LEO to GEO (and Beyond) Transfers using High Power Solar Electric Propulsion (HP-SEP)

IEPC-2017-396

Presented at the 35th International Electric Propulsion Conference
Georgia Institute of Technology • Atlanta, Georgia • USA
October 8 – 12, 2017

Christopher S Loghry[1]
Moog Inc, Chatsworth, CA, 91311, USA

Steven Oleson[2], Jeffrey Woytach[3], Michael Martini[4], David Smith[5], James Fittje[6], John Gyekenyesi[7], Anthony Colozza[8], James Fincannon[9], Aimee Bogner[10], Robert Jones[11], Thomas Packard[12], Elizabeth Turnbull[13]
NASA Glenn Research Center, Cleveland, OH, 44135, USA

Abstract: Rideshare, or Multi-Payload launch configurations, are becoming more and more commonplace but access to space is only one part of the overall mission needs. The ability for payloads to achieve their target orbits or destinations can still be difficult and potentially not feasible with on-board propulsion limitations. The High Power Solar Electric Propulsion (HP-SEP) Orbital Maneuvering Vehicle (OMV) provides transfer capabilities for both large and small payload in excess of what is possible with chemical propulsion. Leveraging existing secondary payload adapter technology like the ESPA provides a platform to support Multi-Payload launch and missions. When coupled with HP-SEP, meaning greater than 30 kW system power, very large delta-V maneuvers can be accomplished. The HP-SEP OMV concept is designed to perform a Low Earth Orbit to Geosynchronous Orbit (LEO-GEO) transfer of up to six payloads each with 300 kg mass. The OMV has enough capability to perform this 6 km/s maneuver and have residual capacity to extend an additional transfer from GEO to Lunar orbit. This high delta-V capability is achieved using state of the art 12.5 kW Hall Effect Thrusters (HET) coupled with high power "roll up" solar arrays. The HP-SEP OMV also provides a demonstration platform for other SEP technologies such as advanced Power Processing Units (PPU), Xenon Feed Systems (XFS), and other HET technologies. The HP-SEP OMV platform can be leveraged for other missions as well such as interplanetary science missions and applications for resilient space architectures.

This document consists of general capabilities information that is not defined as controlled technical data under ITAR Part 120.10 or EAR Part 772.

[1] Business Development Manager, Moog Space Access & Integrated Systems, cloghry@moog.com
[2] COMPASS Lead Conceptual Spacecraft Design Team, steven.r.oleson@grc.nasa.gov
[3] Systems Engineer, Science and Space Technologies Systems Engineering Branch, jeffrey.m.woytach@grc.nasa.gov
[4] Systems Engineer, NASA Glenn Research Center, michael.c.martini@nasa.gov
[5] Systems Engineer, Vantage Partners LLC, david.a.smith-1@nasa.gov
[6] Propulsion Engineer, Vantage Partners LLC, james.e.fittje@nasa.gov
[7] Mechanical Engineer, ZIN Technologies Inc, john.z.gyekenyesi@nasa.gov
[8] Thermal Engineer, Vantage Partners LLC, anthony.j.colozza@nasa.gov
[9] Power Engineer, NASA Glenn Research Center, james.fincannon@nasa.gov
[10] Avionics Engineer, NASA Glenn Research Center, aimee.l.bogner@nasa.gov
[11] Communication Engineer, NASA Glenn Research Center, rejones@nasa.gov
[12] Configuration Engineer, Vantage Partners LLC, thomas.w.packard@nasa.gov
[13] Cost Engineer, NASA Glenn Research Center, elizabeth.r.turnbull@nasa.gov

Nomenclature

BEO	=	Beyond Earth Orbit
BOL	=	Beginning of Life
bps	=	Data rate, bits per second
dB	=	decibel
dBi	=	Gain of an Isotropic radiator
delta-V	=	Change in velocity between two orbits, km/s
EP	=	Electric Propulsion (specifically plasma thrusters)
GEO	=	Geosynchronous Orbit
GRC	=	Glenn Research Center
GTO	=	Geosynchronous Transfer Orbit
HET	=	Hall Effect Thrust
HP	=	High Power (specifically 30 kW to 50 kW)
Isp	=	Specific Impulse, s
K	=	Temperature, Kelvin
LEO	=	Low Earth Orbit
LRO	=	Lunar Near-Rectilinear Orbit
MEO	=	Medium Earth Orbit
OMV	=	Orbital Maneuvering Vehicle (generic term for a transfer vehicle)
P	=	Power, kW
SEP	=	Solar Electric Propulsion (specifically plasma thrusters using power from solar arrays)
T/P	=	Thrust to Power ratio, mN/kW
T	=	Thrust, N
VDC	=	Volts Direct Current

I. Introduction

Rideshare missions, also known as Multi-Manifest Missions, provide a reduced cost of space access where a secondary payload can utilize excess capacity on a launch vehicle as opposed to procuring a dedicated launch. The concept has been used nearly as long as orbital launch vehicles have existed but not typically as the primary method of space access for programs. In the last decade, CubeSats have taken advantage of this concept and helped create a relatively new economy in Low Earth Orbit (LEO) based on small and rapidly developed platforms mostly focused on Earth Observation, technology development programs, or educational endeavors that have much lower barriers to entry than previous systems. LEO offers an orbital regime that is easiest to achieve allowing for relatively large upmass from many existing launch vehicles compared to other higher energy orbits such as Geosynchronous Transfer Orbit (GTO), direct inject to Geosynchronous Orbit (GEO), or Earth Escape velocities that often have upmass limitations. Regular cargo resupply missions to the International Space Station provide another method of consistent space access with regular supply vehicles. The low orbit is advantageous from an orbital debris mitigation perspective.

As secondary payloads become larger and desire orbits different than LEO (or multiple different ones within LEO), an Orbital Maneuvering Vehicle (OMV) becomes a cost-effective method of achieving these mission orbital requirements. The OMV is a propulsive secondary payload adapter that can simplify many of the mission needs such as power and propulsion that do not often scale to a small form factor to enable payloads and/or spacecraft to remain smaller, simpler and often lower cost. This OMV can be treated like an upper stage on the launch vehicle, an orbital transfer vehicle part of the mission, or in some cases both. Another advantage of separating these requirements from either the launch vehicle or the end spacecraft/payload is each element can focus on that element of the mission. The launch vehicle provider can focus on low cost access to a few identified orbits such as LEO and GTO, the spacecraft/payload can focus on the needs of the payloads rather than adding capability to achieve a given orbit due to limitations from launch, and the OMV can act as a bridge between these two system elements by focusing on the orbit adjustment or many adjustments.

This 3rd element changes the typical "one rocket, one satellite" paradigm that most large spacecraft programs rely on, but is a key change to reduce overall mission cost and complexity by segregating complexity within the overall system. For instance, a typical spacecraft for use in GEO will be launched into GTO and then use an onboard chemical propulsion system to circularize the orbit and reduce inclination. Once into GEO after approximately one week of maneuvers this element of the propulsion system is often not used again. This transfer requires a large amount of propellant that can be over 40% of the overall launch mass and drives to a larger spacecraft structure increasing the mass and cost in addition to other impacts such as processing facility, test chamber, and transportation system sizes. An intermediate OMV specifically designed for this type of transfer would allow for a simpler primary spacecraft and the OMV could be specialized specifically for this task. An OMV with more capability could extend this transfer from LEO to GEO which would reduce the cost and complexity of the launch vehicle as well. The propellant mass fraction for chemical propulsion systems for this type of transfer would be much too great to be efficient, but solar electric propulsion is an ideal candidate and has been used on other systems for Electric Orbit Raise (EOR). For large delta-V maneuvers the overall thrust level becomes important to minimize the transfer time, for both radiation and time to revenue considerations. This drives a system to a relatively high thrust and therefore high power solar electric propulsion (HP-SEP). For the purposes of this paper, high power is defined as greater than 30 kW but less than 50 kW.

This paper provides results of a mission study showing how a flexible OMV platform leveraging HP-SEP can be used for multiple missions, and customer types, each with different goals including cost targets and time to system readiness. Technology needs and state of the art assessments were part of this study as the desire was to field a demonstration mission in the 2020-2021 timeframe. The study begins with a technology demonstration platform that can be expanded to an operational mission element for commercial, civil (such as NASA), and military missions.

II. Moog Orbital Maneuvering Vehicle

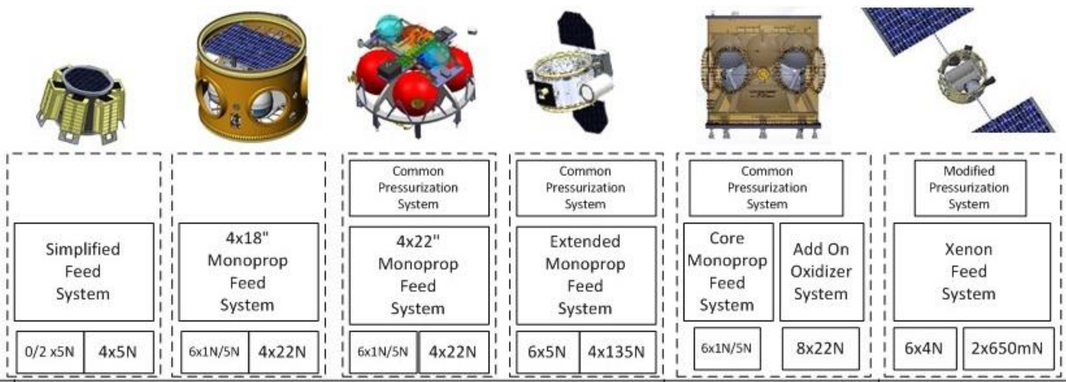
A. Moog Space Access and Integrated Systems Introduction

Moog Space Access and Integrated Systems (SAIS) provides a focal point to harness the breadth and depth of Moog's capability including mission architecture/design, launch strategy, and spacecraft systems engineering. Moog works with customers at the early stage of a program to identify and optimize technical, cost, risk and programmatic trades. Moog has developed spacecraft and smart upper stage concepts from <100 kg to 1,100 kg+ to support a wide range missions such as: constellations of over 1800 small satellites, deep space cubesat deployers, hosted payload platforms, interplanetary probes, NASA's Asteroid Return Mission, a commercial rideshare tug, and a commercial weather satellite constellation.

Moog SAIS draws from within the greater Moog organization for both engineering expertise and flight hardware. Capabilities include spacecraft avionics, science payloads, propulsion components and systems, launch adapters including launch site integration, mechanisms and actuators, Guidance Navigation and Control components and algorithms, and several aspects of launch vehicles including thrust vector control systems and launch adapters/payload accommodation. Moog's global footprint is in 25 countries and 10,500 people with its Space and Defense Group in several countries and 16 sites within the US.

B. Moog OMV Introduction

Moog has been developing its OMV family of capabilities for over 3 years (see Figure 1). The key to the family is a flexible and modular propulsion system configuration in addition to key subsystems, such as avionics, that remain somewhat constant between configurations. Structurally the system is based on the EELV Secondary Payload Adapter (ESPA) which is another key to a flexible configuration with built in rideshare capability. The family can utilize a green propellant such as LMP-103S, Hydrazine as a monopropellant, Hydrazine with Nitrogen Tetroxide (NTO) as a bipropellant, and finally Xenon in the Electric Propulsion variant to support delta-V maneuvers and be part of the Attitude Control System (ACS). The OMV family is designed to be launch vehicle and primary payload agnostic, within the appropriate class, and flexible with respect to the payload(s) it can accommodate. The OMV is particularly useful for future technology demonstration missions because of the reduced costs through rideshare and flexible interface of the ESPA. Many of the key components and subsystems can be sourced from within Moog including subsystems to support the HP-SEP subsystem.



Parameter	Monopropellant		Bipropellant	Electric
Propellant	LMP-103S	Hydrazine	Hydrazine/NTO dV + Hydrazine ACS	SEP: Xenon Cold Gas ACS

Figure 1. Moog OMV Family Overview

The core structure of the OMV is the ESPA ring which provides a flexible and adaptable structure for rideshare missions. This structure has been used to mount and deploy auxiliary payloads, as well as provide the backbone structure for extended space missions. The ESPA Ring was designed to use excess launch capacity on EELV medium-class launch vehicles. The ring is a multi-payload adapter for large primary spacecraft (up to 20,000 lbm (9072 kg)) and six auxiliary payloads on 15” diameter ports (payloads up to 400 lbm (181 kg)) or up to five auxiliary payloads on 24” diameter ports (payloads up to 700 lbm (320 kg)). The ESPA mounts directly to the launch vehicle upper stage, below the primary spacecraft. Stacked ESPA configurations are also possible and have been flight proven.

The maiden flight of the ESPA ring was in March 2007 for the STP-1 mission. Further ESPA options have been developed to offer varying port configurations, ring heights, and increased auxiliary spacecraft carrying capability. The first NASA mission to utilize an ESPA, Lunar Reconnaissance Orbiter (LRO)/Lunar Crater Observation and Sensing Satellite (LCROSS), launched in June 2009. The option to use standard or custom ESPA ports, external brackets that are configurable for a specific mission design, and/or internal mounting features makes the ESPA an ideal baseline for the OMV designs described here. A number of missions have already flown using the ESPA for both long and short durations. Examples include the Lunar Crater Observation and Sensing Satellite (LCROSS), AFRL’s Demonstration and Science Experiments (DSX), the USAF’s ESPA Augmented Geostationary Laboratory Experiment (EAGLE), and Spaceflight’s SHERPA. This heritage lays the groundwork for utilizing the ESPA to meet the requirements of a HP-SEP OMV mission.

An initial HP-SEP OMV concept was studied from a mission perspective to see if further development was warranted[1]. Three major mission types were assessed included a GTO rideshare to a variety of destinations beyond Earth orbit (BEO) and a survey of potential mission applications, deployment in LEO as a platform for regular Space Asset Management (SAM) including acting as a platform for robotic servicing or disposal of orbital debris, and a spiral out from LEO to GEO and continuing to Lunar Orbit In each instance, there was no “off the shelf” solution that could meet these requirements but a relatively common HP-SEP OMV could. This allows for reduced overall costs with a design phase that applies to several missions and potentially a platform that can be produced at one to two units per year consistently reducing production costs. It was the positive results from this study and industry interest that warranted further mission study and definition of the HP-SEP OMV configuration.

III. COMPASS Study

A. Collaborative Modeling for Parametric Assessment of Space System (COMPASS)

The NASA Glenn Research Center (GRC) COMPASS team was formed in 2006. As a result of its success and several subsequent projects, this multidisciplinary concurrent engineering team continues its mission to produce preliminary spacecraft system designs for space missions. COMPASS has performed over 100 studies since its inception for a wide variety of customers and mission types ranging from a lunar robotic lander, a Mars Ascent Vehicle, several CubeSats, and a submarine for use on Titan. Many of the studies involve SEP and leverage much of the power and propulsion technology developed or funded by GRC. The COMPASS team was awarded the NASA Systems Engineering Excellence Award and the Space Flight Awareness Team Award in 2014.

Subsystem design options and technologies need to be integrated into a full vehicle or architecture to assess the impact of each subsystem's design on other systems. These assessments require many different skills from multiple organizations. Thus, forming the team and defining interactions can take significant time and effort. COMPASS studies eliminate rework through consistent processes, tools, and subject matter experts enabling space system design assessments that are conducted rapidly in a collaborative environment. Typically at the conclusion of a study, a final report is prepared which includes the customer's request, outlines the study problem, details the assumptions and requirements used for analysis, and lists the details of the final design.

COMPASS does business across NASA, with industry partners, and other government agencies. COMPASS studies can be tailored to support proposals, project reviews such as Mission Control reviews, system requirements reviews, and implementation of technologies. The products from a design study depend on the scope of the design and the customer agreed upon products. At a minimum, the team produces an annotated chart package detailing the system design, risk, and costs. Chart packages also include the following:

- Master equipment list and costing based on work breakdown structure elements
- Mission design and trajectory optimization; trade space investigations
- Proposal quality final reports, presentation, CAD drawings, figures, plots, tables, and animations

B. Solar Electric Propulsion ESPA (SEP-ESPA) Study Overview

The SEP-ESPA study was performed from November 8-21, 2016 in the COMPASS lab at GRC. In addition to Moog and NASA personnel, the Aerospace Corporation provided on-site support to provide insight from similar studies[2]. The initial trade space was to assess a technology demonstrator concept that would be relatively similar to an operational mission that could perform a LEO to GEO transfer of 5,000 kg of payload for military applications, a commercial variant that could be used for similar transfers from LEO or GTO to a variety of orbits, and finally a very high delta-V system that could be leveraged for NASA science missions. Initially, it was thought a concept let alone a design would not be able to meet the needs of these three customer types, but the flexibility within the OMV concept and keeping with that design rationale made for a system that could meet each need in particular the desired cost point. Each of the key requirements for the four cases are summarized in Table 1. The launch mass, payload mass, delta-V, and transfer time were provided from top level mission needs or market needs relatively to other options.

Table 1. SEP-ESPA Mission Cases Inputs

Parameter	Case 1 - Demonstrator	Case 2 – Transfer Vehicle	Case 3 – Commercial Applications	Case 4 – NASA Applications
Max Vehicle + Payload Wet Mass	5000-6000 kg	~10,000 kg (including 5000 kg Primary)	<2500 kg	TBD, within Case 1-3
Payload(s)	Up to six 300 kg Small Sats (~1750 kg)	~5000 kg	Six 180 kg Small Sats, exploration missions	TBD, NASA science and exploration
Mission Type	Transfer to GEO (and radiation dose equivalent to transfer from LEO to GEO)	Transfer from LEO to GEO	GEO, LEO, Lunar, Asteroids, Debris mitigation, Space Asset Management, LEO Constellation delivery	Near Earth Objects, Mars, Deep space probes
ΔV	6 km/s	6 km/s	<6 km/s (LEO to GEO), ~ 3 km/s (GTO to GEO)	10-15 km/s
Transfer Time	GTO-GEO (5 mos),	6-8 months (transfer time is the priority)	< 1 year (cost is the priority)	1-4 years (delta-V is the priority)

	GEO to 8000 km (4 mos), 8000 km to Lunar Halo (6 mos)			

To minimize the transfer time, a Hall Effect Thruster (HET) technology was selected for each study case but the size and quantity of these engines was variable. HETs were chosen because of their higher Thrust to Power (T/P) ratio than gridded ion technology and availability of high power flight units in the 2019 or sooner timeframe (compared to other advanced EP technologies). Three HETs were selected as part of the study with the NASA 12.5 kW TDU engine that was developed for the Asteroid Return and Redirect Mission (ARRM), the Aerojet Rocketdyne 4.5 kW XR-5 engine, and the NASA 3.8-4.5 kW HiVHAc engine.

The mission transfer time is inversely proportional and linear with continuous thrust through Newton's Second Law so faster transfers required more thrust. More payload mass requires more thrust to maintain the same transfer time. The HETs were traded based on their nominal T/P (e.g. 55 mN/kW). This allowed the independent variables of total launch mass, payload mass, orbit transfer, and maximum transfer time to be dependent on available power for the engines. This greatly simplified the preliminary mission trades without first doing detailed analysis on each configuration. Engine Isp impacted the total launch mass but this was a lower order concern initially as it was related to the T/P (higher T/P usually has lower Isp) so it too became dependent on power. The power was balanced with the mission costs as the solar array power scaled approximately linearly with cost (e.g. constant $/ W) and was the largest variable in system cost, so a trade of all required mission parameters and cost could be made based on power. Once an "engine power budget" was determined based on mission needs this could be compared to the available engine selections. Table 2 shows the outputs for the mission cases to balance the cost requirements. Note that cost requirements and desirement are not shared in this paper for proprietary reasons but were a key part of the study.

Table 2. SEP-ESPA Mission Cases Outputs

Parameter	Case 1 - Demonstrator	Case 2 – Transfer Vehicle	Case 3 – Commercial Applications	Case 4 – NASA Applications
Solar Power (BOL)	35 kW (2 x 17.5 kW Arrays)	35-50 kW (2 x 17.5 kW arrays)	~20 kW (2 x ~10 kW arrays)	TBD kW
EP power	25 kW = Two 12.5 kW HETs	31 kW = Seven 4.5 kW HETs + Spare	~15-20 kW = Three or Four 4.5 kW HETs	TBD, kW
Isp	2600 s	~1800 sec	~ 1800 sec	3000s

Case 1 included a non-standard transfer associated with validating the system capability in a relevant space radiation environment and also looking to architect a mission concept of operations (CONOPS) that would minimize costs. Ideally in an operational mission the SEP-ESPA would be used to transfer from LEO (28.5° inclination) to GEO (0° to 3° inclination) in Case 2 so a similar transfer would be appropriate for a demonstration mission, but there are very few regular launches to LEO and 28.5° inclination with most being 51.6° or polar. GTO launches are regularly at this inclination and could have the needed excess capacity (in particular when coupled with an "all EP" primary spacecraft). In order to decrease the mission costs and essentially provide ballast for the mission, it was assumed that up to six 300 kg small satellites would be included in this launch. These would be carried on an additional ESPA ring as they wouldn't be part of the standard configuration. There are limited opportunities for spacecraft of this size to reach GEO so the value of this access could be used to offset launch and/or mission costs. The initial goal was to deploy these spacecraft to GEO first before continuing with the rest of the mission due to radiation exposure concerns. Therefore the SEP-ESPA would first transfer from GTO to GEO in approximately 5 months. It was assumed these payloads would need to be deployed as soon as possible in the mission to minimize the radiation exposure. Once it had deployed the six payloads in GEO, the SEP-ESPA would "spiral down" to an intermediate Medium Earth Orbit (MEO) intentionally increasing the overall radiation exposure. A circular orbit of 8000 km provided an adequate "radiation soak" to meet an equivalency of a LEO to GEO transfer. Upon completion of this, the SEP-ESPA would then "spiral up" past GEO and end in a lunar halo orbit. Inclination would remain constant in the spiral down and spiral up phases. The goal was to remove the SEP-ESPA from the GEO belt and provide a cooperative target that could be part of a future manned lunar mission.

C. Case 1 Orbit Transfer Planning

Two NASA software packages, Optimal Trajectories by Implicit Simulation (OTIS) and Copernicus, were used to simulate Case 1 phases of the CONOPS to provide mission trades with delta-V estimates and transfer times. The delta-V and transfer times were used with the HET selection and Isp to calculate a propellant mass. Case 1 mission assumptions were 2 x 12.5 kW thrusters, 2600 s Isp, 65% engine efficiency, 90% duty cycle, yielding a total thrust of 1.15 N. Table 3 shows the Case 1 orbit transfer CONOPS.

Table 3. SEP-ESPA Case 1 Orbit Transfer CONOPS

Event	Orbit	Payload Mass	Notes
Launch to GTO	35,786 x 300 km, 28.5° inclination	1750 kg	6000 kg starting S/C mass
Spiral to GEO	35,786 km circular, 0° inclination	1750 kg	Minimum delta-V spiral
Deploy Payloads	35,786 km circular, 0° inclination	0 kg	Assume payloads can phase to desired location
GEO to MEO Spiral	8,000 km circular, 0° inclination	0 kg	Minimum delta-V spiral
Loiter at MEO	8,000 km circular, 0° inclination	0 kg	Loiter duration based on total mission radiation dose
MEO to Lunar Near-Rectilinear Orbit (LRO)	LRO	0 kg	Minimum delta-V spiral
Loiter as required	LRO	0 kg	Loiter duration based on follow on missions

OTIS phases of the mission were run using "Directed Adaptive Guidance" steering to target final conditions on semi-major axis, inclination, and eccentricity with optimal tuning coefficients for each to drive minimal delta-V solutions. OTIS phases also included the effects of shadowing with the assumption that no thrust occurs during shadow. The batteries required to thrust during shadow would have added a large cost and mass to the system. To examine the impact of sun angle losses (beta angle loss), the OTIS portions of the trajectory were run with and without a cosine beta loss parameterization on power. Copernicus was used for the MEO to GEO then NRO transfer.

Table 4 shows the comparison between the beta angle loss for the transfers and the impact on the delta-V and transfer duration. The difference was negligible between the two so for future trades the beta loss term was not used. An assumption of 40 days at MEO was used for the mission planning.

Table 4. SEP-ESPA Case 1 Orbit Transfer CONOPS Trades with (left) and without (right) beta angle loss

no beta angle loss				
Phase	dV	Eng on Time	Duration	Prop
	m/s	mo	mo	kg
GTO to GEO	2863	5.15	5.51	610.78
GEO to MEO	2424	3.93	3.93	466.43
Loiter @ MEO	0	0.00	1.33	0
MEO to LRO	3930	5.83	5.83	500
Total	9218	15	17	1577

cosine beta loss				
Phase	dV	Eng on Time	Duration	Prop
	m/s	mo	mo	kg
GTO to GEO	2870	5.16	5.69	612.13
GEO to MEO	2431	3.94	4.34	467.01
Loiter @ MEO	0	0.00	1.33	0
MEO to LRO	3930	5.83	5.83	500
Total	9232	15	17	1579

The OTIS software provides plots of the transfers that can be used to visualize the transfer. As it is a three-dimensional transfer from GTO to GEO the visualization requires views from three views (X-Y, Y-Z, and Y-Z) to fully appreciate the overall transfer. Figure 2 shows this with Earth located at 0,0 and white portions of the plot are when the system is not thrusting as it is in shadow. Figure 3 shows the GEO to MEO transfer which is coplanar and a ‘flatter’ transfer making it easier to visualize. The full MEO to LRO transfer was not plotted as this was secondary to the mission.

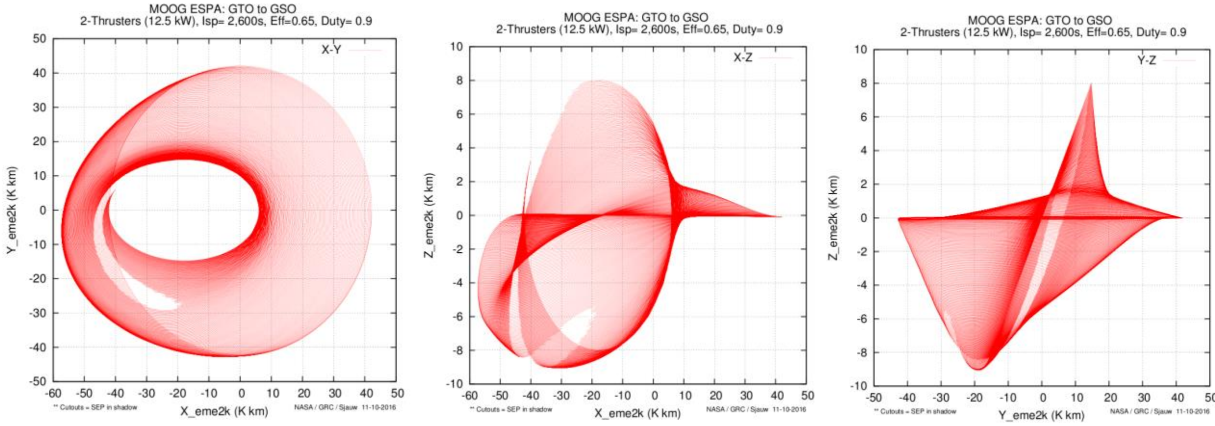


Figure 2. OTIS Simulation for GTO to GEO Transfer (X-Y, X-Z, and Y-Z views)

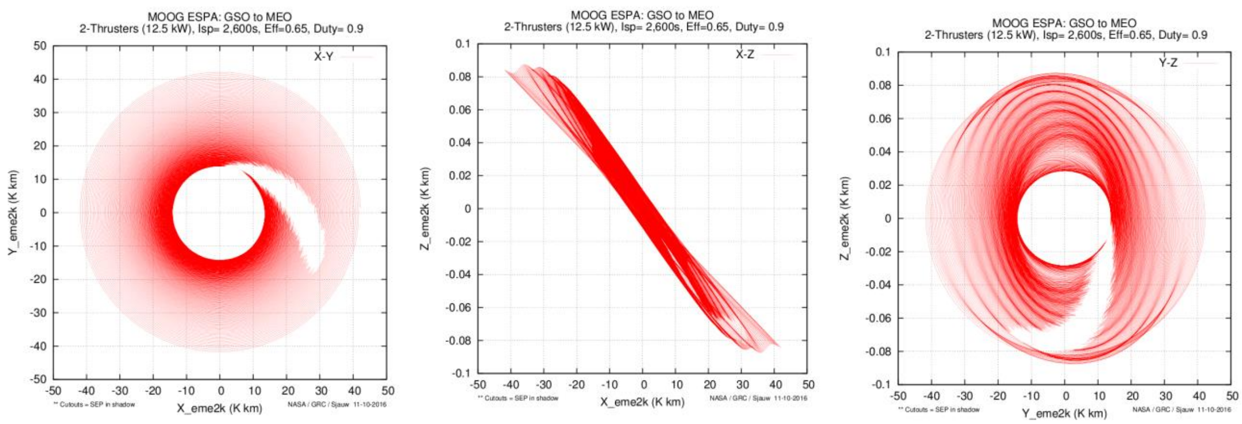


Figure 3. OTIS Simulation for GEO to MEO Transfer (X-Y, X-Z, and Y-Z views)

The 9.2 km/s delta-V budget was the most stressing mission of Cases 1 through 3. Between propellant for the transfer and attitude control (discussed later), the system was designed to hold up to 1890 kg of Xenon propellant. For Case 1 this required seven Xenon tanks (see Figure 4). Case 2 and 3 required less propellant and reduced the number of Xenon tanks to five and three, respectively. Tanks were removed to maintain symmetry to keep the center of gravity in roughly the same location between variations.

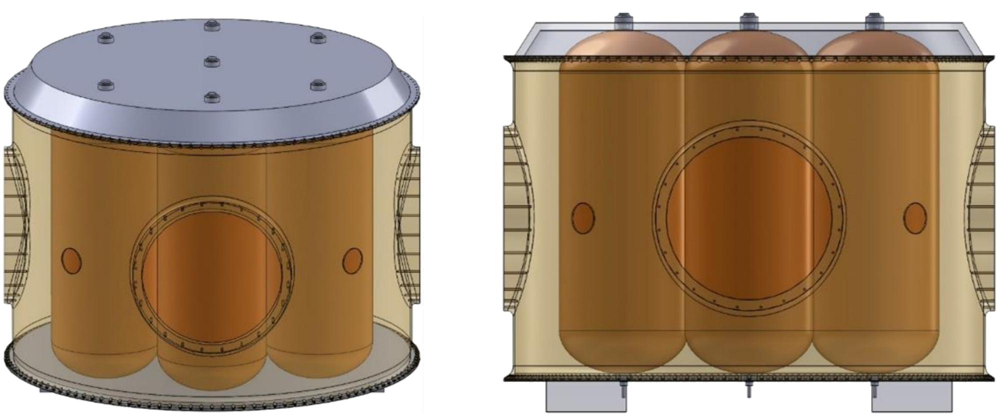


Figure 4. SEP-ESPA ESPA Grande with Seven Xenon Tanks

IV. SEP-ESPA

A. SEP-ESPA Overview

The preliminary study assessment showed the biggest variable would be system power and once that was established, the quantity and type of HETs could be determined, and this in turn would determine the amount of propellant needed. This allowed for a spacecraft block diagram that could meet all four mission cases with variations of the solar array power and quantity of HETs. A series of common Xenon tanks was selected to allow for variation between the cases with a minimum of three used and a maximum of seven depending on the configuration. This “building block” approach allowed for large variations in mission needs with a common design. This is critical in both minimizing development cost for what are multiple configurations and reducing recurring unit price as each system is roughly the same as the other. This is analogous to the automotive industry design methodology. Figure 5 shows a simplified block diagram of the system. One trade was for Case 2, the primary spacecraft could provide the control systems such as the Command and Data Handling (C&DH); Guidance, Navigation, and Control (GN&C), and Communications (Comm). To simplify trades it was assumed that the capabilities of the SEP-ESPA control systems would be the same as if they were in the free flyer or part of the primary spacecraft.

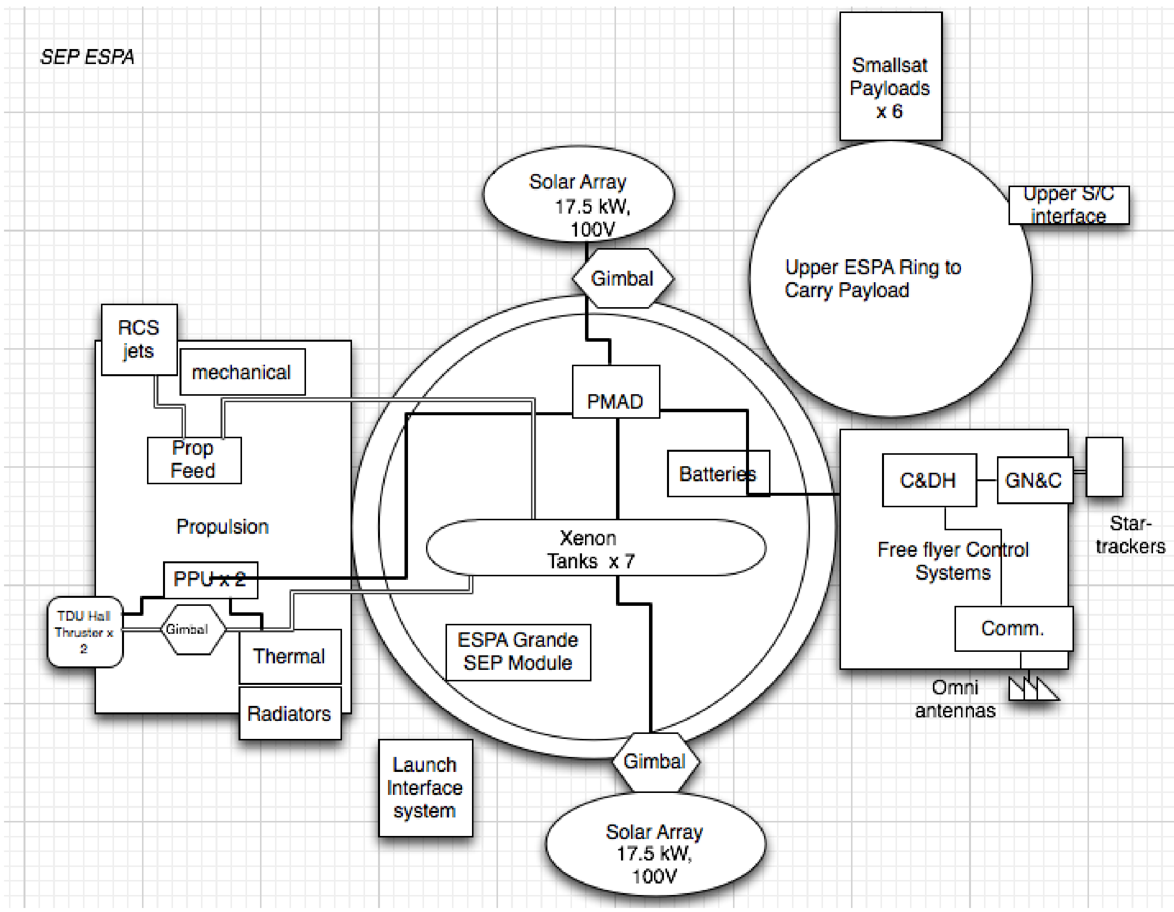
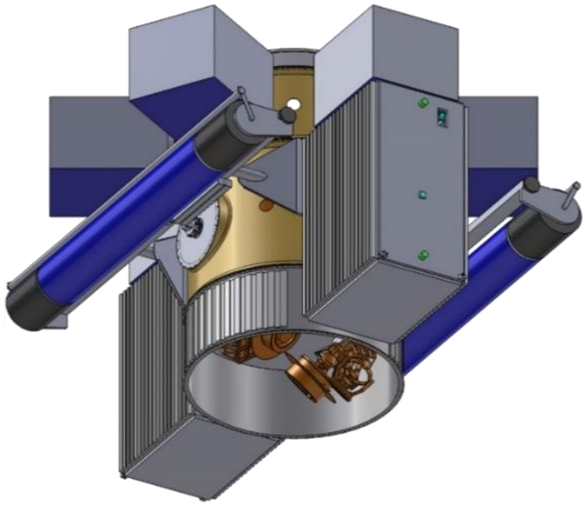


Figure 5. SEP-ESPA Block Diagram (Case 1)

An additional constraint is the system must fit within an existing launch vehicle and payload fairing (PLF). The Falcon 9 launch vehicle was selected for preliminary trades which provided a maximum stowed diameter. The Falcon 9 PLF would roughly encompass the Atlas V 5 meter payload fairing meaning the system could be used on other launch vehicles. This 5 meter class payload fairing is a common design standard. The height of the system was also a limitation as for Cases 1 through 3, and likely Case 4, there would be a large satellite on top of the SEP-ESPA. This created a maximum height consideration. These two constraints led to the conclusion that the Deployable Space Systems (DSS) Roll-Out Solar Array (ROSA) was the best option for this system. Another advantage was this type of design allowed for easier trades of the solar array power. More power means longer arrays and less power means shorter arrays. By designing for the higher power array requirement this would encompass lower power arrays. Figure 6 and Figure 7 show the SEP-ESPA launch configuration and deployed configuration.



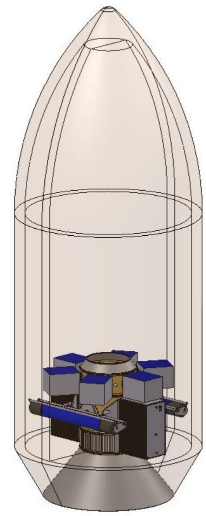


Figure 6. SEP-ESPA Case 1 with six 300 kg payloads, standalone (left), in Falcon 9 Payload Fairing (right)

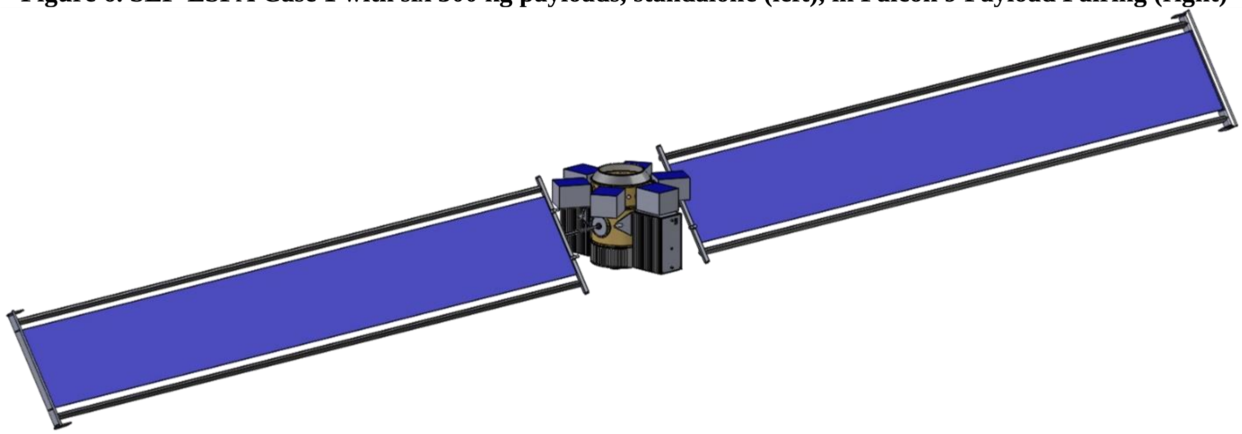


Figure 7. SEP-ESPA Case 1 with six 300 kg payloads and two 17.5 kW solar arrays deployed

The ESPA provided a simple and easily adaptable method of packaging all the required equipment. The SEP-ESPA used three adapters for the build-up of the system. The core of the system is a based on an ESPA Grande that is 42" tall and has four 24" ports. Each port can hold 454 kg of mass[3] providing a method of supporting the solar arrays (each on a port) and the equipment boxes (each on a port). Up to seven Xenon propellant tanks are mounted vertically and internally as discussed previously.

The lower portion uses a C-22 launch vehicle adapter. This is a flight heritage launch adapter commonly used on Atlas V launches. As there would be no payloads located there a lower mass structure was selected than another ESPA. One advantage to the ESPA and C-22 stack is both structures are designed for very large and heavy primary payloads and both have extensive flight heritage. This provided another example of a flexible building block approach to minimize development costs while still being adaptable. The equipment boxes and C-22 adapter provided surface area for mounting radiators which was a key derived requirement. The power processing unit (PPU) for the HETs is not 100% efficient so approximately 7% of the electrical power is converted to thermal power in the form of waste heat that must be radiated to space. Depending on each case this could be greater than 2 kW of thermal power.

The standard ESPA located on the upper portion provides a method of carrying up to six secondary payloads for Case 1. For Case 2 and Case 3 this ESPA may or may not be used depending on the mission needs. The ESPA is a common payload adapter and many 300 kg or less payloads plan for this method of launching. The ESPAs as building blocks allowed for a system to be developed without prior knowledge of the primary payload or secondary payloads.

B. Systems Configuration and Launch Detail

The overall SEP-ESPA for Case 1 was treated as two major elements with the Power and Propulsion Module (PPM) and the Demo Control Module (DCM). The justification was the PPM would be mostly common between each case but for Case 2 there was the potential the primary spacecraft could provide the control system such as C&DH, GN&C, and Comm. Overall Case 1 had a predicted total mass of 3278.9 kg (see Table 5). This included an overall Mass Growth Allowance (MGA) of 6%. The MGA process and margin/contingency process is from an internal COMPASS document[4] and leverages industry standards[5]. Each component and subsystem was assessed based on design and technical maturity per the MGA policy. An additional 112.6 kg of mass as part of system level growth was used bringing the standalone SEP-ESPA launch mass to 3391.5 kg before adding the upper ESPA and secondary payloads (see Table 6). The ESPA, adapter for the primary payload and secondary payloads brought the total launch mass to 5406.4 kg (see Table 7).

Table 5. SEP-ESPA Case 1 Mass Summary

Description	Basic Mass	Growth	Growth	Total Mass
Case #1 ESPA OMV 2016 CD-2016-139	(kg)	(%)	(kg)	(kg)
ESPA OMV	3094.2	6.0%	184.8	3278.9
Power and Propulsion Module	3040.5	5.9%	178.7	3219.2
Electrical Power Subsystem	301.0	32.5%	97.8	398.8
Thermal Control (Non-Propellant)	92.0	18.0%	16.6	108.6
Propulsion (EP Hardware)	337.1	10.0%	33.7	370.7
Propellant (EP)	1890.0	0.0%	0.0	1890.0
Structures and Mechanisms	420.4	7.3%	30.7	451.1
Demo Control Module	53.6	11.4%	6.1	59.7
Attitude Determination and Control	16.3	3.0%	0.5	16.8
Command & Data Handling	16.8	17.2%	2.9	19.6
Communications and Tracking	6.5	3.0%	0.2	6.7
Electrical Power Subsystem	0.0	0	0.0	0.0
Thermal Control (Non-Propellant)	14.1	18.0%	2.5	16.6

Table 6. SEP-ESPA Case 1 Mass Summary with System Level Growth

Moog_ESPA_OMV Summary Mass Calculations	Basic Mass (kg)	Growth (kg)	Predicted Mass (kg)	Aggregate Growth (%)
Moog_ESPA_OMV Total Wet Mass	3094.2	184.8	3278.9	
Moog_ESPA_OMV Total Dry Mass	1204.2	184.8	1388.9	19%
Dry Mass Desired System Level Growth	991.2	297.3	1288.5	30%
Additional Growth (carried at system level)		112.6		11%
Total Useable Propellant	1890.0		1890.0	
Total Trapped Propellants, Margin, pressurant	0.0		0.0	
Total Inert Mass with Growth	1204.2	297.3	1501.5	
Moog_ESPA_OMV Total Wet Mass with system level growth	3094.2	297.3	3391.5	

Table 7. SEP-ESPA Case 1 Mass Summary with System Level Growth at Launch

ESPA OMV Total Wet Mass with System Level Growth	3391.5	kg
Mass, 1194 Adapter	62.0	kg
Mass, Upper ESPA Ring	136.0	kg
Mass, 6 Payloads at 300 kg each, attach to Upper ESPA	1800.0	kg
Mass, 6 Lightband adapters at 2 kg each, ESPA side	12.1	kg
Mass, 6 Lightband adapters at 0.8 kg each, Payload side	4.8	kg
Total Wet Mass with System Level Growth At Launch	**5406.4**	**kg**

The 5406.4 kg launch mass was within the initial goal of 5000 to 6000 kg. This would provide nearly 2700 kg mass for a GTO primary launch vehicle (see Table 8). This available launch mass is on the order of the launch mass for "all electric" spacecraft like the Boeing 702SP[6] or smaller communications satellites like the Lockheed Martin A2100-A size spacecraft. If the maximum launch mass were exceeded, a smaller amount of Xenon could be used as the primary demonstration mission can be achieved with ~1200 kg of propellant so nearly ~700 kg of margin. A notional six 300 kg payloads (~1750 kg) were used, but the actual values are unknown and could be less. These two values could be balanced with matching GTO primary satellites. This could allow for cost sharing of a single launch vehicle reducing costs for both the SEP-ESPA and the primary.

Table 8. SEP-ESPA Case 1 Launch Mass and Margin

Launch Architecture Details, Moog_ESPA_OMV		Units
Launch Vehicle	Falcon 9 FT	
Injected Orbit	GTO	
ELV performance (pre-margin)	9000	kg
ELV Margin (%)	10%	%
ELV performance (post-margin)	8100	kg
Total Wet Mass at Launch with System Level Growth	**5406**	kg
Available ELV Margin	**2694**	kg
Available ELV Margin (%)	**33%**	%

C. Attitude Determination and Control System (AD&CS) Detail

The AD&CS is used to determine and control attitude of the spacecraft for the following phases: null tip-off rates after launch vehicle separation, HP-SEP cruise and coast, thrust vector in the required direction as dictated by guidance, orient solar arrays to provide the required power to the vehicle, safe mode and maintain 3-axis stabilization. Although a demonstration mission, the concept is meant to be expanded into an operational transfer vehicle in Cases 2 through 4 so a single fault tolerant design was used.

For initial sizing the following simplifying assumptions were used: simple geometric shapes in the calculation of moments of inertia, negligible products of inertia, swirl torque from EP thrusters can be offset by gimballing the thrusters a minimal amount (< 1 degree), and SEP thrusters are used for primary means of vehicle control while in use. Table 9 provides a summary of the AD&CS sensors and actuators. To minimize development costs a suite of flight heritage actuators were chosen. For ACS thrusters, a Xenon cold gas solution was used to minimize the cost and complexity of the overall system. Depending on the mission scenario this may have required over 100 kg of Xenon propellant due to the fairly inefficient use of Xenon as a cold gas (Isp ~ 30 sec) that is likely better used as primary propulsion. An alternate hydrazine-based ACS thruster system was developed as an option.

Table 9. SEP-ESPA AD&CS Sensors and Actuators Summary

Sensor/Actuator	Make/Model	Use	Notes
Star Tracker	2x DTU Micro Advanced Stellar Compass Star Trackers	Provides vehicle inertial attitude estimation	2x Optical heads and 2x Electronics units
Inertial Measurement Unit	2x Honeywell MIMU	Gyros estimate vehicle body rates, Vehicle attitude estimated	Radiation hardness capability > 100 Krads
Sun Sensor	8x Adcole coarse analog sun sensors (CASS)	Coarse attitude determination, Knowledge of direction to Sun for safe mode	
Orbit Location	2x Moog NavSBR GPS Receivers + 2x Antennae	Precision Orbit Determination	Works above MEO (e.g. GPS @ GEO)
Attitude Control	16x Moog 58E151 Cold Gas Xenon ACS thrusters	Control of the vehicle when SEP thrusters are not in use, detumble after launch vehicle separation, Eclipse periods	Assumed Isp of 28 s

For Case 1 the two HETs are mounted independently on a two-axis gimbal that can be used to gimbal the HET be used to gimbal the HET 35° within two degrees of freedom (see Figure 8). The two-axis gimbal has a “launch lock” that reduces the loads on the gimbal during launch. This style of gimbal has been successfully deployed on the AEHF spacecraft. The two gimballed engines provide pitch, yaw, and roll control when thrusting. The advantage to having gimballed engines beyond the ability to adjust for a changing center of gravity within the vehicle, is this configuration can account for differences in thrust between the engines. This is useful for an "engine out" scenario or in the event two different engines are used. Case 1 was intended as a demonstration mission for the concept but also some of the key elements like the HET and solar arrays. It is possible to demonstrate two different HETs with the TDU engine such as the Busek BHT-8000 or Aerojet Rocketdyne XR-12. This is useful in Case 3 where 5 kW class engines would be used. Similarly if a different engine technology entirely such as a gridded ion engine were used the overall system can accommodate some thrust mismatch.

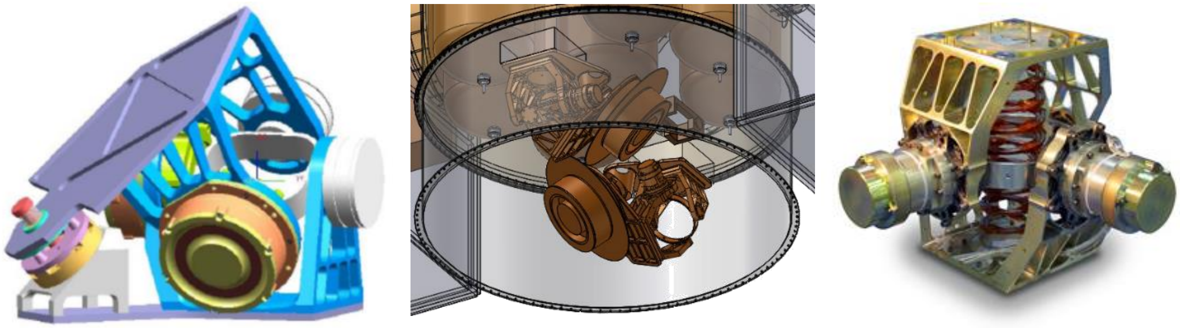


Figure 8. Two-Axis Gimbal, in launch configuration (left), in SEP-ESPA (center), and actual unit (right)

D. Power Subsystem Detail

The HET options all require >100 VDC for operating voltage but many spacecraft bus components use 28 VDC. The PPU for each HET has a DC/DC convertor to increase line voltage to the needed 200 to 700 VDC but the greater the starting voltage the more efficient the design. To allow for the greatest use of existing technology a "split voltage" array configuration was determined to be an optimal solution. The solar array cells would have some strings wired such that they provided 100 VDC and other cell strings provided 28 VDC. This allowed for decoupling of the bus segment and high power system and the bus segment to use existing technologies. Figure 9 provides a block diagram of the power subsystem configuration. The assumption that the SEP system would not be powered during shadow/eclipse parts of the orbit simplifies the power storage assumption. The battery can be sized to just maintain the bus elements and not the HP-SEP elements.

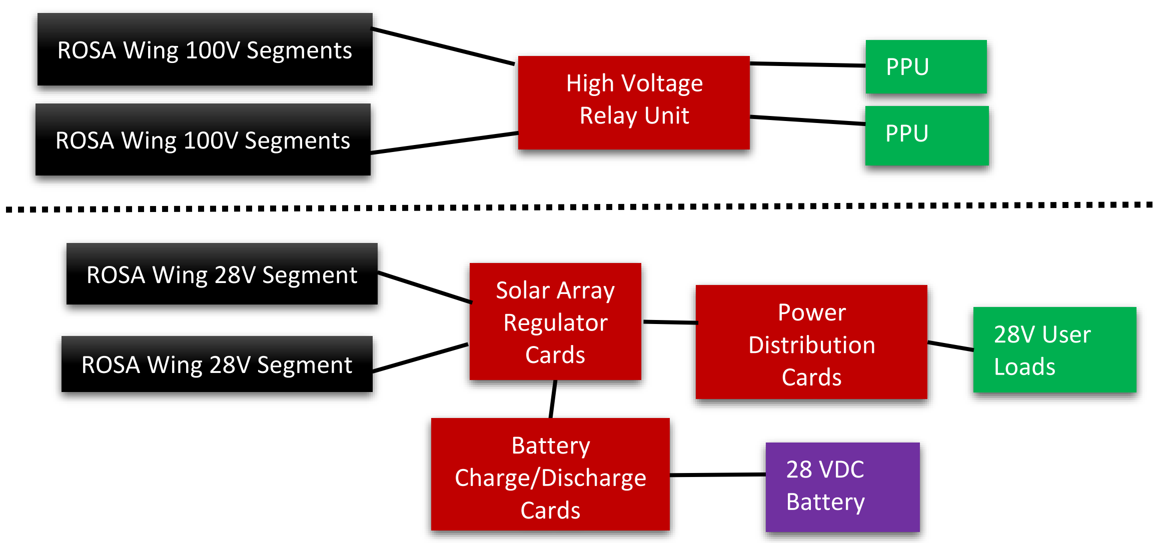


Figure 9. SEP-ESPA Power Distribution Block Diagram

Using the above block diagram a power subsystem was architected and sized to meet the mission needs. As with the other subsystems the preference was to use existing components and subsystem elements, but two portions of the power subsystem did not have "off the shelf" solutions with the High Voltage Power Management and Distribution

(PMAD) and solar arrays (see Table 10). The ROSA solar array recently had a demonstration flight on the International Space Station[8] that proved the roll-out technology in a space environment, but a 17.5 kW variant with the split 100/28 VDC configuration would need to be developed. Similarly the High Voltage PMAD is derived from flight heritage but not in this configuration. Case 1 for the SEP-ESPA would be used to demonstrate these two technologies that are each currently at TRL 6. To reduce overall complexity there is the potential to move the Low Voltage PMAD element into the C&DH avionics (discussed later)

Table 10. SEP-ESPA Power Subsystem Summary

Element	Make/Model	Use	Notes
Power Generation	2x ROSA solar array wings	Generate 17.5 kW of power per wing	29.5% efficiency ZTJ cells, 6 mil coverglass, 160 W/kg, Wing size is 3 m by 20 m; aspect ratio= ~7
Energy Storage	7S2P SAFT VES 180 space qualified cells	Contains 2500 W-hr , designed for 30% depth of discharge	Provides 500 W at 28 VDC with 1.5 hour eclipse
High Voltage PMAD	Custom, based on Dawn spacecraft	Provides 100 VDC unregulated bus to PPUs	Unregulated so PPUs need to account for varying input voltage
Low Voltage PMAD	Terma Space cards: 4 x Array Power Regulation Module 2 x Equipment Power Distribution Module 2 x Battery Charge/Discharge Regulation Module	Provides 28 VDC regulated bus to the remainder of the SEP-ESPA	Also trading using Moog Broad Reach Engineering power avionics
High Voltage and Low Voltage Harness	Custom, based on Dawn for High Voltage and existing 28 VDC spacecraft	Distribute 100 VDC and 28 VDC bus voltage	Assuming 15% of the power subsystem mass

E. Propulsion Subsystem Detail

The propulsion subsystem design and sizing is in response to many of the other system trades and requirements. The mission trades determined the required propellant masses and corresponding propellant tanks, the mission case examples determined the range of HETs (both size and quantity) along with power requirements, the AD&CS sizing determined the HET gimbal and ACS requirements, and power system trades for the 28 VDC bus focused the range of existing equipment. The desire to minimize development costs by using existing hardware solutions helped focus the design. The propulsion subsystem used elements that are regularly flying on many of the hybrid EP communications satellites such as the SSL-MDA LS1300 spacecraft and the Lockheed Martin AEHF spacecraft. Figure 10 shows the SEP-ESPA propulsion system schematic and Table 11 describes the system. This system was designed to be common across all mission cases so includes provisions for more than two HETs.

Propellant storage is through a series of identical composite overwrap pressure vessels. Each use a titanium liner with a T-1000 carbon fiber overwrap. The specific variant chosen was the Orbital ATK Model 80458-1 that has flight heritage on AEHF. Each tank has a volume of 7,928 cubic inches and a maximum operating pressure of 2,700 psia. This allows for 270 kg of Xenon at 80°F. This provides the ability to "right size" the number of tanks to the mission needs by changing the number of tanks in parallel. For instance Case 1 required seven tanks for 1890 kg of total capacity, Case 2 required five tanks for 1350 kg total capacity, and Case 3 required three tanks for 810 kg of total capacity. In each case the feed system and engine controls remain the same. This technique is commonly used in sizing solar arrays and batteries so using the same nomenclature Case 1 would be 1S7P configuration, Case 2 would be 1S5P, and Case 3 would be 1S3P. This modular building block approach is used throughout the SEP-ESPA.

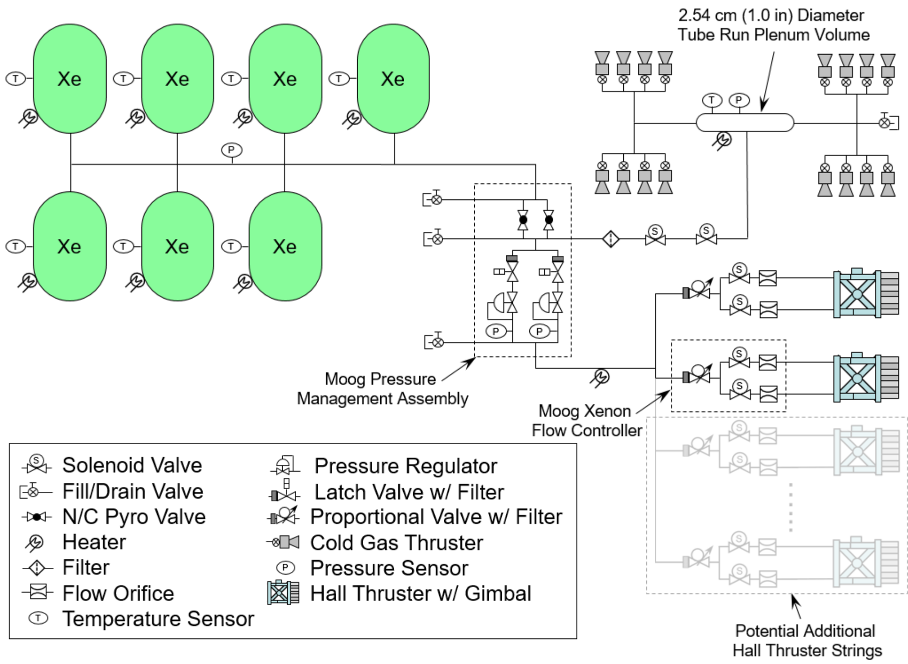
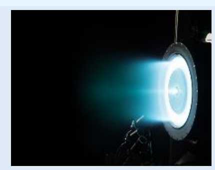
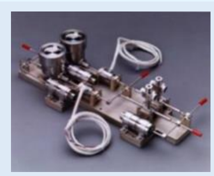


Figure 10. SEP-ESPA Propulsion System Schematic

Table 11. SEP-ESPA Propulsion Subsystem Summary

Element	Make/Model	Use	Picture
Mission Delta-V	2x NASA TDU 12.5 kW HET with provisions for other engines and quantities	Provide > 1 N of thrust during orbit transfers	
Pressure Management Assembly	1x Moog PMA (LS1300 flight heritage)	Reduce Xenon tank storage pressure to useable low pressure for the HETs, parallel redundant design	
Engine Xenon Flow Controller	1x Moog XFC (AEHF flight heritage) per engine	Provides proportional flow control of Xenon to the HET anode and cathode	
Solenoid Valve	2x Moog Solenoid Valve (AEHF flight heritage)	Provides redundant isolation to the Xenon ACS feed plenum	
ACS Cold Gas Thruster	16x Moog Cold Gas Thruster	Provide Nominal Thrust of 4 N (1.0 lbf) for ACS, nominal ISP of 30 s with Xenon	

F. Thermal Subsystem Detail

The thermal system was sized to operate within the environment expected for Mission Case 1. Solar Intensity and view angle as well as the view to warm bodies such as the Earth and spacecraft solar arrays are used to determine the worst case hot and cold conditions. The worst case warm conditions will occur in LEO sunlight conditions with all equipment operating whereas the worst case cold will be in shadow in Earth orbit. Due to the size of the spacecraft and thermal waste heat that needs to be dissipated from the PPUs and PMAD the radiators were distributed over the available surface area to eliminate the need for a deployable radiator.

The radiator is used to reject the waste heat from the spacecraft, the PPUs, and PMAD electronics and use the same radiator system. This is possible since they are operating at the same rejection temperature. The radiator is split up into different sections located on the available surface area on the spacecraft. Each radiator segments is connected together through conductive paths and heat pipes. There is insulation between the radiator and spacecraft body providing a single surface for radiating. The radiator is connected to the cold plates with heat pipes to move heat from the interior to the radiator.

The radiator sizing was based on an energy balance analysis of the area needed to reject the identified heat load to space (see Table 12). From the area a series of scaling equations were used to determine the mass of the radiator. No louvers were utilized for the radiator to simplify the cost and complexity. It is expected that the internal electronics will be operational for the duration of the mission. Variable conductance heat pipes will minimize heat loss during shadow periods. During shadow and if electronics/propulsion system thermal output, decreases heaters will be used to maintain the internal temperature of the spacecraft.

Table 12. SEP-ESPA Case 1 Thermal Subsystem Sizing

Variable	Value
Radiator Solar Absorptivity	0.14
Radiator Emissivity	0.84
Max Radiator Sun Angle View Factor to Earth View Factor to Array & Spacecraft Body	70° 0.25 (worst case) 0.40
Radiator Operating Temperature	280 K to 310 K
Power Dissipation & Radiator Area:	Electronics: 183 W, 0.62 m^2 PPUs: 1875 W, 6.56 m^2

The radiator and variable conductance heat pipe system works as follows: heat is collected by the cold plates from the electronics boxes and other components, the heat is transported to the radiator through the heat pipes, redundant heat pipe loops are used for each cold plate, the radiator dissipates the heat to space, and the radiator is coated to reflect radiation frequencies other than the frequencies associated with the temperature range it will be operating at. This system is passive reducing complexity but does require the addition of heaters to balance out when the PPUs are not operating (i.e. coast mode through shadow). Thermal switches were included to minimize the heat loads back into the radiators when the heaters would be turned on. The radiator surface area was a key parameter and drive to using radiators on the C-22 lower adapter portion of the stack. For smaller heat loads like in Case 3 it is anticipated a smaller radiator surface area will be required.

The remainder of the thermal control subsystem uses traditional elements found in most spacecraft such as multi-layered insulation (MLI) and resistive heaters. The mass of the thermal control subsystem is primarily based on the radiators at nearly 50% with all the remaining elements combined making up the other 50%. Due to the uncertainty at this early point in the design a mass growth of 18% was used for the entire thermal control subsystem.

G. Command and Data Handling Subsystem Detail

The C&DH subsystem had the following design Requirements: avionics perform duties for systems command, control, health management, and data monitoring/acquisition/storage; Radiation Hardened (100 krad) avionics; and

Single Fault Tolerant avionics. One assumption for the single fault tolerant avionics was Single Event Upset (SEU) detection and reset capability. The avionics components were based on military grade commercially available components from proven aerospace system vendors. To ensure ease of flexibility and reuse of existing designs a 3U cPCI form factor and cPCI card cage with backplane is utilized. The system used PowerPC-class processor and several types of I/O cards. The overall enclosure package included any necessary DC-DC converters, filter, and EMI shielding. This Integrated Avionics Unit (IAU) provided the complete C&DH avionics in one box. These requirements could be met through existing Moog Broad Reach (MBR) hardware designed and demonstrated for operations in LEO through Lunar orbit. To meet the single fault tolerant requirements, an identical IAU was used along with a Redundancy Management Unit (RMU) that could detect any SEU faults and switch between the avionics creating a fully redundant A-side and B-side. Figure 11shows this in block diagram form. The redundant IAU is identical to the primary IAU. Overall this meets the mission requirements in a cost effective manner as one set of software and ground support equipment can be used for the system.

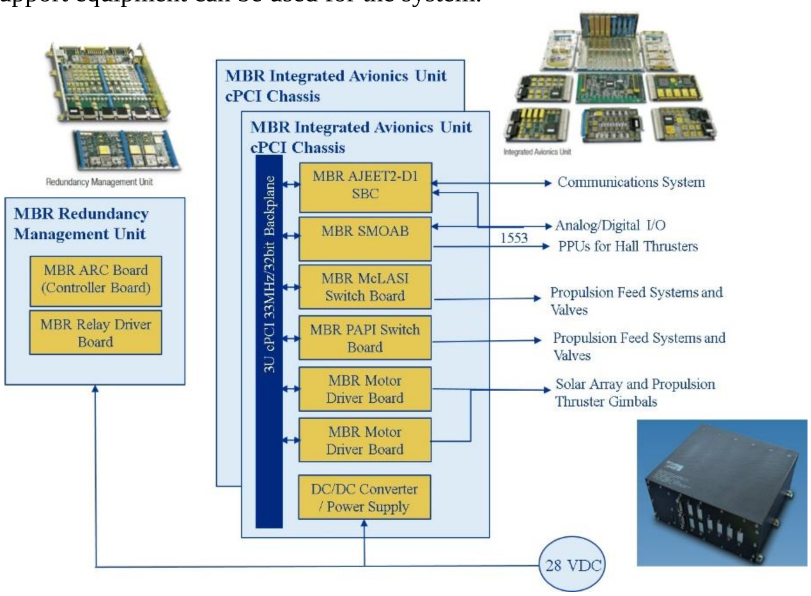


Figure 11. SEP-ESPA C&DH Subsystem Block Diagram

The core of the IAU is the Ajeet single board computer (SBC). The Ajeet is made up of a radiation-hardened BRE440 System on a Chip (SoC) PowerPC Processor and a radiation-hardened Actel RTSX72 Field Programmable Gate Array (FPGA). Data is stored via a 1 GB Flash memory board. A 512 MB double data rate (DDR) dynamic random-access memory (SDRAM) chip provides interim memory storage for computations. The Ajeet has several outputs and inputs to support temperature sensors, reaction wheels, star trackers, sun sensors, transponder, and the IMUs (see Figure 12). Several discrete purpose boards are used for command and control of other elements of the SEP-ESPA such as commanding the PPU, Xenon feed system elements, cold gas thruster valves, HET gimbals, and solar array drives. As discussed in the Power Subsystem section, there is the potential to merge the 28 VDC bus power elements into the IAU reducing the complexity of the PMAD. In this scenario the PMAD would only focus on high voltage operations. This could reduce the development risk and allow for all low voltage elements of the SEP-ESPA to be controlled through a single avionics suite.

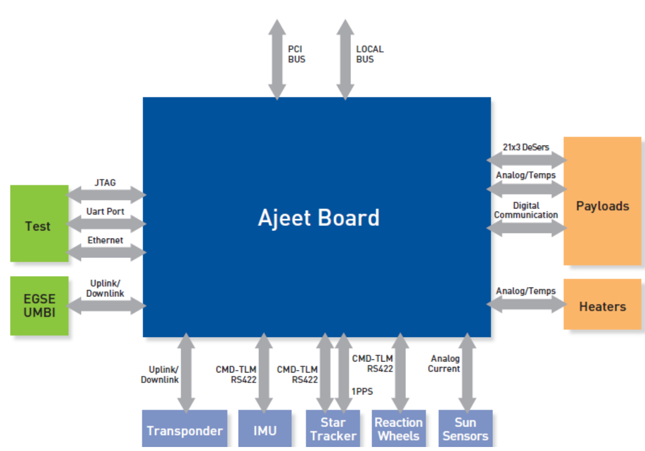


Figure 12. SEP-ESPA Ajeet Block Diagram

H. Communications Subsystem Detail

The purpose of the Communications Subsystem is to provide a data downlink and telemetry support between SEP-ESPA and the Earth during all mission phases. For the different mission types compatibility with Air Force Satellite Control Network (AFSCN) and Tracking and Data Relay Satellite System (TDRSS) was desired along with the ability for commercially available services (Case 3). The initial data budget was a minimum 8 kbps data downlink with a goal of 5 Mbps 900 km to GEO with a link budget requirement of 3 dB communications link margin, standard for typical space applications. An S-band system was selected that could meet these requirements and would be relatively low cost compared to other options like Ka-band. The General Dynamics Multi-Mode Standard Transponder (MST) was selected to meet these requirements. An AntCom S-band hemispherical patch antenna with a greater than 3dBi gain was selected. To meet the redundancy requirements a primary and redundant MST/Antenna configuration was used. The NASA GRC “Link Budget Calculator and Design Tool” was used to perform the link budget verifications at both LEO and GEO (see Figure 13 and Figure 14). As the LRO portion of the mission was not a requirement but a stretch goal this was not analyzed. At LEO, 10 W of transmit power would be needed and a data rate of 1 Gbps could be achieved. At GEO, 20 W of transmit power would be needed and a data rate of 5 Mbps could be achieved. Both of these exceed the goal of 5 Mbps and are within the capability of the MST.

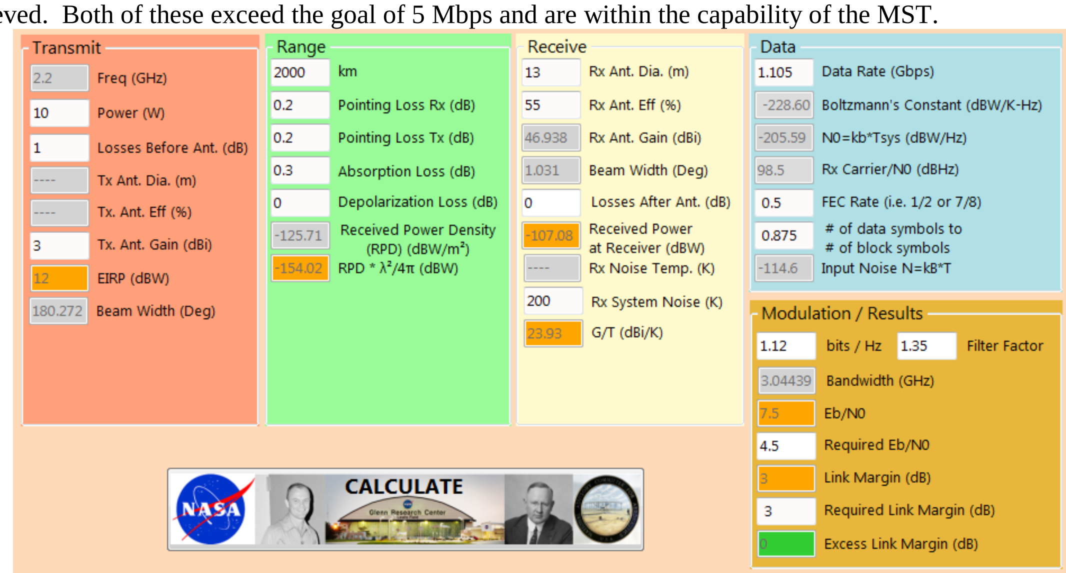


Figure 13. SEP-ESPA Link Budget (at LEO)

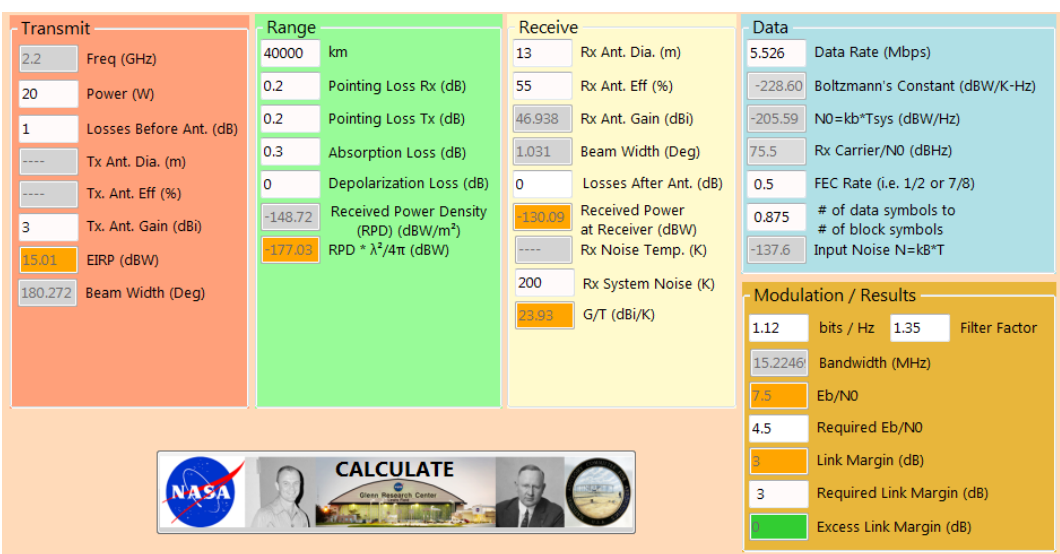


Figure 14. SEP-ESPA Link Budget (at GEO)

I. Mechanical Configuration Detail

The preliminary mission trades determined the stacked ESPA module configuration, the overall launch vehicle PLF requirements, the size and number of Xenon tanks, the type and size of solar arrays, and the size and number of HETs (see Figure 15). The remaining elements were packaged in "boxes" that were mounted on the ESPA Grande ports. The boxes are intentionally oversized to provide available surface area for the radiators as discussed in the Thermal Subsystem section. The boxes in addition to the Xenon tank mounting attachments are the only major structural elements that are unique to the SEP-ESPA and both of these are relatively simple structures to design and fabricate. The equipment boxes provide ample room for integration and test operations reducing the cost and complexity to fabricate the system (see Figure 16).

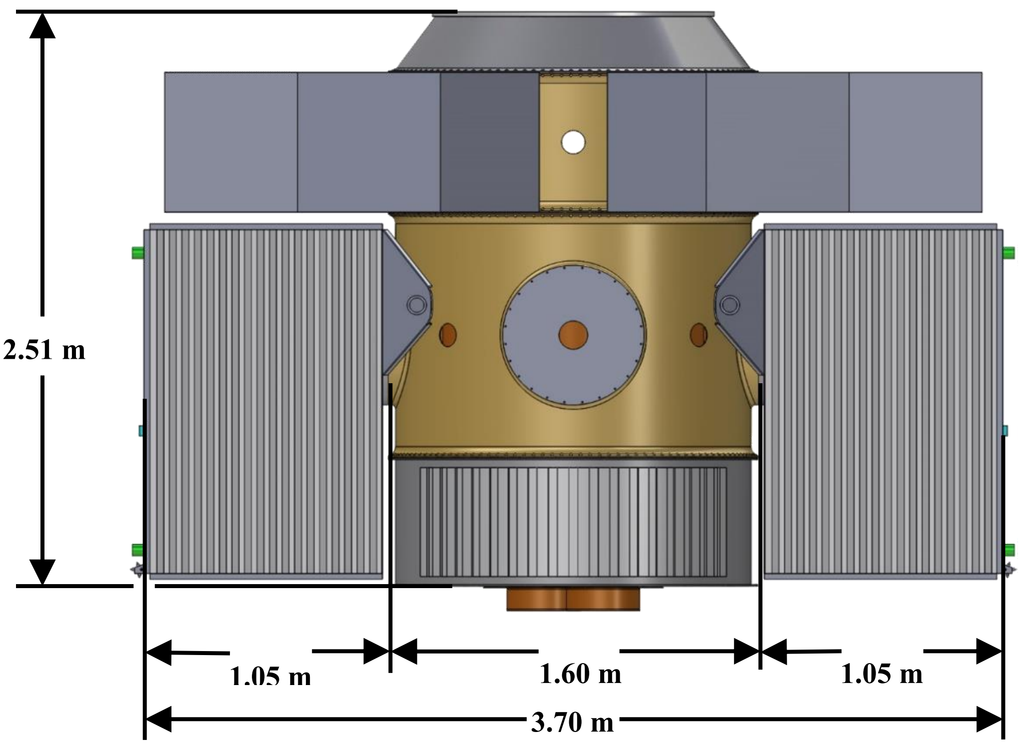


Figure 15. SEP-ESPA Case 1 Dimensions

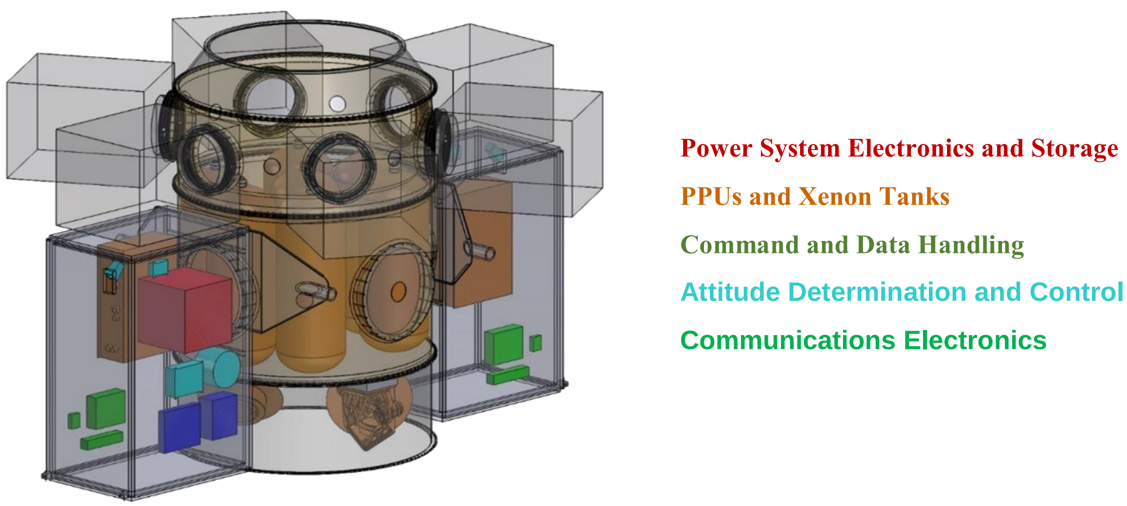


Figure 16. SEP-ESPA Case 1 Transparent View (with subsystem elements identified)

The remaining structures were designed to contain the necessary hardware for avionics, communications, propulsion and power in addition to supporting modules within the stack. The structure needed to withstand applied mechanical and thermal loads across the possible launch vehicles and mission orbits. The equipment boxes were design to limit attached masses as specified by ESPA port carrying capability. The design assumed 6 g axial acceleration during launch and needed to provide minimum deflections, sufficient stiffness, and vibration damping. The structures, except for the ESPAs and C-22, were made from composite sandwich structure with aluminum face sheets & aluminum honeycomb core. The ESPA Grande and C-22 adapter accounted for over half the structure subsystem mass despite no effort to optimize mass. The structures were designed to be cost efficient and easily to fabricate with all of the elements made of some form of aluminum and there are no expensive materials such as carbon fiber composites or titanium.

V. Conclusion

The SEP-ESPA study showed how the OMV concept could be used in a manner to meet very disparate customers and mission types while using relatively common subsystems. This approach minimized both the development costs and the recurring costs. The very large delta-V capability offered by an HP-SEP system enabled mission scenarios that were not feasible using chemical propulsion or other methods. The study showed how a demonstration mission concept (called Case 1) could be flexible enough to be used for many customer types such as military, civil (i.e. NASA), and commercial applications. Designing the system to be “cost flexible” was another key element. By ‘dialing’ the power the system cost can be adjusted while leaving base platform relatively unchanged. The ESPA Grande provides an ideal building block platform for space platforms that can be launched as part of multi-manifest missions. Rideshare compatibility is critical in reducing the overall cost to space access in the future.

Acknowledgments

Moog would like to thank NASA Glenn Research Center, especially Steve Oleson and the COMPASS team, in addition to Michael Barrett for funding the COMPASS team to support this study. Moog thanks Josh Davis from the Aerospace Corporation for his support and insight. Moog would also like to thank NASA Marshall Spaceflight Center in particular Bruce Wiegmann for early support of the HP-SEP OMV concept and supporting mission studies that were precursors to this paper.

References

The following pages are intended to provide examples of the different reference types. You are not required to indicate the type of reference; different types are shown here for illustrative purposes only.

[1] Loghry, C. S., and Stender, M. S., “Rideshare Beyond Earth Orbit using High Power Solar Electric Propulsion (HP-SEP),” *Reinventing Space Conference*, BIS-RS-2016-28, London, UK, October 2016

[2] Penn, J. P., Mayberry, J., Ranieri, C., Rossi, O’B., “Re-Imagining SMC’s Fleet with High-Power Solar Arrays and Solar Electric Propulsion”, *AIAA SPACE 2014 Conference and Exposition*, San Diego, CA, August 2014

[3] Maly, J. M., Sanford, G. E., Williams, A., Berenberg, L., “ESPA Class Redefined”, *31st Annual AIAA/USU Conference on Small Satellites*, SSC17-IV-06, Logan, UT, August 2017

[4] NASA Glenn Research Center, COMPASS document CM-2012-01: "COMPASS Concurrent Engineering Team Design Study Requirements and Operations Manual, Version 2"

[5] American Institute of Aeronautics and Astronautics document, AIAA S–120–2006, *Standard Mass Properties Control for Space Systems*

[6] Boeing Satellite Family information, http://www.boeing.com/space/boeing-satellite-family/ [cited July 30, 2017]

[7] Lockheed Martin A2100 Family information, http://space.skyrocket.de/doc_sat/lockheed_a2100.htm [cited July 30, 2017]

[8] NASA Research and Technology website, Roll-Out Solar Array (ROSA) - 07.19.17, https://www.nasa.gov/mission_pages/station/research/experiments/2139.html