Low-Cost SmallSats to Explore the Outer Solar System

Final Report on NIAC task to define an architecture for Outer Solar System SmallSats (OS4)

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I. INTRODUCTION & EXECUTIVE SUMMARY

Overview: *New Horizons*, *Voyager 1 & 2*, and *Pioneer 10 & 11* are the only missions to venture beyond Saturn's orbit. Each had a spacecraft mass >250 kg (some >>250 kg), mission cost >FY20$500 M (most >>$500 M,[1] and required operations teams with 10s of people. All required radioisotope power to operate at Jupiter and beyond. We propose a completely different approach for focused solar-powered science investigations from approximately the orbits of Jupiter to Neptune, and potentially farther beyond, without need for the complexity of using radioisotope power systems (RPS).

Our investigation has shown the feasibility at TRL2 for SmallSats within the family of Evolved expendable launch vehicle Secondary Payload Adapter (ESPA)-compatible spacecraft to operate as far from the Sun as the orbit of Neptune. Operation to perhaps twice that solar distance may be possible with an upgraded thermal insulation approach. Beyond there, spacecraft using similar techniques, but with solar power collection surfaces larger than can be foreseeably accommodated within the "SmallSat" framework, may be capable of making focused heliophysics measurements to and beyond the Heliopause.

Based on the results of our NIAC Phase I work, we outline a new architecture that could enable many outer Solar System SmallSat (OS4) mission concepts at 1/10th the cost and mass, and 1% of the equivalent continuous power level and operations staffing of such missions today. Inspired by the CubeSat revolution in small, low power electronics and miniature instruments, it appears that a small enough mass and launch size is achievable such that these outer Solar System (OSS) explorers could be launched as secondary payloads along with primary missions to the OSS, such as those to Europa and farther (e.g., New Frontiers round 6 or 7 selections), and use Jupiter swingbys to target different destinations.

The objective of our investigation was "to a) define an architecture; b) produce one existence-proof-level conceptual design to show the feasibility of low-cost OSS spacecraft; c) define a Reference Mission consistent with TRL2-3 to use multiple copies of such a spacecraft to perform Solar System-scale 3D mapping of heliophysics parameters; and d) define an enabling technology maturation path to achieve such a mission." We added an objective to e) consider how such an architecture could be applied to collecting relevant science data at typical destinations closer to the Sun than the heliopause, such as outer planets, planetary satellites, and small bodies.[2]

While we show in this report many elements of an "existence proof-level" design for an example mission, we note that this design is not complete, and far from optimal. We believe that no laws of physics need be violated, and that all required technology improvements are achievable over the next decade within the limits of modest expenditure, and/or their achievement will be motivated by uses and rationale beyond enabling Outer Solar System SmallSats. JPL's Team Xc is noted as a co-author, and while their involvement was essential to reaching the level of definition described here, the overall design and many of its elements are not endorsed by Team Xc. Their experience base and typical application for their products involve concepts for possible implementation much closer to the present than is typical of a NIAC investigation. We are grateful for their support.

Where a specific vendor's product is shown, cited, or referred to, this is intended to contribute to the veracity of an "existence proof," but is not intended as an endorsement of that product or vendor, nor as a source selection for any future mission implementation.

Innovation and System Elements

The following elements were described in the proposal, and where noted by crossout (for deleted language [~~crossout~~]) and underline (for added language [underline]) type, were changed during the course of the Phase I study.

1. A system architecture, including the innovative elements below, utilizing…
2. 5 ~~~3 – 8~~ m, inflatable and UV-rigidizable Paraboloid ~~Sphere~~, e.g., of clear Kapton[tm], metalized on one surface~~~1/3 – 2/3 of its anti-Sun extent,~~ to direct concentrated sunlight and rf energy at its ~~diffuse "aberrated~~ focus,~~"~~ making it a multi-functional structure.
3. RF- ~~Visible~~-transparent visible Diffuser (Fresnel Element) ~~(~75%)[3] rf subreflector~~ before the ~~aberrated~~ focus, shaped to diffuse visible wavelengths to spread over a circular power-producing solar panel, while not deflecting ~~direct~~ rf wavelengths, ~~and~~ from going into (for receiving commands) or out of (for downlinking data) a feed horn to telecommunications equipment. The concentrated sunlight absorbed by solar cells provides electricity and heat, replacing the need for RPS, which have been used on all prior spacecraft going beyond Jupiter (~5 AU).
4. Spider structure to position the primary Bus ("Hub") subsystem package ~~inside~~ outside the ~~Sphere~~ Paraboloid ~~at~~ near the ~~aberrated~~ focus, and maintain constant relative position of all spacecraft elements during slow attitude maneuvers, and small trajectory trim maneuvers.
5. Thermally-accommodating Bus package (Hub) that, in combination with attitude maneuvers to spin slowly about its polar axis, and point off-Sun to modulate heat input when <~~10+~~ ~6+ AU, enables subsystems to operate within thermal limits at all solar distances ~~up to 125 AU~~ from 1–30AU. Phase change materials would recover the waste energy during science/communications operations to augment the thermal state of the spacecraft during sleep/standby mode.
6. Low-power avionics capable of performing needed s/c functions with a very limited duty cycle. Low-intensity/low-temperature (LILT) solar cells covering a surface defined by the ~~aberrated focus~~ Fresnel Element diffuser generate electricity continuously, charging batteries used to operate s/c subsystems when "awake" in the OSS at duty cycles <~1%).
7. Data rate ~300 (at 30 AU Neptune-like distance) – >10,000 (at 5AU Jupiter-like distance) ~~<10-20~~ bps X- ~~Ka-~~band telecommunications, activated briefly at preset times with modest rf power (20 ~~10s of~~ W) using the metalized ~~spherical~~ paraboloid surface in #2 as high gain antenna (HGA).
8. Largely autonomous operations requiring <~1 person-day/month averaged over mission lifetime, after deployment and Jupiter flyby.
9. Packaging as ESPA-~~port-~~compatible s/c for deployment as secondary payload from Jupiter-bound primary mission launches. A large (~200 - 1000 m/sec) deep space maneuver capability plus a small (10s m/sec) TCM capability is to enable targeting a Jupiter or other outer planet flyby to point the trajectory toward different sectors of the heliopause, and/or to a specific planet, asteroid, Centaur object or other small bodies, including ~~tweak~~ to pass Trans-Neptunian

Object (TNO; also known as Leonard-Edgeworth Disk, aka Kuiper Disk)[4] targets-of-opportunity.

10. Small planetary and small body-relevant instruments, plus heliophysics instruments for measuring magnetic field, plasma, energetic ions, dust, and perhaps other ~~heliophysics~~ parameters.

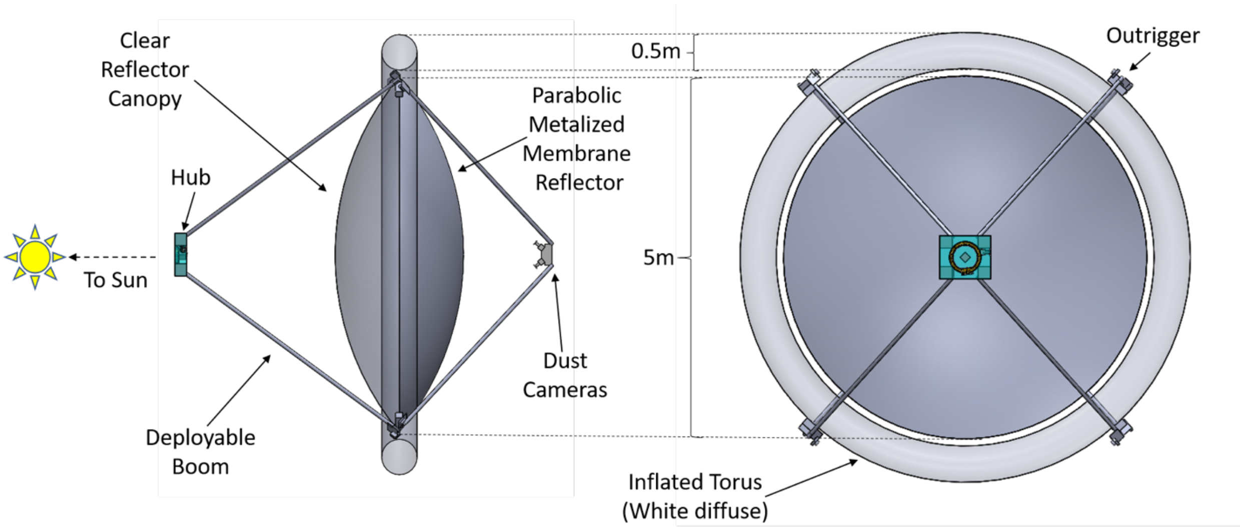


Figure 1. Outer Solar System SmallSat (OS4) cross-section (left) of 5 m diameter paraboloid reflector and clear canopy, suspended within a Torus providing tension for structural stability. A Hub, or bus is located so that solar cells cover an area near the focus of the Parabolic Metalized Membrane Reflector to intercept nearly all of the sunlight beyond ~6 AU. From 1 – ~6 AU, the spin axis is pointed off-Sun to reduce heat loading. Beyond 6 AU the spin axis ("To Sun" in left figure) points to the Sun. Periods of Earth pointing enable telecommunications. RF waves pass through a transparent Fresnel Element between the Hub and the Clear Reflector Canopy (shown in Figure 18) that uses refraction to diffuse visible wavelengths onto a concentrated solar panel on the anti-Sun side of the Hub. Attitude/TCM thrusters are in Outriggers on the ends of the four Deployable Booms. Right side is view from sunward side along spin axis.

Order-of-magnitude Characterizations Indicating Feasibility[5]

Energy: If one assumes a 5-m parabolic reflector diameter, then electrical output at 30 AU is 4 W, assuming LILT solar cells[6] covering a concentrated area of 0.3 m^2 could operate at adequate temperature, with the following efficiencies: 85% concentrator; 25% end-of-life (eol) solar cells; 92.5% packing factor; 92.5% incidence angle effects of light reflected onto cells, 80% for 2x passage through clear canopy, and 80% remaining after visible light blockage by an opaque rf feedhorn and waveguide.

Telecommunications: Using standard DSN parameters and link margin of 3 dB, 5 m diameter s/c reflective surface aperture, X-band to 34 m DSN station yields 3 kbps at 30 AU for 20 W rf transmit power. Scheduled transmissions can be performed at a low duty cycle from a few hours per week to 1 hr/month utilizing energy stored in trickle-charged batteries during "sleep" time. For heliophysics-focused missions with no specific destination, a downlink-only mission appears viable after navigating to a Jupiter swingby that targets escape direction to a particular sector of the outer Solar System. When a specific Solar System body destination is a mission focus, then a standard uplink/downlink mission is envisaged to refine observation parameters up to encounter, and ensure return of the encounter data load. With more risk but lower operations cost, one could imagine setting all those parameters years ahead of encounter, having onboard sensing processed onboard to adjust encounter time (and maybe even make trajectory corrections), leaving us to wait with anticipation for whatever post-encounter data stream comes back.

Structure: Inflatable structures have been flown in space since the 30-meter diameter, ~70 kg Echo 1 built for NASA in 1960.[7] Recent work on inflatable high gain antennas has been performed at JPL and ASU, testing inflation of parabolic antennae.[8] UV rigidization has been explored in this and prior work, so that pressure need not be retained.[9] Ribs built into gores of a paraboloid on the ground can be filled with a liquid that rigidizes in solar UV. While our original intent was to utilize an inflated sphere for manufacturing ease, two factors guided our choice of the inflatable paraboloid. The first was realizing that with the spherical concentrator/reflector we could lose as much as 50% of the intercepted sunlight with a practical configuration to get it to the solar panel on the Hub. The second was learning more about the Spartan 207 technology demonstration deployed in 1997 from the Space Shuttle, and the apparent manufacturability thereby demonstrated creating an accurate-enough paraboloidal surface.[10]

Autonomy: After launch and Jupiter flyby, mission operations could be vastly simplified. With few modes, and "smart" instrumentation, some subsystems will be "asleep" to save power >90% of the time. On Solar System escape trajectories, attitude changes required to maintain Earth pointing are only fractions of a degree per day. Such small adjustments require minimal propellant expenditure. Rather than downlinking raw data, instrument data could be processed onboard while raw data is stored, with the possibility of downlinking selected very small segments of raw data in order to verify onboard processing techniques. Full autonomy of this operations concept is well within flight proven (TRL 9) technology such as the methhods used to operate the Earth Observing One (EO-1) mission for over a dozen years as the Autonomous Sciencecraft Experiment.[11] Beyond the basic operations concept, we note that AI development has proceeded dramatically, and will be drawn on as appropriate to move traditional ground functions onboard in ways that would benefit operations costs for any mission.

Launch availability: Any mission going to the Jovian system as either a destination or trajectory waypoint is likely to have 100s of kg of launch mass margin, which could accommodate one or a few OS4s as secondary payloads. Deployed using an ESPA ring or similar arrangement, OS4s need not ride aboard the primary mission spacecraft, but could instead launch the way MarCO A & B used InSIGHT's upper stage for their launch to Mars.

Instrumentation: Highly sensitive magnetometers and plasma instruments are in development <1U in volume, with chip-scale instruments on the horizon. A magnetometer being developed at JPL was incorporated into the example mission design.[12] Two charged particle instruments from SwRI. were also incorporated into the example mission design for heliophysics measurements: a) the Solar Wind Ion & Electron Sensor (IES) based on the *Rosetta* instrument, and the Miniaturized Electron and Ion Telescope (MeRIT) for higher-energy charged particles. One dust-measuring instrument concept was incorporated into the example design, taking advantage of the large cross section of the Parabolic Metalized Membrane Reflector (PMMR). It employs a mini-camera on booms on the anti-Sun side of the PMMR looking back toward the Sun, autonomously measuring new holes in a segment of the PMMR, backlit by sunlight. A second concept might be further considered employing a micro radio receiver measuring rf discharge from plasma puffs created by micrometeoroid impacts.[13] Feasibility of other instrumentation, e.g., for galactic cosmic rays (GCR) and anomalous cosmic rays (ACR), could be considered.[14] One possibility includes monitoring onboard memory and other electronics to detect, as well as quantify energy and direction, of cosmic ray hits. Finally, a Solar System Portrait Camera was incorporated more for the artistic, educational, and recording-for-posterity value than its likely scientific yield. The example used for this and the Dust Camera was based on the current Mars 2020 Enhanced Engineering Cameras produced at JPL. Future advancements of all relevant instrumentation is considered likely, but these formed an existence proof set for the purpose of this investigation.

<u>Lifetime</u>: Considering what is perhaps the biggest challenge, the *Voyagers* and *Pioneers* demonstrate that multi-decade lifetimes are possible. Several generations of electronics later, it is worth noting that some modern CMOS electronics has proven spaceworthy and rad-tolerant at relevant levels, especially deeply-scaled CMOS technology (smaller than 65 nm) where the surface quality of nanometer transistors creates high robustness to total ionized dose (TID) through fewer trapped charges, and better single event rejection from a much-reduced transistor cross section area.[15] Additionally, on-chip background calibration techniques allow for remaining radiation effects on analog circuitry to be digitally corrected.[16] Emerging CMOS system-on-chip (SoC) technology is unlike any other electronics technology and has the ability to co-integrate digital, analog, mixed-signal (ADC/DAC) memory and rf circuitry all together on a single chip. This integration capability of CMOS has transformed our world by enabling the markets for personal computing, gaming, and smartphones. This capability provides a pathway to build all spaceborne electronics as a single system on a chip (SoC), including telecom, power handling, thermal/attitude control, C&DH, instrument data, and everything else. As demonstrated on MarCO, watchdog timers, firmware and software can automatically clear some faults and errors. SoC's size enables redundancy and voting logic in a still-tiny package. While we would expect to employ higher-quality electronics, it is noteworthy that one of the three very first CubeSats launched to orbit (in 2003), *XI-IV*, built at University of Tokyo, was still functioning in LEO as of August 2018. Dust, solar UV, and other environmental effects on the PMMR's and Clear Canopy's material will also be considered in a future phase beyond what was done for Phase I.[17]

In the case of missions envisaged to the orbit of Neptune, mission times of 10-15 years are likely to be more the norm. Therefore, while recognizing the likely advancements cited above, our example mission employs a set of electronics available and in development today that has sufficient capability, with dramatically lower power draw than typically used in spacecraft today.

Potential Impact: We did not investigate the cost of our example mission. However, applying the SmallSat paradigm and typical cost-per-kg values, a target cost of FY2020$50 M each for multiple units to be launched ~2030 appears reasonable. If that were to prove true, then significant numbers of OS4s (e.g., 6 to 12, spread among different launch opportunities) could be launched onto multiple escape trajectories to provide a truly multidirectional view of the interplanetary medium, and to fly by and take measurements at a selection of outer Solar System destinations. Within the 30 AU heliocentric distance capability of the probes, measurements by each probe, coupled with those taken in the inner Solar System from other platforms, could go a long way toward completing our understanding of how the solar wind propagates in all directions, throughout the solar cycle, by comparing the measurements from different probes at similar distances in different directions from the Sun, adding to measurements we have from missions to date. With a minimum set of six OS4s targeted to Solar System escape in different directions spaced around the heliosphere, the magnetic field, plasma, and dust environment could be mapped across the volume of our Solar System to a distance represented by the orbit of Neptune, and potentially beyond.

II. DESIGN DISCUSSION AND DETAILS

System Architecture & Overview

Our driving requirement was to make and return to Earth useful scientific measurements of interest from the outer Solar System.

The most fundamental parameter of the proposed architecture is the heliocentric distance capability. Choosing this parameter sets the environmental constraints of available solar power, thermal control, and telecommunications, which in turn drive configuration, size, and operations concept. Thus this section begins with our rationale for specifying a 30 AU heliocentric distance. This capability is dramatically beyond that of any SmallSat seriously considered to date lacking radioisotope power. A later subsection describes some possible methods of increasing heliocentric distance capability, potentially as far as interstellar space, just beyond the heliopause, to 125 AU.

Deciding Heliocentric Distance Capability

Our original aspiration was to define an architecture that would enable useful scientific measurements beyond the heliopause (up to 125 AU from the Sun). During the course of our Team Xc session in 2020 January, it became clear that in order to maintain internal spacecraft electronics temperatures at or above -40 C would require almost 4 W of electrical power expended inside our "Hub" (see configuration and nomenclature below), insulated by Dual-stack multi-layer insulation (MLI). Thus, for the size spacecraft we envisage as suitable being tuna-can shaped and 40 cm in diameter, with electronics that require -40 C or warmer operating temperature, our heliocentric distance is limited to ~30 AU, where a 5-meter parabolic reflector can intercept and solar cells convert intercepted sunlight to ~4 W total electrical power that can be dissipated by electronics and/or heaters inside the insulated Hub.

Configuration, Structure and Nomenclature

The driving environmental factor that influences design is the availability of solar power, diminishing as the inverse square of heliocentric distance. Our configuration and operations concept are seen as one solution to this intersection of science requirements and the environment where we wish to gather new data over long time periods and large distance scales. There may be other solutions, but this is our starting point. Many of the details were developed during a JPL Team Xc design session that was an integral part of this investigation.

Capturing solar power requires a collecting area, and at the same time, telecommunications requires a high gain antenna to achieve useful data rates. Thus, we have combined these two functions utilizing a single 5 m diameter Parabolic Metalized Membrane Reflector (PMMR). This reflector would be manufactured on the ground, and deployed by inflation using a technique demonstrated on the 1997 Spartan 207 test deployed from the Space Shuttle for a 14 m reflector (Figs 2, 3 & 4).[18] The hardware was built at L'Garde, Inc., in Tustin, California, in a task managed by NASA/JPL.[19] A ground test version, pictured in Fig 4, happens to be approximately the 5-meter aperture size we have utilized in our example mission concept. While a partial success, we consider this demonstration to provide sufficient proof-of-concept as a jumping off point for our design.

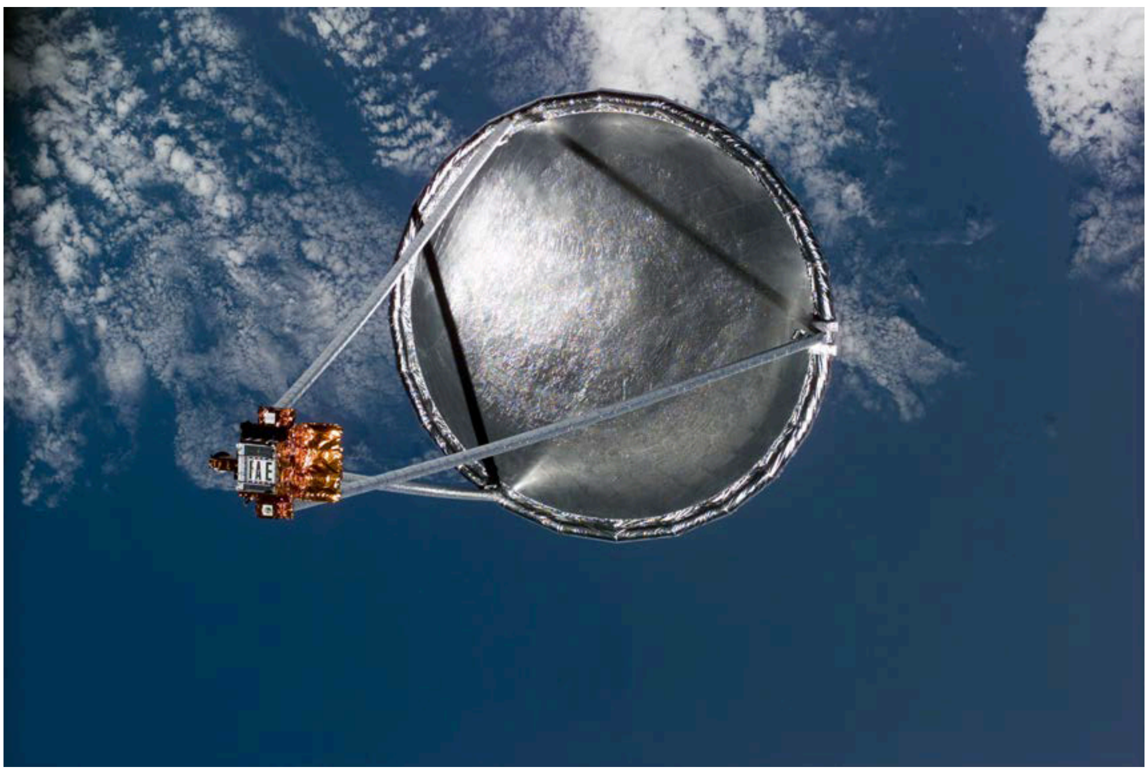


Figure 2. Spartan 207 14-meter diameter paraboloidal reflector deployment was a partial success in a 1997 technology demonstration carried out from the Space Shuttle. NASA photo.

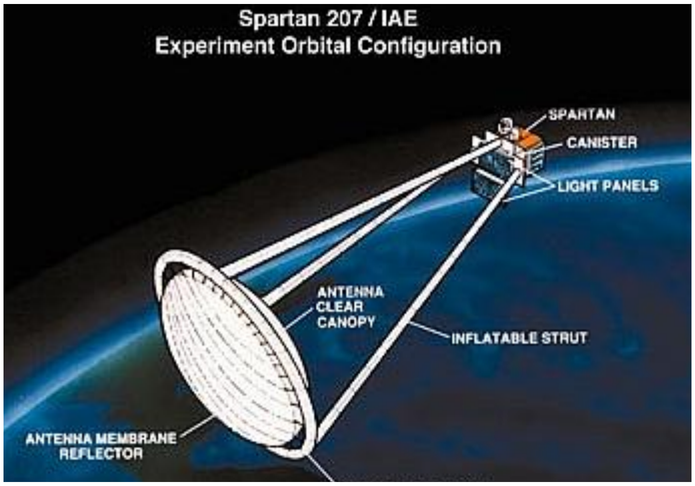


Figure 3. Unlike Spartan 207's inflatable struts, the example OS4 architecture would use Deployable Booms. OS4 would use the same arrangement of an inflatable Torus to provide tension around the perimeter of the Parabolic Metalized Membrane Reflector (PMMR) and Clear Canopy. NASA/JPL/L'Garde artist's illustration.

Figure 4. This sub-scale ground version of the Spartan-207 hardware is approximately the 5-meter diameter specified for the OS4 concept. NASA/JPL/L'Garde photo.

The Hub is the primary science, communications, power, and avionics module. It is insulated with MLI blankets and thermal switches to reduce the total heat loss of the system in the outer Solar System. It also houses attitude sensors. The Hub, schematically depicted in Figure 5, would have instruments and a low-gain antenna on its sunward and cylindrical faces, while the anti-Sun face faces the PMMR. Solar panels would wrap the outer cylinder surface of the Hub to provide electrical power over a wide orientation range during early maneuvers, deployment, and post-deployment of the Inflated Assembly.

Figure 1 depicts the configuration after inflation and deployments, as in the outer Solar System, with spin axis pointed at the Sun. the "Inflated Assembly" is defined as the Parabolic Metalized Membrane Reflector + Clear Reflector Canopy + Torus + Tensioning Strings. Before deployment, they are contained in the Inflatables Launch Package, which is jettisoned upon inflation/deployment.

Small pulsed plasma thrusters (PPT) would be used for attitude control as well as minor trajectory corrections. They do not require external propellant plumbing, valving, or storage, because they contain solid Teflon fuel that is held in the required position by springs. Four thruster modules with two orthogonal thrusters each would be mounted around the perimeter of the Torus at the end of each boom on the Outriggers. This allows for the greatest moment generated by the thrusters and the least propellant to be used for the needed attitude maneuvers.

Radiator(s) are used in the inner solar system to reject waste heat from the spacecraft. Because there is so much more sunlight per unit area in the inner in Solar System, radiator(s) are required to operate the spacecraft sunward of ~6 AU.

The Outer Solar System solar panel (OS3P) is estimated to be 40 cm in diameter, populated with low-intensity low-temperature (LILT) solar cells mounted on the anti-Sun side of the Hub. It would face the solar collector (PMMR) from 0.5 m behind the focus. Sunlight would be collected from reflection off the PMMR, after which this light passes through the Fresnel Element, which spreads the light collected more evenly over the OS3P array. The solar array is separated and insulated from the Hub because although it receives sunlight, it is colder than the Hub in the outer Solar System, and therefore would act as a radiator when this is not desirable.

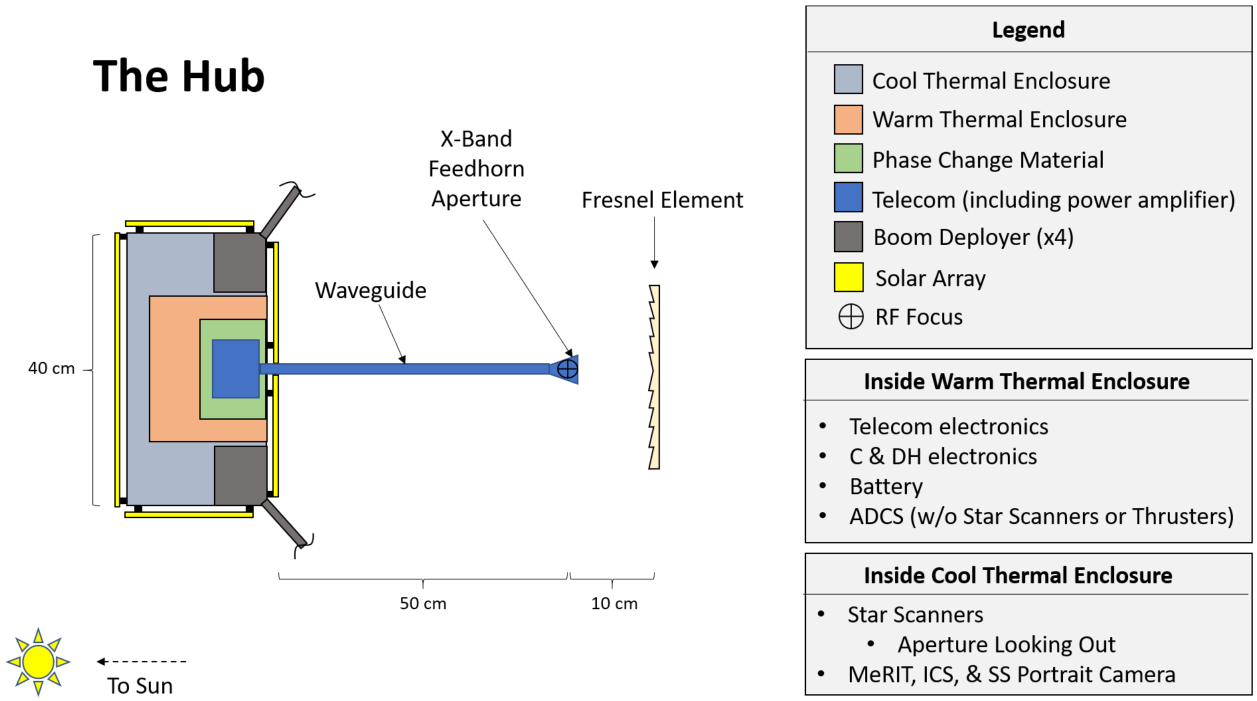


Figure 5. The Hub contains most spacecraft electronics in a tuna-can shaped cylinder 40 cm in diameter and ~20 cm long. The Parabolic Metalized Membrane Reflector (PMMR) is not shown to the right; its focus is at the X-band Feedhorn aperture. The side of the Hub facing the PMMR (facing to the right) is covered by the Outer Solar System Solar Panel (OS3P) depicted here attached to the Hub body as a thin yellow vertical panel, which converts sunlight, concentrated onto the array from the PMMR and diffused through the Fresnel Element, into electricity beyond 6 AU from the Sun. The Inner Solar System Solar Panel (IS3P) consists of the other thin yellow panels depicted on the Hub exterior, which provide power up to 6 AU heliocentric distance, when the PMMR is not pointed directly at the Sun, to avoid overheating the OS3P. In the inner Solar System, a Radiator (not shown) is to reject heat from a portion of the cylindrical surface. As temperatures drop in the thermal enclosures, Reverse Thermal Switches disconnect the heat path to the Radiator, conserving heat inside the Hub. See also System Block Diagram.

During launch and until separation from the propulsive ESPA ring kick-stage (~T+30 - 60 days), the OS4 would remain in its stowed configuration. While stowed, the inflatable assembly would remain compressed within its cover and the booms on their spools before deployment. This will allow the spacecraft to remain rigid during primary maneuvers as well as to fit into the launch vehicle fairing.

System Block Diagram

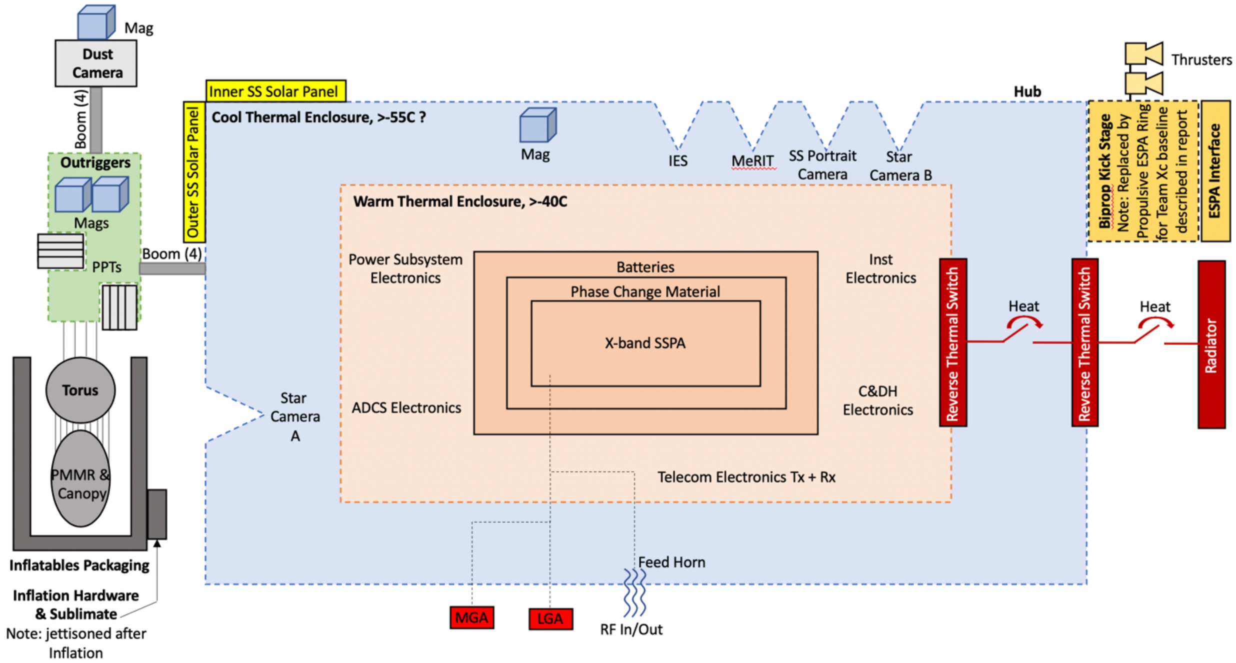


Figure 6. Schematic view of mechanical interfaces among the OS4 elements, which are not to scale. Instrument and sensor apertures from the outside of the Hub into the Cool Thermal Enclosure are shown as dashed V's. In this depiction, a Biprop Kick Stage is shown in a dashed box near the upper right. For the Team Xc analysis described in this Phase 1 report, this Stage's function was replaced by a Propulsive ESPA Ring with capability as an independent spacecraft, carrying the OS4 as a payload released after the 200 – 1000 m/sec deep space maneuver 30 – 60 days into the mission. In a later phase, a trade study could be performed to determine which arrangement is better suited to a specific opportunity. The Propulsive ESPA Ring might be a better solution in a situation where two or more OS4s could be carried as secondary payloads on a single primary mission launch.

Thermal Control

Several key components are necessary to handle the thermal environment that changes dramatically from 1 AU to 30 or more AU. Reverse operation thermal switches[20] would be used to connect or disconnect the thermal path from equipment inside the Hub to the Radiator assemblies on the Hub's exterior. These would allow the Radiator(s) and Hub to reject heat as needed in the inner Solar System, while remaining insulated for heat retention in the outer Solar System. Phase change material (PCM) is to be located around the transmitter power amplifier unit in the Hub. It would be used to capture the waste heat from the transmitter and other electronics to redistribute it evenly over the telecommunications duty cycle of the spacecraft. As designed, the PCM would be roughly 60% ethanol, 40% water to achieve a melting point of -37°C, providing a small margin above the -40C lower operating limit assumed for the batteries and some other equipment inside the Hub.

Initial Propulsion to Get Onto Mission Trajectory

In order to accomplish the Jupiter flyby that will send OS4 to 30 AU in not much over a decade, a 200-1000 m/s delta-v deep space maneuver (DSM) would need to be completed within the first 30 – 60 days after launch. In the configuration examined with Team Xc, OS4 lacks the capability to make such a burn, so it was assumed that the spacecraft would be mounted on a kick stage that can perform the maneuver. Additionally, as OS4 would be unable to deploy the solar reflector for the duration of the mission up until right after the high-thrust DSM, so the OS4 would need to rely on the kick stage for both power and telecommunications. A version of the Orbital Maneuvering Vehicle (OMV) offered by Moog[21] was selected as an example apparently capable of satisfying all of these requirements. As shown in Figure 7, carrying a single OS4 with a mass of 220 kg, the OMV would be capable of producing ~1700 m/sec of delta-v using a monopropellant, or 2300 m/sec of delta-v using a biprop system.

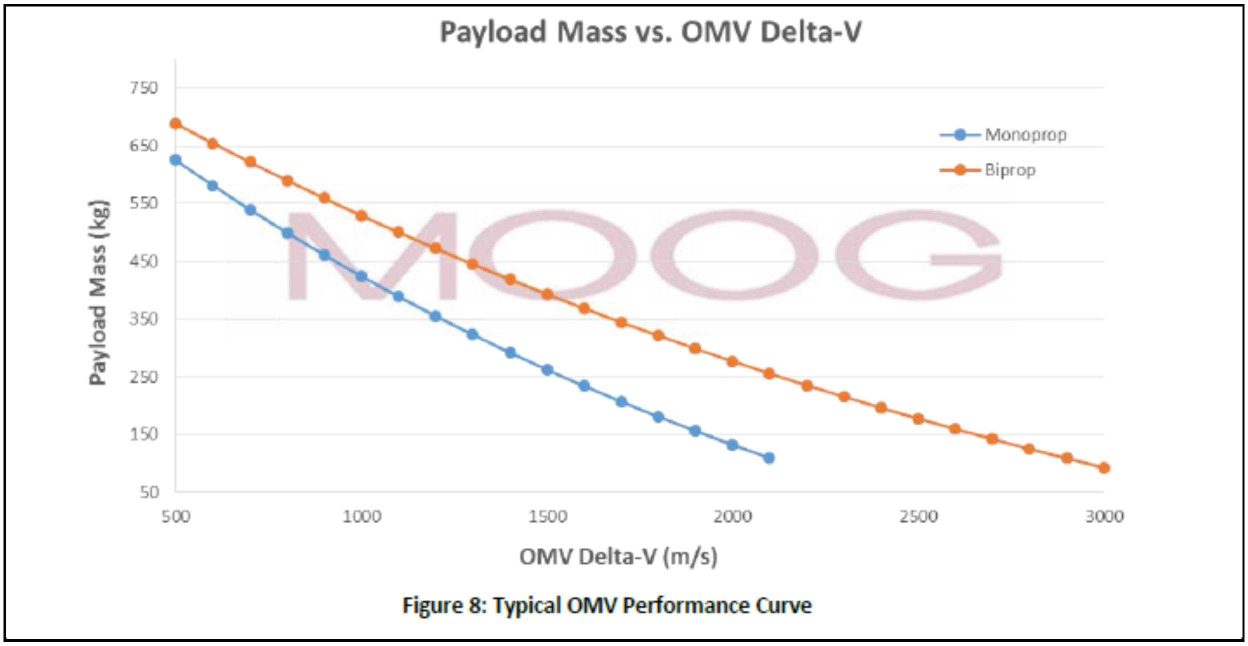


Figure 7. Typical Orbital Maneuvering Vehicle (OMV) performance is shown for a version of the example Propulsive ESPA Ring. Moog illustration used with permission.

This performance would be adequate for the Deep Space Maneuver of 200 – 1000 m/sec required for the Jupiter 2038 example trajectory, based on the referenced paper by Marissa Stender, Chris Loghry, Chris Pearson, Eric Anderson, and Joseph Maly. The OMV's variable configuration would enable it to provide the necessary power for the duration of the mission until after the DSM, as the batteries and solar array size can be altered to suit an OMV's needs.

Future Work to Increase Heliocentric Distance Capability

We made numerous assumptions that led to our 30 AU baseline design limitation. We were guided by a desire to keep our assumptions conservative enough that one might envision a modest institution with space flight experience building and operating a successful Outer Solar

System SmallSat (OS4) by 2030. With a change to one of more of the following assumptions, it is plausible for an OS4 to operate at greater heliocentric distances (see Table 1). But each of the following changes adds to technological risk and/or technology maturation cost that makes implementation of an OS4 without radioisotope power less likely. Our assumptions and their possible relaxations include:

a) Assumed operating temperatures need to be >= -40 C (233K). Relaxation: some electronics today of the kind we envisage for system-on-a-chip (SOC) implementation can operate over a range from -55 to +125 C.[22] However this is only one part, and the inside-Hub temperature must accommodate all parts. It is possible by 2030 that a subsequent generation of this part series, and an entire shipset of all the different parts for OS4, could be operable at a lower temperature. However, our mission will not drive requirements for any electronics manufacturer, and we haven't identified a large-market commercial reason for which such low-temperature parts would be justified, so we did not want to make this stretch at this time.
b) Assumed spacecraft enclosed Hub size defined by 40 cm diameter solar panel. Relaxation: with the solar panel thermally isolated from the Hub, if all the necessary componentry could be fit into a much smaller volume, with correspondingly lower surface area, perhaps a factor of 4-8X reduction could be realized in the required internal heat to offset leakage. We did not pursue this path, but consider it promising.
c) Assumed dual stack MLI, with effective emittance = 0.01. Relaxation: Switch to "thermos bottle" Hub insulation approach, based on experience with in-space cryogenic dewars (as on Spitzer Space Telescope and other missions), yielding an effective emittance = 0.0025, or a factor of 4X improvement. However, such an architecture is likely much more difficult to design and build, especially at a detailed level such as cable feedthroughs and structural elements that provide parasitic heat loss paths.
d) Assumed moderately conservative packaging volume estimate for the inflatable hardware that, with 25% margin on the volume of plastic film layers, UV-setting rigidizing resin that permeates "veins" of the Venous Web in the inflated layers, and especially space between layers in the launch packing, appears to enable packing all the Inflatables associated with a 5 m Reflector into a 90 cm diameter cylinder 70 cm long. This may be a little beyond the standard ESPA payload envelope already, but it was noted that standard ESPA volumetric margins can sometimes be exceeded depending on the needs of the host mission. Relaxation: One could imagine reducing the assumed spacing of 2 mm from one layer to the next (including veins filled with soft resin on launch), and a reduction in the assumed 50% packing factor (how much of the available volume can actually be used with a large surface having many folds and a shape that does not necessarily pack optimally in the available form factor). With this relaxation, one could imagine adding 1-2 m to the Reflector aperture for the same packaged size. However, such packaging problems are notoriously tricky, especially for something that must deploy properly after having been packed on Earth for months before launch, then survive launch loads and uneven temperature variations, and then deploy reliably 30 – 60 days after launch.
e) Assumed ESPA or ESPA Grande secondary payload adapter, up to and including the possibility of the ring itself being part of the OS4 spacecraft. Relaxation: if a dedicated (and presumably much more expensive) launch were available, then based just on

available solar power areal density, a minimum Reflector aperture of ~10.5 m appears able to collect enough solar power to enable travel and operation to the target 125 AU. We did not perform this branch of a design study, but if we were to do so, we would probably start with a Reflector diameter of 11-12 m to provide some margin.

Some combination of the above relaxations might be able to significantly extend achievable heliocentric distance within a SmallSat form factor launchable as a secondary payload utilizing or incorporating an ESPA or ESPA Grande ring. Performing such an optimization remains for future work, beyond the "existence proof" level design architecture described here.

If (and a big if) some combination of the above steps could be taken, then it is conceivable that the original objective of reaching just beyond the heliopause *might* be feasible within a SmallSat/Rideshare paradigm, perhaps in the 2040s. If that were the case, then specific impacts of the concept could include: 1) enable a constellation of SmallSats to collect data volumetrically mapping the heliosphere and its heliopause boundary, and then a little beyond, into the interstellar medium. This heliophysics goal[23] has heretofore been unaffordable; 2) current models of GCR (e.g., CRÈME-MC)[24] only provide estimates near 1 AU and inside Earth's magnetosphere. OS4s could lead to engineering estimates over the entire solar cycle of the GCR ions, ACR, and GCR electrons for the regions beyond 1 AU and extending out to the interstellar medium (ISM) beyond 150 AU, enabling better-informed future-generation electronic parts selection for a variety of NASA missions; 3) enable smaller entities, such as universities, to build payloads, or OS4, to make measurements throughout the heliosphere; and 4) as NASA pioneers the technology needed to build OSS SmallSats, invitations to the international community to employ similar techniques, with some subsystems supplied by US businesses, could expand the measurement constellation.

Table 1. Different thermal insulation approaches and paraboloid reflector sizes can afford different theoretical heliocentric distance capabilities. The first and most conservative was used for the example mission in this Phase I report.

Thermal Design	Minimum Power Generation	Inflatable paraboloid Diameter	Maximum Solar Distance
Dual Stack MLI	4.6W	5 meter	30 AU
Thermos Bottle	3.6W	5 meter	44 AU
Thermos Bottle	1W*	5 meter	60 AU
Thermos Bottle	1W*	10.5 meter	125 AU

*1W power generation assumes that pulsed plasma thrusters can be built to require no thermal control and that the spacecraft energy budget will close with 1W continuous power. This determination was not made in Team Xc, but may be possible.

Science

With a paucity of platforms from which to make in situ measurements, there is a desire for frequent sampling of spatially separated data from the inner Solar System to outside the Heliopause. The following measurement specifications were used to scope the instrument payload and operations concept for the example mission:

- Vector magnetic field ~1 nT sensitivity.
- Low-energy plasma (~1 – 30 keV).
- High-energy plasma to Galactic cosmic rays (~30 keV to 1 GeV).
- All three of the above measured simultaneously.
- Not essential that these measurements be continuous; may have gaps up to days in measurement frequency as heliocentric distance increases.
- Dust flux and particle size data may have gaps of weeks in data as long as cumulative amounts are measured.

In order to follow on the Voyager tradition of a "Solar System Portrait," we suggest the capability take one portrait set including each planet every year after ten years.

While originally envisaged to make heliophysics measurements without the ability or need to target specific measurement locations, the OS4 architecture offers the possibility of flying by one or more targets that could be coarsely targeted during the Jupiter flyby, with tracking and small TCMs used to refine flyby geometry and timing. While this clearly increases operations complexity, this capability could create the ability to collect imaging, compositional, magnetic field, and other observations of a variety of small bodies, all the way out to Trans-Neptunian objects (TNOs), satisfying Solar System science goals that might not rise in priority enough to justify a more expensive mission with a more capable spacecraft, broader instrumentation, and huge data-taking capability. An OS4 could carry 10 – 20 kg of instrumentation plus a modest pointing capability for taking images, spectra, and a variety of other measurements within the capability of small instruments. 3-axis stabilization, at least for an encounter phase if not the whole mission, could make such measurements easier.

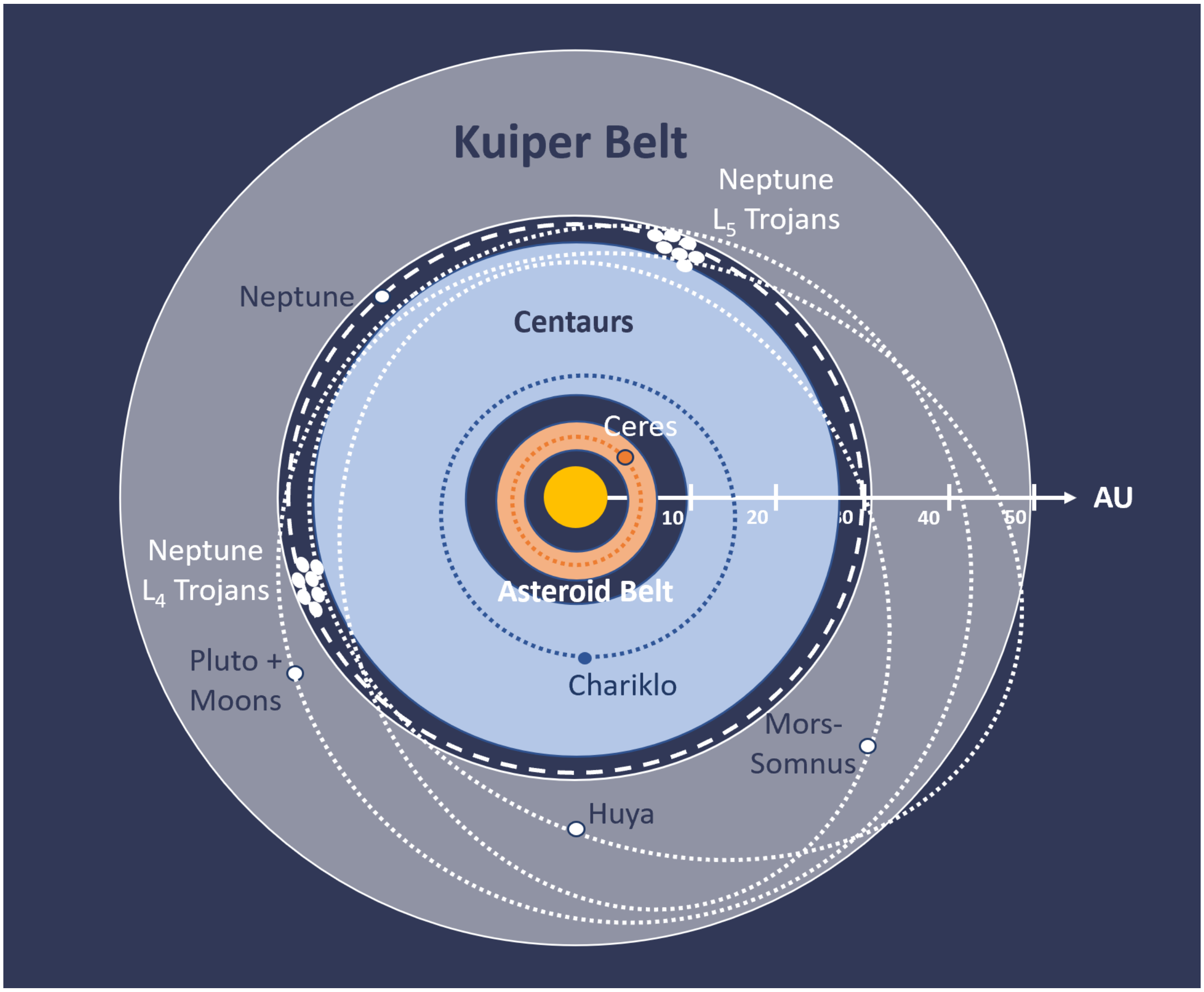


Figure 8. A variety of small bodies are available for flyby investigation by the OS4 architecture, some of which are shown schematically here. In this illustration, the example trans-Neptunian objects are shown with notional orbits, for which the true anomalies and argument of perihelion of all the objects were arbitrarily chosen to fit on the chart. Jupiter's orbit is at ~5.2 AU, and Saturn's orbit is at ~9.6 AU. From database: https://minorplanetcenter.net/iau/lists/TNOs.html

Instrumentation

Our example mission was designed so it could make the measurements listed in the above Science subsection. Accordingly, the OS4 instrument suite would include an Ion & Electron Sensor (IES), a Miniaturized Electron and Ion Telescope MeRIT, a Dust Camera, and a Solar System Portrait Camera. Four of the very small SiC Solid-State Quantum Magnetometers are also envisaged, with their spatial separation affording calibration. We postulate that the SiC Solid-State Quantum Magnetometer can be built to not require heaters or insulation.

Magnetometer placement: mounted two on Outriggers, one on the Hub, and one near the Dust Camera to gather data throughout the mission from these points on the spacecraft.

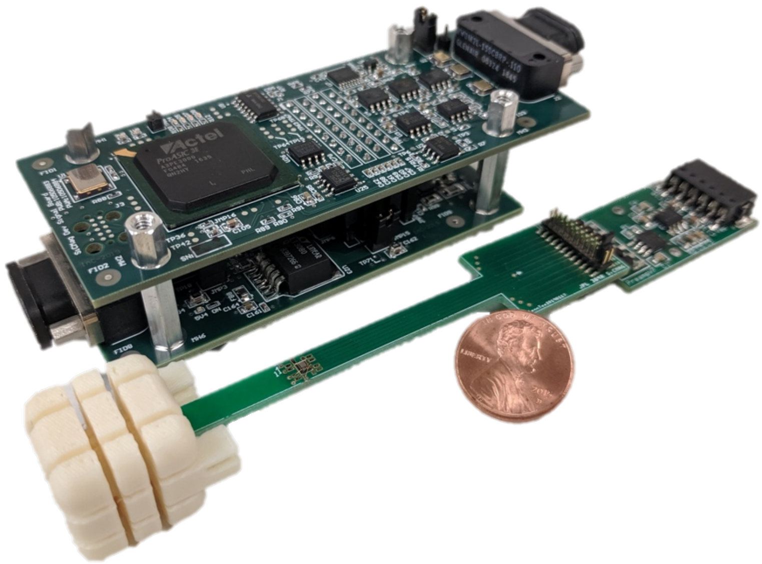


Figure 9. Present TRL3 JPL SiC Solid-State Quantum Magnetometer. Electronics are to be miniaturized to ASIC/SoC. We assume that the electronics package shown as the upper assembly of the two shown will shrink dramatically, to fit as two more boards stacked on the one that is the wider part of the right-hand portion of the lower assembly. The actual sensor is coils wrapped in the white cube to the left of the lower assembly. The distance between the left and right portions would be maintained for now, for magnetic cleanliness. Four identical magnetometers could be placed in dispersed locations: one in the Hub, one in each of two orthogonal Outriggers, and one with the Dust Camera. Because the OS4 spacecraft's power modes are simple, this tetrahedral geometry may aid in mutual calibration of the magnetometers to enable separation of the spacecraft-induced fields from the natural field without the use of dedicated magnetometer booms.

One Southwest Research Institute (SwRI) Solar Wind Ion & Electron Sensor (IES) would be mounted on the sunward side of the spacecraft Hub. Additionally, one Miniaturized Electron and Ion Telescope (MeRIT) would be mounted to the cylindrical sunward face of the spacecraft Hub aimed radially, perpendicular to the spin axis. IES and MeRIT are not to be pointed in the same direction, but they would be run at the same time and their data would be time-correlated with a record the attitude at that time of each measurement.

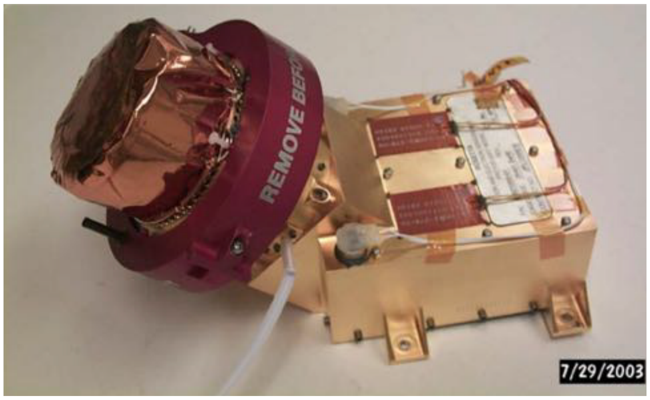


Figure 10. The completed SwRI Ion & Electron Sensor (IES) shortly before integration with ESA's comet-bound Rosetta *spacecraft. The red "Remove Before Flight" cover protects the entrance grid, while a thermal blanket cap covers the upper portion of the detector assembly. SwRI photo.*

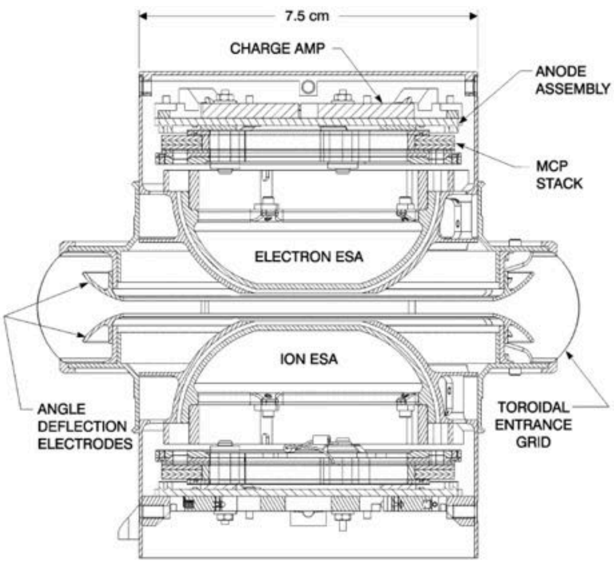


Figure 11. Cross section of Ion & Electron Sensor. SwRI

IES placement: The red collar in Figure 10 covers the instrument entrance aperture. The instrument has a pancake field of view. In the part of the s/c closest to the Sun, and not shaded from the Sun, this toroidal aperture would be placed such that the s/c spin axis, which is to be pointed within a small angular distance of the Sun (for the outer Solar System part of the mission) is parallel to a diameter across the toroid. Thus, the Sun would be located within the pancake field of view, and the pancake is rotating end-over-end about the axis roughly defined by a line from the Sun to the spacecraft, or more precisely defined by the s/c spin axis, or a line parallel and close to the s/c spin axis. The anti-Sun 90 degree segment of the pancake FOV is to be blocked, that is, +/-45 deg from the anti-Sun direction, to avoid light reflected back from the PMMR, Torus, and/or blockage by the s/c. No part of the remaining 270 deg field of view (FOV) should be blocked by any part of the s/c.

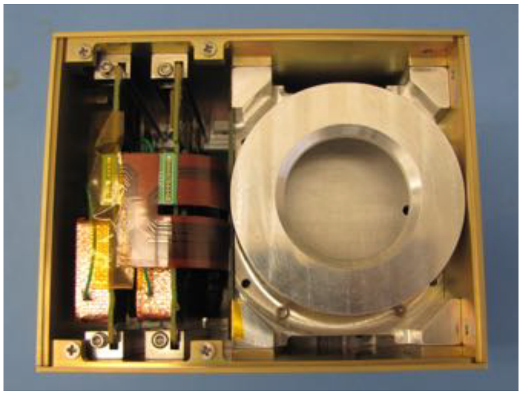
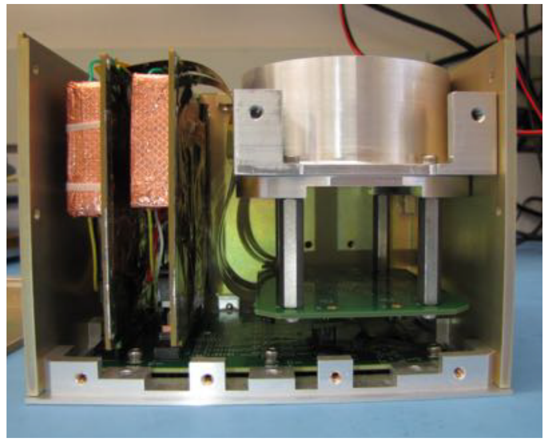



Figure 12. SwRI Miniaturized Electron and Ion Telescope (MeRIT), as developed for the Cubesat mission to study Solar Particles (CuSP), scheduled for launch aboard Artemis 1, and Compact Radiation belt Explorer (CeREs). SwRI photos.

MeRIT placement: The axis of the centerline of MeRIT's aperture and the cylinder behind it should point perpendicular to the s/c spin axis, and be clear of any view of elements of the s/c +/- ~20 deg on either side of this pointing direction. Thus MeRIT scans 360 deg with each rotation of the s/c about its spin axis.

The Dust Camera would be mounted to deployed boom supports on the anti-sun side of the PMMR. As the spacecraft travel through various parts of the solar system, dust particles will pierce small holes in the reflector. This camera will capture the sunlight through these holes to help characterize the particulate size and density across the Solar System.

Dust Camera placement: on the anti-Sun side of the spacecraft, looking back in the sunward direction at a segment of the Parabolic Metalized Membrane Reflector. Its data would be a count of the holes, position, and intensity of light showing through each hole, and how the hole population increases over time, e.g., from one month to the next.

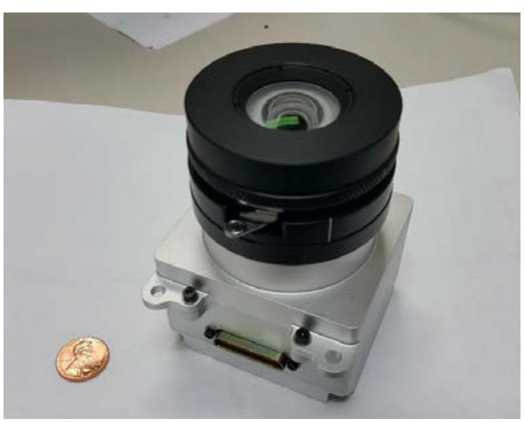


Figure 13. Mars 2020 Enhanced Engineering Camera (EECAM) Engineering Development Unit. EECAMs deployed on Mars 2020, Orbiting Carbon Observatory-3 aboard the International Space Station, and Near-Earth Asteroid SCOUT are representative of cameras that could meet the Dust Camera

requirements for OS4. The same camera back with a longer focal length lens could meet the OS4 Solar System Portrait Camera requirements.

Table 2. While spaceborne camera capabilities are expected to evolve considerably during the coming decade, the Enhanced Engineering Camera (EECAM) performance would meet OS4 camera requirements. Different (and considerably larger) optics would be used for any OS4 imaging application to fly by Trans-Neptunian Objects or other outer Solar System flyby targets.

Mars2020 EECAM Camera Specifications	
Sensor Capabilities	
Type	20M Pixel CMOS Image Sensor
Array Size	5120 x 3840
Pixel Size and Pitch	6.4um2 on 6.4um Pitch
Full well charge	15ke-
Pixel Dark Noise	8e- RMS
Windowing	Yes
Shutter	Global
Color	Bayer RGB Color
Pixel Quantization	12bit
Electrical Interface	
Commanding & Data	LVDS
Protocol	MER/MSL/Mars2020 NVMCAM
Power Input	+5.5V (+/- 0.4V)
Power	< 3 W
Memory	1Gbit SDRAM
FPGA	MicroSemi Rad-Tolerant ProASIC3
Camera Specifications	
Mass (CBE, no optics)	< 425g
Volume (CBE, no optics)	65 mm x 75 mm x 55 mm
Operating Temperature Range	-55C to +50C
Survival Temperature Range	-135C to +70C
Optics Configurations	
Navigation Camera	95°X 71°(H x V), f/12, iFOV < 0.32 mrad/pix
Hazard Camera	134°X 110°(H x V), f/12, iFOV < 0.46 mrad/pix
Sample Caching System Camera	0.49 magnification, 130mm stop to plane-of-focus, +/- 5mm Depth of Field

One Solar System Portrait Camera would be integrated into the Hub. This camera would primarily be used to take Solar System portraits throughout the mission but could also be used to gather images of Jupiter, Jovian moons, or other bodies during flyby. For missions to specific body flybys, this camera could be replaced with cameras having suitable larger optics and mounted on a gimbal.

Solar System Portrait Camera placement: somewhere on the sunward side of the Hub, with boresight approximately along the spin axis, pointing in the sunward direction.

Master Equipment List

We generated a top-level Master Equipment List (MEL) to record rough estimates of the mass of each component or group of components, with uncertainties expressed as contingency percentages. Our estimated total mass, including a system contingency of 25%, is just under 220 kg, meeting the standard ESPA interface mass limit.

Subsystem	Component (Example, if any)	Qty	Contingency	TRL	Mass (kg)/unit	Total Mass w/ contingency (kg)
ADCS	Star Tracker (Blue Canyon Technologies)	2	50%	6	0.35	1.1
	IMU (Sensonor STIM300)	2	30%	7	0.055	0.1
	Sun Sensors	2	100%		0.02	0.1
Propulsion	PPT Thrusters for spinup, precession (Busek BmP-220, augmented w/ 120g propellant)	8	33%	4	0.75	8.0
	Liquid Hypergolic Kick Stage empty (Stellar Exploration per Svitek & Veber, JANAF 2019/12) [not used in baseline; propulsive ESPA instead]	0	50%	5	5	0.0
	Hypergolic propellants (Stellar Exploration)	0	0%	9	17	0.0
	Mounting hdw, controller, interlocks	0	30%	6	3	0.0
EPS	Outer Solar System Solar Panel (LILT cells + coverglass + mounting + electrical connections)	1	50%	5	1	1.5
	Inner Solar System Solar Panel (triple junction cells + coverglass + mounting + electrical connections)	1	50%	9	2	3.0
	Battery (Li-ion)	1	30%	7	25	32.5
	Power Processing/Conversion Unit	1	50%	7	0.2	0.3
Structure	Hub Structure (Al 3D printed or tbd)	1	50%	7	3	4.5
	ESPA interface & separation hdw	1	50%	7	3	4.5
	Booms (Roccor)	8	100%	5	1	16.0
	Radiation Shielding	1	50%	7	6	9.0
Thermal	MLI	1	30%	9	2.2	2.9
	Radiator	1	100%	7	1.5	3.0
	Phase change material (water/ethanol & container)	1	100%	4	2.2	4.4
	Thermal Switch (Bugby/JPL)	2	100%	5	0.5	2.0
C&DH	Avionics pkg, incl items below (all will change by 2030)	1	100%	4	1	8.0
	Microprocessor/ Controller (Amtel ATS128)	7	3%		.0048 (each)	
	Memory (AS6C)	1	3%		0.008	
	Interface (JPL development)	4	3%			
	Memory Mux (74CBT)	1	3%		0.00064	
	Programmable wake-up timer (ABLIC)	1	3%		0.00023	
Telecom	X-band Solid State Pwr Amplifier	1	50%	4	1	1.5
	Feed Horn, cables & waveguides	1	50%	6	1	1.5
	Medium & Low Gain Antennae set	1	50%	6	0.25	0.4
	Feedhorn Extension Waveguide	1	50%	4	0.25	0.4
	Transponder electronics and housing	1	50%	4	0.5	0.8
Optics	Parabolic Metalized Membrane Reflector + Clear Reflector Canopy (Aluminized and clear Kapton [tm]	1	100%	4	6.25	12.5
	Inflated Torus	1	100%	4	0.62	1.2
	Diffuse White Annulus (coating on sunward siode of Torus)	1	100%	5	0.35	0.7
	Fresnel Element	1	50%	3	2.6	3.9
	Fresnel Element/Feedhorn Deployment Mechanism	1	50%	4	2.6	3.9
	Tensioning Strings & fasteners	1	50%	4	3	4.5
	Rigidizing Fluid (polymethylmerthcrylate per R. Gilbert Moore/Thiokol patent)	1	100%	4	4.9	9.8
	Inflated Hardware Launch Housing	1	50%	6	10	15.0
	Inflation Hardware (incl. sublimate)	1	50%	4	5	7.5
Instruments	Ion & Electron Sensor (IES)	1	30%	6	2	2.6
	Miniaturized Ion & Electron Telescope (MeRIT)	1	30%	4	1	1.3
	SiC Quantum Sensor	4	200%	3	0.1	1.2
	SiC Magnetometer electronics	4	200%	3	0.25	3.0
	Solar System Potrait Camera	1	30%	6	0.425	0.6
	Dust Camera	1	30%	6	0.425	0.6
Subtotal						173.6
Sys Conting			25%			43.4
Total incl Contingency						216.9
	ESPA Capacity (std)					220.0
	Mass Margin to standard ESPA interface					3.1

Inflated Hardware

The Inflation Package contains the flowing equipment *before* inflation and deployment. This package attaches to anti-Sun side of Hub. Structural packaging is jettisoned during inflation.

- Parabolic Metalized Membrane Reflector
- Clear Reflector Canopy
- Torus
- Tensioning Strings and fasteners
- Inflation sublimate gas source, control box, inflation gas plumbing, and other ancillary equipment to make it work.
- Venous Web attached to inflatable surfaces, containing UV-rigidizing liquid.
- Annular diffuse white skirt to reflect sunlight into Outer Solar System Solar Panel before Sun-pointing to provide power when spin axis >~5 deg off-Sun, as is always the case during Inner Solar System Cruise.

The inflation deployment is conceptually straightforward, but the Spartan 207 demonstration was reliant on maintaining inflation gas pressure to retain shape. This may be sufficient for even a few days, but not for a mission of a decade or more. Thus, we propose combining the 1990 sunlight-illuminated UV rigidization invention of R. Gilbert Moore [Ref 9] with recent work by co-author Alessandra Babuscia to describe a vein-like web structure embedded between two layers of the plastic inflatable structures. Before launch, the veins, making up the Venous Web, would be filled with the UV-rigidizing liquid before the structure is folded and packaged. Some folds would pinch out the fluid from segments of the ~1 mm diameter veins for distances up to ~5 mm, but it is assumed (needing verification in test) that the moderately low viscosity fluid would refill these sections that were compressed in folds within seconds after inflation. Deployment and sunlight is not planned until 30 – 60 days into the mission, after all high-thrust maneuvers would have been completed. While we have done no structural analysis for this particular configuration, we estimate that under the very low loads of interplanetary cruise and slow spin, with infrequent very gentle maneuvers using pulsed plasma thrusters, that a sufficiently accurate shape would be maintained to provide adequate RF and visible wavelength performance. As a photon-collecting “light bucket,” no optical quality accuracy is required for collecting solar power. At X-band wavelengths, the expected 1 mm accuracy of the L’Garde 1997 structure is several times better than what is required to serve as a good high gain antenna surface for RF wavelengths of 25 – 40 mm.

Figure 1 shows the Parabolic Metalized Membrane Reflector (“Parabola” or PMMR), Clear Reflector Canopy (“Canopy”), Torus and tensioning strings between the Torus and the periphery of the PMMR. The Hub, where most spacecraft subsystems and instruments are situated, is located near the focus of the PMMR, held in position by 4 booms that deploy soon after the Inflatables Launch Package is jettisoned. After inflation and while gas pressure is still maintained above a high enough level to keep shape, the spacecraft would execute attitude maneuvers to bathe the entire structure in sunlight long enough for the UV-setting fluid to rigidize.

Figure 14 shows the geometry of were the PMMR, Canopy, Torus, Outriggers, and Deployable Booms come together.

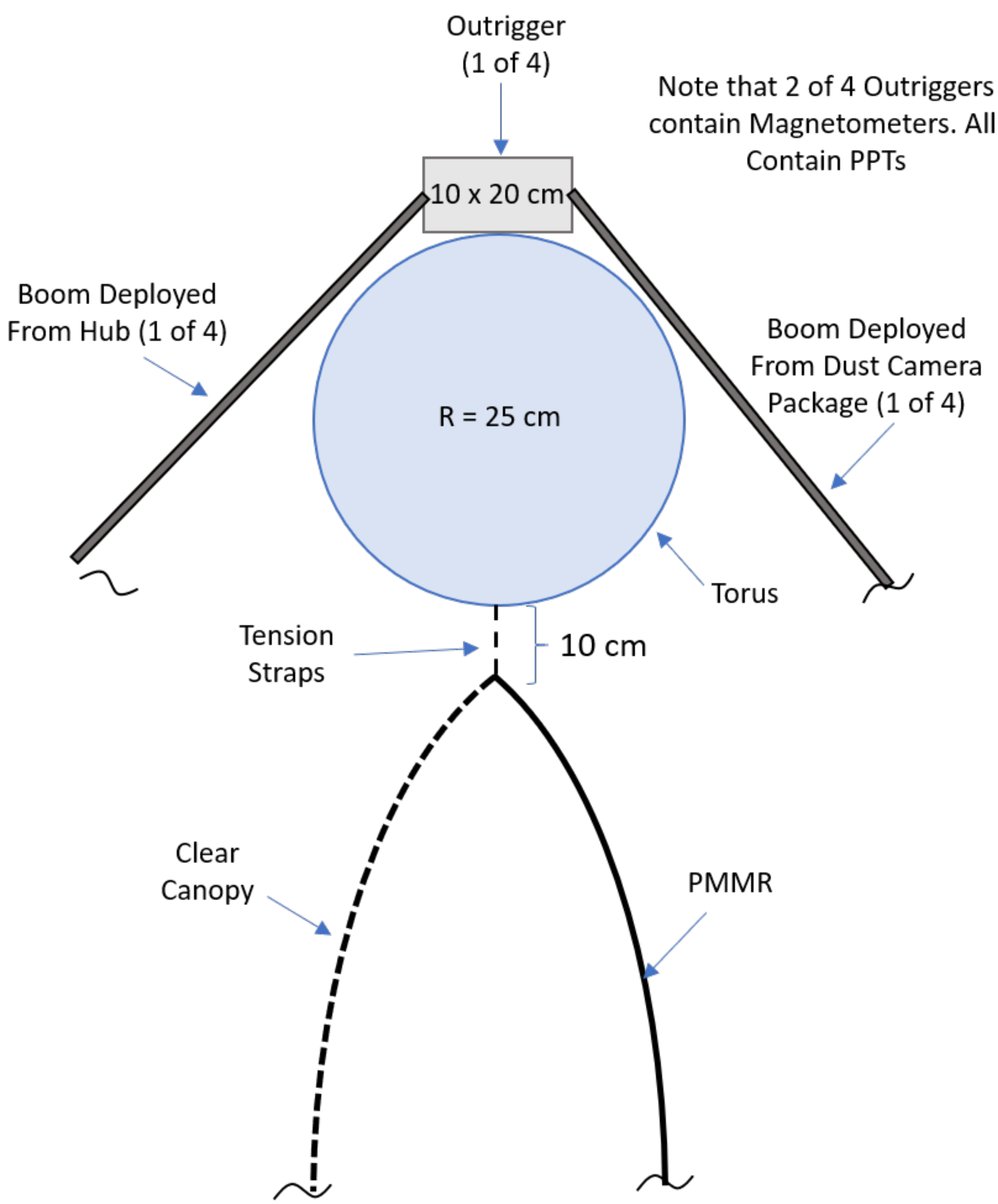


Figure 14. Cross section of the outer part of OS4 in fully-deployed configuration, showing approximate geometry of the inflatable equipment's (PMMR, Clear Canopy and Torus) connection, via the four Outriggers, to the four Booms deployed from the Hub, and the four Booms that connect to the Dust Camera package. Profiles of the Clear Canopy and PMMR are not to scale. Design of detailed mechanical connections is left for a later phase of development.

The following describes our assumptions and rough calculations for the Inflated Structure Mass & Packaging:

- Assume (conservatively) PMMR + Canopy = surface area of sphere A = 4*pi*R^2 = 78.5 m^2 for 2.5 m radius
- Torus surface area = 4*pi^2*R*r = 8 m^2 for R = 3.25 m and r = 0.25 m
- t = thickness of one layer; assume 2 mil = 0.002" = 0.05 mm = 5E-5 m
- 2nd layer, for veins and overlap for bonding gores, covers 10% of surface area
- A*t = 0.0044 m^3 ➔ 6.25 kg assuming Kapton density = 1420 kg/m^3.
- Then assume that veins filled with UV-rigidizing polymer cover 2% of entire inflated surface area at 2 mm thickness. Assume the polymer also has same density as Kapton.
- Thus, polymer filling mass = 4.9 kg
- 72 Tensioning strings 60 cm long (including knots) 1 mm dia steel ➔ round up to 3 kg including fasteners.
- Inflation hardware including sublimate assume 5 kg
- So total Inflated hardware mass = 6.25 + 4.9 + 3 + 5 = 19.2 kg (we put 100% contingency on this in MEL)
- For packing, assume equivalent material thickness is 2 mm (to accommodate filled veins), with packing factor of 0.5
- Thus, A = 86.5 m^2 * t = 2E-3 m → 0.17 m^3/0.5 packing factor
- ➔ 0.35 m^3 packed volume, or a cube 70 cm on a side
- Putting 25% margin on the volume enables packing into 90 cm diameter cylinder 70 cm long.

Electrical Power & Energy

In the inner Solar System cruise phase, the spacecraft off-points from the Sun to avoid overheating the spacecraft. The farther away from the Sun, the less solar energy is reflected onto the cells, and thus the spacecraft will lose power. This can be mitigated slightly with a diffuse white torus around the solar reflector and by a secondary set of cells around the cylindrical external surface of the Hub. Without either mitigation, the percent of maximum power generated by distance is shown below in Figure 16, along with the angle of off-pointing relative to the Sun over distance in Figure 15.

The following array parameters were used in our example design:

- Primary "Outer Solar System Solar Panel (OS3P)" 40 cm diameter facing focused reflection of Sun from Parabolic Metalized Membrane Reflector, when in outer Solar System.
- Auxillary "Inner Solar System Solar Panel (IS3P)" wrapping around Hub to provide power soon after separation from launch vehicle, and for at least part of power while near enough to Sun that the dish cannot be pointed directly at the Sun (threshold estimated at 6 AU)
- Total Area (m^2):
  - Primary OS3P = 0.12 m^2;

    - Auxiliary IS3P = 0.36 m^2
- Auxiliary IS3P output at 1 AU, assuming 1/3 of array is generating power at a time, yields 168 W max power when spin axis is perpendicular to Sun line
- Configuration: all body-fixed per above; no array deployments
- Array technology:
    - IS3P: triple-junction, like used on geostationary communications satellites
    - OS3P: LILT, like Juno or as developed since

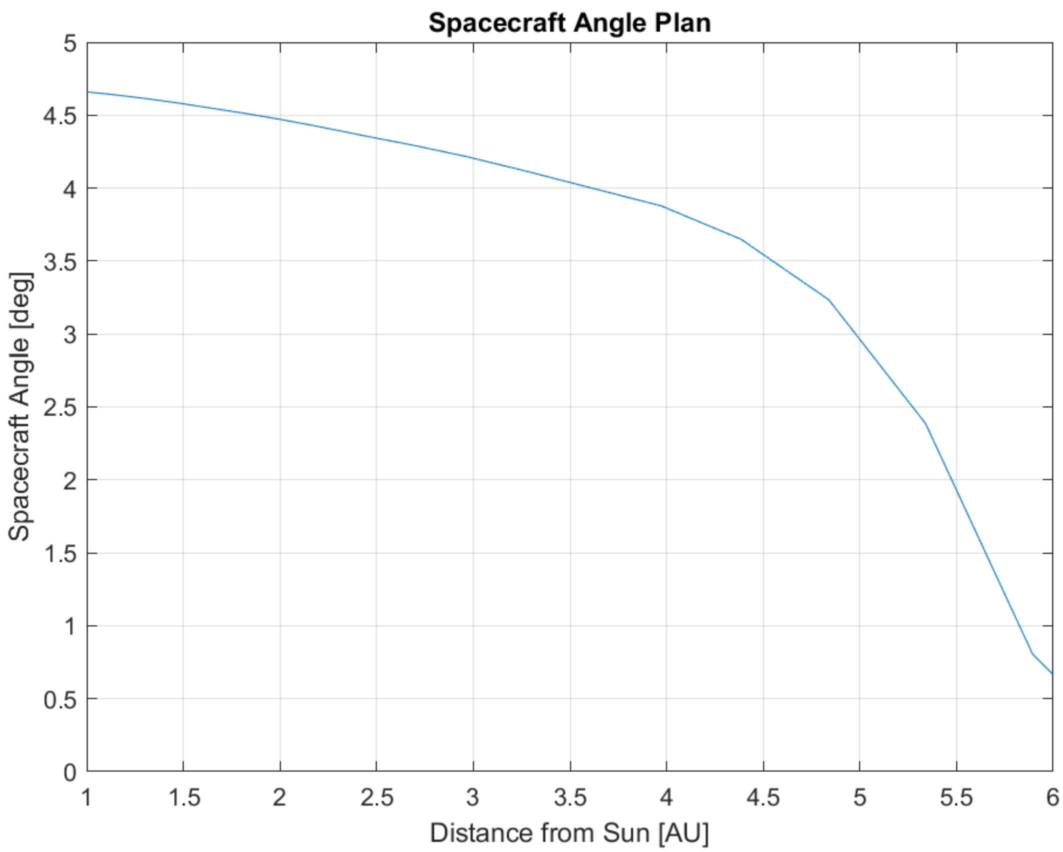


Figure 15. Inside of 6 AU, the OS4 spin axis would be pointed off the Sun line. One possible profile for this pointing appears here, but will depend on the final geometric layout of the PMMR, its focus, the Hub and its attached Outer Solar System Solar Panel. Pointing near but not on the Sun line can prevent overheating, while allowing some illumination of the OS3P by diffuse sunlight reflection off the white surface of the Torus.

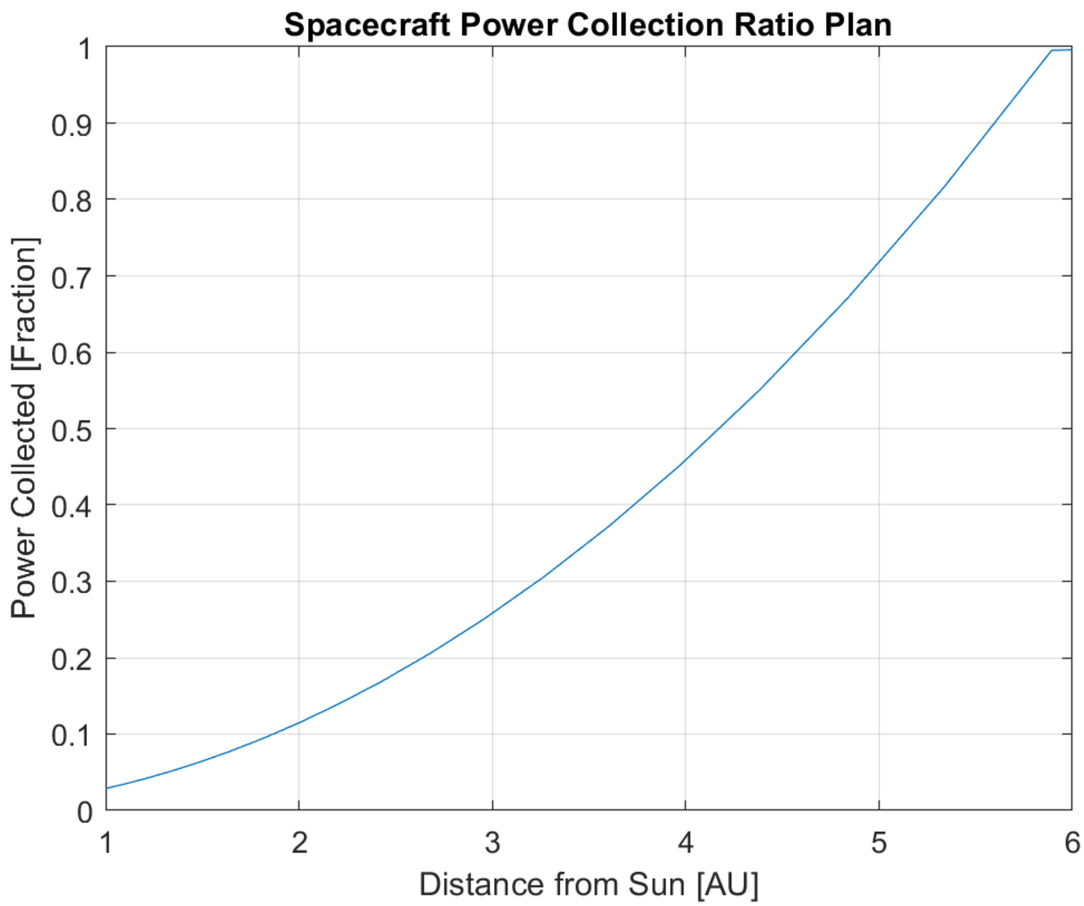


Figure 16. Inside of 6 AU, only a fraction of the power theoretically available from the 5-meter PMMR aperture would be used, with the amount controlled by off-pointing from the Sun line (see prior figure). A detailed optimization analysis would be appropriate for a later design phase among the PMMR focal length, Fresnel Element, OS3P, spacecraft moments of inertia and other parameters.

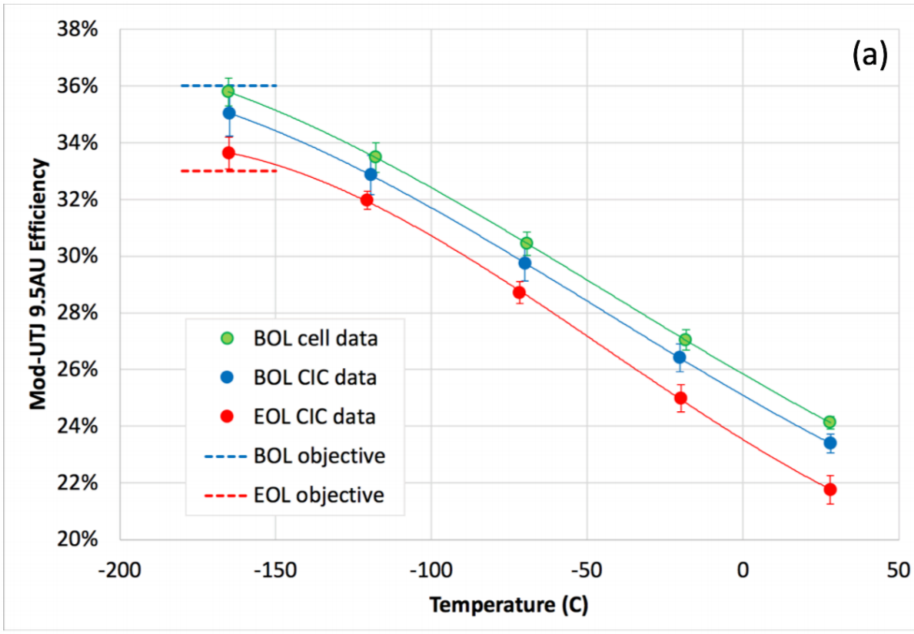
Batteries would need to last for the duration of the mission. Adding some margin to the 10.5 -year example mission lifetime to 30 AU, the batteries are sized to last for 12 years. Because most of the electronics would operate at a voltage much below the "standard" 28 vdc, a future analysis could consider whether to operate the whole spacecraft at a lower voltage.

Battery parameters for the 12-year mission were estimated as follows:

- Frequency: 63% DOD every month (total cycles: 144) almost meets JPL Design Principles (with such few cycles 63% is assumed acceptable and can be reduced with adjusting margin on the loads)
- 1,200 Wh needed per cycle (added margin and used 1733 W-h for sizing)
- -40 degrees C assumed for battery temperature and was ignored for battery sizing. Assuming future lithium ion batteries will be able to meet this temperature.
- This yields a potential solution involving:
    - →24A-h Lithium Ion Batteries

- →4 Battery Units
- →8 Cells in series for each battery unit bus voltage range of 22-36 V
- →2,765 W-h total
- →25.6 kg total
- →18 liters

The current baseline is to use low intensity, low temperature (LILT) Solar Cells for power generation. The LILT solar cells are a recent development intended for deep space missions. As implied by the name, they are designed to perform well in low temperature conditions and make use of limited solar areal power density. In their referenced paper, Andreea Boca, Clara MacFarland, and Robert S. Kowalczyk show that solar cells designed by JPL for LILT conditions demonstrated 33% end of life efficiency and 36% beginning of life efficiency. [25]



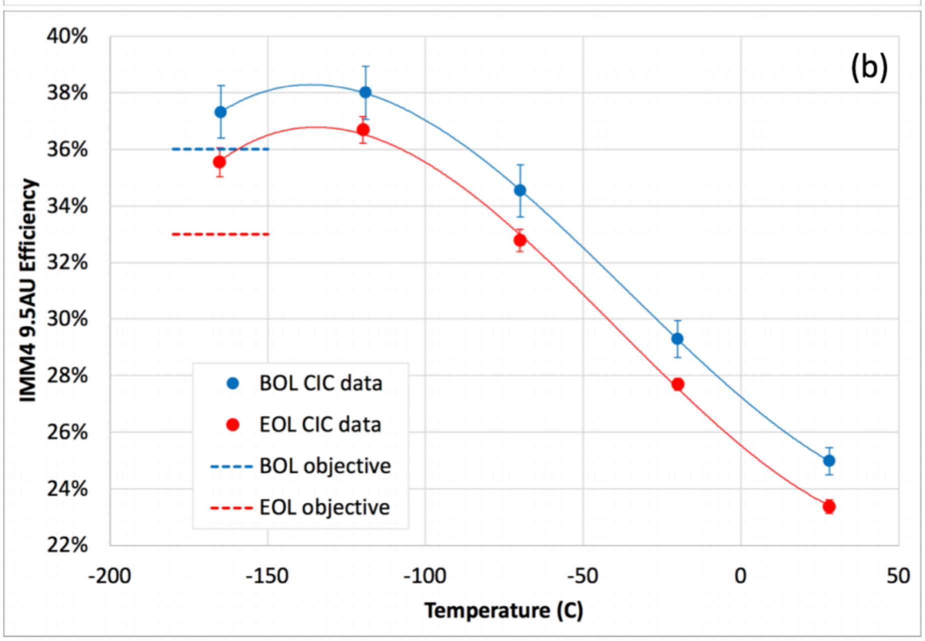
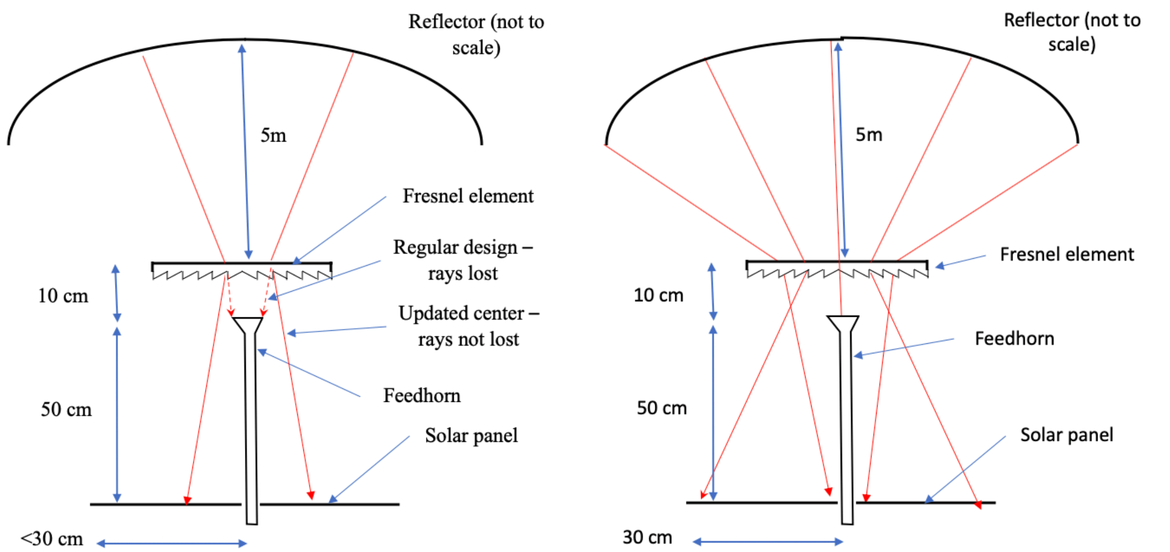


Figure 17. Low-intensity/Low-temperature (LILT) solar cell technology appears sufficient to enable power generation from a concentrator array in the outer Solar System. Plots are from the referenced 2019 paper by Andrea Boca, Clara MacFarland, and Robert S. Kowalczykof JPL.

Visible/RF Optics

Figure 18 outlines the method by which we propose to separate the paths of RF energy and visible light so they go to the right respective places on the spacecraft.

Figure 18. RF rays go to the focus of the PMMR, where the conical Feedhorn is located. Before reaching the same focus, visible rays are diffused by the Fresnel Element separated from the Feedhorn, so that they fall more evenly over the OS3P on the side of the Hub facing the PMMR. Solar Panel size and Fresnel Element groove angles remain an open trade; two options shown here.



Spider Structure

The four booms deployed from the Hub would each have an Outrigger. The four Outriggers are attached at 90 deg intervals around the periphery of the Torus, and would be the only mechanical connection between the Hub and Inflation Assembly after inflation and deployment. Instead of inflatable struts as used on Spartan 207 experiment, these would be replaced with deployable booms coming from, e.g., Roccor, Astronika, or other expert vendor.

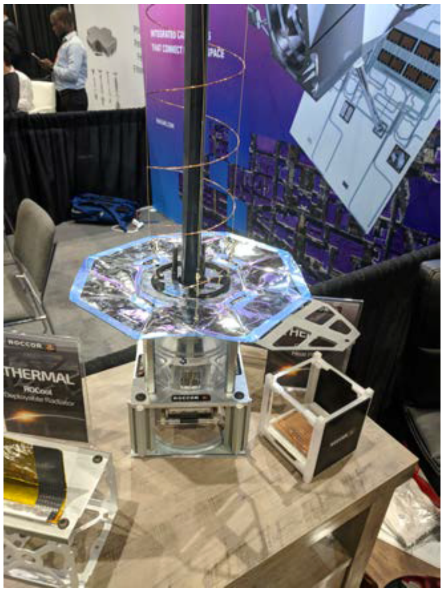


Figure 19. Example Deployable Boom from Roccor, displayed at 2019 USU Small Satellite Conference. Thanks to Roccor for providing additional information. (The boom is the black vertical shaft, deployed from the aluminum box at the bottom; the helix is not part of the boom, but is part of what is deployed by the boom for the particular application shown in this display.)

<u>Thermally Accommodating Bus: the Hub</u>

Thermal control of OS4 is a driving requirement of the design because the system must withstand large amounts of heat near Earth while keeping the electronics warm in deep space. The primary thermal concern is establishing a design that keeps the avionics and the pulsed plasma thrusters within operational temperature limits.

In the outer Solar System, it is necessary to insulate the electronics as much as possible to prevent heat loss. Team Xc proposed two approaches to accomplish this, assuming a bus size that corresponds to ESPA/ESPA Grande. The first technique would consist of dual stack multi-layer insulation surrounding the Hub housing the electronics. This configuration requires 3.6W of continuous power dissipation inside the Hub to maintain the operational temperature range. The second approach would house the electronics in a Thermos Bottle, or Dewar system. This configuration requires 0.9 W of continuous power dissipation inside the Hub. In both approachhes, an additional 1W of heater power would be required to keep the pulsed plasma thrusters within operating temperature range on the booms. From this analysis, OS4 must generate either 4.6W or 1.9W of electrical power constantly, that is then dissipated as thermal power.

Achieving effective emittance values required for this high performance architecture will be an implementation challenge. The entire spacecraft will have to be architected with thermal isolation in mind, incorporating low conductance harnesses, G10 tension bands, and minimal insulation penetrations.

Table 3 Thermal insulation options considered by Team Xc defined the minimum power dissipation levels needed to maintain a minimum temperature inside the Hub (left and center) and inside the Outriggers (right). While insulation type is a major determinant of effective emittance, the survival power levels can be lowered significantly by either reducing the survival temperature of the internal components, and/or reducing the emitting surface area by reducing the size and improving the packaging efficiency of the equipment inside. The lower the survival power, the greater the heliocentric distance at which an OS4 can operate with a given size PMMR.

Dual Stack MLI		
Side Length	0.6	m
Surface Area	2.16	m^2
Effective Emmitance	0.01	
T_{low}	233	K
P_{surv}	**3.61**	**W**

Thermos Bottle		
Side Length	0.6	m
Surface Area	2.16	m^2
Effective Emmitance	0.0025	
T_{low}	233	K
P_{surv}	**0.90**	**W**

Prop Outrigger		
Side Length	0.125	m
Surface Area	0.09375	m^2
Effective Emmitance	0.01	
T_{low}	223	K
P_{surv}	**0.13**	**W**

In order to maintain operation, we assumed that the avionics would need to be held between -40°C and +50°C. A radiator rejects heat in the inner Solar System. Thermal switches disconnect the heat path to the radiator when internal heat needs to be conserved.

The Busek pulsed plasma thrusters do not currently have a specified temperature range, so this is an area in need of more research.

Figure 20 shows the power generated (without off-pointing) is plotted against the minimum and maximum thermal power values.

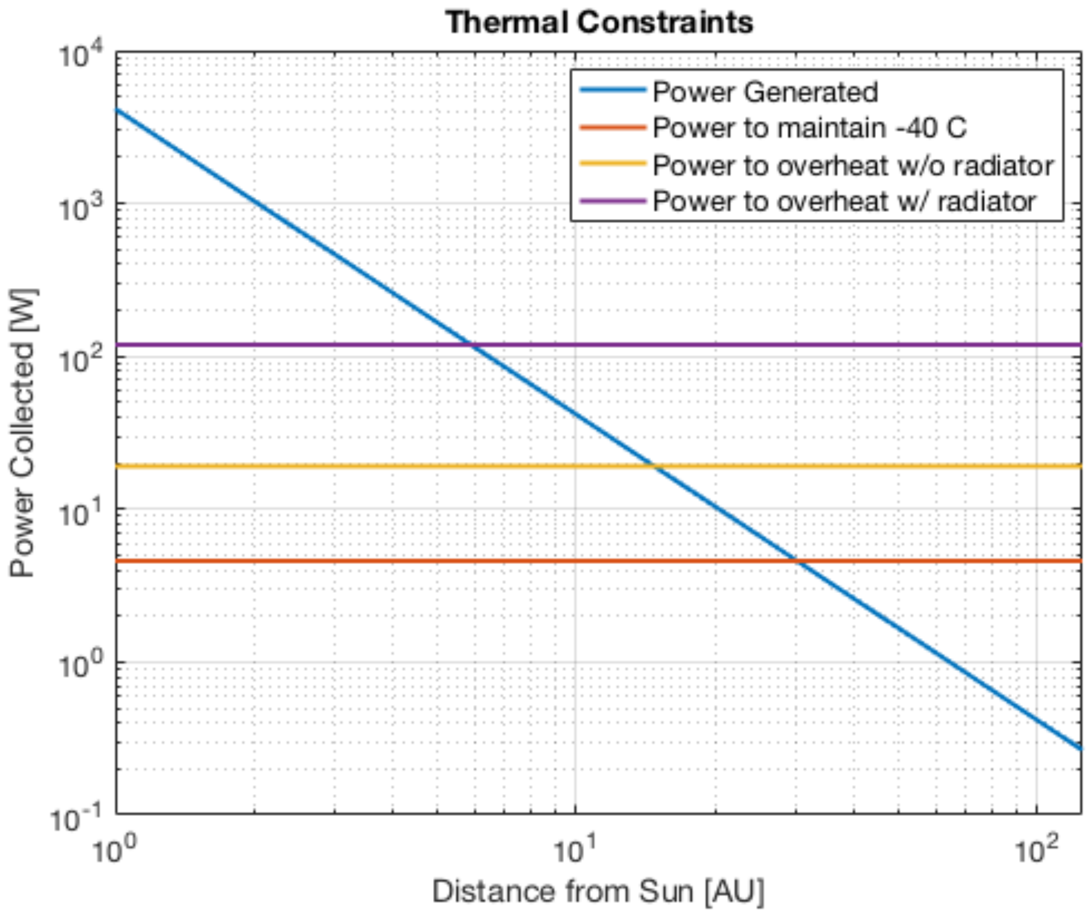


Figure 20. Our OS4 example design assumes dual stack multi-layer insulation (DS-MLI) surrounding the Hub housing the electronics. This configuration requires 3.6W of continuous power dissipation inside the bus to maintain the operational temperature range (red horizontal line), which intersects the electrical power that can be produced from sunlight reflected by the 5-meter reflector onto the Outer Solar System Solar Panel (blue diagonal line). These lines intersect at ~30 AU, or about the orbit of Neptune. Also shown are maximum power levels that can be dissipated inside the Hub with or without an external radiator, not accounting for insolation onto the Hub itself.

With the MLI-based thermal design and the intended 5-meter inflatable paraboloid, the heliocentric distance capability would reach 30 AU. With the thermos-based design and the 5-meter paraboloid, heliocentric distance capability could reach 44 AU. If PPTs were developed that did not require thermal control and the spacecraft energy budget could close with 1W dissipated inside the Hub, the design could reach 60 AU with a 5-meter paraboloid or the originally proposed 125 AU with a 10.5-meter paraboloid.

Low Power Avionics

An example avionics approach was focused on low power consumption, taking advantage of the revolution in consumer demand for high-performance cell phones and other long-life battery-powered devices. During a largely uneventful cruise for years in the outer Solar System, not much needs to happen most of the time. Therefore, long periods of "hibernation" are possible for most electronics functions, interrupted by brief periods to take data, perform maneuvers, and communicate with Earth at intervals.

At the heart of our example avionics approach is an Internet of Things (IoT) microcontroller and other low-power components (see Table 4 and Figure 21). A Clock must operate all the time, but high accuracy (better than 1 sec/month) is unnecessary. The Clock must wake up instruments and equipment in their signal chain to memory every day to take heliophysics data, and other subsystems every month for DSN passes. Onboard clock drift can be measured from DSN pass to DSN pass to maintain accurate on-ground predictions of when to start the next pass.

The key elements from which an example core avionics set was assembled are:

1. Programmable wakeup timer: ABLIC part
2. Microprocessor/Controller: Atmel part
3. SRAM Memory: Alliance Part
4. Interface chip to instruments: JPL custom part
5. Memory mux chip: IDP part
6. Non-volatile Rad-Hard Memory
7. Flight SW Magnetoresistive Random Access Memory (MRAM)
8. Various custom interface chips
9. Other elements shown in red in the Avionics Block Diagram, Figure 21

[Note: Only #2, #6 & #7 are "space" parts. #4 is being designed as a space part.]

Table 4. A selection of current microelectronics is capable of supporting the relatively simple functions of the OS4 architecture with very low power consumption. Further analysis and testing are required to determine if these parts, or alternatives, can operate in all the environments and for the mission duration required for a successful OS4 mission. However, the rapid advancement of microelectronics technology driven by consumer applications, and the serendipitous improvement in radiation tolerance that has been experienced to date, suggests that a) by a 2030 launch date, none of these parts are likely to be available, and b) higher performance electronics will be available that can meet OS4 requirements, maybe even in the form of one or a few custom System-on-a-Chip (SoC) implementations.

Ultra Low Power IoT Microcontroller Families							
Series	Vendor	Grade	Architecture	Sleep Power (mW)	Active Power (mW)	min temp active ºC	max temp active ºC
PIC32MX2	Microchip Technologies	Commerical	RISC_V-32	0.297	165	-40	85
AtmegaS128	Atmel	Space Grade	AV-RISC-32	0.0825	36.3	-55	125
SAM L11	Microchip Technologies	Commerical	ARM/Cortex	0.00165	2.64	-40	125
Atmega328P	Atmel	Automotive	AV-RISC-32	0.00495	6.6	-55	125
SAM D	Microchip Technologies	Commerical	ARM/Cortex	0.14256	14.19	-40	125

Other parts included in the example are:

- AT-S128 solely for Nav/Guide/ACS functions and interfaces + custom interfaces board/chip
- AT-S128 for EPS and Thermal/voltage telemetry (JPL REU-like) processing
  - custom board/chip for power interfaces/switching
  - FPGA for telemetry concentration/digitizing
- AT-S128-based board (outside warm inner box) for inflation/deployment/kick stage control/interface
  - Can be powered off and ignored once fully deployed/inflated
- Interfaces undetermined now so just rough assumptions

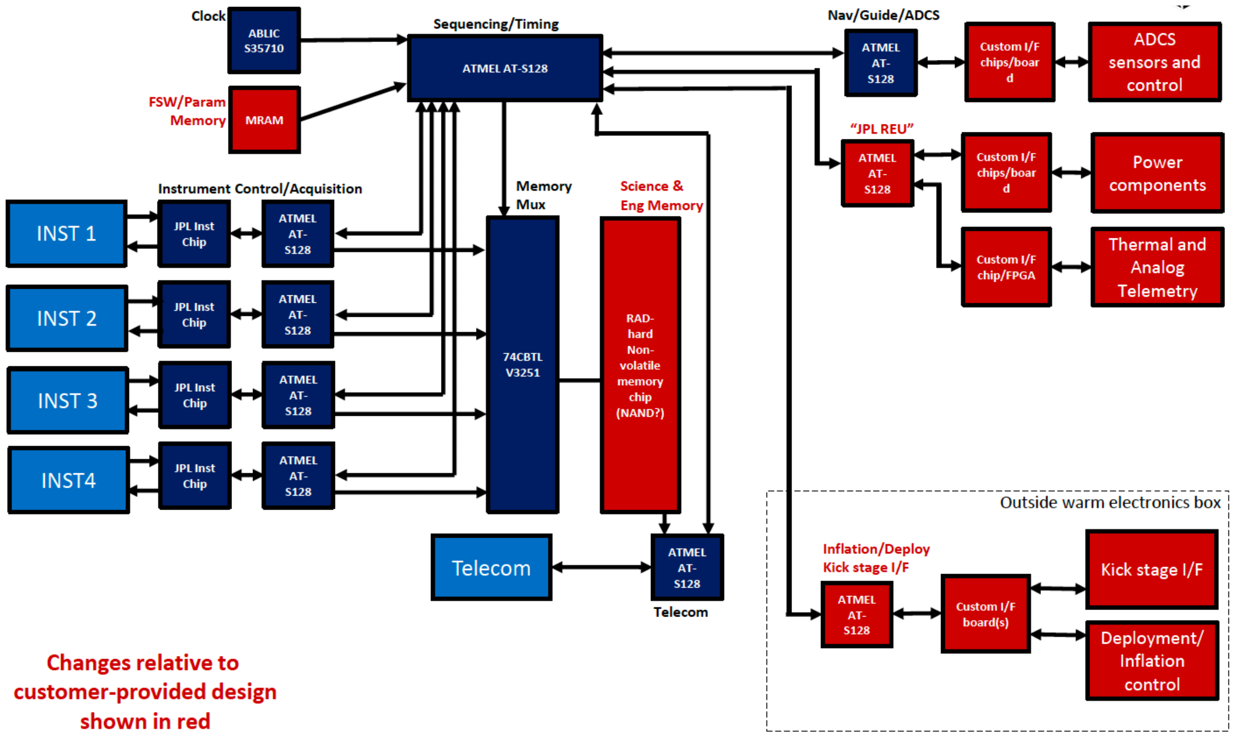


Figure 21. Top-level avionics block diagram as used for the example OS4 mission. Major changes are inevitable for any mission a decade later.

The following assumptions were made to estimate a minimum average power requirement for the outer Solar System. Keep in mind that at all closer heliocentric distances, more power and energy can be obtained and utilized; these assumptions were used simply to describe a limiting case.

- Assumed can power off the NAND when sleeping and using rough assumed numbers for other new custom parts
- Used instrument and telecom Duty Cycles:
  - 2 instruments on for 15 min 12 times/month
  - 1 instrument on for 15 min 3 times/month
  - 90 min of telecom (no instruments) per month
  - 30min/month of non-telecom and non-instrument's on for engineering time (memory scrubbing, cross-string checkout, etc.)
- Assuming backup string is always sleeping
- Assuming kick-stage/deploy interface/control turned off completely
- Averages out to ~32.5mW continuous in deep-space mode

We set a goal to provide radiation survivability to 200 krad [Si] for CMOS and other electronics. We had a further desire that the design to provide sufficient shielding down to 100 krad [Si] or better. Our basis for this was that as feature sizes decrease, some CMOS electronics is proving hardier to radiation exposure than legacy electronics, to levels of 100 krad and beyond. Missions to Europa, and even GEO, provide NASA and industry strong motivation to understand the underlying physics, and design hardware and software to accommodate radiation environments at these levels.

Most systems except the Clock may be turned off during the Jupiter flyby if useful to limit radiation damage.

Our conceptual design is very early, and outside the "design family" for any prior missions. Among the risks and concerns brought up were:

- Complexity of timing and interactions between distributed software-controlled central processing units (CPU) is non-trivial.
- Flight software (FSW) development and testing cost/schedule risk.
- Performance/complexity trade-off has not been performed;
    - Could be mitigated with a common avionics bus but at the cost of performance.
- Complexity of developing custom software needed (above) on full custom, all-new hardware is also non-trivial and significant risk to cost/schedule.
    - Could be partially mitigated with heritage hardware, but likely at the cost of more power.
- Lack of visibility or control during "hibernate" state between science and telecom ops.
- As a matter of design philosophy, JPL doesn't presently power off (or hibernate) a non-landed spacecraft during cruise due to risks of off-nominal situations and need for active control;
    - Could be mitigated by redesign with "always on" FDU, basic processor and self-contained minimal attitude determination & control (ADCS) hardware for maintaining pointing.
- Mitigations add technology development risks of their own, as well.
- All parts specified are unproven in long-term deep-space environment.
- Analysis possible, but such a newly-proposed approach is usually a mix of known and unknowns.
- No cross-string interfaces accounted for (if there were to be a redundant avionics string);
    - Unsure if there are available interfaces as-is.
    - Nothing present for cross-string synchronization.
    - Greater power needed to have both strings up at the same time or develop new scheme.
- Bandwidth of "Sequencing/Timing" ATMEL chip to be the interface to memory for spacecraft systems and engineering data in addition to main role.
- Connecting other processors directly to the memory multiplexer (mux) increases timing complexity further.
- Potentially not enough processing power for worst-case ADCS computing needs with one AT-S128;
    - Could be mitigated with stronger processor there (more power).
- Power needs of space-grade ATMEL chip not verified, but probably close enough for now. Likely variable with processing load.

Some additional comments included:

- More power savings could be achieved with power OFF of subsystems instead of "sleep mode" (still not recommended approach);
    - Requires separation of electronics into separate power domains and addition of switches.
- More nonvolatile memory would be preferred with limited capability to extend telecom pass times (power limited).
- Overall this approach could *theoretically* work with ample time for design, analysis and a larger power budget for avionics while awake and asleep.
- Many concerns are easily mitigated with more power budget.
- The 30 AU mission architecture studied in Team Xc, with a higher allowed C&DH power budget, seems feasible.
- At 30AU, the power budget closes with up to 2.5W average C&DH power, which would even allow the use of Sphinx-based avionics.

Table 5. Power modes for example avionics arrangement suggest that OS4 operations are feasible out to 30 AU.

Part	**On Power (mW)**	**Sleep power (mW)**	**Notes**	**Modes:**	***Sleep***	***Everything On***	***2 inst on and normal w/o telecom***	***1 inst on and normal w/o telecom***	***normal w/ telecom***	***normal w/o telecom***
AT-S128	252	0.574	from customer		4.592	2016	1261.722	1010.296	1010.296	758.87
Inst. IF chip	100	0.12	from customer		0.48	400	200.24	100.36	0.48	0.48
NAND	45	0	based on slightly more than customer's SRAM chip, turn off instead of sleep		0	45	45	45	45	45
MRAM	45	0.045	based on slightly more than customer's SRAM chip		0.045	45	45	45	45	45
Sleep clock	0.00066	0.00066	from customer		0.00066	0.00066	0.00066	0.00066	0.00066	0.00066
ACS IF board	200	0.5	Based on AT-S128		0.5	200	200	200	200	200
Power I/F	200	0.5	Based on AT-S128		0.5	200	200	200	200	200
TLM I/F	150	0.5	Based on AT-S128		0.5	150	150	150	150	150
Memory Mux	2	0.001	from customer		0.001	2	2	2	2	2
Power Conditioning (assume 15% overhead)					0.993	458.700	315.594	262.898	247.916	210.203
Total power need (mW)					7.611	3516.701	2419.557	2015.555	1900.693	1611.553
	Minutes in a month:	43200								
Minutes on per month			Mostly from conOps on slide 55 of updated customer packet		42855	0	180	45	90	30
Monthly energy need (mWh)					5436.48	0.00	7258.67	1511.67	2851.04	805.78
Total energy needed per month (mWh)				17863.639						
Average power needed per month, active string (mW)				24.811						
Power needed for backup always asleep string (mW)				7.611						
Total Subsystem power average (mW)				**32.422**						

Telecommunications

We examined X- and Ka-band telecommunications. While Ka-band theoretically yields greater bit rate performance for a given power level and antenna size, the deciding factor was that Ka-band pointing requirements are tighter than the X-band requirements (0.1 degrees vs. 0.3 degrees). The rough pointing budget in Table 6 no longer closes for Ka-band. Refinement may be possible, but an additional drawback for Ka-band is that the 70 m DSN stations do not handle Ka-band. It is intended that OS4 spacecraft be able to complete their standard mission using only 34 m stations, but the potential availability of 70 m apertures providing 4X the "standard" data rate at X-band would be attractive for handling spacecraft emergencies, or getting all of an encounter data load down in 4X less transmission time.

Table 6. Early OS4 spacecraft pointing budget, showing that this is driven by antenna beamwidth. While Ka-band telecommunications performance is impressive, the tighter beamwidth that would be required drove the choice to X-band.

Pointing Budget Component	**Value**	**Notes**
Pointing Requirement	0.322 deg	Derived from X-Band Beamwidth
Sun Sensor Accuracy Spin	< 0.1 deg	Based off NFSS-411 Sun Sensor
Stabilization Accuracy	0.1 deg	From SME: The New SMAD[26]
Minimum Impulse Bit Accuracy	1.45E-4 deg	BGT-X1 Thruster
Total Accuracy	**0.2001 deg**	
Margin	**0.1219 deg**	

Primary elements of the telecommunications subsystem design are (see also Figure 22):

- Radio: Iris-like (with development)
- Amplifiers:
  - Solid state power amplifier (SSPA, Iris like) for 20 W RF output (40 W consumption, assuming future development)
  - Low-noise amplifier (LNA, Iris like, no development required)
- Antennae
  - High gain antenna (HGA, mounted on the inflatable surface, assuming 5 m at 30% efficiency)
  - Medium gain antenna (MGA, horn with 8-11 dB of gain, used as feed for the HGA)
  - Low gain antenna (LGA, patches, MarCO-like)

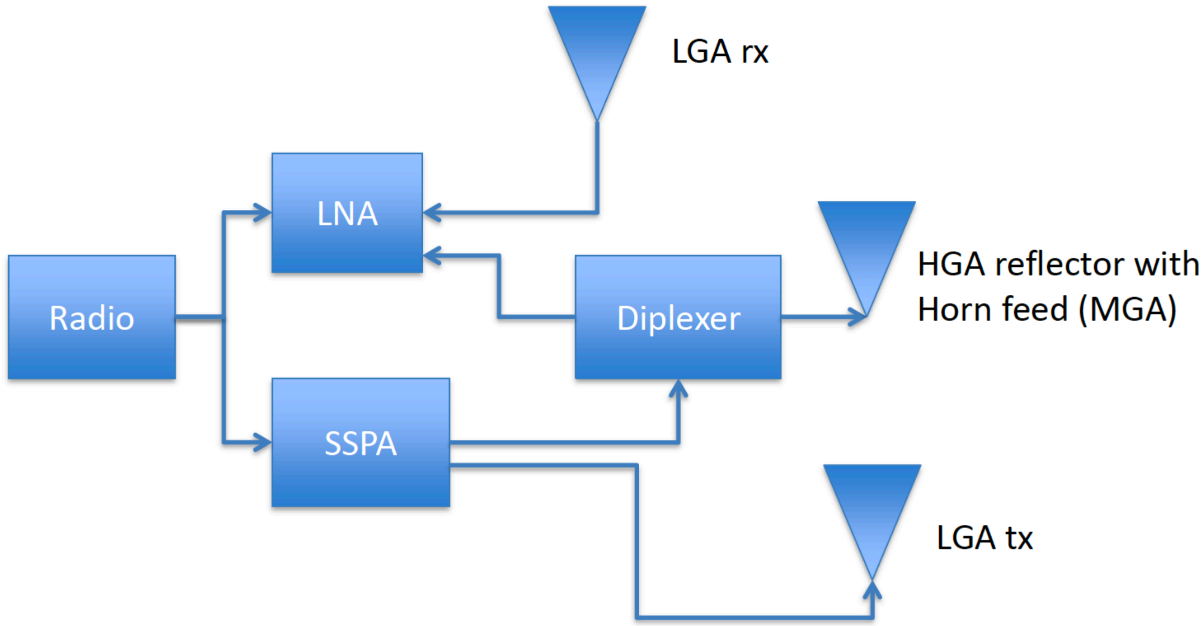


Figure 22. Top-level block diagram for telecommunications equipment.

Link budgets for 30 AU (Table 7) and 125 AU (Table 8) were found to be more than adequate to meet science data return needs.

Table 7. OS4 Link budget closes at 30 AU. The noted data rate is higher than required for a heliophysics-driven mission, so one degree of freedom for a future more detailed design would be to reduce the power level of the solid state power amplifier (SSPA), thus reducing its size, and thence maybe the size of the Hub, that in turn could extend heliocentric distance capability by reducing the Hub's required internal power dissipation needed offset thermal radiation losses.

Item	Symbol	Units	Downlink X-Band	Downlink Ka-Band	Uplink X-Band
EIRP:					
Transmitter Power	P	dBW	13.00	13.00	43.00
Line Loss/Waveguide Loss	L_l	dB	-1.00	-1.00	-0.70
Transmit Antenna Gain (net)	G_t	dBi	46.51	57.05	73.09
Equiv. Isotropic Radiated Power	EIRP	dBW	58.51	69.05	115.39
Receive Antenna Gain:					
Frequency	f	Ghz	8.40	32.00	7.10
Receive Antenna Diameter	D_r	m	70.00	34.00	
Receive Antenna efficiency	η	n/a	0.75	0.75	
Receive Antenna Gain	G_r	dBi	74.55	79.89	46.51
Free Space Loss:					
Propagation Path Length	S	km	4,488,000,000.00	4,488,000,000.00	4,488,000,000.00
Free Space Loss	L_s	dB	-303.98	-315.59	-302.52
Transmission Path and Pointing Losses:					
Transmit Antenna Pointing Loss	L_{pt}	dB	-1.00	-1.00	-0.40
Receive Antenna Pointing Loss	L_{pr}	dB	-0.40	-0.40	-1.00
Atmospheric Loss (H2O and O2 losses)	L_{atmo}	dB	-0.20	-0.20	-0.20
Loss due to Rain	L_{rain}	dB	-2.00	-2.00	-2.00
Implementation, additional losses		dB	-1.00	-1.00	-1.00
Total Additional Losses		dB	-4.60	-4.60	-4.60
Data Rate:					
Data Rate	R	bps	3,000.00	5,000.00	2,000.00
Data Rate	10 log(R)	dBbps	34.77	36.99	33.01
Boltzman's Constant:					
Boltzman's Constant	10 log(k)	dBW/(Hz*K)	-228.60	-228.60	-228.60
System Noise Temperature:					
System Noise Temperature	T_s	K	33.00	50.00	930.00
System Noise Temperature	10 log(T_s)	dBK	15.19	16.99	29.68
E_b/N_o		dB	3.13	3.37	20.69
E_b/N_o required		dB	0.10	0.10	9.60
Margin		dB	**3.03**	**3.27**	**11.09**

We also considered tones (sometimes called semaphores, or a variant on JT65 multi-tone frequency shift keying (FSK)) in terms of detectability and rates. There are two issues with this. The smaller one is that we would have to come up with a block coding scheme like JT65 did.[27] This isn't difficult but it's not in the standard catalog now. The bigger issue is that binary phase shift keying (BPSK) is 3 dB better than frequency shift keying (FSK, aka tones) because of the coherence/non-coherence detection issue. A factor of two is a lot when you're on the edge anyway, so we stayed with the standard DSN service catalog.

Table 8. OS4 Link budget also closes at 125 AU with an adequate data rate.

Item	Symbol	Units	Downlink X-Band	Downlink Ka-Band	Uplink X-Band
EIRP:					
Transmitter Power	P	dBW	13.00	13.00	43.00
Line Loss/Waveguide Loss	L_l	dB	-1.00	-1.00	-0.70
Transmit Antenna Gain (net)	G_t	dBi	46.51	57.05	73.09
Equiv. Isotropic Radiated Power	EIRP	dBW	58.51	69.05	115.39
Receive Antenna Gain:					
Frequency	f	Ghz	8.40	32.00	7.10
Receive Antenna Diameter	D_r	m	70.00	34.00	
Receive Antenna efficiency	η	n/a	0.75	0.75	
Receive Antenna Gain	G_r	dBi	74.55	79.89	46.51
Free Space Loss:					
Propagation Path Length	S	km	18,700,000,000.00	18,700,000,000.00	18,700,000,000.00
Free Space Loss	L_s	dB	-316.37	-327.99	-314.91
Transmission Path and Pointing Losses:					
Transmit Antenna Pointing Loss	L_{pt}	dB	-1.00	-1.00	-0.40
Receive Antenna Pointing Loss	L_{pr}	dB	-0.40	-0.40	-1.00
Atmospheric Loss (H2O and O2 losses)	L_{atmo}	dB	-0.20	-0.20	-0.20
Loss due to Rain	L_{rain}	dB	-2.00	-2.00	-2.00
Implementation, additional losses		dB	-1.00	-1.00	-1.00
Total Additional Losses		dB	-4.60	-4.60	-4.60
Data Rate:					
Data Rate	R	bps	170.00	300.00	500.00
Data Rate	10 log(R)	dBbps	22.30	24.77	26.99
Boltzman's Constant:					
Boltzman's Constant	10 log(k)	dBW/(Hz*K)	-228.60	-228.60	-228.60
System Noise Temperature:					
System Noise Temperature	T_s	K	33.00	50.00	930.00
System Noise Temperature	10 log(T_s)	dBK	15.19	16.99	29.68
E_b/N_o		dB	3.20	3.19	14.31
E_b/N_o required		dB	0.10	0.10	9.60
Margin		dB	**3.10**	**3.09**	**4.71**

50% DC-to-rf transmitter power amplifier conversion efficiency was assumed as a (somewhat optimistic) technology advancement from the present. In the event that 25% efficiency is realizable in the OS4 time frame, then the downlink rate would be cut in half, but is still considered viable. This is another way of illustrating that the solar collector/rf antenna size in this case would be determined by the thermal balance needed to keep equipment at or above minimum operating temperature, rather than telecom needs. In the case of this proof of concept point design, aperture size is limited by our assumptions regarding the ability to pack the inflatable equipment, plus the rest of the spacecraft, into a volume and mass suitable within the present comprehension of a “SmallSat” using a rideshare with a larger primary mission payload.

Packaging for ESPA-Compatible Rideshare Launch

Figure 23 shows a Moog concept for a propulsive ESPA ring Orbital Maneuvering Vehicle (OMV), on which the performance shown in Figure 7 is based.[28] Addition of solar panels and other subsystems give the OMV full capability to carry one or more OS4s from launch vehicle separation through the deep space maneuver 30-60 days after launch. This was the configuration examined and reported out of Team Xc. Alternative possibilities are discussed in Propulsion.

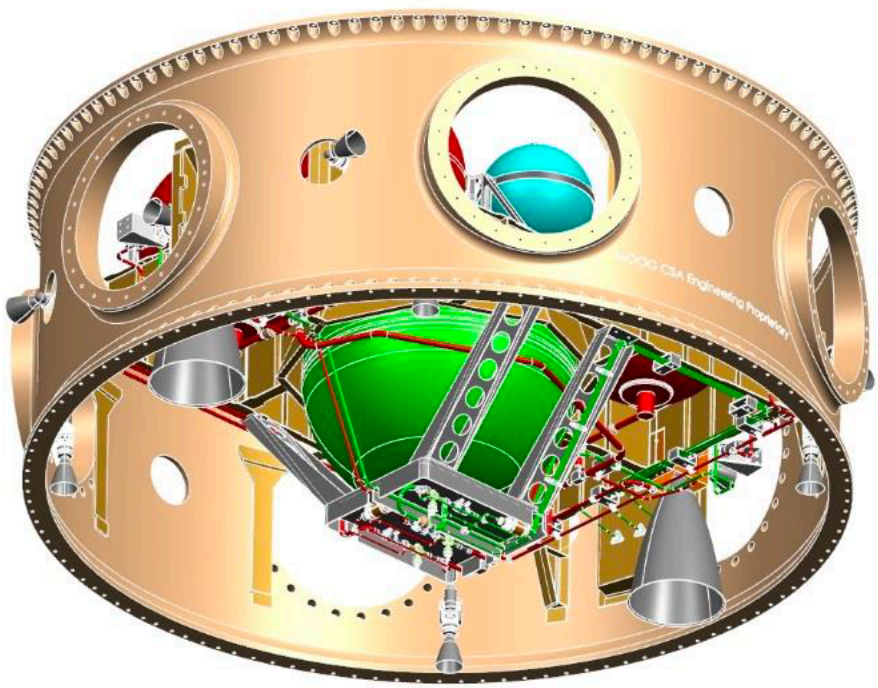


Figure 23. Example Propulsive ESPA Ring shown in a Moog Orbiting Maneuvering Vehicle (OMV) configuration. In the Team Xc session, a self-contained OMV-like stage was assumed to provide all spacecraft services up through the 200 – 1000 m/sec deep space maneuver 30 – 60 days after launch, with an OS4 as its payload that separates to fly on its own after the Deep Space Maneuver (DSM). Moog CAD illustration.

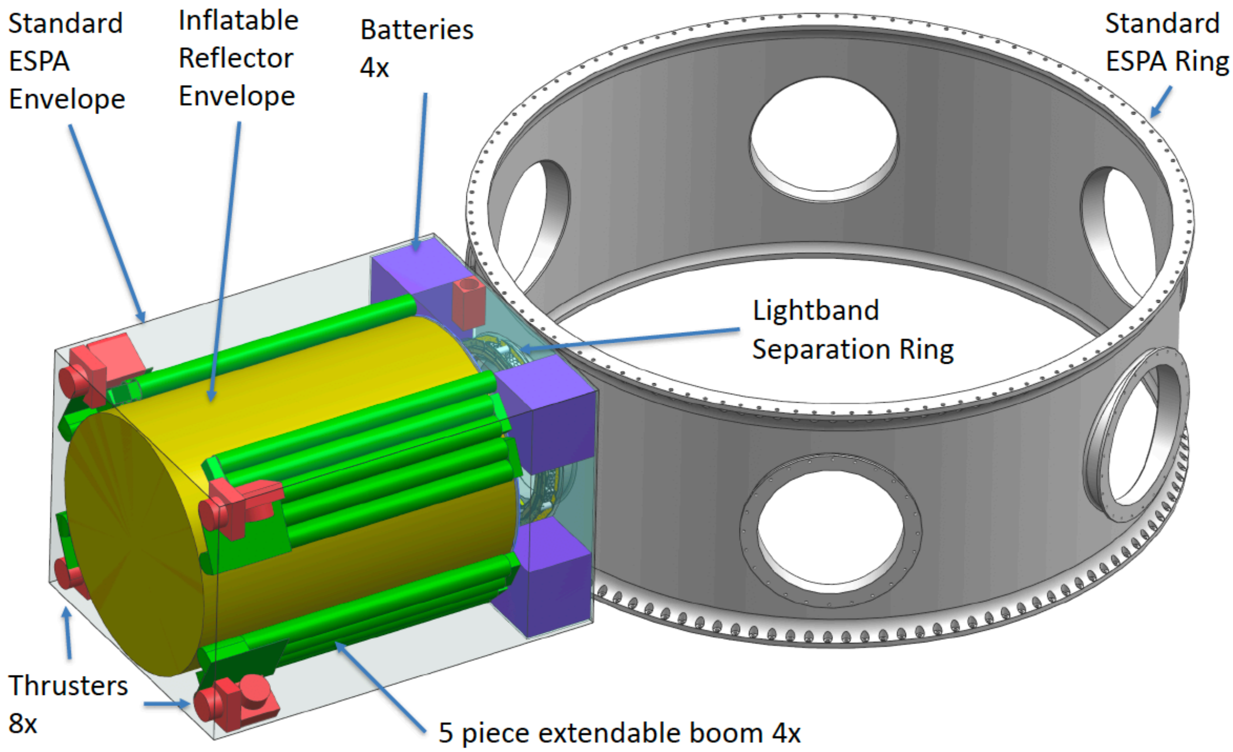


Figure 24. One or more OS4 spacecraft could be attached to an ESPA or ESPA Grande ring. The interface is attached to the Hub (not called out in this illustration), which attaches to the Lighjtband Separation Ring. The Inflatables Packaging cannister is the gold-colored cylinder. The "5 piece extendable boom" shown would be replaced by the Deployable Boom type referred to in Figure 19. Team Xc CAD illustration.

Propulsion

As an expedient to get needed definition completed during two 4-hour Team Xc sessions, it was assumed that the post-launch Deep Space Maneuver could be accomplished using a propulsive ESPA ring such as the Moog Orbital Maneuvering Vehicle (see prior section). This arrangement essentially uses the OMV to supply all operations capability until shortly after the DSM. Another option examined, that could potentially save significant mass and possibly some cost if multiple OS4 units were to be built for different launches, is to add a propulsive stage between a non-propulsive ESPA ring and the rest of the OS4. The example considered was a variant of the Nanosat Hypergolic Propulsion System concept (Figure 25) developed by Stellar Exploration, Inc, including miniature electric pump-fed qualified hydrazine/nitrogen tetroxide (NTO) 3 N thrusters.[29] Another example considered used a small solid rocket motor. The Biprop system was considered superior to the solid rocket motor because of its ability to provide its own thrust vector control by off- and on-pulsing across its set of 4 thrusters.

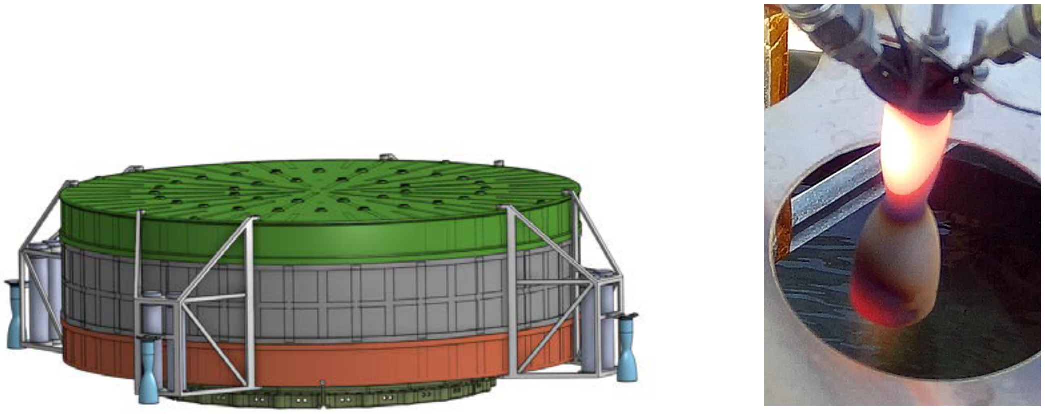


Figure 25. An alternative bipropellant kick stage configuration was considered in place of a propulsive ESPA ring. The Stellar Exploration CAD concept shown is similar to a size that could provide the needed delta-V to the OS4, but does not show how the structure could be modified to also serve as the structural connection between the Hub and ESPA ring interface. A hot firing of an associated miniature bipropellant thruster is shown. CAD illustration and photo from the referenced JANNAF 2019 paper, used by permission from Stellar Exploration, Inc.

A very low-thrust capability with well-controlled minimum impulse bits was desired for the outer Solar System, for spin-up/spin-down, attitude maneauvering, and potentially small trajectory correction maneuvers (TCM). Pulsed plasma thrusters (PPT) were selected for our example mission primarily because of their apparent ability to manage their mechanically simple solid propellant without high sensitivity to cold temperatures.

Our example design settled on:

- Qty (8) Busek BmP-220 thrusters[30] would be mounted in 4 clusters of 2 each.
- Clusters mounted 90 degrees apart on the circumference of the inflatable Torus, in each Outrigger.
- Each Cluster would consist of an axial thruster and an orthogonal radial thruster;
    - Axial Thrusters for TCM and slewing.
    - Radial Thrusters for spin-up and spin down.
- Thrust values could vary, but 0.35 mN per thruster was assumed.
- Propellant Loads –To Be Reviewed and evaluated;
    - 200 g Teflon for each axial thruster (this would require to a re-design/development effort, or need for multiple thrusters used in sequence).
    - 40 g Teflon (nominal load) for each orthogonal thruster.
- Beginning-of-life (BOL) thruster system mass w/ 40 g PTFE load is 0.5 kg, per manufacturer spec sheet.
- Busek has flight heritage with PPT thrusters;
    - Power demand for this thruster appears considerably less than other offerings surveyed.
    - 7.5W per unit (latest spec sheet referenced states 3 W per unit) versus 50W per unit.

Attitude Control

Spin stabilization was chosen because of its apparent ability to passively stabilize while Sun-pointed. While this was the baseline, a future investigation should perform a trade study between this option and 3-axis stabilization. Since periodic attitude maneuvers are required to accommodate the changing geometry of the Sun and Earth positions along the OS4's hyperbolic trajectory, and to accommodate telecommunications, it may turn out that a 3-axis solution is superior in important respects to the chosen baseline.

The attitude control system (ACS) of OS4 would be responsible for slewing between charging batteries via solar power and communication with the Earth. There must be a +/- 0.1° accuracy of knowledge per axis and +/- 0.3° accuracy for pointing control in each axis for downlink capabilities. ACS would also support minor trajectory correction maneuvers (TCM) and be capable of spin stabilization. The spacecraft will be spin stabilized along the center axis (the line to the Sun in Figure 1). At this point it is assumed that OS4 will achieve stability at 1 rpm after 1 month. This number will be verified by further analysis as the design matures. The current configuration for the ACS thrusters would be four groupings of two pulsed plasma thrusters (PPT) at each Outrigger. Each group would have one axial thruster and one radial thruster. All four axial thrusters would point along the spin axis in order to provide the thrust for slewing and minor TCMs. Two of the radial thrusters would be used for spin up and the other two for spin down.

In order to slew as a spin stabilized system, without precession or nutation, the system is assumed to be symmetric about the spin axis, therefore having no products of inertia, and such that $I_{axial} >> I_{transverse}$, where I_{axial} is the moment of inertia about the spin axis and $I_{transverse}$ is the transverse moment of inertia. The spin axis is also assumed to be a principal axis. These are acceptable assumptions for the purpose of determining a time estimate for slewing. However, based on a mass model of the system, I_{axial} was found to be 291 kg-m^2 and $I_{transverse}$ to be 363 kg-m^2 meaning the constraint of $I_{axial} >> I_{transverse}$ is not met, by far, for the current design. As the mechanical design matures, further analysis into the need for ballast masses may be required. A different architecture might be devised to distribute most of the mass to the periphery of a spinning extremity. Alternatively, a switch to 3-axis stabilization might eliminate the problem, if that is not infeasible for other reasons. (We note that the *Voyager* spacecraft are 3-axis stabilized.)

Rotational inertia comes primarily from mass of 4 Outriggers (1 kg each), Torus (~4.5 kg w/ Kapton, rigidizing fluid, tensioning strings, fasteners, diffuse white skirt), Paraboloidal Metalized Membrane Reflector (PMMR) and Clear Reflector Canopy (~7 kg distributed from spin axis out to 5 m diameter). Transverse inertia comes primarily from the Hub.

The following design parameters were utilized or determined:

- The ACS PPTs would provide 4 hours of radial thrust per spin up or spin down and 450 cumulative hours of axial thrust for slewing during the entire mission.

- Mission design estimated an extra 3 m/s delta V for TCMs, equaling approximately 120 more hours of axial thrust broken down into 1 m/s burns.
- The selected example Busek BmP-220 PPTs would carry 40 g of propellant and have 220 N-s of total impulse.
- Based on the thrust capabilities, 0.35mN, and the total impulse, a total of 628,500 seconds can be provided by the Busek BmP-220 thruster, or 175 hours.
- This provides enough total impulse for many spin up and down maneuvers, but would need to be 3.5x this for the axial thrusters.
- To achieve the axial requirements the PPTs would need around 140 g of Teflon, which would require thruster modification or redesign and qualification.
- ACS thrusters are to be the sole propulsive system following detachment from the OMV, and responsible for stabilization following the separation.
- Analysis has not been performed of the need to offset disturbance torques, e.g., from solar pressure asymmetries.
- Radiation testing for Teflon at close to -50C and perhaps colder will need to be done.

In order to maintain pointing during telecommunications, knowledge of the spacecraft's attitude would need to be accurate to within +/-0.1 degrees per axis. Results from the NIAC-OS4 Team Xc session indicate that two IMUs, STIM300s produced by Sensonor, as well as two star trackers would be necessary in order to attain this level of accuracy. Further development in star tracker technology is required to meet the requirements of OS4. The nominal rotation rate of the spacecraft is at least 6 deg/sec, however the BCT NanoSliceStar Tracker is only rated up to 4 deg/sec slew rate thus not assuredly capable of tracking our attitude accurately. To satisfy the attitude sensing requirements it may be necessary that a star tracker be developed for small satellites that can perform at the desired slew rates. Another option would be development of a star scanner for small satellites. The Mars 2020 star scanner produced by Ball Aerospace heritage and modified by JPL Section 323 is capable of operating at 12 deg/s nominally, however it is not suitable for use on a smallsat.

The approach used to estimate the total mass of propellant needed for each axial and radial thruster was determining the total spin up and slew time. Spin up time will be constant throughout the life of the mission. Given OS4 would need to maneuver between charging batteries from the Sun and communication with Earth, as OS4 travels away, the angular distance between these bodies generally decreases, and so does the angle change for the slew. This was taken into account in determining the total slew time required. The time for spin up/down was calculated based on the target angular momentum, H_t, and torque imparted on the body by two of the axial thrusters. Assuming no initial momentum,

$$H_t = I_{axial} \cdot \omega_s$$

where ω_s is the angular velocity of the spin stabilization, 1 rpm, and a spin up/down time,

$$t_s = \frac{H_t}{N_{PPT} \cdot F_{PPT} \cdot L_R}$$

where N_{PPT} is the number of thrusters (2 radial), F_{PPT} is the force per thruster and L_R is the radial moment arm. For the PPTs selected the thrust values very, but for analysis purposes 0.35mN was assumed. From this, the time required for one spin up or down would be around 4 hours. For the slew calculations, turn time,

$$t_s = \frac{\Delta\theta \cdot H_t}{N_{PPT} \cdot F_{PPT} \cdot L_R}$$

is based on the target angular momentum above, 4 axial thrusters, a slew angle of ΔΘ and a duty cycle of 12.5%, which translates to an 11.25° angular sector for each PPT pulse. While the spacecraft spins, each thruster would pulse with appropriate timing to reach the orientation required for the slew.

In order to calculate the necessary propellant for the mission, the cumulative sum of slew angles for the entire mission was calculated. The slew angle could be approximated by finding the angle between the Earth and the Sun from the perspective of the spacecraft. The majority of the spacecraft's lifetime will be spent with the solar reflector pointing near the Sun, however to communicate with the DSN it must slew to within 0.3 degrees of Earth. Tabulating the slews the spacecraft must make over its lifetime provides an estimate for the propellant required for the mission. Figure 26 shows the cumulative angle change and the variations to track Earth after the OS4 starts tracking the Sun at 6 AU, for an idealized Solar System escape via Jupiter flyby trajectory.

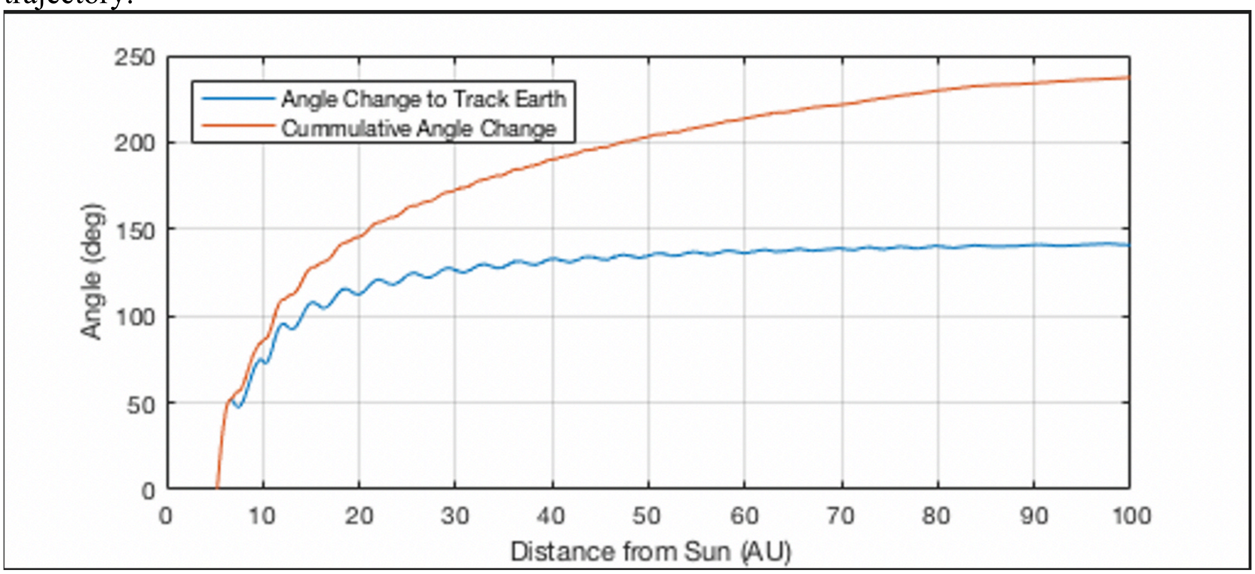


Figure 26. Cumulative angle change and the variations to track Earth after the OS4 starts tracking the Sun at 6 AU, for an idealized Solar System escape via Jupiter flyby trajectory.

Following the methodology described above over the range of slew angles given over the lifetime, the total slew time would be approximately 440 hours, rounded to 450 hrs. This neglects the initial angles greater than 25° as they would happen within 1.25 AU, at which point OS4 would still be dependent on the propulsive ESPA. This would significantly reduce the demands of the ACS thruster.

Within 6 AU from the Sun, there is a "keep out zone" around the Sun where the solar reflector should not point due to the risk of overheating the Hub. For the example trajectory, which is near the ecliptic plane, there is every year a period of solar interference where the Sun is between the spacecraft and Earth and interferes with radio communication.

Operations Concept Overview

The example launch phase operations sequence could be as follows:

- Packaged spacecraft accommodated as secondary payload aboard launch vehicle with a primary payload bound for Jupiter.
- Delivered to launch site >=3 months before primary mission launch.
- Verified safe and alive before launch. Clock set on ground.
- L+ tbd minutes: Primary spacecraft separates from launch vehicle
- Primary Spacecraft Separation + tbd minutes: OS4 separates.
- L+2 hrs: OS4 (or its host OMV-like propulsive ESPA ring) orients independently to power-positive, telecom enabled. (events shown below are as if OS4 is not hosted by an OMV, but must operate independently. In the case of hosting by an OMV, similar functions would need to be provided by that vehicle, but were judged to be within its capability with one (or perhaps more) OS4 as its payload.)
    - Battery provides power until this time; max 100W since launch
    - ADCS using, e.g., star camera, reaction wheels, target orientation loaded on ground into software memory.
    - Power provided by Inner Solar System Solar Panel.
    - 3-axis stabilized to +/-1 deg.
- Telecom via low-gain antenna(s) to be geometrically tolerant, housekeeping-only downlink, command uplink to set up for kick stage burn.
    - 1 kbps assumed adequate both uplink & downlink.
- Tracking, telemetry and commanding continues to Biprop Kick Stage Deep Space Maneuver (DSM) burn at L+30 – 60 days.
- Verify navigation, system health.
- Orient OS4 to Biprop Kick Stage DSM burn direction.
- T+30 – 60 days: Fire to specified delta-V in range of 200 – 1000 m/sec (would be known >1 yr before launch).
- Track and verify, calculate clean-up maneuver.
- DSM+~5 days: Perform cleanup maneuver.
- Track & verify.
- DSM+~8 days: Deploy Inflatables.
    a) Point to attitude favorable for inflation.
    b) Jettison Inflatables Packaging cover.
    c) Inflate Torus, then volume between Parabolic Metalized Membrane Reflector (Parabola) and Clear Reflector Canopy (Canopy).
    d) UV harden (rotate on two axes to cure all surfaces; keep metalized

reflector portion of Parabola pointed >15 deg from Sun).

e) Stabilize in starting orientation, pointing mirror >15 deg off-sun while close to 1+ AU

- Track & verify.
- T+40 – 70 days (DSM +~10 days): end Launch/Deployment Phase

Inner Solar System Cruise Phase operations could be as follows:

- Switch to Medium Gain Antenna (MGA - if needed) as distance from Earth increases.
- Spin up to 1 rpm along axis that keeps Earth in sight of MGA more or less continuously.
- Calibrate torques and responses to dump excess momentum against solar pressure asymmetries (as done with Mars Cubesat One (MarCO)).
- If necessary, dump momentum against pulsed plasma thrusters.
- Re-orient spin axis (1/month) as Earth's angular position in sky changes, to enable monthly DSN pass with PMMR pointed at Earth +/- 0.3 deg for X-band link to 34 m stations.
    - There would be energy to do downlink passes more frequently than 1/mo while closer to the Sun than 30 AU. However, spin-up/spin-down and attitude maneuvering propellant for the PPTs would also need to be adequate. For this example mission, the one telecom session every 30 days cadence was assumed.
- Collect instrument data and transmit in 1 8-hr DSN pass/month.
- Total F&P instrument data rate = 2300 bps. Assume s/c housekeeping is same.
- In order to fit within 1 DSN pass/month, sample data at regular intervals (that lengthen as telecom performance decreases with distance) to fit available data volume.
- Provide data overlap between downlink passes to ensure that >98% of science data is received at least once on the ground.
- Don't point within 5 deg (tbc) of Sun, in order to avoid overheating Outer Solar System Solar Panel.
- When this prohibits a DSN pass, either utilize MGA (and lower data rate), or summarize data for including during next available downlink pass.

With our example mission launching from Earth in 2038, for trajectory corrections, we consider 5 TCM's as presented in Table 9 below.

Table 9. Example trajectory correction maneuver (TCM) plan for OS4. Their estimated magnitudes after separation of the propulsive ESPA ring appear small enough to be executed with the example pulsed plasma thrusters.

Maneuver	Epoch	Magnitude (m/s)	Engine	Reason
TCM-1	L+15	10	ESPA Ring	Injection Covariance matrix uncertainty clean up
TCM-2	L+50	2	ESPA Ring	TCM-1 clean up
Jupiter-Approach 1	Jupiter – 180 days	1	Plasma	Targeting Jupiter flyby
Jupiter-Approach 2	Jupiter – 90 days	1	Plasma	Approach 1 clean up and targeting Jupiter flyby
Jupiter-Approach 3	Jupiter – 15 days	1	Plasma	Approach 2 clean up and targeting Jupiter flyby

Outer Solar System Cruise Phase operations could be as follows:

- Keep spinning at 1 rpm along axis now pointed at Sun.
- Solar torques may now be negligible?
- Because Earth is farther away (approaching and then passing 10 AU), attitude excursions between Sun-point (for power) and Earth-point (for telecom) will become smaller and keep shrinking.
- Each month, re-orient spin axis as Earth's angular position in sky changes, to enable monthly DSN pass with Parabola pointed at Earth +/- 0.3 deg for X-band link to 34 m stations.
- Then point back to Sun to continue collecting energy.
    - Point so that for the entire month until the next monthly downlink pass, sunlight will provide enough energy.
    - So, this may involve pointing the s/c spin axis at a midpoint along the line of where the Sun will appear from the s/c in inertial space over the course of the upcoming month.
- Collect instrument data.
- Total field & particles (F&P) instrument data rate = 2300 bps gets stored into memory. Assume s/c housekeeping is same.
- In order to fit within 1 DSN pass/month, sample data at regular intervals (that lengthen as telecom performance decreases with distance) to fit available data volume.
- (Option considered for Heliophysics mission): Two years into Outer Solar System Cruise, make last uplink to spacecraft. Ops now autonomous. Modeling based on measured performance now predicts s/c behavior for ground to schedule and set parameters for DSN passes.
- (Option considered for Heliophysics mission): Drop all but 100 bps s/c housekeeping. This remainder is for failure & degradation understanding. If the uplink option is not maintained, then with no uplink, there is nothing the ground can do to send commands to correct a situation.

- When collected energy can no longer support one full-duration (8-hr) DSN pass, reduce pass duration to fit available energy. Target minimum duration is 90 minutes once/month at 125 AU end-of-mission (eom). For missions ending at 30 AU because of thermal constraints, this minimum DSN pass duration is likely to be much longer.

Instrument Operations: power, energy, and bits

Full autonomy of this operations concept is well within flight proven (TRL 9) technology such as operated the Earth Observing One mission for over a dozen years as the Autonomous Sciencecraft Experiment.[31] Instrument operations could proceed as follows:

- Instruments, attitude control, computing, and other electronics wake up, make measurements, operating <=15 minutes every other day at average total 16 W(e) for this duration, then go back to Sleep Mode.
    - Can fractionate the allocation.
    - Assume Mag, IES & MeRIT 160 W-minutes every other day, except every 10th day, which is reserved for Dust Camera.
    - Dust Camera 40 W-minutes every 10th day.
    - All other functions use 60 W-minutes every other day, except every 10th day.
    - All other functions use 180 W-minutes every 10th day.
    - Reserve 20 w-min every other day, except reserve 60 W-min every 10th day.
- For IES, MeRIT, assume 1 kbps coming out of 1 IES and 1 MeRIT, 10W operating power each instrument.
- For Magnetometer, assume 100 bps coming out of 4 sensors simultaneously, 2.5 W operating power for each sensor.
- For camera: assume EECAM, read out and store one frame 20 Mpixels, 8 bits deep.
- Instrument data processing
    - Assuming (to start with) IES, MeRIT, and Mag each can operate 8 min every other day…(If this breaks the energy allocation, reduce instrument on-time by 1+ minute).
    - Then in every 10-day interval (4 measurement days for these instruments), 32 minutes of operation generating 1000 (IES) + 1000(MeRIT) + 4*100(Mag) = 2400 bits/sec ➔ 19,200 bits fields & particles data per 10-day "frame".
    - 3 Frames make 1 month = 57,600 bits *plus parity, overhead, repeating some data down from prior transmissions, etc.* to downlink.
    - Each downlink pass (at 125 AU) has a capability of 108 kbits (20 bps x 90 minutes). (Downlink passes at 30 AU have much higher capacity.)
    - For Dust Camera, onboard processing examines present and prior pictures, downlinking location and size of new holes by seeing Sun through Parabolic Metalized Membrane Reflector (PMMR). TBD bits required for this. If too much, will reduce the Dust Camera measurement frequency.

Solar System Portrait Camera

- Takes a portrait every year starting Year 10.

- Assume same EECAM as Dust Camera, but looks directly at Sun. From beyond 30 AU, Sun is a really bright star; might use crude occulting disk. Makes multiple exposures to get light level right for different bodies and to handle slow s/c rotation, sort out noise from stellar and planetary images.
- Image processing to determine centroid, number of pixels covered (one in case of most bodies, plus bleed and smear), brightness, and color. Downlink is this information, not any of the pictures.
- Data processing done onboard.

Rideshare Opportunities and Trajectory Availability to Outer Solar System

In order to stay within the SmallSat paradigm including affordable launch, we would need to be carried as a secondary payload, thereby avoiding most of the cost of a dedicated launch. Thus we need to assume that there will be at least an occasional primary mission to the Outer Solar System that can accept a secondary payload of our mass (ESPA Ring + OMV + OS4). Our analysis indicated that, assuming that we rideshare aboard a launch of something on its way to Jupiter:

A. Uranus or Neptune trajectories come around once every ~12 yr (Jupiter's orbital period). There appear to be opportunities in early 2030s then mid 2040s.
B. Neptune should be available at least once during the two decades 2030 - 2049, and we can time our trajectory to get a close up of Triton terra incognita.
C. There are enough trans-Neptunian objects (TNO) that we could go by one if we are not too picky about which one.
D. If a constellation of a few OS4's were to go out on a single launch, then they could spread out at Jupiter, but likely wouldn't spread out more than 90 deg in right ascension.
E. While we did not perform this specific analysis, there are probably trajectories to Jupiter Trojans and Centaurs, and also Hildas in the outer Main Belt that are also of scientific interest. A trajectory search would be needed to determine what's actually available. It may require more delta-V to intercept one of these via targeted jovian flyby.
F. The trajectory driver here is entry speed for the primary mission at its destination, which an OS4 sponsor won't have much control over. That is why we need the DSM.

For our example mission trajectory, we chose one of the trajectories launching to Jupiter in 2038, such as might be used by a primary mission to Europa or another of the Galilean satellites. Two views of mission opportunities the latter 2030s can be seen in Figure 27.

A significant factor in choosing the Jupiter 2038 example was the modeled radiation exposure during the Jupiter flyby, as shown in Figure 28.[32] This example limited total ionizing dose exposure to <175 krad between the contemplated 200 to 1000 m/sec deep space maneuver.

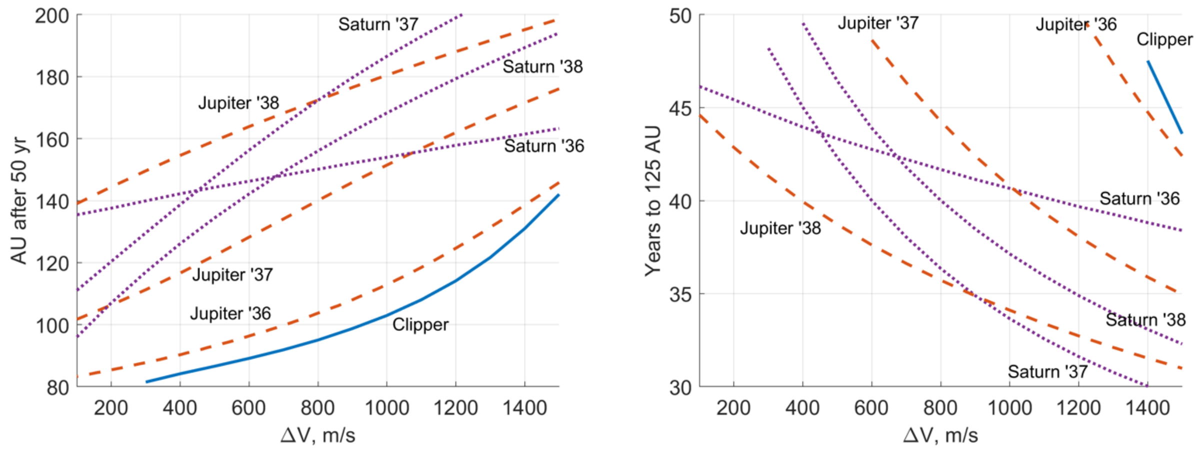


Figure 27. Time to heliocentric distance is shown for a family of trajectories that primary missions might launch on to Jupiter and Saturn during the later 2030s. OS4, as a ridesharing secondary payload, could apply the noted ΔV in order to achieve the noted time to distance. Primary missions would be expected to utilize trajectories minimizing approach velocity at their destination body, which is not optimal for a secondary payload aiming at a more distant destination. An example of this can be seen where one of the Europa Clipper reference trajectories (in the 2020s) is shown in blue in the first figure as if that mission did not insert into Jupiter orbit, but simply flew by. It can be seen that of the trajectories shown, the Clipper reference trajectory gains the least speed toward escape after the application of the delta-V noted on the x axis 30 – 60 days after Earth departure.

Figure 29 and Figure 30 show our trajectory when remaining in the ecliptic for 200 and 1000 m/sec deep space maneuvers, respectively. Dates shown are for crossing the orbits of the noted planets.

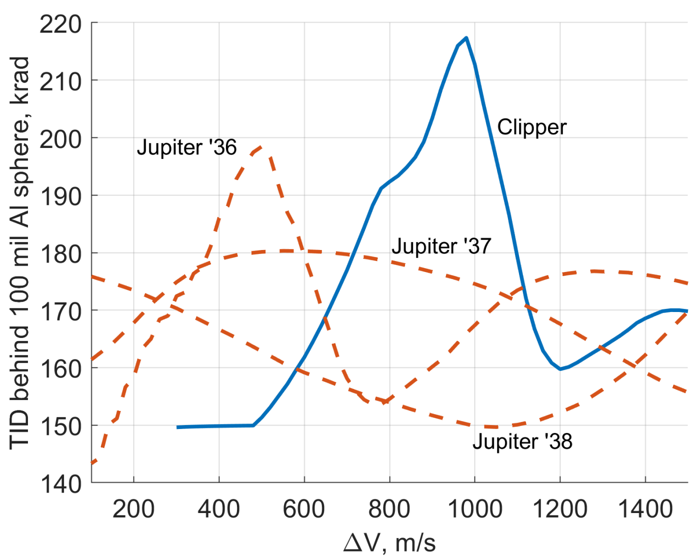


Figure 28. For the example Jupiter 2038 example mission trajectory, the reference radiation dose can vary from 150 - 175 krad behind 100 mil Al, depending on the magnitude of the deep space maneuver.

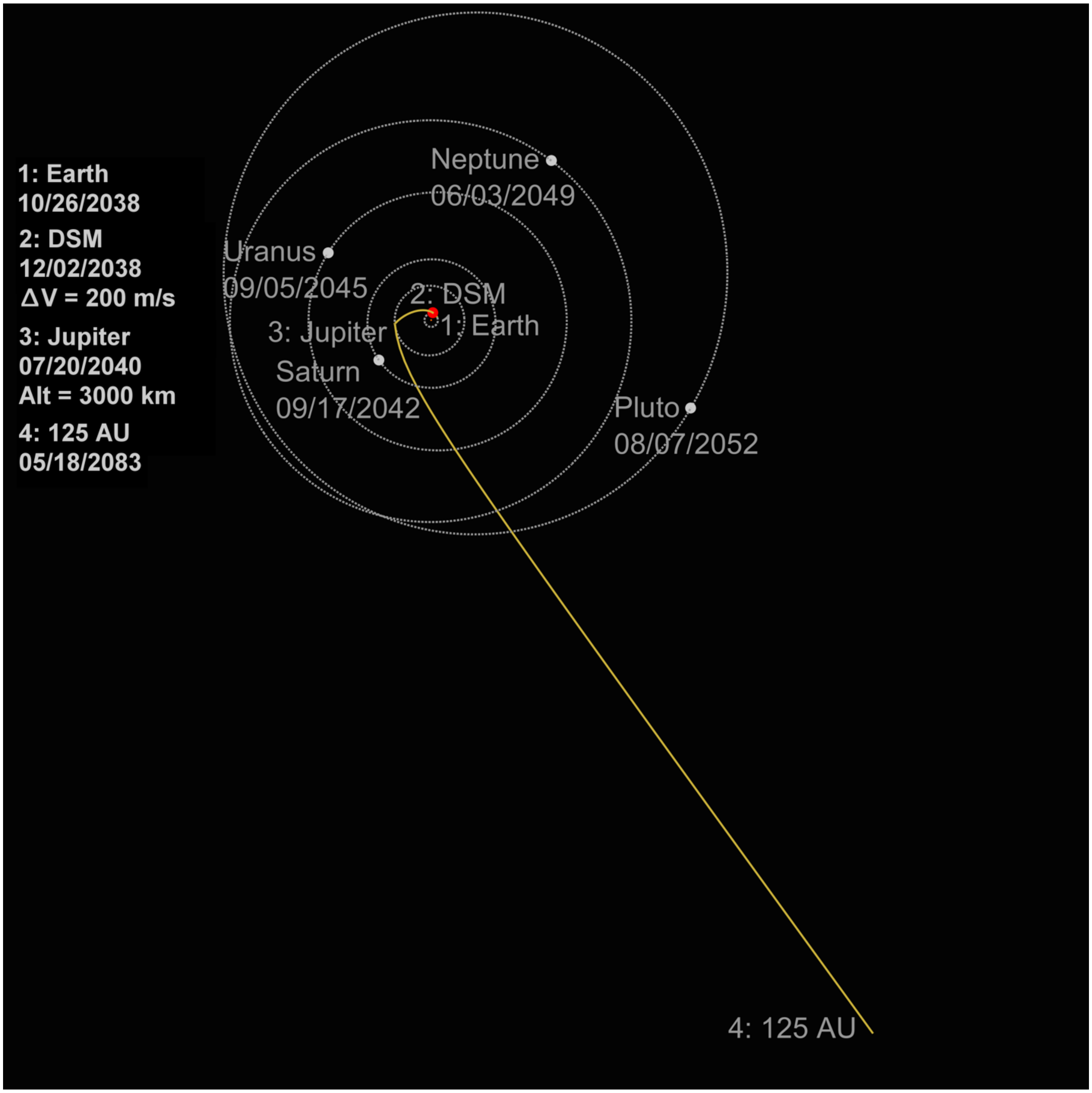


Figure 29 and the next one show the same Earth launch date, but with different deep space maneuver (DSM) magnitudes. Shown here is an example 200 m/s DSM, with associated dates for crossing the orbits of the outer planets, and reaching 125 AU.

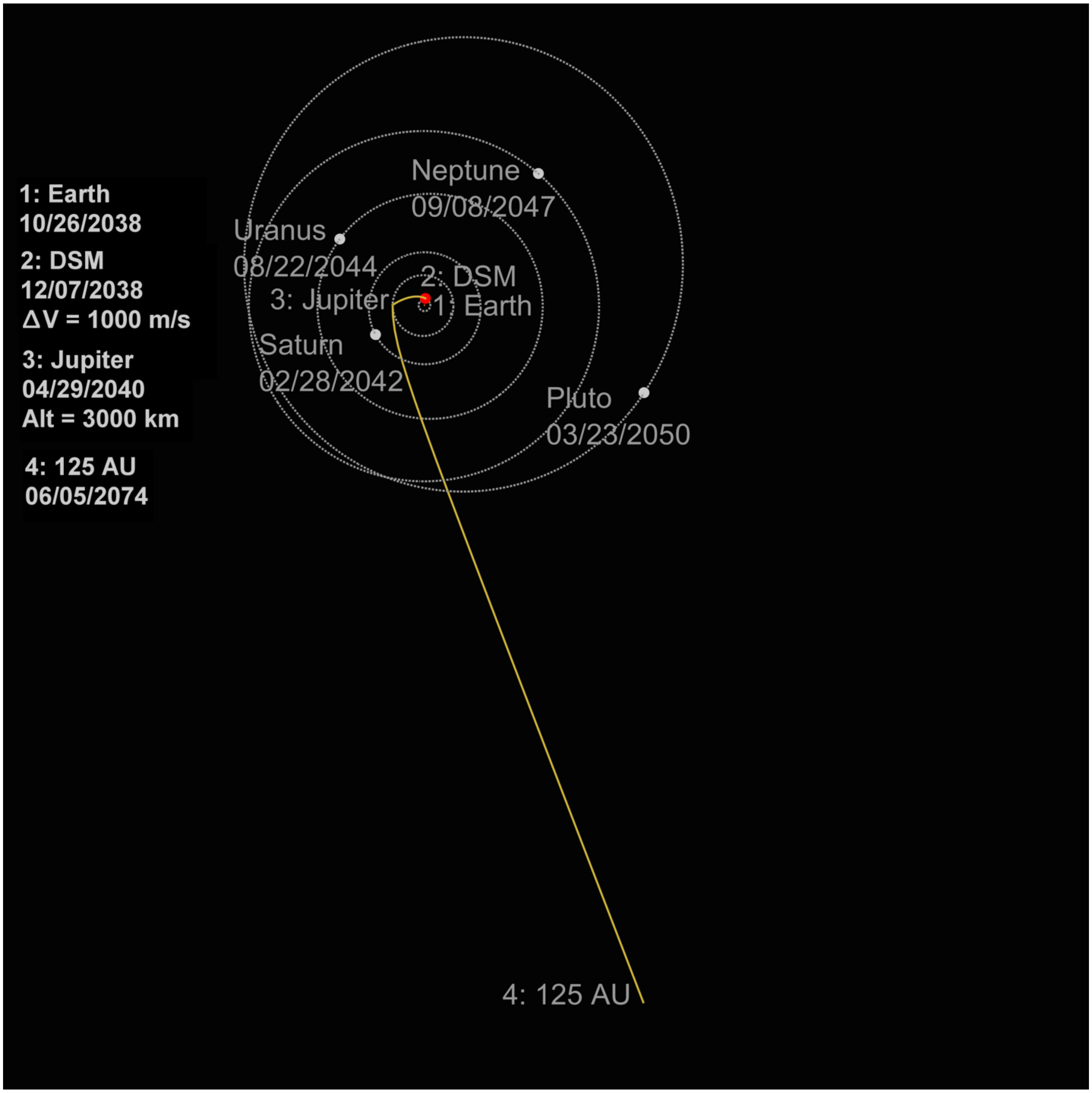


Figure 30 and the prior one show the same Earth launch date, but with different deep space maneuver (DSM) magnitudes. Shown here is an example 1000 m/s DSM, with its earlier associated dates for crossing the orbits of the outer planets, and reaching 125 AU.

For this Jupiter 2038 trajectory, there is limited maneuvering capability outside the ecliptic, as shown in Figure 31, for the same two magnitudes of deep space maneuver.

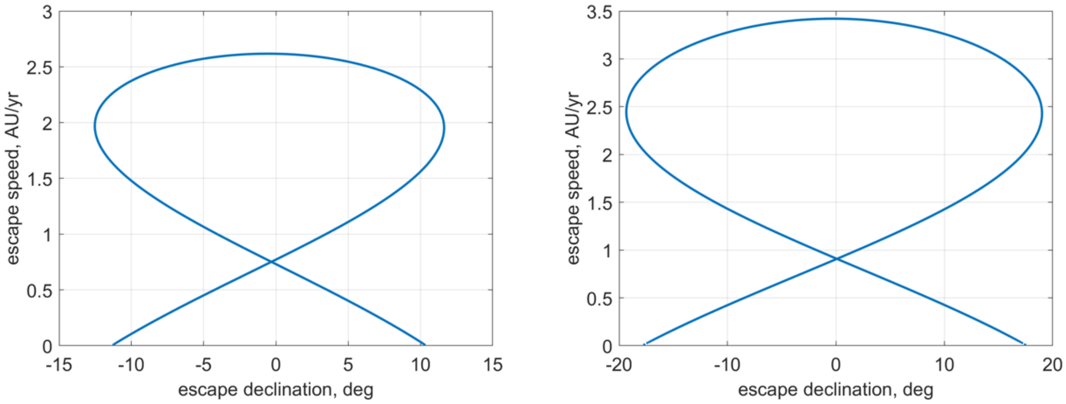


Figure 31. Achievable out-of-ecliptic escape declinations are shown for a 200 m/sec (left) and 1000 m/sec (right) deep space maneuver. It can be seen that escape speed can be traded for achievable escape declination, up to a limit.

Our configuration target was for the OS4 to stay within the ESPA secondary payload standard envelope. Mission-specific allocations beyond this envelope may be possible, but we chose to try to fit the standard as a starting point. Figure 24 shows a tight fit developed during the Team Xc session.

III. REQUIRED TECHNOLOGY ADVANCEMENTS

There are four technologies that would need to be advanced beyond the present state-of-the-art in order to enable the concept of Outer Solar System SmallSats (OS4) as described in this example. A fifth would be useful, but is not viewed as essential. While solutions might be possible that are quite different from the example noted here, our intent is to show a path by which the outer Solar System could be visited and explored by scientifically productive SmallSats fulfilling focused objectives. It is not our intent to suggest that solar-powered SmallSats can take the place of radioisotope-powered missions that address a much broader range of scientific objectives with a single mission. The combination of the two types of missions would significantly broaden and accelerate the range of objectives that could be affordably sought in the outer Solar System.

Cold-tolerant equipment:

> In the outer Solar System, everything that is not actively heated will become cold. There is also a significant range of cold temperatures over which equipment would need to operate. Rapid and frequent changes in temperature are not expected, making this challenge likely to be somewhat easier to overcome than for missions, e.g., on the martian surface, where many wide temperature cycles are required over a mission's duration.
> Not everything can be contained in the Hub with its thermal enclosures. This applies primarily to the instruments (especially the Magnetometer and Dust Camera), and the pulsed plasma thrusters.

Schemes might also be valuable where only the smallest volumes of electronics and other sensitive componentry would be maintained at adequate temperatures by micro-heating, while the larger housing is connected in a manner to avoid mechanical failure from differential thermal expansion effects.

Inflatables:

The proposed scheme for rigidizing the inflatables would need to be matured, involving the Venous Web, UV-rigidizing liquid, folding, storage, and deployment over time, etc.

For rigidizing inflatables: there would be a need for a slow-curing alternative to Industrial Adhesives Dymax Multi-Cure 6-621 Series,[33] which is said to cure in seconds. There may already be slow-curing fluids that could be used; we did not do a thorough search.

All surfaces of inflatables should be conductive if possible to avoid large non-conductive areas with possible build-up of charge. This may not be possible for the Clear Canopy, as it needs to be RF- and visible-transparent. In this case, a method of dissipating charge buildup would likely need to be sought.

Radiation tolerance will need to be verified for the Jupiter flyby, especially regarding degradation of reflective and clear surfaces.

Inflatables as a method of creating large, moderate-precision structures with very low mass could be valuable for a variety of space applications needing large apertures, especially for radio frequency applications.

Attitude Sensing:

The example rotation rate of the spacecraft is 6 deg/s, however the current BCT NanoSliceStar Tracker is rated to track up to 4 deg/s slew rate and would thus not be assuredly capable of tracking our attitude accurately. To satisfy the attitude determination requirement, a star tracker suitable for small satellites would need to be qualified or developed that can perform at the desired slew rates. If a 3-axis attitude control approach were to prove feasible, then this attitude sensing advancement would not be required.

Longer-life and low operating temperature Pulsed Plasma Thrusters:

The example BmP-220 carries 40g of propellant and achieves 220 N-s of total impulse. Given assumed thrust level and performance, the throughput would have to be increased Stick geometry will likely require re-design.

Radiation testing for Teflon and thruster performance at these temperatures is unknown and would need to be tested.

TCMs would require an estimated 3 m/s delta V, consuming approximately 430,000 sec of on time. Note that the TCMs are broken up into 1 m/s burns, which will take approximately 40 hours to complete.

ACS would require approximately 2,500,000 seconds of on time per axial thruster for turns, putting the total requirement at approximately 3,000,000 seconds
This would be 5x the standard life of the thruster, so additional development would be required, or additional thrusters sequenced through the mission. In terms of raw propellant, this will only add 160g of prop to each thruster.

The PPU would be a significant portion of the existing mass, so assume that total wet mass increases from 0.5 to 1kg, but that should be subject of an early trade study.

The high radiation environment at Jupiter should be considered.

Clear Canopy jettison:

While not necessarily essential, it was suggested that a method be found to cut away and jettison the Clear Canopy after all the Inflatables are inflated and rigidized by solar UV. The Clear Canopy would be needed during inflation to enclose a volume of the sublimated gas such that the Parabolic Metalized Membrane Reflector (PMMR) takes its shape determined by the gores and seams out of which it is assembled, as with the larger Spartan 207 Shuttle experiment. But unlike Spartan 207, the OS4 structure would self-rigidize, not needing the sublimated gas to keep its shape stabilized. Thus, if the Clear Canopy could be cleanly jettisoned after the PMMR has rigidized, the losses from two passes of light and RF waves through the Clear Canopy could be eliminated, boosting system efficiency. We did not attempt to devise a scheme to jettison the canopy, and so accepted that it would introduce a factor of 80% in our sunlight efficiency chain.

Our concept for focused, affordable outer Solar System exploration differs significantly from alternative approaches, three of which are noted below:

A) The *Voyagers*, *Pioneers 10/11* & *New Horizons*, while dramatically more comprehensive in their single mission investigation capability, all needed radioisotope power systems (RPS), and cost >$300M plus dedicated launch;
B) Nosinov NIAC describing interstellar solar sails needing RPS, with s/c >>100 kg, requiring 0.2 AU perihelion and resulting 25-Sun thermal load, with cost estimated >$3B;[34] and
C) Pete Worden/Russian-backed StarChip laser sails ("a few grams at most")[35] requires military-class lasers, would require unbelievably small instruments, and may not work.

We consider our architecture, if matured, to provide a new branch of outer System Exploration capability characterized by lower development cost and/or technology advancements that are less demanding than the above alternatives. We are not suggesting the elimination of any of those alternatives, merely that the OS4 approach be developed further to open new a new architectural option.

IV. PATH FORWARD TOWARD PROJECT IMPLEMENTATION

Including Student, Public and International Engagement

The progress made advancing the OS4 concept would not have been possible without the involvement of the 11 students attributed on the title page. All of them were students actively involved in the California Polytechnic State University-San Luis Obispo (Cal Poly) CubeSat Lab at the time of their contribution. As a member of the California State University system, Cal Poly is prohibited from offering doctorate degrees. As a direct result, our team consisted of one master's student and nine undergraduates. In particular, this work provided exposure and experience to the undergraduates that would have been very scarce at larger research institutions. The students successfully researched and iterated on the OS4 concept, providing critical input to the Team Xc session. A subset of them were able to actively participate in the two-day session. All the students reported the session as being both productive for the OS4 design and a very educational experience for them personally.

Further noteworthy is the role that then-student Kian Crowley, at Cal Poly, played before our NIAC proposal was prepared, investigating a 100 AU no-radioisotope mission concept for his MS thesis, completed in 2018 June.

The OS4 project has also provided the opportunity for students to engage with the broader public via conferences and workshops. The Cal Poly student lead for the project, an undergraduate, was instrumental in putting together an OS4 presentation for the 2020 Inter-Planetary Small Satellite Conference. He also gave half of the presentation during the conference itself. In addition, the CubeSat Lab's involvement in the project and interim results are included in posters presented at various venues, including the annual Small Satellite Conference in Logan, Utah, and the talking points for interactions with VIPs and collaborators. This provides the opportunity for many students indirectly involved in the project to understand and convey the exciting possibilities of OS4.

The path toward implementation of OS4 will continue to include student involvement and public engagement on many levels. As a key collaborator in this work, Cal Poly has a long history of involving both undergraduates and masters students in research. In addition to the direct involvement of Cal Poly students, the lab is also active in training the broader small satellite community through general events such as the annual CubeSat Developers Workshop (www.CubeSat.org) and more tailored multi-day in-person training courses. The OS4 team intends to leverage this as part of Phase II by involving more university collaborators focused on specific aspects of the spacecraft design. If this proceeds, the team will use a combination of workshops, focused training, and advisement to bring new participants up to speed on necessary background information they may not have at the outset. It will also continue to use the project as a tool to engage a large public audience and provide networking opportunities for students. This will happen through participation in both domestic venues, such as the Small Satellite Conference, and international venues such as the International Astronautical Congress (IAC) and the Committee on Space Research (COSPAR) assemblies.

Once the basic technology is developed for Outer Solar System SmallSats, it is probable that other countries will want to "join the club" of nations exploring the outer Solar System, just as new-to-space nations have in the last two decades made the financial and intellectual investments to build CubeSats, and even mount missions to the Moon and Mars. As with Interplanetary CubeSats,[36] OS4s will lower the barriers to performing serious science investigations at multiple destinations. Some of those non-U.S. entities seeking to do this will choose to purchase or trade for elements of their missions with U.S. businesses and universities. In addition, they are likely to contribute to a pool of scientifically useful data that the U.S. alone would not collect, thereby hastening the scientific community's investigation of the Solar System's many mysteries waiting to be found and resolved.

V. ACKNOWLEDGEMENTS

The idea on which this investigation was based, that it might be possible to create useful wholly solar-powered Outer Solar System SmallSats (OS4), came from Prof. Jordi Puig-Suari in 2016, who was then at Cal Poly-San Luis Obispo. Reference 2 was the first publication of this concept known to the lead author of this report.

An earlier concept had been advanced to Robert Staehle at JPL, NASA Administrator Dan Goldin, and others in the 1990s for a not-so-smallsat, not using rideshare, for a mission to Pluto, advocated by Caltech Prof. Bruce Murray and Tomas Svitek. That concept may have envisaged use of radioisotope heater units.

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