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Ice Giants Decadal Study

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Planetary Science Decadal Survey

Mission Concept Study Final Report

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Executive Summary

The purpose of this study was to define a preferred concept approach along with the risk/cost trade space for a Uranus or Neptune Mission launched in the 2020–2023 time frame and within a cost range of $1.5B–$1.9B in FY15$. The study was conducted by a team led by William Hubbard with members of both the Giant Planets and Satellites Panels working with the JHU/APL Space Department as the design center. NASA Glenn Research Center's COMPASS team made significant contributions as part of the design team in the areas of mission design, solar electric propulsion stage concept development, and Advanced Stirling Radioisotope Generator performance. NASA Langley Research Center also played a significant role in the areas of aerocapture evaluation and entry probe descent trajectory analysis. Georgia Institute of Technology was also a significant contributor to the mission design team in the area of developing the Uranus satellite tour.

Initial energy trades identified Uranus as more accessible and a lower risk option within the specified launch time frame ,and work on a Neptune option was dropped by the panel after a couple of weeks into the study. Since a Jupiter flyby option was not available during the launch years studied, a Neptune mission would have required unproven aerocapture technology or a cruise time well beyond 15 years.

A low-thrust solar electric propulsion trajectory option was developed to Uranus based on a single Earth gravity assist that could be repeated every year with a 21-day launch window. Using a launch on an Atlas V 531 and allowing a 13-year cruise time, a concept was developed that could accommodate both the floor and enhanced orbiter payload, perform atmospheric science with a fully equipped shallow entry probe, and perform multiple targeted flybys of each of the five Uranian satellites. No new technology is required with the exception of continued development of large parasol solar arrays (similar to Orion) to power the solar electric propulsion stage.

All science objectives (Tiers 1–3) could be technically achieved with compromises needed for viewing periapse from Earth and periapse altitude in order to satisfy the requirement of keeping the orbit trajectory safely outside of the rings. However, communications with Earth can be achieved shortly before and after periapse at a radius of 1.5 RU to help with achieving the gravity measurements. In all other areas, the objectives could fully be achieved.

Cost for the full mission (with enhanced payload, probe, and tour) was estimated at $1,894M, which falls within the cost range requirements of the study. Descope options would include removing the satellite tour, with an estimated cost savings of $26M mostly due to operations and Deep Space Network cost savings realized by reducing the mission by 14 months. The second option would be to reduce to the floor payload on both the probe and orbiter, with an estimated savings of $120M in instrument development cost. Finally, the entry probe could be descoped with a savings of $310M in probe development, integration and testing, and operations costs. These cost estimates assume FY15$ and reserves. Some further savings might be possible if the concept were optimized for the floor solution but would likely be of smaller impact.

Overall, the study has developed a concept that can achieve very robust science at Uranus at a cost below flagship mission levels and with minimal required technology development. Finally, significant descope options are available to ensure that the mission is affordable.

1. Scientific Objectives

Science Questions and Objectives

Uranus and Neptune represent a distinct class of planet. Their composition and interior structure are known much less well than those of the gas giants Jupiter and Saturn. While Jupiter and Saturn are composed mostly of hydrogen (more than 90% by mass) with hydrogen envelopes thought to extend all the way to relatively small rock/ice cores, molecular H_2 beginning a transition to ionized, metallic hydrogen at mega-bar pressures (Guillot 2005; Lissauer and Stevenson 2007), Uranus and Neptune possess much smaller hydrogen envelopes (less than 20% by mass) that never make the transition to metallic hydrogen (Guillot 2005). The bulk composition of these planets is dominated by much heavier elements, oxygen, carbon, nitrogen, and sulfur being the likely candidates based on cosmic abundances. Since these species are thought to have been incorporated into proto-planets primarily as ices, either as solids themselves or as gas trapped in a water-ice clathrate (Hersant et al. 2004), Uranus and Neptune are often referred to as “ice giants.” However, it is thought that there is currently very little ice in these planets, a supercritical fluid being the preferred phase of H_2O at depth.

Science Questions

There are many important science questions that can only be addressed by an orbiter at Uranus or Neptune to build upon data from Voyager 2, which was obtained with technology launched more than a generation ago, as well as more recent data from Earth-based telescopes. Results from an orbiter/probe at Uranus or Neptune, along with similar measurements from the Galileo probe/orbiter and the Cassini orbiter, will be used to constrain models of giant planet and Solar System formation and evolution. Knowledge of the ice-to-rock and ice-to-gas ratios, as well as the absolute abundance of species such as noble gases and water, provides information about the conditions in the planetary nebula and the planet formation process (Hersant et al. 2004). The extent to which the gas and heavier components are segregated or well mixed offers additional clues as to how and when each component was incorporated into the planet and how much mixing occurred, which strongly influences the chemical and thermal evolution of the planet. The ice giants are not only crucial for improving our understanding of the formation and evolution of the solar system but are also applicable to extra-solar planetary systems. In 2004, the first of many ice giant candidates was reported around another star (Butler et al. 2004); now ice giants significantly outnumber gas giants in extra-solar planetary systems.

Either Uranus or Neptune can provide the answers to fundamental questions about ice giants, and each has unique aspects worthy of study. Therefore, two mission studies were originally requested, one for Uranus and one for Neptune, with science and mission constraints being kept the same to the extent possible. The concept for each planet included two components: (1) an orbiter with limited instrument suite, and (2) a shallow atmospheric probe to descend to at least 1 bar and ideally 5 bars. The goal of performing two studies was to identify any requirements imposed by the Uranus or Neptune systems themselves that drive mission cost and/or risk. Early in the studies it became clear that a mission to Neptune entailed higher complexity and carried higher risks than one to Uranus, e.g., (1) Jupiter gravity assists were not considered because they are not always feasible during the range of time anticipated for the launch window, and without a Jupiter flyby, aerocapture is required for orbit insertion at Neptune; (2) aerocapture itself not only adds complexity and risk but also makes probe delivery and orbits that allow Triton encounters more challenging; and (3) a mission to Neptune also requires much longer lifetimes and thus higher risks for instruments and spacecraft components. As a result, it was decided to focus on Uranus alone for the remainder of the study.

Several aspects of Uranus are unique in the solar system:

1. Its low rate of internal heat emission (per unit mass, Uranus's heat flow is ~10 times lower than that of Neptune) suggests that much of the interior may not be convective and has correspondingly higher temperatures (Guillot 2005) (Figure 1-1).

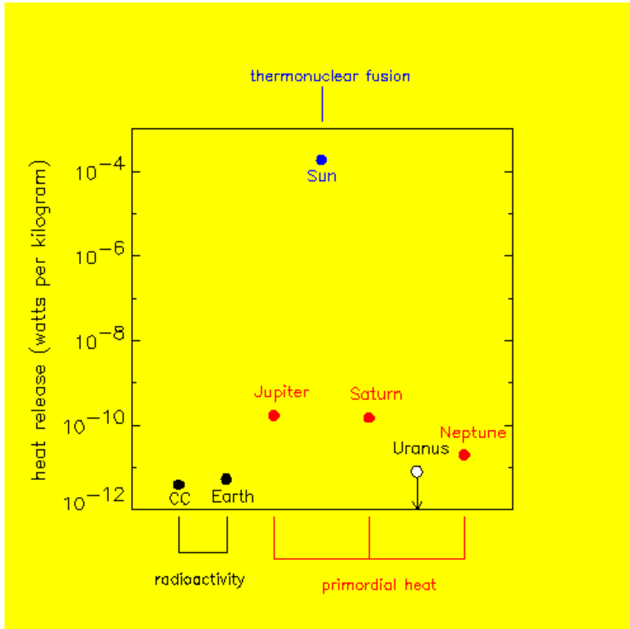

Figure 1-1. Internal heat emission of the ice giants.

2. Its large obliquity (98°) applies an unusual seasonal forcing to the atmosphere. The insolation differences between summer and winter are very large, but the annual averaged insolation as a function of latitude is more uniform than on any other giant planet, although on average the poles do receive more sunlight than the equator (Figure 1-2).

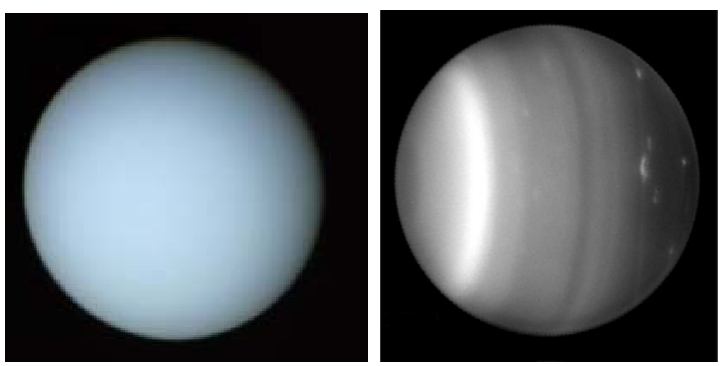

Figure 1-2. Uranus as seen by Voyager 2 in 1986 (left) and Keck in 2003 (right).

3. It has a strongly tilted (~60° from its rotation axis) dipole magnetic field, which, near solstice, leads to a unique geometry that offers an ideal opportunity to study the coupling between the solar wind and planetary atmospheres (Figure 1-3).

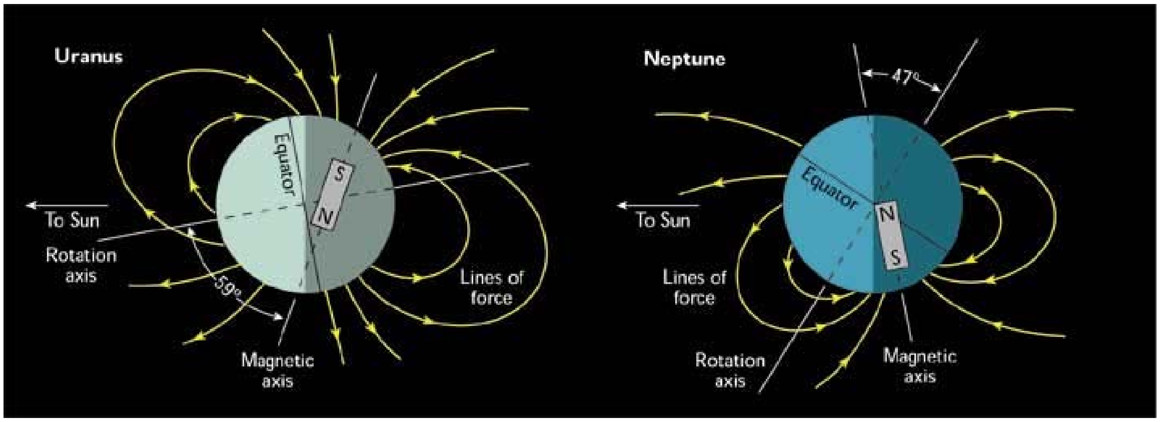

Figure 1-3. Magnetic and rotation axes of Uranus (left) and Neptune (right).

4. Its magnetic field also appears to be generated well outside the planet's core, as much as 70% of the way out from the planet's center.

Uranus's satellite and rings systems are also unique. Uranus lacks the large satellites found around Jupiter and Saturn, and the much lower temperatures allow for different surface ices. Given the different formation and evolutionary paths of ice-giant as opposed to gas-giant systems, the Uranian satellites will shed light on the formation and conditions of the early solar system. The largest Uranian satellites, Titania and Oberon, may have deep interior oceans (Hussmann et al. 2006), so measurements to detect induced magnetic fields of the satellites are desired. Uranus is particularly suited to this type of investigation because its inclined dipole (~60°) induces large field variations in the satellites as the planet rotates. Despite Uranus's distance from the Sun and the satellites' small sizes, there is evidence for significant tectonic activity, especially on small Ariel and smaller Miranda. In the case of Ariel, there is also evidence for viscous cryovolcanic flows, which have not been detected elsewhere in the solar system. Some of the smaller Uranian satellites are also dynamically interacting and perhaps unstable, with orbital changes seen between the 1980s and today (Showalter and Lissauer 2006) (Figure 1-4).

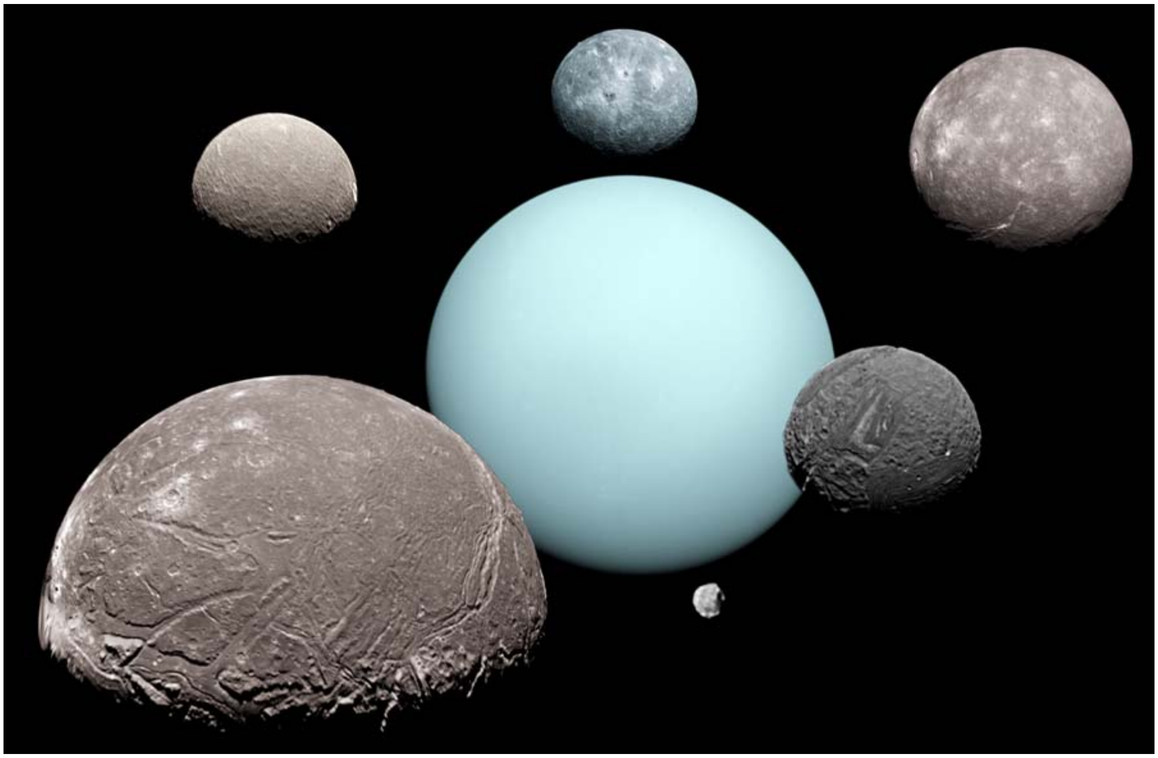

Figure 1-4. Uranus and some of its moons.

Uranus's ring system consists of 13 distinct rings: 9 narrow dense rings and 2 dusty rings in the inner ring system, as well as 2 tenuous dusty outer rings. Faint dust bands are also interspersed between the rings in the inner ring system. The narrow dense rings may highlight dynamical processes not seen in Saturn’s wider rings or in Jupiter’s and Neptune’s under-dense ones. The rings also serve as a probe of Uranus’s gravitational field, providing information about equatorial higher-order terms. There are also several as yet unexplained aspects of the Uranian rings, e.g., a lack of centimeter-sized particles (French et al. 1991), the apparent dynamical instability of some rings and moons (Showalter and Lissauer 2006), the blue color of the newly discovered μ ring (de Pater et al. 2006), and changes since the Voyager encounter of 1986 (de Pater et al. 2007), which may shed new light on ring physics and evolution.

Prioritized Science Objectives

Tier 1 Science Objectives: Orbiter

- *Determine the atmospheric zonal winds, composition, and structure at high spatial resolution, as well as the temporal evolution of atmospheric dynamics.*

Measurements: Requirements should include dayside imaging with ~15-km/pixel resolution. Imaging should include at least one sequence of repeated coverage of the same region at ~2-hour intervals for cloud tracking (necessary to obtain winds, divergence, and vorticity) with 2-m/s accuracy. Wavelengths should include visible and/or near-IR continuum (e.g., out to 2.5 μm) as well as one or more methane bands (e.g., 0.889 μm and another in the near-IR, e.g., 2.3 μm). Imaging strategy must characterize behavior over a range of timescales, including short (1–3 days), medium (~1 month), and long (~1 year) variability. Global or near-global daily coverage for periods of weeks to months is desired. To search for lightning, requirements should include multiple views of all latitudes on the nightside (15 to 100 km/pixel) with clear filter imaging combined with imaging of discrete thunderstorms on the dayside. Measurements require repeated imaging while tracking a feature (usually near 90° phase).

These measurement requirements can be achieved by the mission with the floor payload: the wide-angle camera (WAC) can perform dayside imaging with pixel scales better than 15 km within 2.7 hours of closest approach and tracking of features on the dayside can be achieved at pixel scales better than 100 km within 30 hours of periapse; the visible/near-IR (Vis/NIR) mapping spectrometer can achieve the same pixel scales within 2.5 and 27 hours of periapse, respectively; the WAC can accommodate up to 12 spectral filters between 395 and 1040 nm and the mapping spectrometer has seven spectral bands covering 400 nm to 2.25 µm; near closest approach it is possible to acquire a full-disk mosaic of the nightside, a WAC with a wider-angle field of view would reduce the number of frames necessary to cover the disk, trading this coverage for lower resolution; the planned downlink strategy returns over 1000 WAC images and a few hundred images cubes from the mapping spectrometer per orbit, which is sufficient to accomplish closest approach and more distant monitoring observations.(Further details of the instrumentation are provided in Section 3, Technical Overview, under the Instrument Payload Description.)

- *Understand the basic structure of the planet's magnetosphere as well as the high-order structure and temporal evolution of the planet's interior dynamo.*

Measurements: To understand the structure, composition, and stress balance of the planet's magnetosphere and to determine how magnetic flux is transported in the planet's magnetosphere, requirements should include making continuous measurements of the vector magnetic field (1-s resolution). To study the interactions between the planet's magnetosphere and any large satellites (including any intrinsic satellite magnetic field), requirements include determining the trapped and precipitating fluxes of electrons with energies between 10 eV and 10 MeV, 15° angular resolution or higher, $dE/E = 0.1$, and time resolution of 1 minute or higher.

The orbital tour designed provides appropriate coverage of Uranus's magnetic field in time (details of the magnetometer are provided in Section 3, Technical Overview, under the Instrument Payload Description) and longitude (see figures under the section Concept of Operations and Mission Design). The altitude of periapse is limited to ~1.3 R_U by the need to keep the spacecraft outside of the ring system (>52,000 km) during ring-plane crossing. It would be possible to lower periapse significantly by crossing the ring-plane inside the rings, which might be possible late in the mission if the hazard posed to the spacecraft in this area can be determined by remote sensing of the rings (ring observations are a Tier-3 objective). Periapse occurs over the southern hemisphere. Unfortunately, changing the orbit such that periapse would occur over the northern hemisphere would be prohibitively expensive in terms of delta-V. If it were possible, before the mission, to improve knowledge of the environment between the rings and the planet and to demonstrate that spacecraft transit would be safe, a different orbit insertion scenario is allowed that would result in a periapse near the equator enabling better latitudinal coverage (see Appendix E, page 22).

The satellite tour designed (see section, Concept of Operations and Mission Design) includes at least two close (C/A = 50 km) encounters of each major satellite, making detection of induced or intrinsic magnetic fields possible.

Tier 2 Science Objectives: Enhanced Orbiter + Probe

- *Determine the noble gas abundances (He, Ne, Ar, Kr, and Xe) and isotopic ratios of H, C, N, and O in the planet's atmosphere and the atmospheric structure at the probe descent location.*

Measurement requirements: Probe sampling profile should satisfy the following requirements:

Pressure—two measurements: static and dynamic pressure.

Temperature—two measurements (along and cross flow). The minimum temperature profile (once at subsonic speeds) is a sample every tenth of a scale height (or about one sample every 5 km).

Acceleration—sample rate better than 50 Hz on entry (depends on entry speed and deceleration rates) and then better than 10 Hz once the probe's velocity is reduced to subsonic speeds. From an atmosphere structure standpoint, a desired sample rate is as follows:

2× *z*-accels (high and low gain): 12 bit × 50 Hz supersonic, and 10 Hz subsonic

2× *x-y* accels: 12 bit × 50 Hz supersonic, and 10 Hz subsonic

3× *x-y-z* gyros (role/pitch/yaw): 12 bit × 50 Hz supersonic, and 10 Hz subsonic

The atmospheric probe instrumentation (floor payload) is designed to satisfy these objectives. Data will be acquired over a 1-hour period as the probe descends from 0.1 to 5 bars. Details of the floor and enhanced payloads for the probe are provided in Section 3, Technical Overview, under the Instrument Payload Description, and the probe descent is discussed under the section, Concept of Operations and Mission Design.

- *Determine internal mass distribution.*

Measurement requirements: Special consideration should be given to driving the orbit such that internal gravity measurements can be performed by observing periapses from Earth.

In the course of this study it has been demonstrated that at a range of 1.1 R_U, there is sufficient drag to degrade the gravity measurements. It has also been demonstrated that periapse altitude is limited to ~1.3 R_U by the need to keep the spacecraft outside of the ring system (>52,000 km) during ring-plane crossing. Moreover, periapse occurs while the spacecraft is eclipsed by Uranus. With a periapse of 1.3 R_U, the spacecraft can be tracked to altitudes of ~1.5 R_U outside of eclipse. With this limitation, the mission can make modest improvements in the precision of Uranus's mass and J2 term and a marginal improvement on the existing J4 value; however, it will not be possible to determine J6 and higher terms.

A possible mitigation strategy would be to lower periapse significantly by crossing the ring-plane inside the rings. This scenario might be possible later in the mission if the hazard posed to the spacecraft in this area can be determined by remote sensing of the rings to be sufficiently low (ring observations are a Tier-3 objective). If by making periapse as low as possible (above the altitude at which heating by the atmosphere becomes a concern, which is known to be lower than 1.1 R_U), the spacecraft can be tracked to ~1.2 R_U and J6 may become measurable.

Unfortunately, rotating the orbit such that periapse is visible from Earth was found to be prohibitively expensive in terms of delta-V. However, if it were possible to improve on our knowledge of the environment between the rings before this mission and to demonstrate that spacecraft transit would be safe, a different orbit insertion scenario is allowed that would result in a visible periapse near the equator, enabling tracking closer to the planet to improve the gravity measurements (see Appendix E, page 22).

- *Determine horizontal distribution of atmospheric thermal emission, as well as the upper atmospheric thermal structure and changes with time and location at low resolution.*

Measurement requirements: To assess the thermal emission of the planet, repeated global nadir thermal mapping is required in the 7- to 50-μm spectral range to obtain 80- to 700-mbar (troposphere) and 0.5- to 20-mbar (stratosphere) temperatures to an absolute accuracy of 1.0 K if possible, relative accuracy of 0.4 K; spatial resolution of 100-km/pixel; limb viewing geometry to achieve 10- to 20-km altitude resolution at a wide range of latitudes.

These objectives are met by the mid-IR thermal detector (0.3–400 μm) and the UV spectrograph in the enhanced payload. Pixel scales of 100 km can be achieved within 15 and 76 hours of closest approach, respectively; 20-km pixel scales will be possible within a few and several hours of closest approach, respectively. Details of the instrumentation are provided in Section 3, Technical Overview, under the Instrument Payload Description.

- *Remote sensing observations of large satellites.*

Several large-satellite encounters also would be of interest if they do not preclude the internal gravity measurements.

A satellite tour was developed that allows two targeted flybys of each of the five major satellites (Miranda, Ariel, Umbriel, Titania, and Oberon) at closest approaches of 50 km. Although this mission will arrive near solstice, as did Voyager 2, it will be the opposite season. Therefore, mapping of the previously unseen *northern hemispheres of the satellites will be possible (see Appendix E, pages 42–53). During the tour, observations can be concentrated during a satellite flyby and the data can be downlinked over the remainder of the orbit.*

The floor payload instrumentation WAC, Vis/NIR mapping spectrometer, and magnetometers are sufficient to accomplish fundamental remote sensing observations as well as to detect induced magnetic fields that would be indicative of interior oceans.

Were the satellite tour not performed, the narrow angle camera (NAC) under the enhanced instrument suite, would enable significant satellite science to be accomplished; even remotely, mapping at better than a few kilometer pixel scale would be possible (at apoapse, ~60 R_U, the pixel scale of the NAC is ~12 km). (Details of the instrumentation are provided in Section 3, Technical Overview, under the Instrument Payload Description.)

Tier 3 Science Objectives: Enhanced Orbiter + Enhanced Probe

- *Measure the magnetic field, plasma, and currents to determine how the tilted/offset/rotating magnetosphere interacts with the solar wind over time.*

For magnetospheric observations, desired requirements might include continuous measurements of plasma and energetic charged particles (10 eV to 10 MeV), with full sky coverage and an angular resolution of 15°, dE/E = 0.1, and 10-s time resolution or higher.

These objectives are satisfied by the plasma instruments in the enhanced orbiter payload (see Section 3, Technical Overview, under the Instrument Payload Description).

- *Remote sensing observations of small satellites and rings.*

Observations of the rings and small satellites can be carried out with the orbiter's floor payload. The NAC under the enhanced instrument suite, in particular, would enable significant observations to be accomplished even remotely, imaging at better than a few kilometer pixel scale would be possible (at apoapse, ~60 R_U, the pixel scale of the NAC is ~12 km).

Note that observations of the ring system and the environment inside the orbit of the innermost rings to understand the hazard that would be posed to the spacecraft might reveal that it would be possible to safely lower the ring-plane crossing to inside the rings, thereby making it possible to lower periapse significantly and improve the Tier-2 gravity science.

- *Determine the vertical profile of zonal winds as a function of depth in the atmosphere, in addition to the location, density, and composition of clouds as a function of depth in the atmosphere.*

The enhancements to the probe payload, ultra-stable oscillator (USO) and nephelometer, satisfy these objectives (see Section 3, Technical Overview, under the Instrument Payload Description

Summary

The mission described here is able to meet these science objectives, although the extent to which information on the gravity field and thus the internal mass distribution (a Tier-2 objective above) is limited for a number of reasons. (1) Uranus's exosphere is sufficiently extended that at an altitude of 1.1 R_U, the drag forces exerted by the atmosphere would degrade the gravity measurement. (2) Periapse will occur over the anti-Earth hemisphere; thus tracking will only be possible before and after the spacecraft is occulted by Uranus. The delta-V that would be needed to rotate the orbit such that periapse would be visible is prohibitively high. (3) The periapse altitude is limited to 1.3 R_U by the need to keep the spacecraft outside of the ring system (>52,000 km) during ring-plane crossing. Combined with the time spent in eclipse, this constraint limits spacecraft tracking to >1.4 R_U. It would be possible to lower periapse significantly (<1.1 R_U) by crossing the ring-plane inside the rings, which may prove to be possible late in the mission if the hazard posed to the spacecraft in this area can be determined by remote sensing of the rings (ring observations are a Tier-3 objective). If it were possible, before the mission, to improve knowledge of the environment between the rings and the planet and to demonstrate that spacecraft transit would be safe, a different orbit insertion scenario is allowed that would result in a periapse near the equator, enabling better latitudinal coverage (see Appendix E, page 22).

Science Traceability

The instrument and mission functional requirements derived from each of the Tier 1–3 science objectives are given in the traceability matrix (Table 1-1).

Table 1-1. Science traceability matrix.

Tier 1 Science Objective	Measurement	Instrument	Functional Requirement
Determine the atmospheric zonal winds, composition, and structure at high spatial resolution, as well as the temporal evolution of atmospheric dynamics.	*Spatially resolved scattering properties of the atmosphere*	*Wide-angle visible imager; visible/near-infrared mapping spectrometer*	*Orbiter with 2-year tour*
Understand the basic structure of the planet’s magnetosphere as well as the high-order structure and temporal evolution of the planet's interior dynamo.	*Make continuous measurements of vector magnetic field (1-s resolution)*	*Magnetometer*	*Orbiter*
Tier 2 Science Objective	**Measurement**	**Instrument**	**Functional Requirement**
Determine the noble gas abundances (He, Ne, Ar, Kr, and Xe) and isotopic ratios of H, C, N, and O in the planet’s atmosphere and the atmospheric structure at the probe descent location.	*Measure the noble gases and isotopic ratios in the atmosphere assuming that the atmosphere is well mixed, and measure temperature and pressure as a function of depth*	*Mass spectrometer with sufficient resolution; pressure-temperature sensors sampling at frequent intervals*	*Single probe that reaches at least 1 bar with a stretch goal of below 5 bars*
Determine internal mass distribution.	*Measure higher-order gravitational harmonics from orbiter gravitational moments*	*USO*	*Orbital periapsis visible from Earth*
Determine horizontal distribution of atmospheric thermal emission, as well as the upper atmospheric thermal structure and changes with time and location at low resolution.	*Measure temperature as a function of latitude and longitude, from direct thermal emission and/or UV occultations*	*Mid-infrared thermal detector; UV imaging spectrograph*	*Orbiter*
Remote sensing of large satellites	*Spatially resolved surface reflectance spectroscopy*	*Wide-angle visible imager; visible/near-infrared mapping spectrometer; narrow-angle camera*	*Orbiter with 2-year tour*

Tier 3 Science Objective	Measurement	Instrument	Functional Requirement
Measure the magnetic field, plasma, and currents to determine how the tilted/offset/rotating magnetosphere interacts with the solar wind over time.	*Measurements of plasma and energetic charged particles*	*Plasma and particle instrument*	*Orbiter*
Remote sensing of small satellites and rings	*Spatially resolved surface reflectance spectroscopy*	*Wide-angle visible imager; visible/near-infrared mapping spectrometer; narrow-angle camera*	*Orbiter with 2-year tour*
Determine the vertical profile of zonal winds as a function of depth in the atmosphere, in addition to the presence of clouds as a function of depth in the atmosphere.	*Measure the zonal winds as a function of depth, and determine scattering properties of the atmosphere*	*Ultra-stable oscillator on probe and on carrier; nephelometer*	*Single probe that reaches at least 1 bar with a stretch goal of below 5 bars*

2. High-Level Mission Concept

Study Request and Concept Maturity Level

The objective of this study was to conduct mission studies for both Uranus and Neptune with nearly identical science objectives and constraints. The science team provided three tiers of objectives (described in Section 1) along with floor and enhanced instrument payload options, with the goal of identifying any system-specific requirements that drive cost and risk. This study was conducted at a Concept Maturity Level (CML) of 4 (Table 2-1). The result of this study was to evaluate the trade space and develop a preferred point design that achieved at least the floor mission at either Uranus or Neptune within cost range of $1.5B–$1.9B in FY15$. Secondary goals were to clarify the costs associated with enhancements to the floor mission that could be added (with costs identified) to the mission within the cost cap.

Table 2-1. Concept maturity level definitions.

Concept Maturity Level	Definition	Attributes
CML 6	*Final Implementation Concept*	*Requirements trace and schedule to subsystem level, grassroots cost, verification and validation (V&V) approach for key areas*
CML 5	*Initial Implementation Concept*	*Detailed science traceability, defined relationships and dependencies: partnering, heritage, technology, key risks and mitigations, system make/buy*
CML 4	*Preferred Design Point*	*Point design to subsystem level mass, power, performance, cost, risk*
CML 3	*Trade Space*	*Architectures and objectives trade space evaluated for cost, risk, performance*
CML 2	*Initial Feasibility*	*Physics works, ballpark mass and cost*
CML 1	*Cocktail Napkin*	*Defined objectives and approaches, basic architecture concept*

Overview

The following constraints were defined as part of the study:

- Mission cost $1.5B–$1.9B in FY15$
- Planetary orbit must be achieved with floor payload (not a flyby mission)
- Aerocapture is allowable
- Launch window 2020–2023
- Assume no Jupiter gravity assist (need launch year flexibility)

As part of the study several mission-level requirements were defined by the study team to meet the science objectives.

Tier 1 Science

- 2-year orbital tour (20 orbits)
- Good coverage in magnetic latitude and longitude

Tier 2 Science

- Make atmospheric measurements with a shallow entry probe to at least 1 bar, 5 bars if possible

- *Minimize periapse (~1.1 R_U if possible) for gravity science*
    - Allow Earth tracking during periapse or near periapse
    - Enhanced orbiter payload
    - Encounters with large satellites desired if this objective doesn't interfere with measurements of Uranus's gravity field

Tier 3 Science

- Orbiter payload further enhanced
- Enhanced probe payload

The study began by conducting trajectory energy trades to both Uranus and Neptune based on the above constraints and requirements. For this launch window, the results indicated that Uranus was a lower risk destination since a Neptune mission would require either a prohibitively long cruise time (>15 years), which would greatly push Advanced Stirling Radioisotope Generator (ASRG) and spacecraft design life, or development of an aerocapture system. Based on these inputs, the panel chose to focus the remainder of the study on Uranus.

A robust Uranus mission concept in this time frame was enabled by the use of a separable solar electric propulsion (SEP) stage and an Earth gravity assist. This trajectory approach allowed the entire (enhanced) payload to be carried on an Atlas V 531 launch vehicle with a total cruise time of 13 years. This mission design approach also allows repeatable launch window of at least 21 days every year.

After launch, SEP thrusting would start after an approximate 30-day checkout. The SEP stage would provide velocity corrections to set up an Earth gravity assist approximately 4 years after launch and would continue thrusting the spacecraft for a total of 5 years. At a distance of about 5 AU, the effectiveness of the SEP stage is greatly reduced due to the low solar intensity and is separated from the orbiter.

The orbiter will then go into a spin-stabilized hibernation mode for the remaining 8 years of cruise, with a weekly status beacon and a monthly 8-hour contact. Prior to Uranus arrival, there will be some small targeting maneuvers approximately 1 year and 6 months out.

At 29 days prior to arrival, the spacecraft will release the entry probe on an intersect trajectory with Uranus. One day later, the spacecraft will perform a deflection maneuver to set up for Uranus orbit insertion (UOI). The probe will be in hibernation during most of its free-flight period, waking up only twice for contacts to better estimate the time of entry into the Uranus atmosphere to save on battery mass. The probe will wake up based on a timer about an hour prior to entry and will power up communications and instruments to take measurements.

The probe will hit the atmosphere at approximately 22 km/s at a flight path angle of –68 deg. At about 60 km altitude, its velocity will be slowed adequately to separate the aeroshell and deploy a parachute system. The probe will then make its atmospheric measurements from about 0.1 bar to 5 bars of pressure in 1 hour. During this time, the probe will constantly relay its science data back to the orbiter with the orbiter actively tracking it with the high-gain antenna (HGA).

Once the probe science phase has been completed, the orbiter will have 1 hour to communicate to Earth that it has successfully finished the probe phase and is prepared for UOI. It will then reorient to its preferred attitude for the burn and capture into orbit. The burn duration for UOI is just a little over 1 hour. The burn will be almost entirely occulted from the Earth, and any communications during this phase are not possible since they can only be accomplished with the HGA.

The spacecraft will be captured into a nearly polar orbit (inclination of 97.7 deg) with a 1.3-R_U periapse and a 21-day period. The 1.3-R_U periapse was the lowest that could be achieved while safely crossing the ring plane outside the rings (>52,000 km). Avoiding the rings also prohibited the possibility of achieving an orbit where periapse is directly viewable from Earth. Although if it were possible to improve knowledge of the environment between the rings and the planet and demonstrate safe transit, a different orbit insertion scenario is allowed that would result in a visible periapse and wider latitudinal coverage of the magnetic field (Appendix E, page 22). With the currently known safe orbit, ranging measurements have to be performed outside the eclipse period and thus at a slightly higher altitude (1.5 R_U). The highly inclined orbit allows very good latitude and longitude coverage over 20 orbits. During the prime science phase, the magnetometer and plasma instruments will operate continuously. The imagers will be used when the spacecraft is not communicating to Earth and will operate mostly on the day side of the orbit where higher resolutions can be achieved. The communications system was sized to provide 216 Mbits of data per day over an 8-hour daily contact; 32 Gbits of memory allows for a very robust data storage capability. Data acquisition will be greatest around closest approach to Uranus and will be downlinked throughout the orbit.

After the primary science phase is completed (431 days), there is adequate delta-V in the budget to adjust periapse to 1.1 R_U if the mission team feels that the risk is low enough to attempt to fly inside of the rings. This will potentially help with the gravity measurement goals.

The orbiter is sized to also achieve the lower priority objective of conducting satellite flybys. The orbit will be changed with targeting burns to cross each of the satellites which lie in Uranus's equatorial plane. The current mission design allows 10 targeted flybys along with four untargeted flybys:

- Two Miranda
- Two Ariel
- Two Umbriel + four close untargeted
- Two Titania
- Two Oberon

This tour would extend the mission by another 424 days with targeted encounters achieving closest approach altitudes of 50 km for each satellite. During this phase, significant imagery can be taken during the flyby and slowly downlinked over remainder of the orbit.

The mission is completed after a total duration of 15.4 years, including 2.4 years of science operations, which includes the satellite tour.

Several mission drivers shaped the overall mission concept and spacecraft design. These drivers and mission concept impacts are summarized in Table 2-2.

Table 2-2. Mission concept drivers and impacts.

#	Driver	Mission Concept Impacts
1	Flexible launch year	SEP stage with Earth Flyby
2	Distance for communications	3 ASRG configuration to accommodate 100 W TWTA
3	Shallow Entry Probe	Conduct probe science up to 1 hour prior to UOI
		Communications system that requires mono-pulse tracking of the probe
4	Limited launch capacity	Maximum mission duration of 16 years
5	Chemical propulsion for UOI	
6	Avoiding rings	Initial periapse of 1.3 AU
		Earth occulted at periapse
		Steep entry angle for probe and higher g loading
7	Satellite Tour	Adds significantly to Delta-V budget and propellant load

Technology Maturity

The technology readiness levels (TRLs) of all of the components of the Ice Giants mission to Uranus are shown in the master equipment lists for the solar-electric stage, the orbiter spacecraft, and the atmospheric probe. This mission can be developed with very little technology development. Most of the components are TRL 7 or higher, although some of the components are currently at TRL 6 and one is at TRL 5. All components of the probe and orbiter spacecraft are at TRL 6 or higher. The lowest TRL component is the Ultraflex solar array on the SEP stage, which is currently at TRL 5. The Ultraflex solar arrays are being developed for the Crew Exploration Vehicle and a number of other programs. Their development progress depends on continued funding by at least one of these future missions. A smaller version of this type of solar array has flown on the Phoenix Lander.

Whereas the vast majority of the mission components are at TRL 7, 8, and 9, there are a number of components on each of the three flight systems that are currently at TRL 6. Tables 2-3 through 2-5 list the TRL 6 items by flight element.

Table 2-3. SEP stage TRL 6 Items.

Component	Technology Progress	Flight Readiness
NEXT ion engine	*Under development at NASA GRC*	*2015*
NEXT gas feed system	*Under development at NASA GRC*	*2015*
Power processing unit (PPU)	*Under development at NASA GRC*	*2015*
Power management and distribution	*Update from the DAWN mission*	*2015*

Table 2-4. Orbiter TRL 6 Items.

Component	Technology Progress	Flight Readiness
ASRG	*Under development at NASA GRC*	*2015*
X-Ka-Band Coherent Transceiver	*Under development at JHU/APL*	*2014*

Table 2-5. Entry probe TRL 6 Items.

Component	Technology Progress	Flight Readiness
Mass spectrometer	*Updates to Galileo, currently unfunded*	*4 years after funding*
Probe battery system	*Requires qualification for 400-g deceleration*	*2 years after funding*
Coherent transceiver	*Modification of RBSP design*	*3 years after funding*
Probe antenna	*Modified NEAR patch antenna*	*2 years after funding*

Key Trades

A number of trades were performed both at the mission level as well as within each flight element to arrive at preferred concept. Table 2-6 summarizes the major mission trades performed as part of this study. Additional trade study information is provided in the appendices

Table 2-6. Significant mission trade studies.

Area	Trade Options	Results
Destination	Uranus or Neptune	• Uranus cruise time is 13 years with chemical capture and no JGA • Neptune requires aerocapture or >15 year cruise • Panel selected Uranus as destination
Capture Approach	Chemical vs Aerocapture	• Aerocapture begins to offer mass advantages if capture delta-V is over 2 km/s • For Uranus mission capture delta-V is 1660 m/s • Spacecraft packaging and probe accommodations are much more more challenging with aerocapture • Aerocapture would require $150-$200 M of technology investment to be ready for flight including a previous flight demonstration
Uranus Periapse	Altitude, Viewing Geometry	• Due to ring crossing constraints periapse cannot be below 1.3 Ru and cannot be in view of Earth. • A periapse of 1.1 Ru can be achieved if the trajectory is allowed to come inside of the rings. This is something that could be evaluated during the mission for safety and performed if the risk was deemed acceptable
Data Return	Dish size, power, array ground stations, others	• Basic science can be met with data return capability established in previous study (7.5 kbps, single 34 m ground station, 8 hour per day contacts) • ASRG architecture cannot support higher power TWTA • Dish is sized to 2.5 m to balance pointing requirements, cost, and RF performance • Adequate science can be returned without arraying DSN antennas
Instruments and Satellite Tour	Full Enhanced Mission with Tour Remove Tour Remove Probe Floor Instruments Only	• The full enhanced mission with tour can be achieved from a technical perspective within the cost cap ($1,894M) • Removing probe reduces cost by $310M • Removing the tour reduces cost by $26 • Removing Enhanced Instruments reduces cost by $120M

Some trade-offs were performed in almost all technical areas as a part of this study. The significant flight element level trades that were performed are summarized in Table 2-7.

Table 2-7. Significant element level trade studies.

Area	Trade Options	Results
SEP Power Trades	1-3 NEXT Engines, 15-25 kW	• Full mission can be accomplished with 2 NEXT engines and 20 kW of power • More power does provided modest increase in delivered payload performance
Probe Accommodation with Spacecraft	Spin stabilize with spacecraft rotation or spin table, probe placement	• Probe was placed on the bottom and the spacecraft will spin up to release
Probe Power, Activation, and Thermal Control	Activate on timer or command RHU vs battery power	• Use 4 RHUs to minimize power during probe free flight • Checkout probe 2 times prior to entry • Activate probe based on timer
Attitude Control	Wheels vs thruster control on orbiter, SEP stage or both.	• Wheels will meet science requirements • Thrusters provided no advantage in power • Wheels should be able to last 16 years • Orbiter wheels sufficient to control SEP stage

3. Technical Overview

Instrument Payload Description

The instrument complement is described in Tables 3-1 through 3-11 and text broken into four categories as defined by the study.

1. Floor Science Orbiter Payload
2. Enhanced Science Orbiter Payload
3. Floor Science Probe Payload
4. Enhanced Science Probe Payload

Floor Science Orbiter Payload Complement

Table 3-1. Wide angle camera.

Item	Value	Units
Type of instrument	*Wide angle camera (WAC)*	
Number of channels	*12 filters*	
Size/dimensions	*7.1 × 7.1 × 26.6*	*cm × cm × cm*
Instrument uncompressed average science data rate	*4.2*	*kbps*
Instrument fields of view	*10.5*	*degrees*
Pointing requirements (knowledge)	*350*	*μrad*
Pointing requirements (control)	*0.1*	*degrees*
Pointing requirements (stability)	*140*	*μrad/s*

The science objectives of the WAC are to determine the atmospheric zonal winds, composition, and structure of the Uranian atmosphere at high spatial resolution, as well as the temporal evolution of atmospheric dynamics, by spatially resolving the scattering properties of the atmosphere. Other science objectives addressed include remote sensing of the satellites and rings. For the purpose of this study, the WAC design is based on the MESSENGER WAC, without gimbal. The camera has a 10.5-degree field of view (FOV) and consists of a refractive telescope using a dogmar-like design having a collecting area of 48 mm. A 12-position multispectral filter wheel provides color imaging over the spectral response of the detector (395–1040 nm). The detector is a CCD array with 1024 × 1024 pixels, producing an image per filter of 1.05 Mbit with 8× compression. Assuming a strategy wherein on average 28% of the remote sensing data downlinked is dedicated to imaging with a WAC, over each 21-day orbit the spacecraft would return at least 907 images from the WAC (based on the average value of 7.5 kbps given under concept of operations). A pixel scale of 15 km is achieved when the spacecraft is closer than 4.4 R_U for ~5 hours per orbit. The instrument FOV is co-aligned with other remote-sensing instruments.

Table 3-2. Visible/near-IR mapping spectrometer.

Item	Value	Units
Type of instrument	*Vis/NIR mapping spectrometer*	
Number of channels	*7 on MVIC; 2 on LEISA*	
Size/dimensions (for each instrument)	*40.6 × 49.5 × 29.5*	*cm × cm × cm*
Instrument uncompressed average science data rate	*7.8*	*kbps*
Instrument fields of view	*5.7 × 0.037; 0.9 × 0.9*	*degrees*
Pointing requirements (knowledge)	*350*	*µrad*
Pointing requirements (control)	*0.1*	*degrees*
Pointing requirements (stability)	*20*	*µrad/s*

The science objective of the Vis/NIR mapping spectrometer is to determine the atmospheric zonal winds, composition, and structure of the Uranian atmosphere at high spatial resolution, as well as the temporal evolution of atmospheric dynamics, by spatially resolving the scattering properties of the atmosphere. Other science objectives addressed include remote sensing of the satellite and rings. For the purpose of this study, its heritage was based on the Ralph instrument on New Horizons. The instrument consists of a single 75-mm telescope that feeds two sets of focal planes: MVIC (7 512 × 32 pixel CCD arrays) and LEISA (128 × 128 pixel HgCdTe detector). MVIC provides panchromatic (400–975 nm), blue (400–550 nm), red (540–700), near-IR (780–975 nm), and methane (860–910 nm) channels, operating in a pushbroom mode. LEISA is a spectral mapper with two channels (1.25–2.5 µm, and 2.1–2.25 µm). The total amount of data per image per filter for the mapping spectrometer, assuming 8× compression, is 3.36×10^7 bits. For this study, data allocated to the spectrometer was 39% of total data volume; thus, over a 21-day orbit, the spacecraft can downlink as many as 53 image cubes. The 15-km/ pixel (100-km/pixel) scale is achieved when the spacecraft is closer than 4 R_U (21 R_U) for ~5 hours (52 hours) each orbit. The instrument FOV is co-aligned with other imaging instruments.

Table 3-3. Magnetometer.

Item	Value	Units
Type of instrument	*Magnetometer*	
Size/dimensions (sensor)/(electronics)	*8.1 × 4.8 × 4.6/13 × 10.4 × 8.6*	*cm × cm × cm*
Instrument average uncompressed science data rate	*150*	*bps*
Instrument fields of view (if appropriate)	*360*	*degrees*
Range	*0.1–2000*	*nT*

The science objective of the magnetometer is to understand the basic structure of the planet's magnetosphere as well as the high-order structure and temporal evolution of the planet's interior dynamo by making continuous measurements of vector magnetic field at 1-s resolution. During close satellite flybys (closest approach within a fraction of a satellite radius from the surface), the magnetometer would determine constraints on the presence and strength of induced magnetic fields indicative of the presence of subsurface oceans, which have been hypothesized for Oberon and Titania. To accommodate the full range of the magnetic field measurements, from the background interplanetary field at ~20 AU, which is ~0.1 nT, to the magnetic field strength at closest approach to Uranus, up to thousands of nanoteslas, the measurement range of ±2000 nT with 16-bit quantization was chosen. A boom (~10 m) and dual magnetometers are required to reduce the effects of local spacecraft fields. Moreover, a spacecraft magnetic cleanliness program is required to ensure that stray fields from spacecraft current loops are minimized. The magnetometer chosen is a low-noise, tri-axial, fluxgate magnetometer (MESSENGER heritage). The boom, composed of thin, nonmetallic rods, would be folded during launch and deployed after UOI insertion. The magnetometer data rates are based on a three-axis measurement, which will produce ~60 bits per sample. Assuming the sampling rates will vary from 1 sample/s when far from the planet to 20 samples/s when close to the planet or near magnetospheric boundaries, the average sampling rates comes out to 2.5 samples/s, or 6.48 Mbits/day, assuming 2× lossless compression.

Enhanced Orbiter Payload Complement

Table 3-4. Mid-IR thermal detector.

Item	Value	Units
Type of instrument	*Mid-IR thermal detector*	
Number of channels	*9*	
Size/dimensions (for each instrument)	*15.4 d × 30.5 l*	*cm × cm*
Instrument average uncompressed science data rate	*0.18*	*kbps*
Instrument fields of view	*15 × 0.10*	*degree*
Pointing requirements (knowledge)	*350*	*μrad*
Pointing requirements (control)	*0.1*	*degrees*
Pointing requirements (stability)	*0.05*	*degree/s*

The science objective of the Mid-IR thermal detector is to determine horizontal distribution of atmospheric thermal emission by measuring the temperatures as a function of latitude and longitude, as well as the upper atmospheric thermal structure and changes with time and location at low resolution. The mid-IR thermal detector achieves this objective by measuring reflected solar and emitted IR radiation in nine spectral channels with wavelengths ranging from 0.3 to 400 μm. For the purpose of this study, its heritage was based on the LRO Diviner and MRO MCS instruments. Two identical three-mirror off-axis telescopes are co-boresighted and mounted within an optical bench assembly. At the telescope focal planes are nine 21-element 256 × 256 pixel thermopile arrays, each with a separate spectral filter. The instrument FOV is co-aligned with other imaging instruments, operating as a multi-spectral pushbroom mapper. For one thermal image, per one filter, and assuming an 8× compression, the data is 1.3×10^5 bits. Allocating just 0.9% of the total data downlink to the thermal detector, ~311 observations can be returned from the spacecraft over a 21-day orbit. A 100-km/pixel scale is achieved when the spacecraft is closer than 14.3 R_U, for ~28 hours each orbit.

Table 3-5. Narrow angle camera.

Item	Value	Units
Type of instrument	*Narrow angle camera (NAC)*	
Number of channels	*5*	
Size/dimensions (for each instrument)	*27.4 d × 61.1 l*	*cm × cm*
Instrument average uncompressed science data rate	*1.05*	*kbps*
Instrument fields of view (if appropriate)	*0.29*	*degrees*
Pointing requirements (knowledge)	*350*	*μrad*
Pointing requirements (control)	*0.1*	*degrees*
Pointing requirements (stability)	*5*	*μrad/s*

The science objective of the NAC is to support remote sensing of the satellites, rings, and Uranus. For the purpose of this study, the NAC design is based on the New Horizons LORRI instrument. The camera has a 0.29-degree FOV and consists of a Ritchey-Chrétien telescope with a 20.8-cm diameter primary mirror, a focal length of 263 cm, and a three-lens field-flattening assembly. The imager provides panchromatic imaging over a bandpass that extends from 350 to 850 nm. The detector is a CCD array with 1024 × 1024 pixels, producing an image in one filter of 1.05 Mbit with 8× compression. About 7% of the total data volume is allocated to the NAC over each 21-day orbit. Therefore, the spacecraft can downlink as many as 226 images from the NAC, with resolution greater than 13 km/pixel during all of Uranus's orbit. During a satellite tour, the data allocation can be increased depending on the target body.

Table 3-6. UV imaging spectrograph.

Item	Value	Units
Type of instrument	*UV imaging spectrograph*	
Number of channels	*5*	
Size/dimensions (for each instrument)	*46.3 × 15.8 × 14*	*cm × cm × m*
Instrument average uncompressed science data rate	*20*	*Bps*
Instrument fields of view (if appropriate)	*6*	*degrees*
Pointing requirements (knowledge)	*350*	*Mrad*
Pointing requirements (control)	*0.1*	*degrees*
Pointing requirements (stability)	*0.1*	*deg/sec*

The science objective of the UV spectrograph is to observe the aurora and the Uranian upper atmosphere, measuring the upper atmospheric structure and changes with time and location at low resolution. For the purpose of this study, the UV spectrograph design is based on New Horizons ALICE. The instrument is an off-axis telescope feeding a Rowland-circle spectrograph with a 520- to 1870-Å spectral passband and a FOV of 6 degrees. The focal plane detector is an imaging microchannel plate double delay-line detector (1024 × 32 pixels) with dual photocathodes (KBr and CsI) and a focal surface that matches the instrument's 15-cm diameter Rowland circle. The UV spectogaph's allocation for the duration of one orbit is 138 images, with the total data volume of 4.54 Mbits, assuming 8× compression.

Table 3-7. Plasma instrument 1.

Item	Value	Units
Type of instrument	*Plasma Instrument 1*	
Energy range (ESA)/(RPA)	*(35–7500)/(0–2000)*	*eV/V*
Energy resolution	*(0.085)/(0.5 steps)*	*unitless/V*
Size/dimensions (for each instrument)	*24.4 d × 36.4 l*	*cm × cm*
Instrument average science data rate	*138*	*bps*
Instrument fields of view (if appropriate)	*276 × 10*	*degrees*

Science objectives for the plasma instruments require measuring the plasma and energetic charged particles in the Uranian magnetosphere to determine how the tilted/offset/rotating magnetosphere interacts with the solar wind over time. Two instruments were selected to satisfy these objectives.

The plasma instrument 1 provides measurements of plasma and ions in the Uranian atmosphere in the range of 35 eV–7.5 keV. For the purpose of this study, its heritage was based on the New Horizons SWAP instrument. The heritage instrument was a combination of a cylindrically symmetric retarding potential analyzer with small deflectors, a top-hat analyzer, and a redundant/coincidence detection scheme. The instrument operates continuously. Its total data volume allocation per orbit is 250 Mbits.

Table 3-8. Plasma instrument 2.

Item	Value	Units
Type of instrument	*Plasma instrument 2*	
Ion energy range/electron energy range	*(15 keV/n–1 MeV/n)/ (25–500 keV)*	
Energy resolution	*<5*	*keV*
Size/dimensions (for each instrument)	*19.7 × 21.5 × 12.7*	*cm × cm × cm*
Instrument average science data rate	*112*	*bps*
Instrument fields of view (if appropriate)	*160 × 12*	*degrees*

The plasma instrument 2 provides measurements of ions with compositional information and electrons from 15 keV to 1 MeV in a 160 × 12 degree fan-shaped beam in six sectors. For the purpose of this study, its heritage was based on the New Horizons PEPPSI instrument. Its total data volume allocation per orbit is 204 Mbits, and it operates continuously.

Floor Probe Payload Complement

Table 3-9. Mass spectrometer.

Item	Value	Units
Type of instrument	`	
Size/dimensions (for each instrument)	*16 d × 38 l*	*cm × cm*
Instrument average science data rate	*64*	*bps*
Ion source	*Electron impact/dual filament/variable energy (75 eV, 25 eV, 15 eV)*	*degrees*
Ion detector	*Secondary electron multiplier/pulse counter*	
Mass range (quadrupole mass analyzer)	*2–150*	*amu*
Dynamic range	10^8	
Detector threshold range	*(10 H_2O)/(1 Kr & Xe)*	*(ppmv)/(ppbv)]*

The science objective of the mass spectrometer is to determine the noble gas abundances of He, Ne, Ar, Kr, and Xe and isotopic ratios of H, C, N, and O in the atmosphere. The measurements will be performed by *in situ* sampling of the ambient atmosphere in the pressure range from approximately 100 mbar to 5 bar, together with batch sampling of the noble gas composition and isotopic ratio determination. For the purpose of this study, the mass spectrometer design is based on the Galileo Probe mass spectrometer.

The instrument will be operated using a controlled descent sequence with (16 bits/step) during the hour it takes to reach 5 bars, filling first the rare gas cell and then two enrichment cells starting at 0.5 bar to 2 bars, and between 3 and 5 bars. The nominal scan time is 75 s for the full mass range.

Table 3-10. Atmospheric structure instrument.

Item	Value	Units
Type of instrument	*Atmospheric structure instrument (p, t, accels)*	
Number of sensors	*2 temperature, 3 pressure, 4 accelerometers (x, y, redundant z)*	
Size/dimensions	*6 d × 6 l*	*cm × cm*
Instrument average science data rate	*50*	*bps*
Range (accel/temp/pres)	*(3 μg–409g)/(0–500 K)/(0.1–5 bars)*	

The science objective of the atmospheric structure instrument is to make *in situ* measurements of temperature and pressure profiles of the atmosphere of Uranus, starting at about 10^{-10} bar level for accelerometers and at 100 mbar for pressure and temperature, and continuing through its parachute descent to the 5-bar level. For the purposes of this study, the design of the instrument was based on the Galileo atmospheric structure instrument. Pressure and temperature are sampled at 1-s interval during descent (10-bit resolution), with absolute uncertainty of ~1 K at higher temperatures, and <0.1 at 100 K. Acceleration is sampled at 50 Hz until the probe reaches 100 mbar (62 s), and then at 10 Hz to 5 bar (~1 hour) at 12 bit/sample. The data is then losslessly compressed with FAST algorithm for accelerometers, with pseudo-logarithmic compression for pressure, and with DPCM for temperatures, resulting in 2x–4x total data volume compression.

Enhanced Probe Payload Complement

Table 3-11. Nephelometer.

Item	Value	Units
Type of instrument	*Nephelometer*	
Number of channels	*3 (1 backscatter channel, 2 radiometer channels)*	
Size/dimensions	*10 x 15 x 15*	*cm x cm x cm*
Instrument average science data rate	*10*	*bps*

The science objective of the nephelometer is to determine the presence of clouds as a function of depth in the atmosphere, by measuring the atmospheric scattering properties. For the purpose of this study, the nephelometer design is based on the Pioneer Venus nephelometer. The instrument is composed of a pulsed light source (0.9 μm), a detector to measure scattered light, collimating and collecting optics, spectral filters, and internal calibration systems. During transit of the probe, the instrument windows would be protected by the aeroshell and covered by a hinged hatch, deployed after entry by an explosive pinpuller. The hatch will also carry the calibration target. Data rate is based on obtaining a sample every kilometer of descent, from 100 mbar to 5 bar.

The mass and power of the orbiter instruments are given in Table 3-12.

Table 3-12. Orbiter instrument mass and power.

Orbiter Subsystem/Component	FLIGHT HARDWARE MASSES			FLIGHT HARDWARE POWER		
	Total CBE Mass (kg)	Contingency	Total MEV Mass (kg)	Total CBE Steady-State Power	Contingency	Total MEV Steady-State Power
Instruments	53.50 kg	16%	61.98 kg	41.5	30%	54.0
Wide Angle Camera	3.00 kg	15%	3.45 kg	3.50 w	30%	4.55 w
Narrow Angle Camera	8.60 kg	15%	9.89 kg	5.00 w	30%	6.50 w
Visible/Near-IR Mapping Spectrometer	10.50 kg	15%	12.08 kg	7.10 w	30%	9.23 w
Mid-Infrared Thermal Detector	8.00 kg	15%	9.20 kg	7.00 w	30%	9.10 w
UV Imaging Spectrograph	4.40 kg	15%	5.06 kg	4.40 w	30%	5.72 w
Plasma Instrument 1 (SWAP)	3.30 kg	15%	3.80 kg	3.00 w	30%	3.90 w
Plasma Instrument 2 (JEDI)	1.70 kg	15%	1.96 kg	2.80 w	30%	3.64 w
Magnetometer	1.00 kg	15%	1.15 kg	2.70 w	30%	3.51 w
Magnetometer Boom	10.00 kg	15%	11.50 kg	n/a	30%	
USO	3.00 kg	30%	3.90 kg	6.00 w	30%	7.80 w

The mass and power of the probe instruments are given in Table 3-13.

Table 3-13. Probe instrument mass and power.

Probe Subsystem/Component	FLIGHT HARDWARE MASSES			FLIGHT HARDWARE POWER		
	Total CBE Mass (kg)	Contingency	Total MEV Mass (kg)	Total CBE Steady-State Power	Contingency	Total MEV Steady-State Power
Instruments	14.70 kg	17%	17.13 kg	21.7	30%	28.2
Mass Spectrometer	8.00 kg	15%	9.20 kg	10.00 w	30%	13.00 w
Atmospheric Structure Inst. (ASI)	4.00 kg	15%	4.60 kg	6.30 w	30%	8.19 w
Nephlometer	1.20 kg	15%	1.38 kg	2.40 w	30%	3.12 w
USO	1.50 kg	30%	1.95 kg	3.00 w	30%	3.90 w

Flight System

The flight system for this mission consists of three major elements: a solar electric propulsion (SEP) stage, an orbiter, and an entry probe. Each element is described in more detail in the following sections.

SEP Stage

The function of the SEP stage is to provide the propulsion to deliver the orbiter and probe on a 13-year trajectory to Uranus. After the SEP propulsion phase of the mission is completed (~5 years), it will be jettisoned. The stage concept is shown in Figure 3-1.

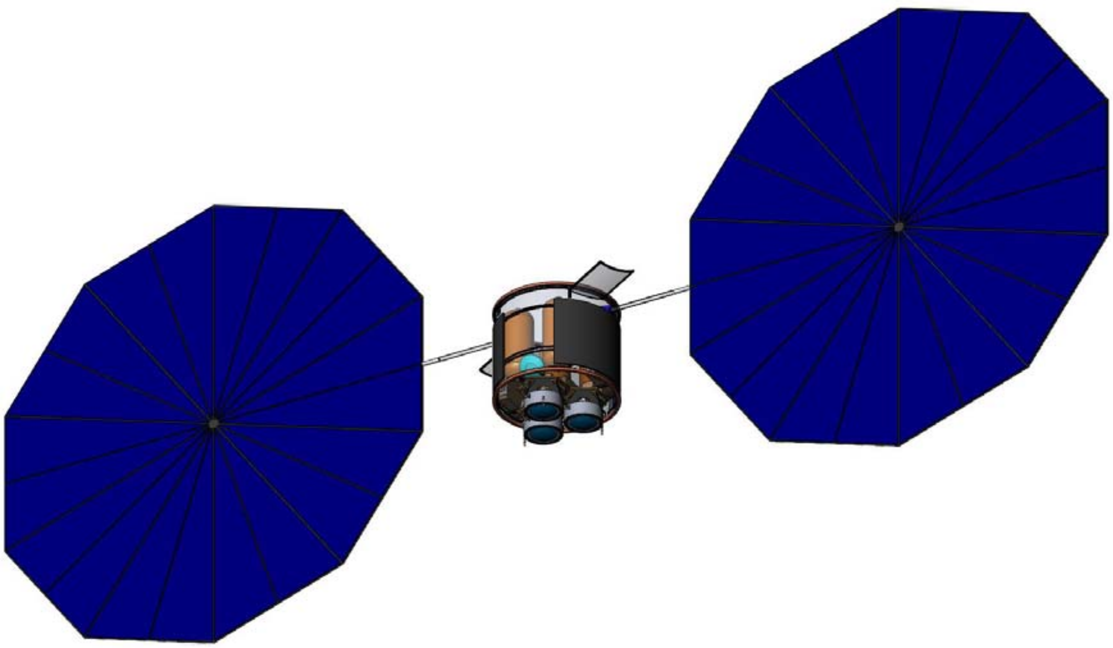

Figure 3-1. SEP stage concept.

Launched by an Atlas 531, the SEP stage design used a 13-year Earth flyby trajectory enabled by operating up to two NEXT xenon gridded ion thrusters using two 7-m-diameter Ultraflex solar arrays using IMM triple junction cells. The propulsion system consists of two active and one cold spare NEXT ion thruster each connected to a dedicated power processing unit and two xenon feed systems. The stage is kept in a tight thermal control band (due to the long duration in deep space) using multi-layer insulation (MLI) and heaters while the excess heat, primarily from the NEXT power processing units, is dumped from louvered radiators on surfaces of the spacecraft that point out of the ecliptic plane. Structurally the stage is a 1.6-m thrust tube made to match the Atlas 551 launch adapter. Communications and command and data handling (C&DH) are minimized on the stage, consisting of only external antennas connected to the orbiter communication system and remote interface units, which handle the lower level controls that are ultimately commanded by the Orbiter C&DH system. The control of the SEP stage is provided by a combination of NEXT thruster gimbals and the orbiter's hydrazine reaction control system (RCS) and four reaction wheels. The NEXT thrusters should be able to control the vehicle except during coast periods. The SEP stage design is similar to past work on outer planetary SEP stages, which, if developed, would greatly reduce the non-recurring cost of the SEP "stage" for the Uranus mission. Further work is needed to assess the interfaces, thermal impacts, and Uranus/SEP stage separation. The only new technology item is the 7-m Ultraflex solar arrays (TRL 5) (which can be based on the flight-proven 2-m Phoenix Ultraflex arrays).

A block diagram of the SEP stage is shown in Figure 3-2.

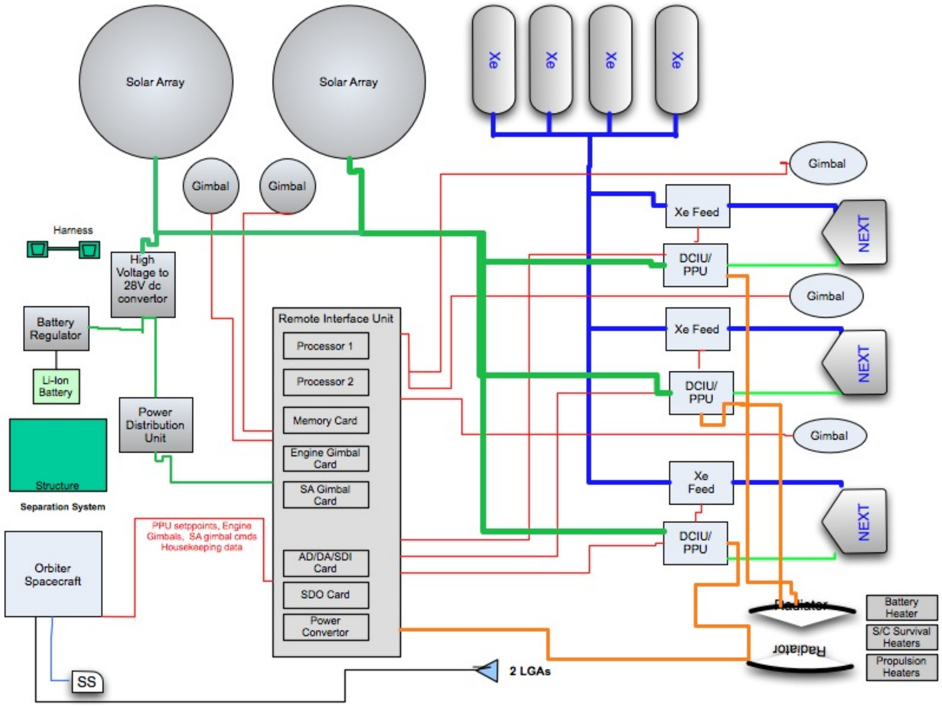

Figure 3-2. SEP stage block diagram.

A roll up of the mass and power is provided in Table 3-14. A more detailed master equipment list and power phasing table is provided in Appendix B. Also note that the power numbers in Table 3-14 represent a summation of all the power items for each subsystem and do not represent actual average power for any system mode. Power phasing by mode is provided on the power phasing table in Appendix B.

Table 3-14. SEP stage mass and power.

SEP Subsystem/Component	FLIGHT HARDWARE MASSES			FLIGHT HARDWARE POWER		
	Total CBE Mass (kg)	Contingency	Total MEV Mass (kg)	Total CBE Steady-State Power	Contingency	Total MEV Steady-State Power
Structures & Mechanisms	129.62 kg	18%	152.36 kg	N/A	N/A	N/A
Propulsion (Dry Mass)	315.07 kg	8%	338.78 kg	7115.0	30%	9249.5
Command & Data Handling (C&DH)	11.48 kg	16%	13.33 kg	47.0	5%	49.4
Electrical Power (EPS)	188.00 kg	30%	244.40 kg	70.0	30%	91.0
Guidance, Navigation, and Control	2.48 kg	3%	2.55 kg	N/A	N/A	N/A
Thermal Control (TCS)	87.22 kg	18%	102.92 kg	15.4	5%	16.2
RF Communications	2.20 kg	39%	3.06 kg	N/A	N/A	N/A
Harness	29.00 kg	46%	42.25 kg	N/A	N/A	N/A
SEP DRY MASS/POWER	765.08 kg	18%	899.66 kg			
Dry Mass Margin	30% (Note 1)		328.98 kg			
SEP Maximum DRY MASS			1094.06 kg			

Note 1: Margin is calculated based on Decadal Mission Study Ground Rules.

Dry Mass Margin% = (Maximum Dry Mass–CBE)/(Maximum Dry Mass)

Table 3-15 gives the SEP stage characteristics.

Table 3-15. SEP stage characteristics.

Flight System Element Parameters (as appropriate)	Value/Summary, units
General	
Design life, months	5 years
Structure	
Structures material (aluminum, exotic, composite, etc.)	1.6 m Al cylinder, Al fittings
Number of deployed structures	2 solar arrays
Thermal Control	
Type of thermal control used	Heat pipe radiators with louvers
Propulsion	
Estimated delta-V budget, m/s	7000 m/s
Propulsion type(s) and associated propellant(s)/oxidizer(s)	Xenon ion NEXT thrusters
Number of thrusters and tanks	3 NEXT thrusters,
Specific impulse of each propulsion mode, seconds	~4000 s
Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.)	3-axis (controlled by science orbiter)
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	
Attitude control capability, degrees	1° pointing accuracy for arrays
Attitude knowledge limit, degrees	from science orbiter
Agility requirements (maneuvers, scanning, etc.)	~180° in 60 minutes
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	Single axes solar arrays
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	from science orbiter
Command & Data Handling	
Flight element housekeeping data rate, kbps	100 kbps
Power	
Type of array structure (rigid, flexible, body mounted, deployed, articulated)	Ultraflex, single-axis pointed
Array size, meters × meters	7 m diameter (×2)
Solar cell type (Si, GaAs, multi-junction GaAs, concentrators)	IMM triple junction
Expected power generation at beginning of life (BOL) and end of life (EOL), watts	11 kW each EOL
On-orbit average power consumption, watts	14 kW
Battery type (NiCd, NiH, Li ion)	Li ion
Battery storage capacity, amp-hours	85 amp-hours

Orbiter

The orbiter is the most complex flight element of the mission and has the following functions:

- Command the SEP stage
- Provide attitude control system (ACS) for the integrated stage
- Provide power for the probe during cruise
- Communication functions with Earth (except for a low-gain antenna [LGA] on SEP stage)
- Spin-up and release the entry probe
- Track probe and act as communications relay for probe data
- Perform all maneuvers post SEP separation
- Provide a stable platform for science measurements

The orbiter concept is depicted in Figure 3-3.

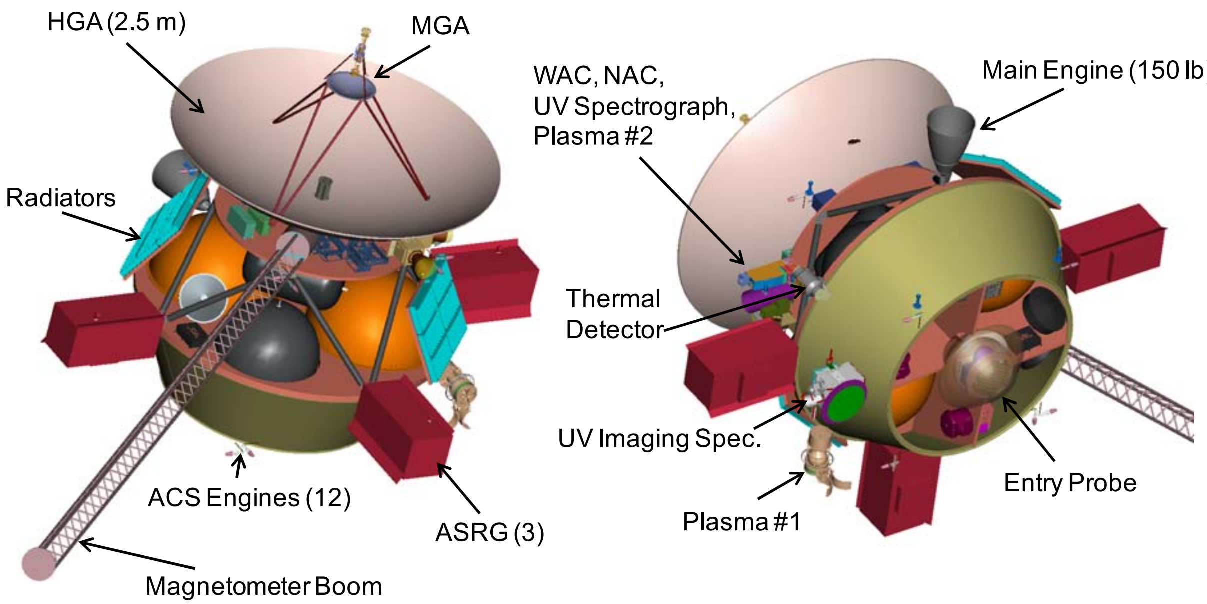

Figure 3-3. Uranus orbiter concept.

The orbiter is designed to operate within a power envelope of three currently developed ASRGs. This power level requirement is levied by the communications requirements to send a reasonable amount of imagery data at distances of 20 AU. The additional ASRG also provides some redundancy for the 15.4-year design life.

The structure is aluminum and is built around the large 2.5-m fixed HGA and the large dual-mode propulsion system needed for UOI and the orbital tour. The entry probe is mounted on the bottom of the spacecraft about its primary spin axis for hibernation mode and probe release. All instruments are mounted to the body of the spacecraft with the exception of the magnetometers, which require a 10-m boom with one mounted halfway out and the other on the boom tip.

The guidance and control (G&C) system consists of redundant star trackers, inertial measurement units (IMUs), and Sun sensors for attitude determination. Attitude control is managed by reaction wheels and hydrazine thrusters. The G&C system is designed to provide three-axis control for launch, detumble and SEP maneuvers, spin-stabilized control for hibernation and probe release, and three-axis control for Uranus orbit and probe tracking. The G&C system will also use mono-pulse tracking information provided by the communications system to actively track the probe during entry and ensure that the HGA can stay pointed adequately to the probe.

The propulsion is a dual-mode system with a 667-N bipropellant engine for large burns such as UOI, four thrusters at 22 N for UOI steering and smaller maneuvers, and 12 engines at 4.45 N for momentum management and large rate ACS maneuvers.

The thermal control system uses a “thermos bottle” approach by maintaining a minimum bus power level of at least 230 W to keep the core components warm. Heat pipes are used to distribute heat throughout the bus. Louvers are incorporated to open to manage peak thermal loads. Thermostatically controlled heaters provide thermal control throughout the bus as needed.

The main processor, interface electronics, and data recorder are housed in redundant integrated electronics modules (IEMs). A standard 32-Gbit recorder will be adequate to store up to almost 150 days of mission data (although the current CONOPS assumes daily data downlink). All of the C&DH and G&C processing will be done in the IEM.

The power system consists of three ASRGs, shunts, power system electronics, a Li ion battery, and a power distribution unit (PDU). ASRG controllers were assumed to be mounted separately from the ASRG units.

The communications system uses Ka band for science data downlink and X band for commanding and low-gain communications. On the orbiter, the RF system can switch between a 2.5-m HGA and 0.3-m medium-gain antenna (MGA) that are coaxially mounted. A low-gain antenna (LGA) (mounted to the SEP stage) can be used at distance of <1.5 AU and would primarily be used for emergency modes very early in the mission (e.g., after launch). The MGA can support emergency operations out to 21 AU. A 40-W Ka-band traveling wave tube amplifier (TWTA) provides adequate performance to a 7.5-kbps average rate during the mission.

All electronics subsystems are redundant and cross-strapped to accommodate the 15.4-year mission design life. The block diagram for the orbiter is provided in Figure 3-4.

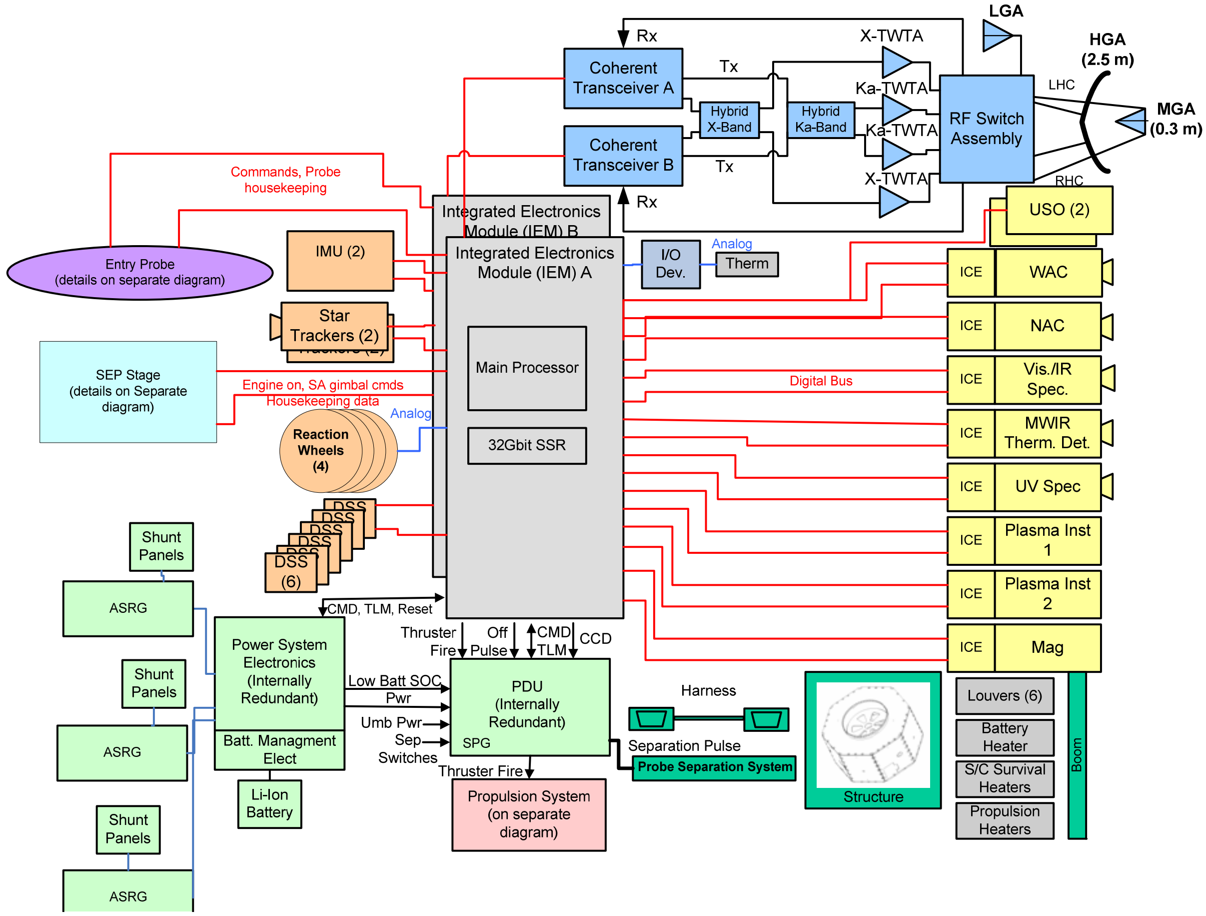

Figure 3-4. Uranus orbiter block diagram.

A roll up of the mass and power is provided in Table 3-16. A more detailed master equipment list and power phasing table is provided in Appendix B. Also note that the power numbers in Table 3-16 represent a summation of all the power items for each subsystem and do not represent actual average power for any system mode. Power phasing by mode is provided on the power phasing table in Appendix B.

Table 3-16. Orbiter mass and power.

Orbiter Subsystem/Component	FLIGHT HARDWARE MASSES Total CBE Mass (kg)	Contingency	Total MEV Mass (kg)	FLIGHT HARDWARE POWER Total CBE Steady-State Power	Contingency	Total MEV Steady-State Power
Instruments	53.50 kg	16%	61.98 kg	41.5	30%	54.0
Structures & Mechanisms	185.00 kg	15%	212.75 kg	N/A	N/A	N/A
Propulsion (Dry Mass)	119.16 kg	6%	125.89 kg	155.7	14%	177.5
Command & Data Handling (C&DH)	14.65 kg	15%	16.78 kg	10.2	5%	10.7
Electrical Power (EPS)	88.50 kg	14%	101.13 kg	13.5	5%	14.2
Guidance, Navigation, and Control	46.10 kg	5%	48.40 kg	45.5	5%	47.8
Thermal Control (TCS)	35.00 kg	13%	39.65 kg	40.0	10%	44.0
RF Communications	59.99 kg	14%	68.33 kg	232.2	7%	247.9
Harness	32.00 kg	15%	36.80 kg	N/A	N/A	N/A
ORBITER DRY MASS/POWER	633.89 kg	12%	711.71 kg			
Dry Mass Margin	30% (Note 1)		272.57 kg			
ORBITER Maximum DRY MASS			906.47 kg			

Note 1: Margin is calculated based on Decadal Mission Study Ground Rules.

Dry Mass Margin% = (Maximum Dry Mass–CBE)/(Maximum Dry Mass)

The orbiter characteristics are given in Table 3-17.

Table 3-17. Orbiter characteristics.

Flight System Element Parameters	Value/Summary, units
General	
Design life, months	15.4 years
Structure	
Structures material (aluminum, exotic, composite, etc.)	Aluminum truncated cone adapter, aluminum honeycomb decks, aluminum struts
Number of deployed structures	1 magnetometer boom
Thermal Control	
Type of thermal control used	"Thermos bottle" approach with heat pipes, louvers, and thermostatically controlled heaters
Propulsion	
Estimated delta-V budget, m/s	2500 m/s
Propulsion type(s) and associated propellant(s)/oxidizer(s)	Dual-mode Hydrazine and nitrogen tetroxide
Number of thrusters and tanks	1 667-N thruster 4 22-N thrusters 12 4.4-N thrusters
Specific impulse of each propulsion mode, seconds	332 s (bi-prop) 210 s (mono-prop)
Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.)	3-axis: SEP phase Spin: hibernation Spin: probe deploy 3-axis: maneuvers 3-axis: science phases
Attitude determination	Star tracker IMU Sun sensors
Attitude knowledge requirement	±100 microrad
Attitude control	Reaction wheels (0.2 Nm, 50 Nms) Thrusters
Attitude control capability	0.1 deg
Pointing stability	5 microrad/s
Agility requirements (maneuvers, scanning, etc.)	~180° in 60 minutes
Command & Data Handling	
Flight element housekeeping data rate, kbps	300 bps (science phase)
Data storage capability	32 Gbit
Power	
Primary power source	3 ASRG
Expected power generation at BOL and (EOL, watts	438 W BOL 367.5 W EOL (1 year fueled storage)
On-orbit average power consumption, watts	314–363 W (science phase)
Battery type (NiCd, NiH, Li ion)	Li ion
Battery storage capacity, amp-hours	16.8 amp-hours

Entry Probe

The entry probe concept was developed using the Pioneer Venus Small Probe as a starting point. The probe is encased by an aeroshell/thermal protection system (TPS) to protect it during high-speed entry and descent as well as to provide the drag necessary to slow it to subsonic speeds. After this entry phase, a parachute is deployed and the aeroshell is separated from the probe (Figure 3-5). The instruments, electronics, and batteries are housed in a titanium pressure vessel with xenon gas at 1 bar pressure. Instrument ports are opened using pyrotechnically deployed covers during the measurement phase that begins at 0.1 bar and extends to 5 bars of pressure.

In addition to the instruments described earlier, the probe contains an integrated electronics box, an X-band communications system, and primary batteries for power. To minimize required power and battery mass for the 29 days of free flight prior to entry, the probe incorporates four radioisotope heater units (RHUs). The electronics are put into hibernation with only a timer running. The timer will wake up the avionics and communication system at least twice to provide the orbiter with a navigation update and get a better estimate as to the final wake-up time to prepare for entry science.

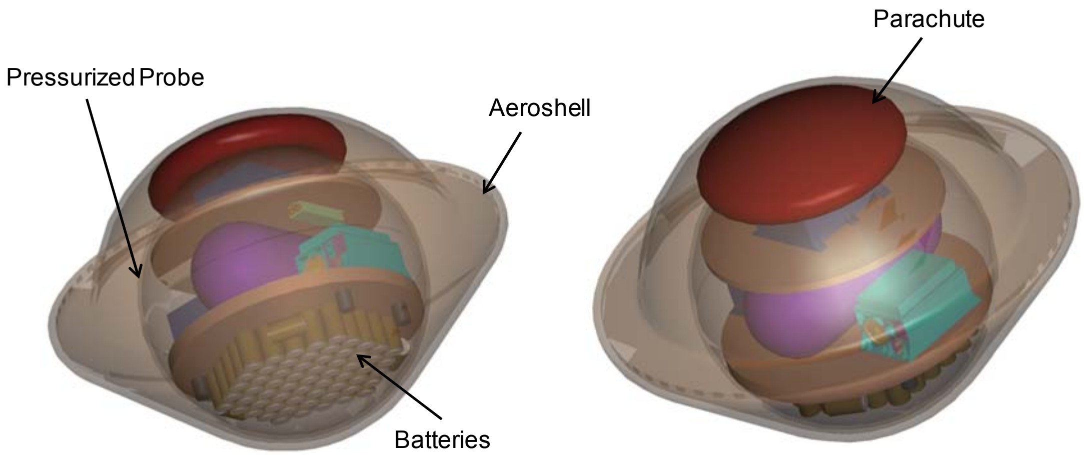

Figure 3-5. Entry probe configuration.

A block diagram depicting the entry probe components is shown in Figure 3-6.

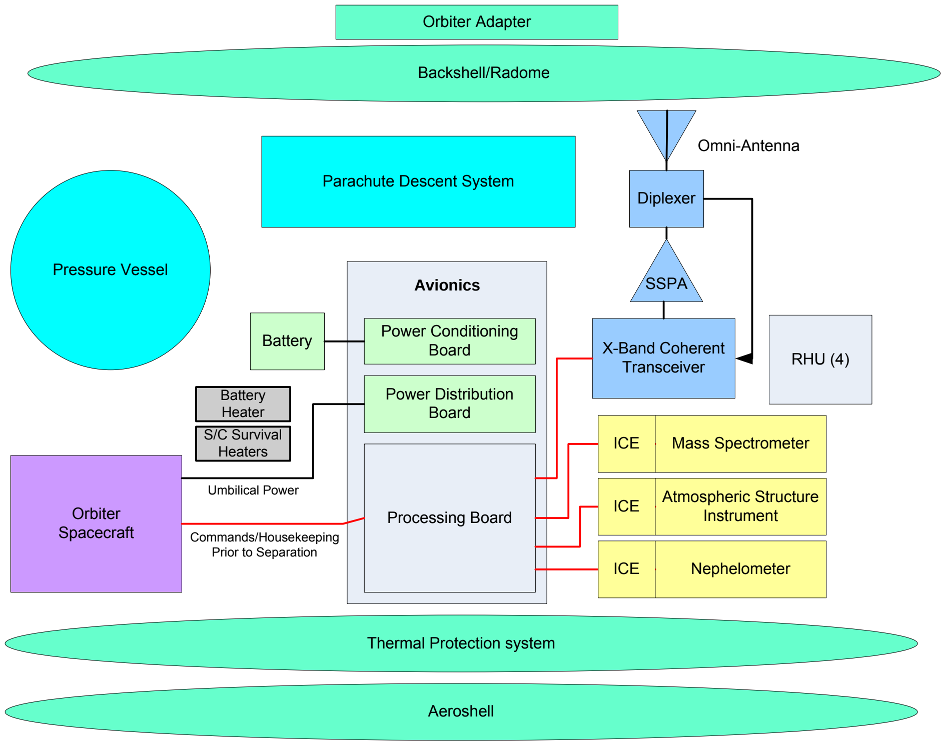

Figure 3-6. Entry probe block diagram.

A roll up of the mass and power is provided in Table 3-18. A more detailed master equipment list and power phasing table is provided in Appendix B. Also note that the power numbers in Table 3-18 represent a summation of all the power items for each subsystem and do not represent actual average power for any system mode. Power phasing by mode is provided on the power phasing table in Appendix B.

Table 3-18. Entry probe mass and power.

Probe Subsystem/Component	FLIGHT HARDWARE MASSES			FLIGHT HARDWARE POWER		
	Total CBE Mass (kg)	Contingency	Total MEV Mass (kg)	Total CBE Steady-State Power	Contingency	Total MEV Steady-State Power
Instruments	14.70 kg	17%	17.13 kg	21.7	30%	28.2
Structures & Mechanisms	49.50 kg	15%	56.93 kg	N/A	N/A	N/A
Command & Data Handling (C&DH)	4.00 kg	15%	4.60 kg	3.8	30%	4.9
Electrical Power (EPS)	8.70 kg	30%	11.31 kg	N/A	N/A	N/A
Thermal Control (TCS)	1.36 kg	15%	1.56 kg	7.0	10%	7.7
Communications	5.21 kg	9%	5.66 kg	21.1	19%	25.2
Harness	5.40 kg	15%	6.21 kg	N/A	N/A	N/A
PROBE DRY MASS/POWER	88.87 kg	16%	103.40 kg			
Dry Mass Margin	30% (Note 1)		38.21 kg			
PROBE Maximum DRY MASS			127.08 kg			

Note 1: Margin is calculated based on Decadal Mission Study Ground Rules.

Dry Mass Margin% = (Maximum Dry Mass–CBE)/(Maximum Dry Mass)

Entry probe characteristics are given in Table 3-19.

Table 3-19. Entry probe characteristics.

Flight System Element Parameters	Value/Summary, units
General	
Design Life	Cruise (off): 13 years Free flight: 29 days Science: 2 hours
Mechanical	
Aeroshell	Aluminum, 45 deg blunted cone 76 cm diameter
Pressure vessel	Titanium, 46 cm diameter
Deployments	Pyrotechnically controlled instrument inlet covers Aeroshell Parachute
Thermal Control	
Type of thermal control used	Orbiter for cruise 4 RHUs for free flight Carbon phenolic TPS for high-speed descent Foam insulation for post-aeroshell deployment
Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.).	Spin stabilized for free flight Aerodynamically stabilized for entry and descent
Parachute system	Conical ribbon parachute (CD = 0.55) Diameter = 3.25 m
Command & Data Handling	
Data rate, kbps	200 bps
Data storage capability	1 Gbit
Power	
Primary power source	Primary battery
On-orbit average power consumption, watts	1–69 W
Battery type (NiCd, NiH, Li ion)	Li-thionyl chloride
Battery storage capacity, amp-hours	49 amp-hours

Integrated Flight System

The integrated flight system is shown in Figure 3-7, both in its launch and SEP cruise configurations. The entire system is accommodated on an Atlas V 531 launch vehicle.

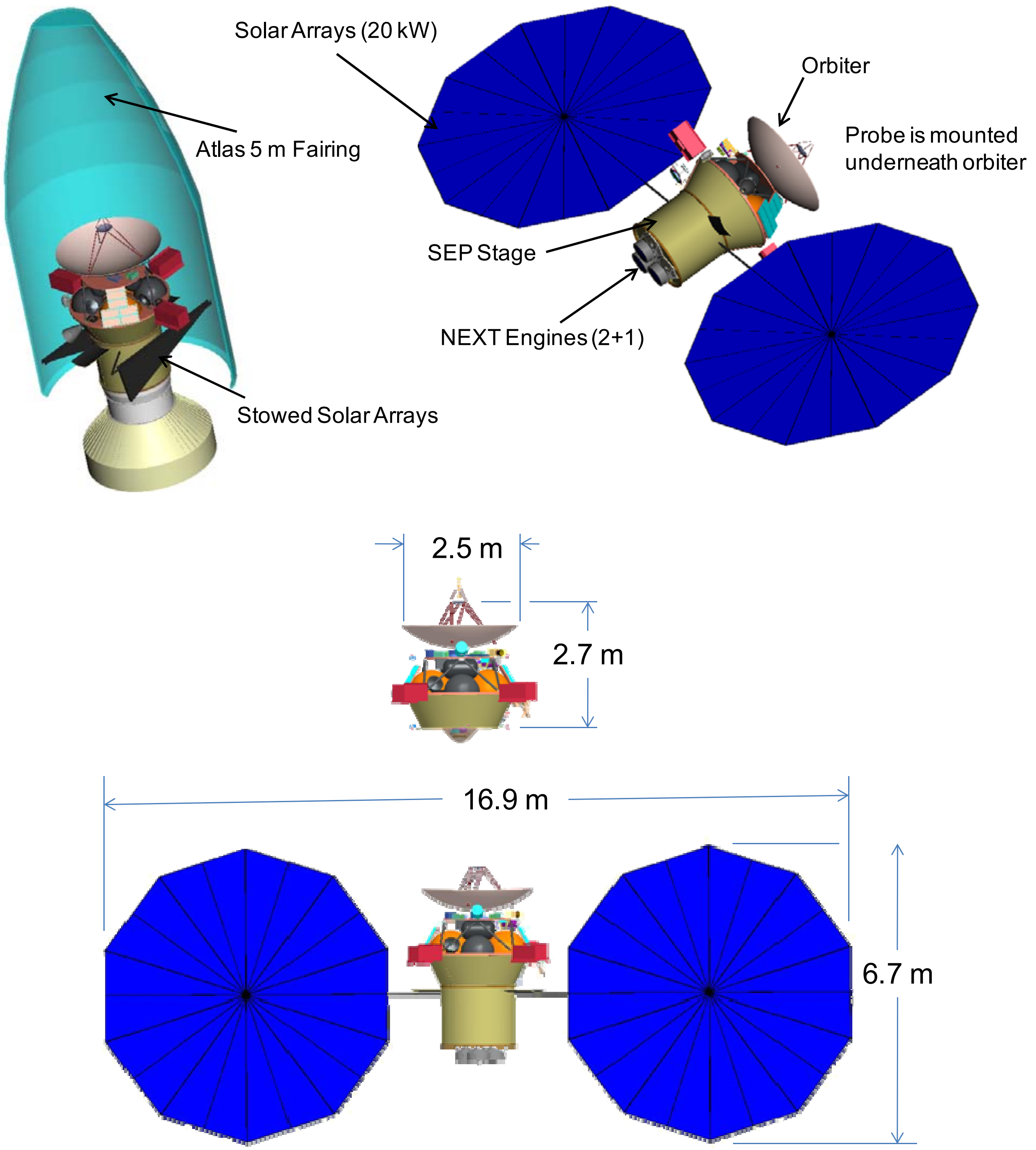

Figure 3-7. Integrated flight system: launch (upper) and SEP (lower) cruise configurations.

Concept of Operations and Mission Design

Mission Design

As part of this study and the previous Uranus decadal study effort, many trajectory trades were performed on how to deliver significant payloads into a Uranus orbit. Purely ballistic options were possible using Jupiter gravity assist but were found to have limited payload delivery capability for the launch years examined. Low thrust trajectories using solar electric propulsion were found to deliver significant payload levels into Uranus's orbit. Use of a Jupiter gravity assist with SEP enables large payloads consistent with this study and modest cruise durations of 10–12 years. However, constraints imposed for this Decadal Study precluded the use of a Jupiter gravity assist option since the last opportunity for that decade was in 2021. If a Jupiter gravity assist were available, additional concept work would need to be done to determine the overall mission benefit.

For this study, the mission design team developed a trajectory option using SEP with an Earth gravity assist (>1000 km altitude) that could enable the full enhanced mission. A 13-year cruise is required to limit the arrival velocity enough so that capture is possible with a chemical bipropellant system with enough remaining propellant to perform a robust satellite tour. A shorter cruise time of 12 years could be possible but would have limited the mission to the primary mission orbit only. This trajectory design also had the advantage that it could be accomplished in any launch year with a 21-day launch window. Figure 3-8 illustrates this trajectory approach showing the baseline launch date of July 2020.

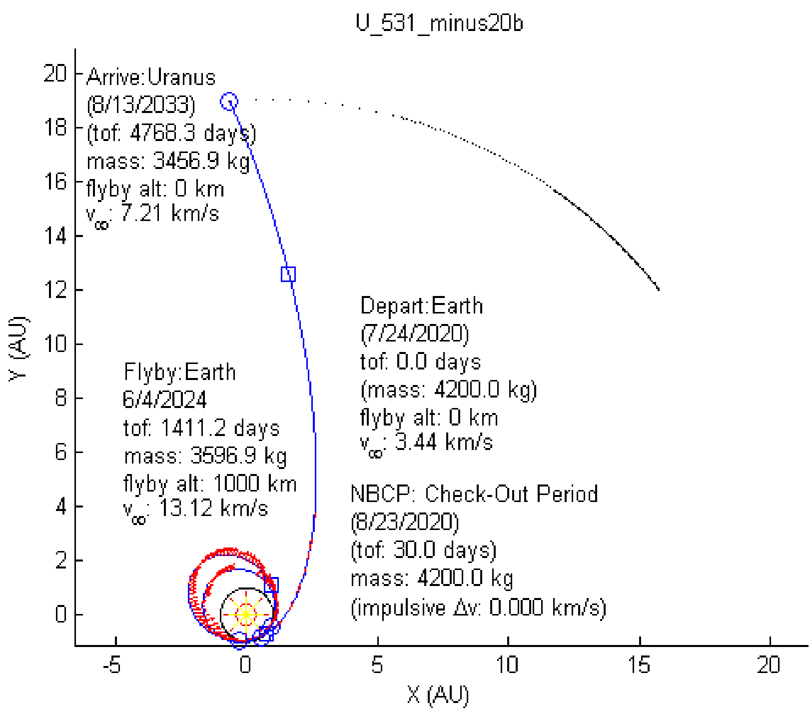

Figure 3-8. Baseline SEP trajectory for study

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Parameters for launch and the trajectory are provided in Tables 3-20 and 3-21.

Table 3-20. Mission design: Launch parameters.

Parameter	Value	Units
Launch location	CCAFS	
Launch vehicle	Atlas V 531	
Launch window (7/24/20–8/13/20 baseline case)	21	days
Launch C3	11.83	km^2/s^2
Launch mass with required 30% margin	4129	kg
Launch vehicle lift capability	4205	kg
Propellant contingency (available margin above estimated load required for mission)	71	kg

Table 3-21. Mission design: Interplanetary trajectory.

Parameter	Value	Units
Trajectory type	SEP with Earth gravity assist (EGA)	
EGA altitude	1000	km
Powered cruise duration	5	years
Total cruise duration	13	years
Repeatability	Every year	
SEP engines	2 NEXT, 90% duty cycle	kg
Cruise power at 1 AU	20	kW
Checkout coast period	30	days
EGA coast period	42	days
Additional coasts	7@ 30	days

To accommodate science using an entry probe, the probe must be released prior to UOI on an entry trajectory to the planet. Achieving a safe UOI outside the rings with the lowest possible periapse of 1.3 R_U puts significant constraints on the probe entry conditions. The design was established to release the probe 29 days prior to Uranus arrival, allowing enough time for a deflection burn with the orbiter at a modest delta-V of 30 m/s 28 days before arrival. This geometry (shown in Figure 3-9) allows 1 hour of viewing the probe's descent within the atmosphere with another hour to prepare for UOI.

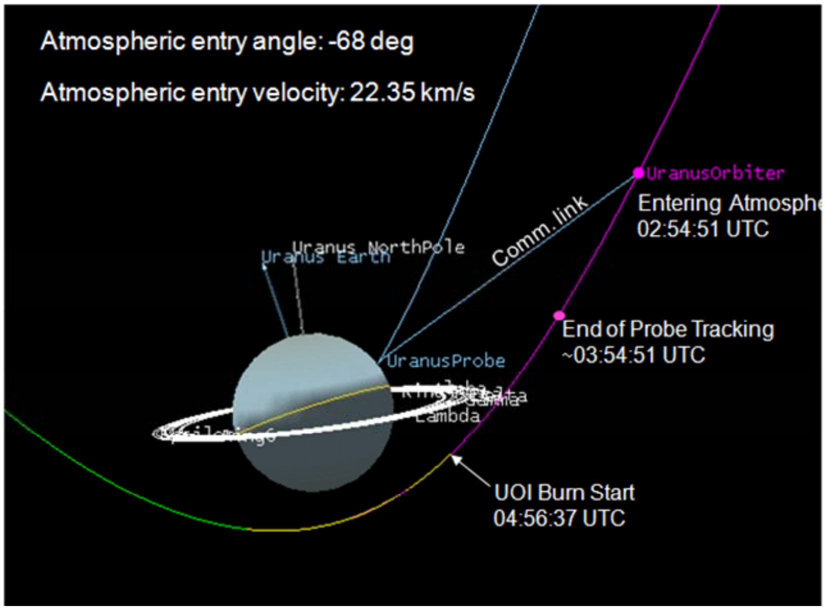

Figure 3-9. Probe entry and UOI geometry.

The probe will enter using aerodynamic braking to slow from 22 km/s at entry interface (~500 km altitude) to Mach 1 at about 60 km altitude (0.1 bar pressure) in about 1 minute. At that time a parachute will deploy, allowing a slow enough descent rate for science to be taken from 0.1 bar to 5 bars over a 1-hour period (approximately 120 km of altitude change). Parameters for the probe entry are provided in Table 3-22.

Table 3-22. Mission design: Probe entry.

Parameter	Value	Units
Probe release prior to Uranus arrival	29	days
Probe entry velocity	22.3	km/s
Entry angle	–68	deg
Peak deceleration loads	372	g
Peak heat rates	5511	W/cm^2
Peak heat loads	38.1	kJ/cm^2
Time to deploy parachute from entry interface	62	s
Measurement pressure range	0.1–5	bar

UOI is conducted 1 hour after probe science has been completed. Probe science will not be transmitted to Earth until after UOI. The UOI burn requires 1661 m/s of velocity correction using a steered trajectory and includes finite burn penalties. The total burn time is 66.7 minutes. Due to the required approach and orbit geometry, almost the entire burn will be occulted from Earth view (the very beginning and end of the burn are within Earth view). In addition, at 20 AU a link cannot be established with an LGA, making communications impossible during this event.

After UOI is complete, the orbiter begins its primary mission phase. The science goal of determining Uranus's internal structure from its gravity field led to a desire for the orbit to have as low a periapse as possible and to have it be viewable from Earth. Constraints of maintaining a safe ring-plane crossing range (≥52,000 km) limited periapse to 1.3 R_U and it is occulted from the Earth. However, the orbiter can viewed from Earth shortly before and after periapse at a radius of 1.5 R_U. Figure 3-10 illustrates the primary orbit.

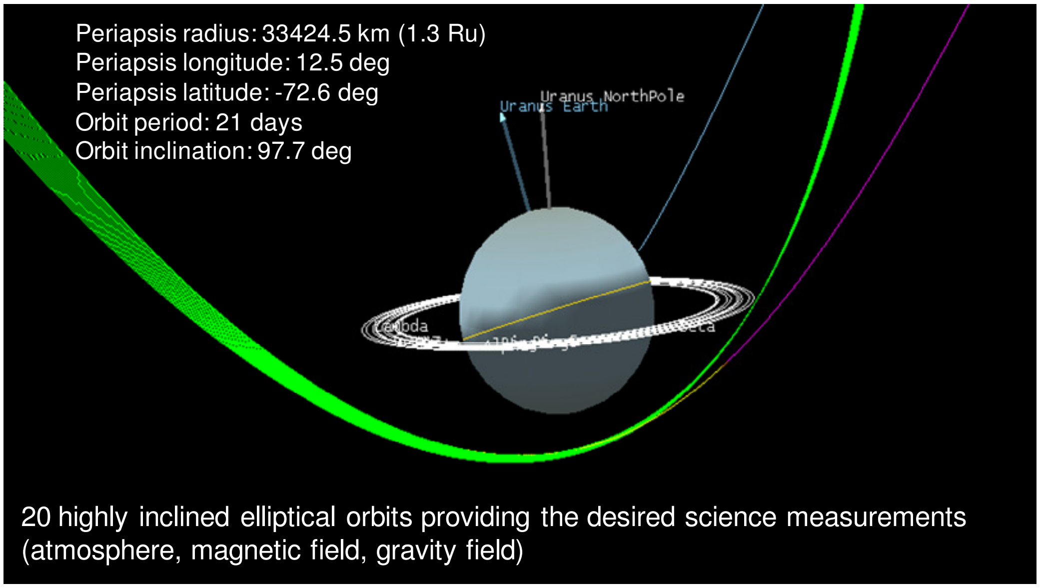

Figure 3-10. Primary science orbit.

Since Uranus has a large tilt angle relative to the ecliptic, it was not difficult to provide a near polar orbit to allow for the desired diversity in latitude and longitude coverage. Over the course of 20 orbits, significant coverage of the planet is obtained, illustrated in Figure 3-11.

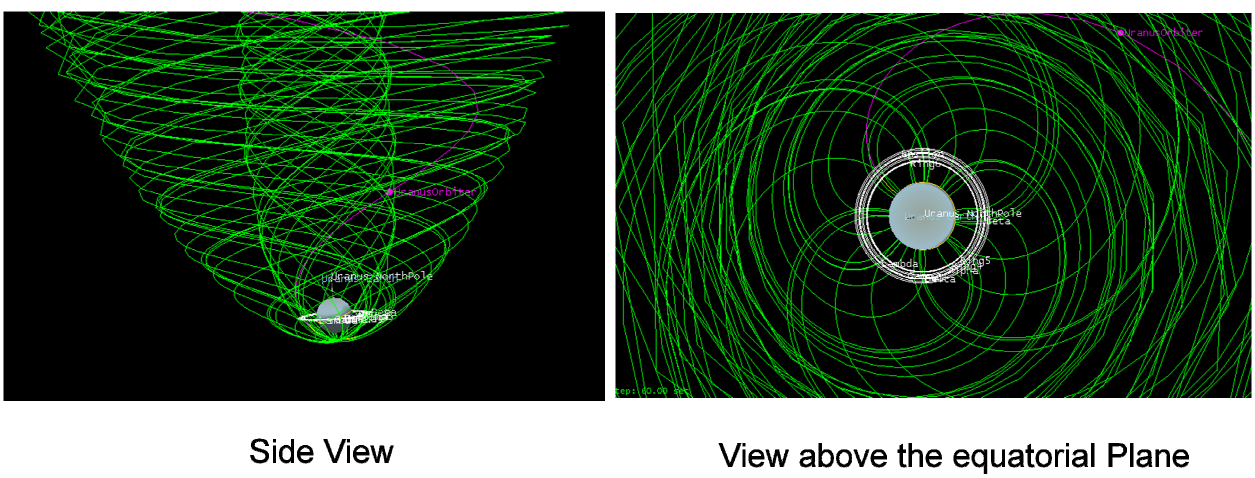

Figure 3-11. 20 Science orbits plotted in Uranus fixed coordinate system (purple line: incoming trajectory, green line: science orbits).

The basic orbit parameters and mission phase duration are summarized in Table 3-23.

Table 3-23. Mission design: Primary science orbit.

Parameter	Value	Units
Periapse	1.3	R_U
Apoapse	51.3	R_U
Inclination	97.7	deg
Orbital period	21	days
Primary science phase duration	431	days

An option was studied to potentially lower periapse at the end of the mission to 1.1 R_U, allowing the trajectory to pass inside of the rings. This was not considered an acceptable risk for the primary mission since it is expected that a significant debris hazard exists inside the known rings. However, at the end of the mission, this risk could be reassessed based on the available mission data. If possible, this option would be of high scientific priority and could be achieved at a modest delta-V penalty of 56 m/s, which has been budgeted into the mission concept.

Furthermore, if it were possible to improve knowledge of the environment between the rings and the planet before the mission and demonstrate that spacecraft transit through this region would be safe, a different orbit insertion scenario is allowed that would result in a visible periapse near the equator, thereby enabling tracking closer to the planet to improve the gravity measurements. (See Appendix E, page 22.)

The final phase of the mission was to accomplish the lower-priority objective of conducting a tour of the Uranian satellites. A tour was developed allowing two targeted flybys of each of the five major satellites (Miranda, Ariel, Umbriel, Titania, and Oberon) at a closest approaches of 50 km. Figure 3-12 depicts the geometry of the satellite tour, and Table 3-24 addresses the performance parameters. Additional information including ground tracks is provided in Appendix F.

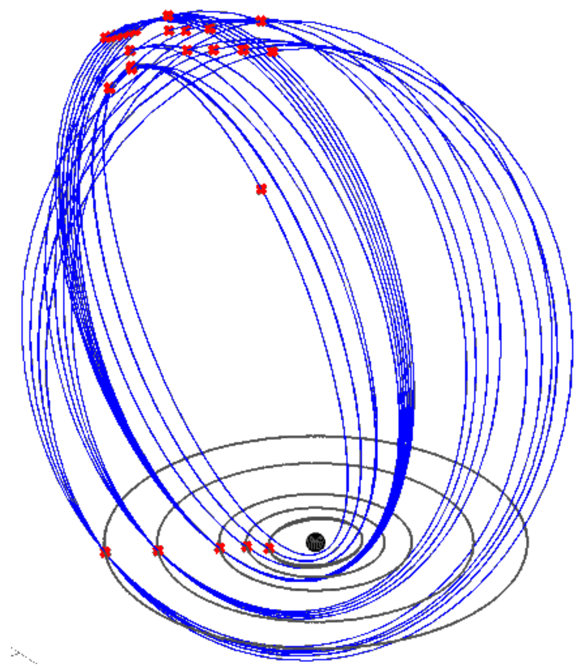

Figure 3-12. Satellite tour geometry.

Table 3-24. Mission design: Satellite tour.

Parameter	Value	Units
Number of satellites encountered	5	
Number of targeted flybys per satellite	2	
Number of additional untargeted close flybys of Umbriel	4	
Satellite tour duration	424	days

Concept of Operations

Operations begin with a 5-week checkout after launch. Initial G&C is done using ASRG power with solar arrays stowed. This will be followed by 12 weeks of instrument checkout. Staffing levels will be higher through EGA building and testing sequences to be used for future critical operations. SEP requires some continuous caretaking of propulsion navigation. Based on previous experience, the staffing can decrease to levels less than MESSENGER after EGA. After SEP separation, the spacecraft will spend a significant amount of time in hibernation and the operations team can be reduced significantly. The lack of cruise science also allows for a smaller operations team size. Currently available ground systems and Deep Space Network (DSN) support are more than adequate to support this mission.

Probe science and UOI will put the highest demands on the operations team since both events are completed back to back over the course of a few hours. Due to the distance and severe communications delays (2.8 hours one way), the mission will be very dependent on time-tagged commands, and a significant amount of autonomy will be required in the orbiter. The operations team will be able to communicate prior to the start of probe entry, receive a status message between probe entry and UOI, and begin communications after UOI.

Primary science will require one 8-hour contact per day on average for the 20 orbits. Most of the imagery science will be performed well before apoapse so there will be a significant amount of time for additional contacts near apoapse if needed.

Downlink rate will average 7.5 kbits/s over the year but will be variable depending on the station and the weather. Ka-band communications is very sensitive to weather and elevation angle so the actual data rate at any pass could vary from 2 to over 10 kbps. Figure 3-13 shows examples of the best and worst seasons. These charts show all three DSN stations, and the two curves are for 10% and 90% cumulative distribution of the weather. Table 3-25 shows the mission operations and ground data systems.

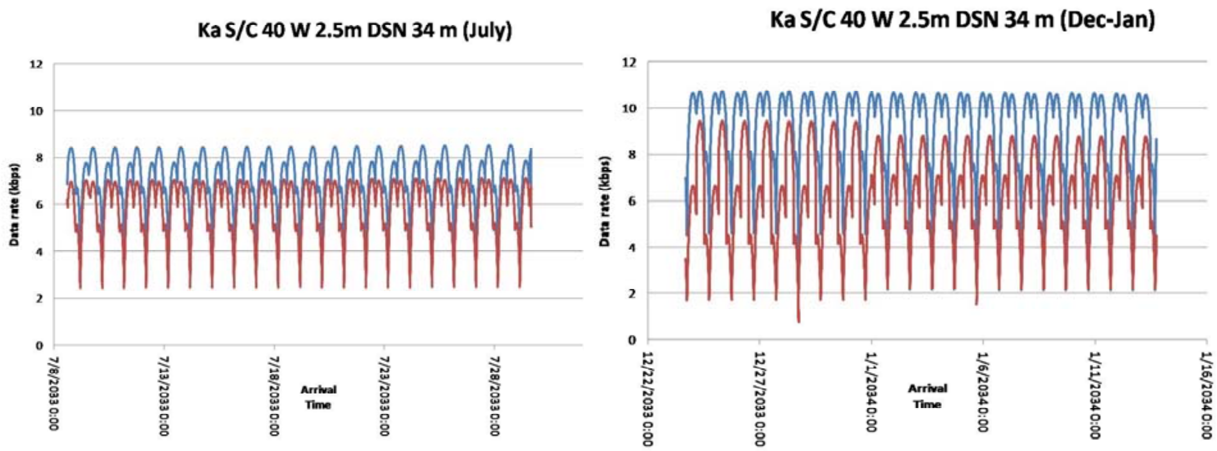

Figure 3-13. Data rate variations for two seasons with DSN.

Table 3-25. Mission operations and ground data systems.

Downlink Information	Early Ops/ Instrument Commissioning	SEP Cruise	Earth Gravity Assist	Hibernation Cruise	Uranus Encounter Prep inc UOI	Uranus Probe Ops (Probe to S/C)	Orbital Science	Satellite Science
Number of contacts per week	21-3	2	2-21-2	1	3-21	21-7-21	7	7
Number of weeks for mission phase, weeks	17	254	12	324	12	5	61	60
Downlink frequency band	X/Ka	X	X	X	Ka	X/Ka**	Ka	Ka
Telemetry data rate(s), kbps	0.10/280/600	0.25/0.50	0.10/280/600	0.010/1.0	1.0*	0.20/1.0	7.5*	7.5*
Transmitting Antenna Type(s) dBi	LGA(X)/MGA(X) /HGA(X/Ka)	MGA/HGA	LGA/MGA/ HGA	MGA/HGA	HGA	HGA	HGA	HGA
Transmitter DC power, watts	116(X)/100(Ka)	116	116	116	100	116/100	100	100
Downlink receiving antenna gain, dBi	5/25/42/55	25/42	5/25/42	25/42	55	42/55	55	55
Transmitter RF output, watts	60(X)/40(Ka)	60	60	60	40	60/40	40	40
Total daily data volume, (MB/day)	0.086(LGA)/8.064 (MGA)/17.2(HGA)	0.0072	0.0864(LGA)/ 8.064(MGA)/ 17.2(HGA)	Beacon(MGA)/ 0.028(HGA)	0.028–0.086	0.36/0.086	27‡	27‡
Uplink Information								
Number of uplinks per day	2-3	2 per week	3	1 per month	3	3	1	1
Uplink frequency band	X	X	X	X	X	X	X	X
Telecommand rate, kbps	0.125(LGA)/2.0 (MGA)	0.125(MGA)† 0.25(HGA)	0.125(LGA)/2 (MGA)	0.25 (HGA)	0.25 (HGA)	0.25 (HGA)	0.25 (HGA)	0.25 (HGA)
Receiving antenna type(s) and gain(s), dBi	34m	34m	34m	34m	34m	34m	34m	34m

- * Downlink data rate varies with weather and station. Hourly rate table coordinated between Mission Operations Center and DSN.
- ** X band for probe-orbiter communications—every 5 days; Ka for orbiter to Earth.
- † No ranging with MGA @ 21 AU. Ranging available at stated data rate with HGA.
- ‡ Based on 8-hour contact time.

Risk List

The top 10 risks have identified likelihood and consequence levels along with a summarized mitigation strategy (Table 3-26). They are also shown on a 5 × 5 risk matrix (Figure 3-14).

Table 3-26. Top ten risks.

#	Risk	Type	L	C	Mitigation
1	If the complexity of conducting the probe science prior to UOI is too high, the probe science would be descoped	Technical	2	4	• Comprehensive analysis and test program • More rehearsals prior to event
2	If there are problems in the development of a new SEP stage then the schedule could be impacted	Schedule	2	3	• Develop a SEP stage demonstrator as a technology risk reduction
3	If mass spectrometer has development problems, then schedule will be impacted	Schedule	2	3	• Mass spectrometers are complex instruments and development will need to begin early
4	If significant issues with an Earth flyby develop, then mission schedule could be impacted.	Schedule	1	4	• Keep flyby distance ≥1000 km altitude • Begin approval process as early as possible • Jupiter or Saturn gravity assisted trajectory mitigates Earth flyby
5	If UOI becomes technically complex due to no contact, then mission schedule will be impacted	Schedule	1	4	• Assess UOI approach early • Assess required autonomy and associated cost and schedule to develop and test
6	If there is difficulty developing a large (20 kW) array, then schedule could be impacted	Schedule	1	3	• Develop and demonstrate a large parasol array as part of a technology risk reduction effort
7	If the orbiter cannot track the probe, then mission performance will be impacted	Technical	1	3	• Develop and test tracking system early • Plan for significant modeling and analysis of probe entry and tracking problem
8	If an ASRG fails, the mission will be degraded	Technical	1	3	• Qualify ASRG for longer life • Limit storage time prior to launch with fueled ASRG to 1 year
9	If system reliability is not shown to be adequate for long mission, then cost will be impacted	Cost	1	3	• Perform reliability analysis early to assess weaknesses in system • Assess system architecture options to improve reliability
10	If the peak acceleration loads have a large impact on the probe design, mission cost could be impacted.	Cost	1	3	• Qualify electronics and batteries for environment

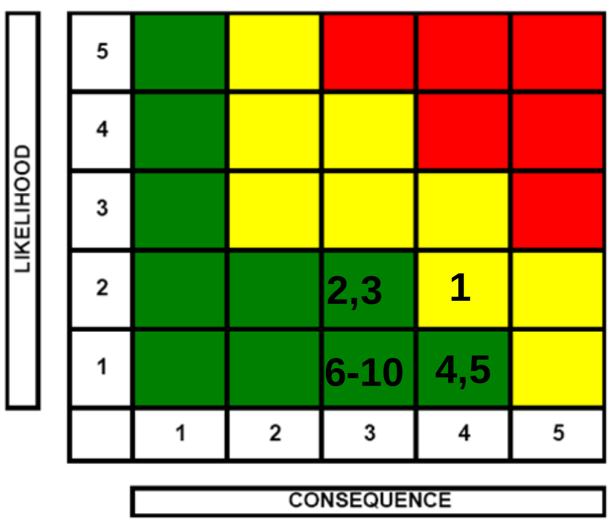

Figure 3-14. Risk matrix.

4. Development Schedule and Schedule Constraints

High-Level Mission Schedule

The following high-level mission schedule (Figure 4-1) is based on a previous schedule for a relevant mission in the same class and also recent concept development efforts, which was deemed a good approximation to the anticipated schedule for the Uranus Mission. The schedule was also updated to take into account the required additional SEP cruise stage development.

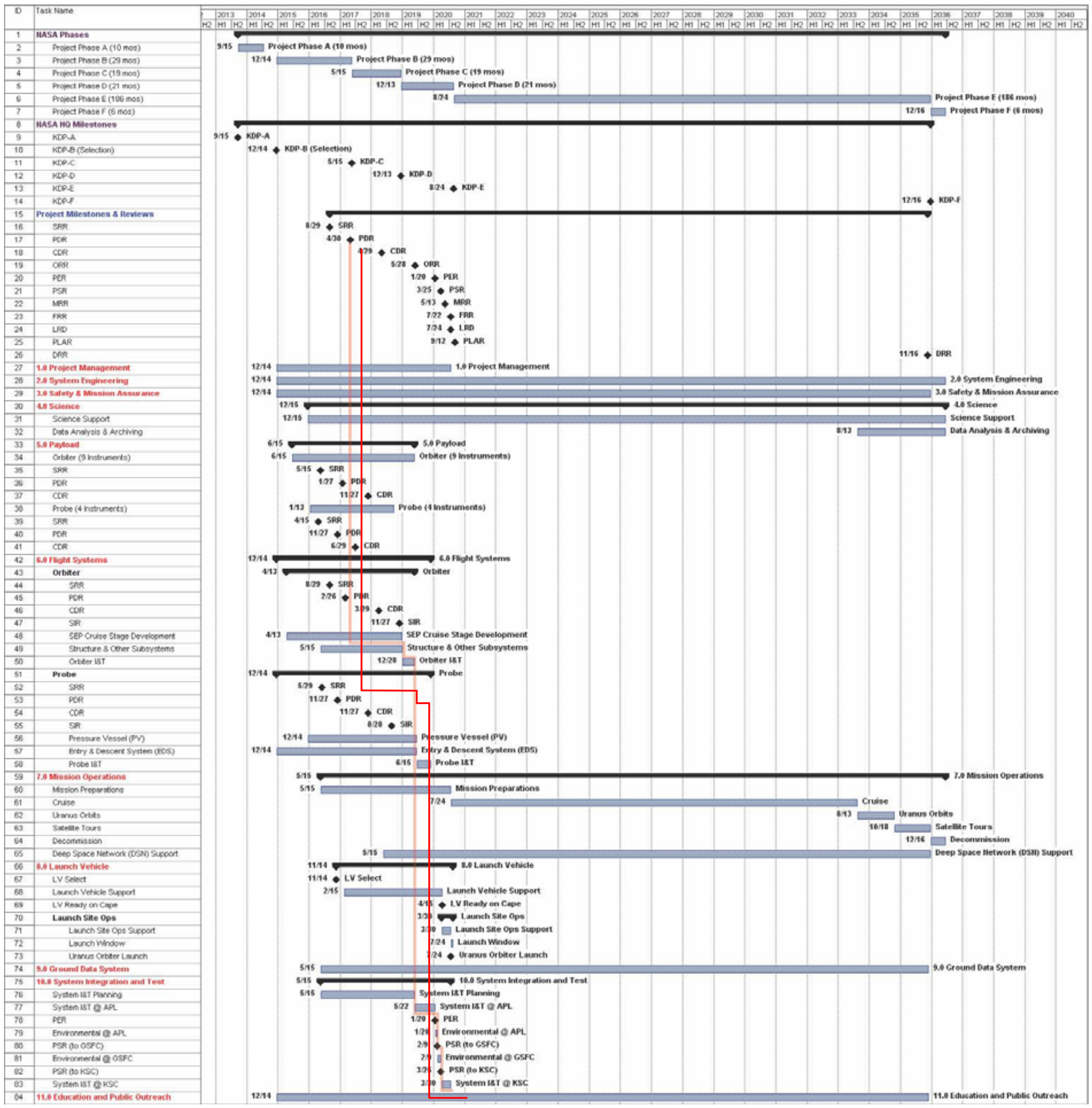

Figure 4-1. High-level mission schedule.

Key phase duration is shown in Table 4-1.

Table 4-1. Key phase duration.

Project Phase	Duration (Months)
Phase A – Conceptual Design	10
Phase B – Preliminary Design	29
Phase C – Detailed Design	19
Phase D – Integration & Test	21
Phase E – Primary Mission Operations	186
Phase F – Extended Mission Operations	6
Start of Phase B to PDR	29
Start of Phase B to CDR	41.1
Start of Phase B to Delivery of Instrument #1 (Orbiter Instr. Payload)	54
Start of Phase B to Delivery of Instrument #2 (Probe Instr. Payload)	46
Start of Phase B to Delivery of Flight Element #1 (Orbiter)	54
Start of Phase B to Delivery of Flight Element #2 (Probe)	60.4
System Level Integration & Test: Orbiter Probe System Total	 5 6 +14 25
Project Total Funded Schedule Reserve	59.7
Total Development Time Phase B–D	69

Technology Development Plan

All technology discussions are provided earlier in Section 2.

Development Schedule and Constraints

A detailed schedule for Phases C/D of the mission is shown in Figure 4-1. Both Payload and Flight Systems are broken down into respective Orbiter and Probe components. The schedule contains a total of 59.7 months of funded schedule reserve (a total of 23 weeks on the critical path) and includes identification and early order of long lead materials. The launch opportunity is from 7/24/2020 to 8/13/2020. Additional launch opportunities with 21-day launch windows exist every subsequent year (see Mission Design section).

5. Mission Life-Cycle Cost

Costing Methodology and Basis of Estimate

The cost estimate prepared for the Ice Giants Decadal Survey mission is of CML 4. It expands in fidelity and detail on the type of cost estimates prepared for the CML-3 Uranus Decadal Survey in winter, 2010. A CML-4 estimate describes the resources required for a preferred design point and takes into account subsystem level mass, power, performance, and risk. In some cases, our estimate takes into account the technical and performance characteristics of components. Estimates for Mission Operations elements whose costs are primarily determined by labor take into account the Phase A–D schedule and Phase E timeline.

The result is a mission estimate that is comprehensive and representative of expenditures that might be expected if the Ice Giants mission is executed as described as above.

Background

A team of JHU/APL and Glenn Research Center (GRC) estimators evaluated mission costs using technical, schedule, performance, and risk inputs from information presented in this report. GRC was responsible for the SEP cruise stage; APL, for all other elements.

The starting point for the mission cost estimate was the cost estimating model for Option 4C of the prior Uranus Decadal Study. The model was updated and expanded. Subsystem elements were updated to reflect selected technologies and changed mission characteristics. The SEP cruise stage estimate is the result of a NAFCOM (NASA/Air Force COst Model) model developed specifically for the Ice Giants study. The cost of the Uranus cruise stage was based on a NAFCOM model developed for an Enceladus mission.

The Ice Giants orbiter differs from the Uranus Option 4C orbiter in propulsion technology and electrical power. The revised orbiter cost model includes an estimate for an advanced bipropellant propulsion subsystem using AMBR engines and an electrical power subsystem that derives its power from multiple ASRGs.

The Uranus Decadal Study probe was described as similar to the Galileo mission probe, and the latter's cost was used as a cost analogy for a Uranus probe. The Ice Giants probe is smaller and required to withstand much less pressure, more like the Pioneer Venus Small Probes. Accordingly, cost data from the latter was used to estimate the Ice Giants probe.

Mission Ground Rules and Assumptions

- Estimating ground rules and assumptions are derived from revision 2 draft of "Groundrules for Mission Concept Studies In Support of Planetary Decadal Survey (Groundrules)."
- Mission costs are reported using the level-2 work breakdown structure (WBS) provided in Revision D of NPR 7120.5. Additional details are available on request.
- Responsibility for the mission is spread throughout the NASA community. JHU/APL will lead the Ice Giants mission and design, develop, manufacture, integrate, and test the orbiter and probe. It will also lead mission operations during Phase E. GRC will design and direct the development and delivery of the SEP cruise stage. GRC will also be responsible for delivering the orbiter's ASRGs. A number of organizations, including JHU/APL, will design, develop, and deliver for integration orbiter and probe instruments.
- Cost estimates are reported in Fiscal Year 2015 (FY15) dollars and Then-Year dollars. Most costs were estimated in FY10 dollars, enabling comparison with current activities and vendor prices and other crosschecks. The FY10 estimates were transformed into FY15 and Then-Year dollars using inflation rates presented in the Decadal Survey Groundrules.
- The 2009 NASA New Start inflation index for 2010 was used to adjust historical cost and price data and parametric results to FY10 dollars.
- The mission does not require Technology Development dollars to advance components to TRL 6 as all Ice Giants mission components will be at or above TRL 6 when required. At present, lightweight Ultraflex arrays of sufficient dimensions to support the Ice Giants mission would be considered TRL 4–5. Our assumption is that at least one planned flagship mission will advance the technology to TRL 6.
- Phase A–D cost reserves are calculated as 50% of the estimated costs of all components except for the launch vehicle and ASRGs.

- Phase E cost reserves are calculated as 25% of the estimated costs of all Phase E elements excluding DSN charges.

Method

The Ice Giants mission cost estimate is a combination of parametric, engineering (bottom-up), and analog techniques. The following paragraphs describe the basis of estimate for each element.

Phase A. $6 million is assumed to be available for Phase A consistent with past large-scale missions. Such a budget will provide sufficient funds for both JHU/APL and GRC managers and engineers to begin the mission concept work. Because all components and subsystems will be at or above TRL 6 at the start of the mission, no Technology Development funds will be required.

WBS 01 Management. This element covers business and administrative planning, organizing, directing, coordinating, analyzing, controlling, and approval processes used to accomplish overall project objectives that are not associated with specific hardware or software elements. This element includes project reviews and documentation, non-project owned facilities, and project reserves. It excludes costs associated with technical planning and management and costs associated with delivering specific engineering, hardware, and software products during Phases B–D.

Project management of the Ice Giants mission is estimated as the sum of two components:

1. **NASA-Glenn Research Center's project office and technical oversight,** which is estimated as 10% of the cruise stage hardware cost and
2. **JHU/APL's program management** effort, which is estimated as a percentage (12%) of orbiter and probe hardware costs.

The factors are based on analysis of the management costs for spacecraft development led by the two organizations. The JHU/APL factor exceeds JHU/APL's share of MESSENGER, STEREO, and New Horizons program management by 2–4%. The difference accounts for various NASA management costs not incurred by JHU/APL and for the costs of meeting current NASA re Earned Value Management (EVM) requirements.

WBS 02 Systems Engineering. The systems engineering element covers technical and management efforts during Phases B–D of directing and controlling an integrated engineering effort for the Ice Giants mission. It includes the efforts to define the project spaceflight vehicle(s) and ground system, conducting trade studies, the integrated planning and control of the technical program efforts of design engineering, software engineering, specialty engineering, system architecture development and integrated test planning, system requirements writing, configuration control, technical oversight, control and monitoring of the technical program, and risk management activities. It also covers mission design and analysis and navigation support (MD&A/NS) through the end of Phase D.

General systems engineering effort is estimated as the sum of two components:

1. **JHU/APL's portion of the Ice Giants systems engineering**, calculated as approximately 14% of orbiter and probe costs. The factor is based on analysis of JHU/APL's MESSENGER, STEREO and New Horizons missions, adjusted for additional effort to comply with additional documentation requirements from revision D of NPR 7120.5.
2. **GRC's portion**, estimated using its NAFCOM parametric model for the cruise stage.

Estimated MD&A/NS cost is based on an engineering estimate of labor requirements by activity over the Phase A–D timeline. The effort averages 2.5 mission analysts and 2.5 contracted navigation experts through Phase D, increasing to an average of 2.5 mission analysts and 3.5 navigation experts during Phase E.

WBS 03 Safety and Mission Assurance (S&MA). This element covers technical and management efforts of directing and controlling the safety and mission assurance elements of the mission during Phases B–D. It includes design, development, review, and verification of practices and procedures and mission success criteria intended to assure that the delivered spacecraft, ground systems, mission operations, and payloads meet performance requirements and function through the end of Phase E.

S&MA is estimated as the sum of two components:

1. **JHU/APL's portion**, estimated as approximately 12.5% of hardware and software costs. That percentage, which is based on an analysis of recent JHU/APL missions including RBSP.
2. **GRC's oversight of the cruise stage contractor**, estimated as 2% of applicable hardware and software costs. The percentage is based on analysis of recent JHU/APL experience managing major subcontractors.

Oversight of the instrument providers is included in the Payloads element.

WBS 04 Science/Technology. This element covers the managing, directing, and controlling of the science investigation aspects, as well as leading, managing, and performing the technology demonstration elements through Phase D. It includes the costs of the Principal Investigator (PI), Project Scientist (PS), science team members including Co-Investigators (Co-Is), and supporting personnel such as payload planners.

Science/technology costs for the Ice Giants mission are estimated using a bottom-up approach. The cost per staff month for each labor category, including an adjustment for travel, is multiplied by the number of scientists in the labor category and the activity duration in months. Labor costs are then summed. The activity of the science team increased from Phase A to one PI, one PS, an average of five Co-Is, and five instrument engineers and planners.

Approximately $1.5 million is included in this element to cover the oversight and development of the Science Operations Center. This is a requirement distinct from the Ground Data Systems.

WBS 05 Payloads. Payloads (WBS 05) includes the probe and instruments hosted on the orbiter and the probe. The basis for estimating each is described below.

Probe. The probe's cost estimate is derived in large measure from cost data provided by the NAFCOM 2008 for the Pioneer Venus Small Probe. As noted above, the pressures to which the Ice Giants probe will be subjected are more comparable to those experienced by the Pioneer Venus Small Probe than by, for example, the Galileo and Pioneer Venus Large Probes. The Ice Giants probe is also much smaller than the latter probes. The three Pioneer Venus Small Probes, while designed and built 4 decades ago, seem more similar technically to the Ice Giants Probe than the recent probes.

NAFCOM 08 provides weight-based cost estimating equations of the form:

$$\text{Design \& Development Cost (FY04\$M)} = a * \text{Weight(in pounds)}^{b}$$

and

$$\text{Flight Unit Cost (FY04\$M)} = a * \text{Weight(in pounds)}^{b}$$

that reproduce the costs of many Pioneer Venus Small Probe subsystems and components. These equations and other Pioneer Venus Small Probe cost data provided the bulk of the estimates for Probe hardware as shown in Table 5-1.

Table 5-1. Probe components: Cost estimating approaches.

Probe Subsystem/Component	Method	Heritage Component (Mission)
Structures & Mechanisms: Entry Descent & Landing		
Aeroshell	NAFCOM 08 weight-based cost equations	Pioneer Venus Small Probe Thermal
Backshell	NAFCOM 08 weight-based cost equations	Pioneer Venus Small Probe Thermal
Thermal Protection System	NAFCOM 08 weight-based cost equations	Pioneer Venus Small Probe Thermal
Separation System (Aeroshell)	Analogies (no weight specified)	Pioneer Venus Small Probe Pyrotechnic Control System
Parachute Decent System	NAFCOM 08 weight-based cost equations	Genesis Probe Parachute
Structures & Mechanisms: Probe Structure		
Pressure Vessel	Analogy (no weight specified)	Pioneer Venus Small Probe Pressure Vessel
Secondary Structure	NAFCOM 08 weight-based cost equations	Pioneer Venus Small Probe Structure & Mechanical
Window/Inlet Release System	Analogy (no weight specified)	Pioneer Venus Small Probe Window
Ultra-Stable Oscillator (USO)	Analogy to USO [APL]	New Horizons USO
Command & Data Handling		
C&DH	PRICE-H model: slice-based module	APL ICE Box electronics
Electric Power		
Battery	PRICE-H model: cell integration & testing	International Lunar Network cost analysis
Thermal Control System (TCS)		
MLI, foam, RHU, etc.	NAFCOM 08 weight-based cost equations	Pioneer Venus Small Probe Thermal
Harness		
Harness	Analogy (no weight specified)	Pioneer Venus Small Probe Harness

Exceptions to the dependence on Pioneer Venus Small Probe cost history are the C&DH subsystem and the battery. The C&DH subsystem design is based on the “ICE Box” slice architecture used for recent JHU/APL instrument electronics. The C&DH cost estimate is based on a PRICE-H parametric model that has been calibrated against slice cost data provided by JHU/APL’s instrument group. A calibrated PRICE-H cost model for batteries constructed of exotic, high-energy lithium ion cells developed for the analysis of lunar power solutions is used to estimate the probe battery.

Factors for management, systems engineering, and S&MA are applied to the summed hardware costs to estimate costs for the probe management and engineering effort. A factor of 4% of hardware costs is also added to the estimate to account for the acquisition of ground support equipment (GSE).

Instruments. The approach used to estimate the costs of each instrument is described in Table 5-2. Baseline mission instruments are bolded. The instrument costs reported in the Ice Giants estimate cover development, design, manufacture, integration and test of one engineering model and one flight unit. It also includes management, engineering and quality assurance efforts.

Table 5-2. Orbiter instruments: Cost estimating approaches (baseline instruments in bold).

Instrument	Method [Source for cost data]	Heritage Instrument (Mission)
Narrow Angle Camera (NAC)	Analogy to NAC portion of MDIS [NICM III, APL]	MDIS (MESSENGER)
Wide Angle Camera (WAC)	**Analogy to LORRI [JHU/APL]**	**LORRI (New Horizons)**
Visible/Near-IR Mapping Spectrometer	**NICM III System-Level Parametric Model--Optical**	**RALPH (New Horizons)**
Magnetometer	**Analogy [NICM III, JHU/APL]**	**MESSENGER MAG**
Magnetometer Boom	**Vendor prices [MESSENGER, RBSP cost data]**	**Cassini boom**
Mid-Infrared Thermal Detector	Analogy to Mars Climate Sounder (MCS) – nearly identical to DIVINER [NICM III cost data]	DIVINER (Juno)
UV Imaging Spectrograph	NICM III System-Level Parametric Model--Optical	ALICE (New Horizons)
Plasma Instrument #1	Analogy to SWAP [JHU/APL]	SWAP (New Horizons)
Plasma Instrument #2	Analogy to JEDI	JUNO JEDI
Ultra-Stable Oscillator (USO)	Analogy to USO [JHU/APL]	New Horizons USO

Cost data for the instruments with obvious analogies are drawn from two data sources—the NICM III instrument data base and JHU/APL cost files. The NICM III data, which has been normalized to FY04 dollars, is adjusted to FY10 dollars. The JHU/APL cost data, which were provided in spend-year dollars, are adjusted to FY10 dollars. Because of overlaps, some crosschecking of cost data for specific instruments is possible. See Table 5-3.

Table 5-3. Probe instruments: Cost estimating approaches (baseline instruments in bold).

Instrument	Method [Source for cost data]	Heritage Instrument (Mission Probe)
Mass Spectrometer	**Analogy to Mass Spec (Cassini probe)**	**MS (Galileo, Cassini)**
Atmospheric Structure Instruments	**Engineering estimate**	**(Galileo)**
Nephelometer	*NICM III System-Level Parametric Model--Optical*	*Nephelometer (Pioneer Venus)*
Ultra-Stable Oscillator (USO)	*Analogy to USO [JHU/APL]*	*USO (New Horizons spacecraft)*

The anticipated TRLs of the Ice Giant instruments ranged from 7 for most instruments to 8 for the magnetometer to 9 for the USO. In adjusting the analog cost, little cost savings credit was given for TRLs greater than TRL 6. We assume that even high-TRL instruments will have to identify and qualify new parts, adjust drawings, and reconstruct test plans and the like, so discounting costs appears optimistic in most cases. Costs of analogous instruments are, however, adjusted to provide a better estimate of the corresponding Ice Giant instruments mass, power, and performance characteristics, which differ significantly.

WBS 06 Spacecraft. GRC estimated the cost of the Cruise Stage; JHU/APL, that of the Orbiter.

Cruise Stage. GRC's assumption is that a contractor will develop, design, integrate, and check-out the cruise stage under the center's management and oversight. The cruise stage flight unit will be delivered to JHU/APL for integration with the orbiter and probe.

GRC estimated the cruise stage hardware costs using the NAFCOM 08 parametric cost model. The model has been calibrated using SEP component cost data. NAFCOM 08 estimates the design and development and flight unit production costs for the cruise stage. The cost estimate covers a prototype unit and flight unit.

GRC assumes that the contractor responsible for producing the cruise stage will add a 10% fee to its hardware, software, and management costs. That fee is added to the NAFCOM results.

Orbiter. JHU/APL will be designing, developing, manufacturing, and integrating and testing the orbiter.

This element also includes all design, development, production, assembly, test efforts, and associated test beds and ground support equipment to deliver the completed system for integration with the launch vehicle and payload.

Table 5-4 lists the estimating methods for each subsystem.

Table 5-4. Orbiter: Cost estimating approaches.

Subsystem/Component	Method [Source for cost data]	Heritage (crosschecks)
Structure & Mechanical	PRICE-H parametric model	
Propulsion	Engineering estimate including component costs, vendor integration RoMs, JHU/APL engineering labor	Various
Guidance & Control	Engineering estimate based on vendor component costs, JHU/APL engineering labor	MESSENGER (labor costs)
IEM/Avionics	PRICE-H parametric model, vendor ROM (IEM)	
Power System Electronics	PRICE-H parametric model (board level)	RBSP (board costs), mission actuals
Power Distribution Unit (PDU)	PRICE-H parametric model (board level)	RBSP (board costs), mission actuals
Test Beds	Engineering estimate: non-recurring labor, parts counts & prices	
Thermal Control	PRICE-H parametric model	
RF Communications	Analogy to MESSENGER	
Harness Assembly	PRICE-H parametric model	
Flight Software (FSW) Development	Engineering Estimate, based on reuse of software	

The dual-mode propulsion subsystem was estimated using an engineering estimate to ensure that component prices are accounted for properly. The estimated cost of $30.4 million in FY15 dollars is based on a purchased subsystem where the propulsion subcontractor is responsible for procuring components and integrating them into the orbiter structure. The subsystem cost also includes the effort of JHU/APL propulsion engineers, who will be responsible for design of the subsystem and technical management of the contract. The cost also covers expenditures for pneumatic GSE required for leak and functional testing.

Key propulsion subsystem assumptions that affect the estimated cost include the following: All components, including off-the-shelf propellant tanks, are assumed to be at least TRL 6. In other words, no additional qualification testing will be required. If the AMBR engines are not qualified prior to their installation, the mission would have to pay an additional $5 million cost for their qualification. No propulsion subsystem-level thermal balance test will be required, and propulsion thermal vacuum testing will be done at spacecraft level. Propellant loading (including any water loading/off-loading required for dynamic testing) will be included in the launch vehicle contract. Finally, all propellant and pressurant costs are included in the launch vehicle contract.

ASRGs. The Advanced Stirling Radioisotope Generators (ASRGs) that power the orbiter are described in the Decadal Survey Ground Rules. Following the Ground Rules, we assume that “the ASRG will be ready for flight no earlier than March 2014 and will have a unit cost of ~$20M.” That $20M cost is presumed to be in FY15 dollars. We assume that the cost included required engineering, mass, and thermal models when required for testing.

***WBS 07 Mission Operations*.** This element covers the management of the development and implementation of personnel, procedures, documentation, and training required to conduct mission operations. Its efforts span all phases of the mission.

Mission operations includes the following elements:

- Operations personnel
- Launch checkout, early operations support (LCEOS)
- Management, sustaining engineering , and S&MA support (Phase E)
- Mission design & analysis and navigation support (Phase E)
- Science Team activity (Phase E)

The first two elements include effort required before the start of Phase E. Missions operations span Phases D and E as personnel must plan and train before the launch. LCEOS is Phase D only: It begins after environmental testing of the space vehicle is complete and ends after launch with the post-launch space vehicle checkout. The other three activities cover strictly Phase E effort.

The estimated cost of the LCEOS activity is 5% of the spacecraft hardware and software cost. The other activities are engineering estimates, based on a level of effort that reflects the mission timeline and key events.

During the 6-plus-year hibernation cruise, costs and labor are minimized. The science team during hibernation, for example, is reduced to a total of half-time combined for the PI and project scientist, two full-time-equivalent Co-Is, and one full-time-equivalent instrument planner. During the orbital science activity, that team expands to a full-time PI, a full-time project scientist, and 20 Co-Is and 3 instrument planners. Similar adjustments apply to operations personnel.

WBS 08 Launch Vehicle and Services. The mission requires a launch vehicle with a 5-mr fairing and lift capacity equal to the Option 5 launch vehicle described in the Decadal Survey Ground Rules. The cost of that launch vehicle is $257 million in what are assumed are FY15 dollars.

WBS 08 includes the costs of a part-time engineer responsible through Phases B–D for interfacing the space vehicle to the launch vehicle. Per the Decadal Survey Ground Rules (c.f. page 3, Table 2, "Launch Services Pricing Estimates" — caption), because the mission includes a radioactive component, WBS 08 also includes $15 million to ensure that the launch complies with NEPA and other safety requirements.

WBS 09 Ground Data Systems (GDS). This element includes the computers, communications, operating systems, and networking equipment needed to interconnect and host the Mission Operations software. It covers the design, development, implementation, integration, test, and the associated support equipment of the ground system, including the hardware and software needed for processing, archiving, and distributing telemetry and radiometric data and for commanding the spacecraft.

A bottom-up estimate for an outer planetary mission ground data system (GDS) was generated using vendor prices for hardware components and licenses and software development labor cost data. For ground software development, we assume that the L3 InControl satellite control software will provide the GDS framework and that InControl-compatible routines developed for the RBSP mission can be reused with minor modifications.

WBS 10 System Integration and Test (I&T). This element covers the efforts to assemble the orbiter, cruise stage, and probe; integrate the three into the mission space vehicle; and perform space vehicle environmental testing. The costs are based on a detailed analysis of the STEREO mission I&T cost history and comparison with RBSP I&T plans. The analysis provides an estimated cost for integrating and testing a "standard" JHU/APL space vehicle as well as identifies factors that trigger adjustments to the typical space vehicle I&T cost.

The Ice Giant space vehicle I&T effort is estimated as if the orbiter-probe combination and the cruise stage require non-recurring planning efforts. Recurring effort will be required comparable to that for three JHU/APL spacecraft. A 12% complexity factor ($3.2 million) is added to the orbiter-probe effort to cover the additional costs for ASRG-orbiter integration and orbiter-probe coupling. Another $1.3 million is included in the estimate to pay for testing the probe in an extremely-high-*g* environment. The element also includes cruise-stage-specific costs for System Test Operations and Integration, Assembly and Check-Out estimated with NAFCOM.

DSN (Deep Space Network) Charges. This element provides for access to the DSN 34-m communications infrastructure needed to transmit and receive mission and scientific data. Mission charges for use of a 34-m dish are estimated using the current DSN rate schedule and a table of DSN connection requirements derived from the Ice Giants mission timeline. The DSN cost estimate covers pre- and post-contact activity for each linkage.

E/PO (Education and Public Outreach). This element provides for the education and public outreach (EPO) responsibilities of NASA's missions, projects, and programs in alignment with the Strategic Plan for Education. Available E/PO funds are calculated as 1% of the costs of non-reserve mission costs.

Cost Estimate(s)

The total mission cost of the Ice Giants mission will be slightly less than \$1.9 billion in FY15 dollars (FY10\$1.7 billion). The most expensive single cost element is cost reserves, 27% of the total mission cost. The next most expensive cost elements are the spacecraft—the combination of orbiter and cruise stage—and the launch vehicle.

Phase E/F costs, including mission operations and DSN charges and cost reserves, account for \$201 million, or 11% of the mission’s total estimated costs. The relatively low Phase E/F cost projection depends on achieving the low mission operations tempo projected for the six-plus-year hibernation cruise.

Cost estimates are shown in Table 5-5.

Table 5-5. **Cost estimates**.

	Uranus Orbiter	Costs by year in spend-year dollars																									
NASA WBS	Description	FY2013	FY2014	FY2015	FY2016	FY2017	FY2018	FY2019	FY2020	FY2021	FY2022	FY2023	FY2024	FY2025	FY2026	FY2027	FY2028	FY2029	FY2030	FY2031	FY2032	FY2033	FY2034	FY2035	FY2036	Total in SY $M	Total in FY15 $M
	Phase A	1	6	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	6	6
	Technology Development	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
01	Project Management	1	5	6	8	8	9	9	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	48	48
02	Systems Engineering, incl. MD&A & Nav.	1	4	7	14	15	13	13	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	68	67
03	Safety & Mission Assurance	0	1	3	10	11	12	12	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	53	51
04	Science/Technology (Phases A-D)	0	1	1	2	3	4	4	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	15	15
05	Payloads	0	0	7	35	37	17	17	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	118	116
05	Probe	0	0	9	48	51	23	23	6	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	160	157
06	Spacecraft	2	3	19	86	155	122	4	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	393	381
07	Mission Operations	0	0	0	0	0	0	0	27	10	11	11	11	9	9	9	9	9	9	10	14	20	17	16	9	209	159
08	Launch Vehicles & Services, incl. LVA, I/F, NEPA	0	1	1	3	3	53	112	114	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	288	273
09	Ground Data Systems	0	0	0	2	3	5	6	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	18	17
10	Systems Integration & Test	0	0	1	2	3	20	23	10	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	60	57
DSN	Space Communications Services (DSN)	0	0	0	0	0	0	0	2	1	1	1	4	1	2	1	1	1	1	2	2	6	10	10	2	48	34
E/PO	E/PO	0	0	0	0	1	3	4	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	12	11
	Subtotal	4	22	55	211	290	280	225	180	12	12	12	15	10	11	9	10	10	10	12	17	25	27	26	11	1496	1392
	Phases A-D	*4*	*22*	*55*	*211*	*290*	*277*	*222*	*167*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*0*	*1248*	*1208*
	Excluding LV	*4*	*20*	*53*	*205*	*267*	*177*	*120*	*153*	*12*	*12*	*12*	*15*	*10*	*11*	*9*	*10*	*10*	*10*	*12*	*17*	*25*	*27*	*26*	*11*	*1228*	*1119*
	Cost Reserves	2	10	27	102	133	87	58	72	3	3	3	3	2	2	2	2	2	2	3	4	5	4	4	2	537	502
	Phases A-D (excl. LV) : 50%	2	10	27	102	133	87	58	70	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	490	467
	Phase E: 25%	0	0	0	0	0	0	0	2	3	3	3	3	2	2	2	2	2	2	3	4	5	4	4	2	47	35
	Total, including Reserves	6	32	82	313	423	367	284	252	14	15	15	17	13	13	12	12	12	13	14	20	30	31	30	13	2033	1894

Confidence and Cost Reserves

Per the Decadal Study Ground Rules, the estimate includes cost reserves equal to 50% of the estimated costs of all Phase A–D elements except for the launch vehicle, ASRGs, and DSN charges plus 25% of the estimated costs of Phase E/F elements, excluding Phase E DSN.

A probabilistic cost estimate was not prepared for the total mission, and a numeric confidence level was not assigned.

Compared with cost estimates prepared for the CML-3 Uranus Decadal study, the biggest single change is the decrease in the estimated cost of the probe. The Uranus study assumed that the probe would be analogous to the Galileo probe, a large probe designed to survive at very high atmospheric pressures. The Ice Giants probe has been based on the smaller, less expensive Pioneer Venus Small Probes whose pressure vessels were designed to withstand the lesser pressures of the upper Venusian atmosphere.

Estimated Savings from Mission Descopes

Approximate cost savings for three mission descopes are as follows:

Elimination of the Satellite Tour. Savings from not conducting the 61-week satellite tour at the end of Phase E accrue from elimination of

- Management and engineering support
- Mission design, analysis and navigation activity
- Science Team activity
- DSN contacts
- Mission operations personnel

Including its prorated share of cost reserves, total savings would be approximately $26 million in FY15 dollars.

Elimination of the Probe. Savings from the probe would result in the following:

- No probe
- No probe instruments
- Reduced mission management, systems engineering and S&MA effort
- Reduced, simplified space vehicle I&T
- Elimination of the probe operations activity after arrival at Uranus

The baseline savings is estimated to be about $215 million in FY15 dollars. Taking into account cost reserves, total savings would be about $310 million.

Elimination of Non-Baseline Instruments. The mission could save a total of more than $120 million in FY15 dollars, including reserves, by eliminating the following non-baseline instruments:

(Orbiter)

1. Narrow-angle camera (NAC)
2. UV imaging spectrograph
3. Mid-IF thermal detector
4. Plasma Instrument #1
5. Plasma Instrument #2
6. USO

(Probe)

7. USO
8. Nephelometer

Appendix A: Ice Giants Decadal Study Team

Role	Name
Uranus / Science Champion	William B. Hubbard
Neptune / Science Champion	Mark Marley
Giant Planets Chair	Heidi B. Hammel
Giant Planets Vice-Chair	Amy A. Simon-Miller
Panel Members	Krishan Khurana
	Brigette Hesman
	John Clarke
NASA HQ POC	Leonard Dudzinski

Role	Name	Organization
Decadal Program Manager	Kurt Lindstrom	JHU/APL
APL Science POC	Elizabeth Turtle	JHU/APL
Project Manager	Helmut Seifert	JHU/APL
Lead Systems Engineer	Doug Eng	JHU/APL
Systems / Instruments	Rob Gold Elena Adams	JHU/APL
Systems Engineer	Darryl Royster	JHU/APL
Systems Engineer / SEP	Steve Oleson Melissa Mcguire David Grantier	NASA/GRC
Mission Design	Yanping Guo John Dankanich Chris Scott Ryan Russell	JHU/APL NASA GRC JHU/APL Georgia Tech
SEP Configuration	Jon Drexler	NASA/GRC
Cost	Larry Wolfarth Sally Whitley Jon Drexler	JHU/APL JHU/APL NASA/GRC
Risk/Reliability	Melissa Jones Anita Tenteris	JHU/APL NASA/GRC

Role	Name	Organization
Aerocapture/Aeroentry	Mary Kae Lockwood Dick Powell David Way	JHU/APL NASA/LaRC NASA/LaRC
RF	Brian Sequeira Joe Warner	JHU/APL NASA/GRC
GN&C	Robin Vaughn Wen-Jong Shyong Mike Martini	JHU/APL JHU/APL NASA/GRC
Operations	Rick Reinders	JHU/APL
Integration & Test	Jay White	JHU/APL
Mechanical Design	Scott Cooper Dave Napolillo John Gyekenyesi	JHU/APL JHU/APL NASA/GRC
Stress Analyst	Marc Briere	JHU/APL
Thermal Analyst	Elisabeth Abel Tony Colozza	JHU/APL NASA/GRC
Software	Steve Williams	JHU/APL
Avionics	Marty Fraeman Glenn L. Williams	JHU/APL NASA/GRC
Power	Marty Fraeman James Fincannon Kristen Bury	JHU/APL NASA/GRC NASA GRC
Propulsion	Stewart Bushman James Fittje	JHU/APL NASA/GRC

SEP Stage Master Equipment List

SEP Subsystem/Component	Unit Mass CBE (kg)	# OF UNITS: Flight Units	Flight Spares	EMs & Proto-types	FLIGHT HARDWARE MASSES: Total CBE Mass (kg)	Contingency	Total MEV Mass (kg)	FLIGHT HARDWARE POWER: Total CBE Steady-State Power	Contingency	Total MEV Steady-State Power	OTHER COMPONENT INFORMATION: Description (Vendor, Part #, Heritage Basis)		TRL	Other Characteristics/ Issues
Structures & Mechanisms					129.62 kg	18%	152.36 kg	0.0	-	0.0				
Primary Structure	39.95 kg	1	0	0	39.95 kg	18%	47.15 kg	n/a	-				7	
Secondary Structure	46.02 kg	1	0	0	46.02 kg	18%	54.30 kg	n/a	-				7	
Mechanisms (Solar Array, Separation,	43.65 kg	1	0	0	43.65 kg	17%	50.92 kg	n/a	-		Moog (may need resizing)		8	
Propulsion (Dry Mass)					315.07 kg	8%	338.78 kg	0.0	-	0.0				
Propulsion Hardware (EP)	55.50 kg	1	1	1	55.50 kg	8%	59.94 kg	n/a	-		NEXT		6	Under development - NASA GRC
Propellant Management (EP)	131.67 kg	1	1	1	131.67 kg	5%	138.69 kg	35.00 w	30%	45.50 w	VACCO AXT		6	Under development - NASA GRC
Power Processing Unit (PPU)	127.90 kg	1	0.2	1	127.90 kg	10%	140.15 kg	7080.00 w	30%	9204.00 w	NEXT PPU		6	Under development - NASA GRC
Command & Data Handling (C&DH)					11.48 kg	16%	13.33 kg	0.0	-	0.0				
Command and Telemetry Computer	1.00 kg	2	1	1	2.00 kg	18%	2.36 kg	20.00 w	5%	21.00 w	Various		8	
Command and Control Harness (data)	5.00 kg		0	0	0.00 kg	25%	0.00 kg	n/a	5%				7	Tallied in Harness section at bottom
cPCI enclosure with power supply	5.00 kg	1	0	0	5.00 kg	15%	5.75 kg	3.00 w	5%	3.15 w	Various		8	
Instrumentation & Wiring	4.48 kg	1	0	0	4.48 kg	17%	5.22 kg	24.00 w	5%	25.20 w			7	
Electrical Power (EPS)					188.00 kg	30%	244.40 kg	0.0	-	0.0				
Solar Arrays (Ultraflex)	68.50 kg	2	0	1	137.00 kg	30%	178.10 kg	40.00 w	30%	52.00 w	Phoenix, ST-9, Orion		5	Crew Exploration Vehicle et al
Power Management & Distribution	27.00 kg	1	0.5	1	27.00 kg	30%	35.10 kg	30.00 w	30%	39.00 w	Dawn		6	
Power Cable and Harness Subsystem (C	24.00 kg		0	0	0.00 kg	50%	0.00 kg	n/a	30%				7	Tallied in Harness section at bottom
Battery System	24.00 kg	1	1	1	24.00 kg	30%	31.20 kg	n/a	30%		RBSP, Mars Phoenix lander		7	LiIon
Guidance, Navigation, and Control					2.48 kg	3%	2.55 kg	0.0	-	0.0				
Sun Sensors	0.62 kg	4	0	0	2.48 kg	3%	2.55 kg	n/a	30%				9	
Thermal Control (TCS)					87.22 kg	18%	102.92 kg	0.0	-	0.0				
Active Thermal Control (heaters ...)	3.16 kg	1	0	0	3.16 kg	18%	3.73 kg	15.40 w	5%	16.17 w	Flown		9	
Passive Thermal Control (MLI ...)	67.36 kg	1	0	1	67.36 kg	18%	79.48 kg	n/a	5%		Flown		7	
Semi-Passive Thermal Control (Louvers ...	16.70 kg	1	0	1	16.70 kg	18%	19.71 kg	n/a	5%		Flown		8	
RF Communications					2.20 kg	39%	3.06 kg	0.0	-	0.0				
Omni-Antennas	0.60 kg	2	0	0	1.20 kg	30%	1.56 kg	n/a	5%		Various		9	
Cables	0.50 kg	2	0	0	1.00 kg	50%	1.50 kg	n/a	5%		Various		8	
Harness					29.00 kg	46%	42.25 kg	N/A	N/A	N/A			7	
Command and Control Harness (data)	5.00 kg	1	see above		5.00 kg	25%	6.25 kg	n/a	5%					from C&DH section
Power Cable and Harness Subsystem (C a	24.00 kg	1	see above		24.00 kg	50%	36.00 kg	n/a	5%					from EPS section
SEP DRY MASS/POWER					765.08 kg	18%	899.66 kg							
Dry Mass Margin					30% (Note 1)		328.98 kg							
SEP Maximum DRY MASS							1094.06 kg							
SEP Consumables							827.97 kg				Consumables	0.00		
Useable Xexon							762.40 kg				SEP Inert Gas	0.00		
Residual Xexon							65.57 kg				SEP Inert Gas	0.00		
SEP (CBE) WET MASS							1593.04 kg							
SEP (Maximum) WET MASS							1922.02 kg							

Appendix B: Master Equipment List and Power Table

Orbiter Master Equipment List

		# OF UNITS			FLIGHT HARDWARE MASSES			FLIGHT HARDWARE POWER			OTHER COMPONENT INFORMATION		
Orbiter Subsystem/Component	Unit Mass CBE (kg)	Flight Units	Flight Spares	EMs & Proto-types	Total CBE Mass (kg)	Contingency	Total MEV Mass (kg)	Total CBE Steady-State Power	Contingency	Total MEV Steady-State Power	Description (Vendor, Part #, Heritage Basis)	TRL	Other Characteristics/ Issues
Instruments					53.50 kg	16%	61.98 kg	0.0	-	0.0			
Wide Angle Camera	3.00 kg	1	0	1	3.00 kg	15%	3.45 kg	3.50 w	30%	4.55 w	MDIS, MESSENGER	7	
Narrow Angle Camera	8.60 kg	1	0	1	8.60 kg	15%	9.89 kg	5.00 w	30%	6.50 w	LORRI, New Horizons	7	
Visible/Near-IR Mapping Spectrometer	10.50 kg	1	0	1	10.50 kg	15%	12.08 kg	7.10 w	30%	9.23 w	RALPH, New Horizons	7	
Mid-Infrared Thermal Detector	8.00 kg	1	0	1	8.00 kg	15%	9.20 kg	7.00 w	30%	9.10 w	DEVINEA, LRO	7	
UV Imaging Spectrograph	4.40 kg	1	0	1	4.40 kg	15%	5.06 kg	4.40 w	30%	5.72 w	ALICE, New Horizons, ROSETTA	7	
Plasma Instrument 1 (SWAP)	3.30 kg	1	0	1	3.30 kg	15%	3.80 kg	3.00 w	30%	3.90 w	SWAP, New Horizons	7	
Plasma Instrument 2 (JEDI)	1.70 kg	1	0	1	1.70 kg	15%	1.96 kg	2.80 w	30%	3.64 w	JEDI, JUNO	7	
Magnetometer	1.00 kg	1	0	1	1.00 kg	15%	1.15 kg	2.70 w	30%	3.51 w	MESSENGER Mag	8	
Magnetometer Boom	10.00 kg	1	0	1	10.00 kg	15%	11.50 kg	n/a	30%		Cassini	8	10 meter
USO	1.50 kg	2	0	0	3.00 kg	30%	3.90 kg	6.00 w	30%	7.80 w	Cassini, New Horizons	9	
Structures & Mechanisms					185.00 kg	15%	212.75 kg	0.0	-	0.0			
Primary Structure	140.00 kg	1	0	0	140.00 kg	15%	161.00 kg	n/a	-		STEREO, New Horizons	7	
Secondary Structure	25.00 kg	1	0	0	25.00 kg	15%	28.75 kg	n/a	-		STEREO, New Horizons	7	
Probe Separation System	5.00 kg	1	0	0	5.00 kg	15%	5.75 kg	n/a	-		Launch Vehicles	7	4 point pyrotechnic
Miscellaneous	15.00 kg	1	0	0	15.00 kg	15%	17.25 kg	n/a	-		Various	7	
Propulsion (Dry Mass)					119.16 kg	6%	125.89 kg	0.0	-	0.0			
TCM Engine (AMBR, 150 lbf)	5.50 kg	1	0	0	5.50 kg	5%	5.78 kg	46.00 w	30%	59.80 w	AMBR	6	
5 lbf Thruster	0.73 kg	4	1	0	2.92 kg	5%	3.07 kg	25.30 w	5%	26.57 w	MR-106E, MESSENGER	9	
ACS Engines (1 lbf)	0.36 kg	12	1	0	4.32 kg	5%	4.54 kg	8.26 w	5%	8.67 w	MR-111C, New Horizons	9	
AMBR Valve Heater	0.00 kg	1	0	0	0.00 kg	0%	0.00 kg	10.00 w	30%	13.00 w	MESSENGER	9	Mass included with thrusters
AMBR Injector Heater	0.00 kg	1	0	0	0.00 kg	0%	0.00 kg	50.00 w	5%	52.50 w	MESSENGER	9	Mass included with thrusters
Cat Bed Heaters (5 lb)	0.00 kg	4	0	0	0.00 kg	0%	0.00 kg	6.54 w	5%	6.87 w	MESSENGER	9	Mass included with thrusters
Cat Bed Heaters (1 lb)	0.00 kg	12	0	0	0.00 kg	0%	0.00 kg	3.86 w	5%	4.05 w	MESSENGER	9	Mass included with thrusters
Thruster Valve Heaters (5 lb)	0.00 kg	4	0	0	0.00 kg	0%	0.00 kg	3.27 w	5%	3.43 w	MESSENGER	9	Mass included with thrusters
Thruster Valve Heaters (1 lb)	0.00 kg	12	0	0	0.00 kg	0%	0.00 kg	1.54 w	5%	1.62 w	MESSENGER	9	Mass included with thrusters
Pressure Transducer	0.23 kg	5	0	0	1.15 kg	5%	1.21 kg	0.90 w	5%	0.95 w	Paine, MESSENGER	9	
Electrical Connectors	0.25 kg	1	0	0	0.25 kg	15%	0.29 kg	n/a	5%		MESSENGER	9	
Propulsion Diode Box	1.50 kg	1	0	0	1.50 kg	15%	1.73 kg	n/a	5%		Modified MESSENGER	9	
Hydrazine (N2H4) Tanks	17.16 kg	2	0	0	34.32 kg	5%	36.04 kg	n/a	5%		ATK 80340, GEA S-5000	9	
Oxidizer Tanks	13.65 kg	2	0	0	27.30 kg	5%	28.67 kg	n/a	5%		ATK 80339, GEA S-5000	9	
Pressurant Tank	12.70 kg	2	0	0	25.40 kg	5%	26.67 kg	n/a	5%		ATK 80436, Astrolink	9	
Latch Valve	0.34 kg	8	0	0	2.72 kg	5%	2.86 kg	n/a	5%		Vacco/Moog, MESSENGER	9	
Hi-Pressure Latch Valve	0.52 kg	2	0	0	1.04 kg	5%	1.09 kg	n/a	5%		Valcor, MESSENGER	9	
Fuel / Ox Service Valve	0.15 kg	12	0	0	1.85 kg	5%	1.94 kg	n/a	5%		Vacco, MESSENGER	9	
Helium Service Valve	0.07 kg	1	0	0	0.07 kg	5%	0.07 kg	n/a	5%		Vacco, MESSENGER	9	
Pyrotechnic Valve	0.21 kg	3	0	0	0.62 kg	5%	0.65 kg	n/a	5%		Conax, MESSENGER	9	
Fuel Check Valve	0.23 kg	3	0	0	0.69 kg	5%	0.72 kg	n/a	5%		Vacco, MESSENGER	9	
Ox Check Valve	0.25 kg	3	0	0	0.75 kg	5%	0.79 kg	n/a	5%		Vacco, MESSENGER	9	
Pressure Regulator	1.20 kg	2	0	0	2.40 kg	5%	2.52 kg	n/a	5%		Stanford Mu, MESSENGER	9	
Filter	0.02 kg	11	0	0	0.22 kg	5%	0.23 kg	n/a	5%		Vacco, MESSENGER	9	
Test Port	0.03 kg	2	0	0	0.06 kg	5%	0.06 kg	n/a	5%		MESSENGER	9	
Cabling	3.00 kg	1	0	0	3.00 kg	15%	3.45 kg	n/a	5%		MESSENGER	9	
Orifice	0.03 kg	3	0	0	0.09 kg	5%	0.09 kg	n/a	5%		MESSENGER	9	
Tubing/Fasteners/clamps/etc.	3.00 kg	1	0	0	3.00 kg	15%	3.45 kg	n/a	5%		MESSENGER	9	

Command & Data Handling (C&DH)					14.65 kg	15%	16.78 kg	0.0	-	0.0				
Integrated Electronics Module	7.00 kg	2	1	5	14.00 kg	15%	16.10 kg	10.00 w	5%	10.50 w	Modified STEREO, MESSENGER		7	
Remote Input output devices	0.13 kg	5	1	5	0.65 kg	5%	0.68 kg	0.15 w	5%	0.16 w			7	
Electrical Power (EPS)					88.50 kg	14%	101.13 kg	0.0	-	0.0				
ASRG	24.00 kg	3	0	0	72.00 kg	15%	82.80 kg	n/a	5%		Under development - NASA GRC		6	Purchase includes models (mech,
Power System Electronics (SRU)	5.00 kg	1	1	1	5.00 kg	15%	5.75 kg	3.50 w	5%	3.68 w	New Horizons		7	
Battery	5.00 kg	1	1	1	5.00 kg	15%	5.75 kg	n/a	5%		RBSP		7	LiIon (16 A-Hr)
Power Distribution Unit (PDU)	6.50 kg	1	1	1	6.50 kg	5%	6.83 kg	10.00 w	5%	10.50 w	RBSP redundant		9	
Guidance, Navigation, and Control					46.10 kg	5%	48.40 kg	0.0	-	0.0				
Star Tracker	1.50 kg	2	0	0	3.00 kg	5%	3.15 kg	6.50 w	5%	6.83 w	AA-STR Galileo		9	
Inertial Measurements Unit	4.44 kg	2	0	0	8.88 kg	5%	9.32 kg	32.00 w	5%	33.60 w	Honeywell MIMU		9	
Sun Sensors	0.04 kg	6	0	0	0.22 kg	5%	0.23 kg	n/a	5%		Aero Astro		9	
Reaction Wheels	8.50 kg	4	0	0	34.00 kg	5%	35.70 kg	7.00 w	5%	7.35 w	Honeywell HR-14		9	
Thermal Control (TCS)					35.00 kg	13%	39.65 kg	0.0	-	0.0				
Thermistors, Thermistats, Heaters, Tape,	4.00 kg	1	0	0	4.00 kg	15%	4.60 kg	40.00 w	10%	44.00 w	Many Missions		7	
Multi-Layer Insulation	15.00 kg	1	0	0	15.00 kg	15%	17.25 kg	n/a	10%		Many Missions		7	
Louvers	1.00 kg	6	1	0	6.00 kg	5%	6.30 kg	n/a	10%		New Horizons		9	
Heat Pipes	5.00 kg	2	0	0	10.00 kg	15%	11.50 kg	n/a	10%		New Horizons		7	
RF Communications					59.99 kg	14%	68.33 kg	0.0	-	0.0				
X-Ka-Band Coherent Tranceivers (A, B)	2.60 kg	2	0	2	5.20 kg	15%	5.98 kg	16.20 w	30%	21.06 w	Connect derivative (JHU/APL)		6	Frontier Radio
X-Band TWTA (with EPC)	2.56 kg	2	0	0	5.12 kg	5%	5.38 kg	116.00 w	5%	121.80 w	STEREO		9	
K-Band TWTA (with EPC)	3.00 kg	2	0	0	6.00 kg	5%	6.30 kg	100.00 w	5%	105.00 w	LRO		9	
HGA reflector Assembly	28.37 kg	1	0	1	28.37 kg	15%	32.63 kg	n/a	5%		New Horizons		7	2.5 m dish
X-Band LGAs	0.35 kg	2	0	1	0.70 kg	5%	0.74 kg	n/a	5%		MESSENGER		9	
X-Band MGA	0.60 kg	1	0	1	0.60 kg	15%	0.69 kg	n/a	5%		New Horizons		8	
RF (Switch) Plate Assembly	5.50 kg	1	0	1	5.50 kg	15%	6.33 kg	n/a	5%		STEREO, MESSENGER		8	
Coax Cables, Waveguide	5.00 kg	1	0	0	5.00 kg	15%	5.75 kg	n/a	5%		Various		8	
Monopulse Combiner Network	3.50 kg	1	0	0	3.50 kg	30%	4.55 kg	n/a	5%		Various		7	
Harness	32.00 kg	1			32.00 kg	15%	36.80 kg	N/A	N/A	N/A			7	5% of dry mass
ORBITER DRY MASS/POWER					633.89 kg	12%	711.71 kg							
Dry Mass Margin					30% (Note 1)		272.57 kg							
ORBITER Maximum DRY MASS							906.47 kg							
Orbiter Consumables							1173.69 kg							
Usable Fuel N2H4							615.68 kg							
Residual Fuel N2H4							4.71 kg				4.62			0.75%
Usable Oxidizer							544.65 kg							
Residual Oxidizer							4.17 kg				4.08			0.75%
Pressurant							4.48 kg							GHe
ORBITER (CBE) WET MASS							1807.58 kg							
ORBITER (Maximum) WET MASS							2080.16 kg							

Entry Probe Master Equipment List

Probe Subsystem/Component	Unit Mass CBE (kg)	# OF UNITS: Flight Units	Flight Spares	EMs & Proto-types	FLIGHT HARDWARE MASSES: Total CBE Mass (kg)	Contingency	Total MEV Mass (kg)	FLIGHT HARDWARE POWER: Total CBE Steady-State Power	Contingency	Total MEV Steady-State Power	OTHER COMPONENT INFORMATION: Description (Vendor, Part #, Heritage Basis)	TRL	Other Characteristics/ Issues
Instruments					14.70 kg	17%	17.13 kg	0.0	-	0.0			
Mass Spectrometer	8.00 kg	1	1	1	8.00 kg	15%	9.20 kg	10.00 w	30%	13.00 w	Galileo, Cassini	6	
Atmospheric Structure Inst. (ASI)	4.00 kg	1	0	1	4.00 kg	15%	4.60 kg	6.30 w	30%	8.19 w	Galileo	7	
Nephlometer	1.20 kg	1	0	1	1.20 kg	15%	1.38 kg	2.40 w	30%	3.12 w	Pioneer Venus	7	
USO	1.50 kg	1	0	0	1.50 kg	30%	1.95 kg	3.00 w	30%	3.90 w	Cassini, New Horizons	7	High G crystal screening
Structures & Mechanisms					49.50 kg	15%	56.93 kg	0.0	-	0.0			
Entry Descent & Landing					35.50 kg	15%	40.83 kg	0.00	-	0.00			
Aeroshell	12.00 kg	1	0	1	12.00 kg	15%	13.80 kg	n/a	-		Scaled from Pioneer/Venus	7	
Backshell	3.50 kg	1	0	1	3.50 kg	15%	4.03 kg	n/a	-		Scaled from Pioneer/Venus	7	
Thermal Protection System	11.00	1	0	0	11.00 kg	15%	12.65 kg	n/a	-		Gallileo, Pioneer	7	Uranus/Neptune TPS paper
Separation System (Aeroshell)	5.00	1	0	0	5.00 kg	15%	5.75 kg	n/a	-			7	
Parachute Descent System	4.00	1	0	1	4.00 kg	15%	4.60 kg	n/a	-		Gallileo	7	
Probe Structure					14.00 kg	15%	16.10 kg	0.00	-	0.00			
Pressure Vessel	5.00 kg	1	0	1	5.00 kg	15%	5.75 kg	n/a	-		Pioneer Venus	7	
Secondary Structure	8.00 kg	1	0	0	8.00 kg	15%	9.20 kg	n/a	-		Pioneer Venus	7	
Window/Inlet Release System	1.00 kg	1	0	0	1.00 kg	15%	1.15 kg	n/a	-		Pioneer Venus	7	
Command & Data Handling (C&DH)					4.00 kg	15%	4.60 kg	0.0	-	0.0			
IEM (processor & power)	4.00 kg	1	1	2	4.00 kg	15%	4.60 kg	3.75 w	30%	4.88 w	RBSP	7	3u card (4x6x4)
Electrical Power (EPS)					8.70 kg	30%	11.31 kg	0.0	-	0.0			
Battery	8.70 kg	1	1	1	8.70 kg	30%	11.31 kg	n/a	5%		SAFT LSH20-150, Delta 181, Ampte	6	64 D cells, Lithium-thionyl Chloride
Thermal Control (TCS)					1.36 kg	15%	1.56 kg	0.0	-	0.0			
MLI Blankets	0.70 kg	1	0	0	0.70 kg	15%	0.81 kg	n/a	10%		Various	7	
Foam	0.50 kg	1	0	0	0.50 kg	15%	0.58 kg	n/a	10%		Mars MER	7	
RHU	0.04 kg	4	0	0	0.16 kg	15%	0.18 kg	n/a	10%		Cassini	9	
Pro					5.21 kg	9%	5.66 kg	0.0	-	0.0			100 bps
Coherent Transceiver	2.20 kg	1			2.20 kg	5%	2.31 kg	12.00 w	30%	15.60 w	RBSP derivative	6	Frontier Radio w/ inverted frequencie
SSPA (X-Band)	0.81 kg	1			0.81 kg	5%	0.85 kg	9.10 w	5%	9.56 w	MESSENGER derivative	7	
Diplexer	0.30 kg	1			0.30 kg	5%	0.32 kg	n/a	-		MESSENGER	9	
Antenna	0.40 kg	1			0.40 kg	15%	0.46 kg	n/a	-		New Horizons	6	
Cables	1.50 kg	1			1.50 kg	15%	1.73 kg	n/a	-		New Horizons	8	
Harness	5.40 kg	1	0	0	5.40 kg	15%	6.21 kg	N/A	N/A	N/A		7	6% total dry mass
PROBE DRY MASS/POWER					88.87 kg	16%	103.40 kg						
Dry Mass Margin					30% (Note 1)		38.21 kg						
PROBE Maximum DRY MASS							127.08 kg						

Mass Summary

From Launch Mass Summary			Orbiter	Probe	SEP
Total (CBE) Dry Mass (Orbiter + Probe + SEP)		1487.84 kg	633.89 kg	88.87 kg	765.08 kg
Total (Maximum) Dry Mass (Orbiter + Probe + SEP)	30% (Note 1)	2127.61 kg	906.47 kg	127.08 kg	1094.06 kg
Total Consumables		2001.66 kg	1173.69 kg	0.00 kg	827.97 kg
Total (CBE) Wet Mass (Orbiter + Probe + SEP)		3489.50 kg	1807.58 kg	88.87 kg	1593.04 kg
Total (Maximum) Wet Mass (Orbiter + Probe + SEP)		4129.27 kg	2080.16 kg	127.08 kg	1922.02 kg
Launch Vehicle Capability		4200.00 kg			
Unused launch mass (kg)		70.73 kg			

SEP Stage Power Modes

			Power Mode CBE (W)					
Subsystem/Component	Power CBE (W)	# Units	Coast		Thrust (1 AU)		Thrust (5 AU)	
SEP	Totals			167.40		7247.40		366.99
Propulsion				35.00		7115.00		234.59
Propellant Management (EP)	35.00	1	ON	35.00	ON	35.00	ON	35.00
Power Processing Unit (PPU)	7080.00	1	OFF	0.00	ON	7080.00	ON	199.59
GN&C				0.00		0.00		0.00
Sun Sensors	0.00	4	ON	0.00	ON	0.00	ON	0.00
C&DH / Avionics				47.00		47.00		47.00
Command and Telemetry Computer	10.00	2	ON	20.00	ON	20.00	ON	20.00
cPCI enclosure with power supply	3.00	1	ON	3.00	ON	3.00	ON	3.00
Instrumentation	24.00	1	ON	24.00	ON	24.00	ON	24.00
Power				70.00		70.00		70.00
Solar Arrays (Ultraflex)	20.00	2	ON	40.00	ON	40.00	ON	40.00
Power Management & Distribution	30.00	1	ON	30.00	ON	30.00	ON	30.00
Thermal				15.40		15.40		15.40
Heaters	15.40	1	ON	15.40	ON	15.40	ON	15.40
Harness Loss (3%)				5.02		217.42		11.01
Total Power Loads (CBE)				172.42		7464.82		378.00
Total Power Loads (CBE plus Margin)				246.32		10664.03		540.00
Battery Recharge Power				0.00		0.00		0.00
Power Required (W) (Estimated)				246.32		10664.03		540.00
Power Available from both Arrays				15000.00		15000.00		600.00
Power used from battery (W)				0.00		0.00		0.00
Duration of Battery Power (hr)				1.00		1.00		1.00
Total Battery Load (W-Hr)				0.00		0.00		0.00
Battery Capacity (W-Hr)			TBD					
Depth of Discharge								

Orbiter Power Modes

			Power Mode CBE (W)																	
Subsystem/Component	Power CBE (W)	# Units	Launch		Cruise (inner)		Cruise (outer)		TCM (Mono)		UOI (Biprop)		Communicate		Idle		Probe Contact		Science	
Orbiter	Totals			234.20		254.20		263.20		412.06		395.58		246.70		202.70		285.15		213.70
Propulsion				4.50		4.50		4.50		175.66		159.18		4.50		4.50		36.75		4.50
AMBR Engine	46.00	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	46.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00
5 LB Engines	25.30	2	OFF	0.00	OFF	0.00	OFF	0.00	ON	50.60	ON	50.60	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00
1 LB Engines	8.26	2	OFF	0.00	OFF	0.00	OFF	0.00	ON	16.52	ON	16.52	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00
AMBR Valve Heater	10.00	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	10.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00
AMBR Injector Heater	50.00	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	25.00	OFF	0.00
Cat Bed Heaters (5 lb)	6.54	4	OFF	0.00	OFF	0.00	OFF	0.00	ON	26.16	OFF	0.00	OFF	0.00	OFF	0.00	ON	2.62	OFF	0.00
Cat Bed Heaters (1 lb)	3.86	12	OFF	0.00	OFF	0.00	OFF	0.00	ON	46.32	OFF	0.00	OFF	0.00	OFF	0.00	ON	4.63	OFF	0.00
Thruster Valve Heaters (5 lb)	3.27	4	OFF	0.00	OFF	0.00	OFF	0.00	ON	13.08	ON	13.08	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00
Thruster Valve Heaters (1 lb)	1.54	12	OFF	0.00	OFF	0.00	OFF	0.00	ON	18.48	ON	18.48	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00
Pressure Transducer	0.90	5	ON	4.50	ON	4.50	ON	4.50	ON	4.50	ON	4.50	ON	4.50	ON	4.50	ON	4.50	ON	4.50
GN&C				66.50		66.50		66.50		66.50		66.50		66.50		66.50		66.50		66.50
Star Tracker	6.50	1	ON	6.50	ON	6.50	ON	6.50	ON	6.50	ON	6.50	ON	6.50	ON	6.50	ON	6.50	ON	6.50
Inertial Measurement Units	32.00	1	ON	32.00	ON	32.00	ON	32.00	ON	32.00	ON	32.00	ON	32.00	ON	32.00	ON	32.00	ON	32.00
Reaction Wheels	7.00	4	ON	28.00	ON	28.00	ON	28.00	ON	28.00	ON	28.00	ON	28.00	ON	28.00	ON	28.00	ON	28.00
C&DH / Avionics				11.50		11.50		11.50		11.50		11.50		11.50		11.50		11.50		11.50
IEM	10.00	1	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00
Input/Output Devices	0.15	10	ON	1.50	ON	1.50	ON	1.50	ON	1.50	ON	1.50	ON	1.50	ON	1.50	ON	1.50	ON	1.50
Power				13.50		13.50		13.50		13.50		13.50		13.50		13.50		13.50		13.50
Power Distribution Unit (PDU)	10.00	1	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00	ON	10.00
Power System Electronics (SRU)	3.50	1	ON	3.50	ON	3.50	ON	3.50	ON	3.50	ON	3.50	ON	3.50	ON	3.50	ON	3.50	ON	3.50

RF/Comm				132.20		132.20		132.20		116.20		116.20		116.20		16.20		136.90		16.20
X/Ka-Band Coherent Transceivers	16.20	1	ON	16.20	ON	16.20	ON	16.20	ON	16.20	ON	16.20	ON	16.20	ON	16.20	ON	20.90	ON	16.20
X-Band TWTA	116.00	1	ON	116.00	ON	116.00	ON	116.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	116.00	OFF	0.00
Ka-Band TWTA	100.00	1	OFF	0.00	OFF	0.00	OFF	0.00	ON	100.00	ON	100.00	ON	100.00	OFF	0.00	OFF	0.00	OFF	0.00
Thermal				0.00		20.00		29.00		20.00		20.00		20.00		76.00		20.00		60.00
Propulsion and Instrument Heat	20.00	1	OFF	0.00	ON	20.00	ON	20.00	ON	20.00	ON	20.00	ON	20.00	ON	20.00	ON	20.00	ON	20.00
Internal heaters	20.00	1	OFF	0.00	OFF	0.00	OFF	9.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	56.00	OFF	0.00	OFF	40.00
Instruments	Totals			6.00		6.00		6.00		8.70		8.70		14.50		14.50		0.00		41.50
WAC	3.50	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	3.50
NAC	5.00	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	5.00
Vis/Near IR Mapping Spectrometer	7.10	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	7.10
Mid-IR Thermal Detector	7.00	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	7.00
UV Imaging Spectrograph	4.40	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	4.40
Plasma Instrument 1 (SWAP)	3.00	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	3.00	ON	3.00	OFF	0.00	ON	3.00
Plasma Instrument 2 (JEDI)	2.80	1	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	OFF	0.00	ON	2.80	ON	2.80	OFF	0.00	ON	2.80
Magnetometer	2.70	1	OFF	0.00	OFF	0.00	OFF	0.00	ON	2.70	ON	2.70	ON	2.70	ON	2.70	OFF	0.00	ON	2.70
USO	3.00	2	ON	6.00	ON	6.00	ON	6.00	ON	6.00	ON	6.00	ON	6.00	ON	6.00	OFF	0.00	ON	6.00
Harness Loss (3%)				7.03		7.63		7.90		12.36		11.87		7.40		6.08		8.55		6.41
Total Power Loads (CBE)				241.23		261.83		271.10		424.42		407.45		254.10		208.78		293.70		220.11
Total Power Loads (CBE plus Margin)				344.61		374.04		387.28		606.32		582.07		363.00		298.26		419.57		314.44
Battery Recharge Power				0.00		0.00		0.00		0.00		0.00		0.00		0.00		0.00		0.00
Power Required (W) (Estimated)				344.61		374.04		387.28		606.32		582.07		363.00		298.26		419.57		314.44
Power Available at ASRG (W)				438.00		411.00		390.00		367.50		367.50		367.50		367.50		367.50		367.50
Power used from battery (W)				0.00		0.00		0.00		238.82		214.57		0.00		0.00		52.07		0.00
Duration of Battery Power (hr)				0.00		8.00		8.00		1.00		1.00		0.00		0.00		2.00		8.00
Total Battery Load (W-Hr)				0.00		0.00		0.00		238.82		214.57		0.00		0.00		104.15		0.00
Battery Capacity (W-Hr)				480.00		480.00		480.00		480.00		375.85		480.00		480.00		480.00		480.00
Depth of Discharge				0%		0%		0%		50%		57%		0%		0%		22%		0%

Entry Probe Power Modes

			Power Mode CBE (W)					
Subsystem/Component	Power CBE (W)	# Units	Comms / C&DH		Hibernation		Probe Science	
Probe	Totals			25.10		0.72		46.55
C&DH / Avionics				1.00		0.10		3.75
IEM	3.75	1	OFF	0.00	OFF	0.00	ON	3.75
IEM (medium power mode)	1.00	1	ON	1.00	OFF	0.00	OFF	0.00
IEM (low power mode)	0.10	1	OFF	0.00	ON	0.10	OFF	0.00
RF/Comm				21.10		0.00		21.10
X-Band Coherent Transceiver	12.00	1	ON	12.00	OFF	0.00	ON	12.00
X-Band SSPA	9.10	1	ON	9.10	OFF	0.00	ON	9.10
Instruments	Totals			3.00		0.62		21.70
Mass Spectrometer	10.00	1	OFF	0.00	OFF	0.00	ON	10.00
Atmospheric Structure Inst. (ASI)	6.30	1	OFF	0.00	OFF	0.00	ON	6.30
Nephlometer	2.40	1	OFF	0.00	OFF	0.00	ON	2.40
USO (warm - up prior to TX)	0.62	1	OFF	0.00	ON	0.62	OFF	0.00
USO (operational)	3.00	1	ON	3.00	OFF	0.00	ON	3.00
Harness Loss (3%)				0.75		0.02		1.40
Total Power Loads (CBE)				25.85		0.74		47.95
Total Power Loads (CBE plus Margin)				36.93		1.06		68.50
Duration of Battery Power (hr)				2.50		693.50		2.00
Total Battery Load (W-Hr)				92.33		737.70		136.99
Battery Capacity (W-Hr)			1415.20					
Depth of Discharge			68%					

Ice Giants Decadal Survey Results Summary

May 4, 2010

(Revision — 6/3/2010)

Ice Giants Decadal Study: Appendix C

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Uranus/Neptune Decadal Study Team

Role	Name
Uranus / Science Champion	William B. Hubbard
Neptune / Science Champion	Mark Marley
Giant Planets Chair	Heidi B. Hammel
Giant Planets Vice-Chair	Amy A. Simon-Miller
Panel Members	Krishan Khurana
	Brigette Hesman
	John Clarke
NASA HQ POC	Leonard Dudzinski

Role	Name	Organization
Decadal Program Manager	Kurt Lindstrom	JHU/APL
APL Science POC	Elizabeth Turtle	JHU/APL
Project Manager	Helmut Seifert	JHU/APL
Lead Systems Engineer	Doug Eng	JHU/APL
Systems / Instruments	Rob Gold Elena Adams	JHU/APL
Systems Engineer	Darryl Royster	JHU/APL
Systems Engineer / SEP	Steve Oleson Melissa Mcguire David Grantier	NASA/GRC
Mission Design	Yanping Guo John Dankanich Chris Scott Ryan Russell	JHU/APL NASA GRC JHU/APL Georgia Tech
SEP Configuration	Jon Drexler	NASA/GRC
Cost	Larry Wolfarth Sally Whitley Jon Drexler	JHU/APL JHU/APL NASA/GRC
Risk/Reliability	Melissa Jones Anita Tenteris	JHU/APL NASA/GRC

Uranus/Neptune Decadal Study Team

Role	Name	Organization
Aerocapture/Aeroentry	Mary Kae Lockwood Dick Powell David Way	JHU/APL NASA/LaRC NASA/LaRC
RF	Brian Sequeira Joe Warner	JHU/APL NASA/GRC
GN&C	Robin Vaughn Wen-Jong Shyong Mike Martini	JHU/APL JHU/APL NASA/GRC
Operations	Rick Reinders	JHU/APL
Integration & Test	Jay White	JHU/APL
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Stress Analyst	Marc Briere	JHU/APL
Thermal Analyst	Elisabeth Abel Tony Colozza	JHU/APL NASA/GRC
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Power	Marty Fraeman James Fincannon Kristen Bury	JHU/APL NASA/GRC NASA GRC
Propulsion	Stewart Bushman James Fittje	JHU/APL NASA/GRC

Study Request

- **Two Full Mission Studies – one for Uranus and one for Neptune – but the science and constraints are to be as similar as possible. The goal of the two studies is to identify any system-specific requirements (i.e., requirements imposed by the Uranus or Neptune systems themselves) that drive cost and/or risk.**
  - Neptune dropped due to higher complexity, e.g. aerocapture, lifetime risks
- **Ice Giant Orbiter/Probe Mission: define concept for a floor mission to an Ice Giant System with two components: (1) a orbiter with limited instrument suite, and (2) a shallow atmospheric probe.**
- **The main goal of the concept study is to define a solution with the $1.5B to $1.9B cost range.**
- **Secondary study goals are to: clarify the costs associated with possible variations or additions to the primary mission, i.e., look at potential up-scopes to the orbiter instrument payload (with costs identified) and identify descopes to keep to the cost cap.**

Study Level

Concept Maturity Level	Definition	Attributes
CML 6	Final Implementation Concept	Requirements trace and schedule to subsystem level, grassroots cost, V&V approach for key areas
CML 5	Initial Implementation Concept	Detailed science traceability, defined relationships and dependencies: partnering, heritage, technology, key risks and mitigations, system make/buy
CML 4	Preferred Design Point	Point design to subsystem level mass, power, performance, cost, risk
CML 3	Trade Space	Architectures and objectives trade space evaluated for cost, risk, performance
CML 2	Initial Feasibility	Physics works, ballpark mass and cost
CML 1	Cocktail Napkin	Defined objectives and approaches, basic architecture concept

Prioritized Objectives

1) Deliver a small payload into orbit around the planet and conduct a limited orbital tour of that ice giant system for roughly two years with primary science being atmospheric and magnetospheric characterization.
2) Deliver a shallow probe into the planet's atmosphere.
3) Determine distribution of thermal emission from the planet's atmosphere.
4) If possible, refine the gravitational harmonics of the planet.
5) Conduct close flybys of any large satellites.

These results will build upon Voyager's remote-sensing data obtained with technology launched more than a generation ago, as well as more recent data from Earth-based telescopes. The mission results, along with similar measurements from the Galileo probe/orbiter and the Cassini orbiter, will constrain models of giant planet and Solar System formation and extend our understanding of extrasolar planetary systems, the bulk of which are ice giants.

Study Requirements

- **$1.5B-$1.9B cost range (NOT flagship missions)**
- **Need a basic science payload inserted into planetary orbit. (Flyby trajectories should not be considered.)**
- **SEP is allowable.**
- **Aerocapture is allowable, but must be costed appropriately (perhaps a flight qualification).**
- **Need a shallow probe (depth of 1-5 bars). The data from the probe must be retrievable.**
- **If the orbiter must be encased for aerocapture, consideration should be given to alternatives, e.g., relay data from SEP stage.**
- **Launch window: 2020-2023.**
- **Assume no JGA.**

Study Requirements

- Need to understand how the cost estimate is affected by limiting the probe instrument suite to a mass spectrometer, T-P sensors and accelerometers, versus adding a nephelometer and/or other instruments.
- Need to understand the margins provided by given mission architectures that could allow possible scale back to maintain the $1.9B cost cap.
- Scope the range of feasible missions within the mid-range cost cap (assume $1.5B - $1.9B) that can obtain fundamental new measurements for an ice giant system, including: atmospheric dynamics and chemistry; magnetic field measurements; in situ measurements of elemental and isotopic abundances; constraints on internal structure; and observations of the retinue of rings and satellites, particular the larger satellites.
- Identify any requirements imposed by the Uranus or Neptune systems that drive cost or risk.

Study Results Summary (1)

- **Uranus was selected over Neptune for 2020-2023 time frame**
  - Uranus is much more accessible without a Jupiter gravity assist within a reasonable mission duration
  - For a cruise time of ≤ 15 year, a mission to Neptune would have required aero capture which is still a developing technology
- **A robust mission to Uranus can be technically achieved within this timeframe meeting full set of objectives and instrument suite including a shallow entry probe and a satellite tour.**
  - A trajectory using a solar electric propulsion stage along with a single Earth flyby allows for launch opportunities every year.
  - Capture is achievable with a low risk, chemical propulsion solution for a 13 year cruise time
    - Cruise times of less than 13 years would require an aero capture solution

Study Results Summary (2)

- **The mission was designed to accommodate a descent probe with the full instrument suite**
  - 1 hour of measurements from 0.1 – 5 bars
  - Communications allows for 720 kbits of probe data
- **The primary mission accommodates at least 20 orbits highly inclined orbit (431 days)**
  - Average data return of 3200 images per day + plasma and magnetometer data
- **The mission concept accommodates a total of 10 targeted flybys of 5 Uranus satellites (424 days)**

APL

Mission Requirements

Tier 1 science

- 2-year orbital tour
  - high-inclination, 21-day orbit; nominal mission is 20 orbits
- good coverage in magnetic latitude and longitude
  - well distributed in latitude and longitude

Tier 2 science

- shallow probe: descent to at least 1 bar, 5 bars if possible
  - < ~1-hour descent to 5 bars
- Enhanced Orbiter Payload
- Encounters with Large Satellites Desired
- low periapse, ~1.1 Ru
  - limited by safe ring-plane-crossing range ≥ 52,000 km (outside Epsilon ring)
  - lower to inside of rings at end of mission after **ring hazard assessment**?
  - atmospheric drag at 1.1 Ru is safe, but would degrade gravity measurement
- tracking during periapse
  - delta-v cost to rotate orbit is prohibitive

Tier 3 science

- Orbiter payload further enhanced
- Enhanced probe payload

Payload

Currently all fit w/i mass and power constraints; cost is TBD

Orbiter

Floor

- Wide-angle camera
- Magnetometer
- Visible/Near-IR mapping spectrometer

Enhanced

- USO
- Mid-IR thermal detector
- UV imaging spectrograph
- Narrow-angle camera
- Plasma instrument

Probe

Floor

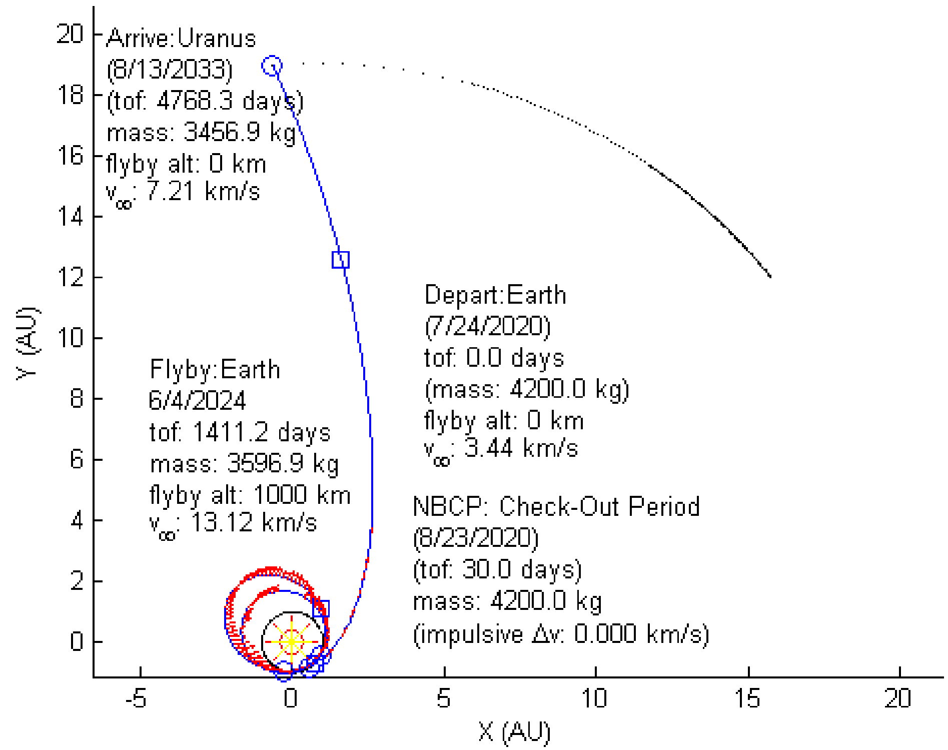
- Mass spectrometer
- Temperature-pressure sensors

Enhanced

- USO
- Nephelometer

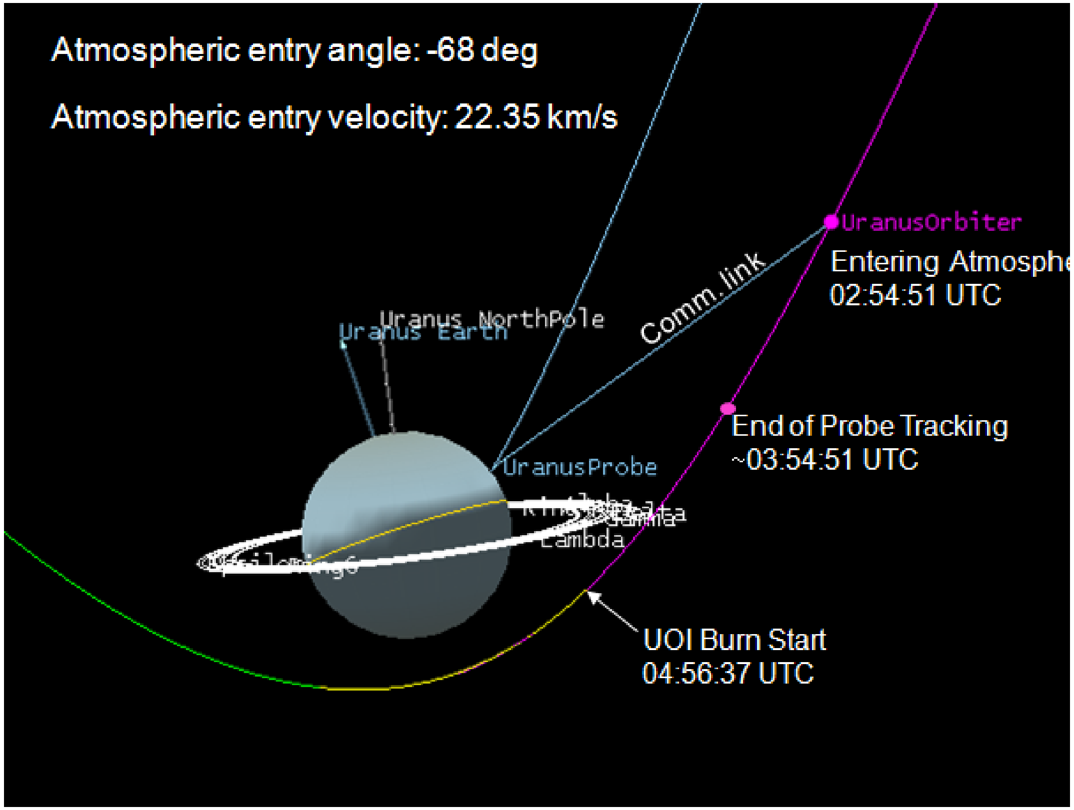
Interplanetary Trajectory

- **Solar electric propulsion trajectory with an Earth gravity assist**
  - 2 NEXT Engines
  - 20 kW Power (1 AU)
- **Assumed launch date of August 2020**
  - 20 day window
  - Launch C3 ≤ 11.83 km^2/s^2
  - Repeatable opportunities every yea
  - Atlas V 551 Launch Vehicle
- **7 Coast Periods**
  - 30 day checkout
  - 42 day prep for EGA
  - 5 other coast periods (20-30 days)
- **13 year cruise period**
  - SEP stage released after 5 years



Uranus Capture and Probe Delivery

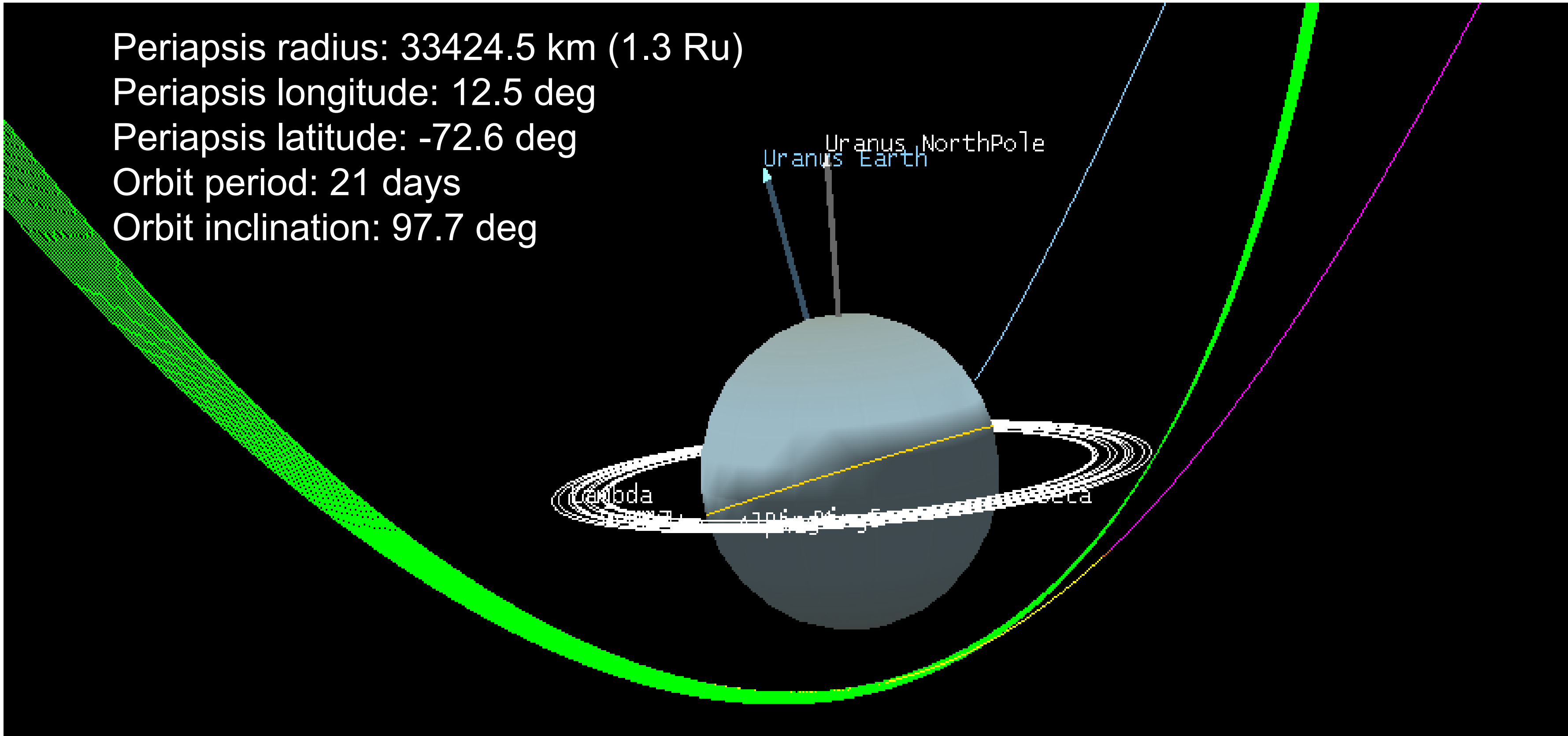
- **Probe is release 29 days prior to UOI**
  - Spin stabilized
- **One day after probe release, the orbiter will perform a deflection maneuver**
- **Probe is tracked by orbiter during entry and data is relayed to orbiter**
- **Probe collects data from 0.1-5 bars pressure for 1 hour**
- **UOI burn is conducted 1 hour after probe entry phase is completed**
  - 66.7 min burn
  - Delta-V = 1660.8 m/s
  - Contact with Earth is not possible during the burn



APL

Primary Science Orbit Phase (431 days)

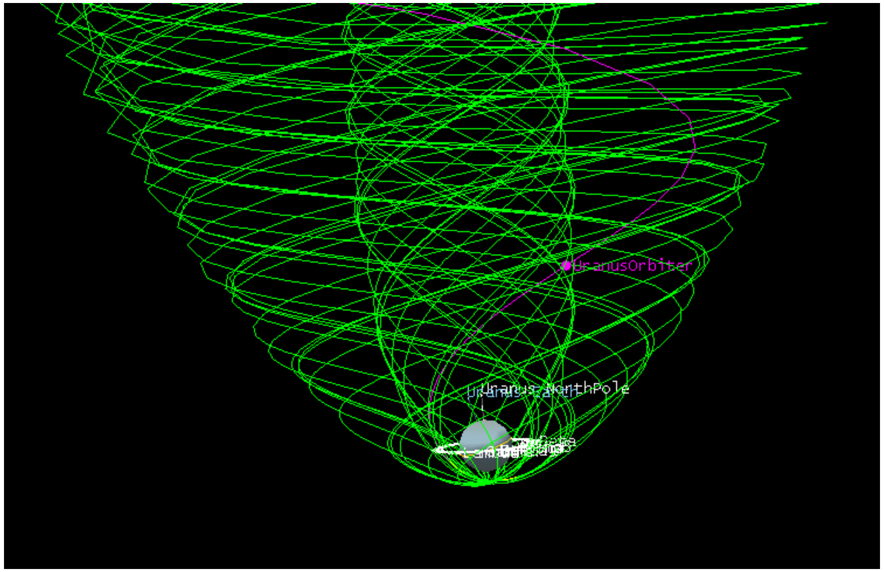
Periapsis radius: 33424.5 km (1.3 Ru)
Periapsis longitude: 12.5 deg
Periapsis latitude: -72.6 deg
Orbit period: 21 days
Orbit inclination: 97.7 deg



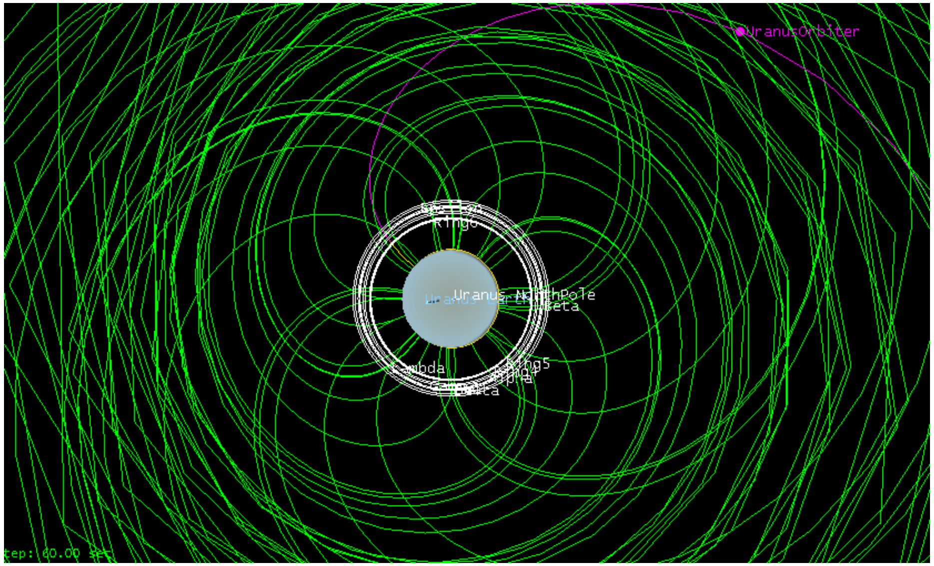
20 highly inclined elliptical orbits providing the desired science measurements (atmosphere, magnetic field, gravity field)

Uranus Observation Coverage

20 Science Orbits Plotted in Uranus Fixed Coordinate System
(purple line: incoming trajectory, green line: science orbits)



Side View

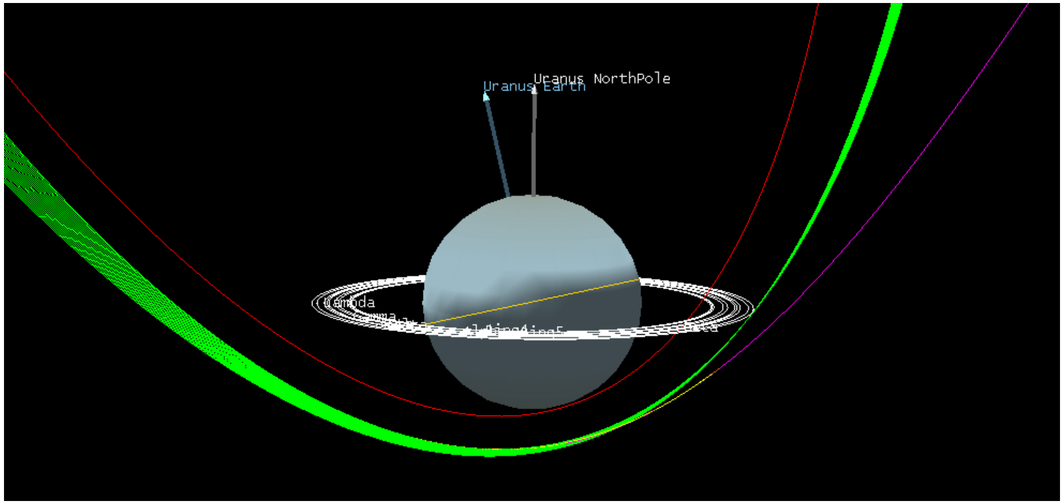


View above the equatorial Plane

Option to Lower Periapse at the End of Science Orbit Phase

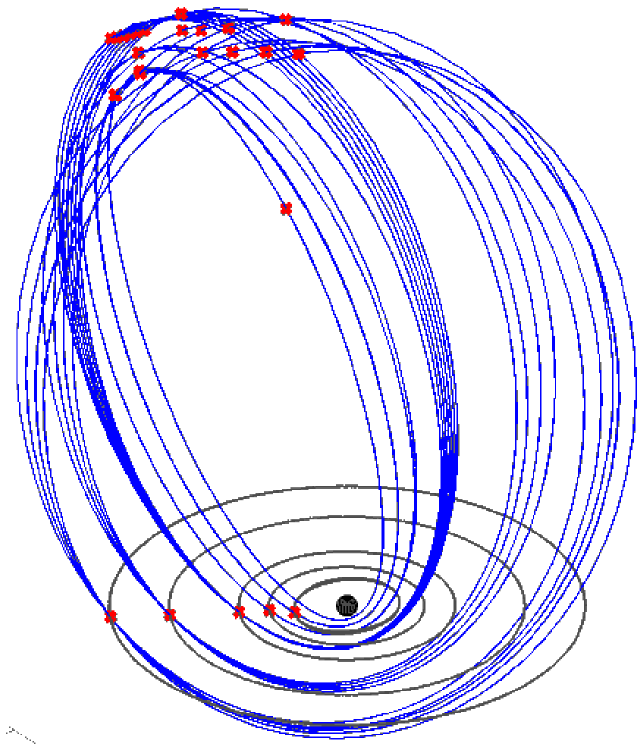
- Periapse closer to Uranus is desired for measuring gravity field J6 term
- Due to the ring crossing constraint, the achievable minimum periapsis radius is 1.3 Ru for the science orbit
- Periapse altitude can be lowered near the end of science orbit phase if desired, after assessment in the early part of the mission of the hazard posted by the environment inside the rings
- The closest periapsis distance possible is limited by the spacecraft heating rate, which is estimated in the range between 25,600 km and 26,000 km

- Delta-V required for periapse change: 56 m/s
- For periapse of 25,600 km, the min altitudes before and post Earth occultation are:
  - Ingress altitude: 6961 km
  - Egress altitude: 6170 km
  - Occultation duration: 2623.3 s



Satellite Tour

- **Tour Duration = 424 days**
- **Total Delta-V = 619 m/s**
- **10 total targeted flybys + 4 close untargeted flybys**
  - 2 Miranda (16:1)
  - 2 Ariel (10:1)
  - 2 Umbriel (6:1) + 4 close untargeted
  - 2 Titania (3:1)
  - 2 Oberon (2:1)
- **Closest approach for targeted flybys is 50 km altitude**



Mission Drivers

#	Driver	Mission Concept Impacts
1	Flexible launch year	SEP stage with Earth Flyby
2	Distance for communications	3 ASRGs to accommodate 40 W RF TWTA
3	Shallow Entry Probe	Conduct probe science up to 1 hour prior to UOI
		Communications system that requires mono-pulse tracking of the probe
4	Limited launch capacity	Maximum mission duration of 15.5 years
5	Chemical propulsion for UOI	
6	Avoiding rings	Initial periapse of 1.3 AU
		Earth occulted at periapse
		Steep entry angle for probe and higher g loading
7	Satellite Tour	Adds significantly to Delta-V budget and propellant load

Major Mission Trade Studies

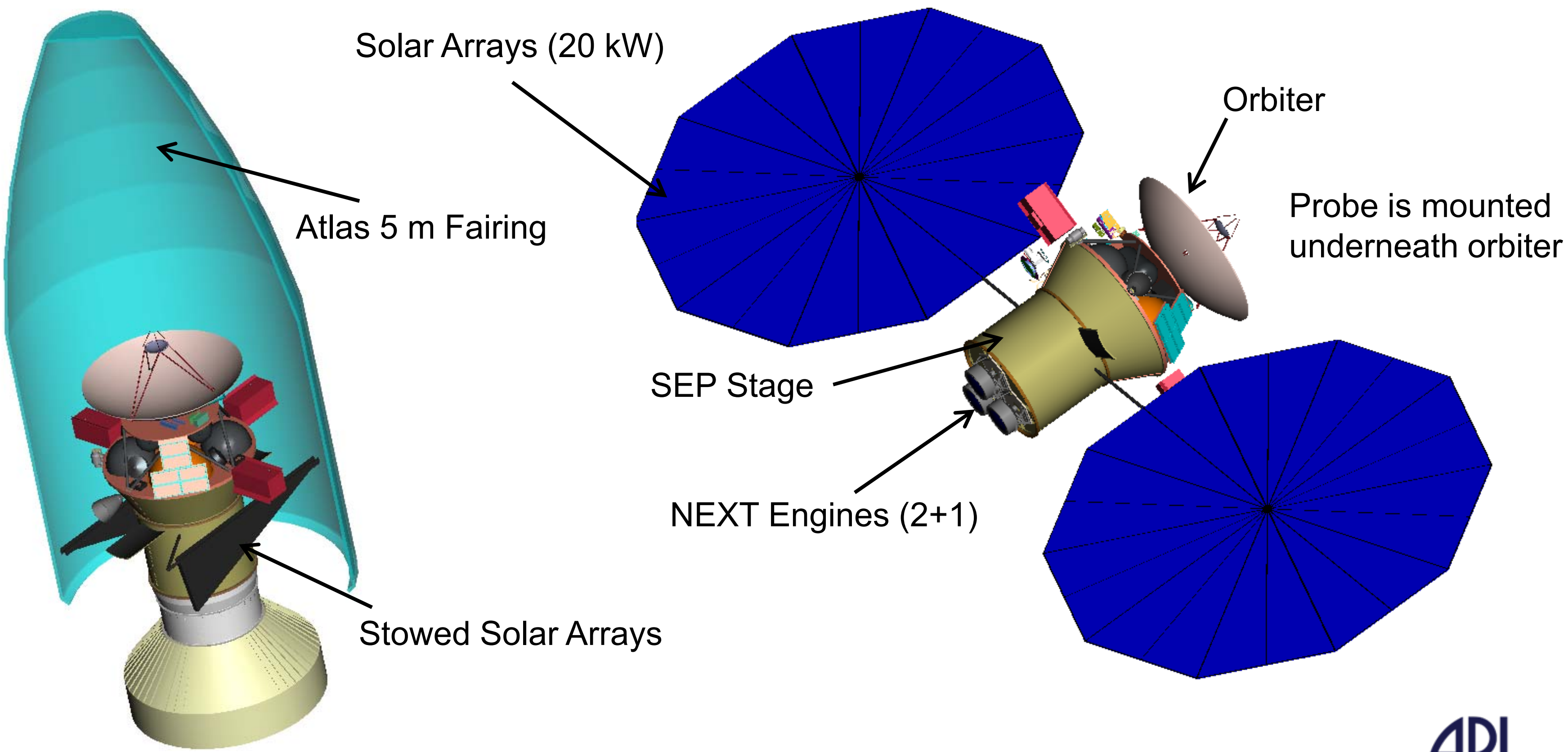
Area	Trade Options	Results
Destination	Uranus or Neptune	• Uranus cruise time is 13 years with chemical capture and no JGA • Neptune requires aerocapture or >15 year cruise • Panel selected Uranus as destination
Capture Approach	Chemical vs Aerocapture	• Aerocapture begins to offer mass advantages if capture delta-V is over 2 km/s • For Uranus mission capture delta-V is 1660 m/s • Spacecraft packaging and probe accommodations are much more more challenging with aerocapture • Aerocapture would require \$150-\$200 M of technology investment to be ready for flight including a previous flight demonstration
Uranus Periapse	Altitude, Viewing Geometry	• Due to ring crossing constraints periapse cannot be below 1.3 Ru and cannot be in view of Earth. • A periapse of 1.1 Ru can be achieved if the trajectory is allowed to come inside of the rings. This is something that could be evaluated during the mission for safety and performed if the risk was deemed acceptable
Data Return	Dish size, power, array ground stations, others	• Basic science can be met with data return capability established in previous study (7.5 kbps, single 34 m ground station, 8 hour per day contacts) • ASRG architecture cannot support higher power TWTA • Dish is sized to 2.5 m to balance pointing requirements, cost, and RF performance • Adequate science can be returned without arraying DSN antennas
Instruments and Satellite Tour	Full Enhanced Mission with Tour Remove Tour Remove Probe Floor Instruments Only	• The full enhanced mission with tour can be achieved from a technical perspective within the cost cap (\$1,894M) • Removing probe reduces cost by \$310M • Removing the tour reduces cost by \$26 • Removing Enhanced Instruments reduces cost by \$120M

Significant Flight Element Trades

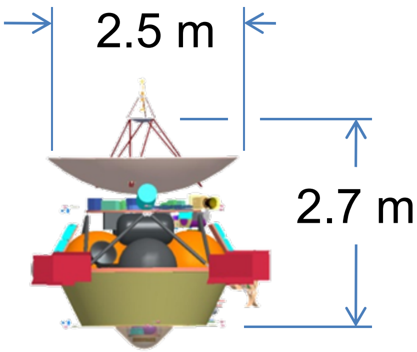
Area	Trade Options	Results
SEP Power Trades	1-3 NEXT Engines, 15-25 kW	• Full mission can be accomplished with 2 NEXT engines and 20 kW of power • More power does provided modest increase in delivered payload performance
Probe Accommodation with Spacecraft	Spin stabilize with spacecraft rotation or spin table, probe placement	• Probe was placed on the bottom and the spacecraft will spin up to release
Probe Power, Activation, and Thermal Control	Activate on timer or command RHU vs battery power	• Use 4 RHUs to minimize power during probe free flight • Checkout probe 2 times prior to entry • Activate probe based on timer
Attitude Control	Wheels vs thruster control on orbiter, SEP stage or both.	• Wheels will meet science requirements • Thrusters provided no advantage in power • Wheels should be able to last 16 years • Orbiter wheels sufficient to control SEP stage

Flight Configuration with SEP Stage

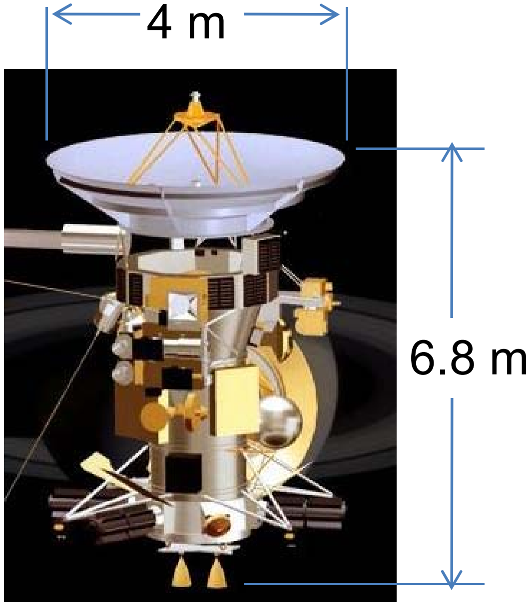
- **Launch mass with Margin = 4290 kg**
- **852 kg of Xenon Carried for SEP Phase**

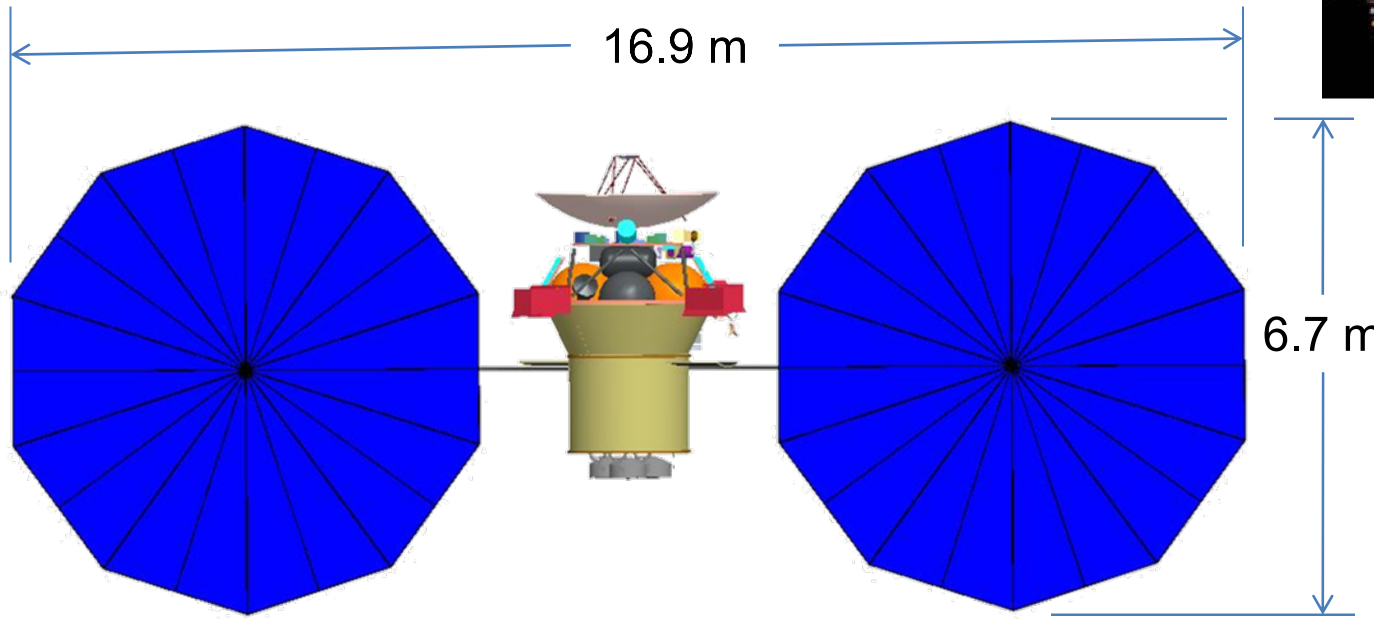


Integrated Spacecraft



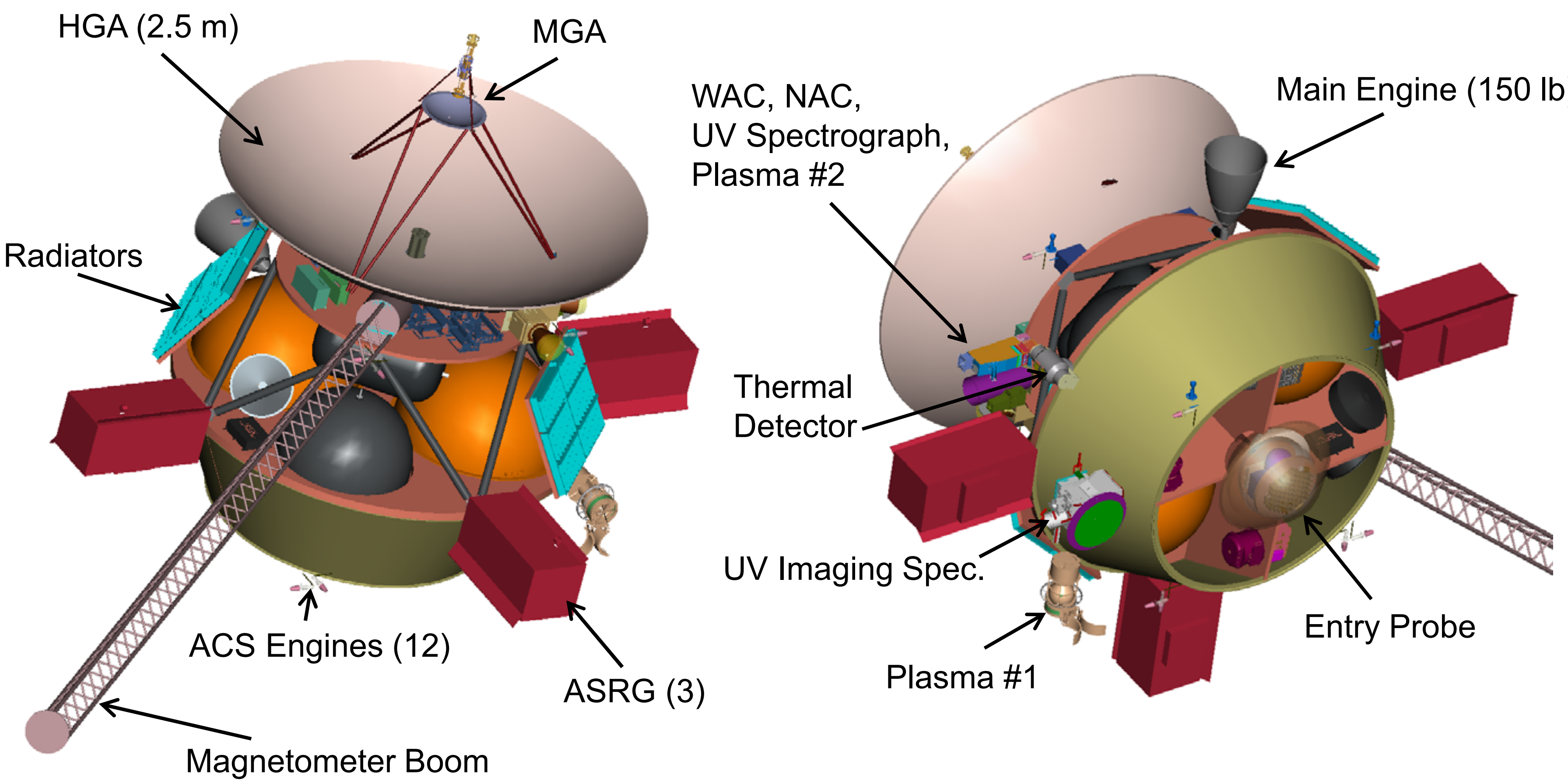
Cassini spacecraft shown for size comparison





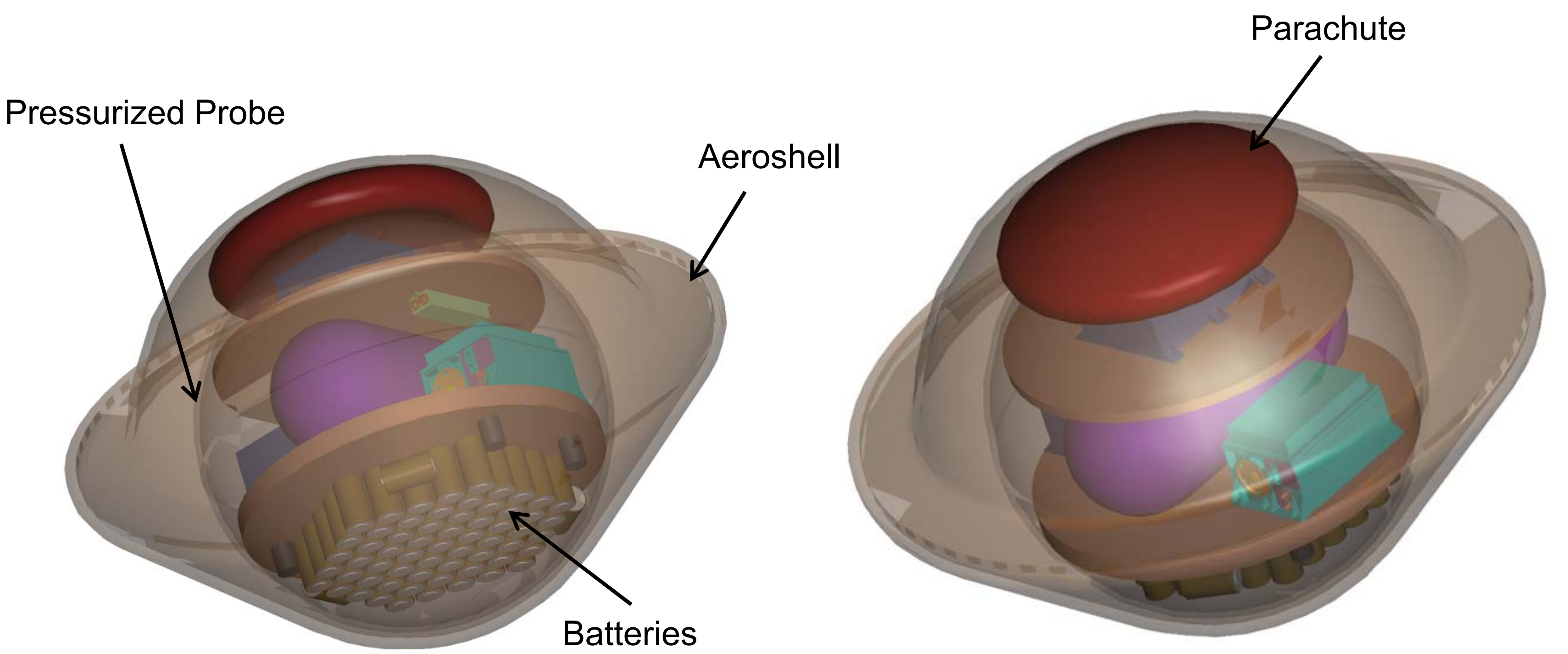
Flight Configuration- Orbiter

- **Orbiter Mass with Margin = 2217 kg (Wet), 906 kg (Dry)**

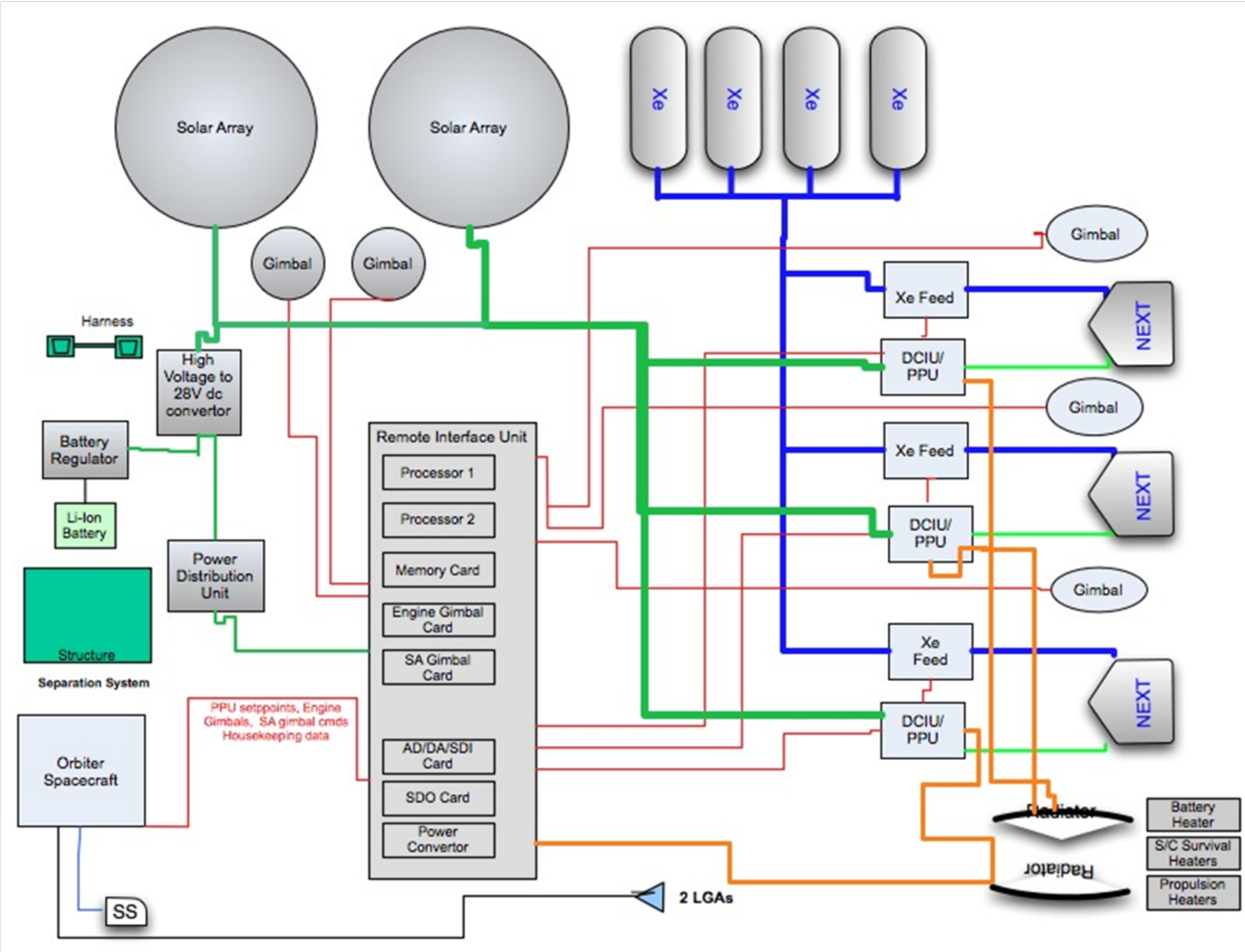


Flight Configuration- Probe

- Probe Mass with Margin = 127 kg
- Aeroshell Diameter = 76 cm
- Probe Diameter = 46 cm
- Parachute Diameter = 3.25 m (Ribbon Parachute Cd = 0.55)



SEP Stage Block Diagram



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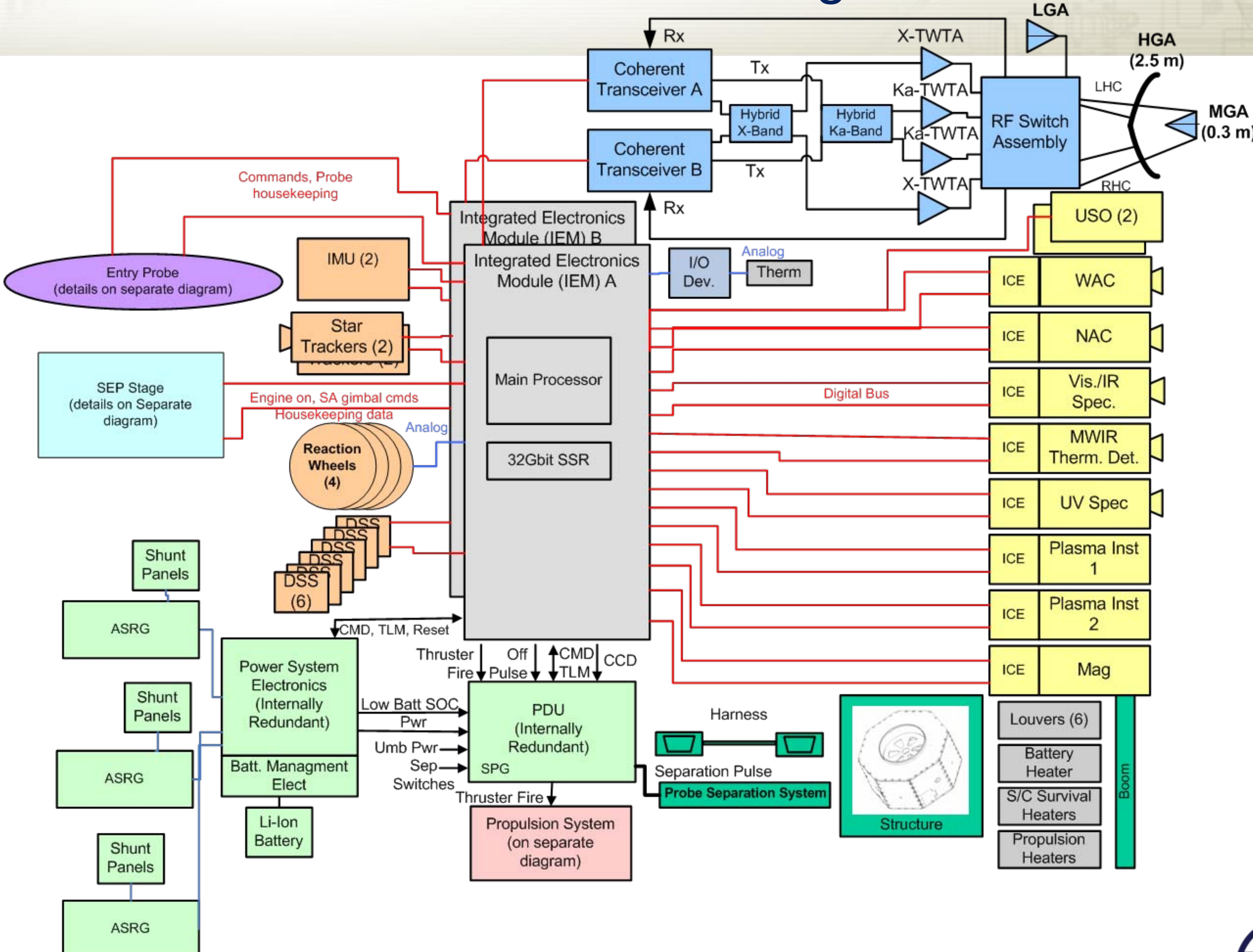
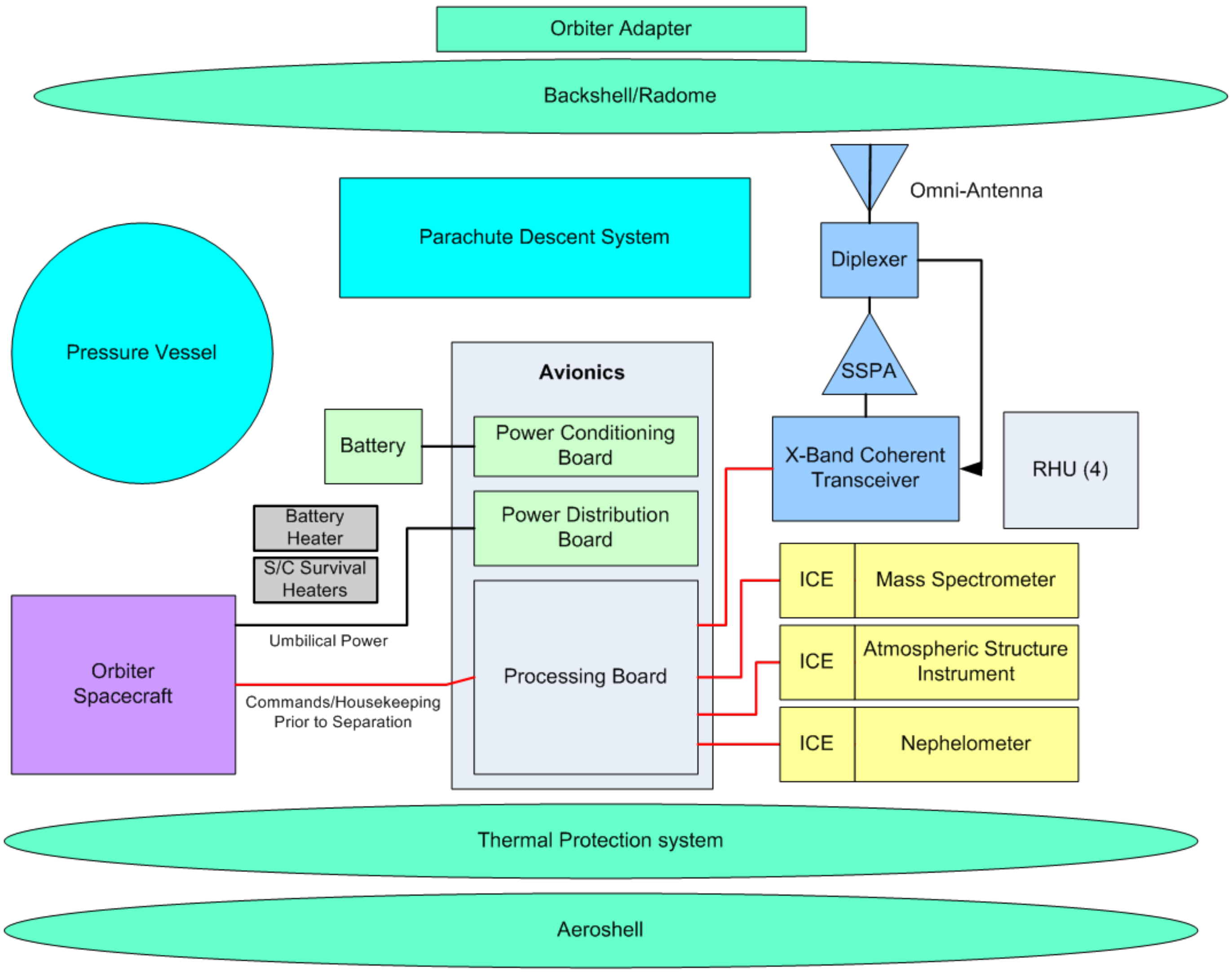
Orbiter Block Diagram

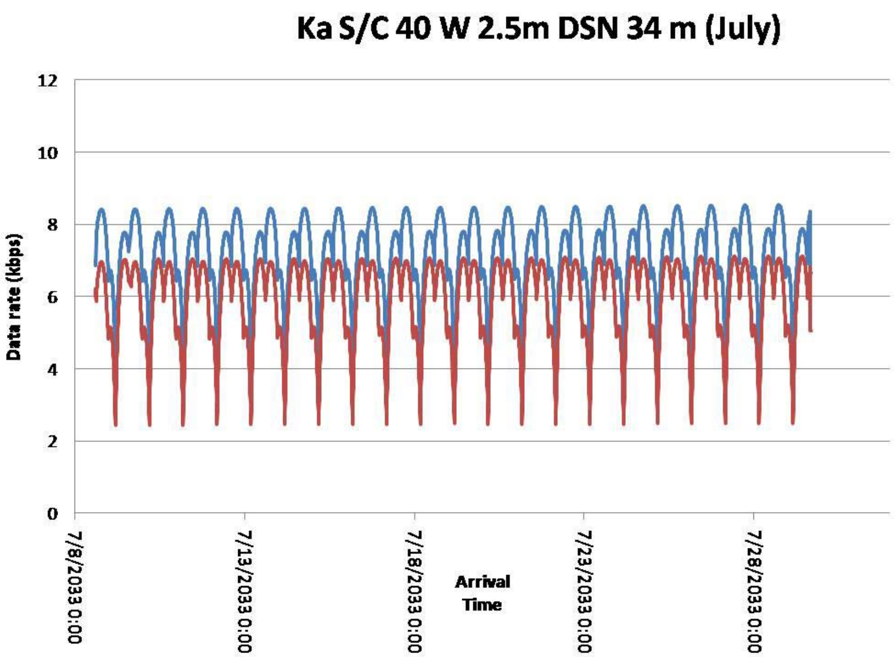


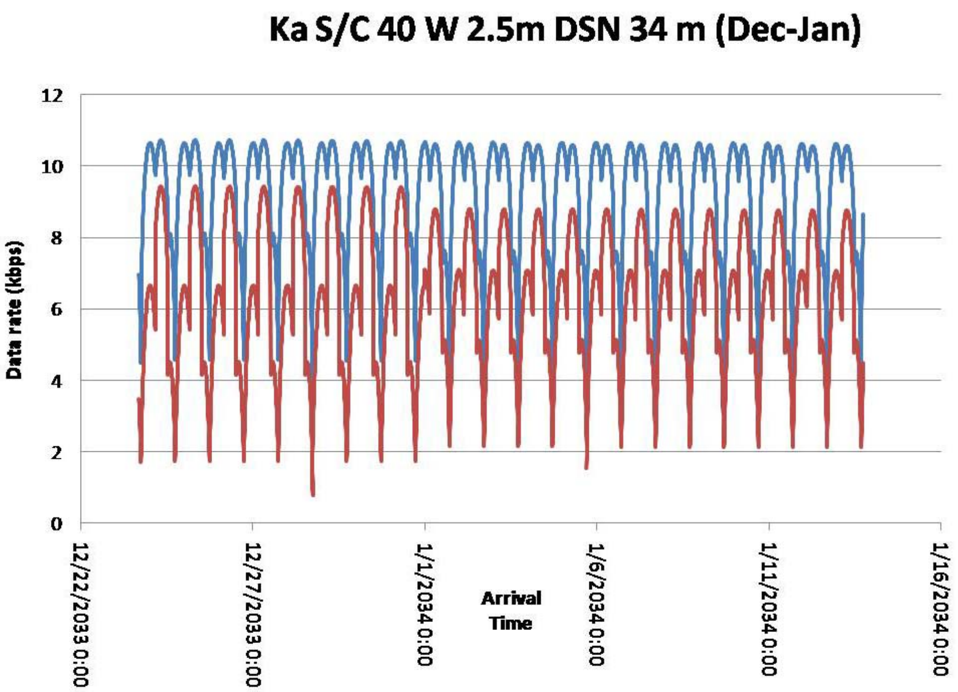
Entry Probe Block Diagram



Data Rates

- Average data rate estimated as 7.5 kbps
- Actual data rate will vary with station, elevation, and season (weather)
- Two example seasons shown below representing worst and best cases





Technology Summary

- **Only the large parasol solar arrays fall below TRL-6**
  - Smaller version has flown on Phoenix
  - Orion development has helped
  - Overall risk is low
- **Below are flight readiness estimates for TRL -6 items. All other components are TRL-7 or higher**

Component	Technology Progress	Flight Readiness
NEXT ion engine	Under development at NASA GRC	2015
NEXT gas feed system	Under development at NASA GRC	2015
Power processing unit (PPU)	Under development at NASA GRC	2015
Power management and distribution	Update from the DAWN mission	2015

Component	Technology Progress	Flight Readiness
ASRG	Under development at NASA GRC	2015
X-Ka-Band Coherent Transceiver	Under development at JHU/APL	2014

Component	Technology Progress	Flight Readiness
Mass Spectrometer	Updates to Galileo, currently unfunded	4 years after funding
Probe battery system	Requires qualification for 400g deceleration	2 years after funding
Coherent transceiver	Modification of RBSP design	3 years after funding
Probe antenna	Modified NEAR patch antenna	2 years after funding

Risk Definitions

Likelihood Bins	Safety (Likelihood of safety event occurrences)	Technical (Estimated likelihood of not meeting mission technical performance requirements)	Cost/Schedule (Estimated likelihood of not meeting allocated Cost/Schedule requirement or margin)
5 Very High	$(P_s > 10^{-1})$	$(P_T > 50\%)$	$(P_{cs} > 75\%)$
4 High	$(10^{-2} < P_s > 10^{-1})$	$(25\% < P_T < 50\%)$	$(50\% < P_{cs} \leq 75\%)$
3 Moderate	$(10^{-3} < P_s > 10^{-2})$	$(15\% < P_T < 25\%)$	$(25\% < P_{cs} \leq 50\%)$
2 Low	$(10^{-5} < P_s > 10^{-3})$	$(2\% < P_T < 15\%)$	$(10\% < P_{cs} \leq 25\%)$
1 Very Low	$(P_s > 10^{-6})$	$(0.1\% < P_T < 2\%)$	$(P_{cs} \leq 10\%)$

Attribute	Value	Description
Consequence	Very High (5)	Potential project cost overrun greater than 20% Schedule slip greater than 3 months Loss of spacecraft, instrument, or payload Death or permanent disabling injury
	Major (4)	Potential project cost overrun greater than 10% Schedule slip of 1 to 3 months Loss of 1 or more level 1 science requirements Major loss of capability of spacecraft, instrument, or payload Severe injury
	Medium (3)	Potential project cost overrun from 3% to 10% Schedule slip affecting critical path but not launch or post launch critical event Major loss of capability of spacecraft or payload Injury with lost work time
	Minor (2)	Potential cost overrun less than 3% Non-critical-path schedule slip Decrease in spacecraft or payload capability/margin but all mission requirements met, or need for requirement definition or design/implementation work-around Minor injury with no lost work time
	Minimal (1)	None of the above

Top Mission Risk Table

#	Risk	Type	L	C	Mitigation
1	If the complexity of conducting the probe science prior to UOI is too high, the probe science would be descoped	Technical	2	4	• Comprehensive analysis and test program • More rehearsals prior to event
2	If there are problems in the development of a new SEP stage then the schedule could be impacted	Schedule	2	3	• Develop a SEP stage demonstrator as a technology risk reduction
3	If mass spectrometer has development problems, then schedule will be impacted	Schedule	2	3	• Mass spectrometers are complex instruments and development will need to begin early
4	If significant issues with an Earth flyby develop, then mission schedule could be impacted.	Schedule	1	4	• Keep flyby distance ≥1000 km altitude • Begin approval process as early as possible • Jupiter or Saturn gravity assisted trajectory mitigates Earth flyby
5	If UOI becomes technically complex due to no contact, then mission schedule will be impacted	Schedule	1	4	• Assess UOI approach early • Assess required autonomy and associated cost and schedule to develop and test
6	If there is difficulty developing a large (20 kW) array, then schedule could be impacted	Schedule	1	3	• Develop and demonstrate a large parasol array as part of a technology risk reduction effort
7	If the orbiter cannot track the probe, then mission performance will be impacted	Technical	1	3	• Develop and test tracking system early • Plan for significant modeling and analysis of probe entry and tracking problem
8	If an ASRG fails, the mission will be degraded	Technical	1	3	• Qualify ASRG for longer life • Limit storage time prior to launch with fueled ASRG to 1 year
9	If system reliability is not shown to be adequate for long mission, then cost will be impacted	Cost	1	3	• Perform reliability analysis early to assess weaknesses in system • Assess system architecture options to improve reliability
10	If the peak acceleration loads have a large impact on the probe design, mission cost could be impacted.	Cost	1	3	• Qualify electronics and batteries for environment

Uranus Decadal Survey ACE Run Science Objectives and Mission Drivers

Zibi Turtle

30 April 2010

Ice Giants Decadal Study: Appendix D

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Study Request

- **Two Full Mission Studies – one for Uranus and one for Neptune – but the science and constraints are to be as similar as possible. The goal of the two studies is to identify any system-specific requirements (i.e., requirements imposed by the Uranus or Neptune systems themselves) that drive cost and/or risk.**
  - Neptune dropped due to higher complexity, e.g. aerocapture, lifetime risks
- **Ice Giant Orbiter/Probe Mission: define concept for a floor mission to an Ice Giant System with two components: (1) a orbiter with limited instrument suite, and (2) a shallow atmospheric probe.**
- **The main goal of the concept study is to define a solution with the $1.5B to $1.9B cost range.**
- **Secondary study goals are to: clarify the costs associated with possible variations or additions to the primary mission, i.e., look at potential up-scopes to the orbiter instrument payload (with costs identified) and identify descopes to keep to the cost cap.**

Draft

Prioritized Objectives

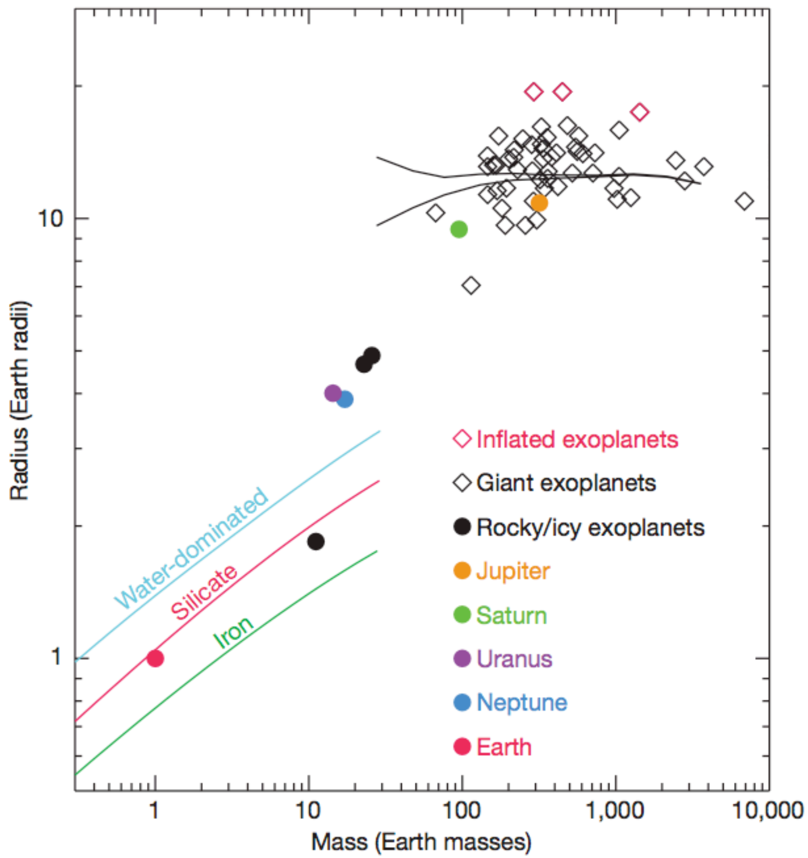
1) **Deliver a small payload into orbit around the planet and conduct a limited orbital tour of that ice giant system for roughly two years with primary science being atmospheric and magnetospheric characterization.**
2) **Deliver a shallow probe into the planet's atmosphere.**
3) **Determine distribution of thermal emission from the planet's atmosphere.**
4) **If possible, refine the gravitational harmonics of the planet.**
5) **Conduct close flybys of any large satellites.**

These results will build upon Voyager's remote-sensing data obtained with technology launched more than a generation ago, as well as more recent data from Earth-based telescopes. The mission results, along with similar measurements from the Galileo probe/orbiter and the Cassini orbiter, will constrain models of giant planet and Solar System formation and extend our understanding of extrasolar planetary systems, the bulk of which are ice giants.

Why Uranus or Neptune?

- **Examples of a class of giant planets visited only once: Voyager 2 flybys in 1986 (Uranus) and 1989 (Neptune)**
- **They have counterparts in the Galaxy (figure from Deming and Seager 2009):**

Figure 3 | Mass–radius diagram for transiting planets. Black diamonds are giant exoplanets. Three inflated giant exoplanets are indicated as red diamonds; from left to right they are: TrES-4, WASP-12 and OGLE-TR-L9. The lower black line is a theoretical mass–radius relation[74] for 1-Gyr-old giant planets orbiting at 0.045 AU from a solar-type star, and having a 10-Earth-mass core of heavy elements, plus a hydrogen–helium envelope. The upper black line is the same, but with zero core mass. The rocky/icy exoplanets (black filled circles) are the two exo-Neptunes (GJ 436b and HAT-12), and the super-Earth CoRoT-7b[22]. The coloured lines are theoretical mass–radius relations[66] for super-Earths lacking a hydrogen–helium envelope, but having a solid composition that is either water-dominated, silicates, or iron. Note the positions of Solar System planets, including Earth.



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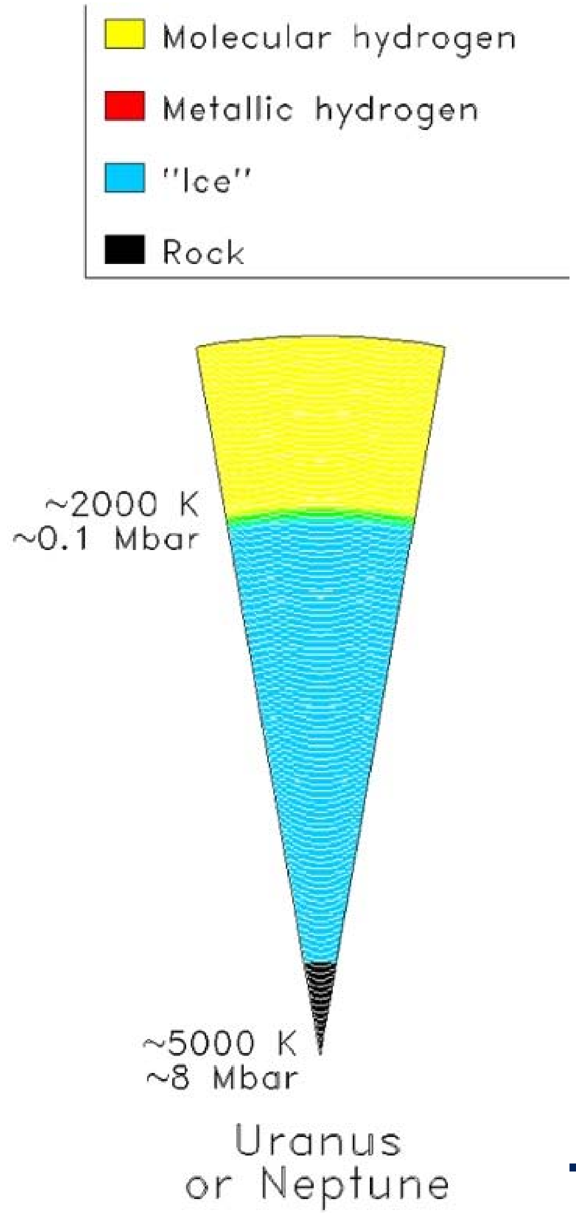
Our current picture of Uranus' interior



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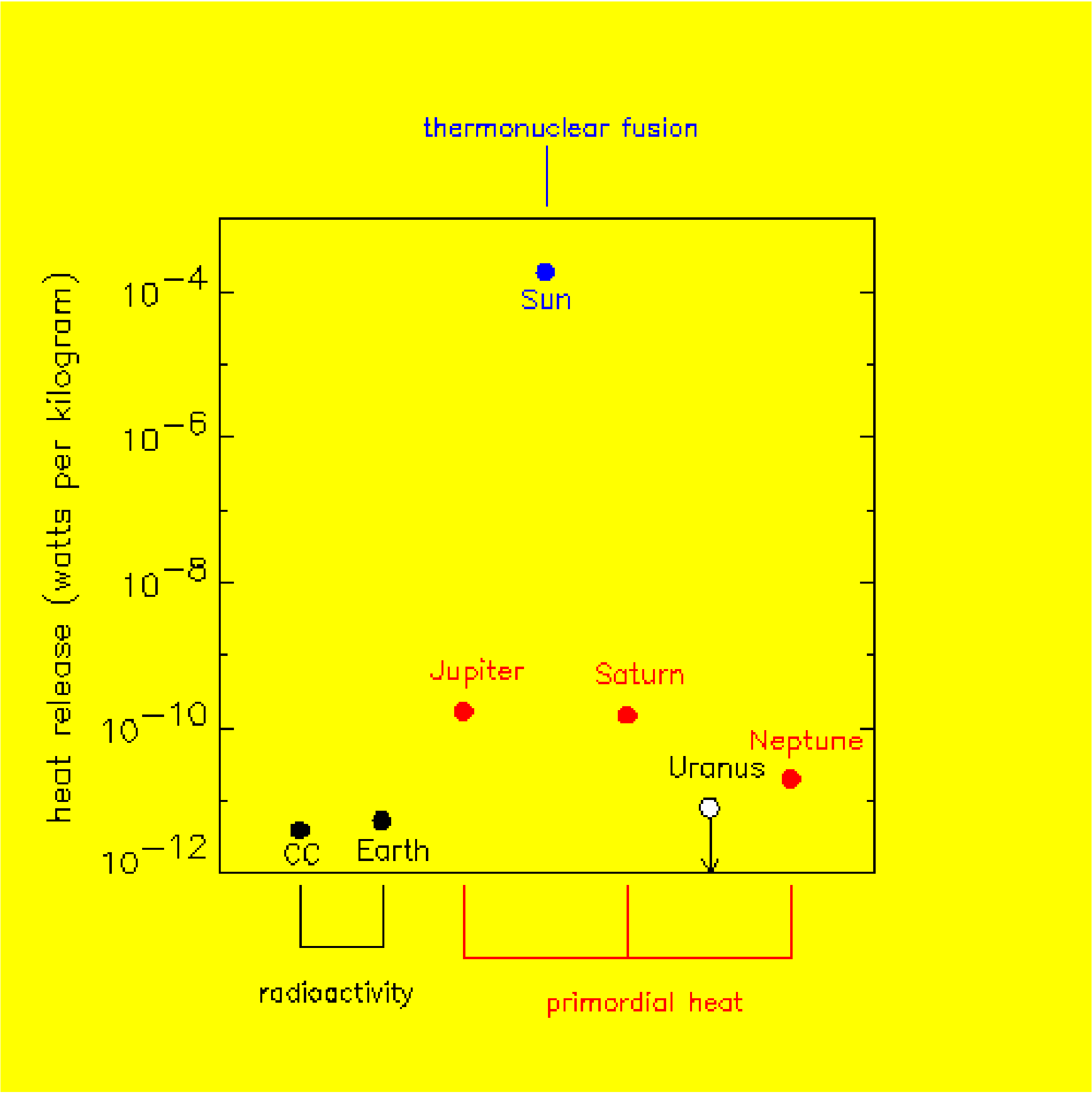
Uranus' anomalous heat flow



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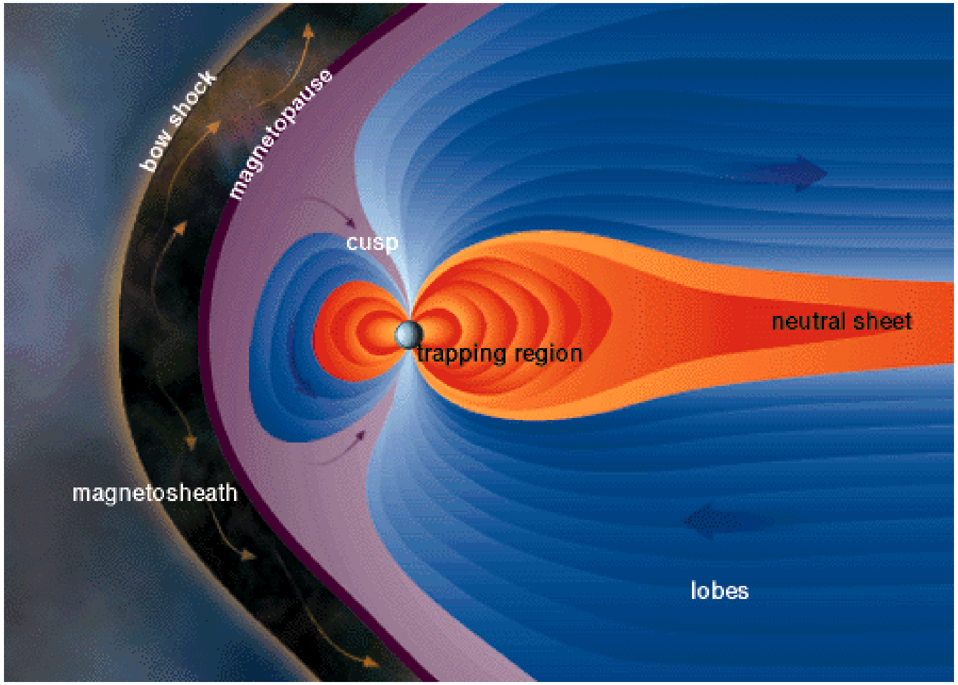
Uranian Magnetosphere

- **Dipole field strength: 0.228 gauss-Ru^3**
- **Dipole tilt to rotational axis: 58.6 degrees**
- **Dipole offset (planet center to dipole center) distance: 0.3 Ru along the rotation axis**



Origin of magnetic field

- **Probably generated by dynamo action in the "ice" layer**
- **Dynamo action implies convective motions resulting from heat flow**
- **Has the dynamo remained stable since 1986?**
- **Detection of high-order structure in magnetic field requires low periapse (similar to gravity field measurement requirements)**

Other indirect evidence of deep heat flow

Draft

- Atmospheric dynamics and zonal flow patterns
- Changes since 1986
- Night-side activity (lightning)?

Draft

Study Requirements

- **$1.5B-$1.9B cost range (NOT flagship missions)**
- **Need a basic science payload inserted into planetary orbit. (Flyby trajectories should not be considered.)**
- **SEP is allowable.**
- **Aerocapture is allowable, but must be costed appropriately (perhaps a flight qualification).**
- **Need a shallow probe (depth of 1-5 bars). The data from the probe must be retrievable.**
- **If the orbiter must be encased for aerocapture, consideration should be given to alternatives, e.g., relay data from SEP stage.**
- **Launch window: 2020-2023.**
- **Assume no JGA.**

Study Requirements

- Need to understand how the cost estimate is affected by limiting the probe instrument suite to a mass spectrometer, T-P sensors and accelerometers, versus adding a nephelometer and/or other instruments.
- Need to understand the margins provided by given mission architectures that could allow possible scale back to maintain the $1.9B cost cap.
- Scope the range of feasible missions within the mid-range cost cap (assume $1.5B - $1.9B) that can obtain fundamental new measurements for an ice giant system, including: atmospheric dynamics and chemistry; magnetic field measurements; in situ measurements of elemental and isotopic abundances; constraints on internal structure; and observations of the retinue of rings and satellites, particular the larger satellites.
- Identify any requirements imposed by the Uranus or Neptune systems that drive cost or risk.

Science Objectives – Tier 1

Tier 1 Science Objectives – Orbiter

- Determine the atmospheric zonal winds, composition, and structure at high spatial resolution, as well as the temporal evolution of atmospheric dynamics.

- Understand the basic structure of the planet’s magnetosphere as well as the high-order structure and temporal evolution of the planet's interior dynamo.

Science Objectives – Tier 2

Tier 2 Science Objectives – Enhanced Orbiter + Probe

- **Determine the noble gas abundances (He, Ne, Ar, Kr, and Xe) and isotopic ratios of H, C, N, and O in the planet's atmosphere and the atmospheric structure at the probe descent location.**
- **Determine internal mass distribution.**
- **Determine horizontal distribution of atmospheric thermal emission, as well as the upper atmospheric thermal structure and changes with time and location at low resolution.**
- **Remote sensing observations of large satellites.**

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Science Objectives – Tier 3

Tier 3 Science Objectives – Enhanced Orbiter + Enhanced Probe

- **Measure the magnetic field, plasma, and currents to determine how the tilted/offset/rotating magnetosphere interacts with the solar wind over time.**
- **Remote sensing observations of small satellites and rings.**
- **Determine the vertical profile of zonal winds as a function of depth in the atmosphere, in addition to the location, density, and composition of clouds as a function of depth in the atmosphere.**

APL

Draft

Traceability Matrix – Tier 1

Tier 1 Science Objective	Measurement	Instrument	Functional Requirement
Determine the atmospheric zonal winds, composition, and structure at high spatial resolution, as well as the temporal evolution of atmospheric dynamics	Spatially-resolved scattering properties of the atmosphere	Wide-angle visible imager; visible/near-infrared mapping spectrometer	Orbiter with 2-year tour
Understand the basic structure of the planet's magnetosphere as well as the high-order structure and temporal evolution of the planet's interior dynamo	Make continuous measurements of vector magnetic field (1 sec resolution)	Magnetometer	Orbiter

Draft

Traceability Matrix – Tier 2

Tier 2 Science Objective	Measurement	Instrument	Functional Requirement
Determine the noble gas abundances (He, Ne, Ar, Kr, and Xe) and isotopic ratios of H, C, N, and O in the planet's atmosphere and the atmospheric structure at the probe descent location.	Measure the noble gases and isotopic ratios in the atmosphere assuming that the atmosphere is well-mixed, and measure temperature and pressure as a function of depth	Mass Spectrometer with sufficient resolution; Pressure-temperature sensors sampling at frequent intervals	Single probe that reaches at least 1 bar with a stretch goal of below 5 bars
Determine internal mass distribution	Measure higher-order gravitational harmonics from orbiter gravitational moments	USO	Orbital periapsis visible from Earth
Determine horizontal distribution of atmospheric thermal emission, as well as the upper atmospheric thermal structure and changes with time and location at low resolution.	Measure temperature as a function of latitude and longitude, from direct therml emission and/or UV occultations	Mid-infrared thermal detector; UV imaging spectrograph	Orbiter
Remote sensing of large satellites	Spatially-resolved surface reflectance spectroscopy	Wide-angle visible imager; visible/near-infrared mapping spectrometer; narrow-angle camera	Orbiter with 2-yr tour

Draft

Traceability Matrix – Tier 3

Tier 3 Science Objective	Measurement	Instrument	Functional Requirement
Measure the magnetic field, plasma, and currents to determine how the tilted/offset/rotating magnetosphere interacts with the solar wind over time.	Measurements of plasma and energetic charged particles.	Plasma and Particle instrument	Orbiter
Remote sensing of small satellites and rings	Spatially-resolved surface reflectance spectroscopy	Wide-angle visible imager; visible/near-infrared mapping spectrometer; narrow-angle camera	Orbiter with 2-yr tour
Determine the vertical profile of zonal winds as a function of depth in the atmosphere, in addition to the presence of clouds as a function of depth in the atmosphere	Measure the zonal winds as a function of depth, and determine scattering properties of the atmosphere	Ultrastable oscillator on probe and on carrier; Nephelometer	Single probe that reaches at least 1 bar with a stretch goal of below 5 bars

Mission Requirements

Tier 1 science

- 2-year orbital tour
- good coverage in magnetic latitude and longitude

Tier 2 science

- shallow probe: descent to at least 1 bar, 5 bars if possible
- low periapse
- tracking during periapse

Tier 3 science

- satellite encounters

Mission Requirements

Tier 1 science

- 2-year orbital tour
  - high-inclination, 21-day orbit; nominal mission is 20 orbits
- good coverage in magnetic latitude and longitude
  - well distributed in latitude and longitude

Tier 2 science

- shallow probe: descent to at least 1 bar, 5 bars if possible
  - < ~1-hour descent to 5 bars
- low periapse, ~1.1 Ru
  - limited by safe ring-plane-crossing range ≥ 52,000 km (outside Epsilon ring)
  - lower to inside of rings at end of mission after **ring hazard assessment**?
  - atmospheric drag at 1.1 Ru is safe, but would degrade gravity measurement
- tracking during periapse
  - delta-v cost to rotate orbit is prohibitive

Tier 3 science

- satellite encounters
  - ~350-day tour with ~10 satellite encounters (~2 each)

Draft

Payload

Currently all fit w/i mass and power constraints; cost is TBD

Orbiter

Floor

- Wide-angle camera
- Magnetometer
- Visible/Near-IR mapping spectrometer

Enhanced

- USO
- Mid-IR thermal detector
- UV imaging spectrograph
- Narrow-angle camera
- Plasma instrument

Probe

Floor

- Mass spectrometer
- Temperature-pressure sensors

Enhanced

- USO
- Nephelometer

Ice Giants Decadal Survey Mission Design

Yanping Guo (APL)
Chris Scott (APL)
John Dankanich (GRC)
Ryan Russell (Georgia Tech)

May 10, 2010

Ice Giants Decadal Study: Appendix E

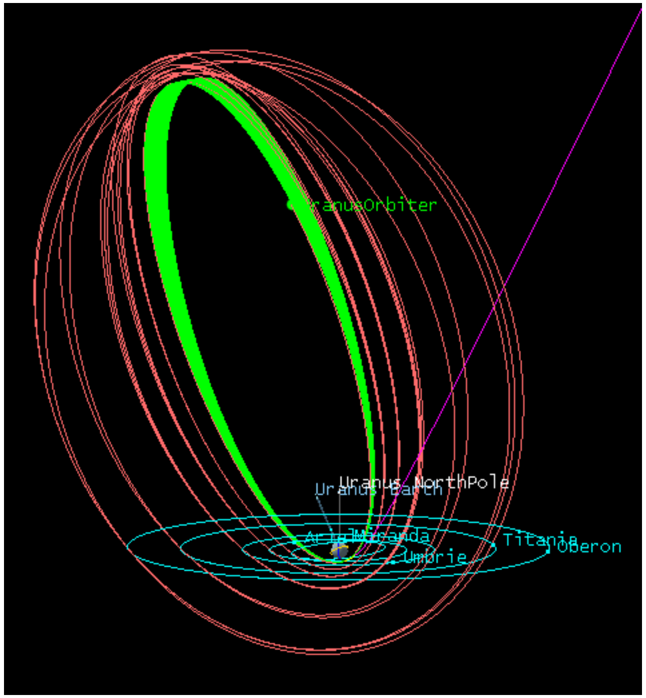
The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Agenda

- Mission Design Overview
- Interplanetary Trajectory
- Uranus Arrival
- Probe Release
- Uranus Orbit Insertion
- Uranus Satellite Tour
- Orbiter Delta-V Estimate
- Neptune Study

Mission Design Overview

- Baseline mission:
  - Launch: 2020 with 21 day launch period on Atlas V (531)
  - Cruise to Uranus: 13 years, SEP with one Earth flyby
  - Uranus arrival: 2033
  - Uranus atmosphere probe: released in a hyperbolic trajectory upon Uranus arrival, 29 days before Uranus orbit insertion
  - Uranus Orbiter: inserted into Uranus orbit after probe descent completion
  - Primary science orbit: 20 high inclined elliptic orbits around Uranus (431 days)
  - Secondary Uranus satellite tour: 10 targeted flybys of 5 major Uranus satellites (424 days)
  - Total Uranus orbit phase: 2.4 years (28.7 months)
  - Total baseline mission duration: 15.4 years
- Back launch: repeatable launch opportunities every year (2021-2023)

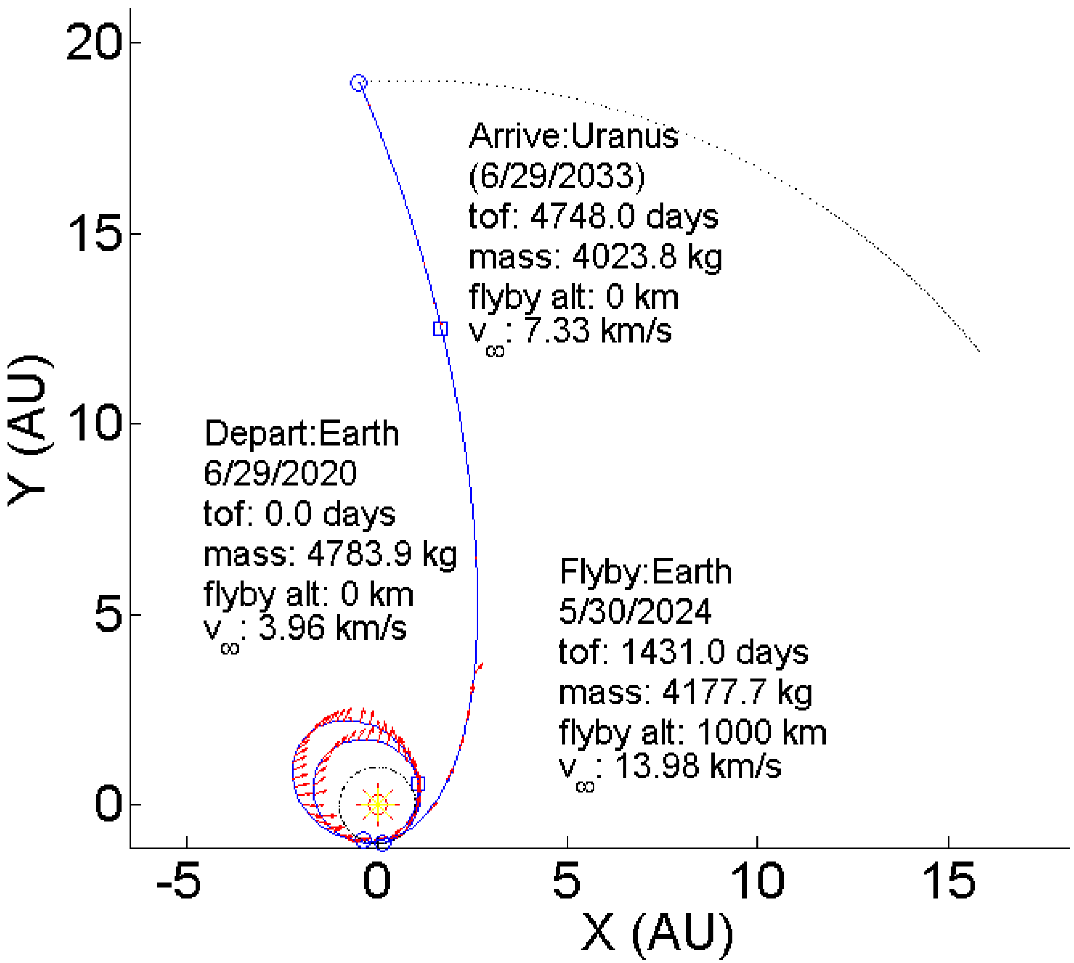


INTERPLANETARY TRAJECTORY

Interplanetary Trajectory Design Requirements

- **Launch window: 2020-2023**
- **Exclude Jupiter gravity assist**
- **Maximum cruise duration: 13 years**
- **Earth flyby altitude: greater than 1000 km**

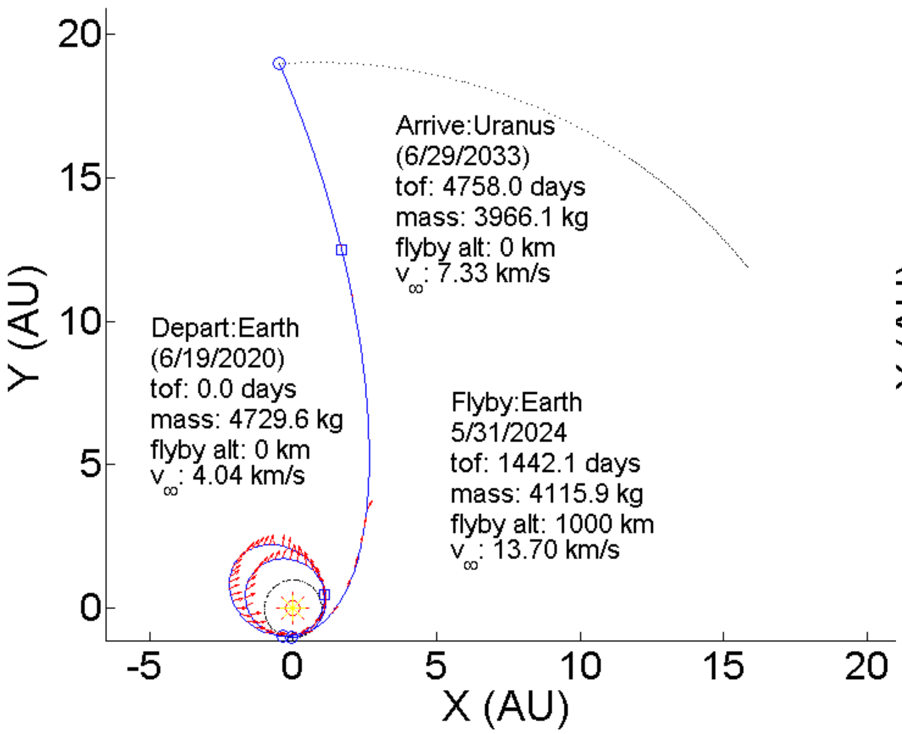
Preliminary Interplanetary Trajectory

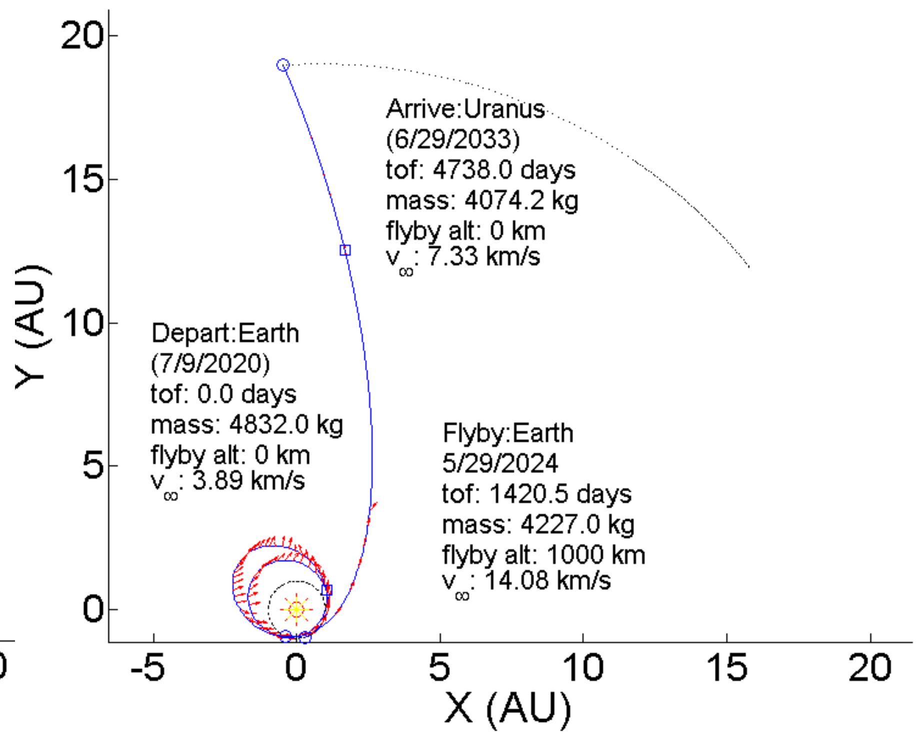


- Cruise: Solar Electric Propulsion (SEP) with an Earth gravity assist flyby
- Engine: NEXT (6000 Isp)
- Number of engines: 2
- Duty cycle: 0.92
- Array power at 1AU: 20 kW

Preliminary Interplanetary Trajectory Launch Window and C3 Requirements

- Launch dates: 6/19/2020 – 7/9/2020
- Launch period: 21 days
- Max C3 over the launch period: 16.322 km^2/s^2



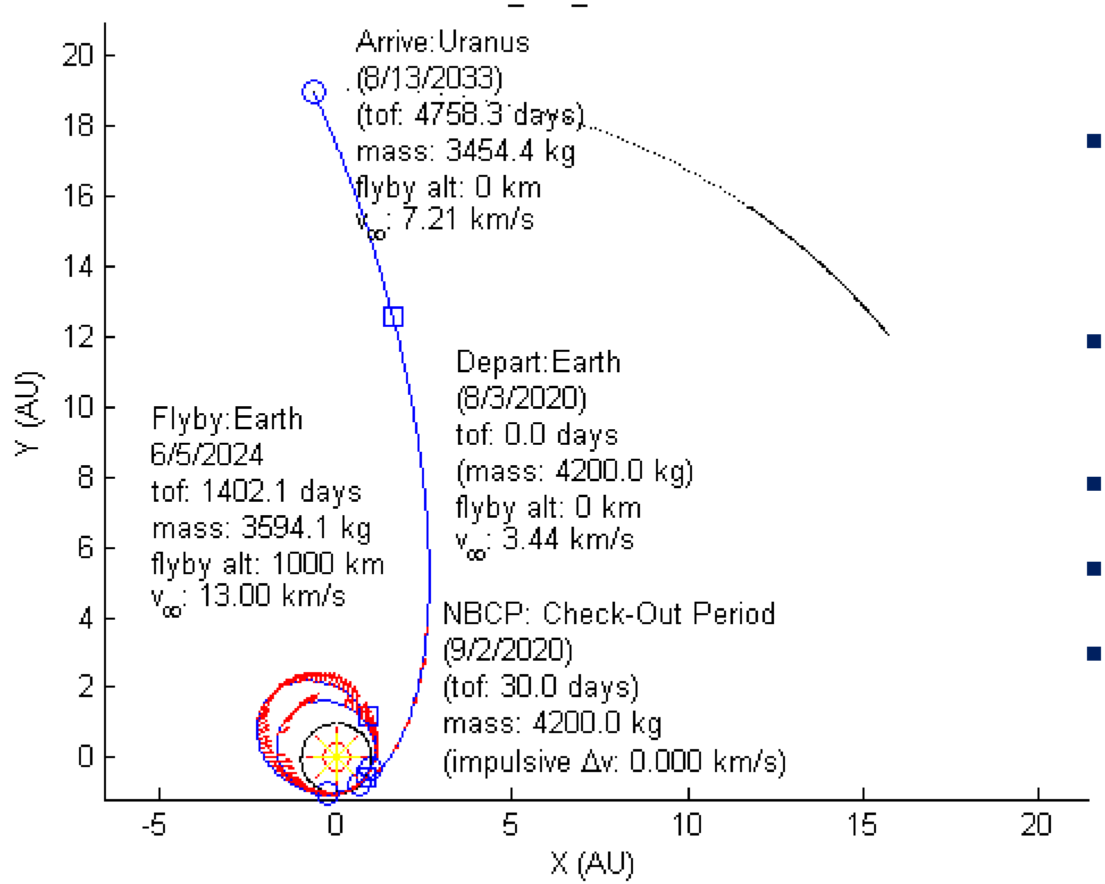


Preliminary Interplanetary Trajectory Launch Mass

- **Max Launch C3: 16.322 km^2/s^2**
- **Launch Vehicle: Atlas V (551)**
- **Max Launch Mass: 4730 kg**

Launch Date	Initial Mass (kg)	Final Mass (kg)	SEP Propellant Usage (kg)
6/19/2020	4729.6	3966.1	763.5
6/29/2020	4783.9	4023.8	760.1
7/9/2020	4832	4074.2	757.8

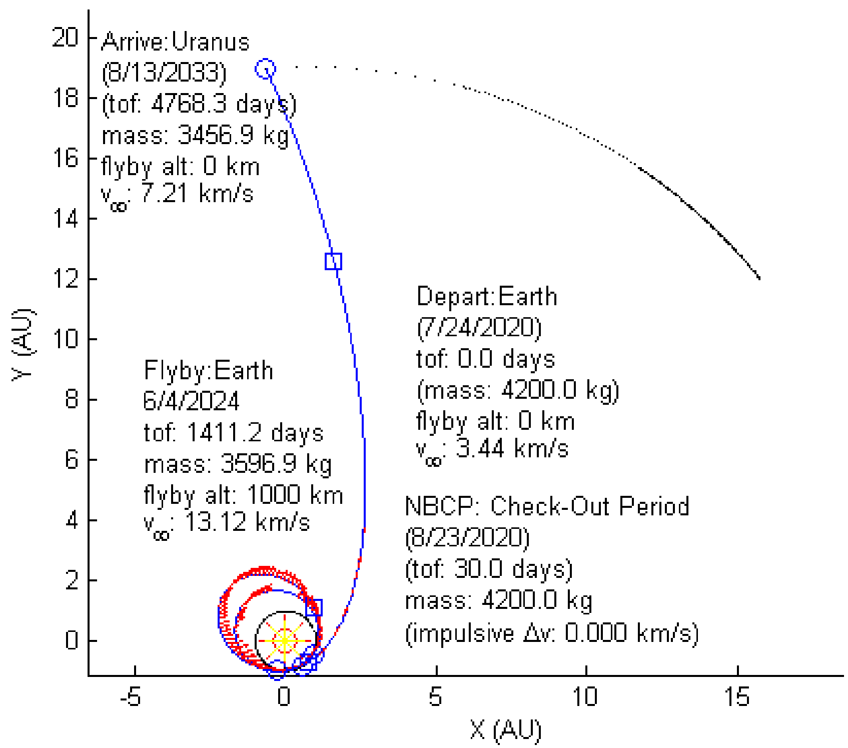
Baseline Interplanetary Trajectory

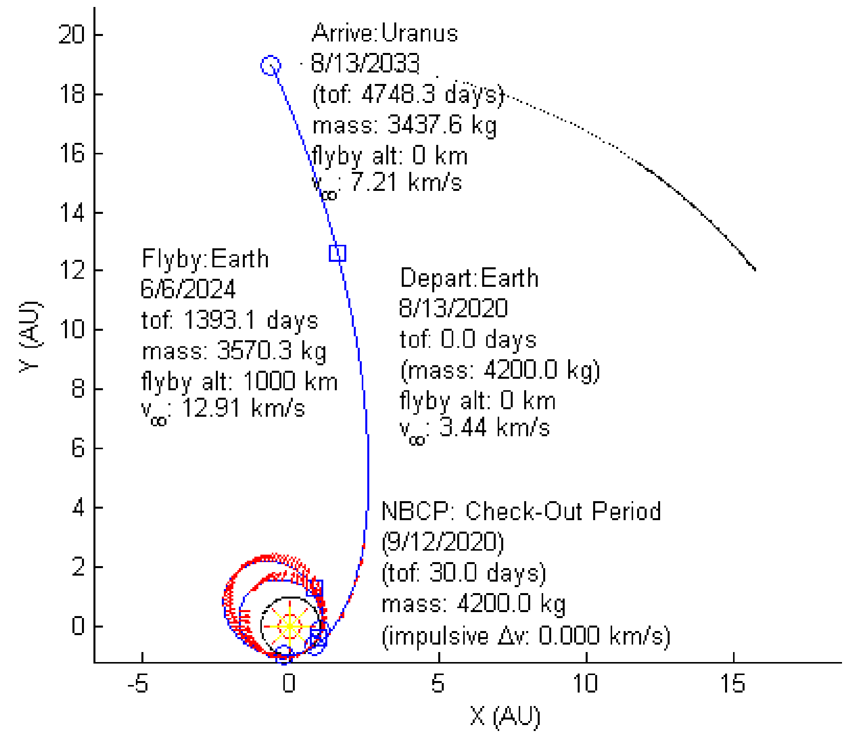


- Cruise: Solar Electric Propulsion (SEP) with an Earth gravity assist flyby
- Engine: NEXT P10_High Isp
- Number of engines: 2
- Duty cycle: 0.90
- Array power at 1AU: 20 kW

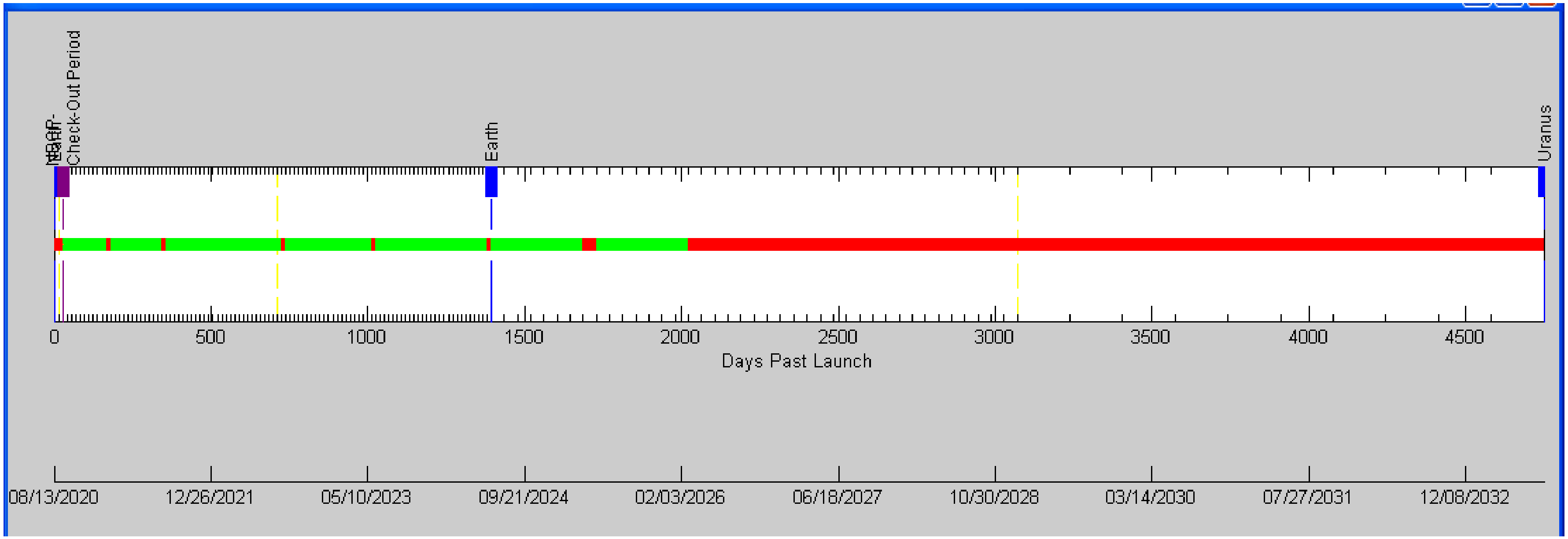
Baseline Interplanetary Trajectory Launch Window and C3 Requirements

- Launch dates: 7/24/2020 – 8/13/2020
- Launch period: 21 days
- Max C3 over the launch period: 11.834 km^2/s^2





Baseline Interplanetary Trajectory SEP Timeline



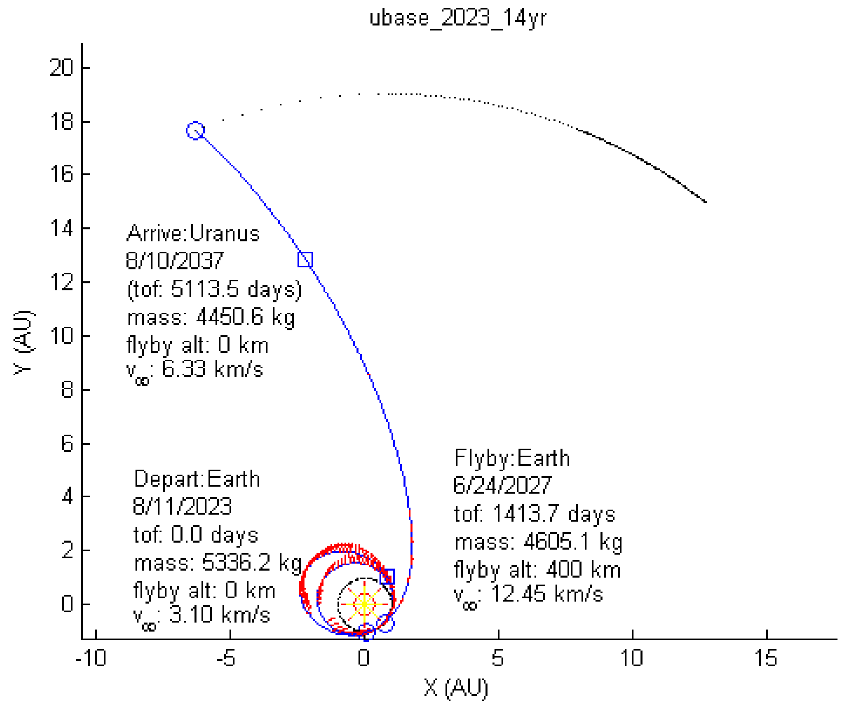
- 7 coast periods
- 30 days check-out after launch
- 42 days prior to EGA
- 5 other 20 – 30 day coast periods

Baseline Interplanetary Trajectory Launch Mass

- **Max Launch C3: 11.834 km^2/s^2**
- **Launch Vehicle: Atlas V (531)**
- **Max Launch Mass: 4205 kg**

Launch Date	Initial Mass (kg)	Final Mass (kg)	SEP Propellant Usage (kg)
7/24/2020	4200	3456.9	743.1
8/3/2020	4200	3454.4	745.6
8/13/2020	4200	3437.6	762.4

Launch Opportunities



Using 20 kW and 3 NEXT Thrusters

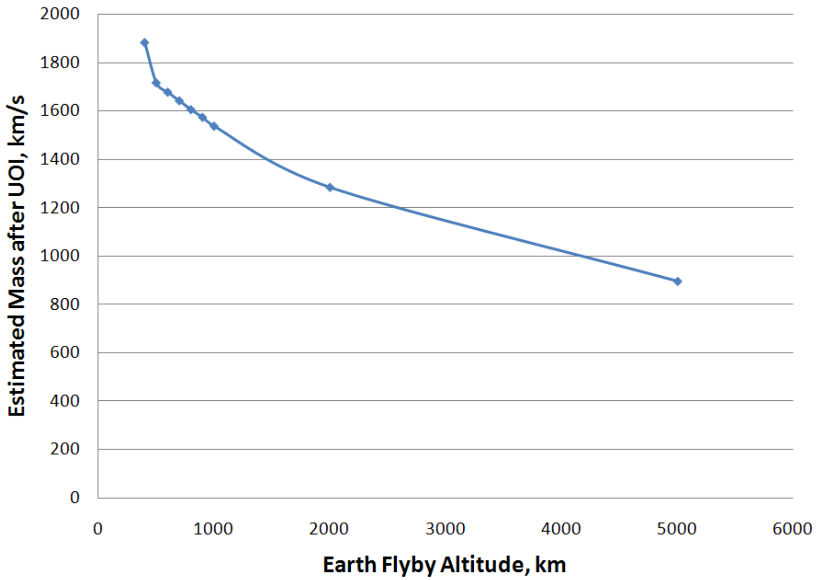
Repeatable launch opportunities

Launch C3 ~ 9.55 – 9.86 km^2/s^2

Iteration will improve repeatability

Comment	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	UOI ΔV, km/s	Chemical Propellant, kg	Estimated Mass after UOI, kg
13yr - Aero	6269	800	430	12.0	Aerocapture	NA	3360
13 yr - 2020	5260	800	853	7.3	1.57	1393	2214
13 yr - 2021	5313	625	909	7.2	1.56	1449	2330
13 yr - 2022	5340	625	926	7.2	1.55	1450	2339
13 yr - 2023	5322	625	910	7.2	1.54	1439	2349
14 yr - 2023	5336	625	886	6.3	1.25	1230	2595

EGA Flyby Altitude Sensitivity



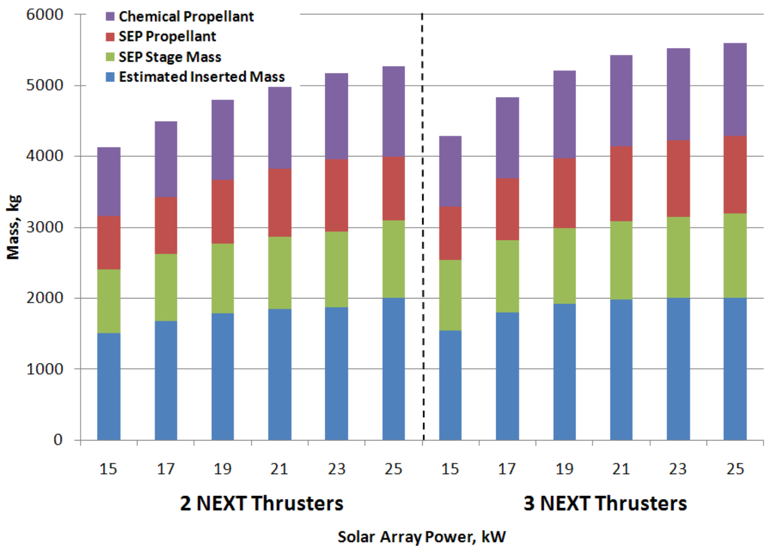
The SEP stage is a constant 2+1 NEXT system with 15 kW solar array power (300W HK)

13 year transfer

Very large performance drop-off with low altitude changes. As the altitude gets higher, changes in performance are less severe.

Flyby Altitude	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	UOI ΔV, km/s	Chemical Propellant, kg	Estimated Mass after UOI, kg
400 km	4421	900	697	7.16	1.53	NA	1882
500 km	4362	900	697	7.17	1.53	1048	1716
600 km	4303	900	698	7.18	1.54	1028	1678
700 km	4244	900	699	7.17	1.53	1003	1642
800 km	4191	900	702	7.18	1.54	984	1606
900 km	4143	900	702	7.19	1.54	967	1574
1000km	4091	900	707	7.20	1.54	947	1537
2000 km	3700	900	717	7.23	1.56	799	1285
5000 km	3101	900	735	7.30	1.58	569	897

Uranus Configuration Options



The transfer time of 13 years is kept constant

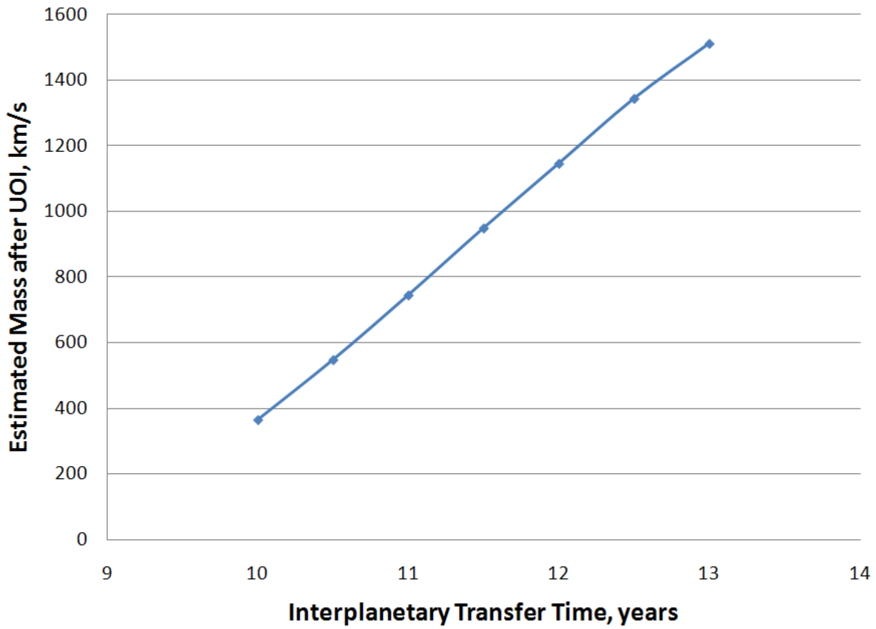
Performance increases significantly with power.

Adding another thruster does not make sense, better to increase power.

Most of these solutions have little to no coast period (90% duty cycle).

System	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	UOI ΔV, km/s	Chemical Propellant, kg	Estimated Mass after UOI, kg
2 NEXT, 15 kW	4133	900	749	7.20	1.54	973	1511
2 NEXT, 17 kW	4491	940	808	7.15	1.53	1065	1678
2 NEXT, 19 kW	4801	980	890	7.15	1.53	1139	1792
2 NEXT, 21 kW	5008	1020	953	7.17	1.53	1184	1850
2 NEXT, 23 kW	5169	1060	1026	7.19	1.54	1209	1873
2 NEXT, 25 kW	5275	1100	890	7.21	1.55	1286	1999
3 NEXT, 15 kW	4281	984	760	7.16	1.53	987	1550
3 NEXT, 17 kW	4832	1024	866	7.16	1.53	1144	1798
3 NEXT, 19 kW	5206	1064	977	7.20	1.54	1241	1924
3 NEXT, 21 kW	5428	1104	1060	7.21	1.55	1283	1980
3 NEXT, 23 kW	5523	1144	1075	7.22	1.55	1302	2002
3 NEXT, 25 kW	5600	1184	1096	7.22	1.55	1308	2011

Uranus Trip Time Sensitivity



The SEP stage is a constant 2+1 NEXT system with 15 kW solar array power (300W HK)

Transfer time primarily only affects the arrival velocity. The arrival velocity and therefore injection ΔV increased significantly with decreases in trip time.

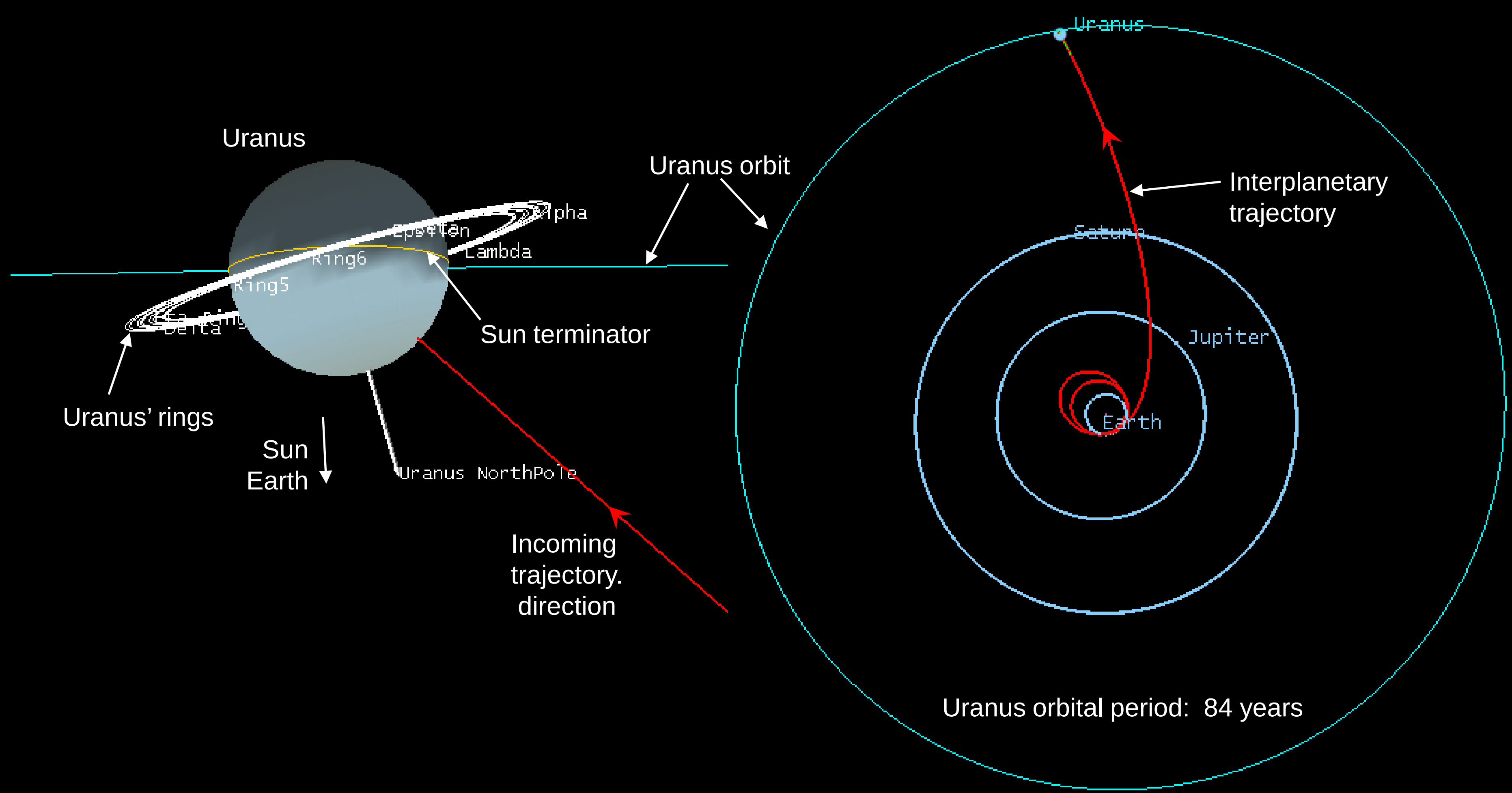
Transfer Time, Years	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	UOI ΔV, km/s	Chemical Propellant, kg	Estimated Mass after UOI, kg
10	2905	900	747	12.27	3.88	894	365
10.5	3183	900	749	11.04	3.22	986	548
11	3428	900	750	10.02	2.72	1034	745
11.5	3651	900	754	9.15	2.33	1048	949
12	3831	900	751	8.40	2.01	1034	1145
12.5	4009	900	753	7.75	1.75	1013	1343
13	4133	900	749	7.20	1.54	973	1511

URANUS ARRIVAL

Uranus Orbit Phase Design Requirements

- Trajectory of Probe and Orbiter outside Uranus rings
- Line-of-sight relay communications from Probe to Orbiter during atmospheric descent
- Probe deceleration limit of 400 g
- Observing periapse from Earth preferred for gravity field measurements
- Periapsis radius close to 1.1 Uranus radii preferred
- 20 primary science orbits for repeated coverage of the same region at ~2 hour intervals for cloud tracking
- Apoapse closer to the planet preferred
- Limited Uranus satellite tour with flybys of major satellites
- Uranus orbit phase duration of 3 years

Uranus Arrival Geometry

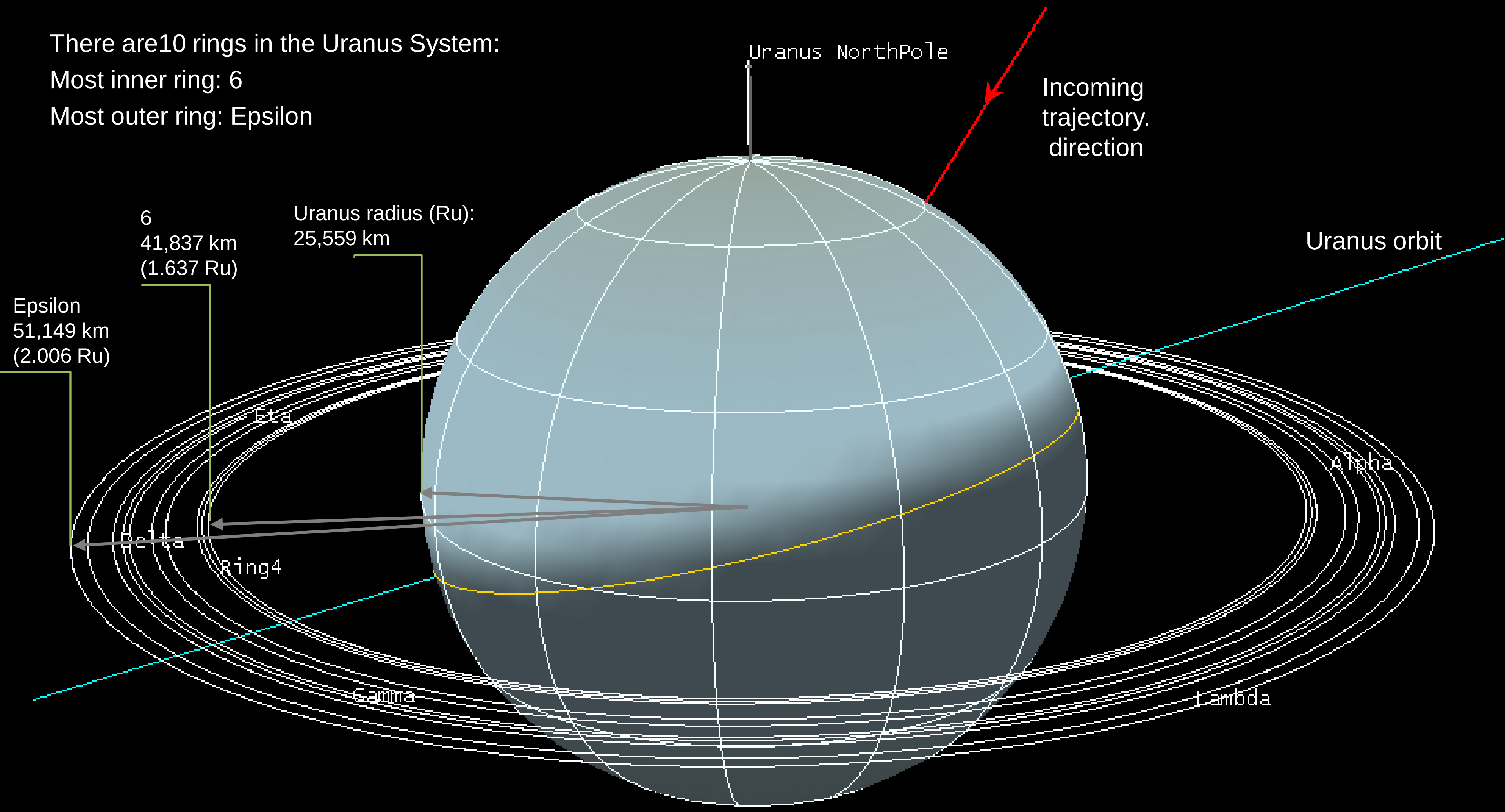


Orbit Insertion Constraint: Rings of Uranus

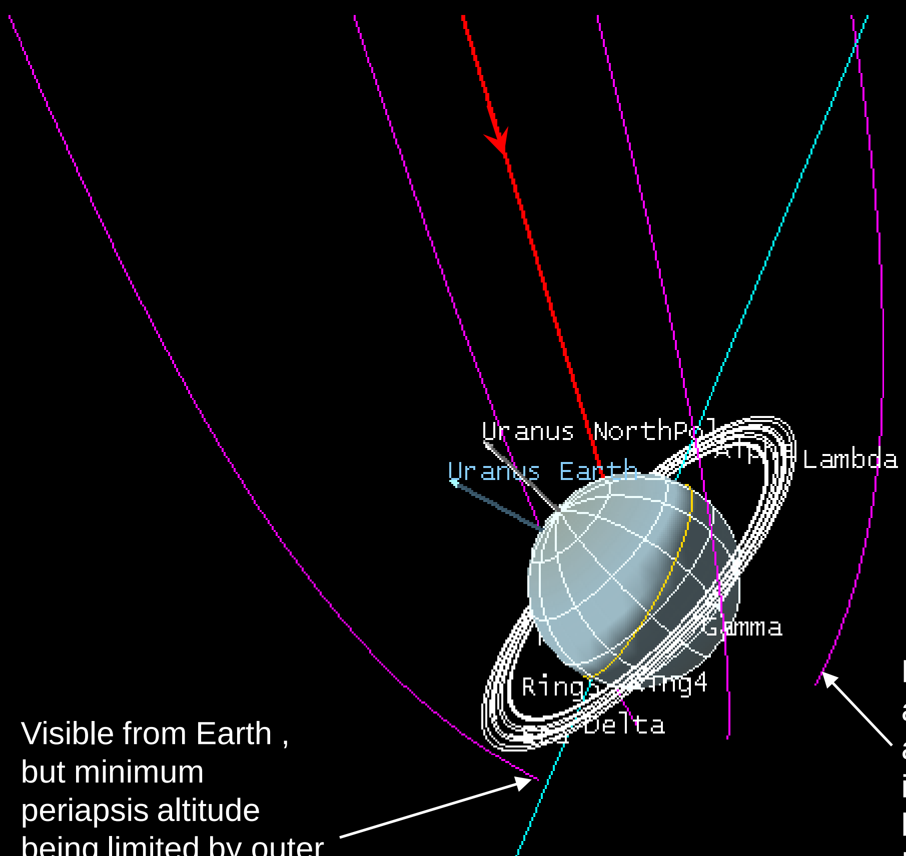
There are10 rings in the Uranus System:

Most inner ring: 6

Most outer ring: Epsilon



Orbit Insertion Constraint: Incoming Asymptote

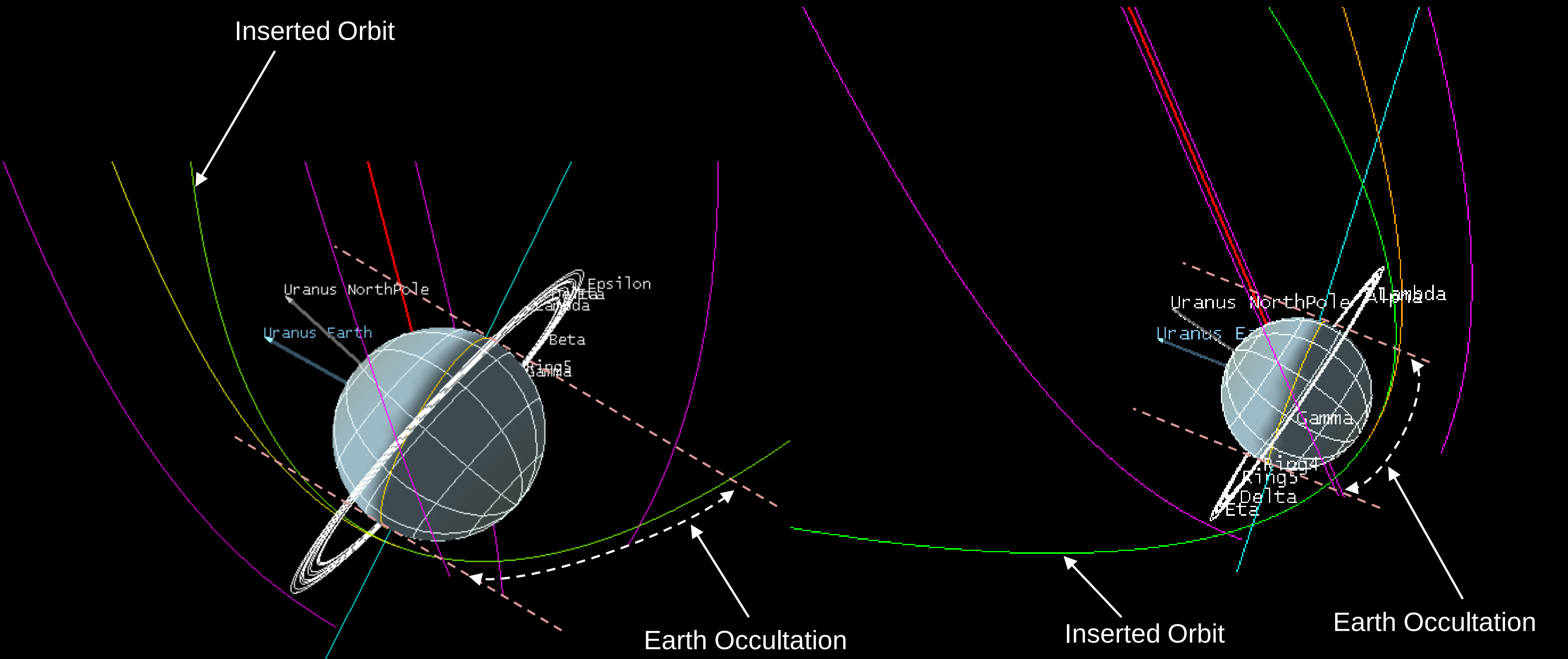


Visible from Earth , but minimum periapsis altitude being limited by outer ring distance (~2Ru)

Lower periapsis altitude can be achieved by targeting insertion on this side , but not visible from Earth

Incoming trajectory. direction

Orbit Insertion Trade



Orbit inserted inside the rings: periapse visible from Earth

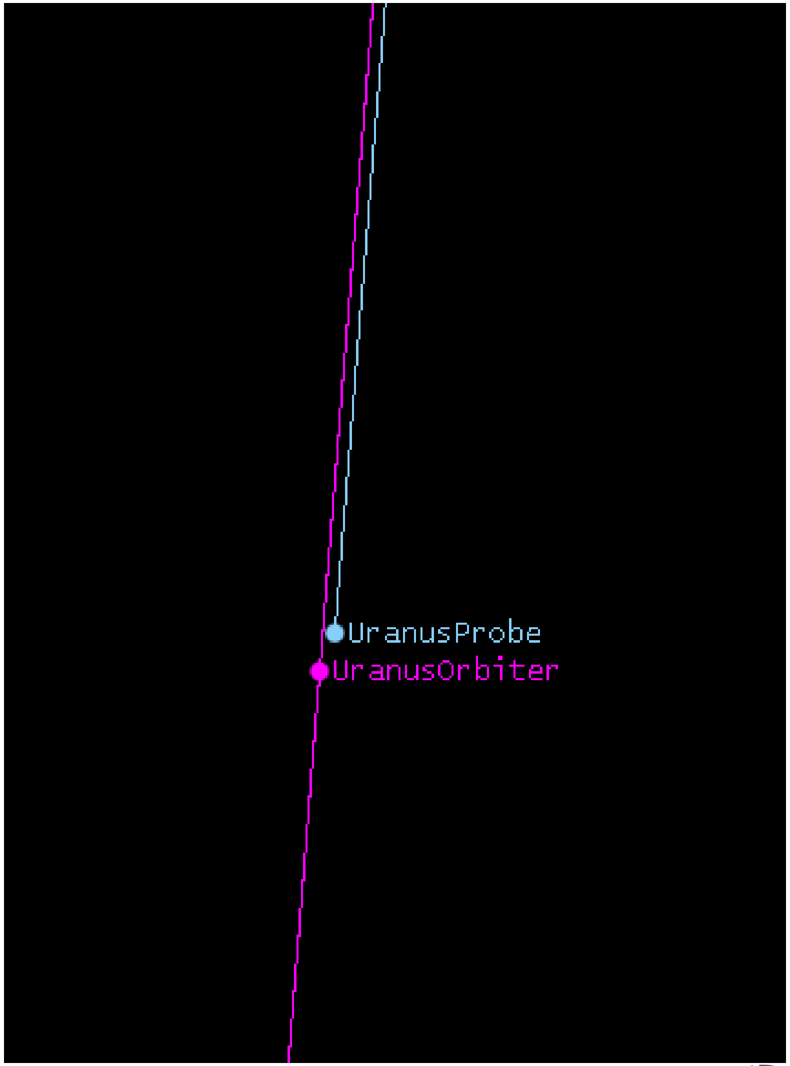
Orbit inserted outside the rings: periapse not visible from Earth

PROBE RELEASE

Probe Release at 29 Days Out

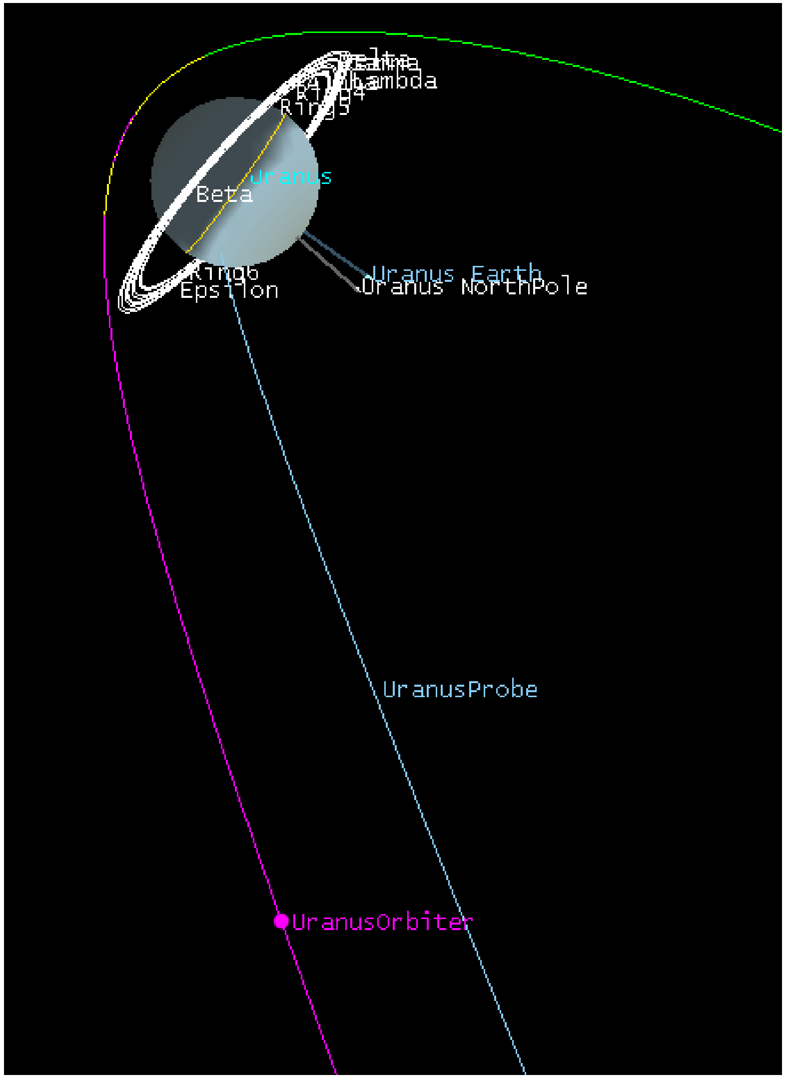
- A TCM will be performed to target at the desired probe entry point
- Prior to releasing the probe, the spacecraft will be spun up to the required spin rate (TBD RPM)
- Probe is released from the Orbiter at 29 days before Uranus arrival*
- After separation the probe will be spinning at an attitude required for entering the atmosphere

* Note: Probe release and Uranus orbit insertion are based on Uranus arrival date from the preliminary interplanetary trajectory



Orbiter and Probe Trajectories Post Orbiter Deflection Maneuver

- One day after Probe release (U-28 days), the Orbiter will perform an Orbiter deflection maneuver (~ 30 m/s) to target at the UOI B-plane aim point
- Orbiter is behind Probe with line-of-sight communication link
- Both Orbiter and Probe are visible from Earth
- Both Orbiter and Probe will not cross the rings

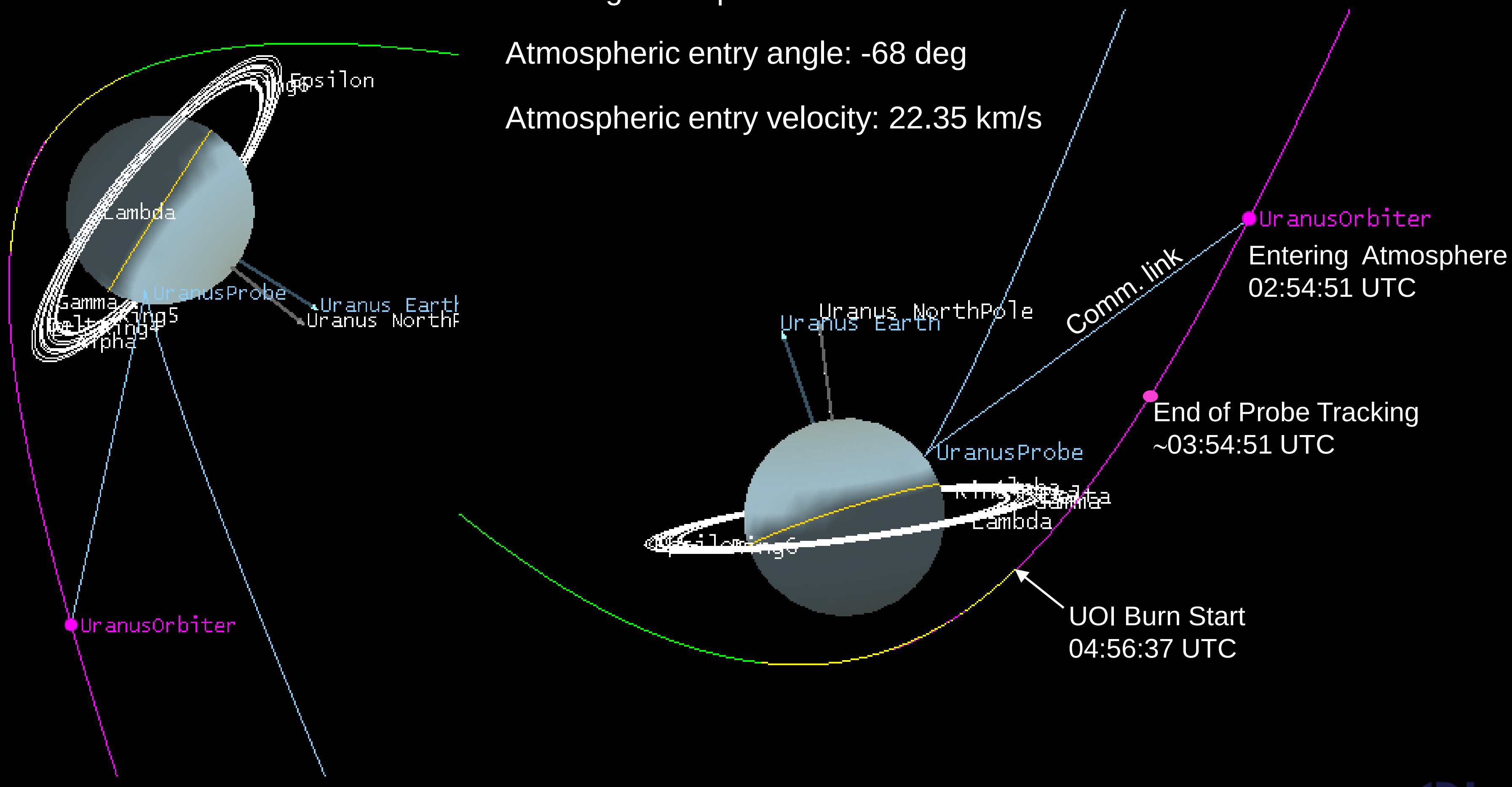


Probe Entering Uranus Atmosphere

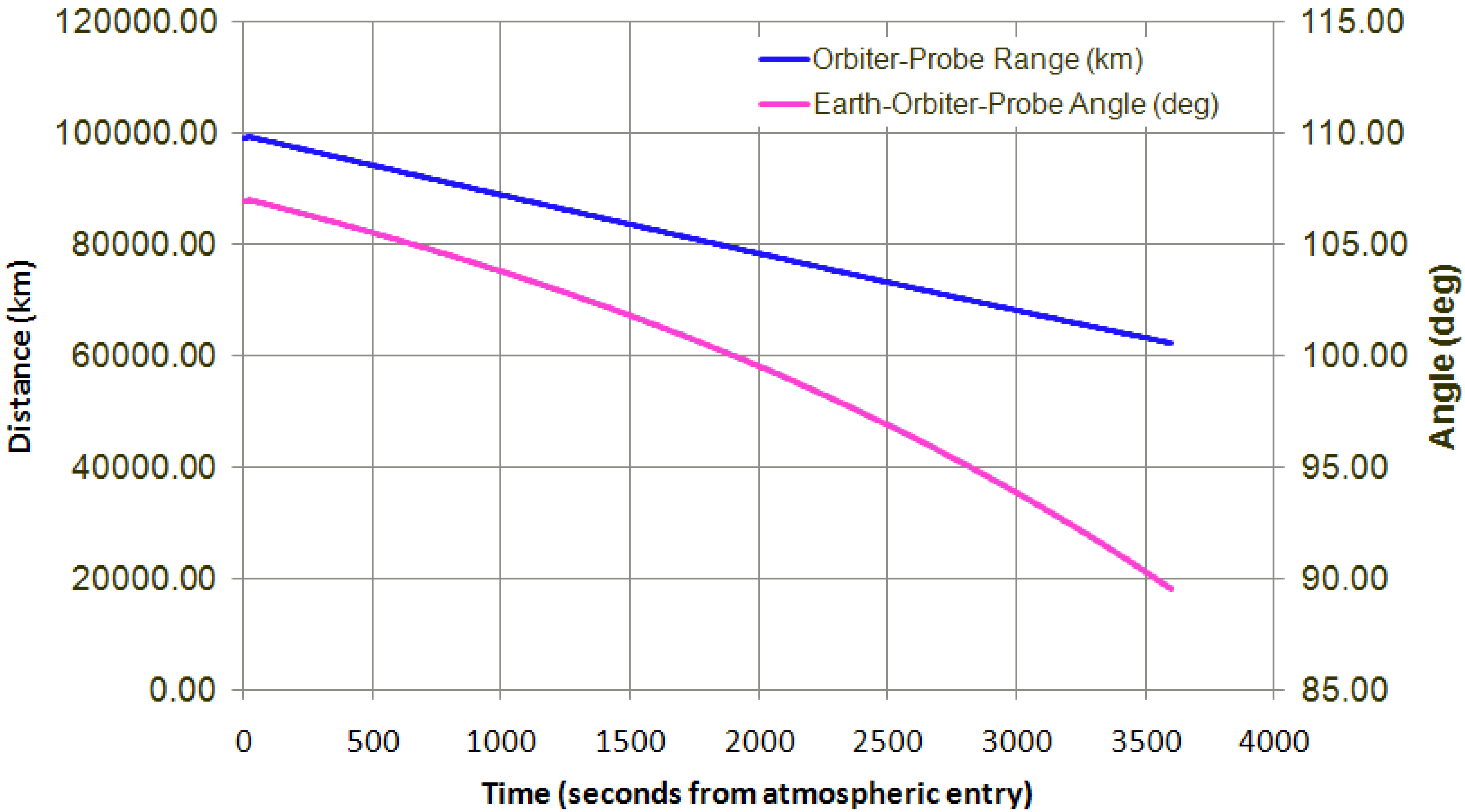
Entering atmosphere: 28 June 2033

Atmospheric entry angle: -68 deg

Atmospheric entry velocity: 22.35 km/s



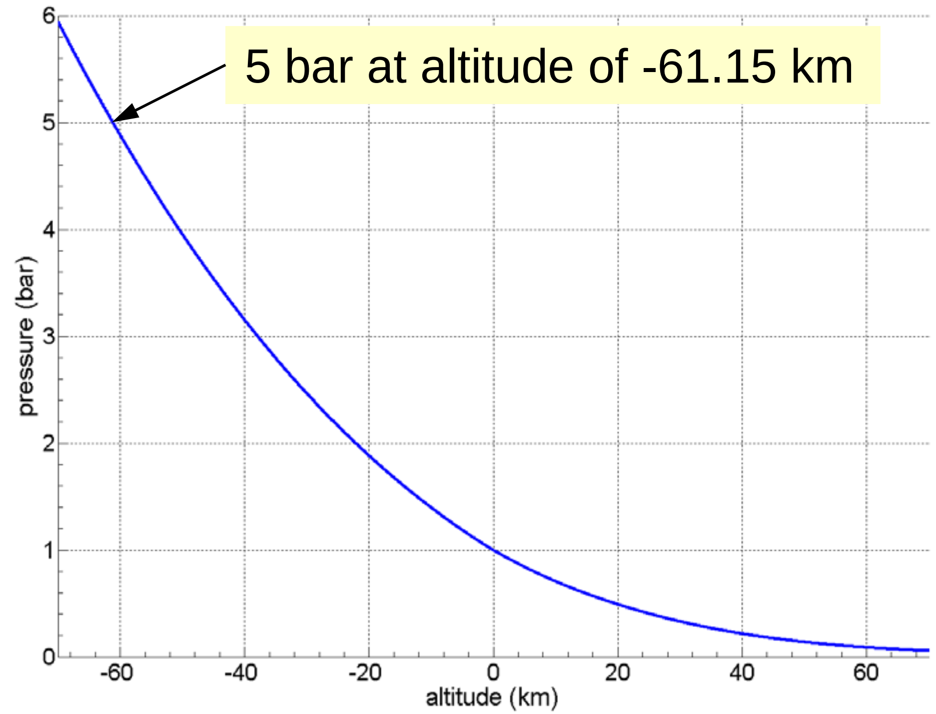
Probe-Orbiter Range During Atmospheric Descent



Probe Atmospheric Entry Modeling

- Probe mass: 127 kg
- Drag coefficient: 1.05
- Area: 0.45 m^2
- Atmosphere model provided by Brigette
  - Model starts at ~550 km altitude
  - Assumed to rotate with planet
  - Other assumptions
    - γ=1.3846
    - R=3,614.91 J/(kg K)

    (Planetary Data System)



$$\mathbf{a}_D = -\frac{1}{2}\frac{c_D A}{m}\rho v_{rel}\mathbf{v}_{rel} = -\frac{1}{2}\frac{\rho}{B} v_{rel}\mathbf{v}_{rel}$$

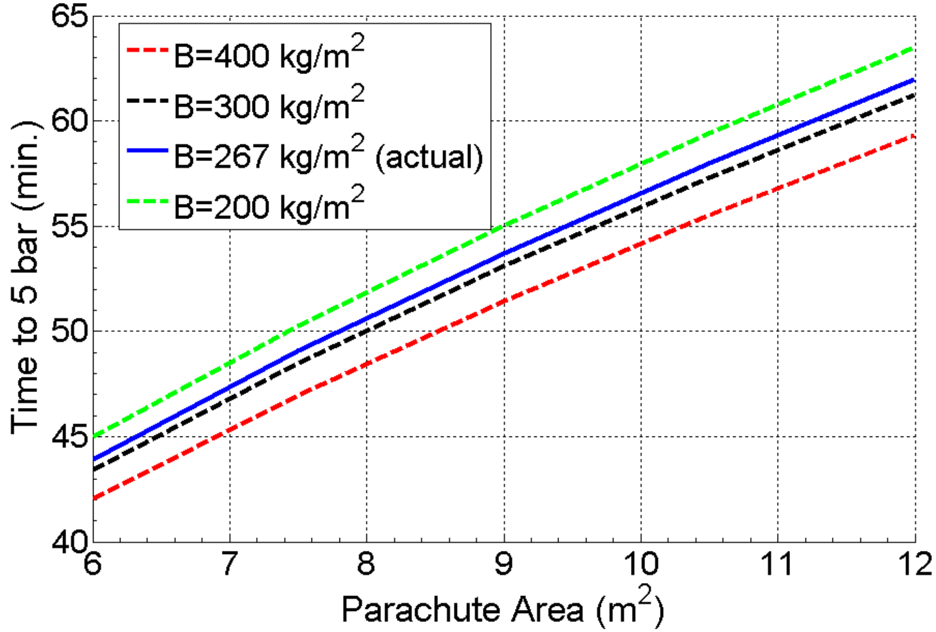
Parachute Area Selection Trade

- Total probe descent duration: ~60 min
- Uncertainty/margin: ~10 min
- Parachute area selected to obtain a nominal probe descent duration at ~50 min

Parachute coefficient of drag: 0.55

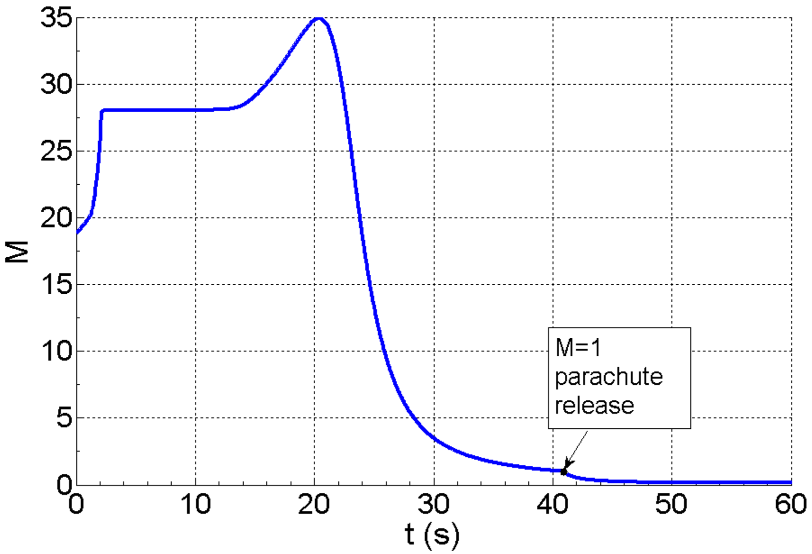
Area of parachute selected: 8.13 m^2

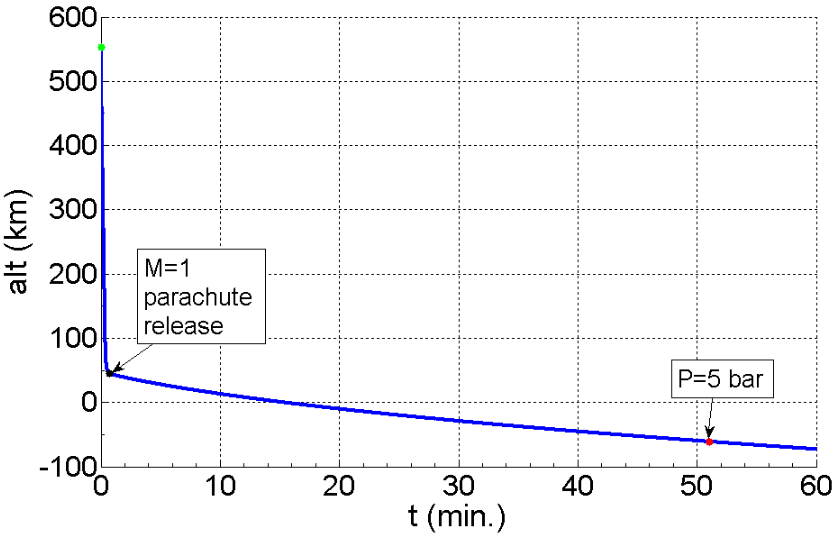
B is the ballistic coefficient



Probe Atmospheric Entry Timeline

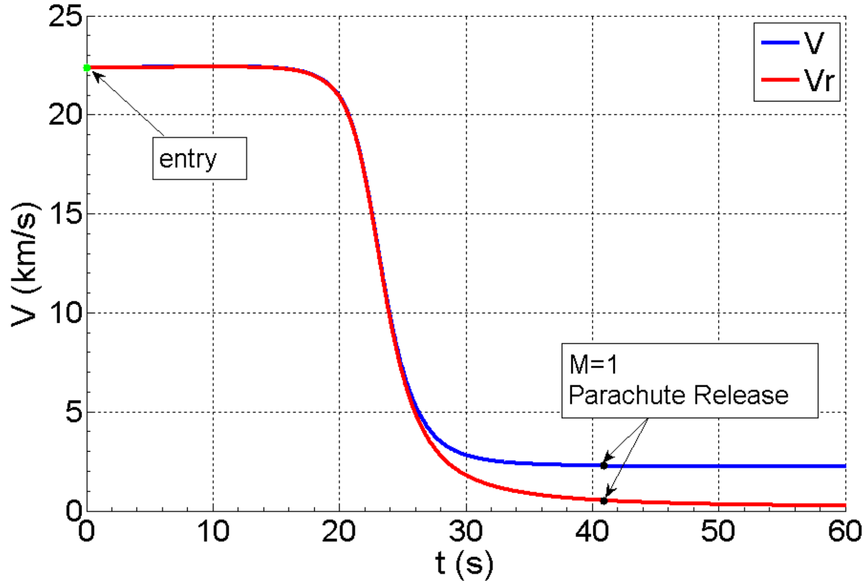
- Time from entry to M=1: ~41 s
- Time from entry to 5 bar: ~51 min
- Drogue parachute opens: <1 min. after entry (not modeled)
- Main parachute opens extending descent to ~1 hr
- (Reference: Galileo's probe descent ~60 min)
- Release parachute at ~45.5 km





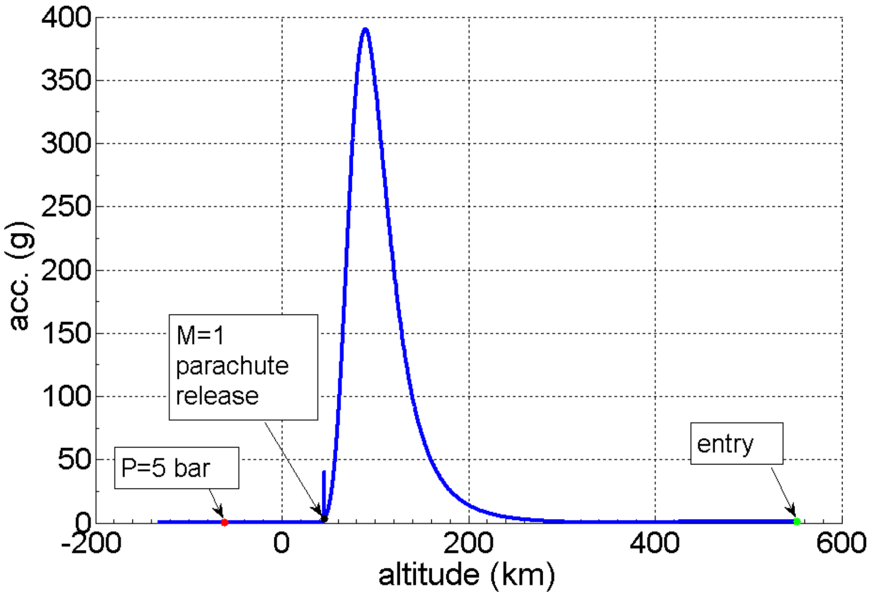
Probe Velocity and Deceleration

Probe Velocity



Blue line: inertial velocity
Red line: velocity wrt atmosphere

Probe Deceleration



Maximum deceleration likely an overestimate due to drag model

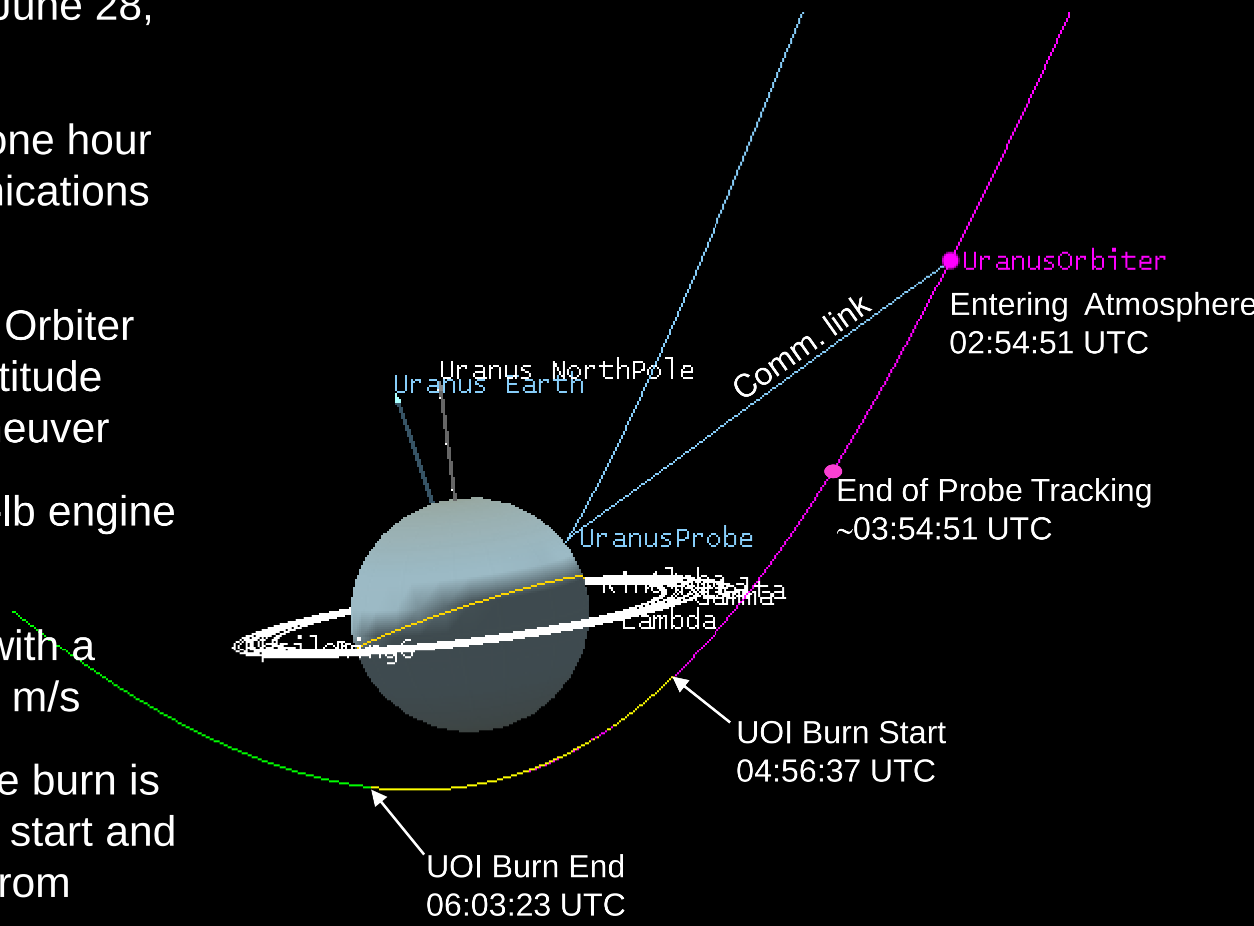
APL-LaRC Comparison

Probe Entry			
	LaRC	APL-MD (case 1)	APL-MD (case 2)
Entry mass (kg)	127	127	127
Flight path angle (deg)	-68	-68	-68
HS Diameter (m)	0.76	0.76	0.76
Parachute diameter (m)	3.25	3.22	3.21
HS Mass (kg)	33	na	na
Vehicle Cd	1.1	1.05	1.1
Parachute Cd	0.55	0.55	0.55
Parachute deploy (Mach Number)	0.9	1	0.9
HS jetison (Mach Number)	0.7	na	na
Peak deceleration (g's)	372	390	389
Atm. Pressure after 1 hr. (bar)	4.83		
Atm. Pressure after 51 min. (bar)		5	5
Peak Heat Rate (W/cm^2)	5511	na	na
Total Heat Load (kJ/cm^2)	38.1	na	na

URANUS ORBIT INSERTION

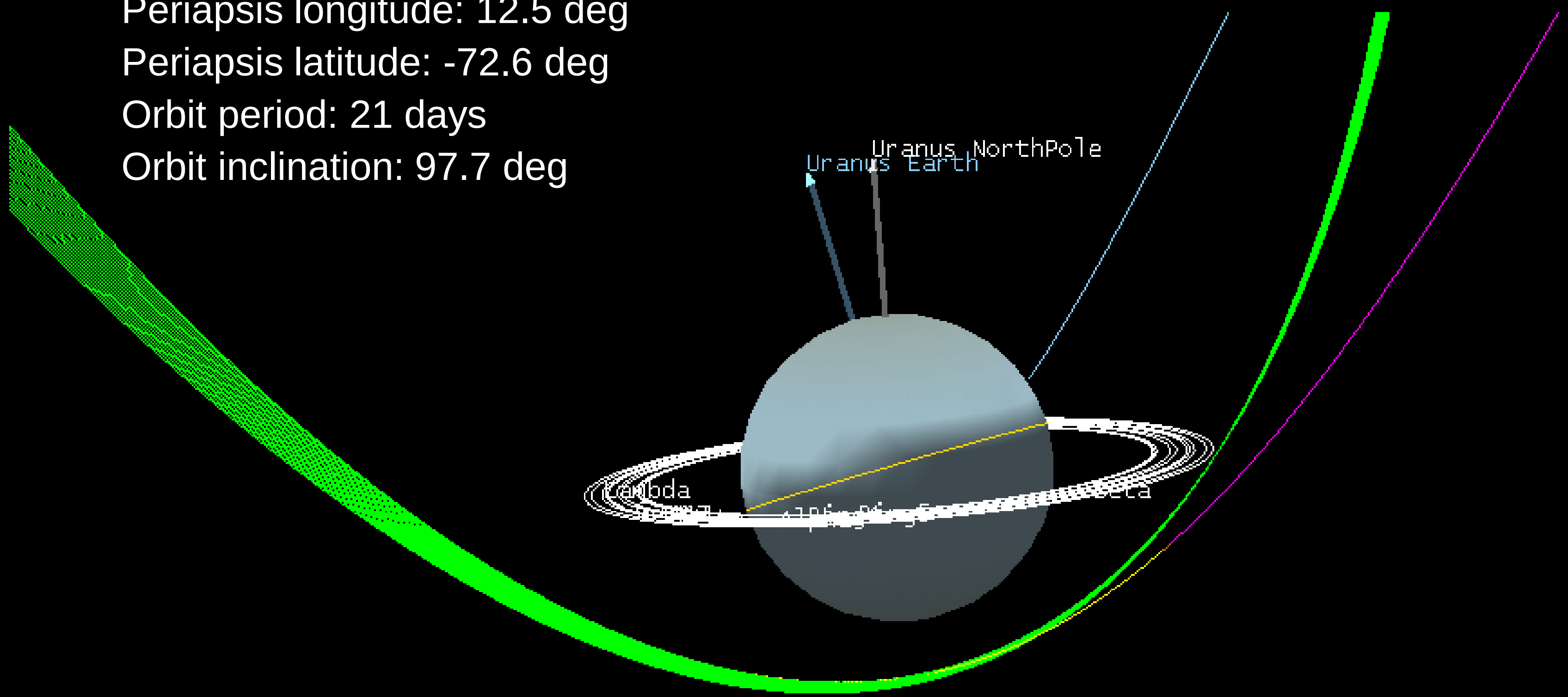
Uranus Orbit Insertion

- Uranus orbit insertion on June 28, 2033
- UOI burn will start about one hour after Probe relay communications completion
- The one-hour time allows Orbiter to slew to the UOI burn attitude and get ready for the maneuver
- UOI burn assumes a 150-lb engine of Isp=332 s
- UOI burn lasts 4006 sec with a burn magnitude of 1660.8 m/s
- Most of the time during the burn is not visible from Earth, but start and end of the burn is visible from Earth



Primary Science Orbit Phase (431 Days)

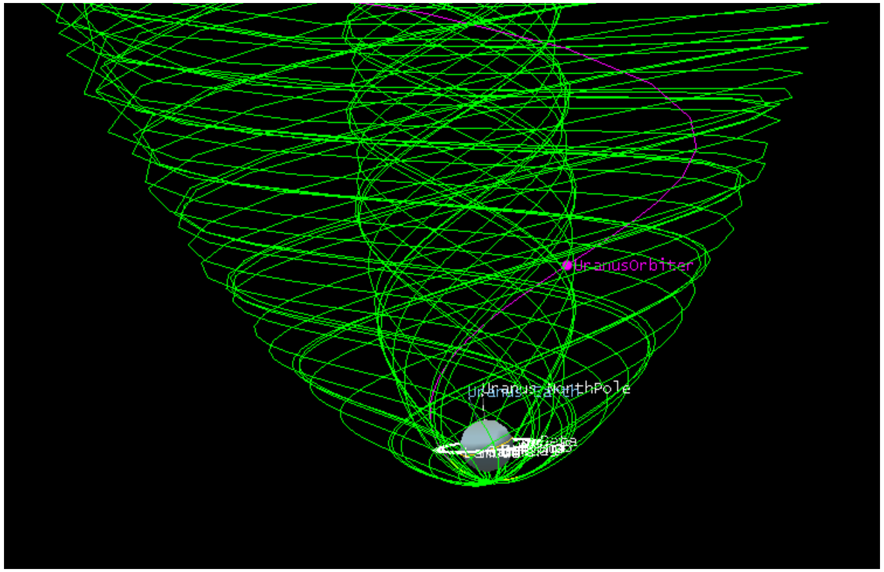
Periapsis radius: 33424.5 km (1.3 Ru)
Periapsis longitude: 12.5 deg
Periapsis latitude: -72.6 deg
Orbit period: 21 days
Orbit inclination: 97.7 deg



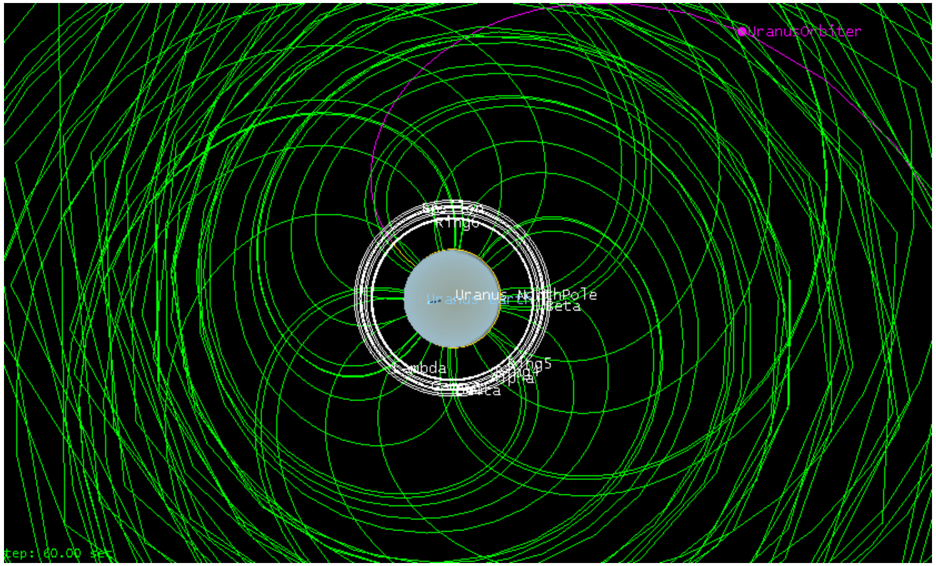
20 highly inclined elliptical orbits providing the desired science measurements (atmosphere, magnetic field, gravity field)

Uranus Observation Coverage

20 Science Orbits Plotted in Uranus Fixed Coordinate System
(purple line: incoming trajectory, green line: science orbits)



Side view



View above the equatorial plane

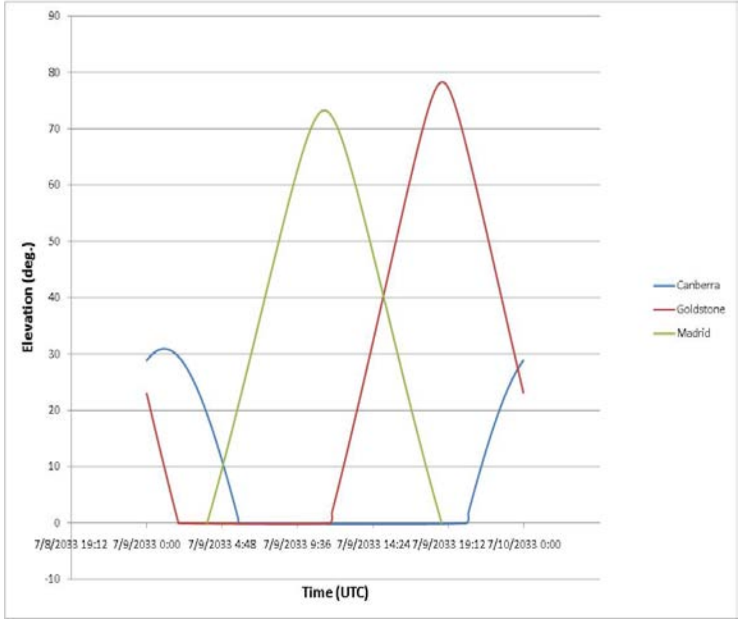
Key Orbit Parameters

Orbit #	Ingress time (Jdate)	Egress time (Jdate)	Duration (s)	Start Lat. (deg)	Start Long. (deg.)	Start Altitude (km)	Stop Lat. (deg)	Stop Long. (deg)	Stop Altitude (km)	Periapse passage (Jdate)	Period (measured, days)	Periapse radius (km)	eccentricity	inclination (deg)	RAAN (deg)	Argument of Periapse (deg)
1	2463797.729	2463797.763	2911	-33.14	24.69	12767	-63.26	-118.51	13197	2463797.746	21.00	33434	0.9508	97.66	354.14	254.30
2	2463818.748	2463818.782	2912	-32.36	118.86	12986	-64.13	-23.88	13071	2463818.765	21.02	33490	0.9507	97.66	354.19	254.17
3	2463839.754	2463839.788	2914	-31.75	-153.63	13171	-64.85	64.05	12996	2463839.772	21.01	33554	0.9507	97.66	354.22	254.02
4	2463860.764	2463860.798	2916	-31.36	-64.32	13298	-65.34	153.69	12971	2463860.781	21.01	33611	0.9506	97.66	354.27	253.89
5	2463881.774	2463881.808	2918	-31.25	25.13	13359	-65.58	-116.69	13014	2463881.792	21.01	33672	0.9506	97.66	354.31	253.75
6	2463902.779	2463902.812	2921	-31.42	111.78	13348	-65.53	-30.03	13125	2463902.796	21.01	33731	0.9505	97.66	354.36	253.61
7	2463923.785	2463923.819	2923	-31.86	-160.72	13273	-65.23	57.33	13298	2463923.802	21.01	33789	0.9504	97.66	354.41	253.48
8	2463944.795	2463944.829	2925	-32.49	-71.31	13155	-64.76	146.52	13522	2463944.812	21.01	33850	0.9504	97.66	354.44	253.34
9	2463965.807	2463965.841	2928	-33.21	18.98	13020	-64.20	-123.45	13765	2463965.824	21.01	33906	0.9503	97.67	354.50	253.21
10	2463986.819	2463986.853	2932	-33.87	109.33	12904	-63.70	-33.29	14006	2463986.835	21.01	33969	0.9503	97.67	354.54	253.06
11	2464007.847	2464007.881	2935	-34.35	-152.25	12828	-63.35	65.04	14194	2464007.863	21.03	34023	0.9502	97.68	354.58	252.94
12	2464028.866	2464028.9	2938	-34.58	-58.42	12813	-63.26	158.85	14325	2464028.882	21.02	34083	0.9502	97.67	354.62	252.80
13	2464049.892	2464049.926	2940	-34.51	38.75	12862	-63.45	-103.84	14375	2464049.908	21.03	34141	0.9501	97.68	354.66	252.66
14	2464070.925	2464070.959	2941	-34.15	140.05	12971	-63.90	-2.29	14345	2464070.941	21.03	34193	0.9501	97.68	354.69	252.54
15	2464091.946	2464091.98	2942	-33.55	-125.11	13141	-64.59	92.93	14264	2464091.962	21.02	34256	0.9500	97.68	354.73	252.39
16	2464112.975	2464113.009	2942	-32.76	-25.77	13349	-65.44	-167.21	14129	2464112.992	21.03	34308	0.9500	97.68	354.77	252.27
17	2464133.995	2464134.029	2943	-31.87	68.47	13589	-66.40	-72.36	13980	2464134.012	21.02	34369	0.9499	97.67	354.80	252.13
18	2464155.015	2464155.049	2944	-30.95	163.32	13840	-67.38	23.22	13824	2464155.033	21.02	34425	0.9499	97.68	354.85	252.00
19	2464176.036	2464176.07	2946	-30.09	-101.75	14083	-68.31	118.91	13684	2464176.054	21.02	34481	0.9498	97.68	354.89	251.87
20	2464197.059	2464197.093	2948	-29.34	-5.48	14299	-69.12	-144.08	13577	2464197.078	21.02	34537	0.9497	97.68	354.93	251.74

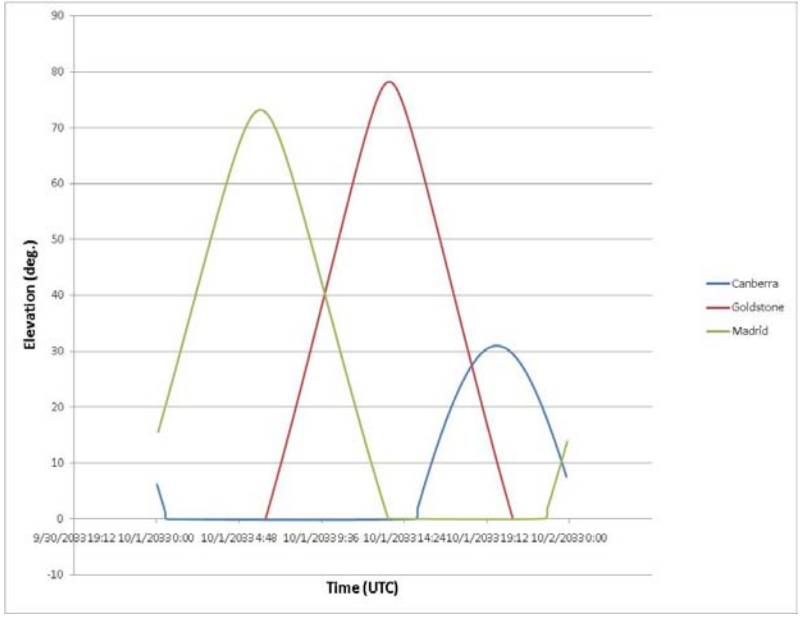
Communications with Orbiter During Orbit Phase (1 of 2)

- Orbiter geometry with respect to Earth investigated for assessment of data downlink capability
- Communication access from the three DSN stations (Canberra, Goldstone, Madrid) to Orbiter analyzed for all seasons around the year
- Earth-Orbiter ranges and elevation angles are computed for 4 full representative orbits at 4 different seasons

Summer 2033, Orbit 1

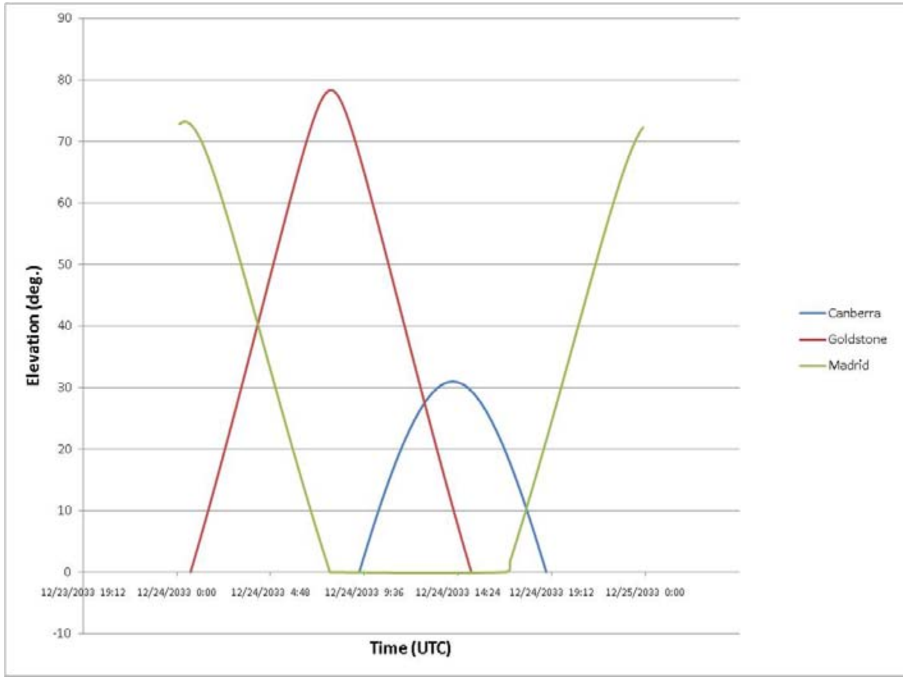


Fall 2033, Orbit 5

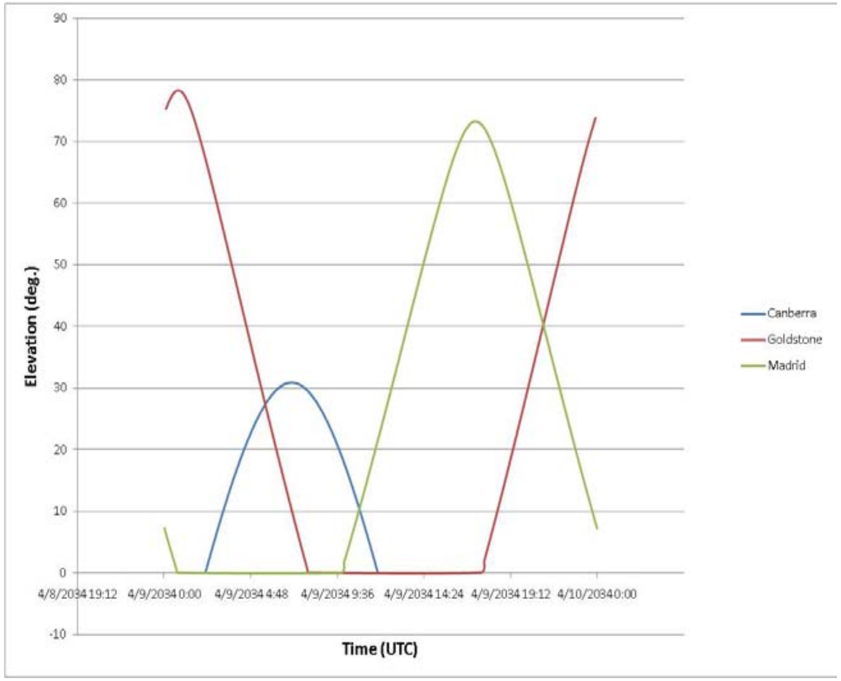


Communications with Orbiter During Orbit Phase (2 of 2)

Winter 2033, Orbit 9



Spring 2034, Orbit 14



Spacecraft Heating and Drag Estimate

Periapsis Radius (km)	Periapse (Ru-Polar)	Periapse (Ru-Equator)	rho (cm^-3)	rho (g/cm^3)	Vp (km/s)	Heating Rate (W/cm^2)	rho (kg/km^3)	drag a (m/s^2)
25597	1.025	1.001		9.82E-11	21.071	4.59E+01	9.82E+01	1.50E-01
25972	1.040	1.016	1.00E+10	1.67E-14	20.917	7.64E-03	1.67E-02	2.52E-05
27470	1.100	1.075	1.00E+09	1.67E-15	20.332	7.02E-04	1.67E-03	2.38E-06
29968	1.200	1.172	1.00E+08	1.67E-16	19.454	6.15E-05	1.67E-04	2.18E-07
31216	1.250	1.221	3.00E+06	5.01E-18	19.055	1.73E-06	5.01E-06	6.27E-09
37460	1.500	1.466	1.00E+05	1.67E-19	17.366	4.37E-08	1.67E-07	1.74E-10
49946	2.000	1.954	1.00E+03	1.67E-21	14.984	2.81E-10	1.67E-09	1.29E-12

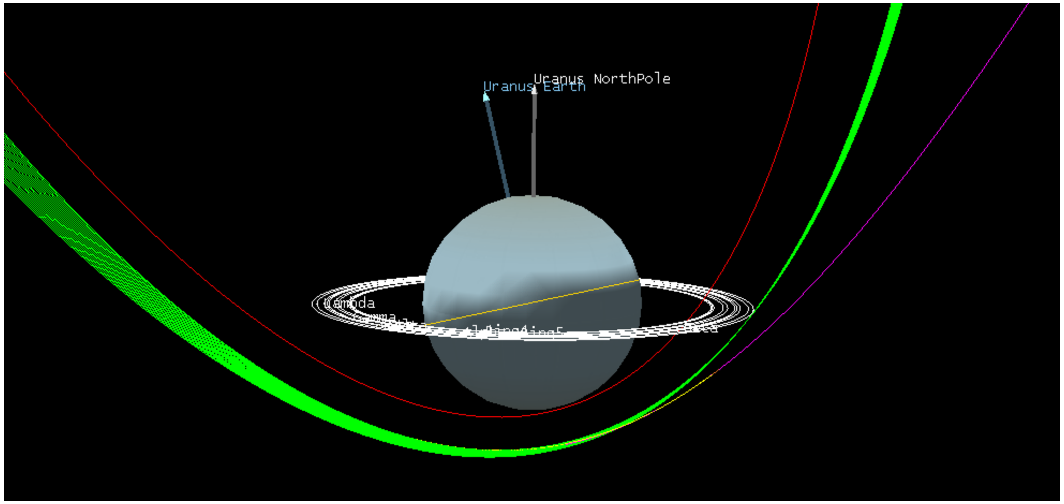
- Parameters used in the estimation:
  - SC mass: 1500 kg
  - SC area: 4.7 m^2
- Atmosphere data at radius 25972 km and greater are based on the Uranus book (Jay Bergstralh)
- Atmosphere data at radius 25597 and lower are based on the Excel table provided by Brigette
- SC heating rate exceeds the limit at a radius between 25597 km and 25972 km

APL

Option of Lowering Periapse at the End of Science Orbit Phase

- Periapse closer to Uranus is desired for measuring gravity field J6 term
- Due to the ring crossing constraint, the achievable minimum periapsis radius is 1.3 Ru for the science orbit
- Periapse altitude can be lowered near the end of science orbit phase if desired, after assessment in the early part of the mission of the hazard posted by the environment inside the rings
- The closest periapsis distance possible is limited by the spacecraft heating rate, which is estimated in the range between 25,600 km and 26,000 km

- Delta-V required for periapse change: 56 m/s
- For periapse of 25,600 km, the min altitudes before and post Earth occultation are:
  - Ingress altitude: 6961 km
  - Egress altitude: 6170 km
  - Occultation duration: 2623.3 s

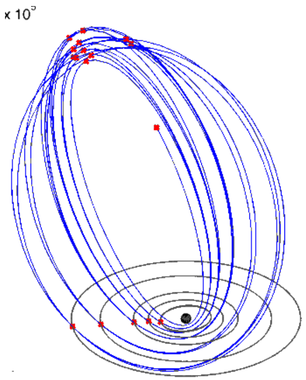


URANUS SATELLITE TOUR

Baseline Tour

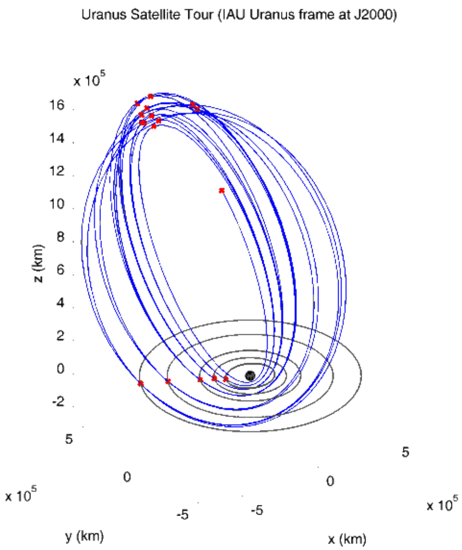
JULIAN DATE	Body	Event	DeltaV(m/s)
2464214.500	-1	ref_orbit	0.000
2464228.631	0	maneuver	88.400
2464239.462	0	maneuver	8.239
2464261.511	705	flyby(16:1)	4.979
2464271.326	0	maneuver	1.401
2464284.127	705	flyby(16:1)	0.000
2464295.355	0	maneuver	79.107
2464307.063	0	maneuver	8.932
2464330.757	701	flyby(10:1)	0.000
2464341.784	0	maneuver	0.984
2464355.961	701	flyby(10:1)	0.000
2464368.416	0	maneuver	99.205
2464381.672	0	maneuver	2.687
2464407.760	702	flyby(6:1)	0.000
2464419.130	0	maneuver	0.956
2464432.626	702	flyby(6:1)	0.000
2464547.636	0	maneuver	181.205
2464557.962	703	flyby(3:1)	0.000
2464569.187	0	maneuver	2.094
2464584.079	703	flyby(3:1)	0.000
2464598.082	0	maneuver	139.335
2464611.565	704	flyby(2:1)	0.000
2464622.640	0	maneuver	1.548
2464638.489	704	flyby(2:1)	0.000

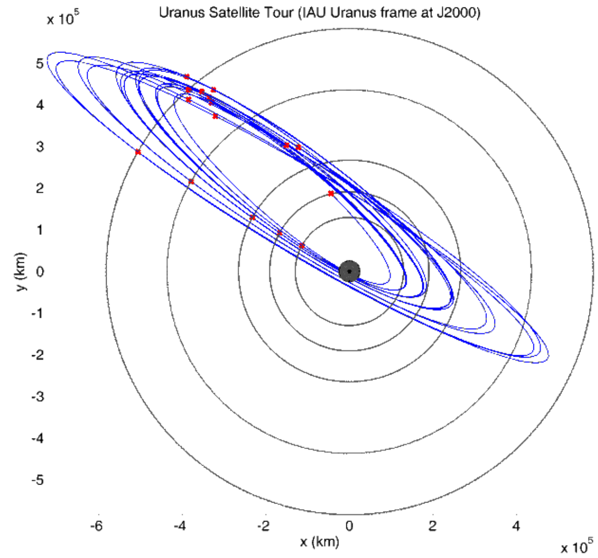
- **TOF total (day) = 424 days**
- **DV total (m/s) = 619 m/s**
- 10 total targeted flybys + 4 close untargeted
  - 2 Miranda (16:1)
  - 2 Ariel (10:1)
  - 2 Umbriel (6:1) + 4 close untargeted
  - 2 Titania (3:1)
  - 2 Oberon (2:1)
- Red labels indicate two-impulse maneuvers that guarantee phasing in ~2 s/c revs. The first maneuver is to target the ring plane crossing, while the second maneuver is at the ring plane and adjusts the period to ensure proper phasing.
- The two-impulse solutions are less favorable for Titania and Oberon because their orbits are much larger than the other moons…Therefore, we use a 1-impulse solution here…
- Note the green labels indicate a very long flight time after the maneuver that achieves the Titania transfer. This was considered an acceptable trade compared to ~50 m/s or so extra required for the min time solution.
- In most cases resonances were chosen based on the closest available (and with just 1 s/c rev when possible)
- Considering the initial phase free cost to transfer to Oberon was found to be ~633 m/s, The 619 m/s total here is energetically consistent with a floor value.
- Most of the small maneuvers are to account for the fact that there are no exact resonances in a real ephemeris (The body doesn't come exactly back to where it started.)
- Any maneuver at a flyby is performed at the sphere of influence
- The large maneuvers are to target the next moon…

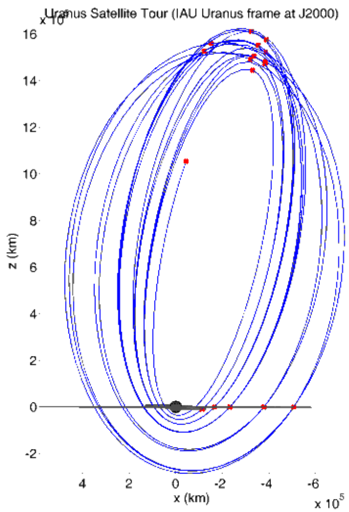


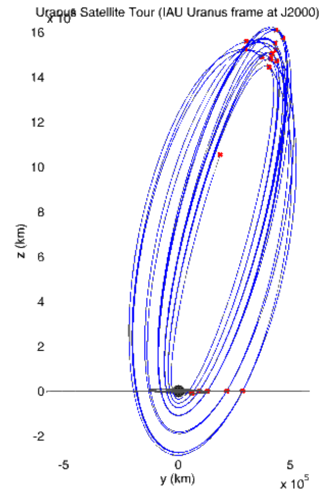
Baseline Tour: Uranus Frame Trajectory

(red dots are locations of Moon Encounters and maneuvers)



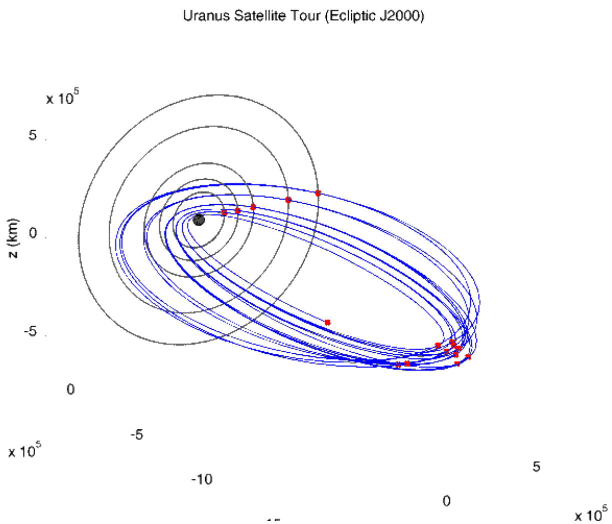


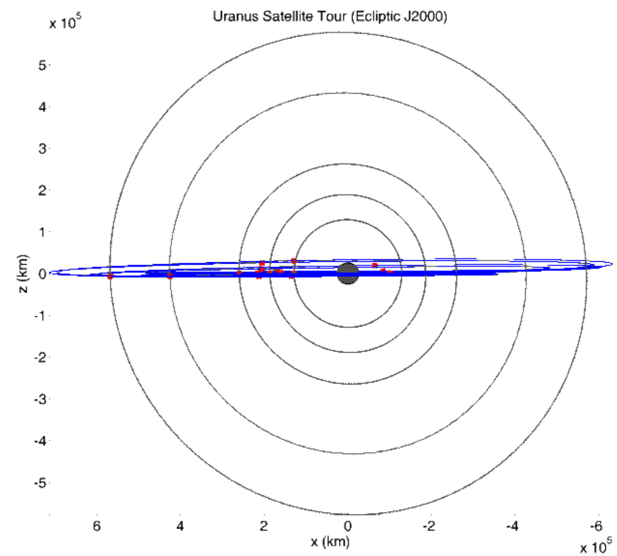


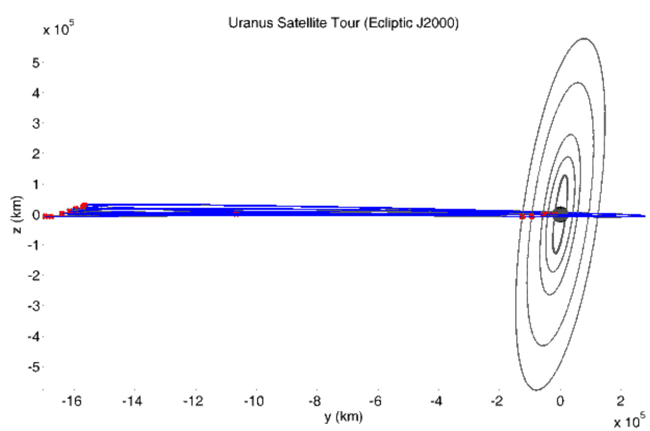


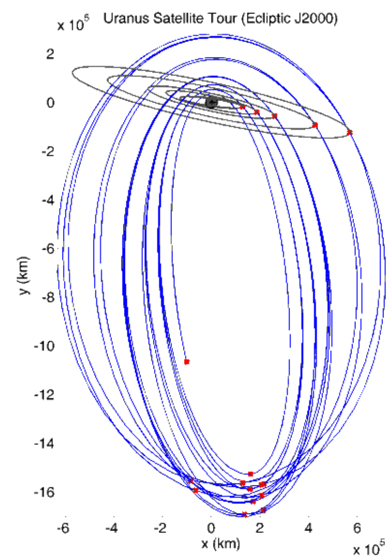
Baseline Tour: Ecliptic Frame Trajectory

(red dots are locations of Moon Encounters and maneuvers)









Baseline Tour: Untargeted Close Approaches

- Values come from integrating the Uranus centric orbit (same patched conic ephemeris model) and checking distances to all moons at each time step.
- Details will change when going to high fidelity design
- Statistically gives an idea of how many untargeted close approaches are expected….
- In high fidelity design we could easily incorporate the low altitude flybys as targeted

ith	body	JD	close approach (km)
1	701	2464306.651	94919.308
2	705	2464330.723	55324.370
3	705	2464355.531	23996.942
4	705	2464380.977	39580.367
5	705	2464431.908	67015.303
6	705	2464432.482	92236.855
7	702	2464457.490	1451.539
8	705	2464482.075	93849.919
9	702	2464482.356	2550.368
10	705	2464506.635	92646.256
11	701	2464507.137	52413.893
12	702	2464507.223	3102.354
13	702	2464532.089	3699.888

APL

Baseline Tour: Details on Flyby Geometries

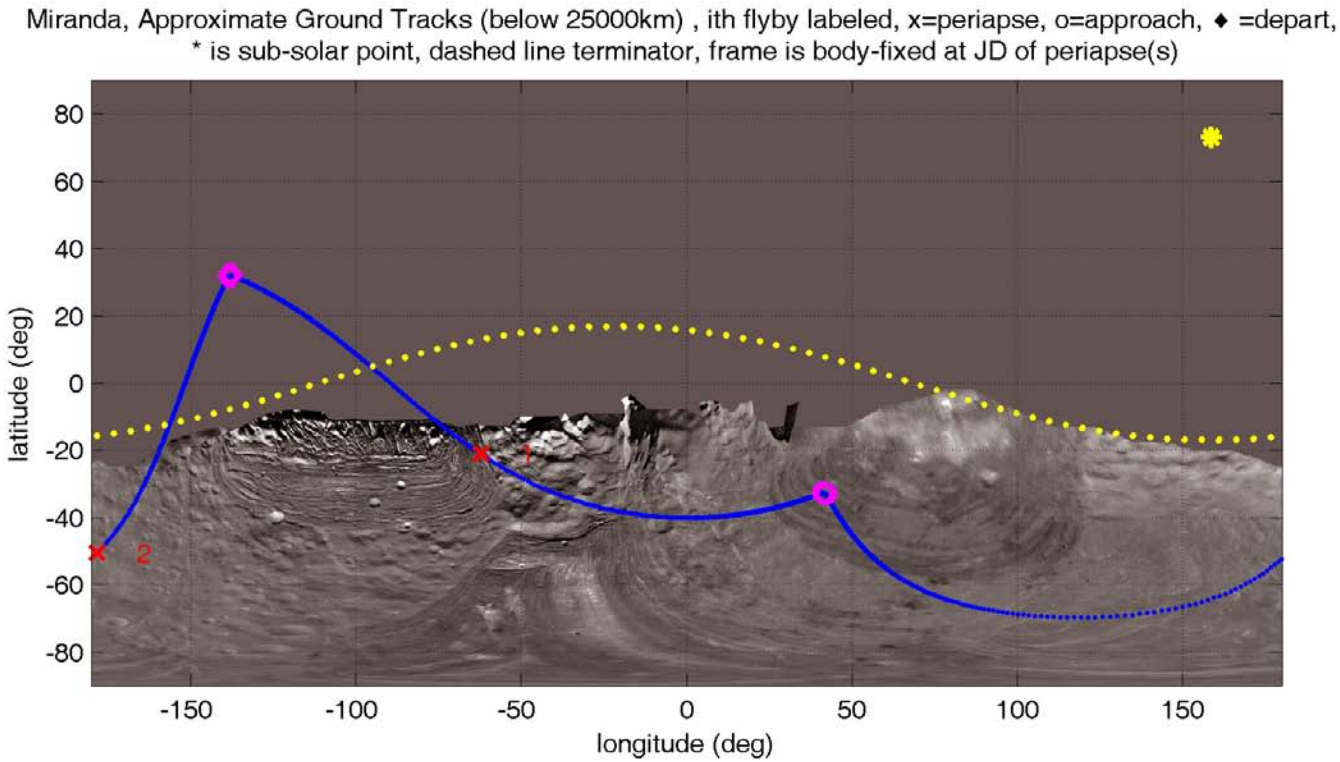
%JD	ifly	body	dv(m/s)	rpalt	vp	rplat	rplon	rpinc	sun_elev_rP	aprSCbodSun_Ang	rtosunRbf	rtosunLONbf	rtosunLATbf
2464261.511	1	705	4.979	50.000	10.880	-21.115	-62.231	139.930	-33.295	128.820	2827415250.471	158.599	73.193
2464284.127	2	705	0.000	50.000	10.878	-50.476	-178.214	69.748	-34.453	128.966	2826829739.900	158.664	72.941
2464330.757	3	701	0.000	50.000	8.808	-32.825	-72.070	130.986	-44.116	131.052	2825593965.470	160.121	67.888
2464355.961	4	701	0.000	50.000	8.808	-50.865	-178.898	69.729	-29.478	131.045	2824943164.436	160.481	67.604
2464407.760	5	702	0.000	50.000	7.339	-11.714	140.176	35.397	9.778	130.544	2823581254.062	160.666	67.104
2464432.626	6	702	0.000	49.997	7.345	-48.937	-179.061	68.442	-26.743	131.853	2822941675.196	161.056	66.821
2464557.962	7	703	0.000	50.000	5.589	-40.491	171.683	59.276	-16.021	132.505	2819664165.031	163.166	65.322
2464584.079	8	703	0.000	50.000	5.586	-44.810	-178.930	65.472	-20.696	133.239	2818997092.283	163.396	65.022
2464611.565	9	704	0.000	50.000	4.698	-29.799	166.662	51.620	-4.931	132.876	2818237419.118	163.398	65.097
2464638.489	10	704	0.000	50.000	4.697	40.105	0.608	117.935	15.742	132.206	2817551209.866	163.566	64.786

rplat= close approach latitude (deg)
rplon = close approach longitude (deg)
rpalt = close approach altitude (km)
vp= close approach velocity (km/s)
rpinc= inclination of flyby hyperbola in body fixed frame (deg)
aprSCbodSun_Ang= SC-Body-Sun Angle when S/C just reaches body sphere of influence
sun_elev_rP= elevation of the sun at close approach
(rtosunRbf,rtosunLONbf,rtosunLATbf)= Position vector in spherical coordinates (km,deg) of the Sun relative to the Body at time of flyby

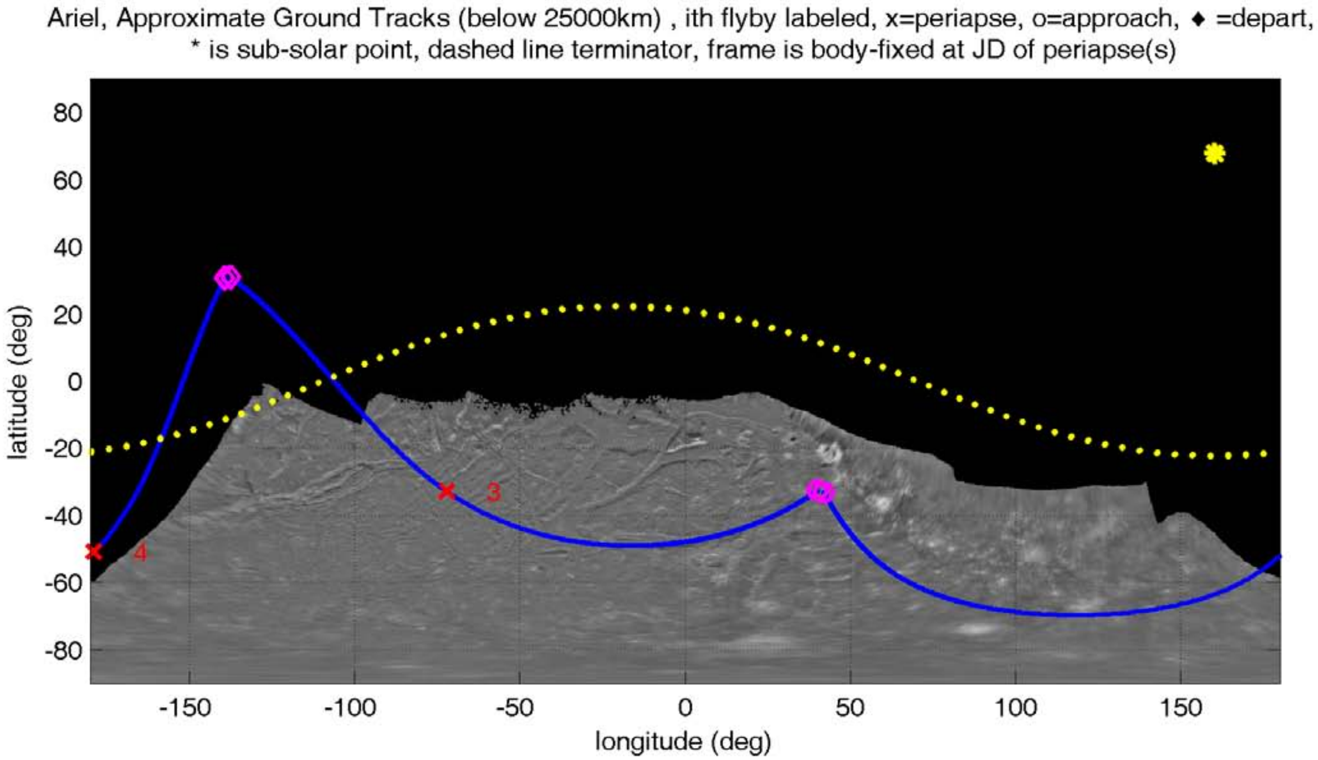
Baseline Tour: Ground Tracks

- Following slides give
  - the ground tracks for each of the moon flybys
  - Close approach location
  - Approximate terminator line
  - Arrival/Departure asymptote direction
- IMPORTANT: detailed design phase will allow better spacing of individual tracks without altering delta-v budget very much. This is possible because the moons are providing very little delta-v because they are low mass. We wont however be able to change the inclination of the flybys (i.e. we are unable to reach higher latitudes without significant design change)

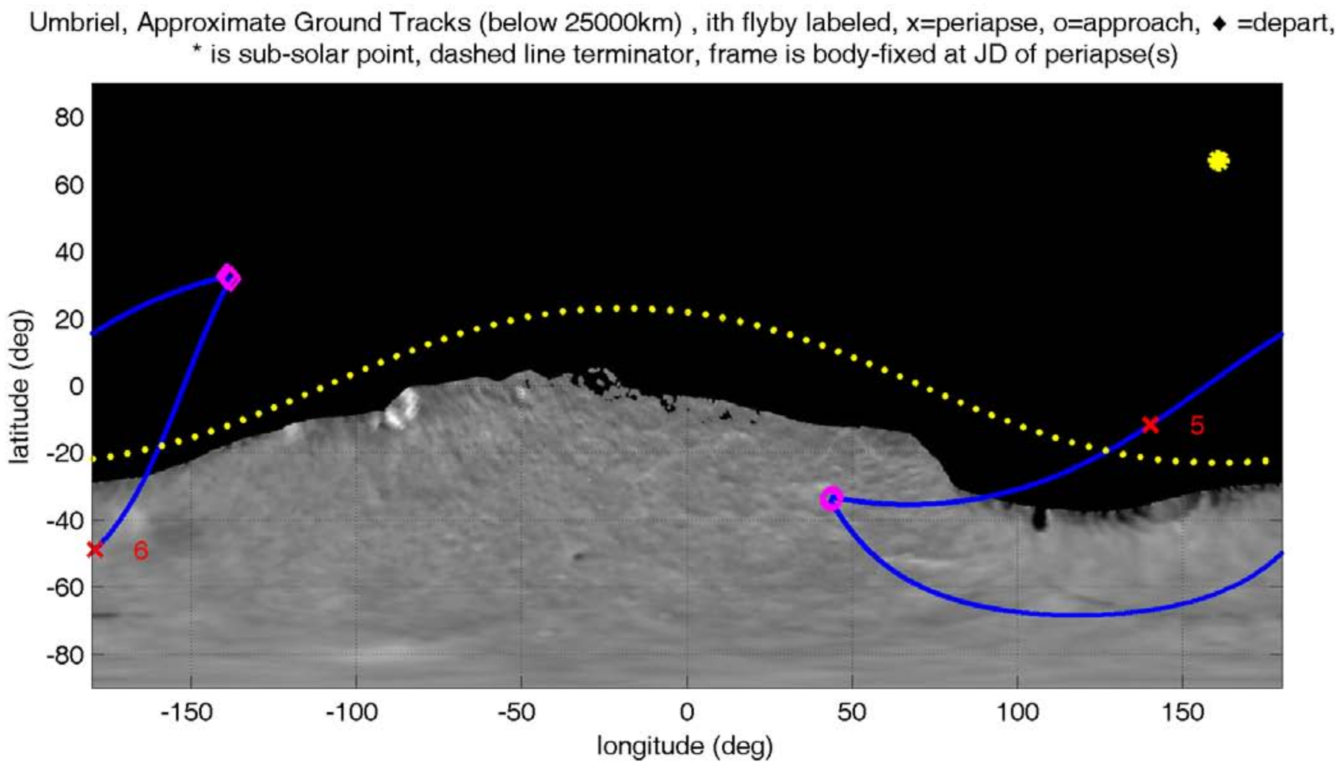
Baseline Tour: Miranda Ground Track



Baseline Tour: Ariel Ground Track

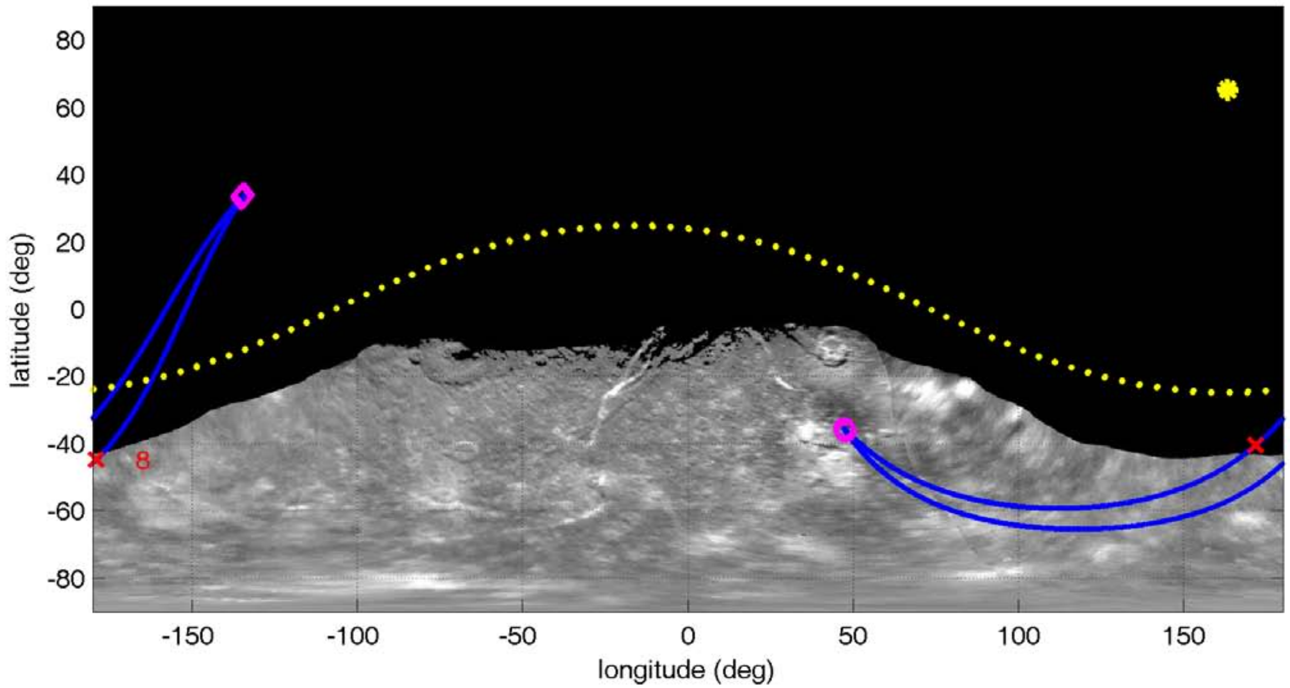


Baseline Tour: Umbriel Ground Track

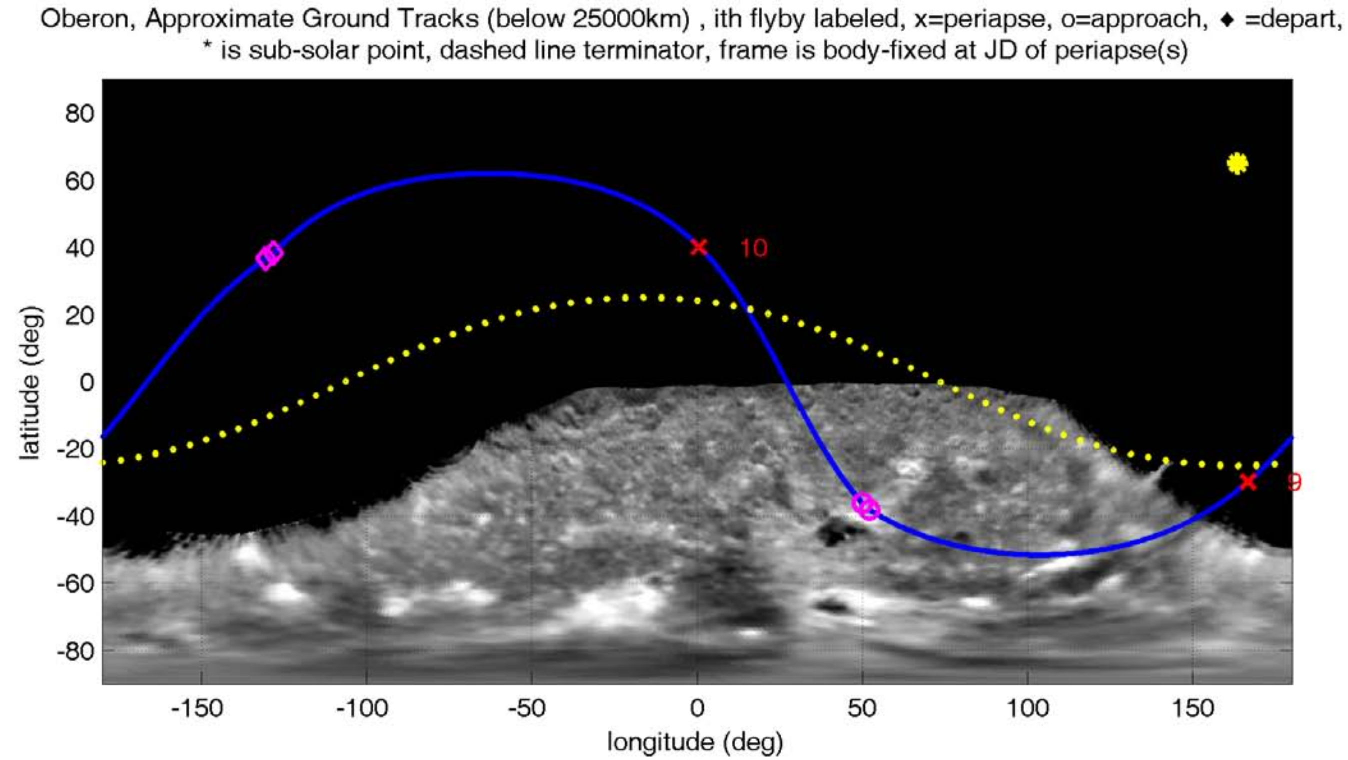


Baseline Tour: Titania Ground Track

Titania, Approximate Ground Tracks (below 25000km) , ith flyby labeled, x=periapse, o=approach, ◆ =depart,
* is sub-solar point, dashed line terminator, frame is body-fixed at JD of periapse(s)



Baseline Tour: Oberon Ground Track

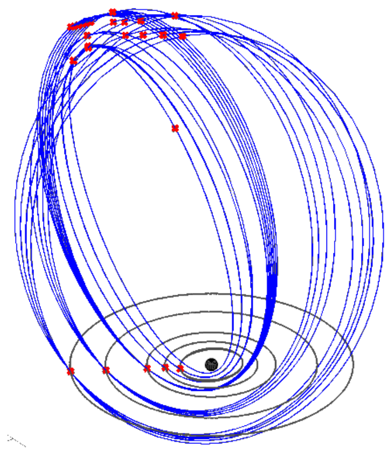


APL

Alternative Tour

JULIAN DATE	Body	Event	DeltaV(m/s)
2464214.500	-1	ref_orbit	0.000
2464291.707	0	maneuver	88.579
2464302.507	705	flyby(15:1)	0.324
2464311.200	0	maneuver	1.244
2464323.709	705	flyby(15:1)	0.000
2464332.403	0	maneuver	1.197
2464344.912	705	flyby(15:1)	0.000
2464353.607	0	maneuver	1.281
2464366.114	705	flyby(15:1)	0.000
2464374.811	0	maneuver	1.500
2464387.316	705	flyby(15:1)	0.000
2464397.745	0	maneuver	83.086
2464408.898	701	flyby(9:1)	0.000
2464442.551	0	maneuver	108.861
2464478.230	702	flyby(6:1)	0.964
2464488.817	0	maneuver	0.376
2464503.097	702	flyby(6:1)	0.000
2464513.624	0	maneuver	0.469
2464527.962	702	flyby(6:1)	0.000
2464538.476	0	maneuver	0.906
2464552.825	702	flyby(6:1)	0.000
2464563.352	0	maneuver	1.022
2464577.691	702	flyby(6:1)	0.000
2464588.226	0	maneuver	0.588
2464602.557	702	flyby(6:1)	0.000
2464613.102	0	maneuver	0.679
2464627.421	702	flyby(6:1)	0.000
2464639.368	0	maneuver	199.215
2464653.772	703	flyby(3:1)	0.000
2464664.993	0	maneuver	0.518
2464679.890	703	flyby(3:1)	0.000
2464690.865	0	maneuver	0.655
2464706.006	703	flyby(3:1)	0.000
2464717.012	0	maneuver	2.349
2464732.123	703	flyby(3:1)	0.000
2464744.057	0	maneuver	153.288
2464759.759	704	flyby(2:1)	0.000
2464770.648	0	maneuver	1.960
2464786.682	704	flyby(2:1)	0.000
2464797.591	0	maneuver	1.887
2464813.607	704	flyby(2:1)	0.000
2464824.489	0	maneuver	1.084
2464840.536	704	flyby(2:1)	0.000
2464851.390	0	maneuver	0.503
2464867.466	704	flyby(2:1)	0.000

- TOF total (day) = 652.96
- DV total (m/s) = 652.5
- 22 total targeted flybys + 1 close untargeted
  - 5 Miranda (15:1)
  - 1 Ariel (9:1) + 1 untargeted Ariel flyby ~6000 km
  - 7 Umbriel (6:1)
  - 4 Titania (3:1)
  - 5 Oberon (2:1)
- In most cases resonances were chosen based on the closest available (and with just 1 s/c rev when possible)
- the resonant flybys were mainly loitering orbits until phasing was favorable to move to next body
- Considering the initial phase free cost to transfer to Oberon was found to be ~633 m/s, The 652 m/s total here is near the minimum possible (for 1 impulse maneuvers)
- The small maneuvers are to account for the fact that there are no exact resonances in a real ephemeris (The body doesn't come exactly back to where it started.)
- Any maneuver at a flyby is performed at the sphere of influence
- The large maneuvers are to target the next moon…



Alternative Tour: Untargeted Close Approaches

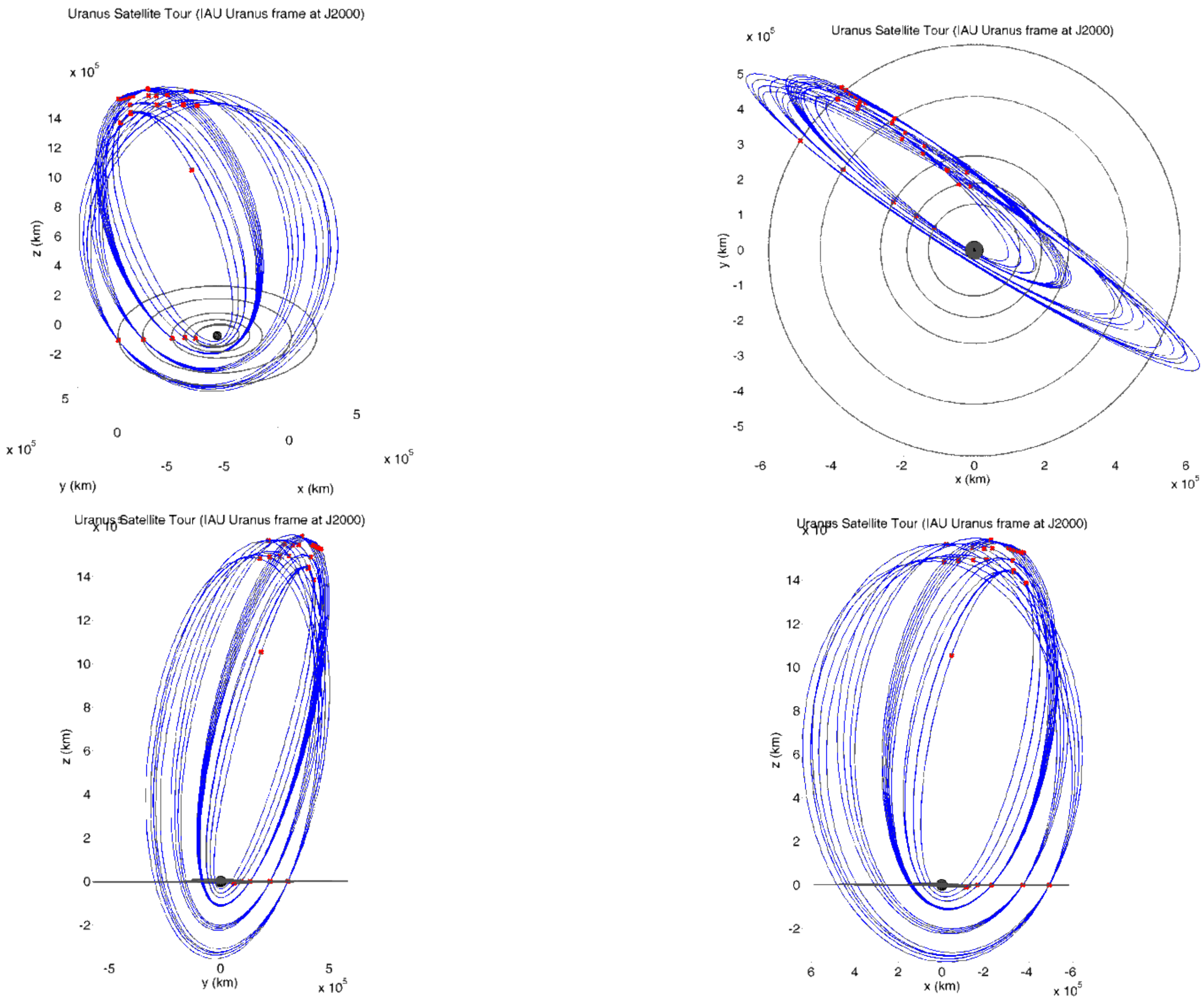
- Values come from integrating the Uranus centric orbit (same patched conic ephemeris model) and checking distances to all moons at each time step.
- Details will change when going to high fidelity design
- Statistically gives an idea of how many untargeted close approaches are expected….
- In high fidelity design we could easily incorporate the ~6000 km flyby of Ariel as a targeted flyby

ith, body, JD, close approach (km)

ith	body	JD	close approach (km)
1	705	2464260.241	81996.073
2	701	2464260.319	81965.829
3	705	2464281.317	28705.931
4	701	2464366.161	58598.917
5	702	2464366.283	89404.230
6	701	2464431.592	5931.562
7	705	2464502.481	50711.945
8	705	2464552.083	86119.500
9	701	2464552.657	77934.406
10	705	2464552.657	80441.603
11	701	2464577.691	86161.249
12	701	2464601.810	54874.956
13	701	2464626.746	85204.023
14	705	2464626.805	89075.336
15	701	2464652.175	99827.481
16	702	2464704.189	84038.897

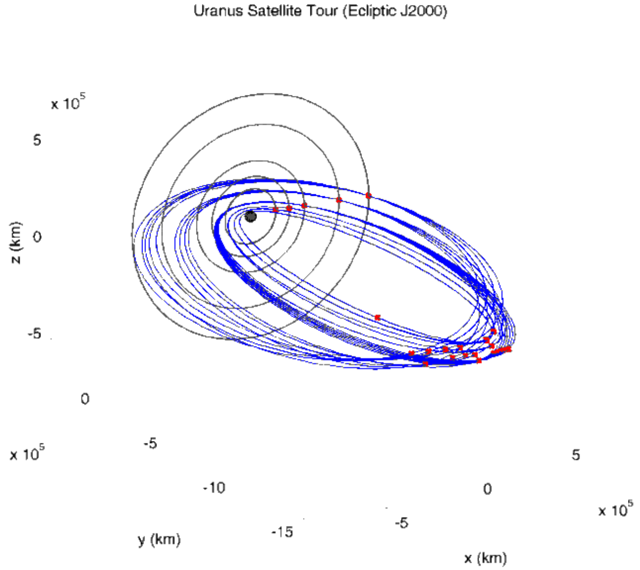
APL

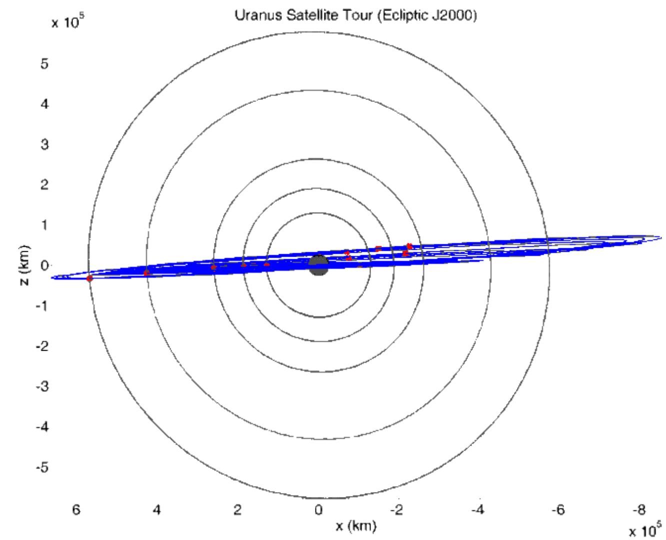
Alternative Tour: Uranus Frame Trajectory
(red dots are locations of Moon Encounters and maneuvers)

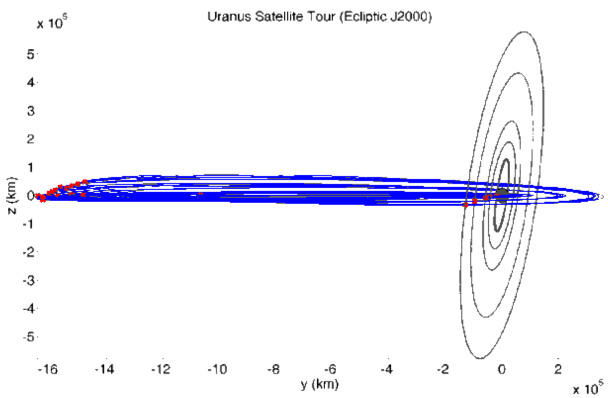


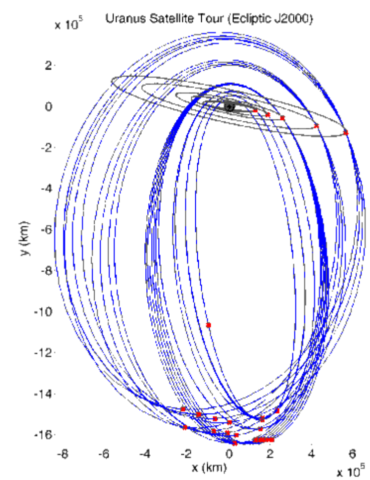
Alternative Tour: Ecliptic Frame Trajectory

(red dots are locations of Moon Encounters and maneuvers)







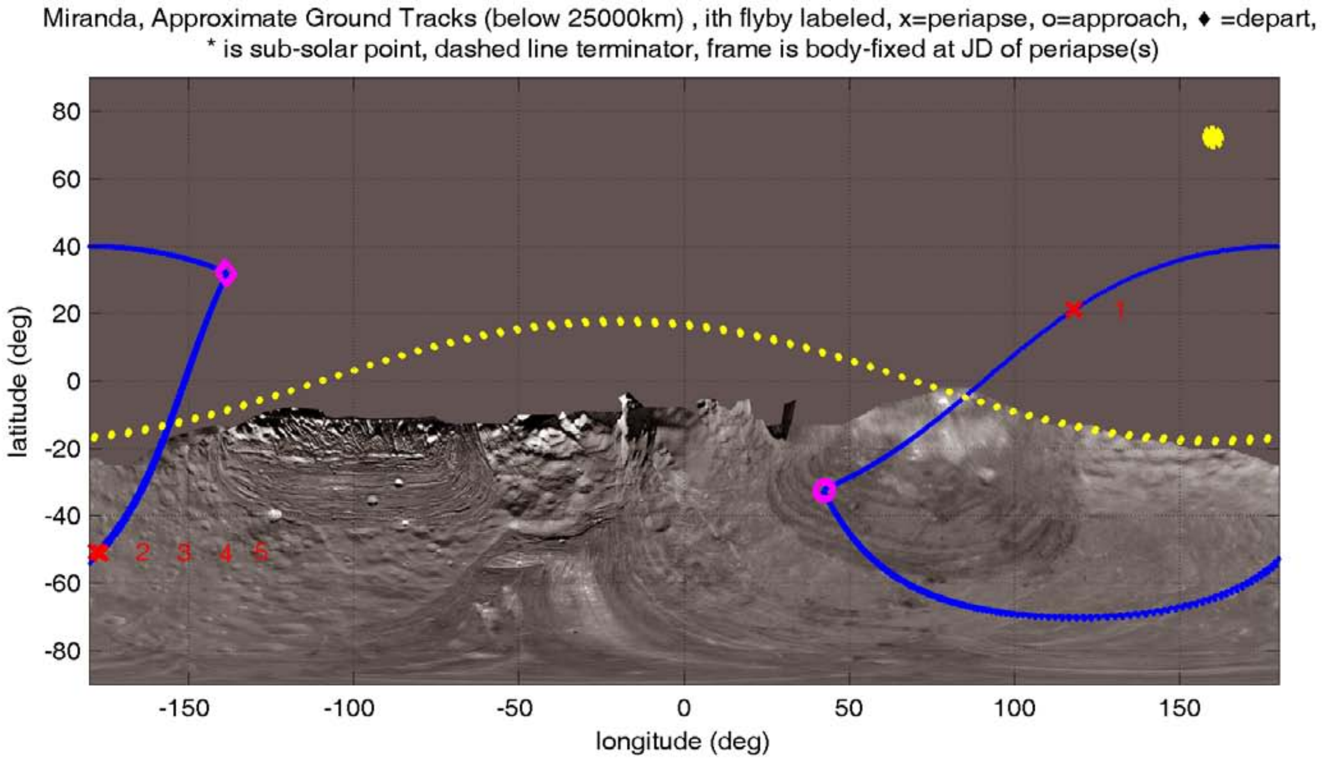


Alternative Tour: Details on Flyby Geometries

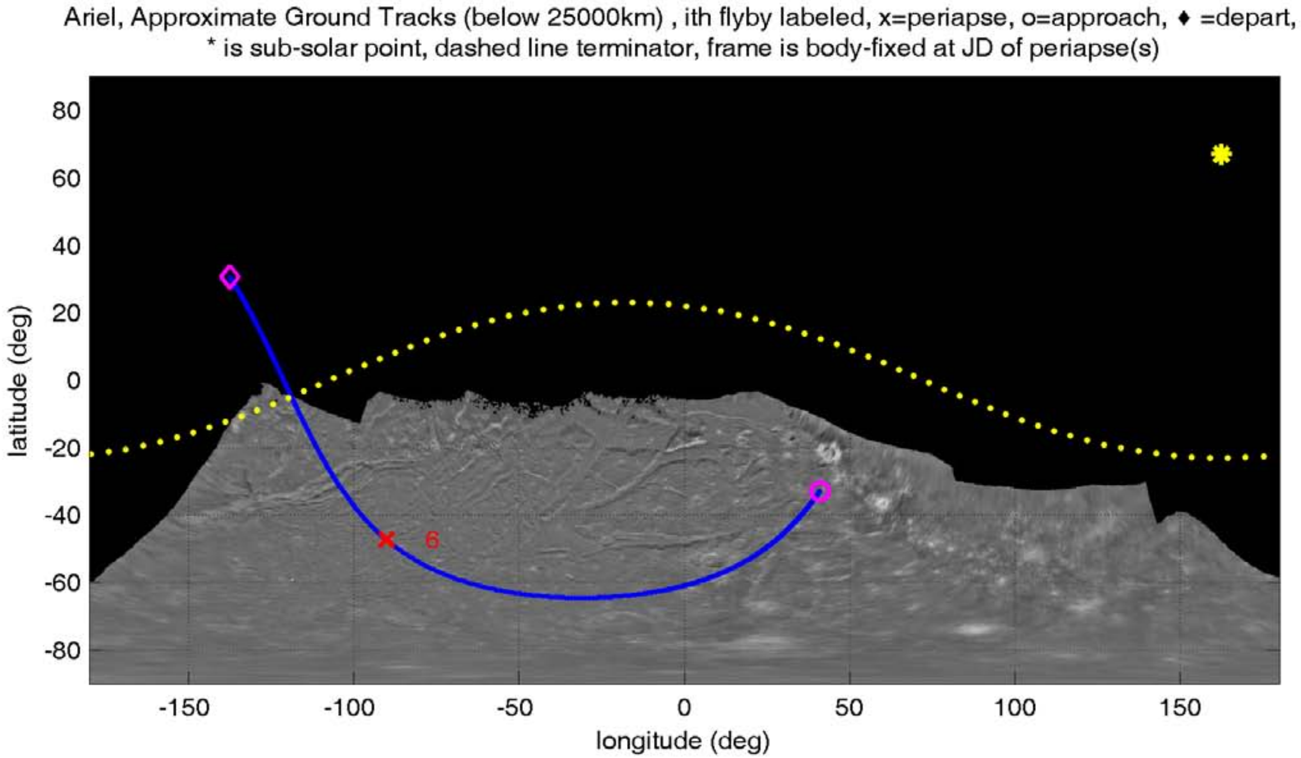
%JD	ifly	body	dv(m/s)	rpalt	vp	rplat	rplon	rpinc	sun_elev_rP	aprSCbodSun_Ang	rtosunRbf	rtosunLONbf	rtosunLATbf
2464302.507	1	705	0.324	50.002	10.884	21.164	117.913	40.019	33.406	128.273	2826354092.392	159.879	72.734
2464323.709	2	705	0.000	50.000	10.877	-50.722	-177.752	69.979	-34.202	129.302	2825806207.057	159.887	72.494
2464344.912	3	705	0.000	50.000	10.871	-50.804	-177.479	70.147	-34.069	129.425	2825258829.458	159.966	72.252
2464366.114	4	705	0.000	50.000	10.866	-51.131	-176.582	70.693	-34.231	129.553	2824711991.335	160.073	72.008
2464387.316	5	705	0.000	50.000	10.861	-50.585	-177.873	69.891	-33.332	129.660	2824165768.059	160.152	71.762
2464408.898	6	701	0.000	50.000	8.812	-47.296	-90.059	115.474	-49.133	132.042	2823578633.959	162.420	67.005
2464478.230	7	702	0.964	50.000	7.335	-21.610	-62.080	139.077	-36.902	132.906	2821770186.949	163.137	66.302
2464503.097	8	702	0.000	50.000	7.338	-48.953	-179.151	68.440	-25.741	132.978	2821133266.748	163.495	66.017
2464527.962	9	702	0.000	50.000	7.343	-48.690	-178.999	68.377	-25.195	133.268	2820497355.311	163.728	65.733
2464552.825	10	702	0.000	50.000	7.340	-48.322	-179.129	68.186	-24.528	133.563	2819862141.112	163.864	65.447
2464577.691	11	702	0.000	50.000	7.336	-47.895	-179.340	67.896	-23.780	133.877	2819227605.258	164.162	65.162
2464602.557	12	702	0.000	50.000	7.341	-47.694	-179.026	67.916	-23.306	134.151	2818594153.737	164.429	64.875
2464627.421	13	702	0.000	50.000	7.342	-47.360	-179.066	67.763	-22.675	134.426	2817961589.249	164.599	64.588
2464653.772	14	703	0.000	50.000	5.578	-36.852	171.430	57.734	-11.164	135.003	2817220151.805	165.963	64.219
2464679.890	15	703	0.000	50.000	5.581	-41.632	-178.818	64.151	-16.214	135.636	2816556874.111	166.202	63.917
2464706.006	16	703	0.000	50.000	5.582	-40.531	-178.564	63.800	-14.833	136.077	2815894524.731	166.362	63.615
2464732.123	17	703	0.000	50.000	5.579	-39.100	-178.927	63.053	-13.066	136.481	2815232749.586	166.552	63.312
2464759.759	18	704	0.000	50.000	4.700	-19.635	162.717	47.557	6.906	135.648	2814468935.967	167.332	63.380
2464786.682	19	704	0.000	50.000	4.700	-32.680	-179.177	59.060	-6.340	136.126	2813787680.602	167.464	63.066
2464813.607	20	704	0.000	50.000	4.696	32.695	0.815	120.917	6.030	135.812	2813106990.686	167.639	62.751
2464840.536	21	704	0.000	50.000	4.693	34.304	0.914	120.255	7.307	135.657	2812427216.349	167.907	62.436
2464867.466	22	704	0.000	50.000	4.694	35.731	0.754	119.782	8.375	135.477	2811748785.492	168.229	62.121

rplat= close approach latitude (deg)
rplon = close approach longitude (deg)
rpalt = close approach altitude (km)
vp= close approach velocity (km/s)
rpinc= inclination of flyby hyperbola in body fixed frame (deg)
aprSCbodSun_Ang= SC-Body-Sun Angle when S/C just reaches body sphere of influence
sun_elev_rP= elevation of the sun at close approach
(rtosunRbf,rtosunLONbf,rtosunLATbf)= Position vector in spherical coordinates (km,deg) of the Sun relative to the Body at time of flyby

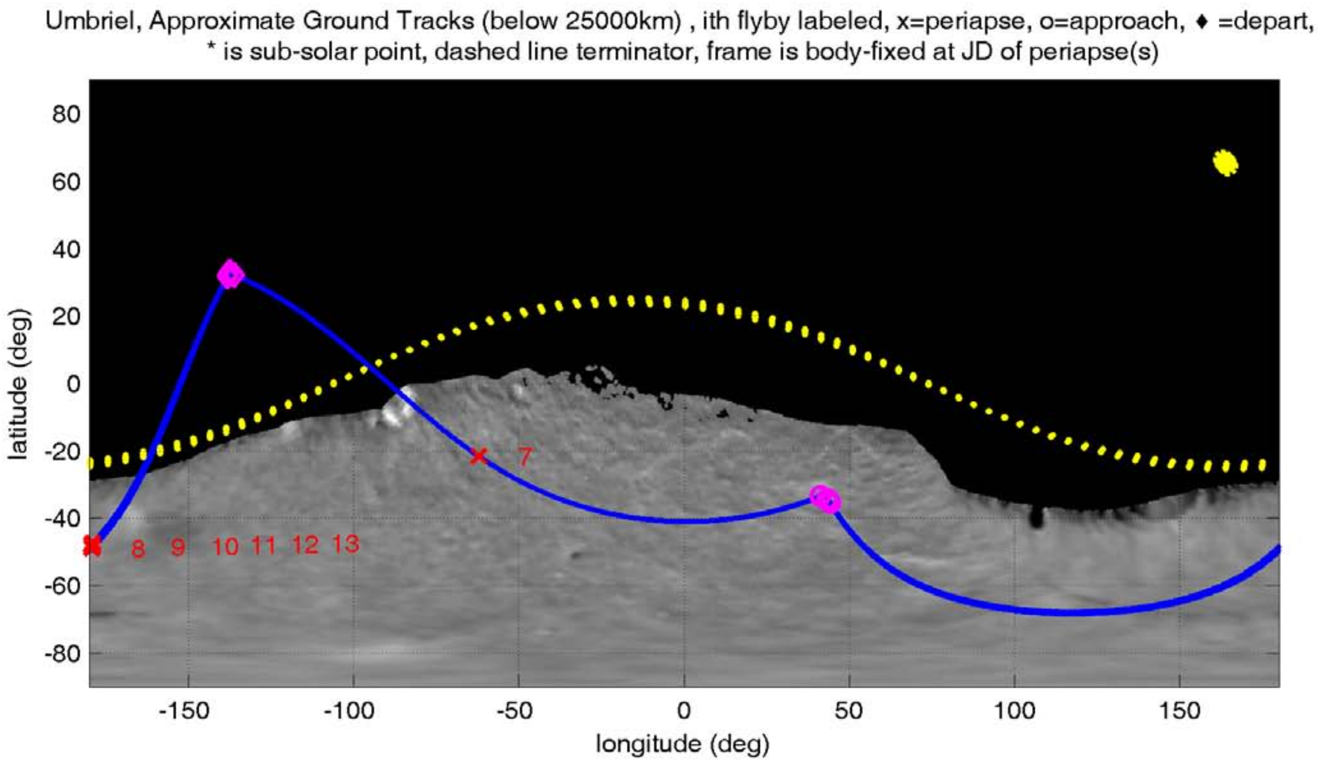
Alternative Tour: Miranda Ground Track



Alternative Tour: Ariel Ground Track

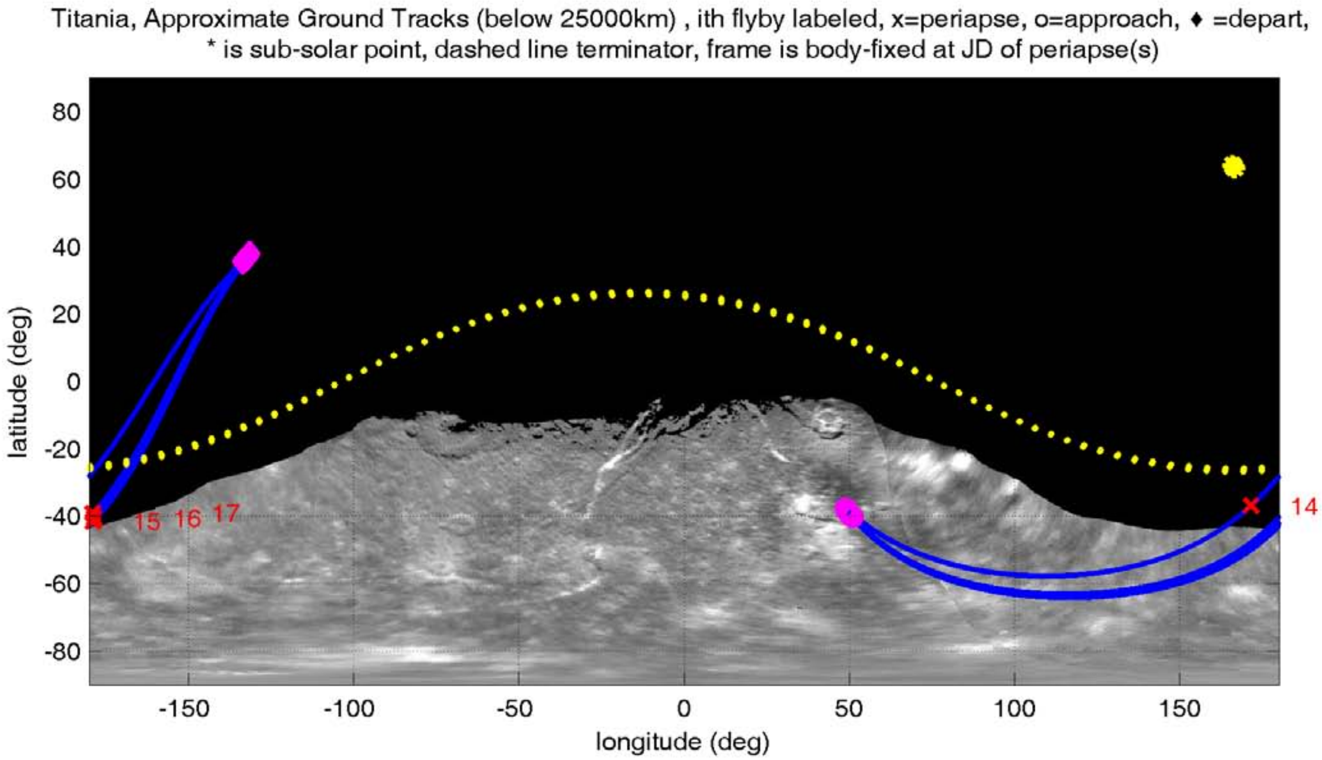


Alternative Tour: Umbriel Ground Track

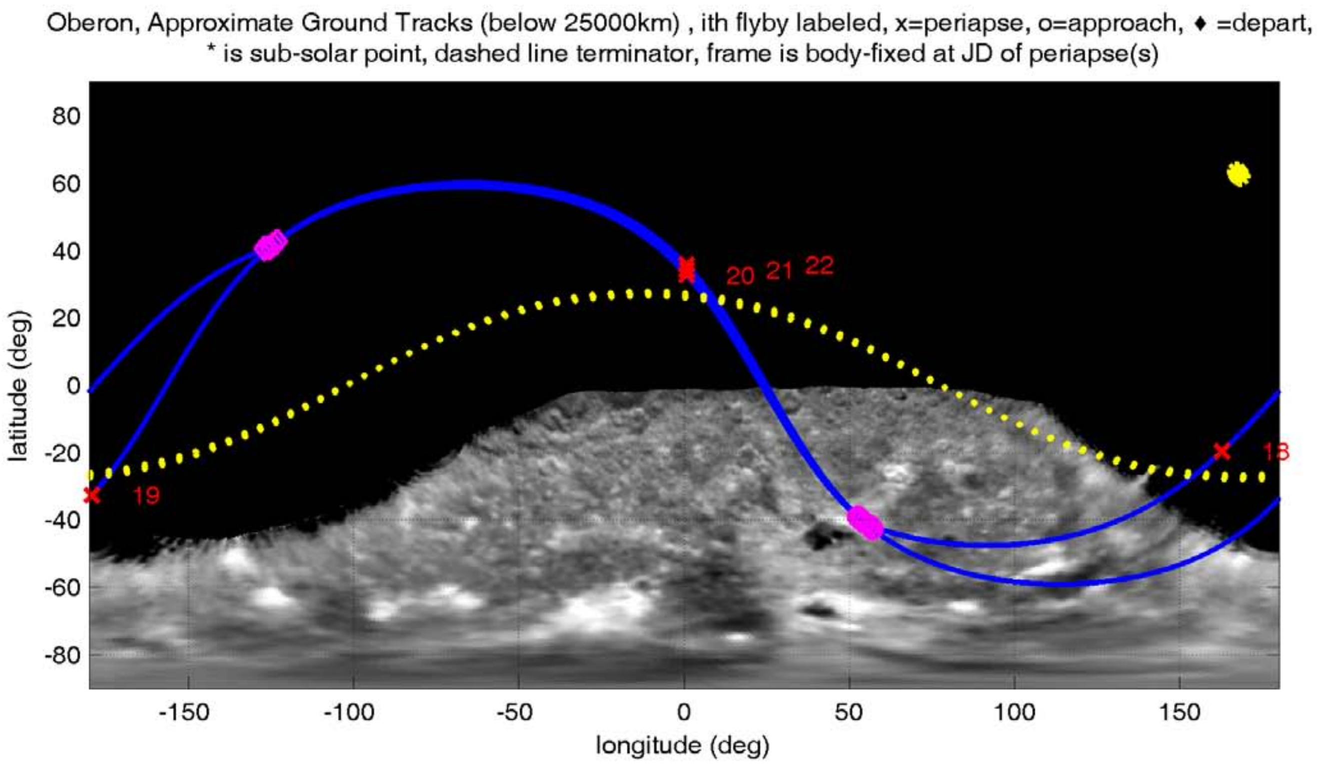


APL

Alternative Tour: Titania Ground Track



Alternative Tour: Oberon Ground Track



Comments on Satellite Tour

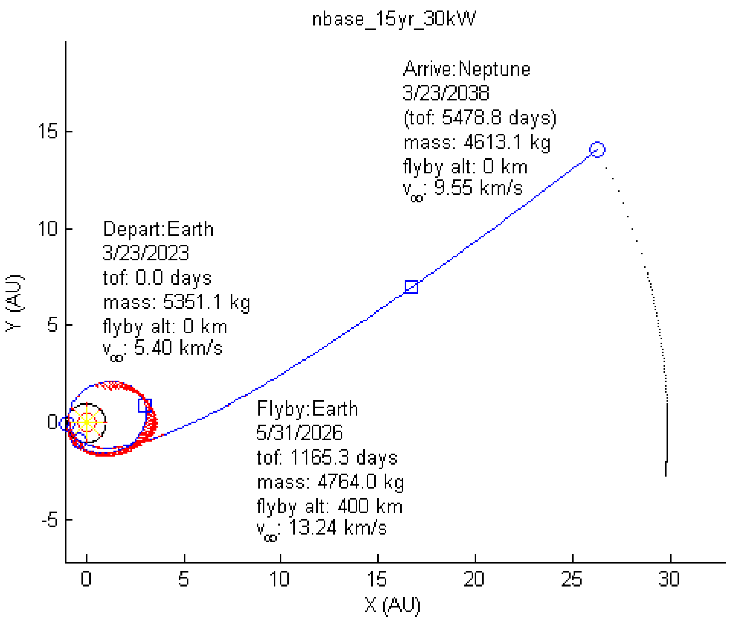
- Emphasize that the tours are not optimized end to end. Each leg is a global minimum for its associated 1 impulse maneuver, or an approximate global min for the two impulse maneuver. It is anticipated that extra maneuvers alongside an end to end optimization may provide some overall delta v savings.
- However, the inclined tour is highly constrained because
  - 1) very high excess velocity limits flyby capabilities and
  - 2) the only potential for moon encounters is at the node crossings of the spacecraft.
- These limitations heavily constrain the design space and therefore the presented solutions are expected to be close to the global minimum for an end to end optimization.
- While non-trivial, J2 effects are not expected to qualitatively change the tour results. Moving to an integrated trajectory instead of patched conics could have delta v penalties (or reductions) on the same order of magnitude as the lack of J2 consideration. An additional 5 m/s/flyby is allocated to accommodate for navigation errors and model fidelity errors (based on numbers from Cassini tour design).
- We could drop some number of the targeted flybys to avoid extra statistical delta-v and simply take what results which would be close but not very close flybys….
- A future study should look at the coupling effect of optimizing the Uranus arrival conditions from the interplanetary trajectory to benefit the moon tour. The moon tour options of the current study was constrained due to the primary science requiring a near polar orbit with very low Uranus attitudes.
- Future studies with moon tours as primary science should initiate with a non-polar orbit and smaller excess velocities to allow for greater maneuverability from flybys.

Delta-V Estimate for Orbiter

Phase	Event	Delta-V (m/s)	Prop Type (1=Hy; 2=bi)	Comment
Cruise	Launch Injection Cleanup	0.0	0	Provided by SEP Stage
	Earth Flyby Targeting	0.0	0	Provided by SEP Stage
	Deep Space Maneuver Deterministic	0.0	0	
	Interplantary Statistical	30.0	1	Cruise post SEP stage seperation
Probe Release	Orbit Deflection	30.0	2	Post Probe release
Orbit Insertion	UOI B-plane targeting	10.0	1	
	Orbital Insertion	1661.0	2	Finite burn
	Clean up	25.0	1	
Science Orbit	Orbit Mantenence	20.0	1	~1m/s per orbit
	Periapsis Reduction	56.0	2	Lower periapsis near end of science orbit phase
Satellite Tour	Miranda Targeting	103.0	2	
	Miranda Statistical	10.0	1	5m/s per flyby (2x)
	Ariel Targeting	89.0	2	
	Ariel Statistical	10.0	1	5m/s per flyby (2x)
	Umbriel Targeting	102.8	2	
	Umbriel Statistical	10.0	1	5m/s per flyby (2x)
	Titania Targeting	183.3	2	
	Titania Statistical	10.0	1	5m/s per flyby (2x)
	Oberon Targeting	140.9	2	
	Oberon Statistical	10.0	1	5m/s per flyby (2x)
Total		**2501.1**		

NEPTUNE STUDY

Neptune – Launch / Arrival Options



There is no significant benefit to using a Star motor with the 551 because of the low C3.

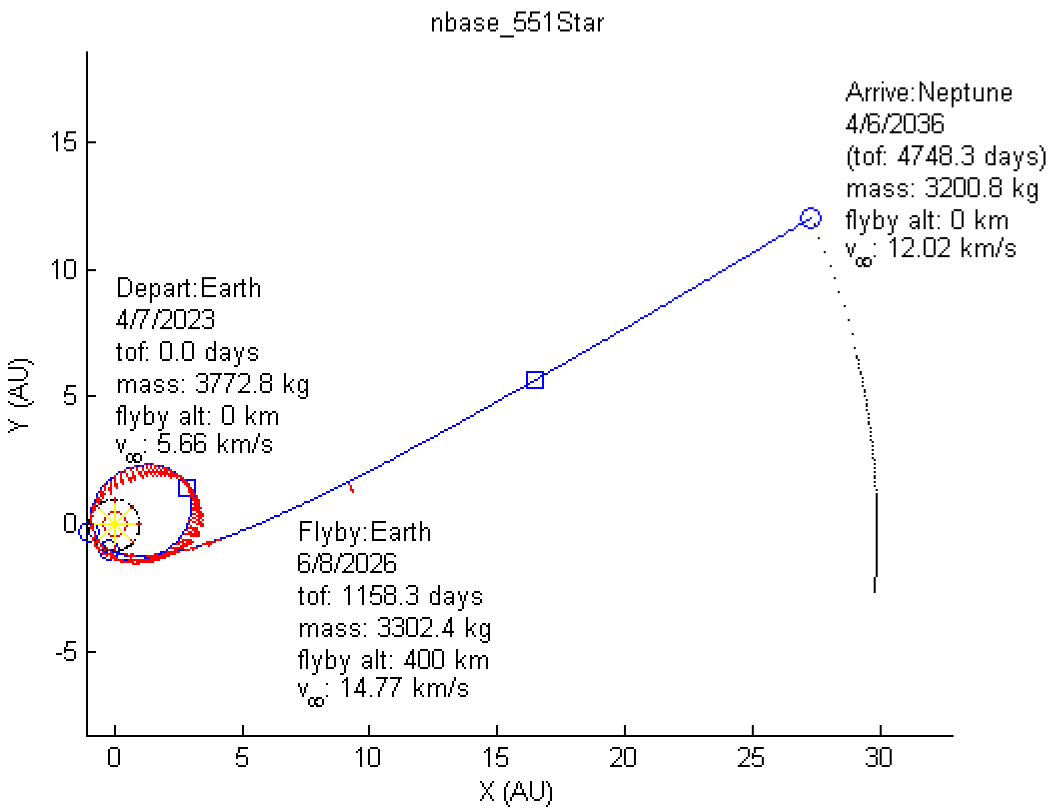
DIVH increases delivered mass ~500kg

Increase in trip time by ~1 yr increases performance ~150kg

Chemical insertion is very challenging, increased trip time is more important

Comment	Launch Wet Mass, kg	SEP Stage Mass, kg	EP Propellant, kg	Arrival V_∞, km/s	UOI ΔV, km/s	Chemical Propellant, kg	Estimated Mass after UOI, kg
Atlas 551	3651	800	577	11.97	Aerocapture	NA	1516
Atlas V - Star	3689	800	542	12.01	Aerocapture	NA	1565
D IVH	4417	800	619	11.98	Aerocapture	NA	1999
D IVH	4417	800	619	11.98	3.62	NA	973
D IVH - 14 year	4640	800	592	10.64	Aerocapture	NA	2165
D IVH - 15 year	4735	800	593	9.53	Aerocapture	NA	2228
D IVH - 15 year, 30 kW	5351	800	738	9.55	Aerocapture	NA	2542
D IVH - 15 year, 30 kW	5351	800	738	9.55	2.57	NA	1716
D IVH - Star -Direct	2908	800	561	10.54	Aerocapture	NA	1032

Baseline Neptune



Using 20 kW and 3 NEXT Thrusters

Trip time and power can be reduced during iteration

Delivers 3200.8 kg, After SEP Stage ~2400kg to Neptune prior to orbit insertion

Uranus Decadal Survey ACE Run Mission Operations

Richard D Reinders

4/29/2010

Ice Giants Decadal Study: Appendix F

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Concept Summary

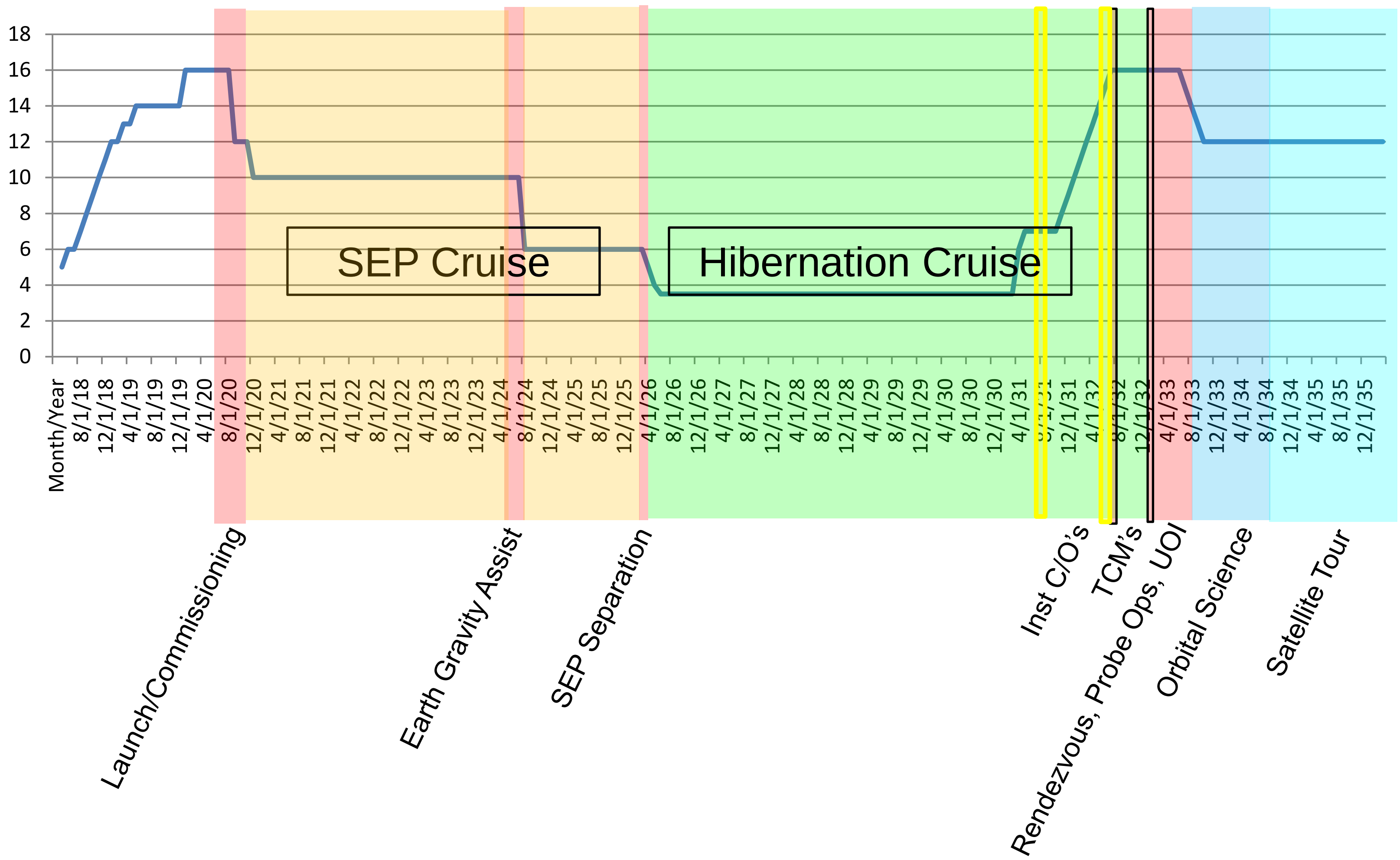
- **Mission segmented into phases – each characterized by specific ops tempo and DSN contact usage**
  - Launch – S/C checkout – Instrument checkout and calibration
  - SEP Operations
  - Earth Gravity Assist
  - SEP stage Separation
  - Hibernation Operations
  - Instrument Checkouts
  - Deterministic TCM's
  - Uranus Rendezvous
    - Probe Release
    - Probe data retrieval
    - Orbit Insertion
  - Orbital Science
  - Satellite Tour

Operations Staffing Summary

(SEI staff only)

Draft

Note: below does not include science ops, navigation, S/C engr support, or mission design.



Requirements and Assumptions

Ground Operations:

- 5 week S/C checkout after launch. Initial checkout of G & C done on ASRG power; delay SA deployment.
- 12 week instrument checkout after S/C commissioning.
- Keep staffing high through Earth GA building and testing sequences to be used for subsequent critical ops.
- SEP requires low level continuous caretaking of propulsion and navigation operations. Size operations team to be less than MESSENGER after EGA.
- Hibernation Cruise allows for still smaller team size.
- Assume NO cruise science.
- 1 year effort during non busy cruise to prepare for orbital operations . Begin ramp up prior to first pre-arrival instrument c/o.
- Uranus rendezvous will be highly stressful requiring many high fidelity rehearsals
- Legacy APL planetary ground system will work fine for mission.

Requirements and Assumptions

Instrument Operations:

- Instruments will be calibrated pre-launch then re-calibrated as necessary during the 12 week transition cruise period
- Instrument sequences will be created pre-launch, retained and tweaked in the two years before Uranus arrival.
- Instruments will not be powered up during cruise phases – including Earth Gravity Assist – except for two one month periods approaching Uranus
- During the science phases, power limitations will preclude simultaneous high data rate instrument operation and data downlink.

Requirements and Assumptions

Spacecraft Operations:

- Due to angular disparities some critical events, SEP separation, probe release, and most of UOI burn, will not be viewable to Earth in real-time.
- Uranus rendezvous will be very high intensity period:
    - probe release at UOI minus 29 days
    - TCM at UOI minus 28 days
    - Probe data collection and relay
    - Final probe data collect at UOI burn minus 1 hour. S/C will be maneuvered to downlink status data then maneuver to burn attitude
    - UOI will be 4000 second “turn while burn”
    - 3000 seconds of burn will be occulted
- Primary Uranus science phase will require one 8-hour contact per day, on average, for each of the orbits (20 orbits)
- Secondary satellite science phase will have similar contact density for the flybys (22) with lower density for phasing orbits
- Variable Downlink data rate during science phases

Draft

DSN Usage by Mission Phase (pt 1)

Mission Phase	Duration	DSN Usage	Operations
Launch and Observatory checkout (8/3/2020)	5 weeks	Continuous: 1 week 2 8-hr contacts per day: 1 week 1 8-hr contact per day: 3 weeks	3-Axis; ASRG initial power. S/C subsystem and maneuvering checkout. Begin instrument checkout
Cruise transition and complete inst cals.	12 weeks	3 8-hr contacts per week	Complete instrument checkout; close covers. Commence SEP ops.
Inner SEP Cruise	170 weeks	2 8-hr contacts per week	Contacts needed for thrusting commands and orbit determination. Cycle thrusters off to maneuver for contact if using MGA (> 1.75 Au)
Earth GA (6/2/2024 CA)	12 weeks	Ramp up to continuous for week of CA then back down. DDORs pre and post CA	42 day SEP coast around CA. TCMS pre and post expected
Outer SEP Cruise	84 weeks	2 8-hr contacts per week	Same as inner SEP Cruise
SEP separation (2/3/2026)	2 weeks	1 8-hr contact per day for 1 week then 2 8-hr contacts per day for 1 week	Separation while in 3-Axis; enter spin mode at some time later. <u>SEP separation may not be viewable, ensure that contact is established immediately before and then after</u>.
Hibernation Cruise	324 weeks	Weekly 1 hour beacon; monthly 8 hr TLM	TLM for S/C checkout and orbit determination
Instrument Checkouts (2)	4 weeks each	5 8-hr contacts per week	2 4-week periods at Arrival – 2 yrs and -1 yr for instrument turn-ons and recalibrations
Pre Arrival TCMs (2)	8 weeks each	Ramp up to 16 hr per day for week of TCM then back down. DDORs to support	TCM#1 to be done at probe release – 12 months; TCM#2 at release -6 months.

DSN Usage by Mission Phase (pt 2)

Mission Phase	Duration	DSN Usage	Operations
Uranus Arrival - Probe Release Prep (UOI-29 days)	12 weeks	Ramp up to continuous through probe release; DDORs to support.	Spin up S/C for probe release, then 3-Axis for remainder of rendezvous. TCM #3 at probe release + 1 day. Continuous DSN support through TCM. Release and TCM probably not viewed in RT.
Uranus Arrival - Probe Science Operations	4 weeks	Reduce to daily 8-hr contacts for following 3 weeks. Increase usage to 16-hr contacts for 1 week; DDORs to support	Periodically maneuver S/C to collect probe data every 5 days for next 4 weeks.
Uranus Arrival (6/28/2033) - Orbit Insertion	1 week	Continuous through UOI for 1 week.	Final probe data will be collected 1 hour before UOI burn initiation. S/C will be maneuvered to broadcast probe sequence completion before being maneuvered back to burn attitude. Burn will be "turn-while-burn." Most of burn period will be occulted by planet. Comms will be re-established immediately as possible.
Uranus Science (20 21-day orbits)	60 weeks	Daily 8-hr contacts	Orbit adjustment maneuvers at apoapsis. Variable DL data rate
Satellite Tour	94 weeks	Daily 8-hr contacts	Orbit adjustment maneuvers at apoapsis. Variable DL data rate.

Backup

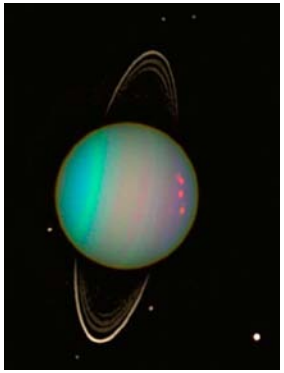
Draft

Draft

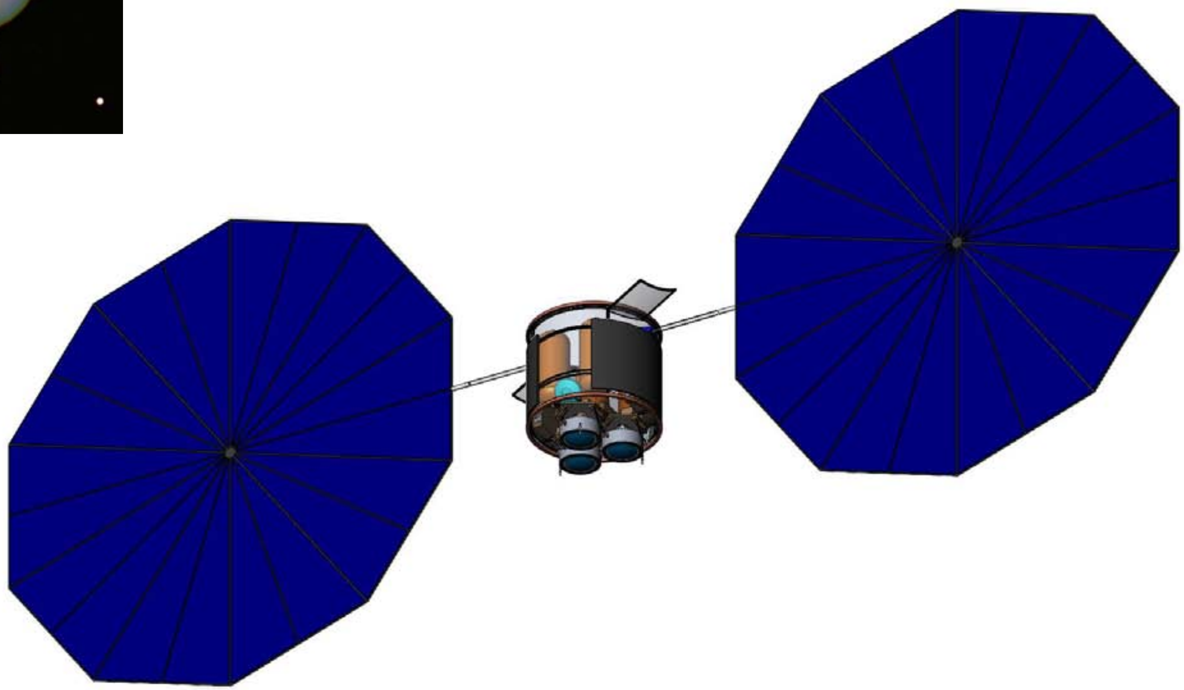
CONOPS

- **Only for operations and Comm.**

Down link Information	Mission Phase 1	Mission Phase 2	Mission Phase …
Number of Contacts per Week			
Number of Weeks for Mission Phase, weeks			
Downlink Frequency Band, GHz			
Telemetry Data Rate(s), kbps			
Transmitting Antenna Type(s) and Gain(s), DBi			
Transmitter peak power, Watts			
Downlink Receiving Antenna Gain, DBi			
Transmitting Power Amplifier Output, Watts			
Total Daily Data Volume, (MB/day)			
Uplink Information			
Number of Uplinks per Day			
Uplink Frequency Band, GHz			
Telecommand Data Rate, kbps			
Receiving Antenna Type(s) and Gain(s), DBi			



Uranus SEP Stage : Preliminary Design



Design by COMPASS

April 20, 2010

Ice Giants Decadal Study: Appendix G

COMPASS Team Roster

•Lead	Steve Oleson	steven.r.oleson@nasa.gov	216-789-2026
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•GN&C	Mike Martini	michael.c.martini@nasa.gov	216-433-3423
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•Mechanical Systems	John Gyekenyesi	john.z.gyekenyesi@nasa.gov	216-433-3639
•Thermal	Tony Colozza	anthony.j.colozza@nasa.gov	216-433-5293
•Power	James Fincannon	james.fincannon@nasa.gov	216-433-5405
•Power	Kristen Bury	kristen.m.bury@nasa.gov	216-433-3150
•C&DH	Glenn L. Williams	glenn.l.williams@nasa.gov	216-433-2389
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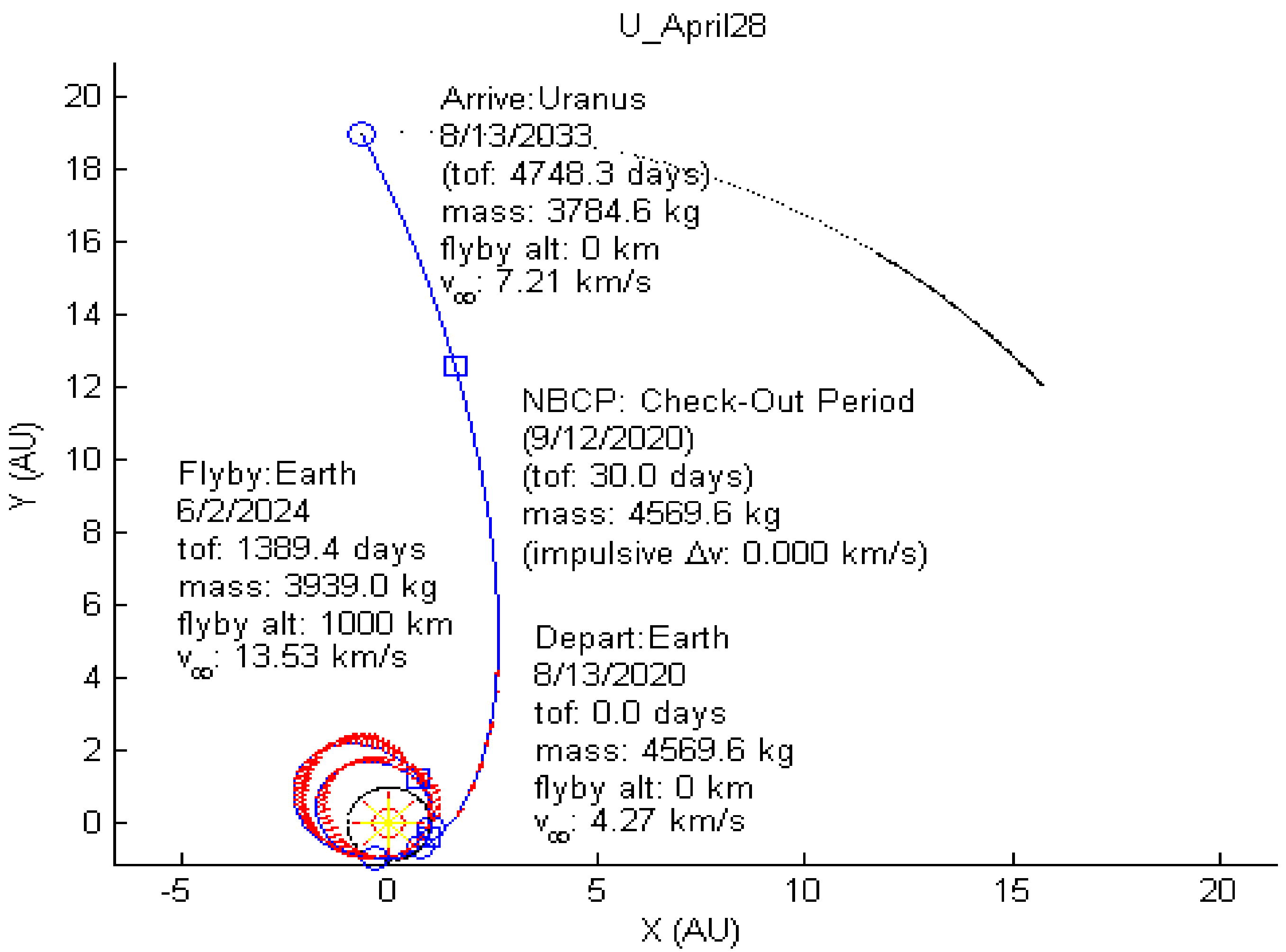
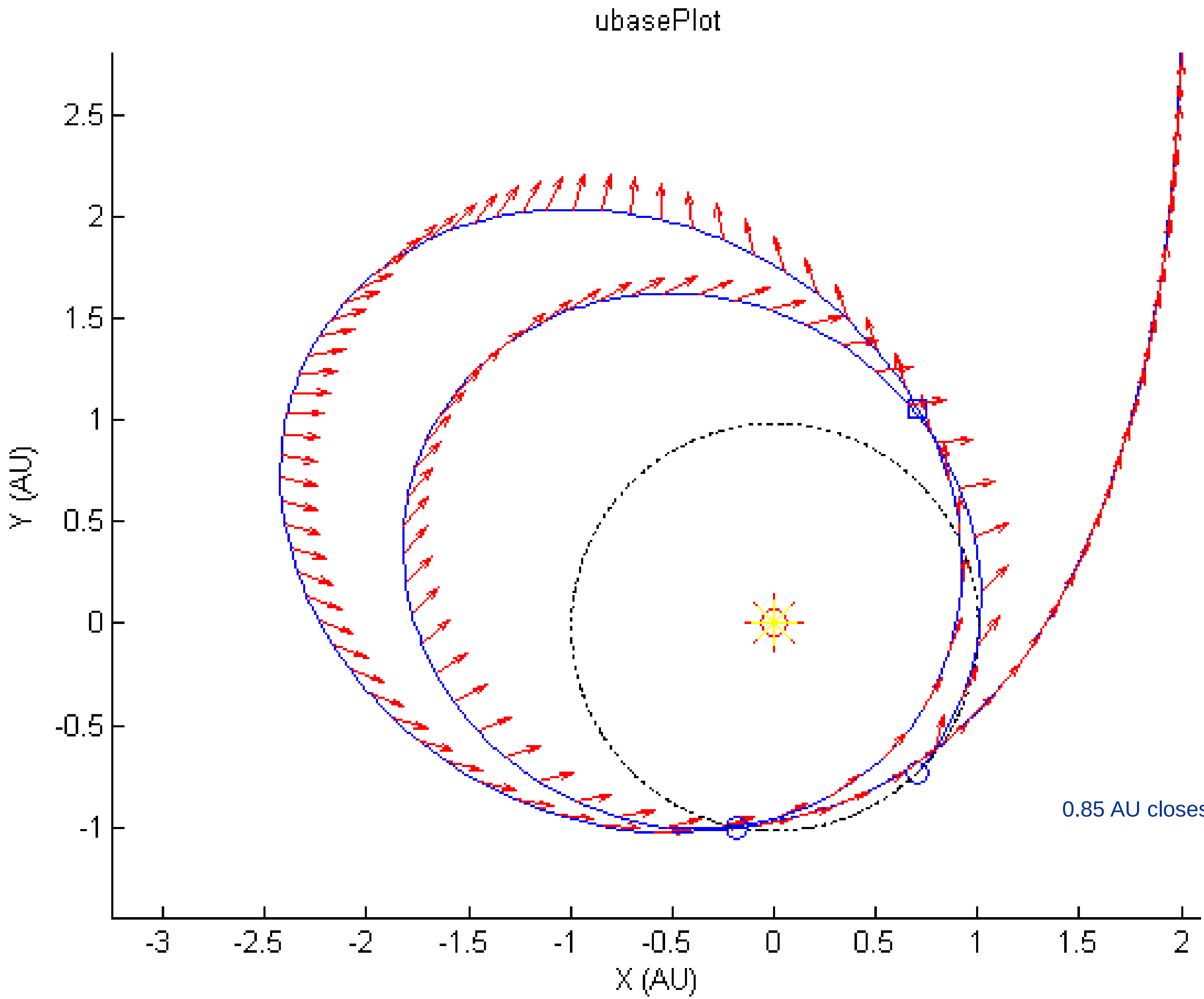
•Configuration	Tom Packard	thomas.w.packard@nasa.gov	216-433-3525
•Cost	Jon Drexler	jonathan.a.drexler@nasa.gov	216-433-6416
•Risk	Anita Tenteris	anita.d.tenteris@nasa.gov	216-433-2803

Mission Background Uranus SEP Stage



- 13 year EEU Trajectory combined with SEP thrusting allows a Vinf=7.2 km/s at Uranus
- 1000 km earth flyby
- Trajectory needs refinement

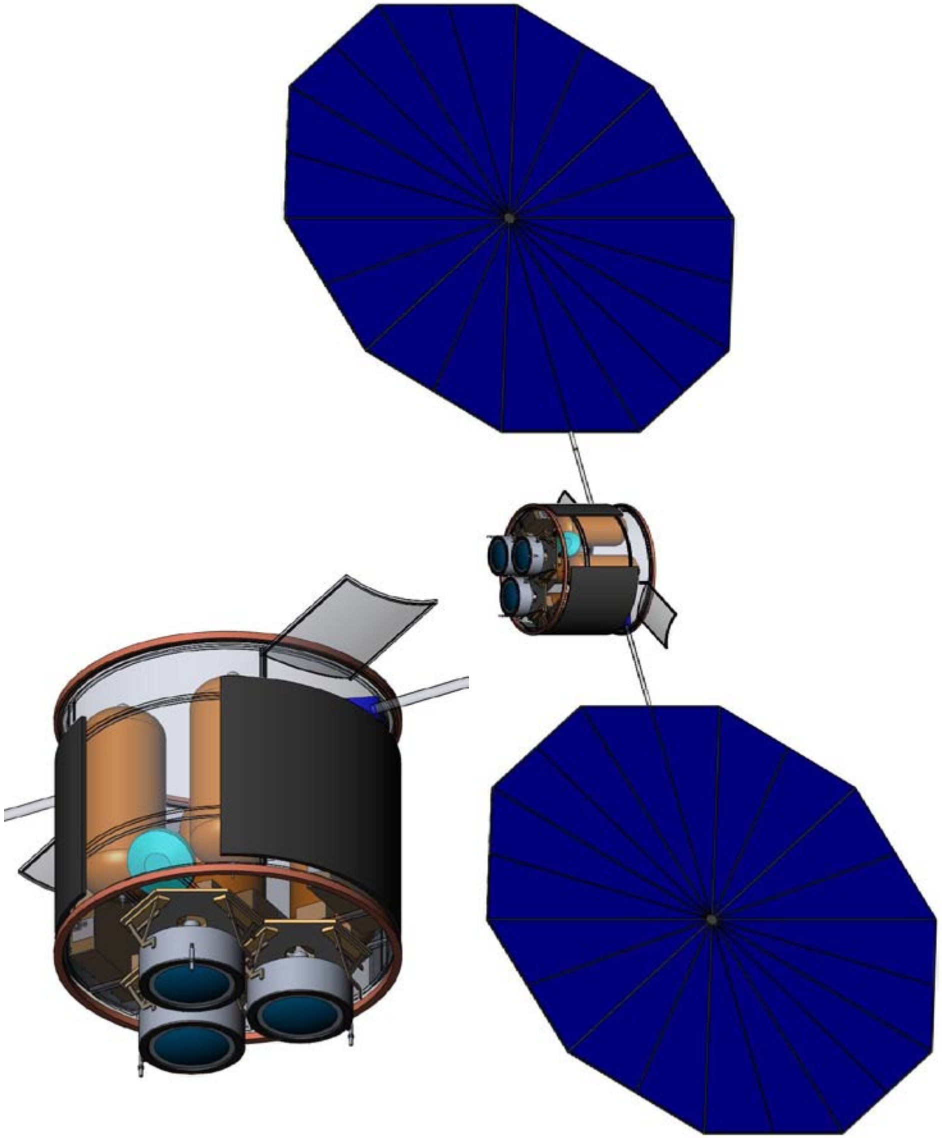
SEP Trajectory



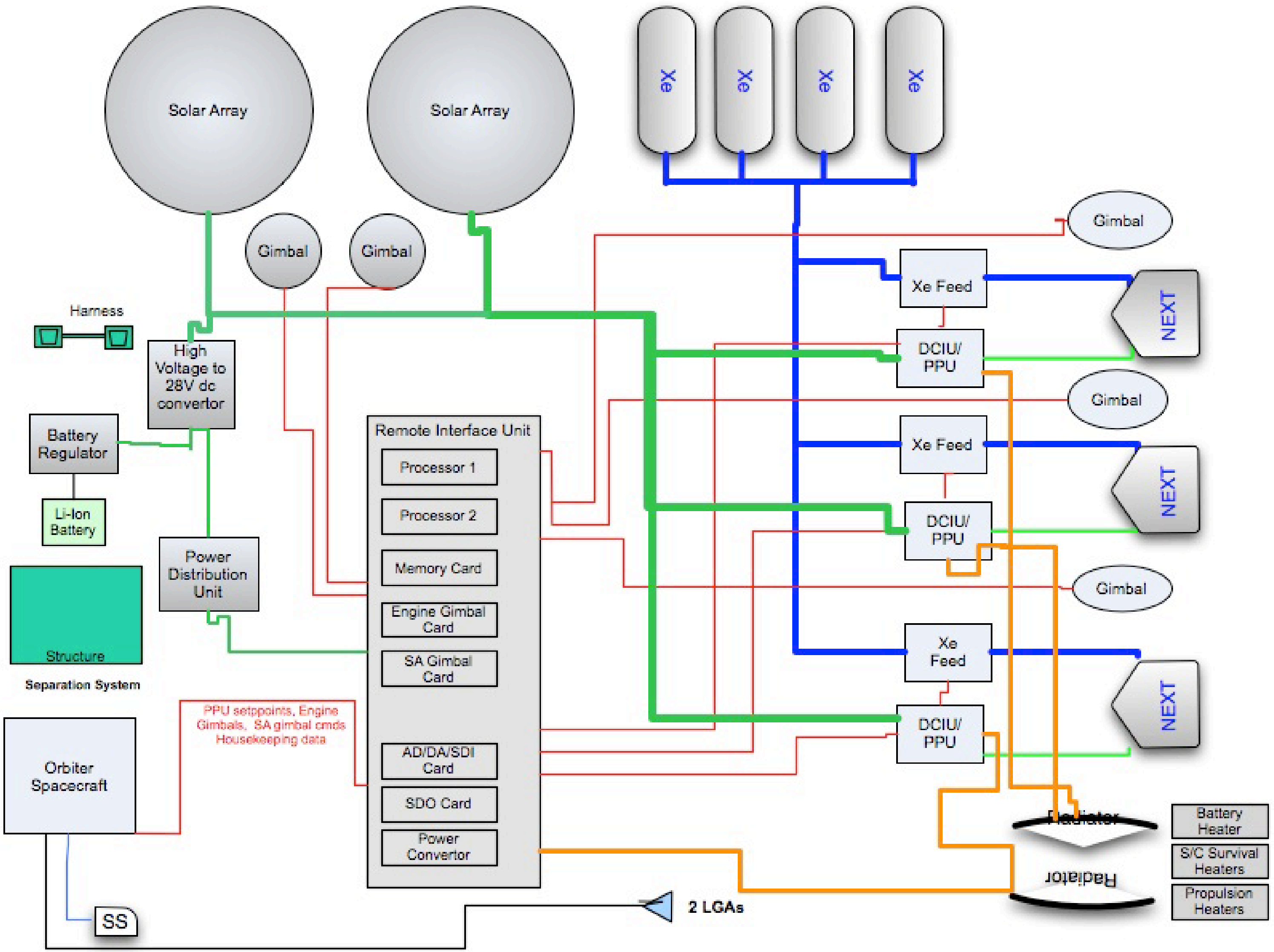
0.85 AU closest distance to Sun

SEP Stage Characteristics

- **Propulsion**: 2+1 gimbaled 15 kW NEXT ion propulsion system with ~750 kg xenon propellant (2 thruster operation, 2-axis [+/- 15°] gimbals can provide yaw/pitch/roll and offset perturbations)
  - Need to keep engines away from sun viewing (avoid overtemp)
  - Most of trajectory has nearly tangential thrusting?
- **ACS**: Use Ion engines to steer vehicle during thrusting
  - Augment with Hydrazine RCS
  - Wheels added (could be dropped)
- **Power**: Two Ultraflex Solar Arrays (total power at 1AU 16.5 kW), Single axis array drives, Li-Ion batteries for launch and contingency ops
- **Structure**: Al-Li thrust tube, supports Uranus Lander during launch
- **Thermal**: Heat pipes to 'north and south' radiator panels with louvers, MLI and heaters xenon tank
- **C&DH**: Digital Control Interface Unit (DCUI) controls power and propulsion system from orbiter commands
- **Communications**: Use Uranus Science Spacecraft (USS) transponders – add 2 LGA antennas to work with USS
- Single Fault Tolerant



SEP Stage Block Diagram



Top Level Uranus SEP Stage Characteristics

COMPASS study: Uranus SEP Stage Study Date 4/28/10

GLIDE container: *Uranus_SEP_Stage: Uranus_case2*

Spacecraft Master Equipment List Rack-up (Mass) - SEP Stage

WBS	Main Subsystems	Basic Mass (kg)	Growth (kg)	COMPASS S/C Design Total Mass (kg)	Aggregate Growth (%)
06	Uranus SEP Stage	1606	131	1737	
06.1	SEP Stage	1606	131	1737	
06.1.1	Science Payload	0	0	0	TBD
06.1.2	Avionics	16	3	20	19%
06.1.3	Communications and Tracking	2	1	3	39%
06.1.4	Guidance, Navigation and Control	2	0	3	3%
06.1.5	Electrical Power Subsystem	200	65	265	32%
06.1.6	Thermal Control (Non-Propellant)	87	16	103	18%
06.1.7	Structures and Mechanisms	130	23	152	18%
06.1.8	Propulsion and Propellant Management	315	24	339	8%
06.1.9	Propellant	852		852	
	Estimated Spacecraft Dry Mass	753	107	884	14%
	Estimated Spacecraft Wet Mass	1606	131	1737	

System LeveL Growth Calculations				Total Growth
Dry Mass Desired System Level Growth	753	324	1077	43%
Additional Growth (carried at system level)		217		29%
Total Wet Mass with Growth	1606	324	1929	

Inert Mass Calculations

SEP Stage Main Mass Calculations	Basic Mass (kg)	Growth (kg)	Total Mass (kg)	
Total Wet Mass	1606	131	1737	
Total Dry Mass	753	107	884	
Dry Mass Desired System Level Growth	753	324	1077	43%
Additional Growth (carried at system level)		217		29%
Total Useable Propellant	785		785	
Total Trapped Propellants, Margin, pressurant	68		68	
Total Inert Mass with Growth	753	324	1144	

Architecture Details	Mass (kg)	units
Launch Vehicle	atlas 551	
Energy, C3	52.0	km2/s2
ELV performance (pre-margin)	4568	kg
ELV Margin	0	kg
SEP Stage Total Wet Mass	1929	kg
Uranus Payload available mass	2638	kg
Total Spacecraft Wet Mass	1929	kg
Thruster name	NEXT	
Thruster Efficiency	0.00	%
Specific Impulse	0	s
Number of Thrusters Operating	2	
Propellant Details		
Main Propellant Details (EP)		
Mass, Propellant Total	852	kg
Mass, Propellant Useable	785	kg
Mass, Prop Nav. & Traj. Margin	39	kg
Mass, Prop Residuals + Misc.	28	kg
RCS/ACS propellant details		
RCS/ACS Used Prop	0	kg
RCS/ACS margin	*N/A*	kg
RCS/ACS Residuals	0	kg
RCS Total Loaded Pressurant (in HW)	0	kg
Total RCS/ACS propellant	0	kg
Spacecraft Totals		
SEP Stage Totals		
SEP Stage Wet mass	1929	kg
SEP Stage Dry mass	1077	kg
SEP Stage Inert mass	1144	kg
Total SEP Stage inert mass	1144	kg

- Bottoms up MGA growth is 14% of the Basic mass.
- Desire 43% growth, carry additional 192 kg at system level (29% of basic mass)
- Available Mass for Uranus Science Spacecraft and Probe = 2366 kg

National Aeronautics and Space Administration

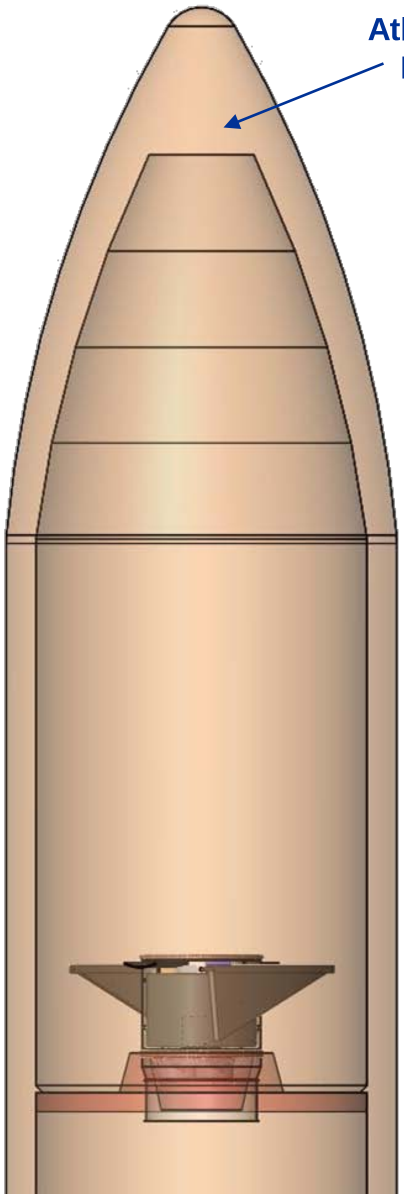
Uranus SEP Stage and Propellant Details

- Delivered Payload Mass: 2638 kg
- Total delta V for mission: 7.45 km/s

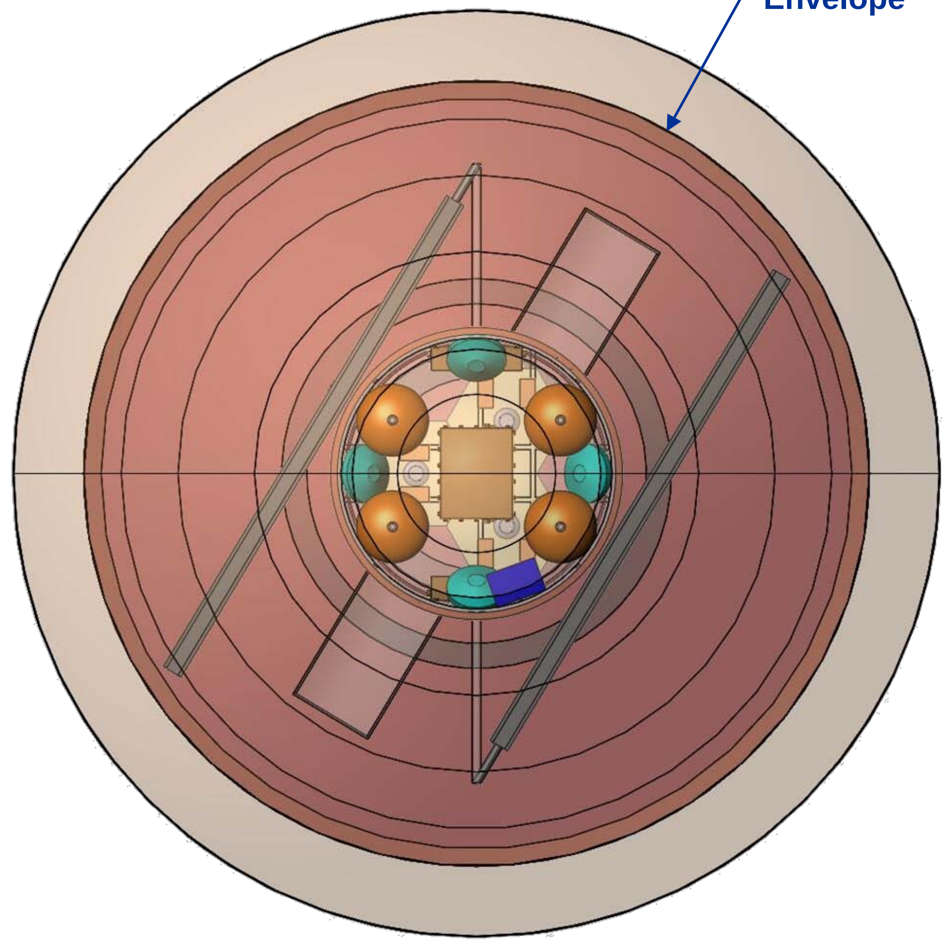
A/ 1 c0 h i t e c t u r e M D a s e s	t u (a n kii	
Launch Vehicle	atlas	55
iES n t e r R g y a, c C 3 k - u p	5 (2 M 0	k a m s
ELV performance	4 (5 p 6 r 8	k g m
ELV Margin	0	kg
SEP Stage Tota l	1W 9 e 2 t 9	kMga
Uranus Payload	2 6 3 8	kgla
Total Spacecraft	1W 9 e 2 t 9	Mga
Thruster name	NEX	T
cThruster Efficie	n c0 y. 0 0	%
Specific Impulse	0	s
tNumber of Thrus	t2 e r s	O p
Propellant Detail s		
gMean Propellant Detai		
Mass, Propella	n t 8 5 2 a	lkg
Mass, Propella	n t 7 8 s 5	a lkg e
Mass, Prop Na	v. &3 9T r	a kj .g
Mass, Prop Res	i d u2 a8 l s	+kg
RCS / ACS propellant d		
RCS / ACS Used	P r 0 p	kg
RCS / ACS marg	i n N / A	kg
RCS / ACS Resid	u a 0 s	kg
RCS Total Load	e d 0 P r e	sks gu
Total RCS / ACS	p r 0 p e	l lka gn
Spacecraft Tota	l s	
SEP Stage To	t a l s	
SEP Stage We	t m 1 a 9 s 2 s	9 kg
SEP Stage Dry	m 1a 0s 7 s	7 kg
SEP Stage Ine	r t 1 m 1 a 4 s	4 kg
a n t		
Total SEP Sta	g e1 1 n4 e	4 r k t g

	Parameter	value	units
Propellant	Mass, Xenon Total	852.44	kg
	Mass, Xenon Useable	784.94	kg
	Mass, Xenon Nav. And Traj. Margin	39.25	kg
	Mass, Xenon Residuals	28.26	kg
	Mass, Xenon Nav. And Traj. Margin percent	5.00%	%
	Mass, Xenon Residuals percent	3.60%	%
Thrusters	Thruster name	NEXT	name
	Thruster Efficiency		%
	Specific Impulse		s
	Quantity, Number of Thrusters Operating	2	
	Duty Cycle	0.9	
Power	Power, BOL Solar Array	20	kW
	Power, Housekeeping	0.335	kW
Trip times & Dates	Time, Transfer to Uranus	4748.25	days
	Date, Launch	8/13/20	
	Date, Earth flyby	6/2/24	
	Date, Uranus Arrival	8/13/33	
ELV assumptions	Launch Vehicle	atlas 551	name
	Energy, C3	51.9841	km2/s2
	ELV performance (pre-margin)	4567.73	kg
	Earth flyby altitude	1000	km
Misc.	Total mission Delta V	7.45	km/s
	Target	Uranus	
	SPK-ID		
Capture Data	Vinf, arrival	7.21	

Atlas V 5-m Medium Payload Fairing



4.58 m Diameter Static Payload Envelope



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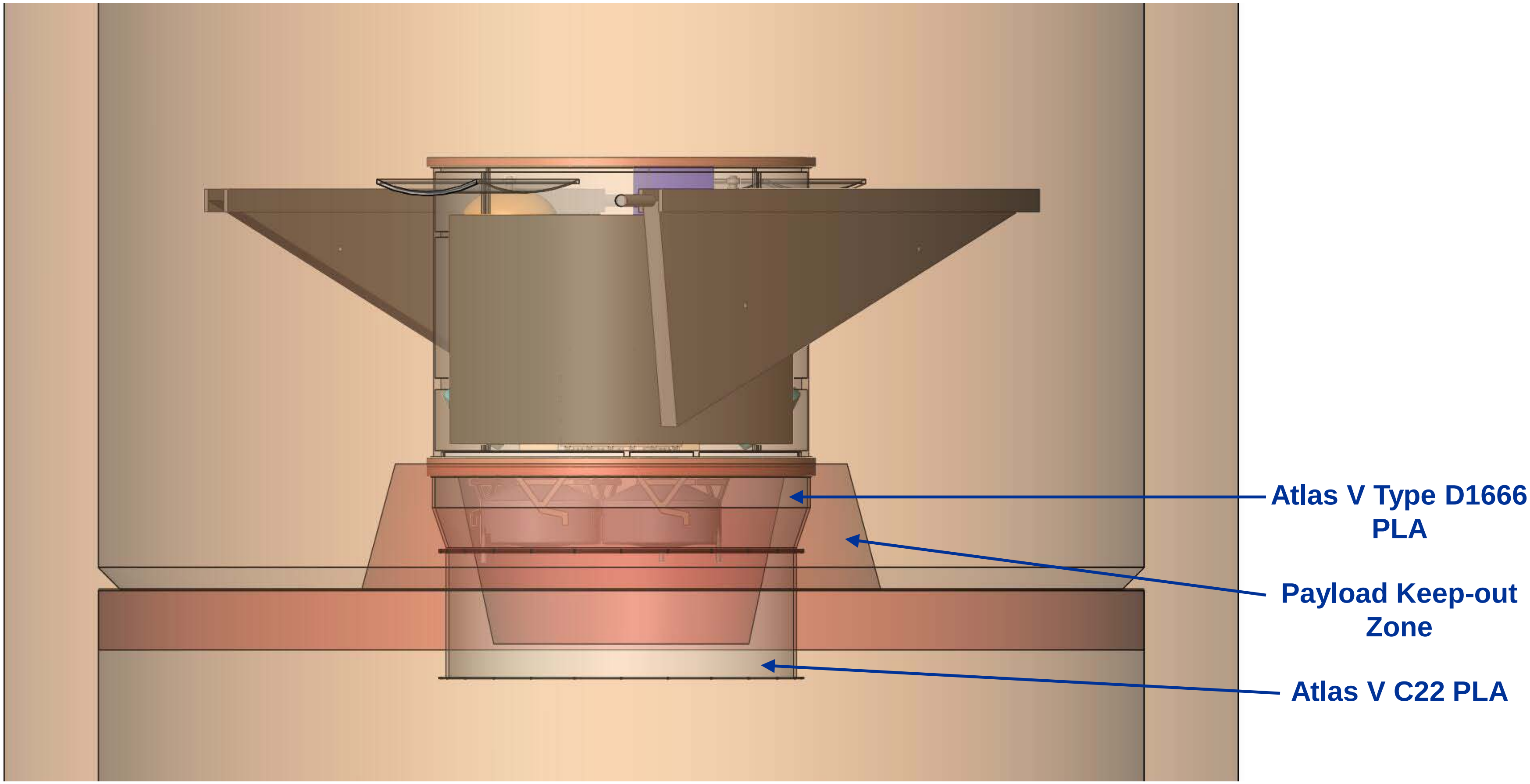
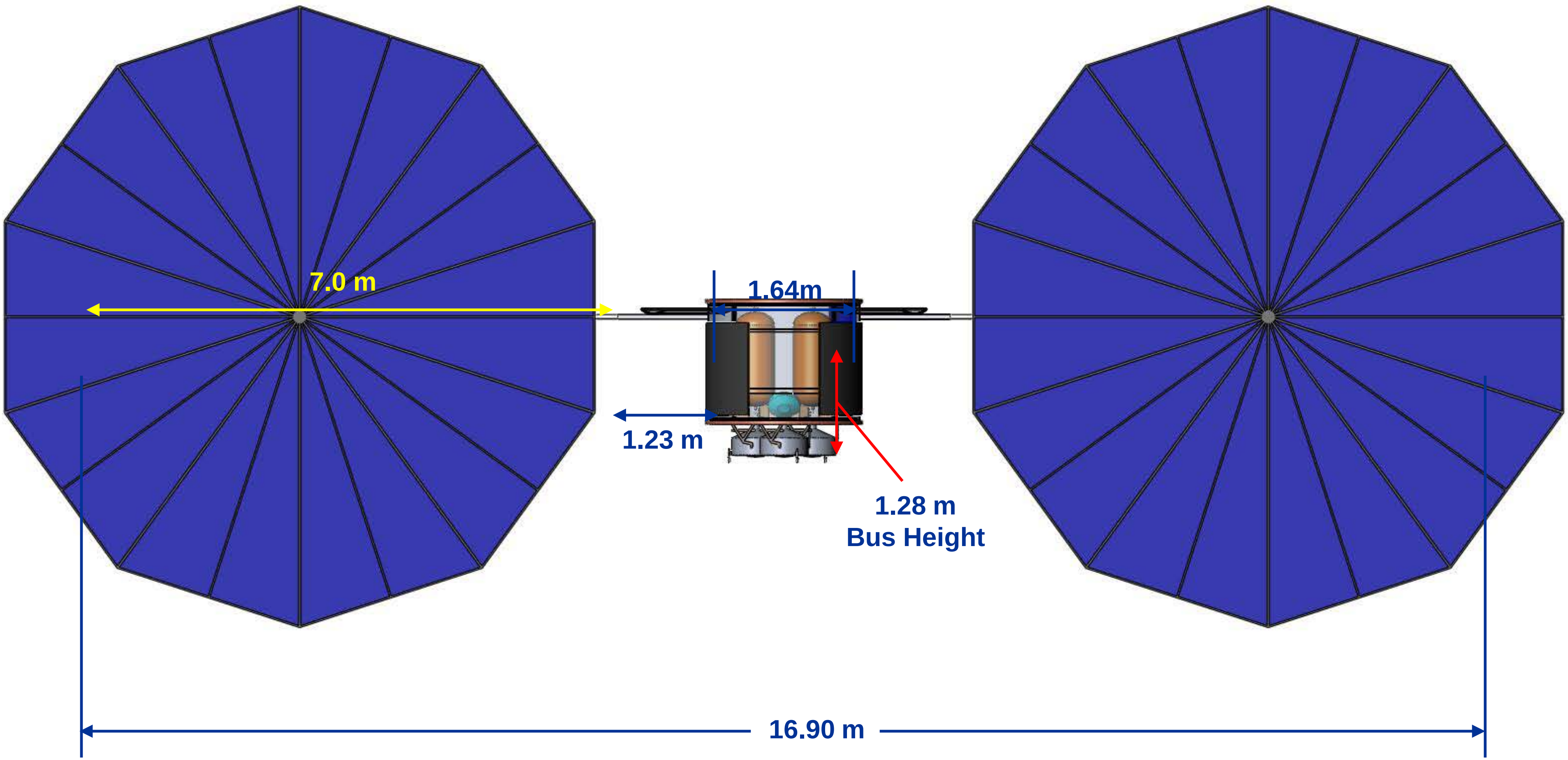
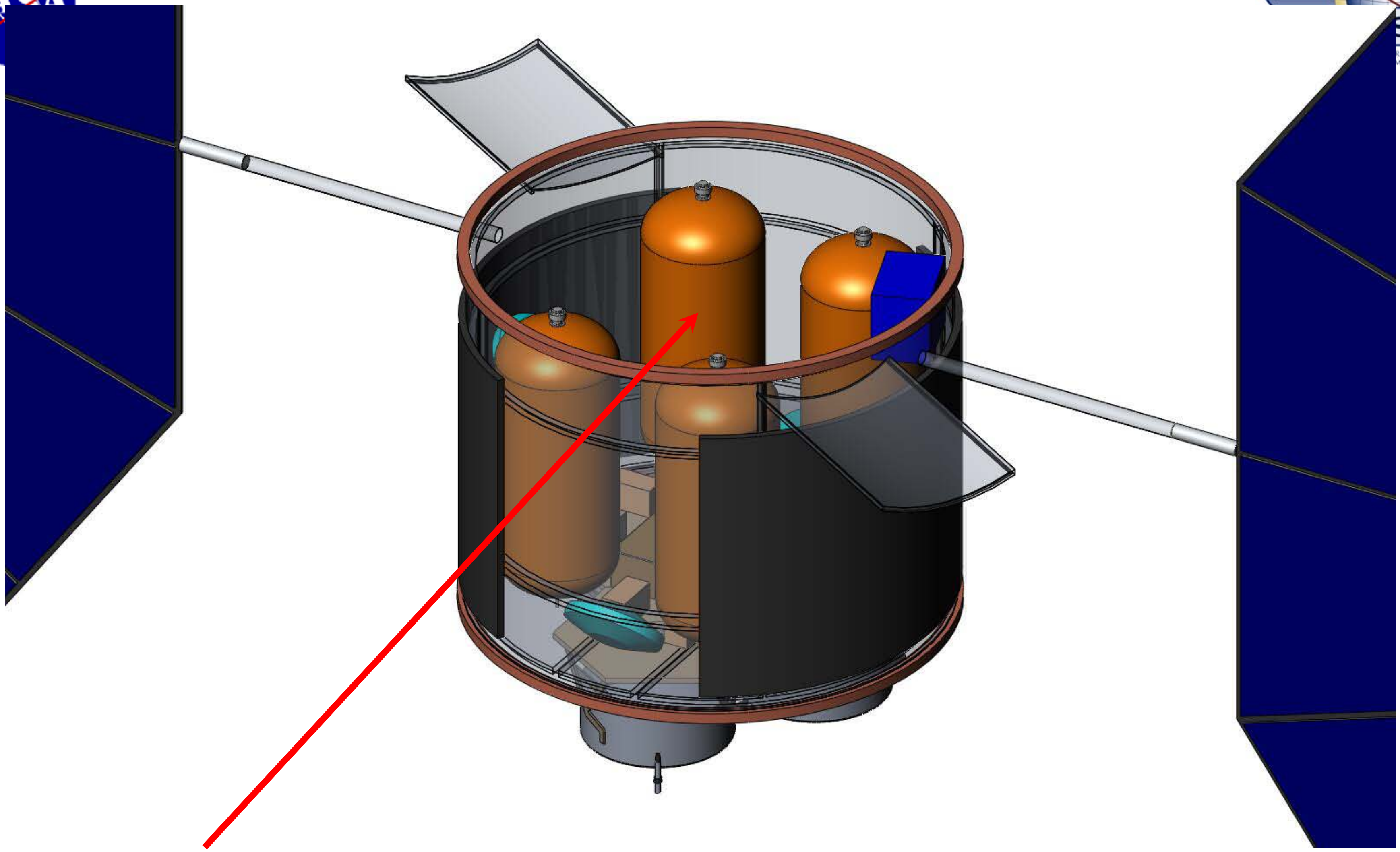




•NEXT Engine Thruster Plumes will not interact with solar arrays

•Preliminary analysis shows communications through thruster plume not an issue.



•May be room to place Uranus Probe into SEP stage to reduce total stack height.

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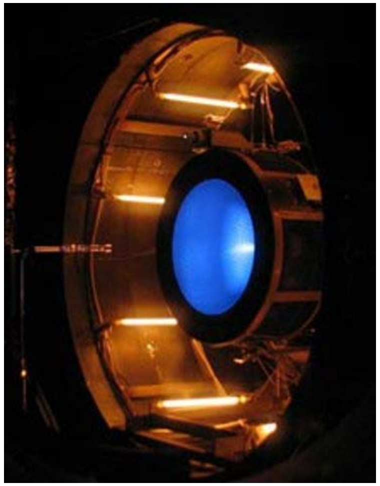
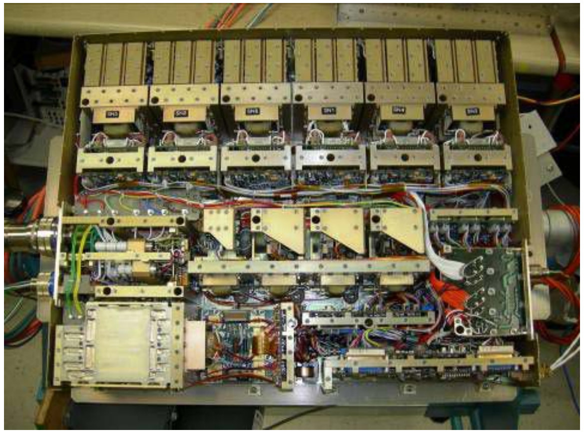
NASA's Evolutionary Xenon Thruster (NEXT)

- 40 cm Diameter
- Xenon Propellant
- 7 kW Electrical Power
- Demonstrated 69% Efficiency
- Specific Impulse > 4050 seconds
- Independent Modular Power Processing Unit (PPU) w/ Integrated Digital Controller
- Extensive Testing at NASA GRC
- Currently processed >400 kg in ground test
- Beam Voltage of 1800V
- Dedicated Propellant Feed System
- 2-Axis Gimbal

NEXT Engine



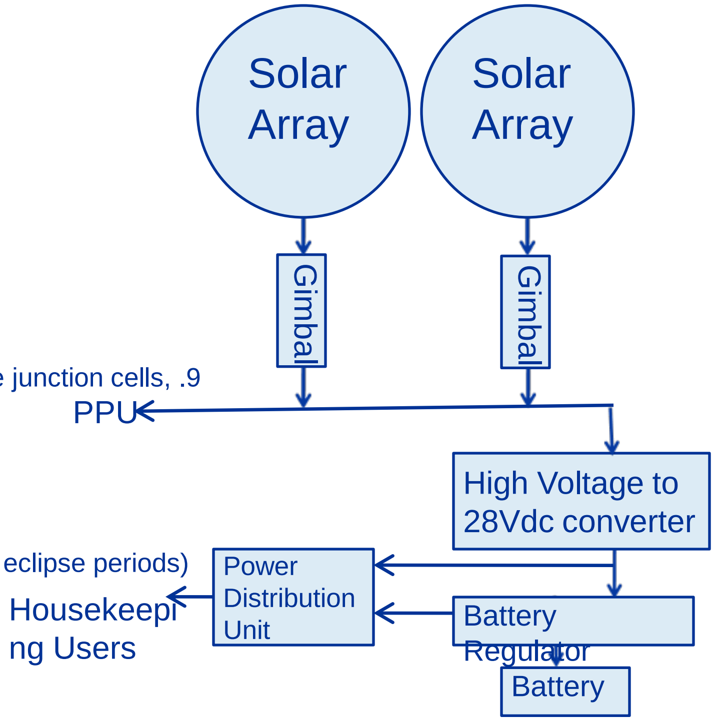
NEXT Engine Vacuum Testing



NEXT Modular Power Processing Unit

Uranus SEP Stage Power System

- Power Requirements
  - Effective sizing: 22,000 W end of life power level at 1 AU
- Power Assumptions
  - One axis gimbal tracking
  - 3.5 hour launch/deployment energy (320 W)
  - Advanced technology for some power components when appropriate
  - Internal redundancy for electronics units and extra battery
  - Peak power regulation
- Power Design
  - Two Ultraflex solar arrays, 7.0 m diameter each
    - 3 mil coverglass, BOL 33% efficiency, EOL 30% efficiency, advanced IMM triple junction cells, .9 kg/m^2
  - Two solar array interface structures
    - Tie downs, yoke, boom, and other ancillary structures for array support
  - Two 1-axis 0.5-g gimbals (drive motors)
  - Advanced lithium ion batteries (required for launch-to-deploy, SEP jettison ops, minor eclipse periods)
    - 200 W-hr/kg, 90% DOD
  - One advanced electronics box
    - Battery electronics – sized for housekeeping power only (130 W/kg)
    - High voltage to 28 Vdc converter – sized for housekeeping power (600 W/kg)
    - Power distribution unit for housekeeping power only– 130 W/kg
  - Harnesses
- Analytical Methods
  - Spreadsheet power system sizing tool
- Power Risk Inputs
  - Large Ultraflex, advanced solar cells and advanced converter may increase the cost and schedule risk
    - Without Orion application, need to validate large Ultraflex
    - Production limits of advanced, IMM solar cells are difficult to predict
- Recommendation
  - Use smaller solar array diameter and traditional cells



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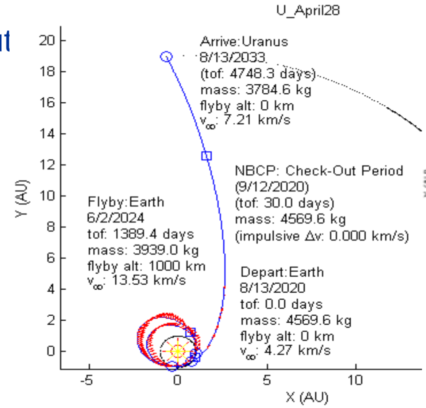
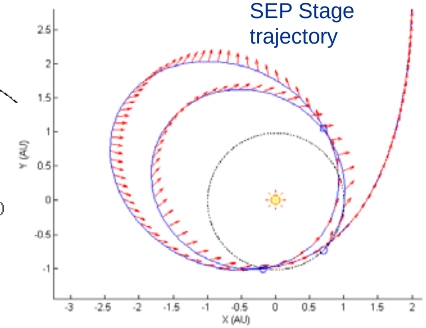
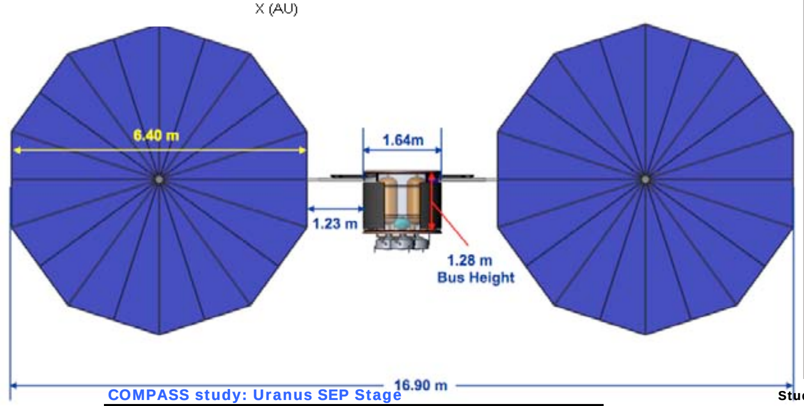
Uranus SEP Stage Summary

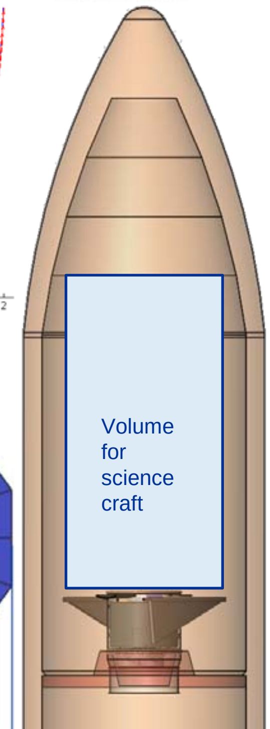
- **Overview**: 20 kW, NEXT Ion propulsion Stage to put a 2638 kg Uranus Science Spacecraft (USS) and Probe on a trajectory to Uranus with a 13 yr transit time.
- **Mission:** 2022 launch, 2026 Earth flyby, 2035 Uranus Arrival
- **Launcher:** Atlas 551 with medium fairing
- **SEP Stage relies on science spacecraft for ultimate control, communications, and ACS**
- **Propulsion**: 2+1 gimbaled 20 kW NEXT ion propulsion system with ~800 kg xenon propellant (2 thruster operation, 2-axis [+/- 15°] gimbals can provide yaw/pitch/roll and solar pressure offset perturbations or dump wheels)
- **ACS**: Use Ion engines to steer vehicle during thrusting
  - Augment with Hydrazine RCS (on USS)
  - Wheels added (could be removed)
- **Power**: Two Ultraflex Solar Arrays (total power at 1AU 22 kW EOL), Single axis array drives, Li-Ion batteries for launch and contingency ops
- **Structure**: Simple 1.6m Al-Li thrust tube, supports Uranus Lander during launch
- **Thermal**: Variable conductance heat pipes to 'north and south' radiator panels, MLI and heaters xenon tank
- **C&DH**: Remote Interface Unit (RIU) controls power and propulsion system from USS commands
- **Communications**: Use USS transponders - add 4 LGA antennas to work with USS if view-ability an issue
- Single Fault Tolerant



Closeup of SEP Stage trajectory







COMPASS study: Uranus SEP Stage

Study Date 4/28/10

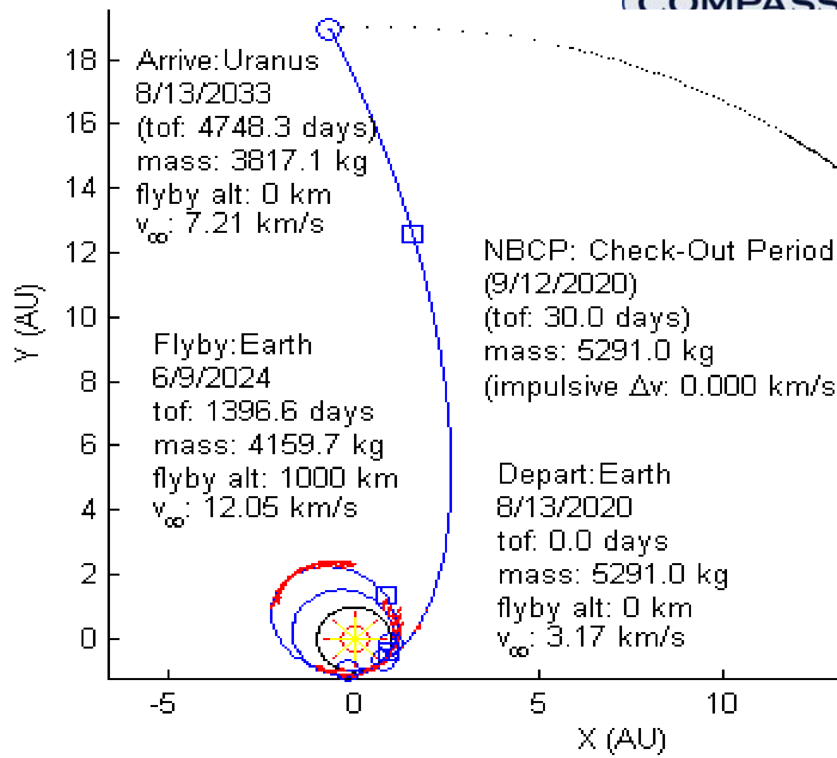
GLIDE container: *Uranus_SEP_Stage: Uranus_case2*

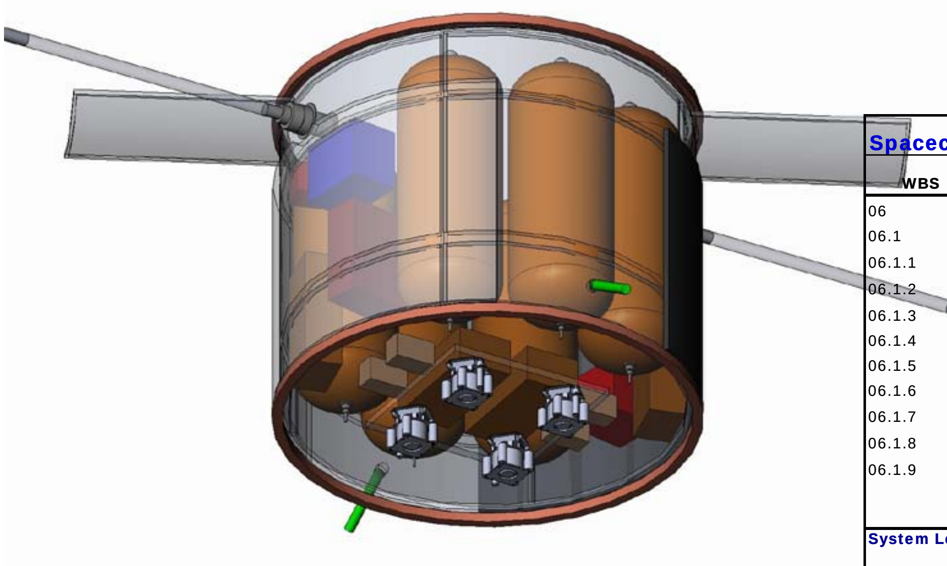
Spacecraft Master Equipment List Rack-up (Mass) - SEP Stage — COMPASS S/C Design

WBS	Main Subsystems	Basic Mass (kg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
06	Uranus SEP Stage	1606	131	1737	
06.1	SEP Stage	1606	131	1737	
06.1.1	Science Payload	0	0	0	TBD
06.1.2	Avionics	16	3	20	19%
06.1.3	Communications and Tracking	2	1	3	39%
06.1.4	Guidance, Navigation and Control	2	0	3	3%
06.1.5	Electrical Power Subsystem	200	65	265	32%
06.1.6	Thermal Control (Non-Propellant)	87	16	103	18%
06.1.7	Structures and Mechanisms	130	23	152	18%
06.1.8	Propulsion and Propellant Management	315	24	339	8%
06.1.9	Propellant	852		852	
	Estimated Spacecraft Dry Mass	753	107	884	14%
	Estimated Spacecraft Wet Mass	1606	131	1737	
System LeveL Growth Calculations					Total Growth
	Dry Mass Desired System Level Growth	753	324	1077	43%
	Additional Growth (carried at system level)		217		29%
	Total Wet Mass with Growth	1606	324	1929	

Uranus SEP Stage Hall Thruster Option

- **3+1, 4.5 kW BPT-4000 Hall provides half Isp (2000 s) of the NEXT system but the mission ∆V is ~7 km/s so Hall can be used**
  - **Twice Propellant of NEXT (1600 kg, six tanks)**
  - **Higher thrust allows greater starting mass (~800 kg)**
  - **Same power level and trip time**
  - **Heavier structure and more thermal control**
  - **Large coast Periods**
  - **Same Payload: 2630 kg**
  - **Needs advanced PPU (for unregulated solar array voltage)**
  - **Should reduce costs $20M (total with Margin)**





WBS	Spacecraft Master Equipment List Rack-up (Mass) - SEP Stage Main Subsystems	Basic Mass (kg)	Growth (kg)	COMPASS S/C Design Total Mass (kg)	Aggregate Growth (%)
06	Uranus SEP Stage	2338	133	2472	
06.1	SEP Stage	2338	133	2472	
06.1.1	Science Payload	0	0	0	TBD
06.1.2	Avionics	16	3	20	19%
06.1.3	Communications and Tracking	2	1	3	39%
06.1.4	Guidance, Navigation and Control	2	0	3	3%
06.1.5	Electrical Power Subsystem	200	65	265	32%
06.1.6	Thermal Control (Non-Propellant)	98	18	116	18%
06.1.7	Structures and Mechanisms	160	28	188	18%
06.1.8	Propulsion and Propellant Management	258	18	277	7%
06.1.9	Propellant	1601		1601	
	Estimated Spacecraft Dry Mass	738	115	871	16%
	Estimated Spacecraft Wet Mass	2338	133	2472	
System LeveL Growth Calculations					Total Growth
	Dry Mass Desired System Level Growth	738	317	1055	43%
	Additional Growth (carried at system level)		202		27%
	Total Wet Mass with Growth	2338	317	2656	

Next Steps

- Finalize Costs: SEP Stage ~$150M (no margin)
  - Seek ways to lower costs
  - Propulsion System (20% of Stage costs)
    - Consider Hall thrusters ($12M savings)
  - Power System (25% of Stage costs)
    - OTS solar arrays and cells ($10M) – will cost mass
  - SEP stage flown previously: $? savings

Backups

Integrated Probe and Orbiter ACE Run Presentations

Ice Giants Decadal Study: Appendix H

The Johns Hopkins University
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Uranus Decadal Survey ACE Run Power

Martin Fraeman

4/30/2010

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Orbiter Power Requirements

- **See 'UR PEL Ace Run Day 5 30April2010' for details**
- **3 ASRGs covers base load in all modes except TCM, UOI**
- **Supplemental power from secondary battery during maneuvers**
  - 170 Wh from battery during TCM
  - 234 Wh from battery during UOI
- **Assume 70% depth of discharge so battery capacity > 330 Wh**
- **Readily satisfied with available Li ion secondary cell technology**
  - Saft MicroSat-8S3P Li-ion MicroSat module
    - 480 Wh nominal energy
    - 16.8 Ah capacity
    - 4.5 kg Mass
- **5 kg battery mass allocated in Orbiter MEL**

Orbiter Power Electronics Design

- **Shunt regulator absorbs excess ASRG power**
  - New Horizons most recent application at APL
  - Internally redundant
- **Battery management electronics**
  - Normal engineering development
  - Internally redundant
- **Power distribution unit**
  - Build to print of APL design
  - Internally redundant

Probe Battery Requirements (1)

- **Approach**
- **29 days total duration**
- **2 power consumption modes during approach**
  - 29 days
  - Communication/Approach
    - Probe transmit/receive active
    - 2.5 hours
    - 36.9 W load (with margin)
    - 92.3 Wh total power for comm mode
  - Hibernation/Approach
    - Minimal electrical activity; timers to wake up for next mode change
    - 29 days – 2.5 hrs
    - 1.06 W load (with margin) average during this mode
      - All electronics off except:
      - 0.1 W for timers
      - USO powered up (~1 W) for warm up (~3 days) prior to comm, science mode
    - 735.1 Wh load power for hibernation mode

Probe Battery Requirements (2)

- **Science mode**
  - 1 hr duration
  - All instruments, communications active
  - 68.7 W load (with margin)
  - 68.7 Wh power for science mode
- **Other mode**
  - Support test, integration, on-orbit checkout
  - At least 2 hrs duration
  - 68.7 W load (with margin)
  - 137.4 Wh power for other mode
- **> 837.2 Wh total battery capacity required**
- **Other requirements**
  - 68.7 W peak load
  - 3 yr pre-launch + 13 yr cruise before probe release
  - Withstand probe deceleration (~400 G for ~20-30 sec)

Probe Battery Selection Issues

- **Chemistry**
  - 181 Wh/kg Hyperbattery Li-polymer (pure commercial secondary cell)
  - 346 Wh/kg Li-thionyl chloride (Saft LSH 20-150)
  - 134 Wh/kg Li Ion Secondary (Saft VES 180)
  - 472 Wh/kg Li Bromide Chloride (Electrochem BCX85)
  - Others variants need further investigation (Li-Sulfur Dioxide, Li-sulfuryl chloride, Li-polycarbon monofluoride,…)
- **Capacity**
- **Self-discharge rate**
  - Typical Li-thionyl chloride degrade at 3%/yr so at ~61% capacity at probe release
- **Peak power**
  - Li-thionyl chloride has limited discharge rate hence extra cells needed to support science mode peak power consumption
  - Extra cells mean ~7 hrs additional operation possible

Probe Battery Point Design

Saft LSH 20-150		Cell characteristics
10	Ah/cell	Cell capacity @ 25°C (from performance curve)
3.6	V	Cell voltage at rated capacity (from performance curve)
36	Wh/cell	
3 %	per yr	Self Discharge rate
16	yr	storage time (3yr pre + 13 yr post launch)
61.4%		Approximate remaining charge factor before use
6.14	Ah/cell	Capacity at arrival/cell
22.1	Wh/cell	Capacity at arrival/cell
0.104	kg/cell	Mass characteristics
0.832	kg/string	
1.3	structure factor	
1.0816	kg/string	
8	cell/string	Battery design
176.9	Wh/string	
28.8	battery voltage	
8	strings	total strings
1415.2	Wh	Battery capacity at release
64	cells	total number of cells
0.3	A	max current/string
2.4	A	max current/battery
69.1	W	peak power/battery
8.7	***kg***	***mass/battery***

Probe Battery Issues

- **Can battery withstand deceleration profile?**
- **Battery performance after long cruise**
  - Assumed 3%/year self-discharge
  - Is that valid for 16 years?
- **Battery performance in probe load/temperature profile**
  - Hibernation for 30 days with comm interruptions
  - Full load after 29 day approach for 1 hour
  - Include temperature changes

Uranus Decadal Survey ACE Run
RF Communications

Brian Sequeira

April 30, 2010

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Requirements and Assumptions

- **X-band LGA for near-Earth (up to 0.15 AU) operations**
  - Maximum Earth-S/C distance for LGA communications: 1.75 AU
- **X-band MGA to support cruise and emergency operations to 21 AU**
  - MGA pointed at Earth during emergency
- **X-band HGA to support TCMs, SEP separation, contacts during hibernation, and communication with probe during descent**
  - Must point HGA at probe
- **Ka-band downlink to support science return to Earth**

Trades

- **Simultaneous contact with probe and Earth not feasible**
  - Required power consumption prohibitive
- **Pointing HGA at probe supported by a monopulse tracking receiver on the orbiter**
- **Radio on probe is frequency inverted with respect to orbiter**

Concept Summary

- **Probe**
  - X-band radio (transmits at 7.2 GHz, receives at 8.4 GHz)
  - Transmits 2 W RF
- **Orbiter**
  - X-band command; X/Ka-band Telemetry
  - 3-LGA (X-band), 1 MGA (X-band), 1 2.5 m diameter HGA (X/Ka-band)
  - Transmits 60 W RF at X-band; 40 W RF at Ka-band
  - Monopulse receiver for communications with probe
  - Orbiter-to-probe: 250 bps; Probe-to-orbiter: 200 bps w/ranging

Data Rates from Uranus Orbits (Sample Seasonal)

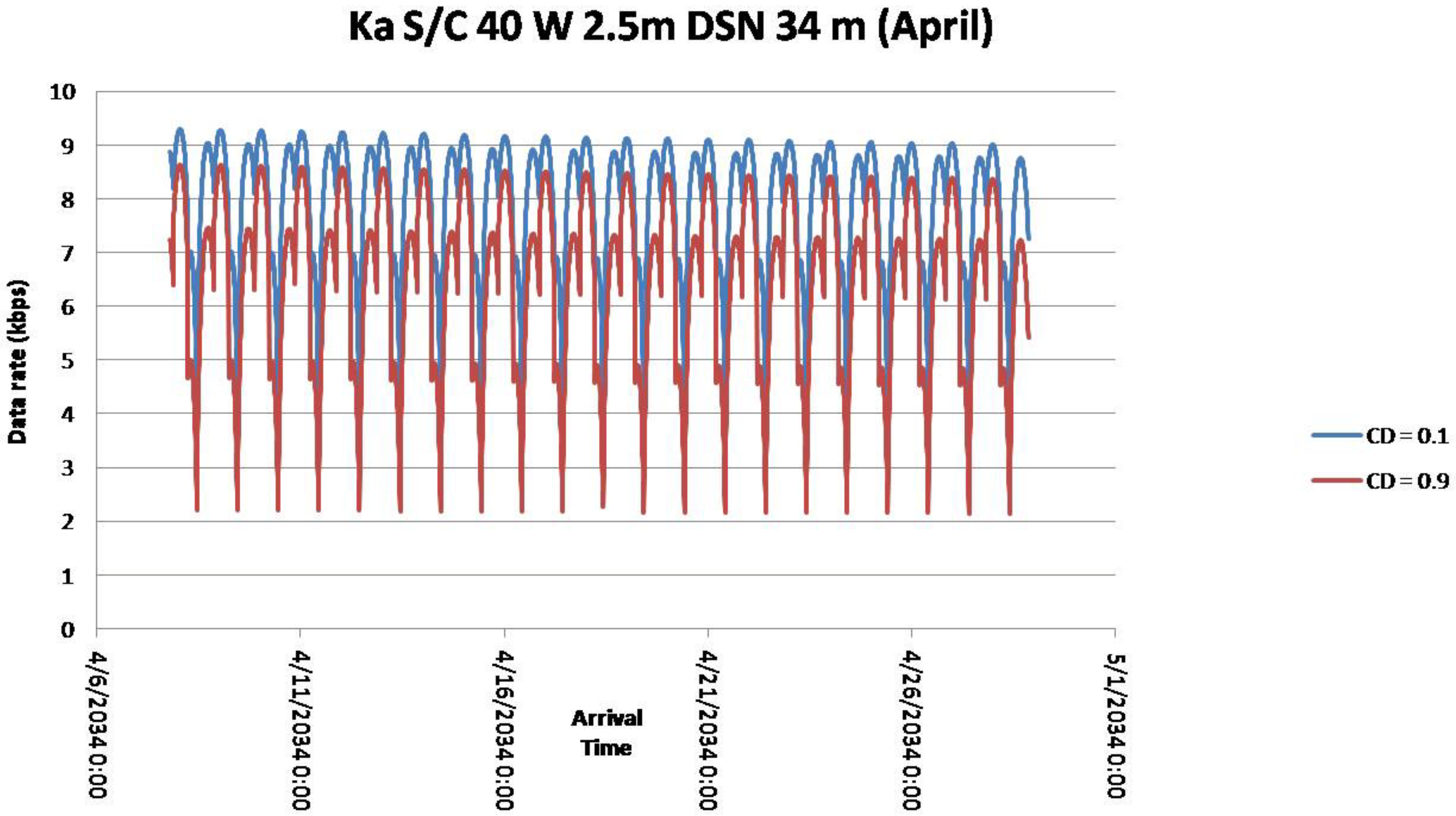
Brian Sequeira

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

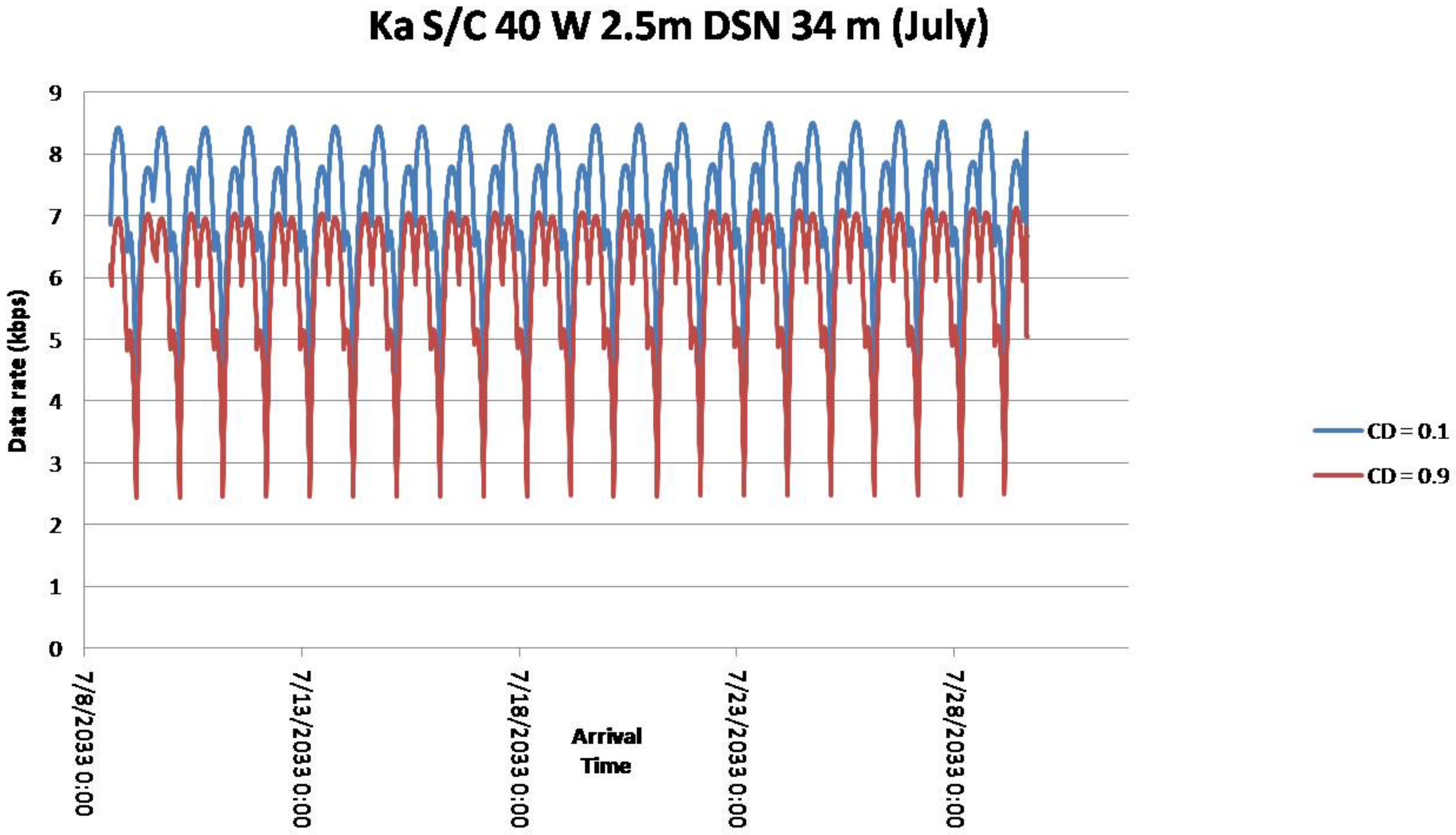
Assumptions

- **Ka-band transmitter: 40 W RF (100 W DC)**
- **High-gain reflector antenna: 2.5 m diameter**
- **JPL weather data from Ka-band radiometer**
- **Mission trajectory to ground station with best view (elevation) at time of signal arrival at Earth**
- **No account for occultation/solar conjunction**
  - These must not occur at least 1 light time ahead of pass and for 8 hours thereafter

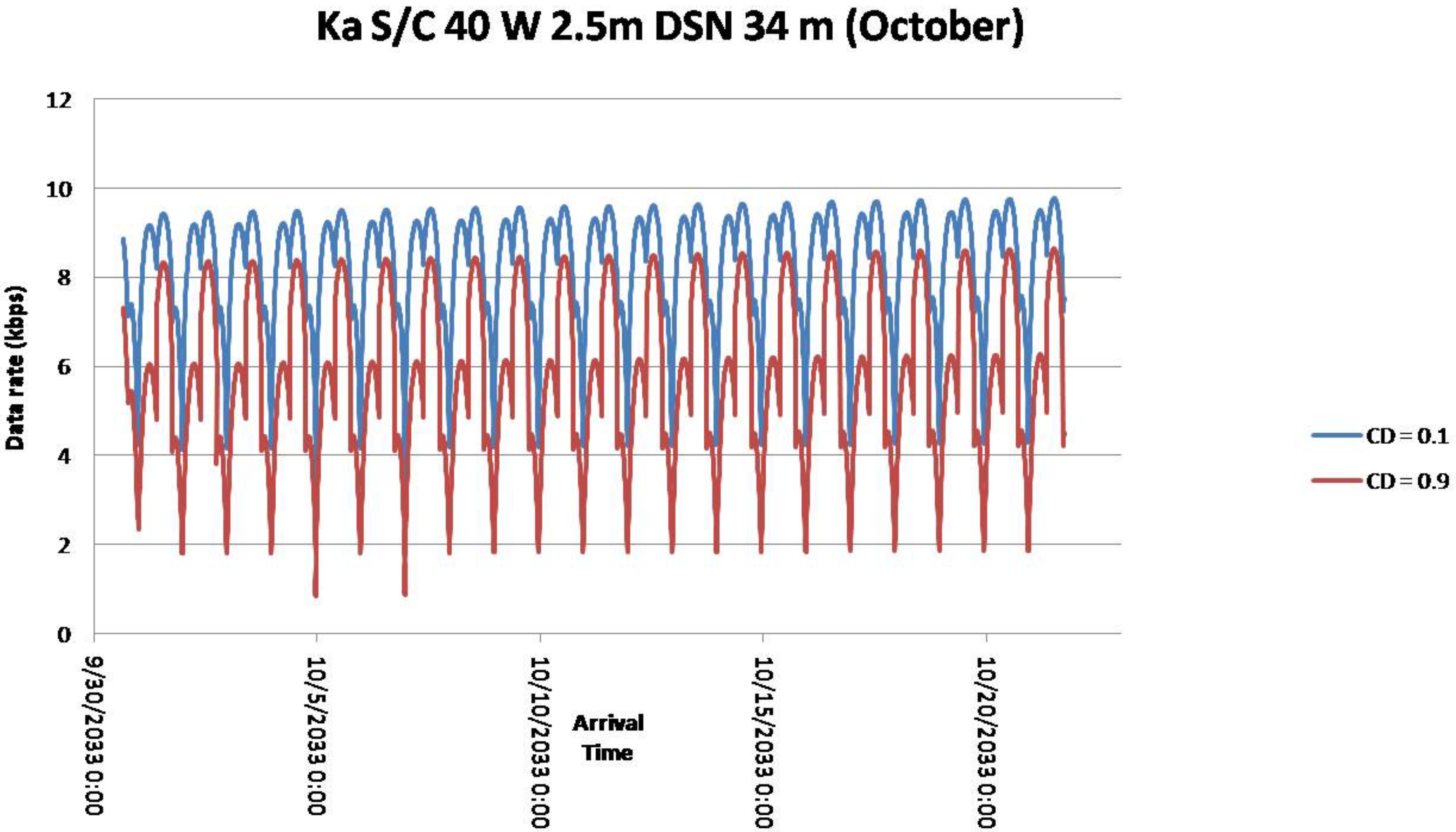
Data Rate from Uranus Orbit (April)



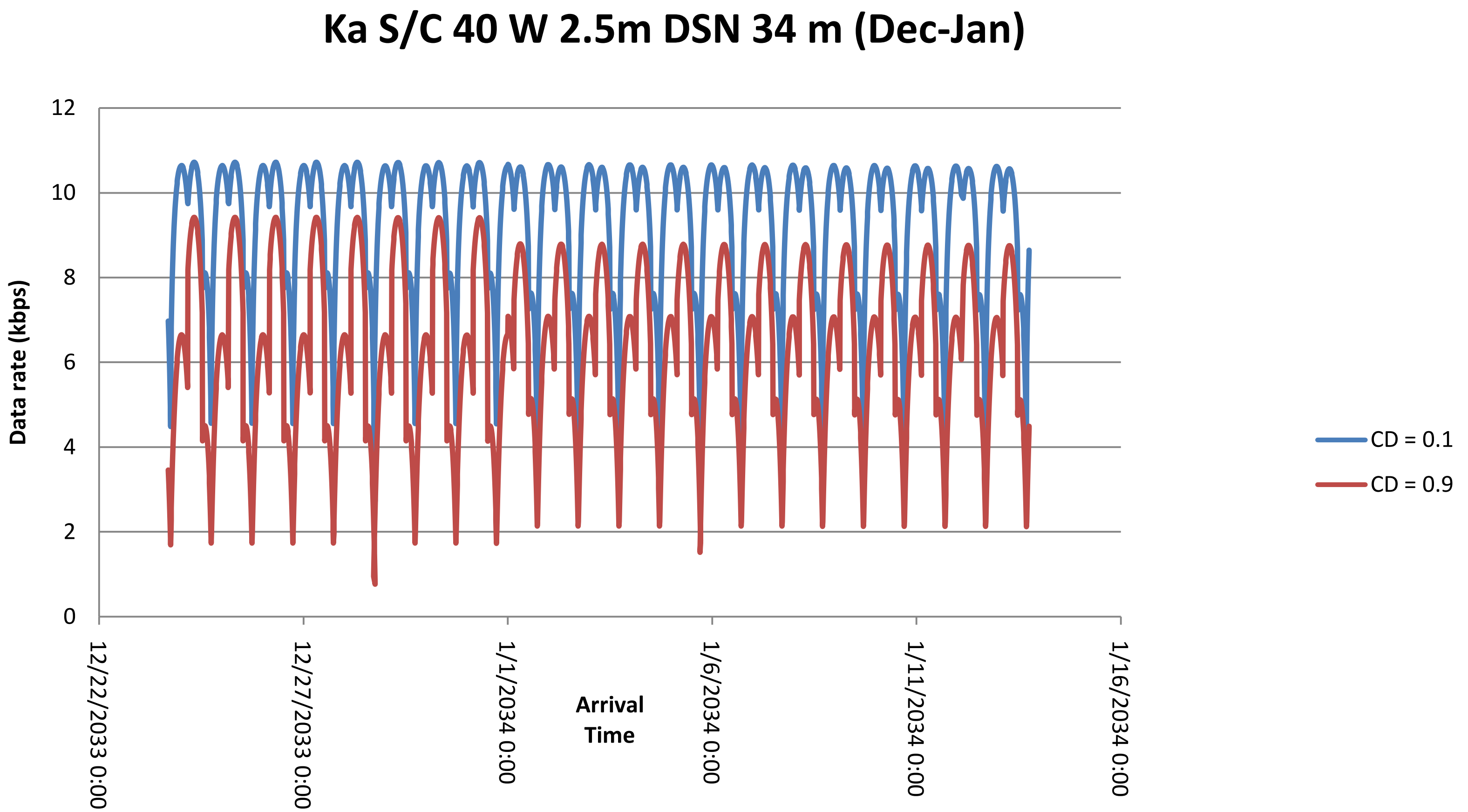
Data Rate from Uranus Orbit (July)



Data Rate from Uranus Orbit (October)



Data Rate from Uranus Orbit (Dec-Jan)



Observations

- **Goldstone is most favorable for preceding cases**
  - Weather effect is smallest
  - Elevation angle is favorable
- **Canberra is worst (weather & elevation)**
- **Goldstone supports ~7 kbps over all four cases and weather (7 kbps → 25.2 Mb/hr → ~200 Mb/day)**

Uranus Decadal Survey ACE Run Propulsion Subsystem

Stewart Bushman

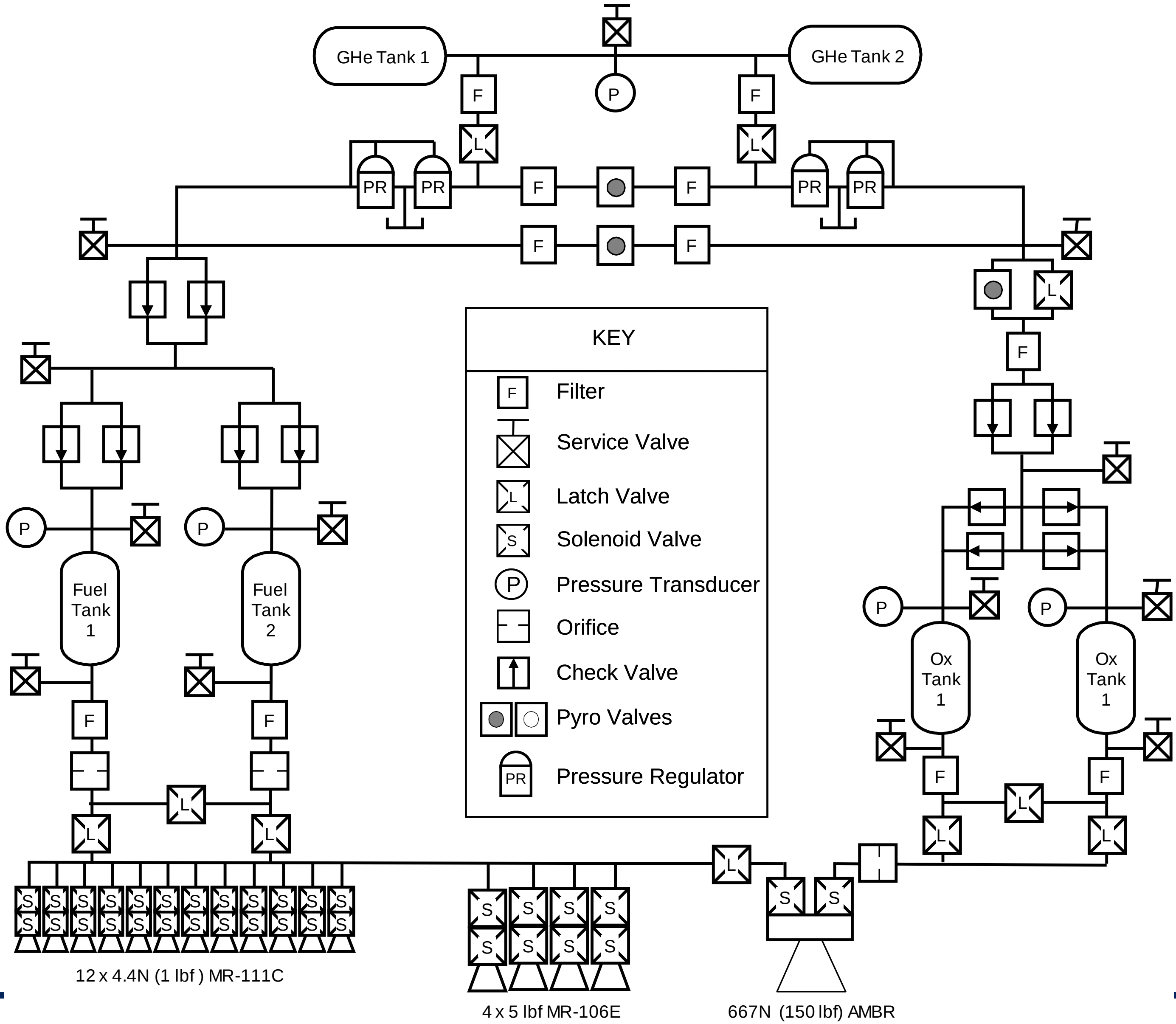
30 April 2010

The Johns Hopkins University APPLIED PHYSICS LABORATORY

Requirements and Assumptions

- **Sufficient Thrust, Isp, Propellant, Pressurant for:**
  - 2594.5 m/s total mission ΔV
    - 2380.1 m/s Biprop (N_2H_4/NTO)
    - 214.4 m/s Monoprop
  - 50 kg N_2H_4 ACS
    - Pointing
    - Spin Rate Changes
    - Precessions
- **Temperature Range: 5 C – 50 C**

Propulsion Subsystem Schematic



Concept Summary

- **Dual Mode System**
  - Thrusters:
    - 667N (150 lbf) Dual Mode (N_2H_4/NTO) Engine (1)
    - 22N (5 lbf) N_2H_4 Monopropellant Steering Thrusters (4)
    - 4.4N (1 lbf) N_2H_4 Monopropellant Thrusters (12)
  - Hardware for propellant flow (COTS baseline, including tanks)

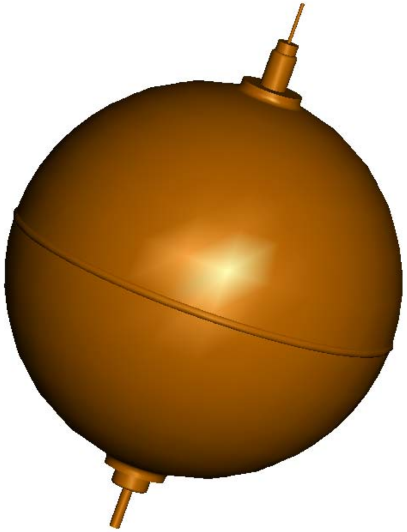
- **Propellant Budget for ΔV = 2594.5 m/s:**

S/C Dry Mass =	906.3	kg
Fuel Mass =	714.1	kg
Oxidizer Mass =	580.5	kg
Orbiter Wet Mass =	2200.8	kg

- **Propulsion System Dry Mass = 135 kg (includes Pressurant and Residual/Margin)**

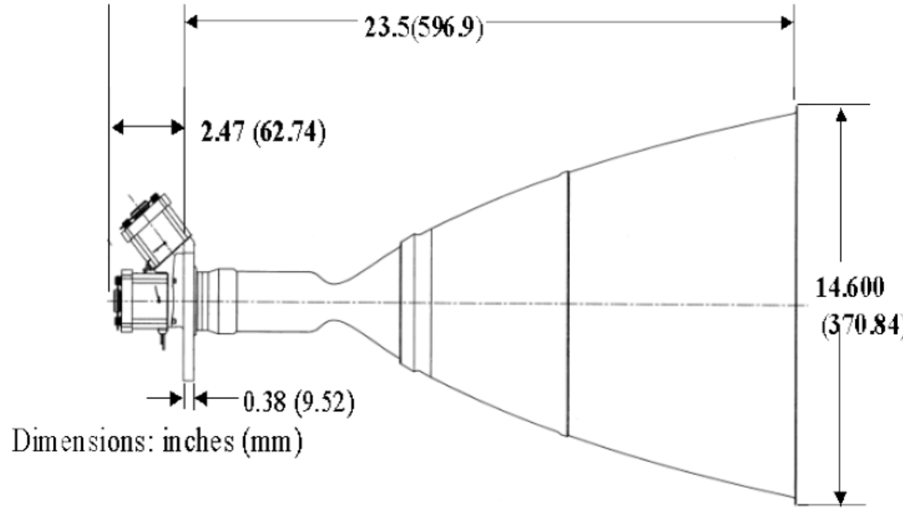
Propellant Tank Baseline

- Current Configuration: All COTS items with flight heritage
  - 1 Fuel Tank, 1 Oxidizer Tank, 1 Pressurant Tank
    - Fuel: ATK-PSI 80340, GEA S-5000 Heritage
    - Ox: ATK-PSI 80339, GEA S-5000 Heritage
    - Pressurant: PSI 80436, Astrolink Heritage



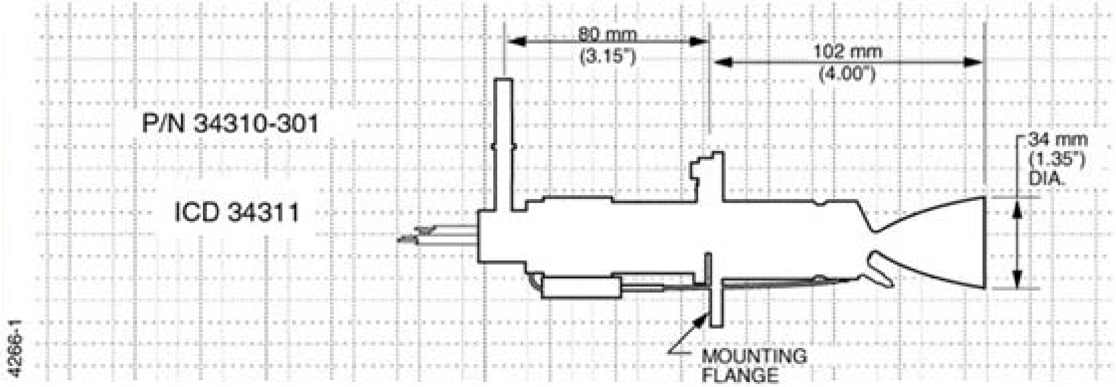
- A Nutation Time Constant analysis may be required early-on in program.

AMBR - 667N (150 lbf) Dual Mode Thruster



Design Characteristic	AMBR
Thrust (lbf)	150
Specific Impulse (s)	333.5
Inlet Pressure (psia)	275
Ox/Fuel Ratio	1.1
Expansion Ratio	400:1
Physical Envelope	
Length (in)	25.97
Exit Diameter (in)	14.6
Valve Power (W @ 28VDC)	46

MR-106E 22N (5 lbf) Rocket Engine Assembly



Design Characteristics

- Propellant.......................................Hydrazine
- Catalyst.......................................LCH-227/202
- Thrust/Steady State.................... 30.7 – 11.6 N (6.9 — 2.6 lbf)
- Feed Pressure....................... 24.1– 6.9 bar (350 — 100 psia)
- Chamber Pressure..................12.4 – 4.5 bar (180 — 65 psia)
- Expansion Ratio.. 60:1
- Flow Rate........... 13.1 – 5.0 g/sec (0.0289 — 0.011 lbm-sec)
- Valve .... Dual Seat
- Cat. Bed Heater Pwr.........6.53 Watts Max @ 28 Vdc & 21° C
- Valve Heater Power .... 3.27 Watts @ 28 Vdc & 21° C
- Valve Power .... 25.3 Watts Max @28 Vdc & 21° C
- Mass .... 0.635 kg (1.4 lbm) max

* Mars Odyssey Test Program
December, 2000

Performance

- Specific Impulse..................................... 235 – 229 sec (lbf-sec/lbm)

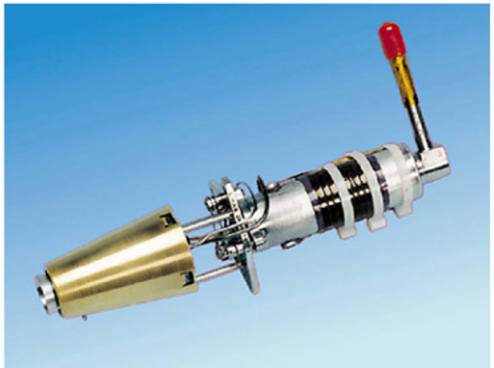
	REA 'A'	REA 'B'	Mars*
Total Impulse...........	120,000 N-sec	125,000 N-sec	90,587 N-sec
....	(26,958 lbf-sec)	(28,044 lbf-sec)	(20,366)
Total Pulses..................	12,405	186	66,631

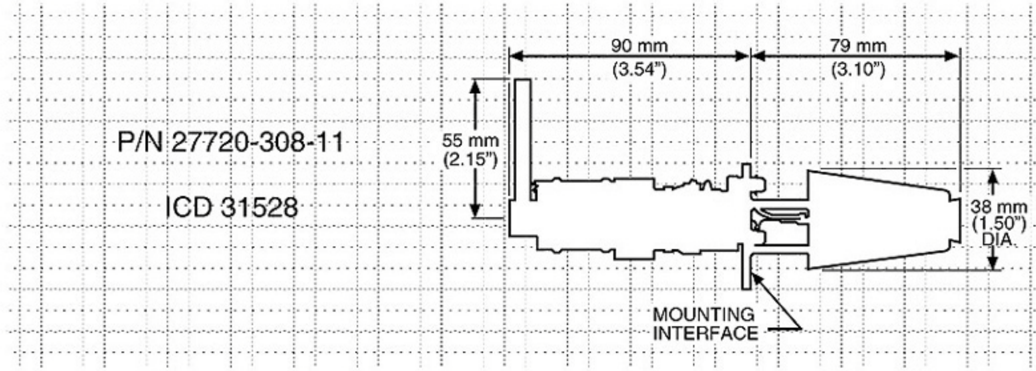
- Minimum Impulse Bit .... 0.46 N-sec @ 12.8 bar & 16 ms ON
  .... (0.103 lbf-sec @185 psia & 16 ms ON)
- Steady State Firing.................................. 2,000 sec — Single firing
  .... 4,670 sec — Cumulative

- APL Heritage: MESSENGER, NEAR, and CONTOUR

MR-111C 4.4N (1 lbf) Rocket Engine Assembly

MR-111C 4 N (1.0-lbf) ROCKET ENGINE ASSEMBLY





Design Characteristics

- Propellant..Hydrazine
- Catalyst... S 405
- Thrust/Steady State..................... 5.3 – 1.3 N (1.2 – 0.3 lbf)
- Feed Pressure...................... 27.6 – 5.5 bar (400 – 80 psia)
- Chamber Pressure................ 12.1 – 3.4 bar (175 – 50 psia)
- Expansion Ratio.. 74:1
- Flow Rate............. 2.4–0.6 g/sec (0.0053–0.0014 lbm/sec)
- Valve.. Dual Seat
- Valve Power..................... 8.25 Watts Max @ 28 Vdc & 21°C
- Valve Heater Power........ 1.54 Watts Max @ 28 Vdc & 21°C
- Cat. Bed Heater Pwr....... 3.85 Watts Max @ 28 Vdc & 21°C
- Mass .... 0.33 kg (0.73 lbm)
  - Engine.. 0.13 kg (0.28 lbm)
  - Valve... 0.20 kg (0.45 lbm)

Performance

- Specific Impulse.................................. 229 – 215 sec (lbf-sec/lbm)
- Total Impulse................................ 260,000 N-sec (58,500 lbf-sec)
- Total Pulses.. 420,000
- Minimum Impulse Bit.................... 0.08 N-sec @ 6.9 bar & 15 ms ON
  ....(0.0171 lbf-sec @ 100 psia & 15 ms ON)
- Steady State Firing........................... 5,000 sec min — Single firing

- APL Heritage: MESSENGER, STEREO, and New Horizons

Subsystem Characteristics

Flight System Element Parameters (as appropriate)	Value/ Summary, units
General	
Design Life, months	
Propulsion	
Estimated delta-V budget, m/s	2594.5 m/s + ACS
Propulsion type(s) and associated propellant(s)/oxidizer(s)	Hydrazine/NTO
Number of thrusters and tanks	1/4/12 Thrusters 2/2/1 Tanks
Specific impulse of each propulsion mode, seconds	Biprop: 332 s Monoprop: 215 s

Uranus Decadal Survey
G&C Summary

Robin Vaughan

Wen-Jong Shyong

The Johns Hopkins University
APPLIED PHYSICS LABORATORY

Guidance & Control Requirements Capabilities

- Provide 3-axis control
  - Launch/detumble and commissioning
  - During inner cruise while using SEP engines
  - For science phases at Uranus starting with probe deployment through end of satellite tour orbits
  - During all TCMs including UOI
- Provide spin-up/spin-down and spin control
  - During outer cruise after separation from SEP stage
  - During probe release
- Provide ΔV maneuver capability
  - UOI – powered turn using bi-prop engine, ~1 hour duration
  - Probe release targeting and orbit adjust maneuvers – fixed direction using bi-prop engine
  - Intermediate ΔV using larger mono-prop thrusters
  - Small ΔVs using smaller mono-prop thrusters

Guidance & Control Requirements Performance

- Attitude Knowledge Requirement
  - +/- 100 arcsec 2σ in each axis (estimated AA-STR capability)

- Attitude Control Requirement
  - Launch and inner cruise
    - Point SEP engine to thrust vector direction with 1.0°
    - Point MGA to Earth with 3.0° for communication
  - Passive attitude control via spin stabilization during hibernation periods of outer cruise
  - Outer cruise through end-of-mission under active control
    - Point HGA to Earth and to probe within 0.1°
      - Control using radio signal feedback from probe required to compensate for probe trajectory dispersions during probe descent
    - Point science instruments to planet and satellites within 0.1°
- Agility/Slew Rate Requirement
  - Slew spacecraft over 180° during ~1 hour orbit insertion burn
  - Track slow evolution of thrust vector direction for SEP engines during inner cruise
- Stability Requirement
  - Pointing stability of 5 μrad/sec for science observations of planet and satellites

Guidance & Control Design Concept

- Guidance and control system design is mixture of New Horizons and MESSENGER (similar to Cassini)
- Attitude determination is handled with 2 star trackers and an IMU
  - Sun sensors provide coarse attitude determination for safe mode operation
  - Sun sensors on SEP stage supplement those on orbiter during inner cruise
- Four reaction wheels are used for precision attitude control
  - Wheels sized to handle momentum build up during inner cruise from solar pressure and SEP thruster “swirl torque”
    - Gimbaling of SEP engines also used to off load momentum where possible
  - Wheels also sized to accommodate changing spacecraft inertia with SEP stage and probe attached and separated from orbiter
  - Wheel torque limited by available power during science observations at planet
- Twelve 4.4 N thrusters are used for attitude control during momentum dumps and TCMs
  - Provide torque around 3 orthogonal axes with coupled pairs
  - Provide ΔV along 2 orthogonal axes (no ΔV parallel to antenna boresight)
- Four 22 N thrusters are used for thrust vector control during bi-prop engine burns and used alone for intermediate TCMs
- One 663 N bi-prop engine used for UOI and a few other orbit adjust maneuvers

Equipment/Product List
G&C Hardware on Orbiter (1)

- **Galileo AA-STR Star Tracker (APS detector)**
  - 2 units
  - Inertial orientation to 100 arcsec 2σ
  - 1.425 kg per unit, 6.5 W per unit (only one tracker on at a time)
- **Honeywell MIMU**
  - 2 complete IMUs, each with 3 GG1320 ring laser gyros and 3 accels (QA3000)
  - Gyro performance
    - Precision: integrated angular rate at 1 microrad/LSB
    - Scale factor error 1-σ knowledge < 1 ppm
    - Uncompensated bias 1-σ stability 0.005 /hr
    - Maximum rate for full accuracy 187 /s (~31 rpm)
  - Accelerometer performance:
    - Precision: integrated linear acceleration at 0.0753 mm/sec/LSB
    - Scale factor error 1-σ knowledge < 50ppm
    - Uncalibrated bias < 5 mg; 1-σ stability 1 μg/hr
    - Maximum acceleration for full accuracy 18.219 m/s^2, or about 1.86 g
  - 4.44 kg per unit, 32 W (only one IMU on at a time)

Equipment/Product List
G&C Hardware on Orbiter (2)

- **AeroAstro Sun Sensors**
  - 6 optical heads, Sun direction to 1 (analog signal processed by IEM to get Sun direction in software)
  - 0.036 kg per optical head
  - Negligible power consumption
- **Honeywell Constellation Series HR14 Reaction Wheel**
  - 4 reaction wheels, 0.1-0.2 Nm max torque, 50 Nms momentum storage
  - 8.5 kg per unit
  - 7.0 W per unit typical at steady state, ~100 W per unit at max torque
- **Thrusters used for ΔV and attitude control listed in propulsion presentation**

Wheels vs. Thrusters

Actuator Choice	Pros	Cons
Wheels	•Smoother pointing for science observations, SEP thrust profile	•Higher power needed for fast turns •Need to manage wheel turn off and on during hibernation periods; periodic exercising needed to maintain lubrication •Mission duration approaching typical wheel design lifetimes •Need to periodically off load momentum using thrusters
Thrusters	•Somewhat lower power than wheels, but not a large gain since catbed heaters must be on •Relatively easy to integrate with other required propulsion hardware	•Difficult to handle jitter requirement with deadband control •Large number of cycles accumulated on small thrusters due to tight pointing requirement and duration of inner cruise and orbit operations •Additional propellant required to handle attitude control needs

G&C Flight System Summary Table Part 1

Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.).	3-axis stabilized during launch, inner cruise, planet operations (probe release through end of satellite tour) Spin stabilized during hibernation periods of outer cruise
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	Inertial (star tracker) for normal operations under active attitude control Probe radio signal for probe data relay during probe descent (compensate for probe trajectory uncertainties) Sun for safe mode
Attitude control capability, degrees	1° for launch and inner cruise 0.1° for planet operations (probe release through end of satellite tour) 0.1° for probe data relay during probe descent
Attitude knowledge limit, degrees	0.03° (100 arcsec) 2σ
Agility requirements (maneuvers, scanning, etc.)	180° powered turn on thrusters for ~1 hour orbit insertion burn Slow slew to follow SEP thrust vector profile during inner cruise Jitter < 5 μrad/s during science observations

G&C Flight System Summary Table Part 1

Attitude Control	
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	2 star trackers, inertial attitude knowledge to 100 arcsec 2σ 3 gyros in each of 2 IMUs, integrated angular rate Precision: 1 μrad/LSB Scale factor error 1σ knowledge < 1 ppm Uncompensated bias 1σ stability 0.005 °/hr Maximum rate for full accuracy 187 °/s 3 accelerometers in each of 2 IMUs, Precision: integrated angular rate at 0.0753 mm/s/LSB Scale factor error 1σ knowledge < 1 ppm Uncompensated bias 1σ stability 0.005 °/hr Maximum rate for full accuracy 18.2 mm/s (~1.9 g) 6 Sun sensor heads, Sun direction to ±1° 4 reaction wheels, 50 Nms momentum storage, maximum torque 0.1-0.2 Nm per wheel Thrusters - twelve 4.4 N, four 22 N, 1 663 N bi-prop

G&C BACKUP

G&C Launch, Initial Deployments, Commissioning

- **Spacecraft configuration**
  - Separation/detumble: SEP stage attached to main spacecraft but large solar panels stowed
  - Commissioning up to start of powered cruise: SEP stage attached to main spacecreaft and large solar panels deployed
- **Launch events**
  - Separation from launch vehicle
    - detumble required to a 3-axis stabilized attitude
    - Probably thrusters for this detumble, automatic transition to wheels when rates get low enough
  - Deployments
    - Need to deploy solar arrays on SEP stage within a few hours of separation; also need to have attitude such that panels are pointed at the Sun
  - No launch injection correction maneuver using chemical propulsion – SEP stage will correct the errors once it starts its thrust profile
- **Commissioning (after launch/separation and before powered cruise)**
  - ~30 days
  - Check out the SEP engines and other spacecraft components; generally confirm all the hardware is working as planned
  - Probably no tight pointing requirements unless we just want to practice for later. Probably will have to try to hit the SEP pointing requirements during the practice sessions
- **Pointing driver is SEP engine pointing and pointing at Earth with MGA**
  - Likely ~1 deg pointing accuracy required – driven by SEP, MGA pointing is ~3 deg accuracy
- **Agility (slew rate, turn time) driver is detumble?**

Launch, Commissioning

Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.).	3-axis stabilized
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	Inertial (star tracker)
Attitude control capability, degrees	1
Attitude knowledge limit, degrees	100 arcsec = 0.03 deg
Agility requirements (maneuvers, scanning, etc.)	Detumble from TBD deg/sec to inertial fixed in a TBD minutes Slow slew to keep antenna pointing at Earth or arrays on Sun
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	Solar arrays on SEP stage – 1DOF, 1 axis Engines on SEP stage – 1 DOF gimbal
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	Star tracker, gyros (IMU) Thrusters - 4.4 N likely for detumble Wheels after detumble

G&C Powered Cruise (SEP Firing)

- **Spacecraft configuration**
  - SEP stage attached to main spacecraft with large solar panels deployed
- **Powered cruise events**
  - Long periods of constant thrust from SEP engine to get on trajectory to Uranus
    - SEP stage attached to main spacecraft with large solar panels deployed
    - Need to point thrust axis at specified direction with arrays kept close to Sun-pointed
    - Communications possible via LGAs without changing spacecraft attitude out to ~1.4 AU
    - Last SEP thrusting period ends at ~5 AU from Sun
  - Earth flyby with no thrust (~42 days)
  - Short periods of no thrust from SEP engine for comm and navigation (~20-30 days each)
    - Need to point MGA (co-boresighted with HGA) at Earth
- **Disturbance torques are solar pressure and "swirl" torque from SEP engine**
    - Largest difference in CP-CM during this time and solar flux is greatest since still close to Earth
    - Momentum dumps needed for wheels
- **Pointing driver is SEP engine pointing**
  - ~1 deg pointing accuracy required – driven by SEP thrust direction accuracy,
  - MGA pointing to Earth only needs ~3 deg accuracy
- **Agility (slew rates, time to complete) driver?**
  - No identified activity that requires fast turns, might be a matter of convenience in being able to slew to Earth for coast periods; probably just take advantage of capability needed for other phases

Powered Cruise (using SEP)

Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.).	3-axis stabilized
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	Inertial (star tracker)
Attitude control capability, degrees	1 deg for SEP pointing, ~3 deg for MGA pointing
Attitude knowledge limit, degrees	100 arcsec = 0.03 deg
Agility requirements (maneuvers, scanning, etc.)	Slow slew to keep SEP engine on controlled trajectory profile Occasional “coasting” periods when antenna pointing back to Earth – faster turns to go back and forth between Earth pointing and SEP pointing
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	Solar arrays on SEP stage – 1DOF, 1 axis Engines on SEP stage – 1 DOF gimbal
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	Star tracker, gyros (IMU) Thrusters - 4.4 N – during momentum dumps Wheels for SEP and MGA pointing outside of momentum dumps

G&C Unpowered Cruise (no SEP stage)

- **Spacecraft configuration**
  - Main spacecraft only (orbiter + probe)
  - SEP stage with its large solar panels has been jettisoned
- **Unpowered cruise events**
  - Separation from SEP stage – not sure if any G&C concerns here
  - Long periods of hibernation similar to NH
  - A few TCMs to target for Uranus – small enough to use mono-prop thrusters
  - A few periods for comm, spacecraft checkout, navigation between TCMs
    - Need to point HGA at Earth
- **Disturbance torques are small**
  - Maybe just torque from ASRG?
  - Solar torque decreasing since moving farther from Sun
- **Spacecraft is spin-stabilized during hibernation periods – no active attitude control**
- **Active attitude control during TCMs and checkout periods**
- **Pointing driver is HGA pointing**
  - ~0.1 deg pointing accuracy required
  - This should be good enough for TCMs as well
- **Agility (slew rates, time to complete) driver?**
  - No identified activity that requires fast turns, might be a matter of convenience in being able to slew to Earth for coast periods; probably just take advantage of capability needed for other phases

G&C Probe Deployment

- **Spacecraft configuration**
  - Main spacecraft only (orbiter + probe) up to time of probe release
  - Orbiter only after probe release
- **Probe deployment events**
  - Final probe targeting TCM (target for Uranus impact trajectory)
  - Probe release: spin up combined orbiter+probe, release probe, put orbiter back in 3-axis stable attitude (no spin)
  - Orbiter deflection maneuver (target to UOI aim point) – first use of bi-prop engine
- **Spacecraft is 3-axis stabilized with active attitude control throughout**
  - Attitude control with thrusters for TCMs and probably also spin up/down for probe release
  - Attitude control with wheels for periods in between
- **Pointing driver is HGA pointing**
  - ~0.1 deg pointing accuracy required to point HGA to Earth to confirm readiness for probe release, TCMs, probe separation, etc
  - This should be good enough for TCMs and probe release as well
- **Agility (slew rates, time to complete) driver?**
  - Turn from probe release attitude to attitude for orbiter deflection maneuver within 24 hours (maybe shorter if want to have some HGA comm time in between these two events)
  - Not sure if any constraint on duration for spin up to probe release rate

G&C Probe Entry/ Orbit Insertion

- **Spacecraft configuration**
  - Orbiter only – probe is diving into Uranus atmosphere
- **Probe entry events**
  - Turn to track probe entry and descent in Uranus atmosphere ~1 hour of tracking required
  - Turn back to point HGA at Earth to relay probe data? or to confirm readiness for UOI?
- **Orbit insertion events**
  - UOI burn – main use of bi-prop engine, powered turn, ~1 hour after end of probe data relay
  - Turn back to HGA at Earth asap after UOI burn complete and Earth is visible
  - UOI clean up burn – on mid-sized mono-prop thrusters
- **Spacecraft is 3-axis stabilized with active attitude control throughout**
  - Attitude control with thrusters for UOI
  - Attitude control with wheels for probe tracking
- **Pointing driver is HGA pointing during probe data relay**
  - ~0.1 deg pointing accuracy required to point HGA to Earth
  - This should be good enough for TCMs as well
  - HGA pointing to probe during relay of descent data to 0.1 deg – will require active control using feedback from probe radio signal to compensate for probe trajectory dispersions
- **Agility (slew rates, time to complete) drivers**
  - Turn nearly 180 deg during 1 hour of UOI burn on thrusters
  - Turn back from probe tracking to Earth point in << 1 hour since only 1 hour between end of probe data transmission and start of UOI burn

Probe Operations and UOI

Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.).	3-axis stabilized
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	Inertial (star tracker)
Attitude control capability, degrees	0.1 (need radio signal feedback from probe comm link to achieve this during probe descent)
Attitude knowledge limit, degrees	100 arcsec = 0.03 deg
Agility requirements (maneuvers, scanning, etc.)	Turn up to 180 deg during ~1 hour orbit insertion burn
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	Solar arrays on SEP stage – 1DOF, 1 axis Engines on SEP stage – 1 DOF gimbal
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	Star tracker, gyros (IMU) Thrusters during orbit adjust maneuvers, wheel momentum dumps Wheels for science pointing, data downlink to Earth

G&C Prime Science – Planet Orbits

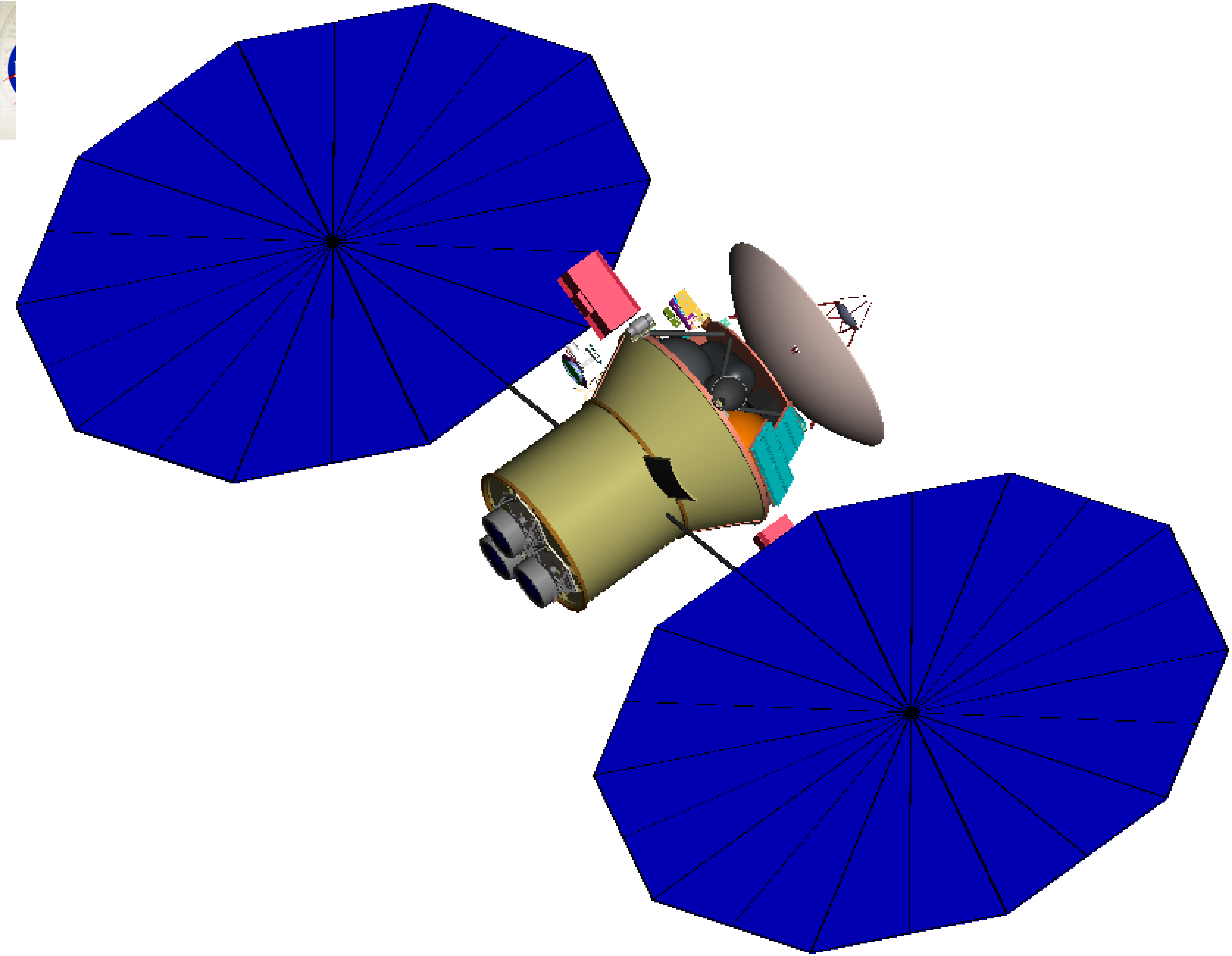
- **Spacecraft configuration**
  - Orbiter only – with a lot of mass gone from propellant tanks
- **Prime science events**
  - Small TCMs at apoapses for orbit control (avoid rings) ~every 21 days, using mono-prop thrusters
  - Brief periods of pointing high-resolution imagers at target features
  - Longer periods of pointing lower-resolution camera at planet
  - ~8-hour daily downlink tracks pointing HGA at Earth
- **Spacecraft is 3-axis stabilized with active attitude control throughout**
  - Attitude control with thrusters for TCMs to adjust orbit, occasional momentum dumps
  - Attitude control with wheels for science
- **Pointing driver is science pointing**
  - ~0.1 deg pointing accuracy required to point instruments at planet
  - This should be good enough for TCMs
  - Jitter requirement for 5 microrad/s for short period with high-resolution instrument
- **Agility (slew rates, time to complete) drivers**
  - No expected high rate tracking for science

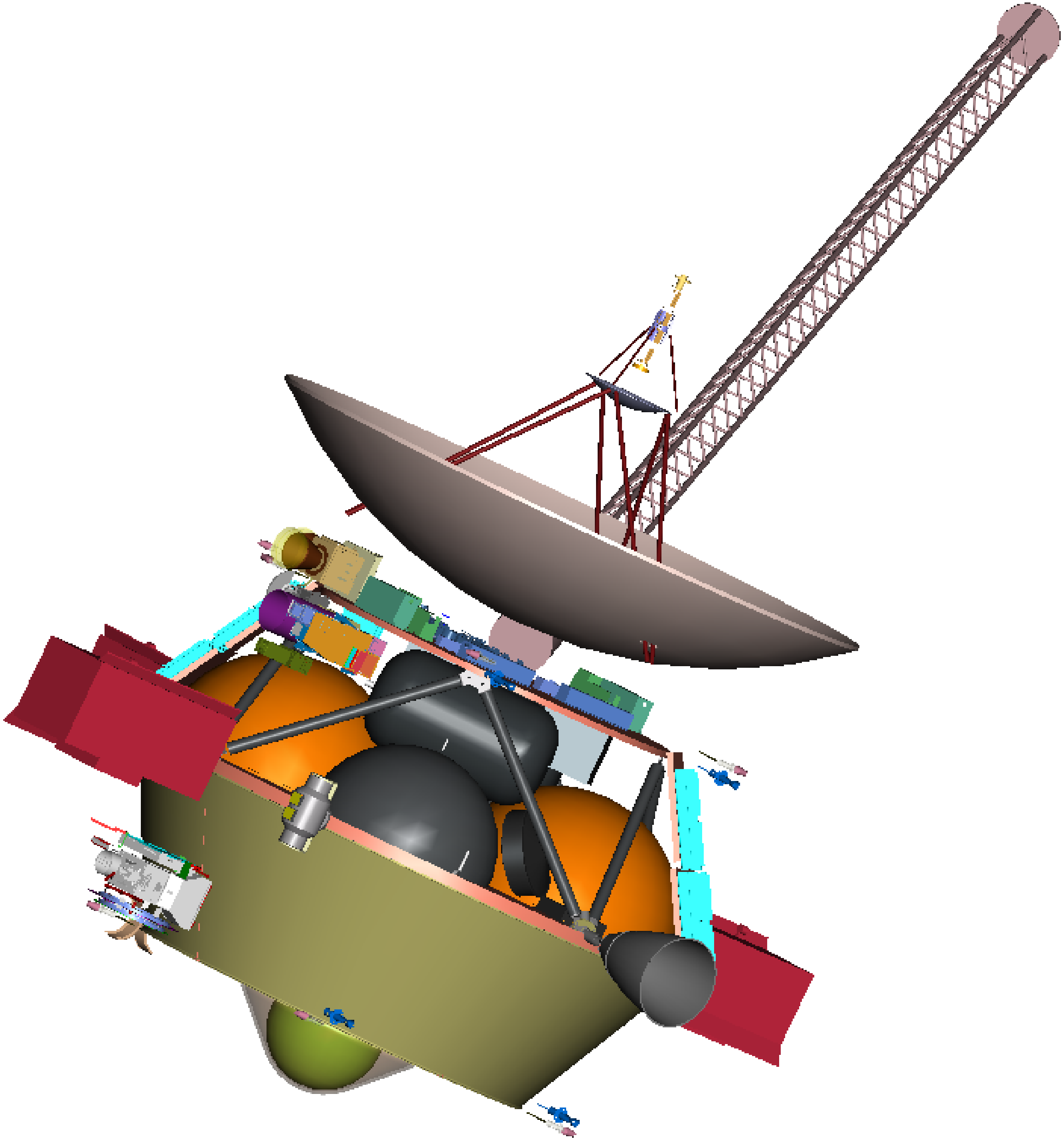
G&C Satellite Tour

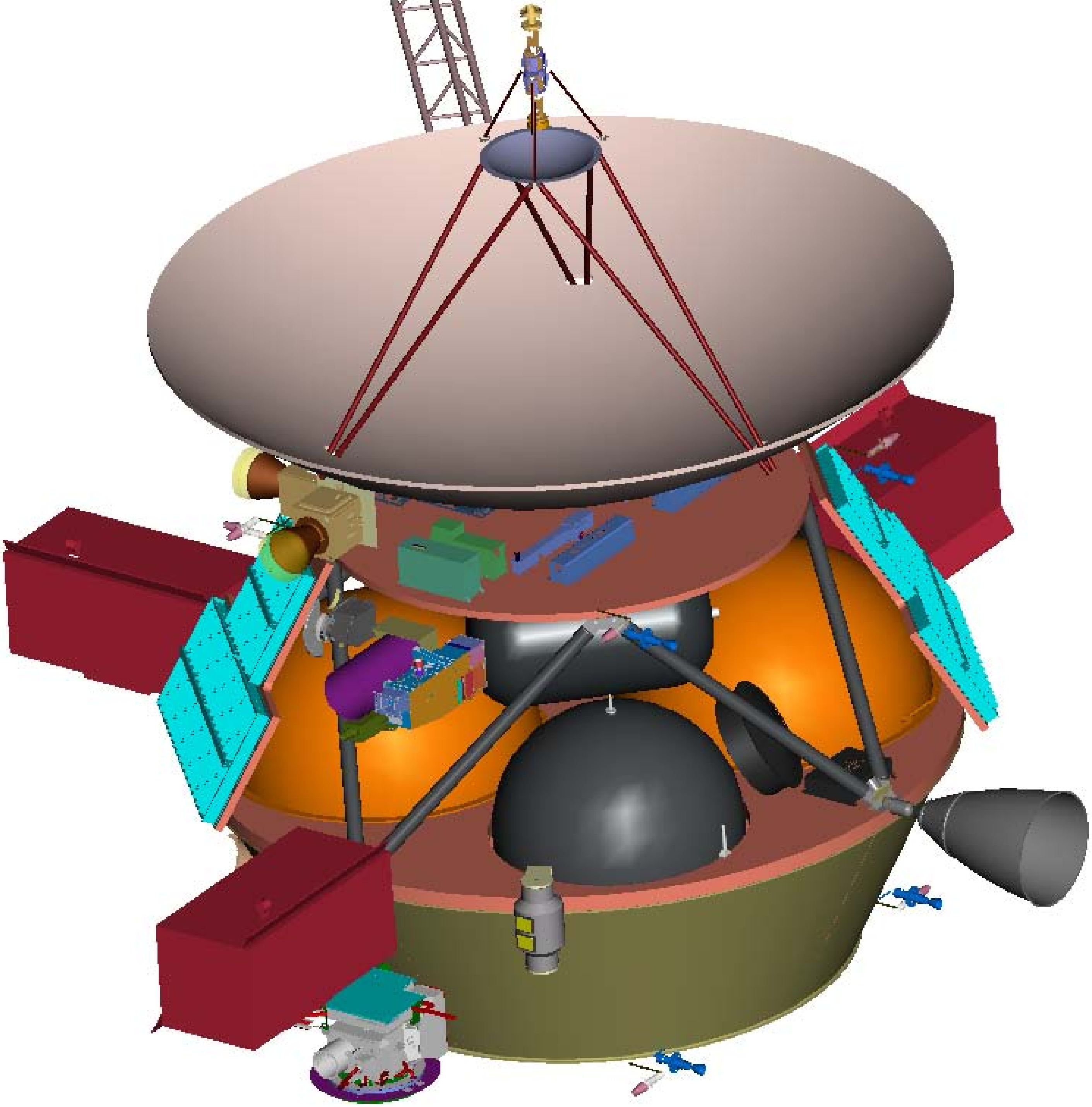
- **Spacecraft configuration**
  - Orbiter only – with a lot of mass gone from propellant tanks
- **Prime science events**
  - Larger TCMs to switch to orbits that reach out to satellite distances from planet – use bi-prop engine?
  - Brief periods of pointing high-resolution imagers at target features
  - ~8-hour daily downlink tracks pointing HGA at Earth
- **Spacecraft is 3-axis stabilized with active attitude control throughout**
  - Attitude control with thrusters for TCMs to adjust orbit, occasional momentum dumps
  - Attitude control with wheels for science
- **Pointing driver is science pointing**
  - ~0.1 deg pointing accuracy required to point HGA to Earth
  - This should be good enough for TCMs
  - Jitter requirement for 5 microrad/s for short period with high-resolution instrument
- **Agility (slew rates, time to complete) drivers**
  - No high rate tracking for science observations is expected

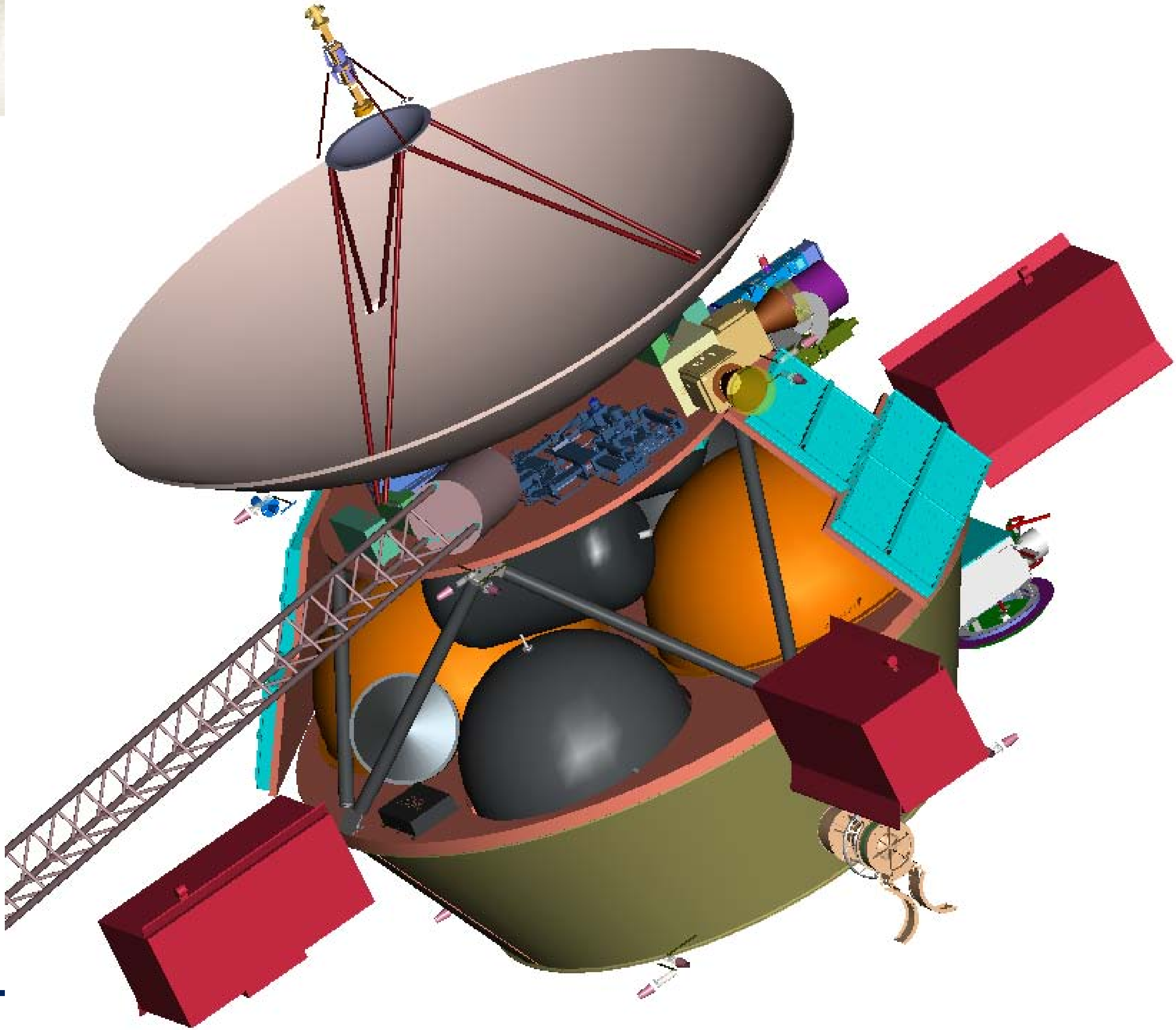
Orbit
(Initial Orbits and Satellite Tour)

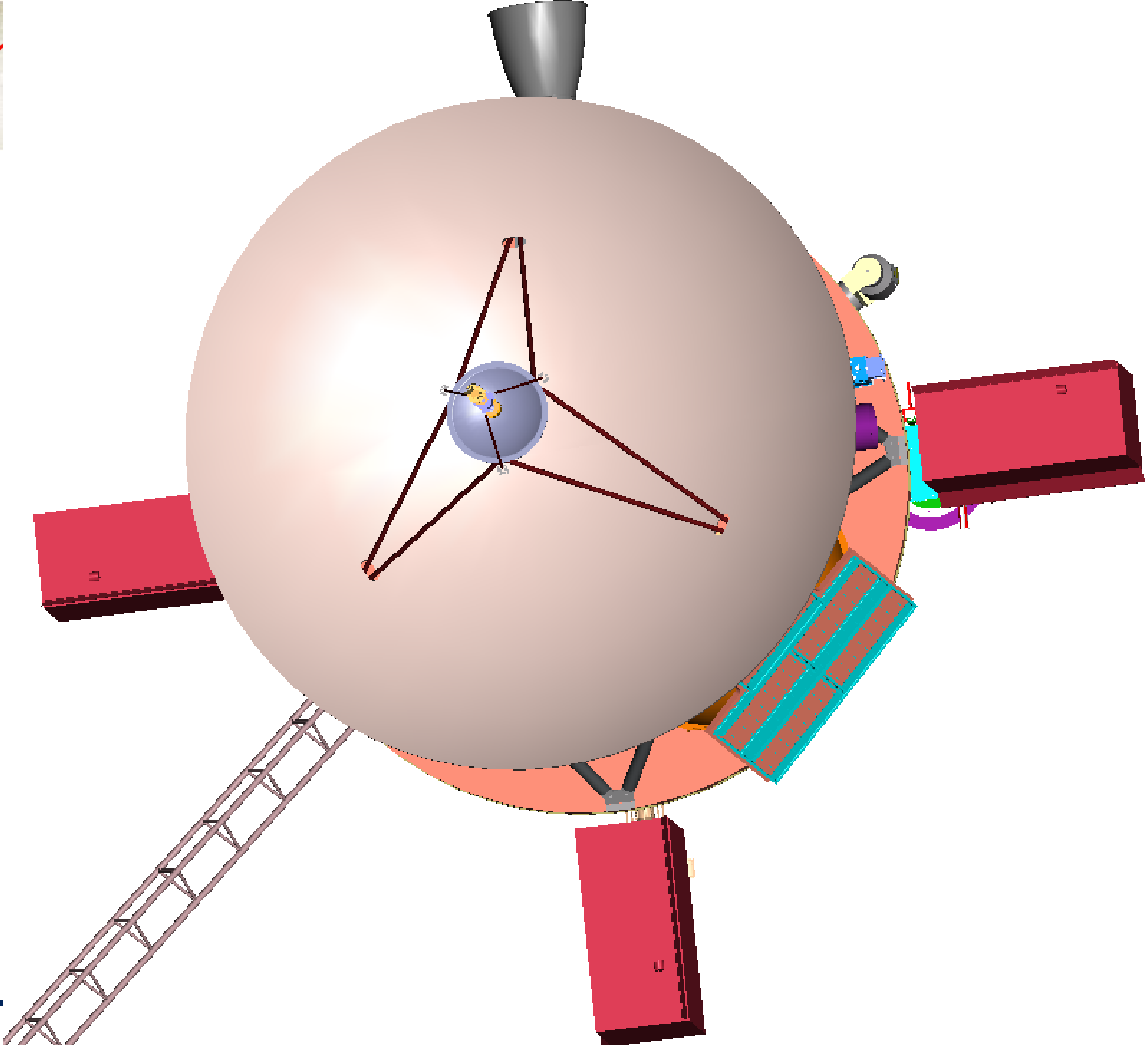
Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.).	3-axis stabilized
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	Inertial (star tracker)
Attitude control capability, degrees	0.1
Attitude knowledge limit, degrees	100 arcsec = 0.03 deg
Agility requirements (maneuvers, scanning, etc.)	TBD
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	Solar arrays on SEP stage – 1DOF, 1 axis Engines on SEP stage – 1 DOF gimbal
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	Star tracker, gyros (IMU) Thrusters during orbit adjust maneuvers, wheel momentum dumps Wheels for science pointing, data downlink to Earth

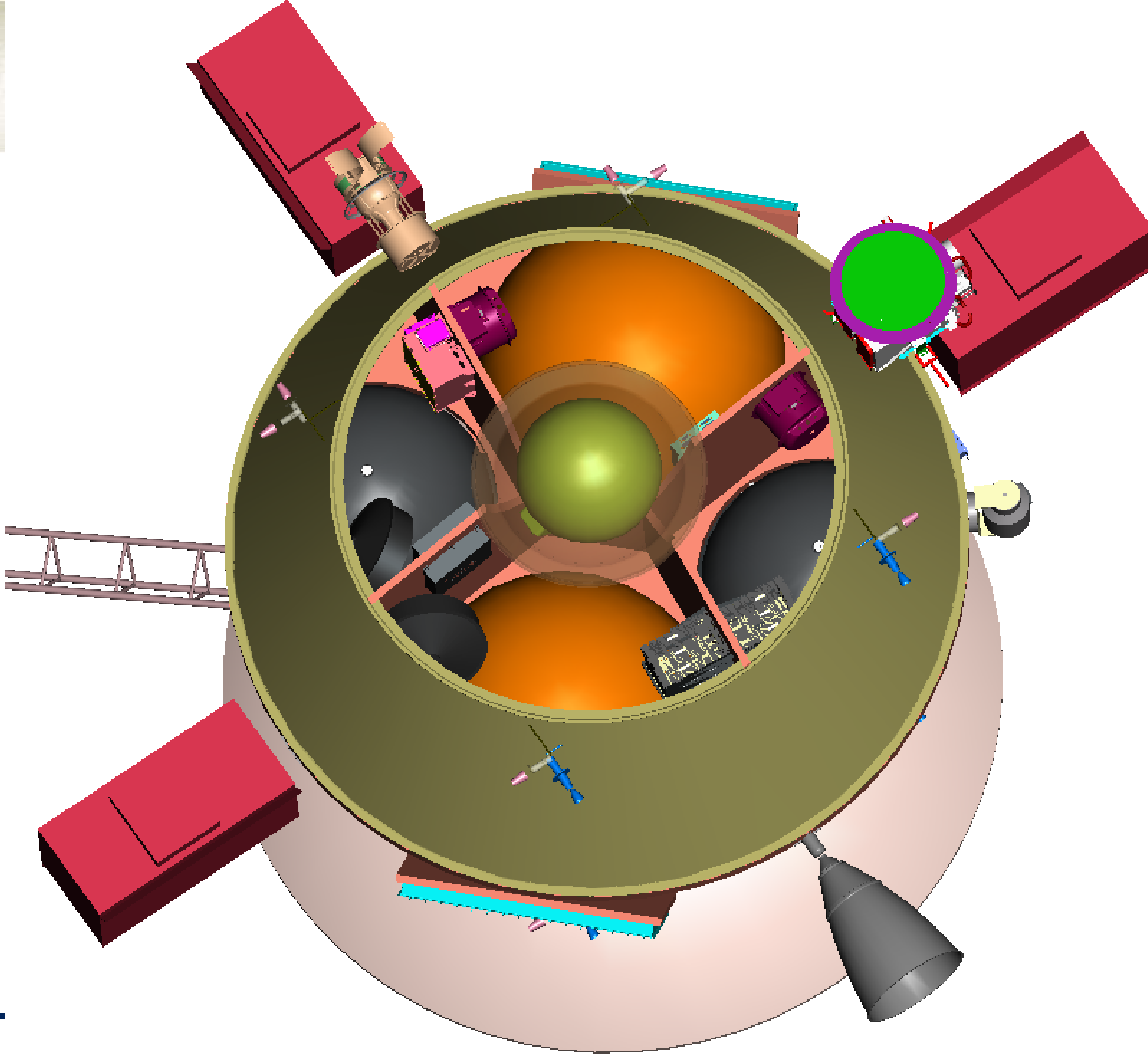












Uranus Decadal Survey ACE Run Thermal

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04/30/10

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Requirements and Assumptions

- **Maintain all component temperatures within limits for the mission duration for sun distances from 0.82 AU to 19.2 AU**

- **Maintain ASRG cold root temperatures after launch**

- **Temperature limits**

➢ S/C Bulk Temperature	10°C to 30°C
➢ Propulsion System:	10°C to 50°C
➢ Antennas (LGA,MGA,HGA):	-160°C to 160°C
➢ Star Trackers:	-20°C to 55°C
➢ ASRG (interface temperature):	0°C to 60°C

Concept Summary

- **Passive "thermos bottle" spacecraft design**
  - Spacecraft will be sealed with MLI to effectively utilize electronics box waste heat
  - Louvers will be used to keep spacecraft core temperature between 10 and 30 °C by balancing electronics waste heat and incident solar energy
  - Assumes a fairly constant electronics dissipation in the bus (~230 W)
  - Software controlled heaters are used to maintain constant dissipation
  - Heat pipes are used to spread heat throughout spacecraft bus
  - Propulsion system thermally coupled to the bus
- **Probe thermal design**
  - An aeroshell will protect the probe from aero heating
  - The outer shell of the probe will be covered in MLI
  - The probe will be pressurized with helium and maintained at 1 atm
  - Light weight foam insulates the probe in atmosphere
  - 4 RHUs are used to maintain minimum temperature during hibernation

Subsystem Characteristics

Define and fill in parameters you think appropriate. Don't just use the ones they have especially if they don't make sense.

Flight System Element Parameters (as appropriate)	Value/ Summary, units
General	
Design Life, months	
Structure	
Structures material (aluminum, exotic, composite, etc.)	
Number of articulated structures	
Number of deployed structures	
Aeroshell diameter, m	
Thermal Control	
Type of thermal control used	MLI, software controlled heaters, heat pipes, lovers, RHUs
Propulsion	
Estimated delta-V budget, m/s	
Propulsion type(s) and associated propellant(s)/oxidizer(s)	
Number of thrusters and tanks	
Specific impulse of each propulsion mode, seconds	
Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.).	
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	
Attitude control capability, degrees	
Attitude knowledge limit, degrees	
Agility requirements (maneuvers, scanning, etc.)	
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	
Command & Data Handling	
Flight Element housekeeping data rate, kbps	
Data storage capacity, Mbits	
Maximum storage record rate, kbps	
Maximum storage playback rate, kbps	
Power	
Type of array structure (rigid, flexible, body mounted, deployed, articulated)	
Array size, meters x meters	
Solar cell type (Si, GaAs, Multi-junction GaAs, concentrators)	
Expected power generation at Beginning of Life (BOL) and End of Life (EOL), watts	
On-orbit average power consumption, watts	
Battery type (NiCd, NiH, Li-ion)	
Battery storage capacity, amp-hours	

Uranus Decadal Survey ACE Run I&T

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4/30/10

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Assumptions

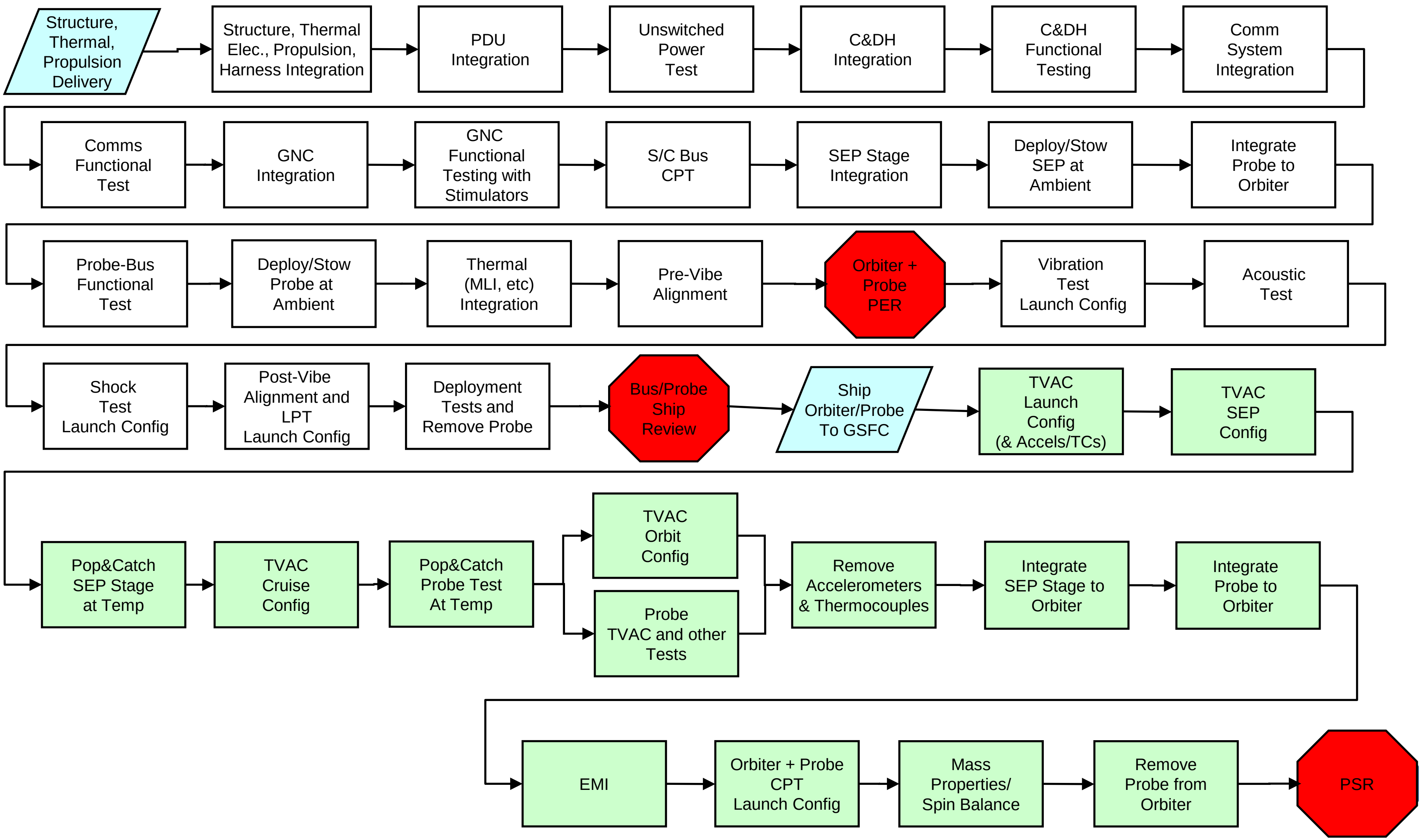
- **Assumed Environmental testing at GSFC**
- **Assumed combined Orbiter + Probe environmental testing for Launch and Cruise configurations**
  - Alternative to separate environmental testing completely
    - Pro: lessens coupling of development schedules
    - Cons: requires a probe mass/thermal model for Orbiter TVAC
- **Assumed separate Probe environmental testing for Uranus atmospheric entry**
  - Assumed Probe environmental testing will NOT be a mixed mode test but will use analysis including temperature effects which requires material testing, over temperature, to determine the property over the range of interest.
  - Would also need to do some large analog tests of the more complex parts to measure the as-delivered strength of the design.
- **Assumed SEP Stage risk reduction performed**
  - Assumed no major integration issues
- **Assumed AstroMast full deployment will be tested at ambient during mast assembly and test**
  - AstroMast deployment at temp assumed to be done in stages during mast assembly and test
- **Assumed ASRGs can be handled, fit checked and installed in parallel**

Concept Summary

- **I&T Flow**
  - Bus Integration flow modeled after MESSENGER and the VRO proposal
  - Probe Integration flow modeled after VRO proposal
  - Environmental Testing
    - Bus + Probe Environmental performed together for Launch and Cruise configurations
    - Probe Environmental performed separately for Uranus atmospheric entry
  - ASRG considerations
    - Will require 3 simulator sets for I&T; i.e., 3 each of…
      - Mass model for static and dynamic mechanical testing
      - Thermal model for TVAC testing
      - Electrical model for electrical interface and power profiles
    - Pathfinder events at least 1 year in advance of Launch Ops start
  - 18-month I&T through PSR

- **Launch Ops**
  - Launch Ops flow modeled after New Horizons
    - but with 3 ASRGs
  - 3.25-month Launch Ops
    - includes handling, fit check and install of 3 ASRGs

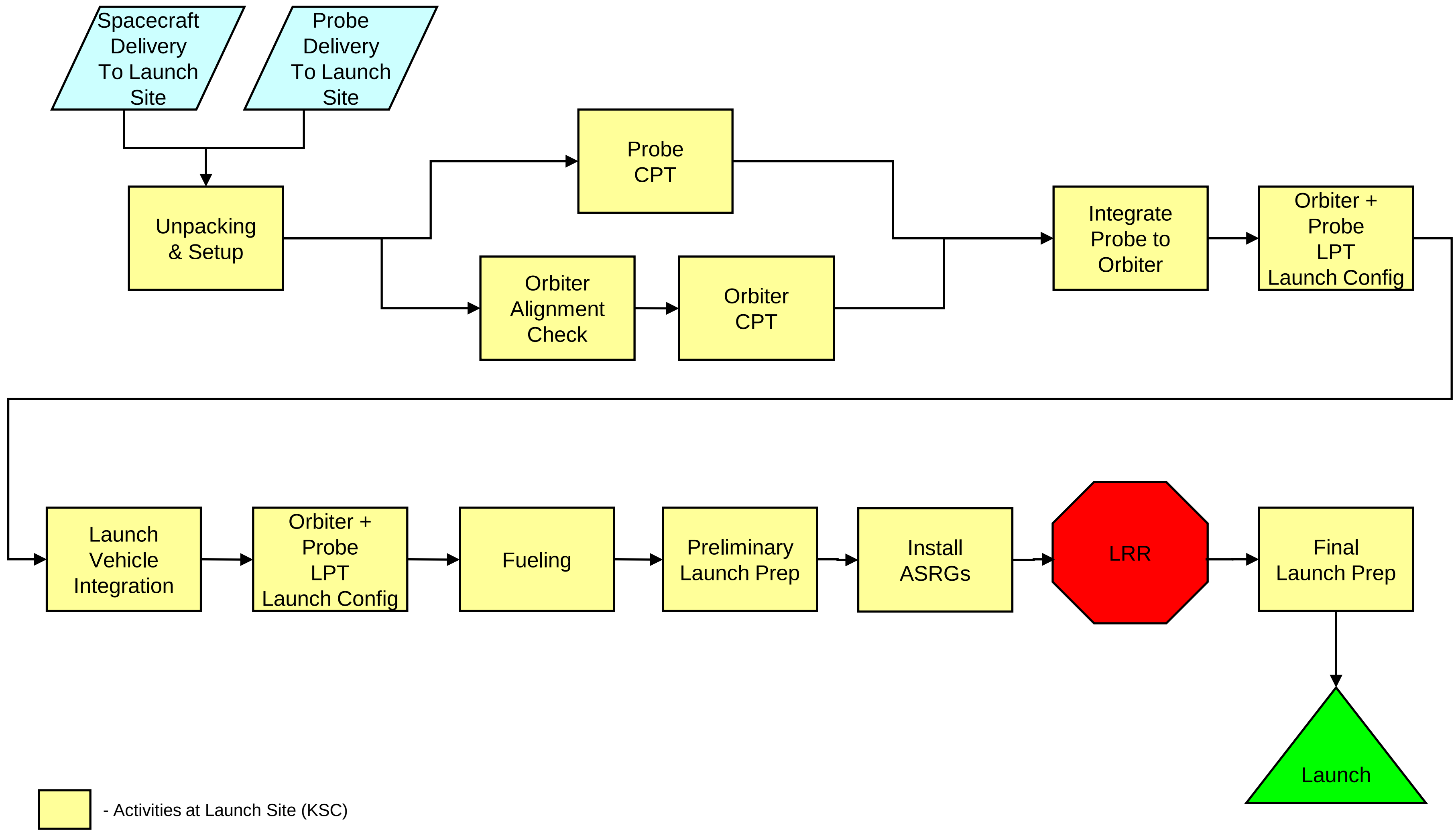
I&T Flow



- Activities at APL
- Activities at GSFC

APL

Launch Ops Flow



Orbiter Simulators/Models/Testbeds

- **Orbiter**
  - Mechanical
    - ASRG Static mass model (Qty 3)
    - ASRG Dynamic mass model (Qty 3)
  - Thermal
    - ASRG thermal model (Qty 3)
  - Electrical/RF
    - Power, discretes, telemetry, data and RF simulators
    - ASRG electrical simulator for interface checkout and tests with high fidelity flight power
  - Testbeds
    - MiniMOC testbed with IEM EM (Qty 5)
    - MiniMOC testbed only [i.e., without IEM EM] (Qty 2)

Probe Simulators/Models/Testbeds

- **Probe**
  - Mechanical
    - Probe static mass model
    - Probe dynamic mass model
  - Thermal
    - Probe thermal model
  - Electrical/RF
    - Power, discretes, telemetry, data and RF simulators
  - Testbeds
    - Testbed with Probe IEM EM (Qty 2)
    - Testbed only [i.e., without Probe IEM EM] (Qty 2)

Costing Input

- **Uranus Decadal I&T vs VRO I&T**
  - I&T effort slightly more than VRO I&T
    - 12% increase over VRO I&T schedule/cost
    - Fewer deployables than VRO (e.g., no MMR)
    - Assumes SEP stage integration is not an issue, but adds schedule time & risk
    - Multiple ASRGs on Uranus Decadal
    - Both missions require environmental outside APL (i.e., GSFC assumed)

- **Uranus Decadal Launch Ops vs ILN Launch Ops**
  - Launch Ops longer duration and more schedule coordination than ILN 2-lander scenario
    - 12% increase over ILN Launch Ops schedule/cost
    - Multiple ASRGs on Uranus Decadal need more time for…
      - Preparation
      - Mechanical fit check
      - Install

Butler R. P., S. S. Vogt, G. W. Marcy, D. A. Fischer, J. T. Wright, G. W. Henry, G. Laughlin, and J. J. Lissauer, "A Neptune-Mass Planet Orbiting the Nearby M Dwarf GJ 436," *The Astrophys. J.*, Vol. 617, No. 1, 2004, pp. 580–588.

de Pater, I., H. B. Hammel, S. G. Gibbard, and M. R. Showalter, "New Dust Belts of Uranus: One Ring, Two Ring, Red Ring, Blue Ring," *Science*, Vol. 312, No. 5770, 2006, pp. 92–94.

Dorfman, S. D., and C. M. Meredith, "The Pioneer Venus Spacecraft Program," *Acta Astronaut.*, Vol. 7, 1975, pp. 773–795.

French, R. G., P. D. Nicholson, C. C. Porco, and E. A. Marouf, "Dynamics and Structure of the Uranian Rings," in *Uranus*, J. T. Bergstralh, E. D. Miner, and M. S. Matthews (eds.), University of Arizona Press, Tucson, AZ, 1991, pp. 327–409.

Guillot, T., "The Interiors of Giant Planets: Models and Outstanding Questions," *Annu. Rev. Earth Planet. Sci.*, Vol. 33, 2005, pp. 493–530.

Hersant, F., D. Gautier, and J. I. Lunine, "Enrichment in Volatiles in the Giant Planets of the Solar System," *Planet. Space Sci.*, Vol. 52, No. 7, 2004, pp. 623–641.

Hussmann, H., F. Sohl, and T. Spohn, "Subsurface Oceans and Deep Interiors of Medium-Sized Outer Planet Satellites and Large Trans-Neptunian Objects," *Icarus*, Vol. 185, No. 1, 2006, pp. 258–273.

Ingersoll, A. P., and T. R. Spilker, "Study of a Neptune Orbiter with Probes Mission," Response to NASA NRA-03-OSS-01-VM, 2 May 2006.

Lissauer, J., and D. Stevenson, *Formation of Giant Planets, Protostars and Planets V*, B. Reipurth, D. Jewitt, and K. Keil (eds.), University of Arizona Press, Tucson, 2007, pp. 591–606.

Lockwood, M. K., *Neptune Aerocapture System Analysis*, NASA Langley Research Center, Hampton, VA 23681, 2004

Marley, M., *Planetary Science Decadal Survey JPL Rapid Mission Architecture Neptune-Triton-KBO Study Final Report,* NASA, Feb. 2010.

Showalter, M. R., and J. J. Lissauer, "The Second Ring-Moon System of Uranus: Discovery and Dynamics," *Science*, Vol. 311, No. 5763, 2006, pp. 973–977.

Spilker, T., and M. M. Munk, "Aerocapture Technology," Decadal Survey Steering Committee Meeting Presentation, 22 Feb. 2010.

Tauber, M., et al., "Uranus and Neptune Atmospheric-Entry Probe Study," *J. Spacecr. Rockets*, Vol. 31, No. 5, Sept.–Oct. 1994, pp. 799–805.