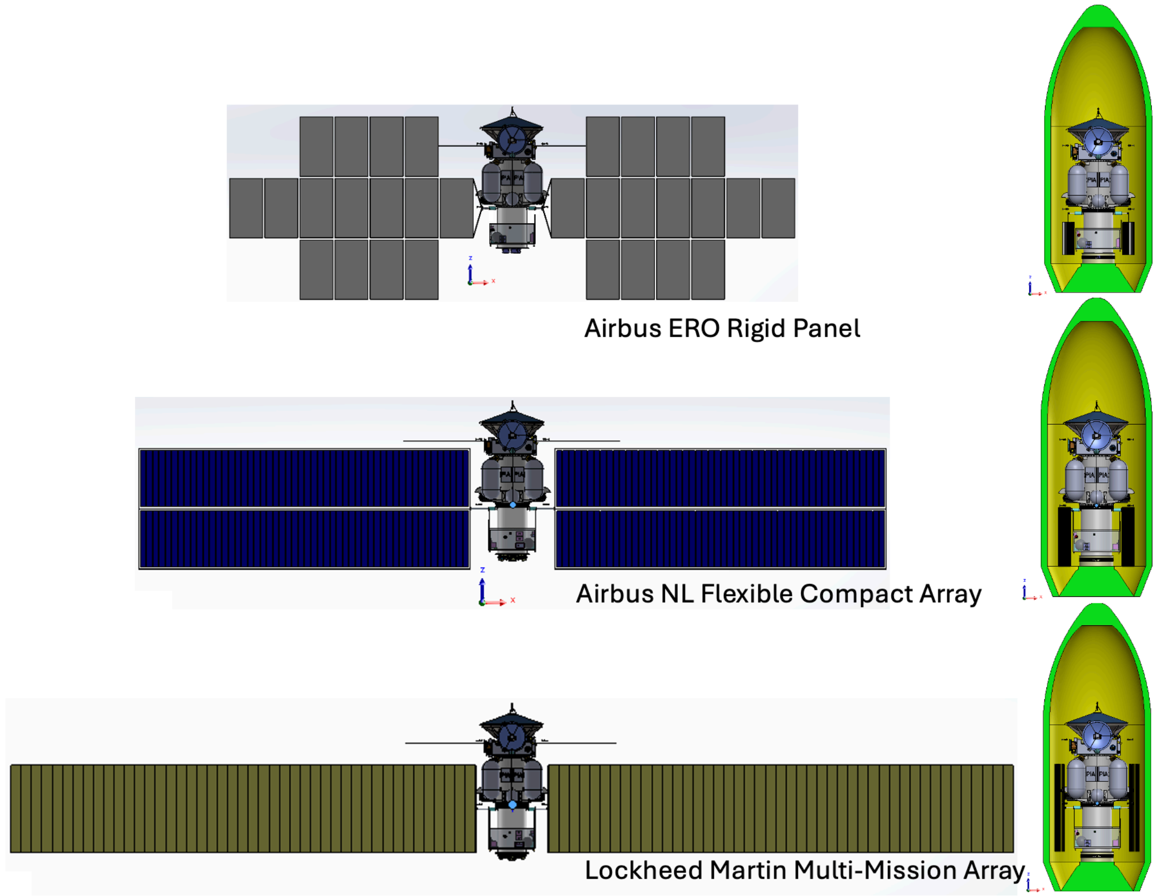

Figure 5. A UOP SEP stage proof of concept design. The updated UOP orbiter stacked with a SEP stage is shown in deployed (left) and stowed (right) configurations for multiple solar array alternatives, all generating 35 kW at 1 AU solar distance.

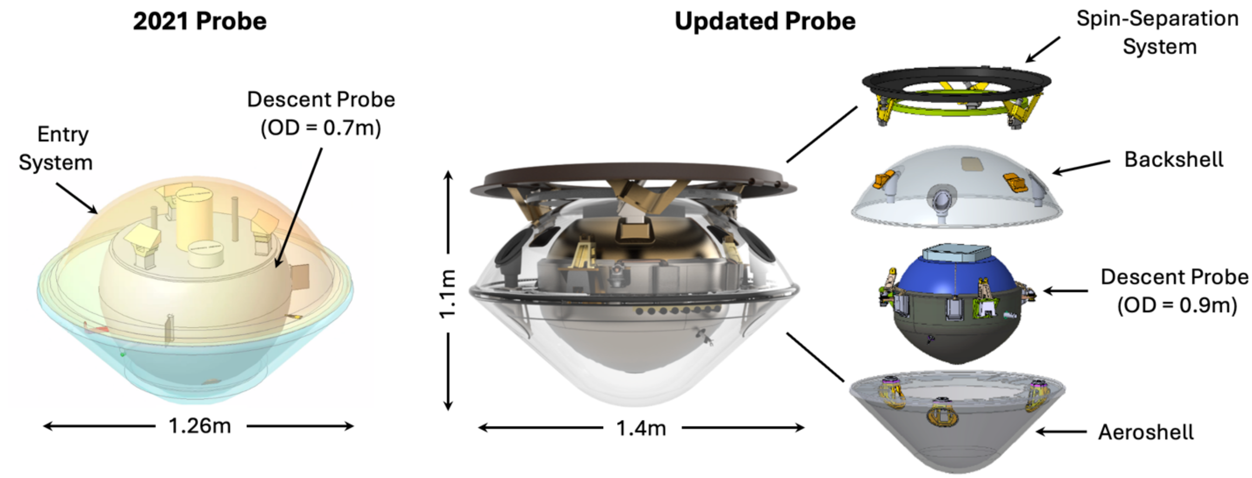

Figure 6. UOP probe configurations in the 2021 Decadal (left) and updated (right) designs. The newer design includes updated packaging and additional components, while accommodating more instruments, resulting in a larger outer diameter (OD) for the descent probe and entry system (M. Amato et al. 2024).

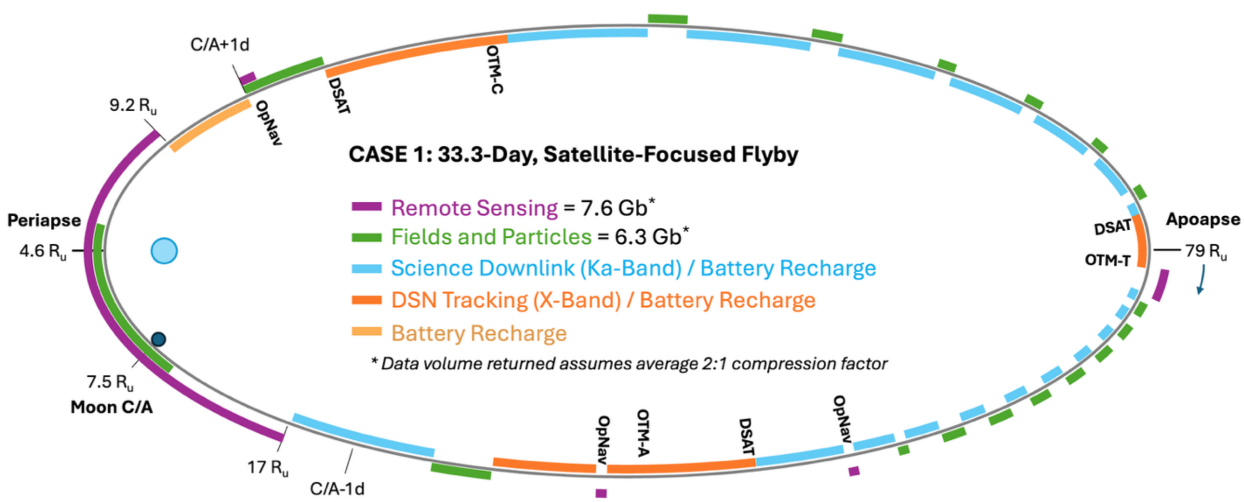

Figure 7. Concept of operations a satellite-focused orbit. Science observations were interleaved with downlinks, battery recharging, optical navigation (OpNav) images, and required orbital trim maneuvers (OTM) and reaction wheel desaturations (DSAT) while balancing the power and data volume available.

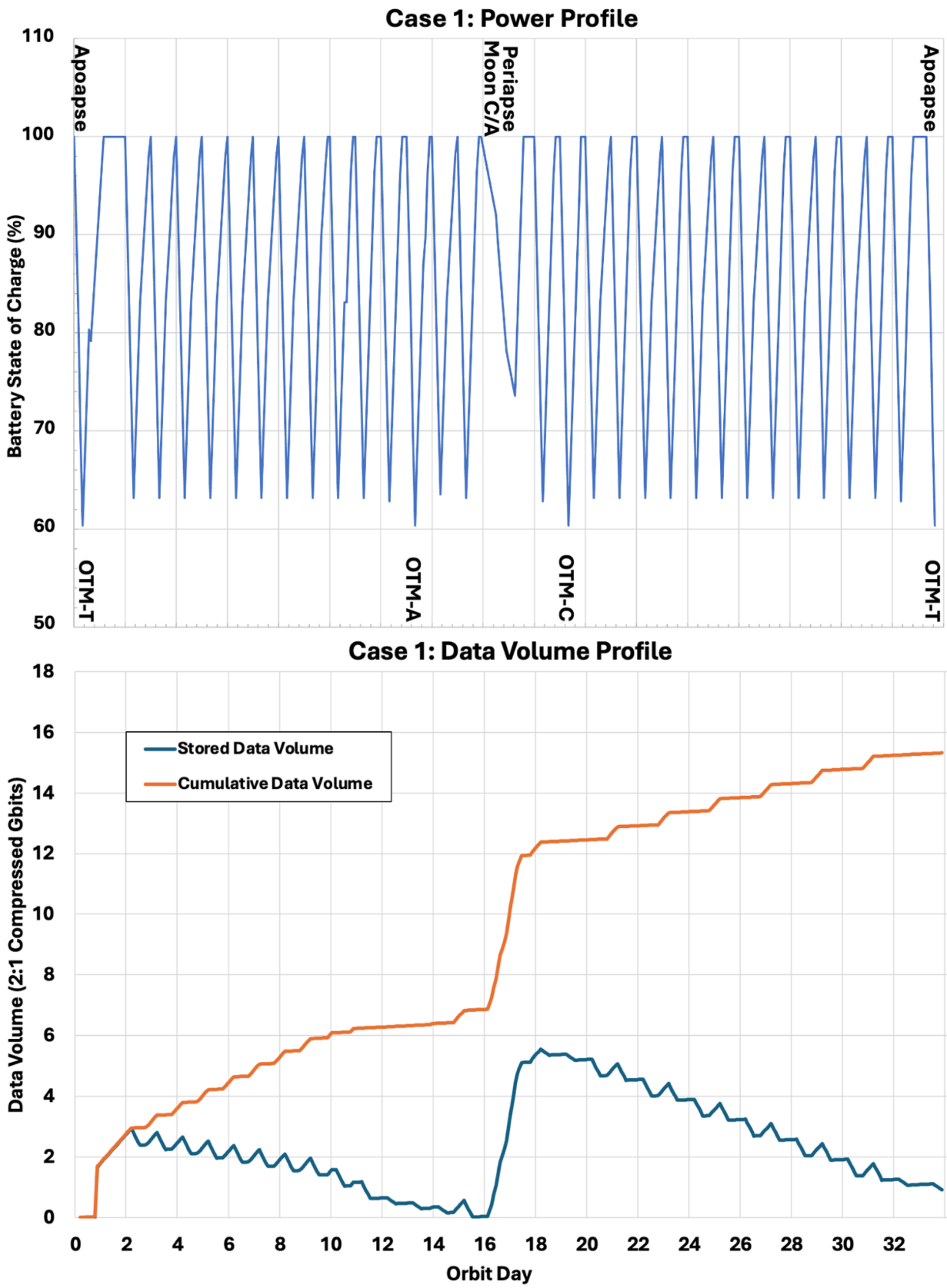

Figure 8. Power and data volume profiles for the case in Figure 7. Top: Power was limited to ~40% depth of discharge to maintain battery health. Bottom: Cumulative data volume (upper line) over the orbit includes remote sensing, fields and particles, OpNav images, and spacecraft telemetry. We required the majority of stored data volume backlog (lower line) to be cleared by the end of the orbit.

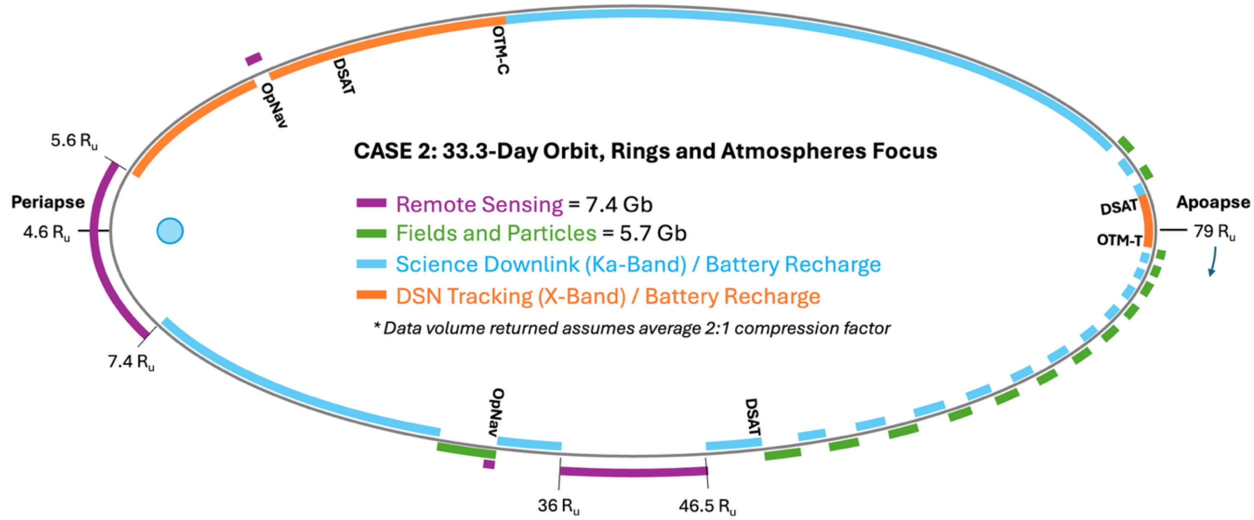

Figure 9. Same as Fig. 7, but for a rings and atmospheric science focused orbit.

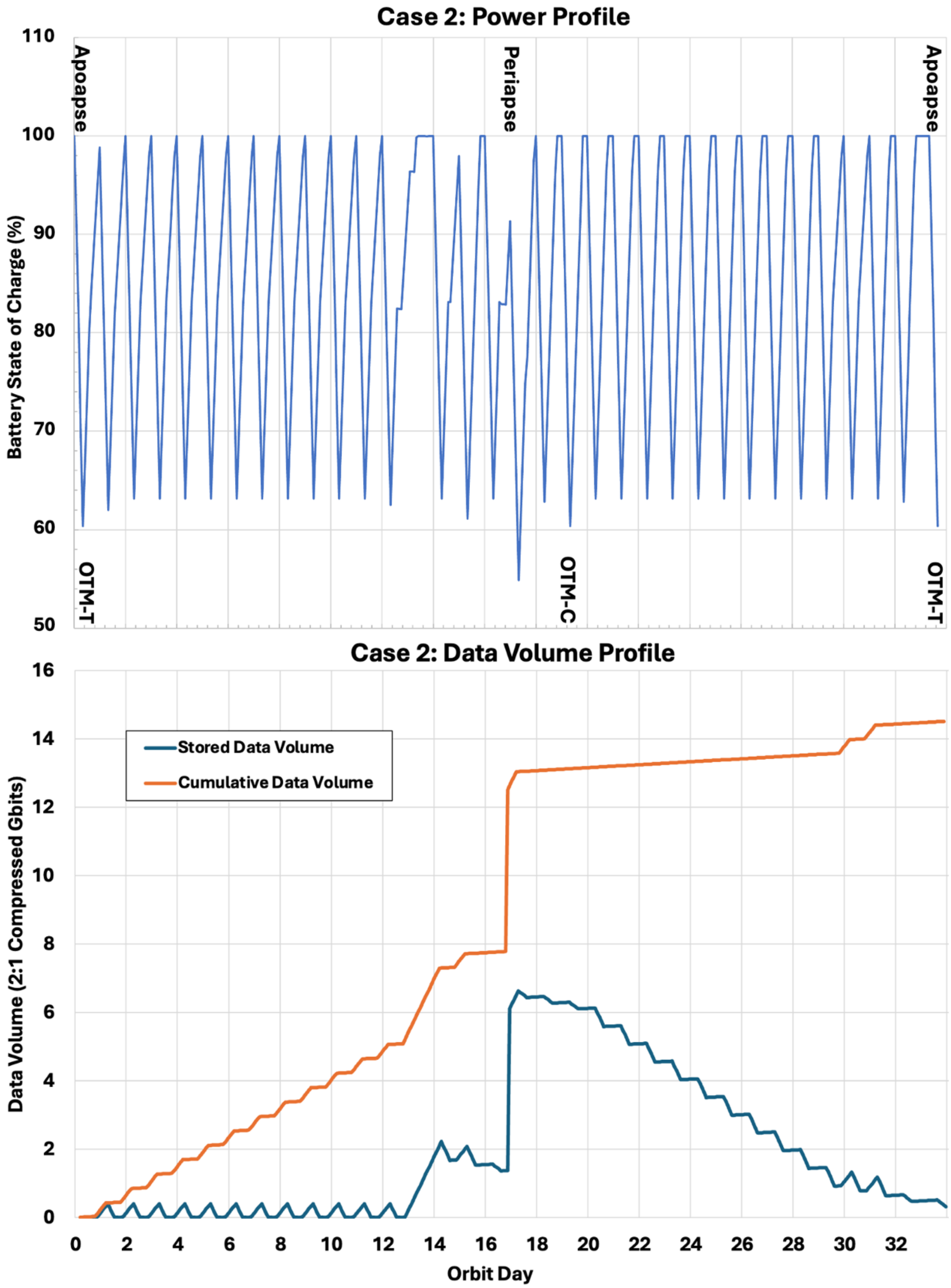

Figure 10. Power and data volume profiles for the case in Figure 9.

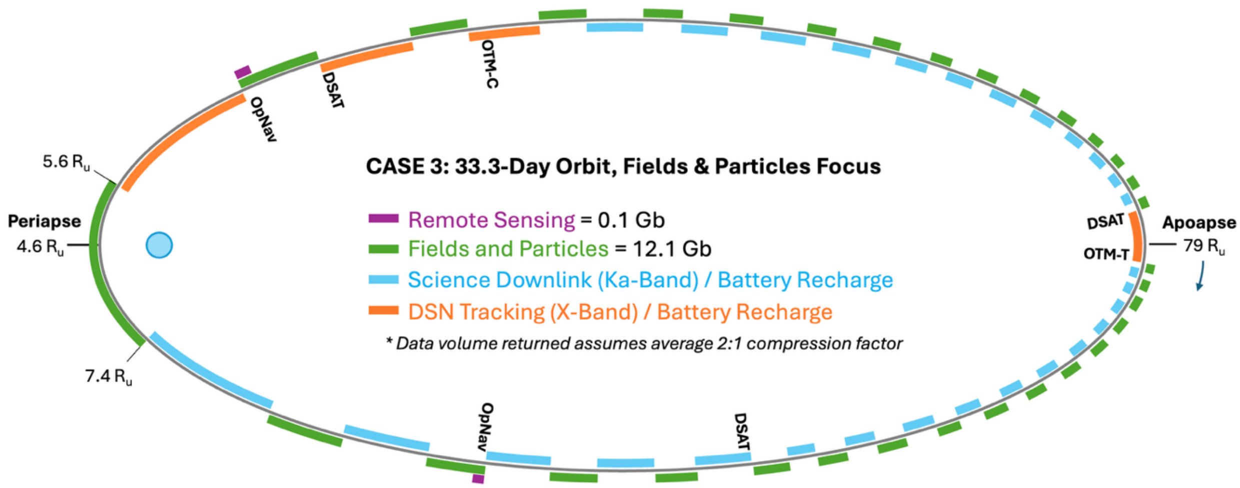

Figure 11. Same as Fig. 7, but for a field and particles focused orbit.

Figure 12. Power and data volume profiles for the case in Figure 11.