Extra-Zodiacal-Cloud Astronomy via Solar Electric Propulsion

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Solar electric propulsion (SEP) is often considered as primary propulsion for robotic planetary missions, providing the opportunity to deliver more payload mass to difficult, high-delta-velocity destinations. However, SEP application to astrophysics has not been well studied. This research identifies and assesses a new application of SEP as primary propulsion for low-cost high-performance robotic astrophysics missions. The performance of an optical/infrared space observatory in Earth orbit or at the Sun-Earth L2 point (SEL2) is limited by background emission from the Zodiacal dust cloud that has a disk morphology along the ecliptic plane. By delivering an observatory to a inclined heliocentric orbit, most of this background emission can be avoided, resulting in a very substantial increase in science performance. This advantage – enabled by SEP – allows a small-aperture telescope to rival the performance of much larger telescopes located at SEL2. In this paper, we describe a novel mission architecture in which SEP technology is used to enable unprecedented telescope sensitivity performance per unit collecting area. This extra-zodiacal mission architecture will enable a new class of high-performance, short-development time, Explorer missions whose sensitivity and survey speed can rival flagship-class SEL2 facilities, thus providing new programmatic flexibility for NASA's astronomy mission portfolio. A mission concept study was conducted to evaluate this application of SEP. Trajectory analyses determined that a 700 kg-class science payload could be delivered in just over 2 years to a 2 AU mission orbit inclined 15° to the ecliptic using a 13 kW-class NASA's Evolutionary Xenon Thruster (NEXT) SEP system. A mission architecture trade resulted in a SEP stage architecture, in which the science spacecraft separates from the stage after delivery to the

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mission orbit. The SEP stage and science spacecraft concepts were defined in collaborative engineering environment studies. The SEP stage architecture approach offers benefits beyond a single astrophysics mission. A variety of low-cost astrophysics missions could employ a standard SEP stage to achieve substantial science benefit. This paper describes the results of this study in detail, including trajectory analysis, spacecraft concept definition, description of telescope/instrument benefits, and application of the resulting SEP stage to other missions. In addition, the benefits of cooperative development and use of the SEP stage, in conjunction with a SEP flight demonstration mission currently in definition at NASA, are considered.

Nomenclature

ΔV	=	velocity change
V_∞	=	hyperbolic excess velocity

I. Introduction

SOLAR electric propulsion (SEP) is often considered as primary propulsion for robotic planetary missions, providing the opportunity to deliver more payload mass to difficult, high-ΔV destinations. However, SEP application to astrophysics has not been well studied. This research identifies and assesses a new application of SEP as primary propulsion for low-cost high-performance robotic astrophysics missions. The mission concept, designated Extra-Zodiacal Explorer (EZE), features use of SEP to deliver an Explorer-class spacecraft to an orbit outside the densest regions of the solar system zodiacal cloud. The study was in part formulated around the possibility of using the planned NASA SEP Flight Demonstration vehicle, in conceptual definition within the NASA Office of Chief Technologist (OCT), to serve as the SEP system. The implications of this approach are described throughout this paper.

II. Science Motivation

The goal of this study is to enable the NASA Astrophysics Explorer Program to access orbits that extend outside the zodiacal dust cloud in order to realize a dramatic increase in cosmic discovery potential and achieve major science goals with no increase in telescope aperture over that which is normally associated with this small payload program. This challenge is motivated by long-term downward budget pressure on space astronomy that precludes development of new large aperture flagship-class payloads for the foreseeable future. We find that for an Explorer Class-EX (700 kg) observatory, this goal can be realized today using solar electric propulsion technology that has reached sufficient maturity (Technology Readiness Level-6) for a Phase-A new start and flight during this decade.

The Earth is imbedded in a cloud of dust grains that are produced by comet outgassing and impact fragmentation of asteroids that surround the inner planets. The general morphology of the zodiacal cloud is illustrated in Figure 1. This zodiacal cloud imposes a dramatic limitation on the sensitivity of all space observatories that operate over the optical to far-infrared spectrum in near-Earth zero inclination orbits.

These zodiacal dust grains impact space astronomy observations by producing a background light through which all space observatories have observed. This zodiacal background adds photon noise to the detector signal from astronomical sources. This added noise entirely limits the sensitivity of observatories that are sited at near-Earth in-plane orbits such as the Sun-Earth L2 libration point (SEL2). This situation is analogous to ground-

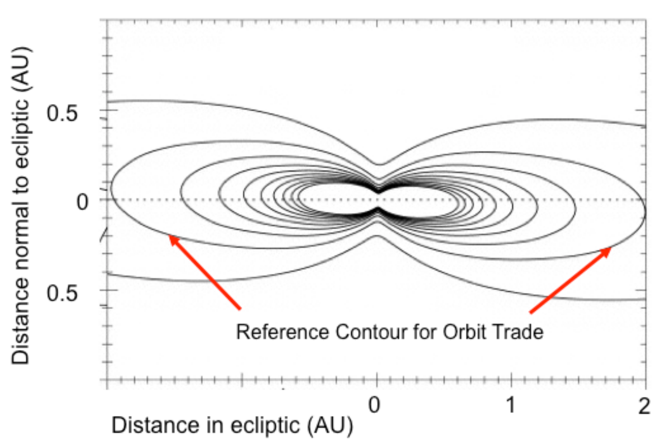

Figure 1. Isodensity contours of the Zodiacal cloud in a plane perpendicular to the ecliptic plane.[1]

based astronomers observing during daylight hours. In a very real sense, it has never been nighttime for optical and infrared space astronomers.

In this study, a mission architecture was developed to deliver an Explorer Class-EX astronomical observatory payload to an orbit outside the high-density region of the zodiacal cloud whose boundary is denoted by the isodensity reference shown in Figure 1.

III. Study Approach and Methodology

Initial analyses demonstrated that the desired spacecraft mass class and orbit could not be achieved, within the cost constraints of an Explorer mission, using chemical propulsion due to the very large mission ΔV's. Attention therefore focused on the use of electric propulsion. The consequent study had the following primary components: low thrust trajectory analyses, spacecraft concept definition, and characterization of science benefits.

A heliocentric target orbit, with semi-major axis of approximately 1.0 astronomical unit (AU) and inclination to the ecliptic plane of 30°, was established for initial low thrust trajectory analyses. Early in the study, a relatively large delivered observatory mass of 1500 kg was assumed. Trajectory searches were performed using available tools. Initial solutions resulted in use of the Atlas V 551 launch vehicle, two Earth gravity assists, and a 30 kW-class ion propulsion system to achieve the objective orbit. This class of mission was consistent with early planning for the SEP Flight Demonstration mission, but lower-cost solutions were sought. Subsequent analysis iterations generated a range of solutions. The team focused on the solution space that was achievable and consistent with Explorer CLASS EX mission payload mass and launch vehicle constraints. A reference case with a 700 kg observatory using an Atlas V 421 launch vehicle, two Earth gravity assists, and a 15 kW-class ion propulsion system provided a baseline with which to advance to the next stage of the study, spacecraft concept definition. Mission analyses further evolved during the concept definition phase as described below. A summary of the final mission analysis results is provided in Section 4.

Spacecraft concept definition was performed through use of the NASA Glenn Research Center (GRC) COllaborative Modeling for Parametric Assessment of Space Systems (COMPASS) team. The COMPASS team is a multidisciplinary concurrent engineering team whose primary purpose is to perform integrated vehicle systems analysis and provide conceptual designs and trades for both Exploration and Space Science Missions. COMPASS study pre-work consisted of mission analysis described above, validation of the 700 kg observatory mass, selection of the spacecraft/SEP architecture, and establishment of concept objectives. The 700 kg observatory mass target was validated by surveying a range of low-cost NASA observatory spacecraft and comparing their sizing and capabilities. The GRC and Goddard Space Flight Center (GSFC) study members reviewed this information and approved the 700 kg assumption as an input to the COMPASS study. The selected architecture consists of an observatory spacecraft with a separable SEP stage that is controlled by the spacecraft. The separated SEP vehicle was preferred for two primary reasons: 1) it is aligned with the general intentions of the SEP Flight Demo project, and 2) to minimize mass and momentum disturbances on the observatory for the science phase of the mission. Control of the SEP stage by the spacecraft reduces the SEP stage development and recurring cost. The primary objectives that influenced the COMPASS study included:

- Ensure that the observatory spacecraft would fit within the Explorer-EX class, including accommodating the impact of flying on the SEP stage assumed to be provided by the OCT project.
- Avoid low-TRL technologies in both the SEP stage and observatory; resulting in selection of the NASA's Evolutionary Xenon Thruster (NEXT) ion propulsion system and Orion-based UltraFlex solar arrays.
- Provide a capability that could be utilized by a variety of observatory missions, and thus is compatible with the competed Explorer-EX program approach.

The COMPASS study generated mission concepts for two destination orbits and performed a top-level comparison study with the representative 700 kg observatory in a more traditional near-Earth orbit. As the study evolved, the second destination orbit with semi-major axis of approximately 2.0 AU and 15° inclination to the ecliptic plane was developed in detail and provided a more favorable result. In addition, collaborative, iterative analyses allowed the team to baseline the Falcon 9 launch vehicle and still meet the mission trajectory objectives. This was a key accomplishment in fitting the resulting mission into an Explorer-EX mission class. The results of the COMPASS concept definition study are described in Section 5 of this paper.

A key element of the overall effort was to characterize the benefits provided by performing science in the resulting mission orbits; however, detailed description of the methods to quantify these benefits are outside the scope of this paper and will be reported in other technical forums. The general results of this assessment are presented in Section VI.

IV. Mission Analysis

The EZE Mission study developed two possible science orbits; a highly inclined (~30^o) heliocentric orbit with a semi-major axis of 1 AU, and a less inclined heliocentric orbit (~15^o) with a semi-major axis of 2 AU. The major figures of merit for mission performance were the time required to reach the science orbit, the flux of the Zodiacal dust as the spacecraft operates in its science orbit, and the communications distance from the spacecraft to Earth during science orbit operations. The 1 AU/30^o case is designed to keep the communication distance to the spacecraft at a minimum, while the 2 AU/15^o case reduces the interference of the Zodiacal dust cloud without requiring such a large plane change. Following the analysis of the two options, the 2 AU/15^o case is baselined, and further reported here, due to its superior science performance.

Mission analysis is conducted using MALTO 5.2.6[2]. The launch opportunity is mid-2020 based on the expected release year of an Explorer-EX announcement of opportunity, and the exact optimal launch date will be shown in the results below. This launch window provides the opportunity for a Mars flyby, which provides a significant benefit to the mission. Based on the dry mass of the spacecraft, MALTO is operated to minimize the total mission flight time while delivering no less than 1439 kg of mass, including the observatory spacecraft and dry SEP stage, to the target orbit.

The launch vehicle is modeled as a Falcon 9 Block 2, with a vehicle adapter mass of 40 kg and a mass contingency of 10%. To ensure that the launch vehicle performance curve is not applied to high declination departures, the declination of the Earth departure asymptote is limited to +/-28.5^o. Before accounting for contingency and adapter mass, the launch vehicle performance is modeled using the following equation:

$$2490.88 - 72.1167 \cdot V_\infty + 0.614262 \cdot V_\infty^{\ 2} - 0.00280389 \cdot V_\infty^{\ 3}$$

The nominal power generation of the solar arrays at 1AU is 13 kW. Of that, 500 W is dedicated to the non-propulsion aspects of spacecraft operation during thrusting periods. A propulsion duty cycle factor of 90% is used to model periodic coast periods for communications, navigation, and other functions best performed when the engines are not firing. The solar array model used is the “Lockheed UltraFlex” model as given in MALTO GUI v2.5.7.

The propulsion system utilizes two NEXT thrusters assuming a “P10 High Thrust” throttle table[3]. The thruster switching strategy is set to run as many thrusters as possible given the current power. In general, ion thrusters tend to run at higher specific impulses at higher input power, so dividing the power between two thrusters will reduce the specific impulse but increase the thrust, reducing the trip time.

The science mission orbit for EZE only has a few loose constraints on its parameters. For this analysis, we assumed a circular orbit of approximately 2 AU in semi-major axis with zero eccentricity. The 15^o inclination is chosen based on what seemed to be an achievable goal in approximately 2 years of flight time. With zero eccentricity, the argument of perihelion of the orbit is effectively irrelevant, so a value of 0^o is used. Two elements of the orbit are thus undefined; the right ascension of the ascending node and the mean anomaly.

The elements of the optimal target orbit are given in Table 1. Table 2 summarizes some of the key features of the resulting trajectory. Not only is the vehicle capable of getting to the science orbit in just over two years, but the propellant throughput is less than the capacity of a single NEXT thruster[4].

Table 1. EZE Science Orbit Elements

Semi-major axis (km)	300 000 000
Eccentricity	0
Inclination (deg)	15
Right ascension of the ascending node (deg)	73.0813
Argument of periapsis (deg)	0
Mean anomaly (deg)	-66.0805

Table 2. Trajectory summary

Earth departure date	June 20, 2020
Earth departure mass (kg)	1823
Earth departure hyperbolic velocity (km/s)	2.48
Mars flyby altitude (km)	1000
Mars flyby hyperbolic velocity (km)	3.51
Science orbit arrival date	June 22, 2022
Science orbit arrival mass (kg)	1439 kg
Total Time of Flight (days)	731
Total xenon propellant expenditure (kg)	384

Figure 2 shows the trajectory inclination history as a function of time. The Falcon 9 at launch provides about three degrees of the inclination change, and roughly four degrees comes from the Mars flyby maneuver, which is the large vertical step in inclination at about 200 days into the flight. The electric propulsion system performs the remainder of the maneuver. The vertical dashed line marks the time of arrival at the science orbit.

With a Mars flyby involved in the trajectory, the geometry of the science orbit is date-dependent. This is complicated by the fact that MALTO allows neither ephemeris-free arrival at an inclined target body, nor for the orbital elements of the target body to be independent variables in the optimization. To overcome this restriction, a Python script is used to wrap MALTO within a Nelder-Mead simplex optimization algorithm via the SciPy *fmin* function[5]. The independent variables for the simplex algorithm were the right ascension of the ascending node and the mean anomaly, whose initial values were taken for a convergent but suboptimal MALTO case. While a simplex optimization algorithm is generally slower to converge than gradient-based methods, gradient methods generally attempt to take steps that were beyond the radius of convergence of MALTO. The algorithm finds the optimal alignment of the target orbit, as well as the correct mean anomaly for insertion into that orbit, which would provide a minimal trip time.

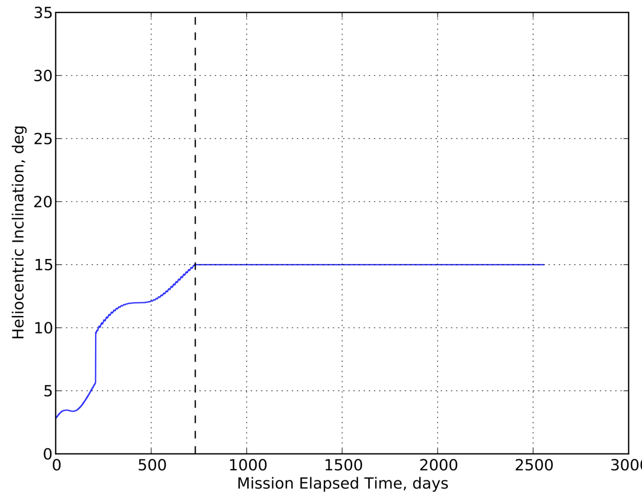

Figure 2. Inclination history for the 2 AU/15° mission option.

Figures 3 and 4 show the trajectory from a top-down from ecliptic-north view and side-on to the ecliptic view, respectively. The thrust vectors are indicative of the thrust acceleration vector at that point in the trajectory. The faint dotted gray lines indicate the orbits of Earth and Mars as well as the science orbit.

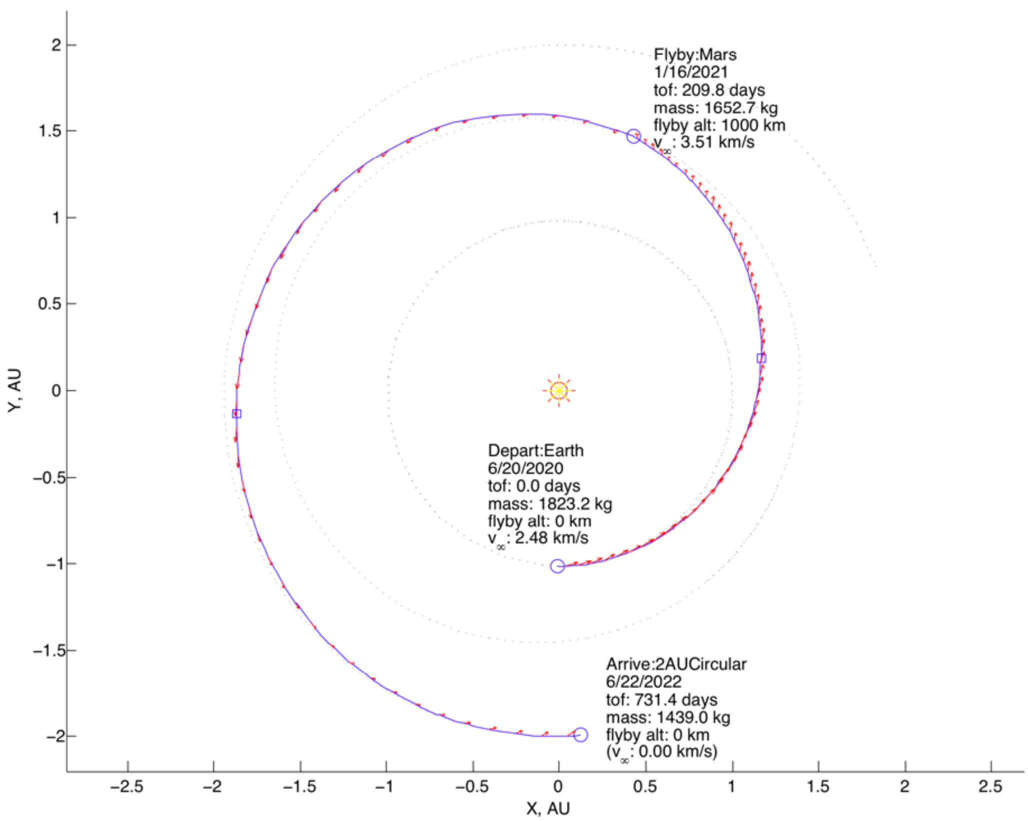

Figure 3. EZE 2 AU/15° trajectory viewed from the ecliptic north pole.

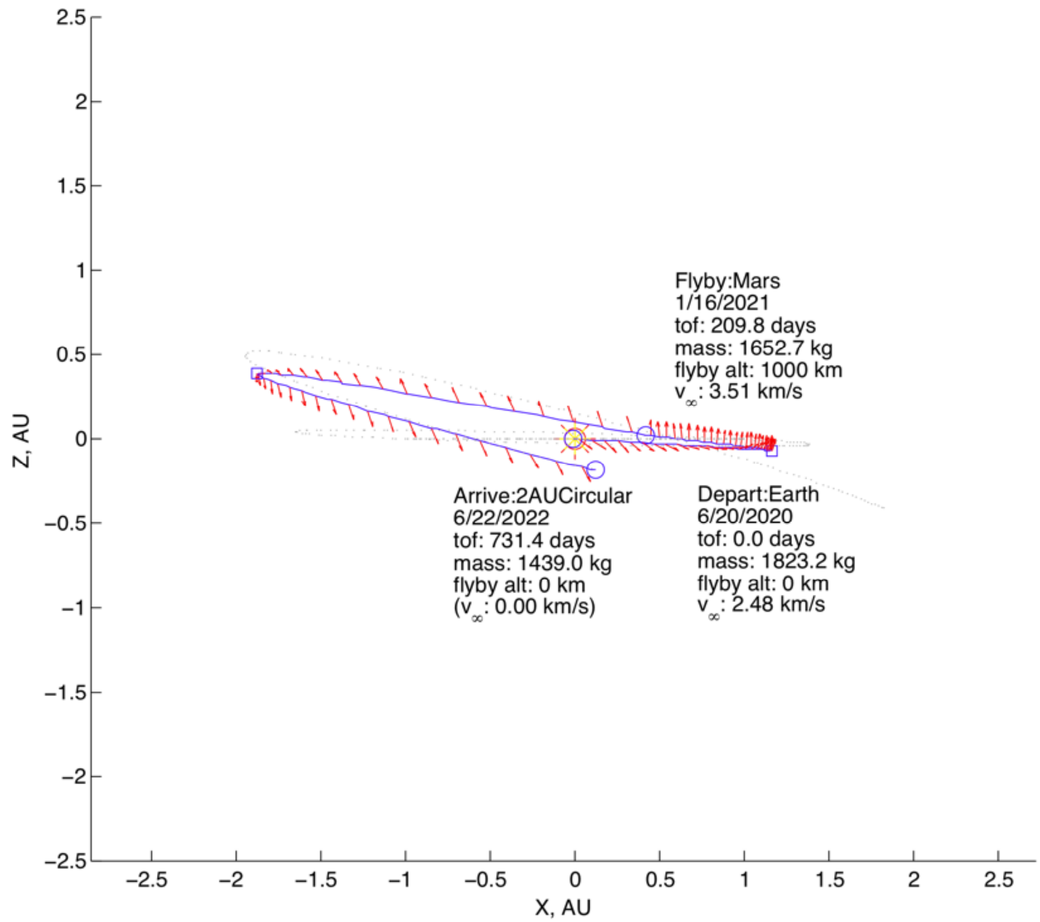

Figure 4. EZE 2 AU/15° trajectory viewed edge-on to the ecliptic

V. Spacecraft Concept

The COMPASS spacecraft/mission concept definition study was structured by the spacecraft architecture defined in pre-work. The expendable SEP stage was fully defined and sized by the COMPASS team. The observatory was separated into the two primary elements, the instrument and the spacecraft bus. Since the spacecraft bus is required to control the SEP stage and otherwise be compatible with the SEP stage, the COMPASS team also defined and sized the bus. The instrument was assumed as a 300 kg "black box", which was representative of a variety of ≤ 1 m diameter-class instruments surveyed during the pre-work described above. Both spacecraft were considered Class C, consistent with Explorer class missions, and thus single string except where additional systems could be added and low mass and low cost. Mission operation time was assumed to be 5 years. Spacecraft concepts were defined for both mission orbits analyzed. The notable features of the SEP stage and spacecraft bus are shown in Figure 5 and described further below.

A. Solar Electric Propulsion Stage

The primary subsystems on the SEP stage are the ion propulsion system and electrical power system. A NEXT-based ion propulsion system was selected to achieve the mission trajectory with a suitable technology readiness level for a mission in the defined timeframe. The system has a 2+0 NEXT configuration – two operating thruster strings and no additional spare strings. This was deemed acceptable given the Explorer mission class and the nature of the potential SEP Flight Demonstration. Each thruster string consists of the NEXT thruster, a gimbal, xenon flow control assembly, and power processing unit (PPU). If a thruster string is lost it was determined that the mission

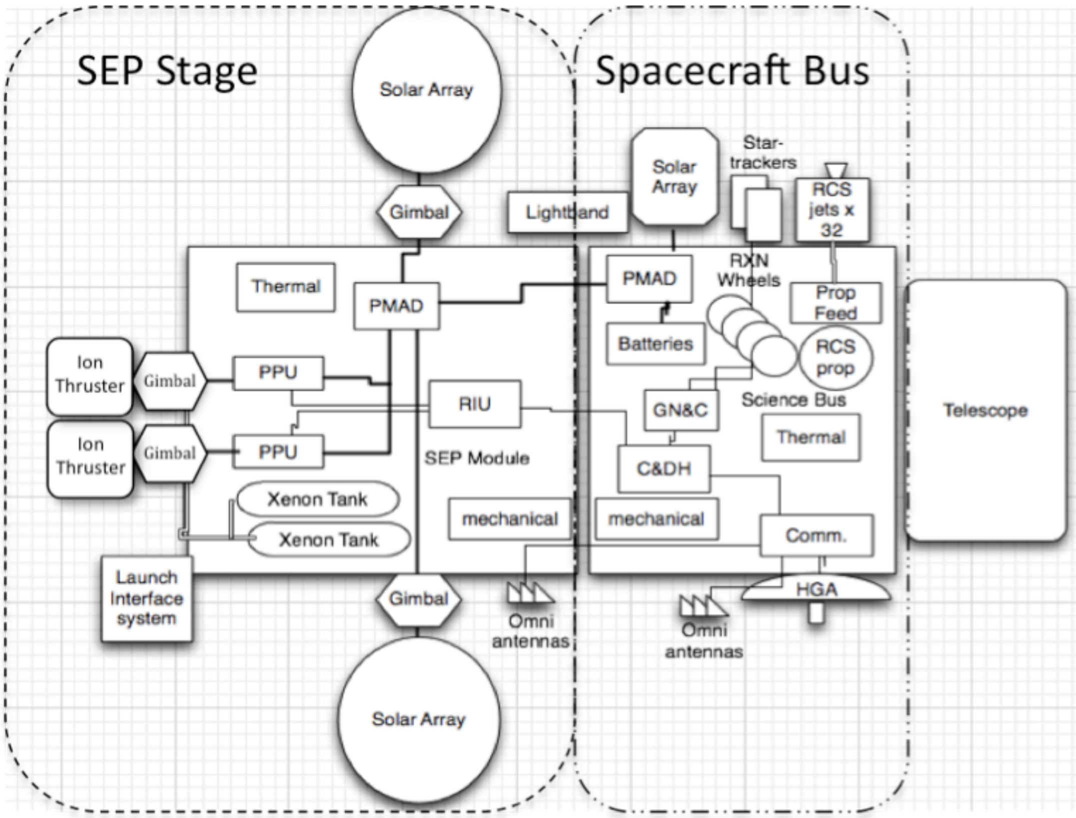

Figure 5. The primary components of the SEP stage and Spacecraft bus, with the telescope instrument shown as a "black box" payload.

could still be achieved with a longer duration transfer to the mission orbit and/or partial sacrifice of heliocentric orbit inclination.

The 13 kW (end-of-life, 1 AU) power for the SEP system (as well as the spacecraft bus during cruise) is provided by provided by two Orion service module development-based 6m diameter UltraFlex arrays. These arrays provided a high specific power approach with reasonable technology readiness. The power level associated with these arrays was accepted as an input to the concept definition to gain the TRL advantage. Array power and technology is a subject for further study, as these have primary impact on the SEP delivery phase of the mission. The power management and distribution (PMAD) electronics distribute high voltage unregulated power to the ion propulsion system and low voltage regulated power to other SEP stage subsystems and the spacecraft bus.

Other components of the SEP stage include:

- Two off-the-shelf composite-overwrap pressure vessel xenon storage tanks to store the approximately 400 kg of xenon,
- A remote interface unit (RIU) to communicate with the spacecraft bus and to control the SEP stage propulsion, power and thermal subsystems in response to high-level spacecraft commands,
- Radiators for thermal management of propulsion and power subsystem waste heat,
- Low data rate omni antennae connected directly to the spacecraft bus communications subsystem to allow a communications link to Earth in all spacecraft attitudes.

The SEP stage is packaged into a 1.6 m diameter thrust tube structure to provide a simple, standard interface to the launch vehicle. This large thrust tube design has been used in many past SEP stage designs and provides ample internal space for propellant, feed systems, PMAD, and power conversion systems.

The SEP stage defined here, with appropriately sized xenon tanks, is compatible with both destination orbits evaluated in this study. It is clear from the design that additional tanks, an additional thruster string, and different solar arrays could be added to adapt the stage to other missions.

B. Spacecraft Bus

The spacecraft bus is standard, with the following features associated with the mission orbit, SEP stage interfaces and SEP operations:

- Since the SEP stage is separated prior to the science operations mission phase, the spacecraft bus has its own solar arrays and electric power system,
- Spacecraft battery sized to support SEP stage and spacecraft bus loads during the launch phase,
- Control and data interface to the SEP stage through the RIU,
- Upsized reaction wheels to accommodate the SEP stage mass properties,
- Control of the ion thruster gimbals to control spacecraft attitude and manage momentum during the SEP mission phase,
- Cold gas propulsion system for momentum management after SEP stage separation,
- Communications system sized to support science downlink from this unique (for astrophysics missions) heliocentric orbit: 0.74 Mbps (compressed, encoded), 40 Gbits per day uncompressed at 3 AU (380 Gb/day at 1 AU).

The impact of these features was found not to have a significant impact on the spacecraft bus. There were some differences in spacecraft bus concept for the two destination orbit missions. The primary differences were in the communications and power subsystems. The 1 AU/30° orbit has a maximum range to Earth of approximately 0.6 AU, allowing a smaller, lower power RF communication system, and a near-circular orbit at 1 AU allowing for a smaller solar array. In other regards, the spacecraft buses for the two orbits were very similar.

C. System Overview

The resulting integrated spacecraft for the preferred 2 AU/15° orbit case, in its deployed SEP cruise and stowed launch configurations, is shown in Figure 6. As shown in the figure, the stowed solar arrays are packaging such that they do not protrude above the spacecraft/SEP stage interface plane, providing a open volume for the EZE, or other, spacecraft. Figure 7 illustrates a graphic depiction of the SEP stage and spacecraft bus. The top-level mass breakdown is shown in Table 3, broken down by major subsystems for each vehicle. Per COMPASS protocol, after growth contingency masses were applied, additional growth mass was added at the system level to ensure 30% overall mass growth allowance. The resulting launch vehicle mass margin of 13% exceeds the 10% margin value that is sought at this stage of analysis and concept definition.

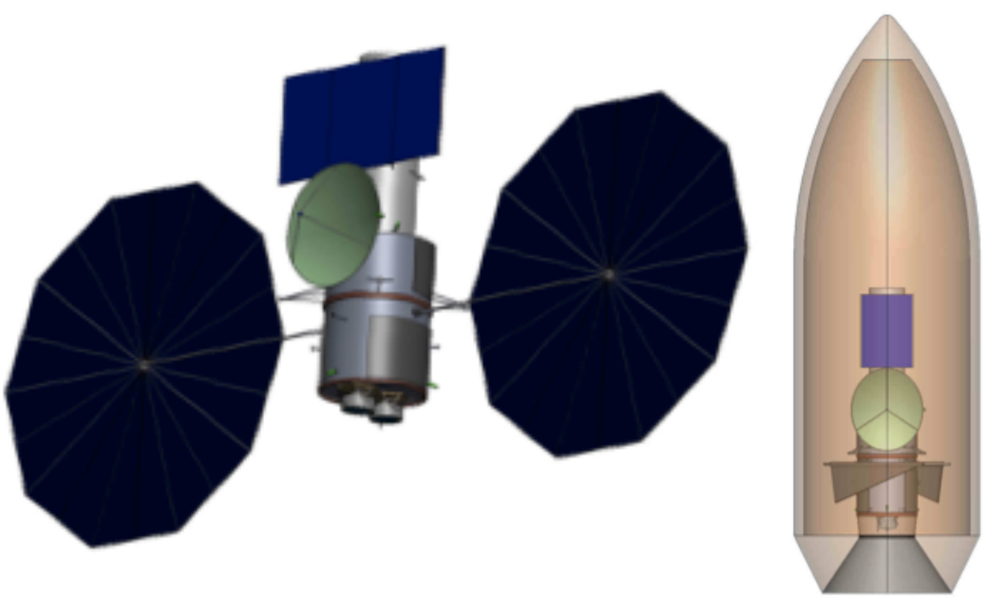

Figure 6. Integrated vehicle, for 2 AU/15° mission orbit, in SEP cruise configuration (left) and stowed in the Falcon 9 payload fairing (right).

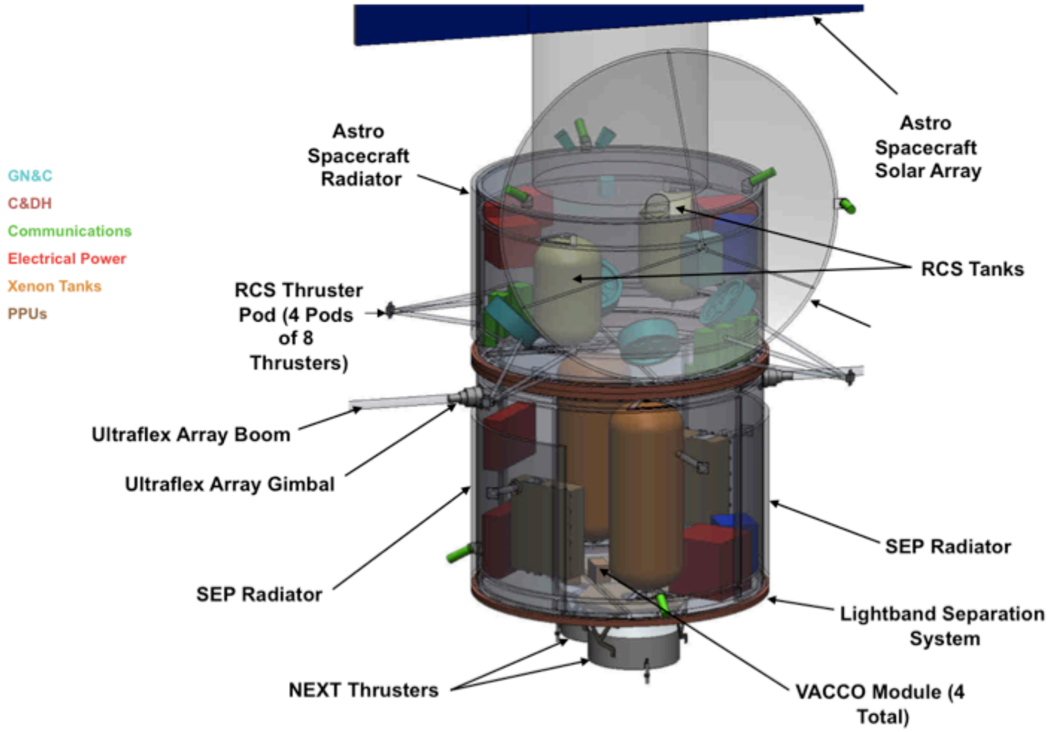

Figure 7. Graphic depiction of SEP stage (lower vehicle) and spacecraft bus (upper vehicle). Instrument package and observatory solar array not fully illustrated.

Table 3. EZE spacecraft mass summary

Spacecraft Master Equipment List Rack-up (Mass)				COMPASS S/C Design	
WBS	Main Subsystems	Basic Mass (kg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
06	EZE Astronomy Spacecraft	1527	207	1734	
06.1	Solar Electric Stage (SEP Stage)	956	82	1038	
06.1.1	Science Payload	0	0	0	TBD
06.1.2	Attitude Determination and Control	1	0	1	3%
06.1.3	Command & Data Handling	35	10	44	29%
06.1.4	Communications & Tracking	4	1	5	30%
06.1.5	Electrical Power Subsystem	182	33	215	18%
06.1.6	Thermal Control (Non-Propellant)	58	9	67	15%
06.1.7	Propulsion and Propellant Management	168	13	181	8%
06.1.8	Propellant	417		417	
06.1.9	Structures and Mechanisms	91	16	108	18%
06.2	Astro Spacecraft Bus	571	125	696	
06.2.1	Science Payload	231	69	300	30%
06.2.2	Attitude Determination and Control	36	5	41	15%
06.2.3	Command & Data Handling	25	7	31	27%
06.2.4	Communications & Tracking	24	2	26	7%
06.2.5	Electrical Power Subsystem	88	20	107	23%
06.2.6	Thermal Control (Non-Propellant)	47	7	54	15%
06.2.7	Propulsion and Propellant Management	36	2	38	7%
06.2.8	Propellant	16		16	0%
06.2.9	Structures and Mechanisms	70	13	83	18%
	Estimated Spacecraft Dry Mass	1094	207	1301	19%
	Estimated Spacecraft Wet Mass	1527	207	1734	
System LeveL Growth Calculations					Total Growth
	Dry Mass Desired System Level Growth	1094	328	1422	30%
	Additional Growth (carried at system level)		121		11%
	Total Wet Mass with Growth	1527	328	1855	

The mission concept developed in this study can serve a wide range of extra-zodiacal orbit astrophysics missions. The two orbits studied are two discrete points in a continuum of orbits that would provide science benefits. The separated SEP stage architecture and stage design is inherently compatible with a broad range of observatory spacecraft. The study team elected to implement a “dumb” stage, controlled by the spacecraft; but, the mission could be similarly implemented with a “smart” stage that provides it’s own navigation, control and communications capabilities. The architecture is robust, and the SEP stage readily adaptable to a variety of science interests.

VI. Science Return

Delivery of an observatory to an orbit outside of the densest regions of the zodiacal dust cloud provides tremendous benefits in capability. The zodiacal background power for the two destination orbits studied, as a function of wavelength, is shown in Figure 8. This assumes the observatory is at the maximum distance normal to the ecliptic plane. Figure 9 illustrates the regions of the zodiacal cloud that the observatory will operate in for each of the two orbit cases. For the 1 AU/30° case, the observatory travels through higher density regions as it passes through the ascending and descending nodes of the orbit, thus peak performance is only available for portions of the science orbit. For the 2 AU/15° case, the observatory is outside of the desired isodensity contour for it’s entire science mission. Between the lower zodiacal background power and orbit conditions depicted in Figures 8 and 9, it is clear that the 2 AU/15° provides the higher science return.

With the zodiacal background power calculated, the resulting observatory performance can be determined. This is depicted in Figure 10, as a function of wavelength, and reported relative to an observatory in a Sun-Earth L2 libration point orbit. This performance increase, which can exceed two orders of magnitude in observing speed and one order of magnitude in sensitivity, can enable an Explorer-class payload to perform at the level of a much larger observatory through orbit choice alone with no increase in telescope aperture.

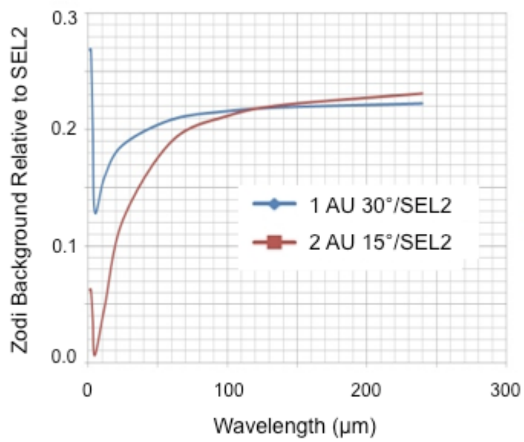

Figure 8. Zodiacal background power for the two destination orbits, as a function of observing wavelength.

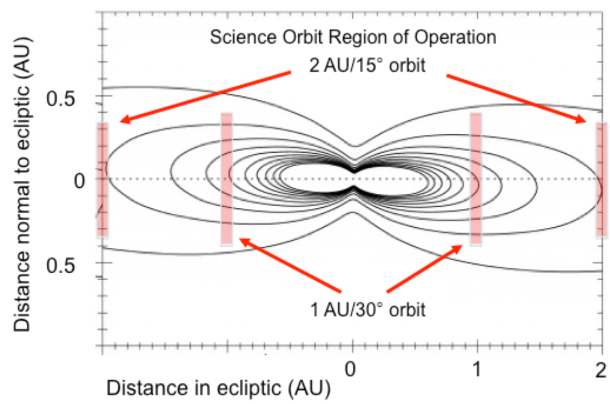

Figure 9. Regions of science operation for the two destination orbits, relative to zodiacal cloud density profile.

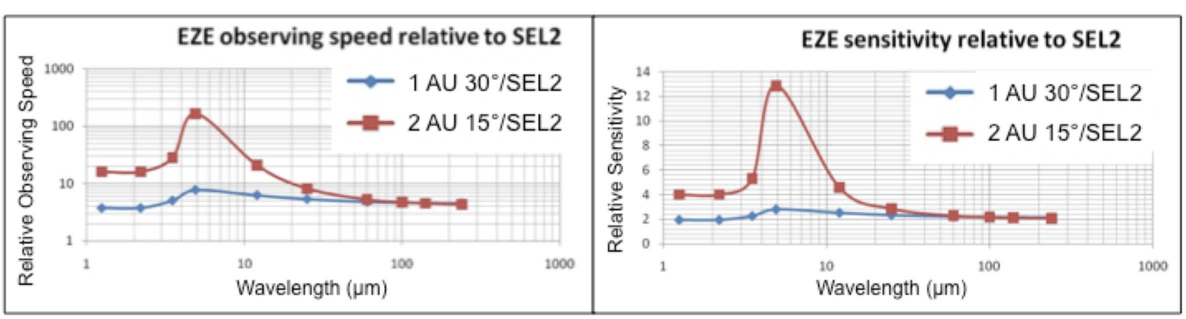

Figure 10. Observatory performance for the two orbits, relative to an observatory in SEL2 orbit.

VII. Other SEP Stage Applications

A. Other NASA Science Missions

The SEP stage defined in this study provides a baseline configuration that can be applied to other NASA science missions with modifications to the vehicle or mission architecture. The use of a separable SEP stage has been widely considered for outer planet missions, in which the SEP stage is used in the inner solar system to inject the spacecraft on the transfer orbit to the outer planet. The Titan/Saturn System Mission baselined a SEP stage with a 2+1 NEXT propulsion system and UltraFlex solar arrays providing 15 kW at 1 AU[6]. The Enceladus flagship mission baselined a 3+1 NEXT system with solar arrays providing 25 kW at 1 AU[7]. A prior study demonstrated a parametric range of Saturn system capability by varying numbers of thrusters and available power[8]. The Uranus Orbiter and Probe[9], Neptune-Triton-KBO[10] and Saturn Ring Observer[11] decadal mission studies all considered use of NEXT-based SEP stages of varying capabilities. In each of these mission concepts, the EZE SEP stage provides a basic configuration that can be tailored to the specific performance required.

Other NASA missions employ SEP as an integrated element of the spacecraft bus. In some of these missions, availability of a SEP stage similar to that defined in this study may result in a mission architecture change and use of such a stage.

B. SEP Flight Demonstration Mission

The NASA Office of Chief Technologist (OCT) is currently defining plans to conduct a technology demonstration project for SEP vehicles. The goal is for a flight demonstration of an integrated SEP stage during this decade. The EZE team has identified an opportunity for cooperation between the OCT and Science Mission Directorate (SMD), in which an OCT-developed SEP stage is used to deliver an Explorer Class-EX observatory to an extra-zodiacal mission orbit. In this scenario, no additional budget lines beyond those in existence are necessary, nor is any exchange of funding between OCT and SMD. The OCT project develops and delivers the SEP stage to the Explorer-EX mission, and performs SEP-related launch site and mission operations. The SMD Explorer project provides the Falcon 9 launch vehicle that would normally be used for this class of Explorer, the observatory spacecraft development, and conducts spacecraft mission and science operations. The SMD benefits from the OCT partnership in accomplishing a mission orbit that provides a major improvement in capability over the usual Explorer-class missions. The OCT benefits from the SMD partnership in receiving launch services, with the result that more resources can be invested in the SEP vehicle technologies and capabilities, and the value associated with delivering a groundbreaking science mission.

OCT development of a SEP stage that is relevant to the proposed EZE mission concept could provide long-term value to the NASA astrophysics community. Were the Explorer program to apply additional resources to subsequent Explorer Class-EX missions to fund the relatively low recurring costs of a previously developed SEP stage, this capability would routinely be available to Principal Investigators.

VIII. Conclusions

The potential use of Solar Electric Propulsion enables the delivery of a small astrophysics observatory to a science orbit beyond the densest regions of the solar system zodiacal dust cloud, providing a step change in science capability by dramatically improving the observing performance of a small-aperture telescope. This is similar in concept to the launch vehicle step-down improvements associated with planetary science missions, in which compelling science normally associated with Flagship missions can be accomplished with small spacecraft and launch vehicles through the use of SEP. The EZE study team has demonstrated low thrust trajectories through which Explorer EX-class missions can be delivered to desirable extra-zodiacal orbits by a modest SEP stage equipped with existing or near-term technologies. Two specific mission orbits have been studied, but a continuum of orbits exist that can be evaluated on a case-by-case basis. The SEP stage and spacecraft bus concepts are viable and consistent with SEP vehicle concepts previously proposed and Explorer class observatories flown to date. A unique opportunity has been identified by the EZE study team, for a partnership between the OCT SEP Flight Demonstration project and the SMD Astrophysics Explorer program. Cooperative application of resources could provide benefits to both parties, with the result of a Principal Investigator's Explorer observatory conducting groundbreaking science outside of our inner solar system's "daytime sky".

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