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Updated Results for Additive Manufacturing: An Enabling Technology for the MoonBEAM 6U CubeSat Mission

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LIST OF ACRONYMS, ABBREVIATIONS, AND SYMBOLS

ACO	Advanced Concepts Office
ADCS	attitude determination and control system
AIAA	American Institute of Aeronautics and Astronautics
AM	additive manufacturing
C&DH	command and data handling
COTS	commercial off the shelf
DRO	distant retrograde orbit
ED04	Advanced Concepts Office
EM	exploration mission
ER23	Spacecraft & Auxiliary Propulsion Systems Branch
ES21	Structural & Mechanical Design Branch
ES22	Thermal & Mechanical Analysis Branch
EV42	Guidance, Navigation, and Mission Analysis Branch
GN&C	guidance, navigation, and control
GR&A	ground rules and assumptions
ICPS	interim cryogenic propulsion system
IMU	inertial measurement unit
I/O	input/output
iOCB	ISIS on-board computer
iSat	iodine satellite
ISIS	innovative solutions in space
ISO	isolation valve

LIST OF ACRONYMS, ABBREVIATIONS, AND SYMBOLS (Continued)

LDRO	lunar distant retrograde orbit
LEO	low Earth orbit
MEL	master equipment list
MEOP	maximum expected operating pressure
MLI	multilayer insulation
MoonBEAM	Moon Burst Energetics All-sky Monitor
MPCV	multipurpose crew vehicle
MPS	main propulsion system
MSA	MPCV stage adapter
MSC	MacNeal-Schwendler Corporation
MSFC	Marshall Space Flight Center
PLPS	postlaunch pressurization system
PM	photomultiplier
PMD	propellant management device
Pro E	Pro/ENGINEER
PRV	pressure-regulating device
RCS	reaction control system
RF	radio frequency
SBC	single-board computer
SiPM	silicone photomultiplier
SLS	Space Launch System
SP	secondary payload
SPDS	secondary payload deployment system

LIST OF ACRONYMS, ABBREVIATIONS, AND SYMBOLS (Continued)

TBR	to be revised
TE	Technical Excellence
TESS	transiting exoplanet survey satellite
Ti	titanium
TM	Technical Memorandum
TRL	Technology Readiness Level
UHF	ultra-high frequency
UTX	U-band transceiver
XP50	Spacecraft & Payload Integration & Evolution Office
μCAT	microcathode arc thruster

NOMENCLATURE

I_{sp}	specific impulse
m_{prop}	propellant mass
T	temperature
t	thickness
V	volume

TECHNICAL MEMORANDUM

UPDATED RESULTS FOR ADDITIVE MANUFACTURING: AN ENABLING TECHNOLOGY FOR THE MOONBEAM 6U CUBESAT MISSION

1. INTRODUCTION

This Technical Memorandum (TM) summarizes updated results from a concept study named Moon Burst Energetics All-sky Monitor (MoonBEAM), Phase 2, in which propulsion components were updated from the previous phase 1 study. Results from the phase 1 study are found in reference 1. The reader is encouraged to examine that TM as background for the work presented below, though this TM has been written to be a stand-alone document.

In early 2016, the Advanced Concepts Office (ACO) at the NASA Marshall Space Flight Center (MSFC) was asked to complete a preliminary design of a 6U CubeSat that could enable the MoonBEAM science mission. This mission would use small detectors to locate gamma-ray bursts and notify other ground- and space-based observatories that could perform more detailed observations. The science team wished to use conventional, high Technology Readiness Level (TRL) components and standard design margins. Unfortunately, preliminary analysis showed the spacecraft would far exceed the 6U volume, extending up to 8U or 9U, mainly due to the conventional propulsion system and spherical tank which is not optimally shaped for packaging into a CubeSat. Interestingly, the conventional design did meet the mass limit of 14 kg, meaning that the issue was almost totally one of volume and not mass. Logically, the next step was to see what additive manufacturing (AM) could offer. A Technical Excellence (TE) proposal was submitted by MSFC to investigate the possibilities of using AM technologies for the MoonBEAM design. The work was proposed in two phases. Phase 1 would illuminate the mass and volume constraints to be placed on the propulsion system, and give the proposers time to work in the laboratory and prove some of the concepts. Phase 2 would incorporate the laboratory results into a redesigned propulsion system, which would then be used by the spacecraft design team in a revision of the configuration.

During early 2017, soon after the proposal was awarded, the ACO began and completed phase 1 of the study. Results looked promising, and provided the mass, volume, and performance constraints for the propulsion system. During the summer, AM techniques were employed to see if the proposed design was feasible, and to provide an updated propulsion system configuration to be used in the spacecraft design. During the fall of 2017, the ACO spacecraft design team took the updated propulsion system elements and updated the phase 1 design while maintaining several of the original elements, including AF-M315E high-performance monopropellant, which is denser than hydrazine, and microcathode arc thruster (μCAT) electric microthrusters for momentum unloading. Other subsystem elements (avionics, power, thermal, etc.) were carried over from the phase 1 study whenever possible. In addition, the design team also examined an alternative 'strong-back' design in which the propellant tank provides the structural support for the subsystems, and a gas generator concept in which pressurant gas is generated on orbit, eliminating the need for pressurized volumes during launch.

The phase 2 design did not meet the target maneuver capability (ΔV) value of 369 m/s from the previous study, but still provides a substantial capability for a small CubeSat. A brief summary of results is shown in table 1. While nearly all subsystems carried over from phase 1 to phase 2, the propulsion concept changed significantly, including an increase in mass and a loss in performance, which will be explained later in this TM. (The phase 1 mass was incorrectly reported as 13.99 kg in the previous TM,[1] but has been corrected here. An error in the mass rollup failed to calculate the mass growth correctly.) Offloading propellant to meet the 14-kg limit results in lower, but still substantial, maneuver capability. Though the current capability of 304 m/s is short of the 369 m/s goal, the original goal was selected precisely because it was so difficult to achieve, and while the current design cannot reach the target lunar distant retrograde orbit (LDRO), it can reach many other orbits, such as a transiting exoplanet survey satellite- (TESS-) type orbit, an Earth-Moon L3 halo, and others, giving science mission planners many options from which to choose.

Table 1. Comparison of phase 1 and phase 2 results.

Mass Characteristics (kg)		
Subsystem	**Phase 1**	**Phase 2**
Structures	1.06	1.06
Thermal	0.23	0.23
Power	1.63	1.63
Avionics	1.47	1.47
GN&C	0.8	0.8
Science instrument	2.77	2.77
Propulsion	3.83	3.97
Total dry mass	11.79	11.93
Dry mass margin	17%	16%
Propellant	2.26	2.26
Total mass (see text)	14.05	14.19
6U deployer limit	14	14
Propulsion Characteristics (ΔV in m/s)		
Type	Piston tank	Blowdown
Unusable volume	2.5%	10%
Propellant	AF-M315E	AF-M315E
I_{sp} (s)	220	220
Desired ΔV capability	369	369
ΔV with total mass	368	334
ΔV with 0.05 kg prop offload	361	–
ΔV with 0.19 kg prop offload	–	308

As emphasized in the phase 1 TM, the goal of the concept study was to show the enabling aspects that AM can provide to CubeSats, and not necessarily as a design specific to the MoonBEAM mission. MoonBEAM was chosen as an example simply because the mission was not possible with conventional technology in a 6U volume. The design study results are more like the show cars of the 1960's, generating interest and showing what may be possible in the future. Many aspects of this proposed CubeSat design may find their way into small satellites of the near future.

This TM is an update to the previously issued TM[1] that summarized the results from the phase 1 study. While the reader is encouraged to examine that publication for background information, much of the text has been repeated here to allow this TM to stand alone. This TM includes a brief introduction to the science mission, a description of the requirements, spacecraft design, and mission concept, and details the design of the various subsystems. It also summarizes the 'strongback' design and the gas generator concept, though subsystem analysis was not completed for those two alternatives. Overall, this TM shows the benefits of bringing AM, green propellants, and other technologies to potential CubeSat missions. With the help of AM, CubeSats can be used for missions beyond low Earth orbit (LEO), missions that have large ΔV requirements such as lunar and interplanetary. Additive manufacturing may indeed be a game changer for CubeSat design.

2. SCIENCE MISSION SUMMARY

The primary purpose of the MoonBEAM mission is the detection of gamma-ray bursts. These highly energetic events, occurring on a daily basis and distributed throughout the sky, can be triggered by the collapse of a massive star or the merger of two compact objects. Since the mission orbit maintains a distance from Earth of 60,000 km or more, the brief time delay between MoonBEAM detecting the event and instruments near Earth detecting the same event allows for more accurate location precision of the gamma-ray source.

In addition to producing a burst of intense gamma rays, the merger of two compact objects is also expected to produce gravitation waves, which was directly detected for the first time in 2015. A gamma-ray counterpart is expected for certain types of gravitational events, and MoonBEAM will improve the gamma-ray sky coverage and increase the joint detection potential. MoonBEAM will provide an additional baseline for better localization if the gravitational wave-related gamma-ray burst is also detected by an instrument in orbit near Earth. The refined location will aid other telescopes in their followup observations searching for the electromagnetic counterpart of gravitational waves.

3. SCIENCE INSTRUMENTS AND OPERATION

The description of the instruments presented below is identical to that from the phase 1 study and is not a detailed description of all components of the instruments and their operation. The data included here are the data that were considered relevant to the spacecraft design, such as instrument operating power, thermal requirements, mass, and similar parameters.

Table 2 provides a top-level summary of the instrument data. The spacecraft has a minimum of four detectors facing in orthogonal directions, each with a minimum area of 126 cm^2 and thickness of 1.5 cm. Note that the thickness is an assumed value and is intended to account for back-end electronics. The detectors are made of silicone photomultiplier (SiPM) groups, 14.2×14.2 mm squares, which themselves are composed of four (2×2 array) SiPM sensors, 6×6 mm each. The fill factor assumed for laying out the instruments was 75%. Allocated power for the instrument suite was 1 W per detector, for a total of 4 or 5 W for four- or five-detector configurations, respectively. While four detectors were allocated in the final master equipment list (MEL) for the study, the configuration can accommodate five detectors, provided the mass limit of 14 kg is not exceeded. In addition, the instruments require electronics cards in the spacecraft. These elements are considered part of the avionics subsystem and are detailed in section 7.5.

Table 2. Basic science instrument dimensions and requirements.

Instrument	Dimensions or Area	Mass (kg)	Power (W)	Temperature (ºC)
Scintillation crystal (each, four minimum, five desired)	126 cm^2	0.5	<1	–40 to 85
SiPM group (made from four SiPMs)	14.2×14.2 mm	N/A	N/A	N/A
SiPM (single element)	6×6 mm	N/A	N/A	N/A

Tables 3 and 4 provide a top-level summary of the instrument data. While a minimum of four detectors was required, a fifth detector was desired. Both versions are tabulated below, though the current design uses the minimum number of four detectors. Electronics boards for the instruments are not included in the MELs here but are part of the avionics subsystem detailed in section 7.5.

Table 3. Science instrument MEL for the four-detector (baseline) configuration.

Component	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
Scintillators	4	0.5	2.02	20	2.42
SiPM and board assembly	4	0.07	0.29	20	0.35
Total			2.31	20	2.77

Table 4. Science instrument MEL for the five-detector configuration.

Component	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
Scintillators	5	0.5	2.52	20	3.02
SiPM and board assembly	5	0.07	0.36	20	0.44
Total			2.88	20	3.46

4. STUDY APPROACH

The goal of phase 2 of the MoonBEAM AM study was to incorporate the updated propulsion elements into the phase 1 design. Having been provided mass and volume constraints and other performance targets for the propulsion system in the phase 1 study in the spring, engineers attempted to show that these elements could be created using AM. Even if some goals could not be met, the results would be a redesigned propulsion system that would be incorporated into an updated spacecraft design during phase 2. This would support the overall goal of demonstrating the benefits of AM to CubeSat missions.

Because CubeSats are, in general, considerably smaller than conventional spacecraft, they are very limited in volume. The subsystem components for a conventional spacecraft easily fit within the allowed volume, with a generous portion of the overall configuration being empty space. For CubeSats, subsystem components usually leave little empty space in the configuration, and laying out the various elements in such a limited volume can be challenging. This volume constraint substantially impacts two areas more than others—thermal management and propulsion. Thermal issues arise due to the dense packaging of the various spacecraft elements in such a small volume, while propulsion issues arise from having such a limited volume in which to store propellant. Since CubeSats are made of 'cubes' by design, using spherical propellant tanks in cubical enclosures wastes space, as it is not easy to place other subsystem elements within this available space around the propellant tank or tanks.

This is where AM can provide a valuable benefit. By printing a propellant tank or propulsion system, and possibly having it be part of the spacecraft structure, propellant volume can be maximized at the same time that wasted space around the tank is minimized or eliminated. Tanks can be of nearly any shape, with additional structure added where needed and reduced otherwise. Additive manufacturing can allow propellant passages, management devices, and perhaps even thrusters themselves to all be part of a single propulsion system element, designed to optimize the use of space, the placement of the thrusters, and the management of thermal soakback from the thrusters into the spacecraft.

The study team chose the MoonBEAM science mission to demonstrate the benefits of AM. A 2016 study for this same mission, but with conventional CubeSat components and design, showed the mission to be infeasible, with insufficient volume to contain the rather large propellant amount required. In fact, the design exceeded the target 6U footprint by over 40%. To see the benefits of AM, the current design team decided to carry forward the 6U volume constraint, the rather high ΔV requirements (for a CubeSat), and the generous science payload volume. The work was completed in two phases, with phase 1 of the study providing the propulsion system performance parameters necessary for mission success. These targets were then used by AM engineers to design a system and demonstrate that AM could create such a system. Phase 2 of the study then incorporated the revised propulsion system design. The results of the phase 2 study, detailed below, show that AM could enable the MoonBEAM mission, providing a platform with a large ΔV capability for a CubeSat. The results also clearly show the goals that must be achieved in AM to make missions such as MoonBEAM successful, and to position CubeSats to take part in missions beyond LEO. While the target phase 1 performance goals were not achieved, the resulting spacecraft performance in phase 2 is still considered a successful demonstration of AM capabilities, as its high ΔV capability can enable many low-cost science missions beyond LEO.

5. MISSION AND SPACECRAFT REQUIREMENTS

As stated previously, the MoonBEAM mission was selected for this design study for several reasons. First, a study from the previous year had shown the mission to be infeasible with conventional CubeSat technology and propulsion. Second, since our office had performed that study, the mission and trajectory were well defined, meaning that our limited resources could be used for other aspects of the study rather than mission analysis and requirements development. And third, the mission would push the limits of 6U CubeSat design, a desired outcome of this study, and hopefully show that a previously infeasible mission can be made feasible with the help of AM. With those reasons in mind, the design team was tasked with designing a spacecraft that would enable the MoonBEAM mission, and to provide a path forward and goals that the AM propulsion and structural elements would have to meet.

Table 5 lists the requirements that guided the design study. Regarding science, the trajectory requirements were few. To meet timing requirements, a position knowledge of 100 km or better is necessary, easily achieved with tracking stations. In addition, the science team wished to maintain a minimum Earth and Moon spacecraft distance of 60,000 km. No other criteria for the orbit were specified. Given the desire to rideshare on one of the Space Launch System (SLS) exploration missions (EMs), the design team carried over the previous study baseline orbit of LDRO, with a rideshare on SLS EM-1. As stated previously, while other orbits are possible, LDRO was carried over precisely because of its high ΔV requirement. (Though the timeline for EM-1 is too soon to be realistic for this mission, it provided a basis for the design and allowed the team to directly compare results with the previous MoonBEAM study.)

Table 5. Mission and spacecraft requirements for MoonBEAM.

Property	Value
Mission duration	1 year required (multiple years better)
Mission class	Risk class D
Orbit	LDRO
Launch vehicle	SLS EM-1 rideshare
Max wet mass	14 kg (12 kg desired)
Instrument pointing requirement	None (full sky)
Science data	250 MB/day of continuous data, downloaded within days
Data storage	250 MB/day $\times$ days before download + margin
Event data transfer	100 kb per trigger, 10 triggers per day to ground within 60 minutes

Table 5. Mission and spacecraft requirements for MoonBEAM (Continued).

Property	Value
Pointing control (driven by spacecraft, not science)	Solar 4–10 deg.; antenna 20 deg; none on instrument
Pointing knowledge	0.1 deg (6 arcmin)
Location accuracy	100 km
Detector	Scintillation crystal with max array of SiPM sensors
Scintillation crystal	126 cm^2 (19.6 in^2) surface area, ~1.5 cm thick
SiPM sensor	6 mm $\times$ 6 mm MicroFC-60035-SMT (36 mm^2); 2$\times$2 array 14.2$\times$14.2 mm^2 ArrayC-60035-4P-BG
Number. of SiPMs	Cover scintillator crystal surface area, 75% fill factor using 2$\times$2 arrays
Number of detectors	4 minimum
Operating temperature	–40 to 85 °C
Power	<1 W per detector

6. MISSION ANALYSIS

Carried over for this study was the assumption that MoonBEAM would be a rideshare on SLS EM-1, which provides a reference trajectory for the mission, including mass limitations and transfer trajectory requirements necessary to place the spacecraft into its operational orbit, which was assumed to be LDRO. LDRO is very stable, requiring little or no orbit maintenance. However, getting to LDRO is challenging[2] from a CubeSat perspective because of a rather large ΔV budget necessary for the transfer. This was actually the reason for selecting this option, as it provided a challenging goal for the design team to meet. However, the final phase 2 design proved to have insufficient ΔV capability to achieve LDRO, so alternative orbits are recommended. The alternative orbits are described near the end of section 7.

A plot of LDRO is shown in figure 1. This is a highly stable orbit, requiring a minimum of stationkeeping to maintain for many years to decades. It also keeps the spacecraft outside the radiation belts and provides a stable thermal environment, factors making it ideal for some science missions. While not required for MoonBEAM, it provides a demonstration for the benefits of AM, as many science missions may be interested in a similar orbit but cannot attain it due to the propulsive limitations of current CubeSat designs.

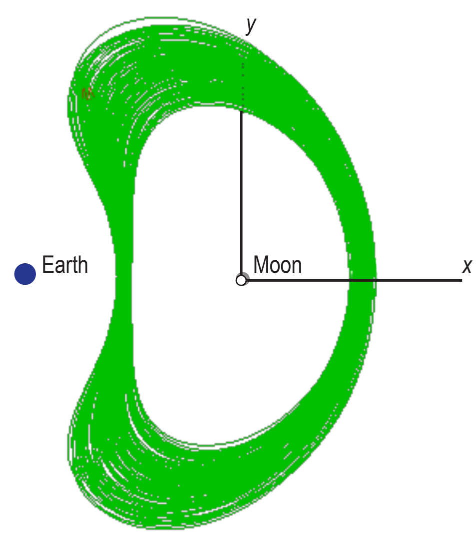

Figure 1. Plot of the lunar distant retrograde orbit in the rotating frame. Total ΔV = 368.5 m/s.

Other orbits considered during the original MoonBEAM study of 2016 include Sun-Earth L2 halos, Earth-trailing drift-away trajectories, and others. The analysis will not be repeated here, but all have lower ΔV requirements than the selected LDRO mission. While the mission could have been achieved with one of these orbits, they would not tax the design, which was the point of this study. In addition, the distance from Earth would have affected communications, though it is not known whether or not that would have been a design hurdle.

The transfer trajectory used in the analysis to determine the maneuver budget is shown in figure 2. After being released from SLS, the spacecraft must despin and then prepare for a midcourse correction maneuver. The major contributors to the maneuver budget are the lunar flyby and the post flyby midcourse, both of which contribute the bulk of the ΔV budget. All maneuvers are shown in table 6. Note that no propellant is allocated for the despin, as this is accomplished using other guidance, navigation, and control (GN&C) components as explained in section 7.3. Also shown in table 6 is the momentum and orbit maintenance values, both of which are zero for this mission, since momentum unloading is accomplished through the use of novel electric thrusters and orbit maintenance is not required. A modest disposal maneuver is included at the end of the mission, pushing the spacecraft out of LDRO and leading to lunar impact after several months. As LDRO is such a stable orbit, disposal may not be necessary, or even advisable. Adding a 10% margin to the total ΔV budget results in a maneuver total of 368.5 m/s. With no orbit maintenance, the spacecraft never exceeds a distance of 500,000 km from Earth, and is never nearer than 60,000 km. Since LDRO is actually a range of values, the mininum distance from Earth can be increased and the maximum decreased by selecting a different LDRO.

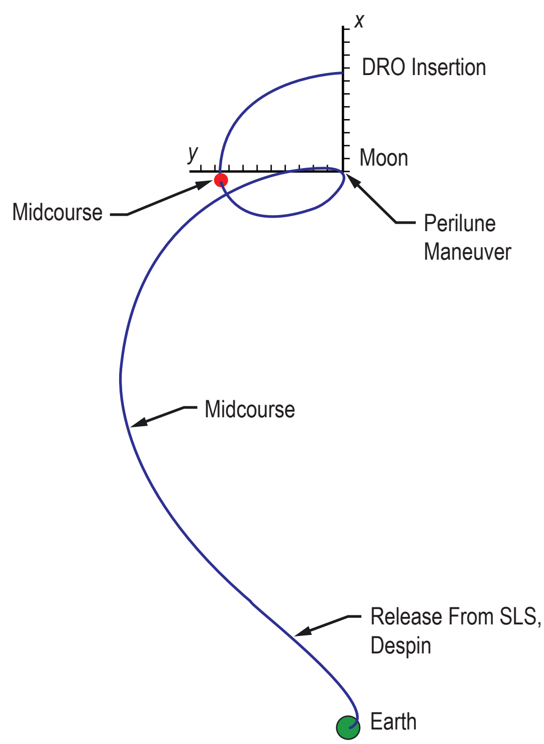

Figure 2. Transfer trajectory to LDRO. Trajectory work was performed on a previous study and is representative of a possible MoonBEAM transfer trajectory.

Table 6. Maneuver (ΔV) budget for the MoonBEAM mission.

Maneuver/Category	Value
Despin	–
Midcourse	5 m/s
Lunar flyby	162 m/s
Midcourse 2	155 m/s
Insertion	3 m/s
Disposal	10 m/s
Maneuver total	335 m/s
Momentum unloading	–
Orbit maintenance	–
Total without margin	335 m/s
Margin	10%
Total ΔV	368.5 m/s
Max distance after 3 years	500,000 km

The SLS EM-1 baseline mission will be using the Planetary Systems Corporation CubeSat dispenser. This dispenser and its location in SLS is shown in figure 3. Sitting between the interim cryogenic propulsion system and the multipurpose crew vehicle (MPCV), the MPCV stage adapter (MSA) contains the secondary payload deployment system (SPDS). As shown in figure 3, the SPDS contains eleven 6U CubeSat deployers, mounted at an angle of 56 degrees relative to the horizontal. CubeSats are released individually at regular intervals, so the release point assumed in the outbound trajectory for our analysis is only an approximation. The 6U dispenser will be certified to accommodate 14 kg maximum payload, with a rail and spring system ejecting the payload. Given the mounting direction, dispenser mechanism, and rotational motion of SLS during payload release, the team assumed a worst-case rotational value of 10 degrees per second for all axes. Though these values will almost certainly be reduced in the future, the GN&C design for MoonBEAM does provide the capability for nulling these rates.

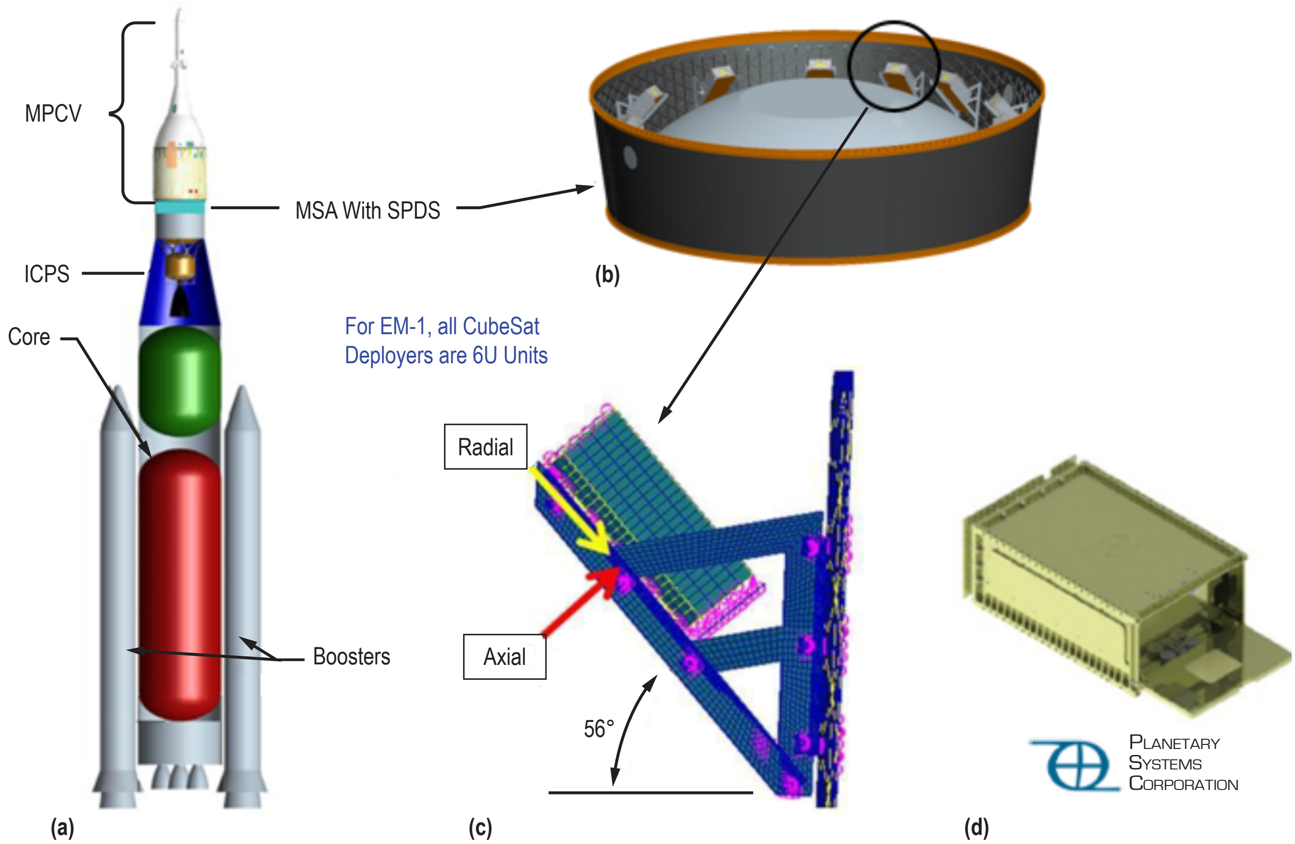

Figure 3. Rideshare accommodations for MoonBEAM: (a) SLS EM-1 configuration, (b) the SPDS showing placement of the CubeSat dispensers, (c) the mounting angle for the dispenser, and (d) the Planetary Systems Corporation dispenser that is being used on the SLS EM-1.

7. SPACECRAFT DESIGN

Details for the phase 2 spacecraft design are presented below, and include descriptions of the overall configuration as well as subsystems. Phase 1 design results can be found in the previous TM.[1] Each subsystem section contains a description of ground rules and assumptions (GR&As) used to guide the design, approach, and methodology, and a MEL showing the components selected for this conceptual design (except for configuration).

Several subsystems use typical components, but propulsion, GN&C, and structures were allowed to use new technologies in order to explore the benefits of AM to the overall design.

7.1 Configuration

The MoonBEAM spacecraft is baselined as a 6U volume CubeSat. Like all CubeSat spacecraft, it is space limited by the chosen 6U payload dispenser volume. The basic approach is to start with the allowable volume and determine the volume of all the required components for a fitment and placement analysis. The primary instruments were the gamma-ray burst monitor detectors. They were placed on five sides to give the maximum viewing capability. Their placement also dictated the amount of space left over for the other spacecraft subsystems. The propulsion subsystem was also a major driver in the design due to more volume requirements than any other subsystem. The propulsion tank size and shape were optimized to provide adequate space to allow for the spacecraft avionics cards. The other less volume-critical system components were placed around the spacecraft in the unused spaces. The allowable volume was almost fully utilized by the various systems. Four solar arrays were added as in a typical 6U CubeSat design.

Basic configuration and dimensions are shown in figure 4. The total size of the spacecraft is 365 mm long by 239 mm wide by 113 mm tall. This size allows the spacecraft to fit within the 6U CubeSat deployer.

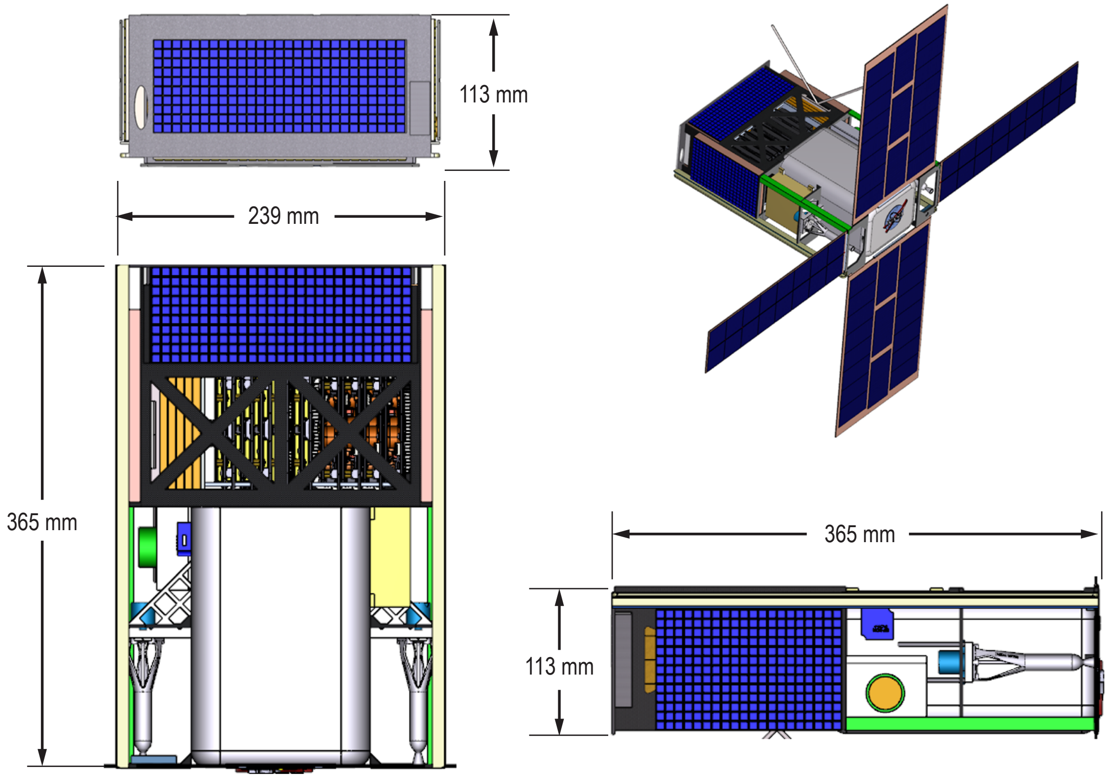

Figure 4. The MoonBEAM spacecraft layout showing the stowed and deployed configurations.

7.2 Mass Properties

The MoonBEAM mass rollup is listed in table 7, with the science instrument values shown being for four detectors (the minimum requirement). A fifth detector may be possible through mass savings elsewhere. Using the AIAA standards[3] for mass contingency, the total predicted mass of 14.19 kg exceeds the 14 kg limit of the deployer. However, as mentioned previously, 0.19 kg of propellant can be offloaded and still provide a substantial maneuver capability of 308 m/s, as shown in table 1. Most components used in the design are at a high TRL with flight heritage, justifying the low margins. Propellant mass was determined with a ΔV budget that includes a 10% margin, and also accounts for residuals. The spacecraft was designed with a 20% power margin, a value that is slightly lower than normally desired at this level of design analysis. Perhaps additional mass can be saved from other subsystems, such as structures, that could be used to increase the power margin. More detailed discussions of propulsion and power design and recommendations are given in sections 7.4 and 7.7, respectively.

Table 7. MEL with full propellant load. Note that 0.19 kg of propellant can be offloaded to meet the 14-kg mass limit and still maintain substantial maneuver capability.

Subsystem	Basic Mass (kg)	Growth (%)	Predicted Mass (kg)
Structures	0.97	10	1.06
Thermal	0.19	24	0.23
Power	1.36	20	1.63
Avionics	1.25	17	1.47
GN&C	0.73	10	0.8
Science instrument	2.31	20	2.77
Propulsion	3.44	15	3.97
Total dry mass	10.24	16	11.93
Propellant	–	–	2.26
Total mass	12.5	13	14.19

The design illustrates the possibilities if the performance targets set forth by the propulsion and structures analyses can be met through AM. It should also be noted that, in the previous MoonBEAM study, the total mass was slightly over 14 kg while the total volume far exceeded the allowable value by over 40%. Thus, the savings through AM are mainly in packaging efficiency for relatively large propellant requirements, though some mass can perhaps be saved through optimized design of structures.

7.3 Guidance, Navigation, and Control

The spacecraft attitude determination and control system (ADCS) architecture design approach consists of primary attitude control actuator sizing, selection of additional actuators to be used for momentum management (if required), and sensor selection. Primary attitude control actuator sizing is based on multiple factors, including the magnitude of environmental disturbances in the spacecraft orbit, required slew maneuvers, and available volume. Because the momentum accumulation due to environmental disturbance torques and slew maneuvers is a strong function of spacecraft geometry and moments of inertia, an estimate of these parameters is required for the ADCS analysis and sizing process. As a first-order approximation, the MoonBEAM spacecraft is modeled as a standard ($10\times20\times30$ cm), 6U Cubesat with a 14-kg mass uniformly distributed throughout the volume. Moments of inertia are estimated for this 'stowed' configuration, as well as for one in which all solar panels are fully deployed.

General GR&As used in the ADCS design are listed in table 8. Three-axis pointing control is required for Sun inertial pointing during nominal operations, driven by the need for continuous solar array power generation when not in eclipse. While no slew maneuvers are required for science operations or thermal management, some may be needed in order to periodically reorient the communications antenna for ground downlink. Because the exact antenna location has not yet been defined, no specific requirement is included at this time. A 180° slew over a quarter of the orbital period about the boresight axis is included in the analysis as a placeholder.

Table 8. GN&C ground rules and assumptions.

Property	Value
Location accuracy	100 km
Pointing control	Three-axis stabilized, nominal attitude is solar arrays toward Sun
Solar	4–10 deg
Communication antenna	20 deg, point antenna toward Earth when possible
Slew requirements	None for science; driven by antenna pointing requirement (TBR)
Pointing knowledge	0.1 deg (6 arcmin)
Thrust alignment error	0.25 deg from cg
Tipoff rate damping/despin	10 deg/s/axis
Reliability	Class D, single reaction wheel per axis

An estimate of the tipoff rate (due to momentum imparted by the launch vehicle and deployer during orbital insertion) of 10º/s/axis is used to inform actuator selection for initial detumble and stabilization prior to solar array deployment and mission operations. This value assumes deployment from the secondary payload ring on SLS EM-1 or EM-2. It should be noted that this tipoff estimate is considered to be highly conservative, and is expected to become more refined as more data are acquired by the deployer manufacturer post EM-1.

Several actuator options were considered in a trade study to determine the best method of tipoff rate damping, including several sizes of reaction wheels and a thruster reaction control system (RCS) using cold gas, warm gas, or electric propulsion. Programmatic constraints require systems using warm gas to make ground contact prior to use, making this method unfeasible for detumble. A cold gas system was not desirable due to the low efficiency and relatively large volume. A µCAT[4] electric propulsion system was carefully considered for detumble maneuver implementation, but it was determined that this would require a significant amount of onboard power prior to solar array deployment (table 9).

Table 9. µCAT detumble maneuver calculations.

Axis	Momentum (Nms)	Moment Arm (m)	Required Total Impulse (Ns)	Maneuver Time (hr)	Power Required (A-Hr)*	
					15 V	25 V
Roll	0.0099	0.12	0.082	38.157	25.438	15.263
Pitch	0.0204	0.12	0.17	78.558	52.372	31.423
Yaw	0.0265	0.12	0.221	102.125	68.083	40.850
Totals	0.057	–	0.473	–	145.893	87.536

* 50 Hz operation requiring 10 W per thruster module; shown for 15- and 25-V controller inputs.

Ultimately, it was recommended that Sinclair Interplanetary 0.03 Nms reaction wheels be used directly for tipoff rate damping, as well as general attitude control, even though the wheel momentum margin is lower than desired at this phase in the mission development (47% in pitch and 13% in yaw; desired is 100%). This is not viewed as a risk, however, since this margin is expected to increase to acceptable levels as the highly conservative tipoff rate estimate is refined (which may be as low as 1° to 2°/s/axis). Because the momentum accumulation due to slew maneuvers and environmental disturbances in LDRO was determined to be relatively low compared to that due to tipoff rates, the 0.03 Nms wheels meet the momentum capacity requirement for these additional operations with ample margin.

While not selected to carry out the initial tipoff rate damping maneuver, the μCAT RCS was determined to be a feasible solution for desaturation of the Sinclair 0.03 Nms reaction wheels selected for the detumble, general attitude control, and slew. The μCAT uses an electric arc discharge to ablate a nickel cathode (see fig. 5), after which the ablated material is accelerated away through the use of a magnetic field. The advantage is that the thrusters do not require additional propellant tanks, pressurant tanks, or plumbing. Separate driver boards are required to produce the necessary current pulse and can be placed in the card stack while the thrusters are positioned as needed on the spacecraft body.

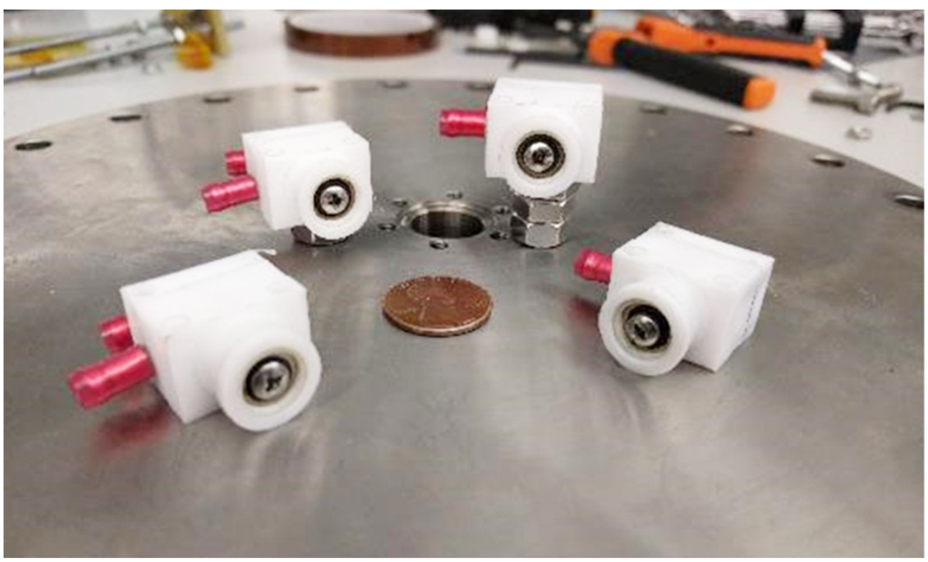

Figure 5. Microcathode arc thruster.

Several attitude determination sensors were selected based on preliminary requirements for pointing accuracy and pointing knowledge. The specific units selected are based on those that will be used on the 12U iodine satellite (iSat) spacecraft. iSat is currently under development and has similar requirements. The inertial measurement unit (IMU) is the M-G362PDC1 model from Epson. A star tracker manufactured by Sinclair Interplanetary provides high-accuracy (5 arcs cross-boresight root mean square) attitude measurements, while a CubeSat Sun sensor from NewSpace Systems allows Sun tracking for Sun inertial pointing. The current location accuracy requirement is lenient enough that it may be met with knowledge of ground station tracking, and a global positioning system receiver was not deemed necessary at this time. The GN&C MEL for the current phase of the MoonBEAM study is shown in table 10.

Table 10. GN&C MEL.

Component	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
Reaction wheel	3	0.185	0.56	10	0.61
Star tracker	1	0.1585	0.16	10	0.17
IMU	1	0.007	0.01	10	0.01
Sun sensor	1	0.005	0.01	10	0.01
Total			0.73	10	0.8

7.4 Propulsion

This update to the propulsion system design was performed several months after the original phase 1 evaluation. Remaining funding from the TE award and availability of the Advanced Concepts team allowed for this update. This provided the opportunity to further refine the design concept and explore those future work items not addressed in the original phase 1 study. In addition to the baseline concept shown in figure 6, two additional concepts were explored—the gas generator as part of the phase 2 configuration and the strongback design as part of a new configuration. Recall that the early 2017 study concluded that a mission using a 6U configuration was feasible with AF-M315E monopropellant and AM. The study focused on the technology enablers rather than the science mission. The evolved system's wet mass was estimated to be 5.7 kg, carrying a 20% performance margin and fitting within the allocated volume.

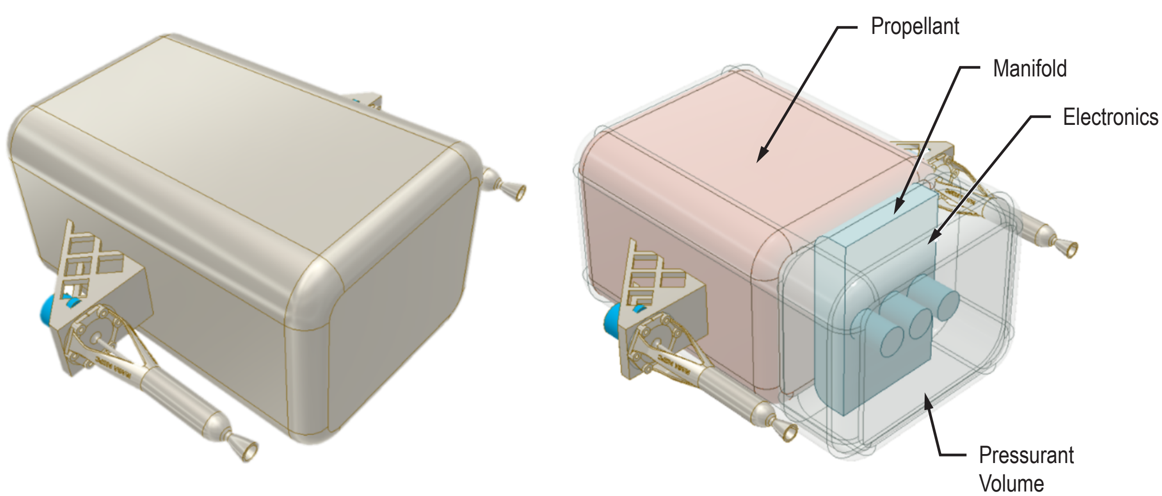

Figure 6. Phase II updated MoonBEAM propulsion system major elements (controller cards not shown).

Two advanced high-performance and low-toxicity 'green' monopropellants were considered for the trade: AF-M315E and LMP-103S. Both propellants exhibit greater density specific impulse (I_{sp}), improved storability, and stability than hydrazine, and can be handled safely without self-contained atmospheric protective ensemble facilities. AF-M315E propellant was selected for the MoonBEAM propulsion system concept. It is a member of a series of storable hydroxylammonium nitrate and hydroxyethylhydrazine blend monopropellants design to possess a significantly lower vapor toxicity than hydrazine, and have a greatly improved volumetric and I_{sp}. The formulation was engineered by the U.S. Air Force Research Laboratory to replace spacecraft hydrazine monopropellants. The same propulsion system GR&As were used in the phase 2 update (listed in table 11).

Table 11. Propulsion system GR&As.

Property	Value
Total impulse	Minimum of 4,650 N-s
Mass allocation	No greater than 5 kg
Thruster impulse bit	No greater than 125 mN-s
Propulsive capability	Three-axis attitude control
Modular propulsion system	Designed and packaged as an integrated, self-contained propulsion module
Initial ullage volume (percentage)	2.5% to 3.5%
Unusable propellant (percentage)	Gallery and lattice structure PMD: 10%
Protection to inadvertent fluid leakage	Two fault tolerance
Fabrication techniques available	All traditional methods, AD of Ti-6-4 and Inconel 718, AD of refractory metals, joining refractory metals to Ti and Inconel
Wetted materials compatibility	Compatible with N_2H_4, AF-M315E, and LMP-103S
Spacecraft regulated power utilization	5 VDC (±5%)
Spacecraft unregulated power utilization	9–12.6 VDC
Thruster firing duration limit	15 minutes firing and 5 minutes off
Pressurant gas	Nonreactive
Storage life (with servicing)	Minimum of 24 months in storage without degradation of system performance
Storage life (no servicing)	Minimum of 12 months in storage without servicing and without degradation of system performance
Thermal operability	Between –10 °C to 50 °C without degradation of system performance
Thermal survivability	Between –34 °C to 60 °C without degradation of system performance
Mission survivability	18 months in a space environment without degradation of system performance
Design factors	Proof =1.5 × MEOP; burst = 2 × MEOP

The dry mass of the propulsion system was driven by the configuration and the associated pressures. The simplest design configuration would be a 'blowdown' configuration. In order to expel the propellant and not drop below the thruster inlet requirements, and remain within the volumetric requirements, the tank would need to be pressured to just above 900 psia. The thrusters and valves could never operate at such inlet pressures. For instance, the 1-N thruster microvalve developed at MSFC for green monopropellant applications is only rated for 400 psia. That analysis does not account for any variation in improvement from the thruster as a result of the higher inlet pressure conditions. The operational conditions were outside of the desired range, therefore, the concept was rejected. It was most appropriate to explore a regulated pressure system. Though it is more complex than a blowdown system, the inlet conditions for the thrusters and valves are regulated to a manageable range. However, it would also require the propellant tank to be pressurized to nearly 1,000 psia. This pressure would be regulated from a high-pressure tank volume into a lower pressure propellant tank. A graphic of this regulated gas system is detailed in figure 7.

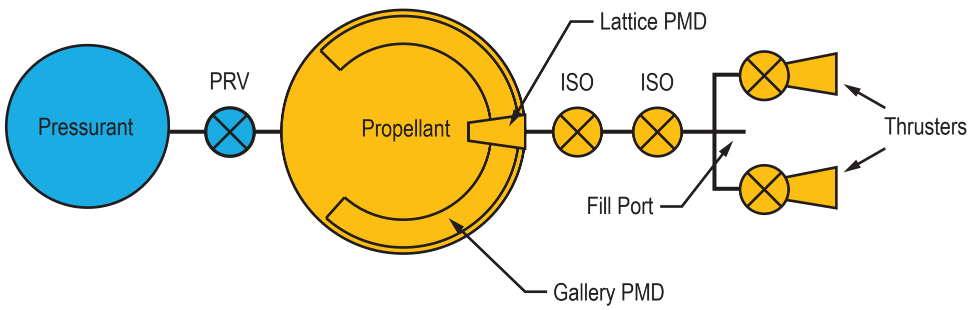

Figure 7. MoonBEAM regulated gas propulsion system.

A standard design factor and credible failure tolerance paradigm was applied for this project. Therefore, the pressure system must be designed to tolerate one credible failure, such as a valve or seal. Applying these two paradigms meant that the propulsion system must be designed to safely tolerate 1.5 times the maximum expected operating pressure of the pressurant tank assuming the failure of the pressure-regulating valve (PRV) between the pressurant tank and the propellant tank. Therefore, the propellant tank must be designed to survive 1,500 psia. This drove the propulsion module walls to be thicker than originally expected. The MEL for the propulsion system is shown in table 12.

Table 12. Propulsion system MEL.

Component	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
Tank structure (printed Ti)	1	2.7	2.7	15	3.11
1N AF-M315E	2	0.08	0.16	25	0.2
Thruster/valve driver board	1	0.05	0.05	25	0.06
μCAT thruster module	4	0.02	0.08	30	0.01
μCAT controller module	1	0.05	0.05	30	0.07
Valves	5	0.04	0.2	–	0.2
Manifold	1	0.2	0.2	15	0.23
Total	–	–	3.44	15	3.88
AF-M315E propellant					2.26

7.4.1 Printing a Propellant Tank

A representative CubeSat propulsion system was designed and built as part of this TE award. The module was designed in a 1U form factor so that it too may be used as an end-item test article itself. It was designed to fit one 1-N thruster and use the very same valves considered in the MoonBEAM propulsion system. There are no differences in the intended operation between the two designs. An image of one of the printed modules is shown in figure 8, and a separate TM is being drafted at the time this TM was being published.

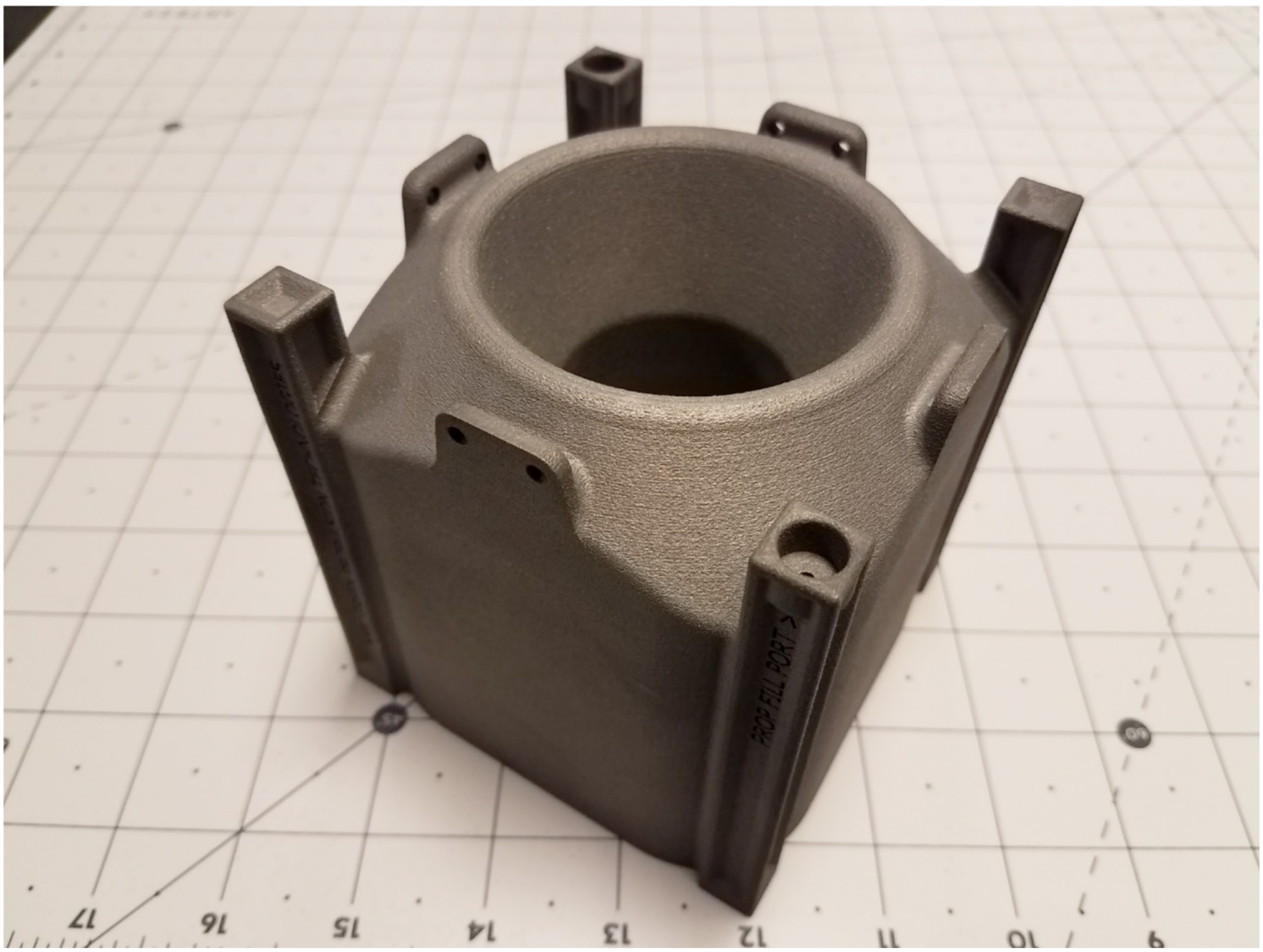

Figure 8. As-printed 1U CubeSat propulsion module.

Three titanium (Ti) modules were printed. The first will be sectioned and microtensile specimen taken from the walls. The other two modules will be postmachined. One of them will undergo hydroproof testing, cyclic testing, and then burst testing. The last module will have valves and a thruster integrated to it, and hot-fire tested. If successful, this will advance the TRL to 4.

7.4.2 Printed Thruster

The printed proof-of-concept 1-N thruster (fig. 9) activity was nearing hot-fire testing at the time this TM was drafted. Two thrusters were completed. Parts for four more thrusters are in-hand and are being held at their currently assembly state awaiting test data to inform adjustments in configuration. A separate TM and Technical Publication were written for the 1-N thruster development and testing.

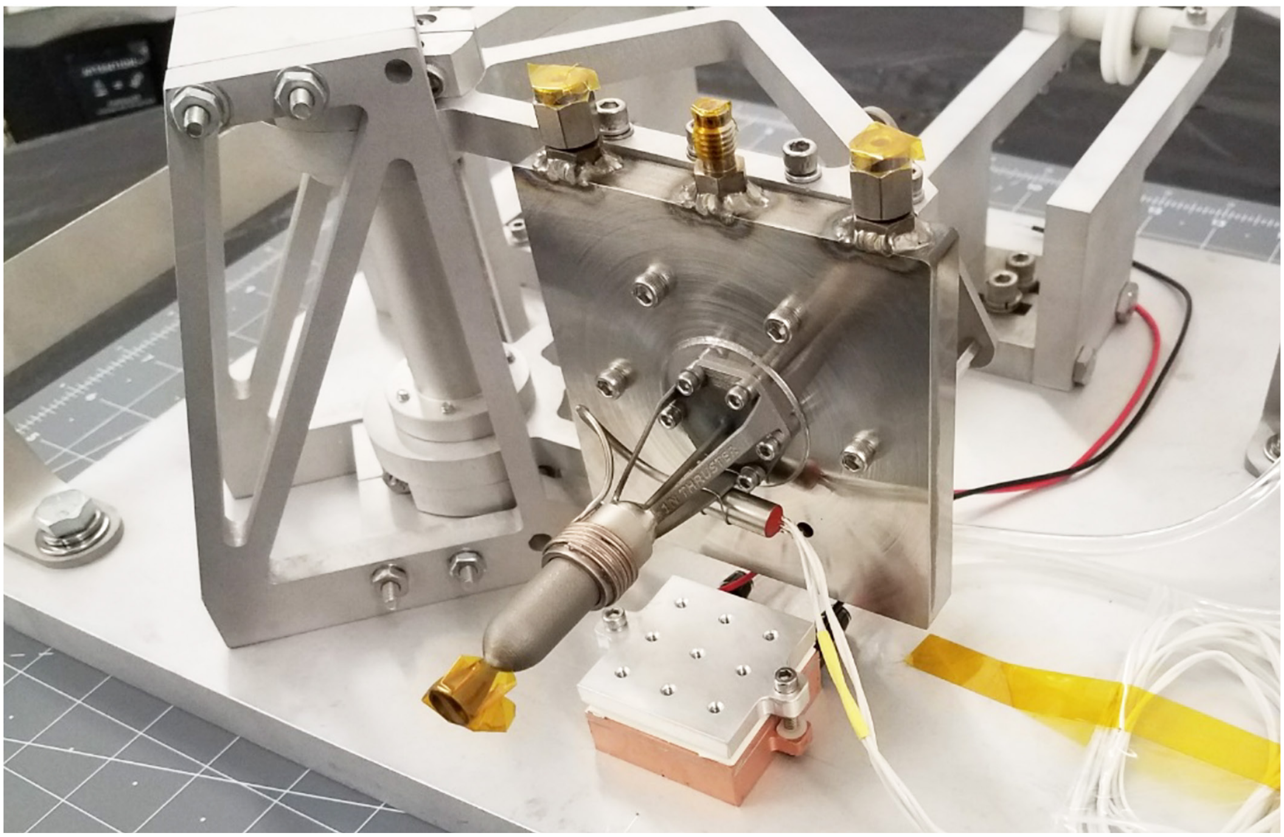

Figure 9. 1-N thruster prepared for testing.

A 1-D thermal analysis was completed for the 1-N thrusters. Concerns stemming from the thruster-generated heat load to the spacecraft at the mounting interface (commonly referred to as heat soakback) entailed transient analyses of the preheat, firing, and cooldown operational phases. In addition to prediction of interface conditions important to the spacecraft, the analyses provides temperature distributions that are beneficial to the propulsion subsystem development.

Figure 10 illustrates the temperature transient averaged across five distinct planes for the two cases examined. For the baseline case, the following significant assumptions are included: (1) AF-M315E propellant, (2) heater power is 20 W, (3) heater shuts off at 673 K, and (4) the thruster fires for 250 s. Relevant results include the following: Heater must be on for 700 s to achieve 673 K and the maximum temperature at mounting interface (i.e., plane A) is 394 K.

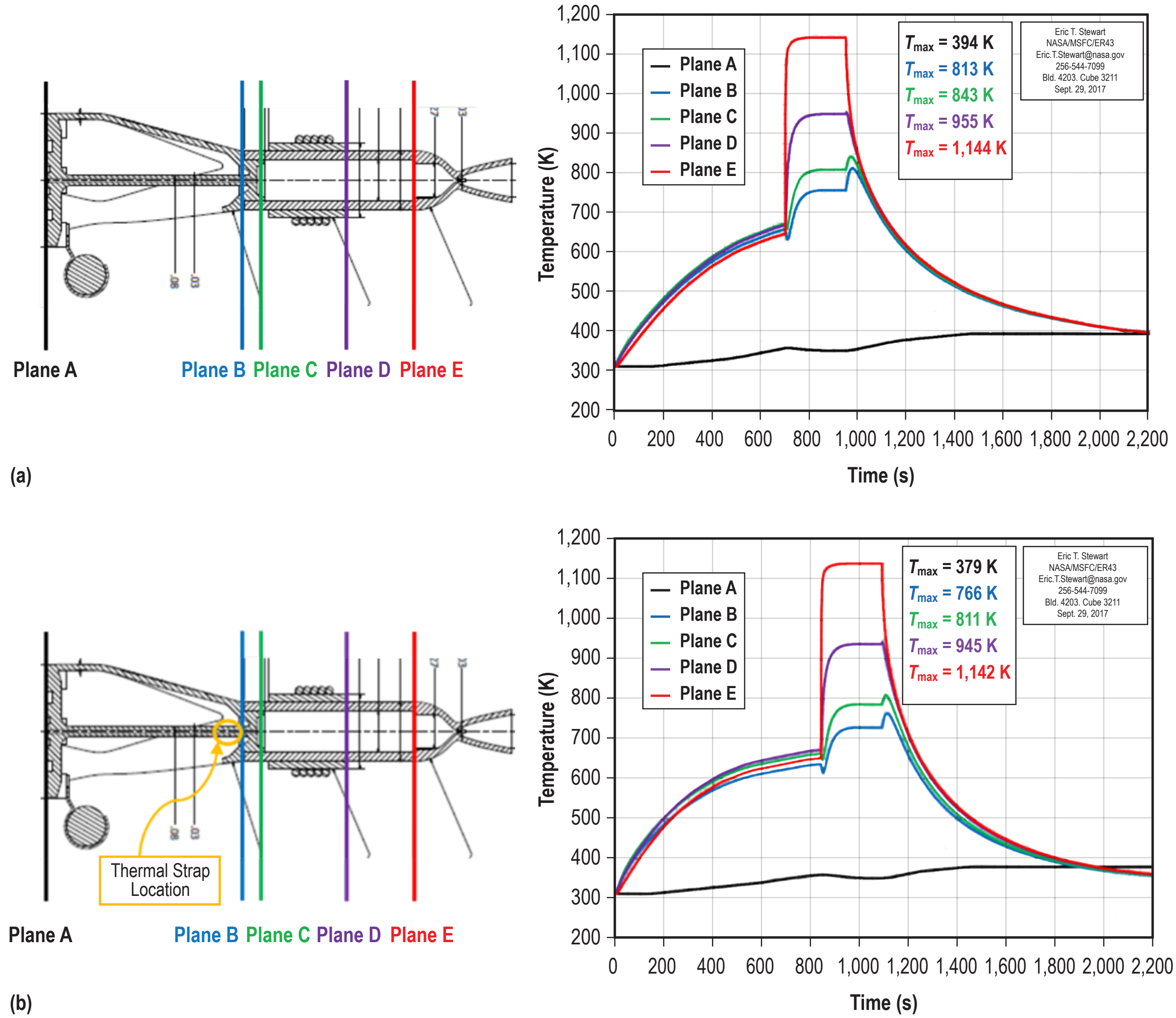

Figure 10. Transient average temperatures across identified cross-sectional plates: (a) Baseline case and (b) thermal strap case.

In addition to the temperature transient at the mounting interface (i.e., plane A in fig. 10), the spacecraft must be designed to handle the heat transfer at this interface, whose transient is illustrated in figure 11. Over the 2,200-s interval for the baseline case, 4,211 J of energy are transferred to the spacecraft with an average heat transfer rate of 1.9 W and a peak heat transfer rate of 5.7 W.

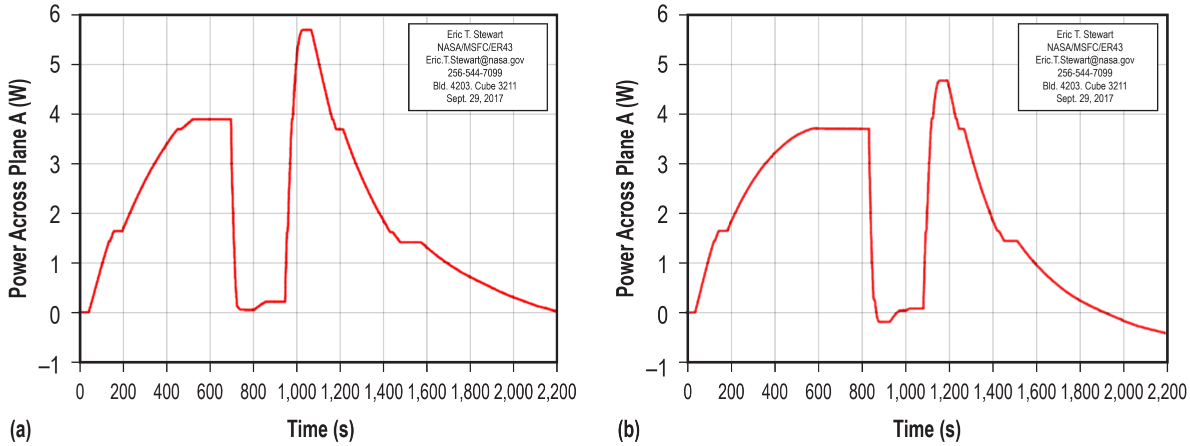

Figure 11. Transient heat transfer rate across the thruster mounting interface: (a) Baseline case and (b) thermal strap case.

An additional case was modeled, which is referred to as the 'thermal strap' case in both figures 10 and 11. The motivation of adding thermal straps arises from a desire by the thruster designer to reduce temperatures at the inlet to the injector face (i.e., plane B in fig. 10). Three thermal straps of 0.05 inch in diameter and 3 inches in length where added to the model, resulting in a 15 and 47 K reduction in the maximum temperatures at the mounting interface and injector inlet, respectively. However, to preheat the thruster to 673 K in similar time as the baseline case, the heater power was raised to 25 W, which still took an additional 140 s of heater-on time. As far as heat transfer across the mounting interface during the 2,200-s interval, 3,705 J of energy are transferred to the spacecraft with an average heat transfer rate of 1.7 W and a peak heat transfer rate of 4.7 W.

Neither case examined appears to add significant risk to the thermal design of the thruster nor its thermal integration into the spacecraft. The maximum interface temperatures and interface heat loads appear to be manageable with anticipated materials, masses, and thermal management strategies. As the designs mature for both thruster and spacecraft, the thruster thermal model should increase in fidelity to minimize any potential risks that remain from development and integration of such a high-temperature component.

7.4.3 Propulsion System Design Conclusions

As part of these studies, the propulsion lead wanted to (1) explore green propellants as options for CubeSat propulsion, (2) explore printed structures for propellant tanks as well as use in integrating the overall CubeSat, and (3) challenge the paradigm for spacecraft integration. At the end of these studies, it is clear that green propellants and AM are key enablers for advanced micropropulsion systems on CubeSats and SmallSats. It was also evident that it is difficult to achieve highly integrated multiuse structures for many subsystems, with little or no benefit for the trouble, and that managing simple interfaces of bulk volumes is preferred. Therefore, it is recommended that the propulsion system be a bulk volume allocation at the back end of the CubeSat, such as illustrated in figure 12. This is what the propulsion lead referred to as a stratified volumetric allocation.

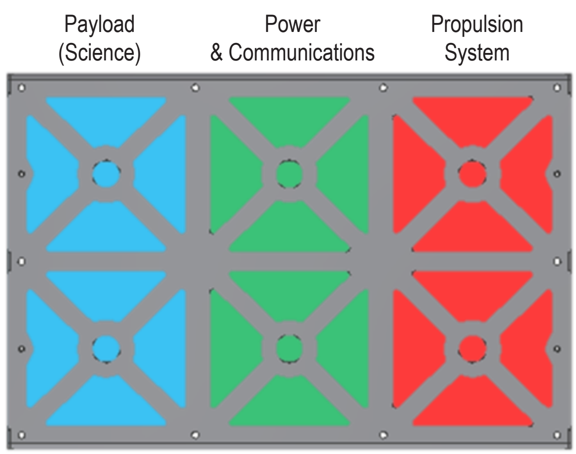

Figure 12. Recommended bulk volumetric allocation approach.

Additive manufacturing of micropropulsion system elements maximizes the miniaturization and packaging of propulsion system elements and features. This is most evident in the printed modules design by examining the propellant flow passages that are integrated into the structure's walls. This could not be reasonably achieved otherwise. Green propellants are both higher performing and pose less risk to the launch vehicle. There is further development to be made. This TE award has moved the needle far in the system level confluence of AM and green propellant. These two technologies are seen as key enablers for micropropulsion, and it is recommended that MoonBEAM baseline an AM and green propellant propulsion system if the goal is to fit within a 6U volume.

7.5 Avionics

Avionics includes command and data handling (C&DH) and communications systems. The GR&As that guided the avionics subsystem design are listed in table 13.

Table 13. Avionics subsystem GR&As.

	Category	Value
C&DH	Instrument data	Bused to processor or grouped into zones on backplane
	Data bus protocols	I2C, RS232, RS422
	Redundancy	2× real time clock
	Processor speed	400 MHz
	Data storage	2×2 GB SD card, 256 kB RAM nonvolatile
	Sensors/instrumentation	Internal temperature sensor
	Command rate	115.2 kbps to 1 Mbps
	Low rate telemetry	50 bps
	Environments	Assume flight-proven COTs
Communications	Downlink	4 kbps, crowd-source ground stations for data collection
	Data storage	250 MB/day for 5 days
	Command uplink	1 kbps

7.5.1 Command and Data Handling

The avionics in the spacecraft is divided into two functional sections, the spacecraft avionics stack and the science instrument stack. Physically, the cards are combined into one stack in the mid-forward part of the bus (see fig. 13). This arrangement allows for optimal data bus management and speed, power distribution across the cards, and thermal management of the stack. The spacecraft stack will perform all C&DH for the spacecraft, and also perform the data storage and downlink operations for the science instruments. It consists of a single-board computer (SBC), a digital input/output (I/O) board, and the avionics stack power supply board. The science instrument stack will perform the science data collection and processing, including analog-to-digital conversion and data compression and filtering. It consists of an instrument data processing board, a data I/O board, and an instrument high-voltage power supply board (see fig. 14, avionics compartment). In addition, the stack will include GN&C and propulsion boards as needed (secs. 7.3 and 7.4, respectively).

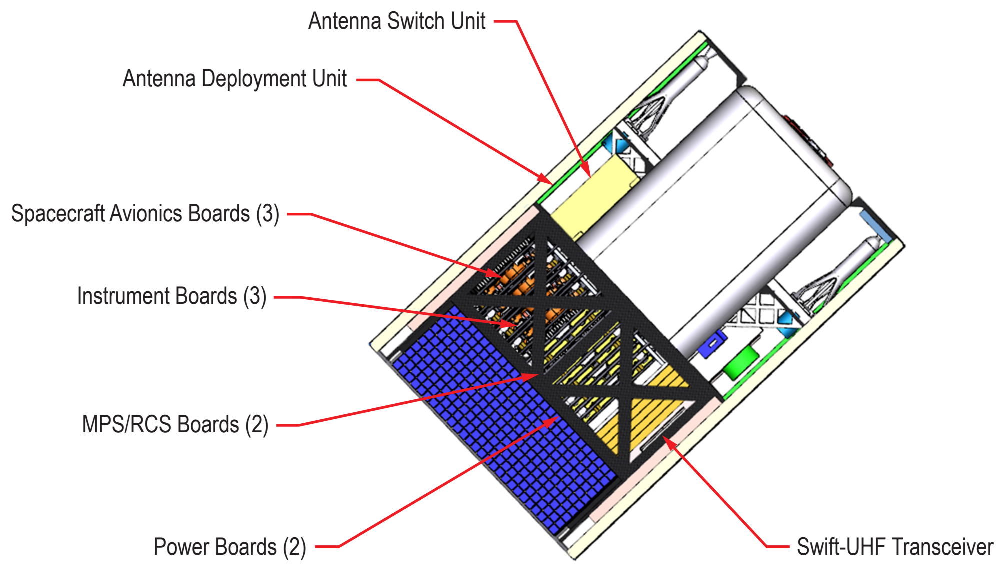

Figure 13. Avionics component locations within the bus—10 PC104 cards.

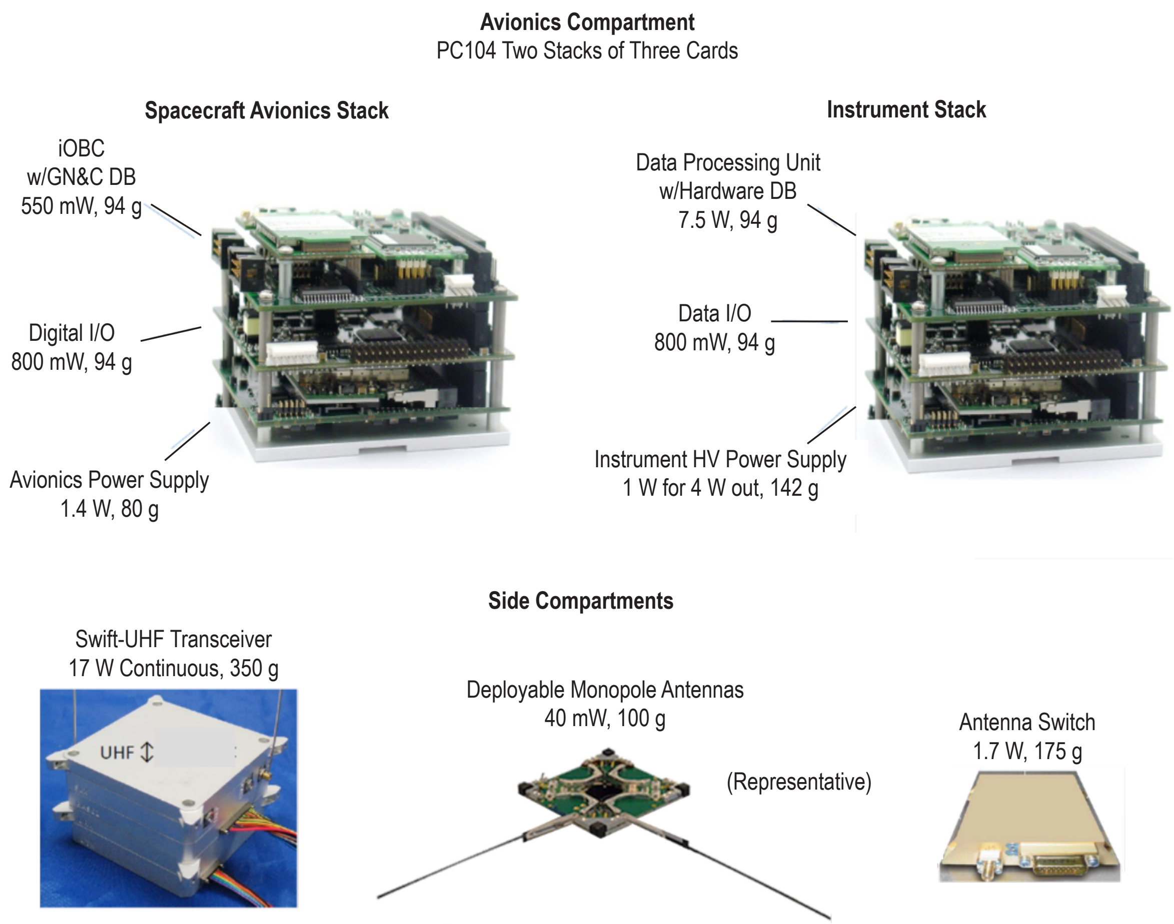

Figure 14. Spacecraft avionics stacks and communications components.

For CubeSats, it is common to use a PC104 form factor for avionics, and that is what was selected for this study. It is a small compact size (95.9×90.2 mm) with stackable printed circuit cards that all conform to the PC104 standard. There is no backplane required. The data buses pass through a common connector set. Many commercial-off-the-shelf (COTS) boards are available and are used in unmanned aerodynamic vehicle, drones, and missiles, some with flight heritage.

The spacecraft SBC was baseline on the innovative solutions in space (ISIS) ISIS on-board computer (iOBC) PC104 CubeSat board. It is a low power board and is a flight-qualified CubeSat design. This SBC meets all the avionics GR&A requirements, including the memory storage with 2×8 GB of flash memory (250 MB/day×5 days of storage requires 1.25 Gbits). The instrument processing SBC, digital I/O, and data I/O boards are based on Diamond Systems PC104 boards. They are representative of prephase A capability, mass, and power. They are ruggedized to MIL-STD-202[5] with a high operating temperature range and high shock and vibration levels. The space environment and radiation capability needs to be determined if these boards are used. The iSat avionics Cortex 160 SBC by Andrews Space was also considered, but that item has been discontinued and no longer supported. The NASA Jet Propulsion Laboratory has an SBC under development that might meet all the requirements and should be considered for this mission. It uses rad-hard components for space environments rated to the total ionizing dose of 30 krads.

In this study it was seen that the proposed science instrument package may present a data volume problem along with questionable power requirements. Depending on the detector configuration chosen, each of the SiPM cells could have up to four outputs. Multiplied by 364 cells per array gives 1,456 signal lines per array (7,280 for five arrays). This is too many discrete lines to bring into the instrument processor. It is assumed that the detector arrays will combine the signals into groups or zones and serial bus the data to the instrument processor via the data I/O board. Although array backplanes were included in this study and accounted for in the configuration, the data busing of the signals was not defined.

The high voltage required for the detectors was assumed to be 56 Vdc based on detector vendor specifications. A representative power supply board (Jupiter-MM) from Diamond Systems was chosen for the mass. However, it is a high wattage supply board, not a high voltage board. A custom-designed, high voltage power supply board may be needed.

7.5.2 Communications

For the communications system, the iSat Swift™ U-band transceiver (UTX) ultra-high-frequency (UHF) transceiver was chosen. It is small, lightweight, with lots of power, and up to 10 W radio frequency (RF) capability. The phase 1 link budget analysis was found to be still applicable for 250 MB/day from an LDRO. It assumed UHF band (430–440 MHz) uplink and downlink, at a distance of 1 million km. The data rate was 4 kbps with one-half binary phase shift keying modulation and a noise bandwidth of 8 kHz. The transmission power was 6 W RF, with a spacecraft antenna gain of 1.5 dBi and a ground receiver gain of 30 dBi.

For a continuous 4-kbps transmission to ground, it was assumed crowd-source ground stations would be utilized for data collection, per phase 1. That would be a consortium of NASA facilities, universities, and interested amateurs worldwide. It is assumed that a public Web site will be made available. On that site, an antenna pointing calculator will provide the current azimuth and elevation angles to the spacecraft for a specified ground station location. The site will also provide a folder to share science data acquired by all receivers, articles about discoveries made by the program to date, and a real-time bulletin board system for user discussion. Operations will be conducted by NASA and selected partners (universities, other space agencies, etc.).

With the spacecraft Sun pointing for solar array power and the gamma array detectors deep space pointing, it was necessary to include two sets of antennas (see fig. 15). To maintain communications when the Earth is on the solar array side of the spacecraft, two one-half-wave dipole antennas were built into the solar arrays (32.25 cm each). For all other positions relative to Earth, two one-fourth deployable low-gain monopole antennas (16.25 cm each) are employed. The antennas are orthogonal to each other and orientated so that one beam will point forward, while the other beams will point to the sides of the spacecraft. A deployment mechanism from ISIS CubeSat is suggested for the monopole antenna deployment (see fig. 11, side compartments). In addition to the Swift transmitter and antennas, a three-way antenna selection switch is required. A representative switch from L3 Cincinnati Electronics was used for mass allocation, but a custom antenna switch design may be needed.

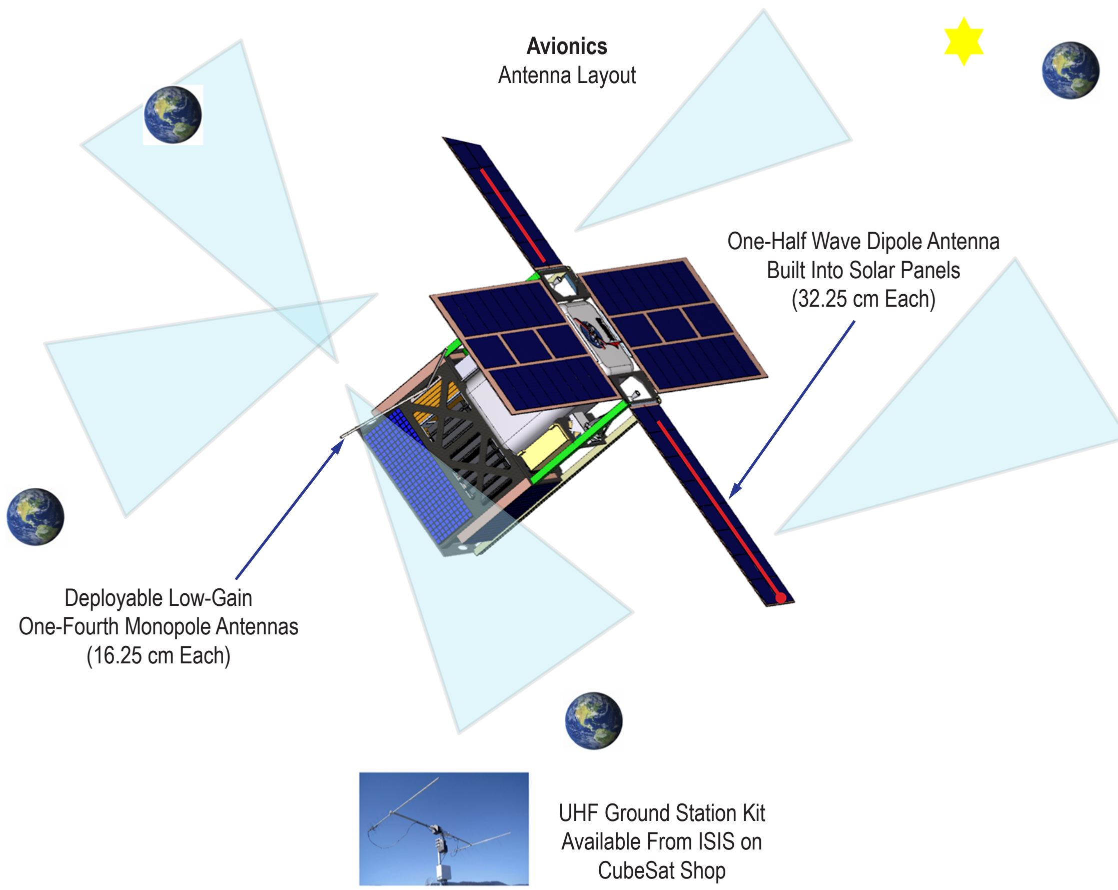

Figure 15. Spacecraft antenna locations and Earth field-of-view layout.

Figure 16 shows a block diagram of the Avionics and Science Instrument systems of the spacecraft. The diagram is baselined off the Fermi gamma-ray monitor spacecraft block diagram, where the instrument side is very similar to MoonBEAM. A major difference on that side is the use of the newer silicon photomultiplier (PM) arrays using much lower power (56 Vdc). The Fermi sodium iodide detectors and 5-inch-long PM tubes used up to 1,243 Vdc. The diagram includes GN&C components and spacecraft power for completeness, but refer to sections 7.3 and 7.7, respectively, for details of those systems. Table 14 is the avionics MEL for the study.

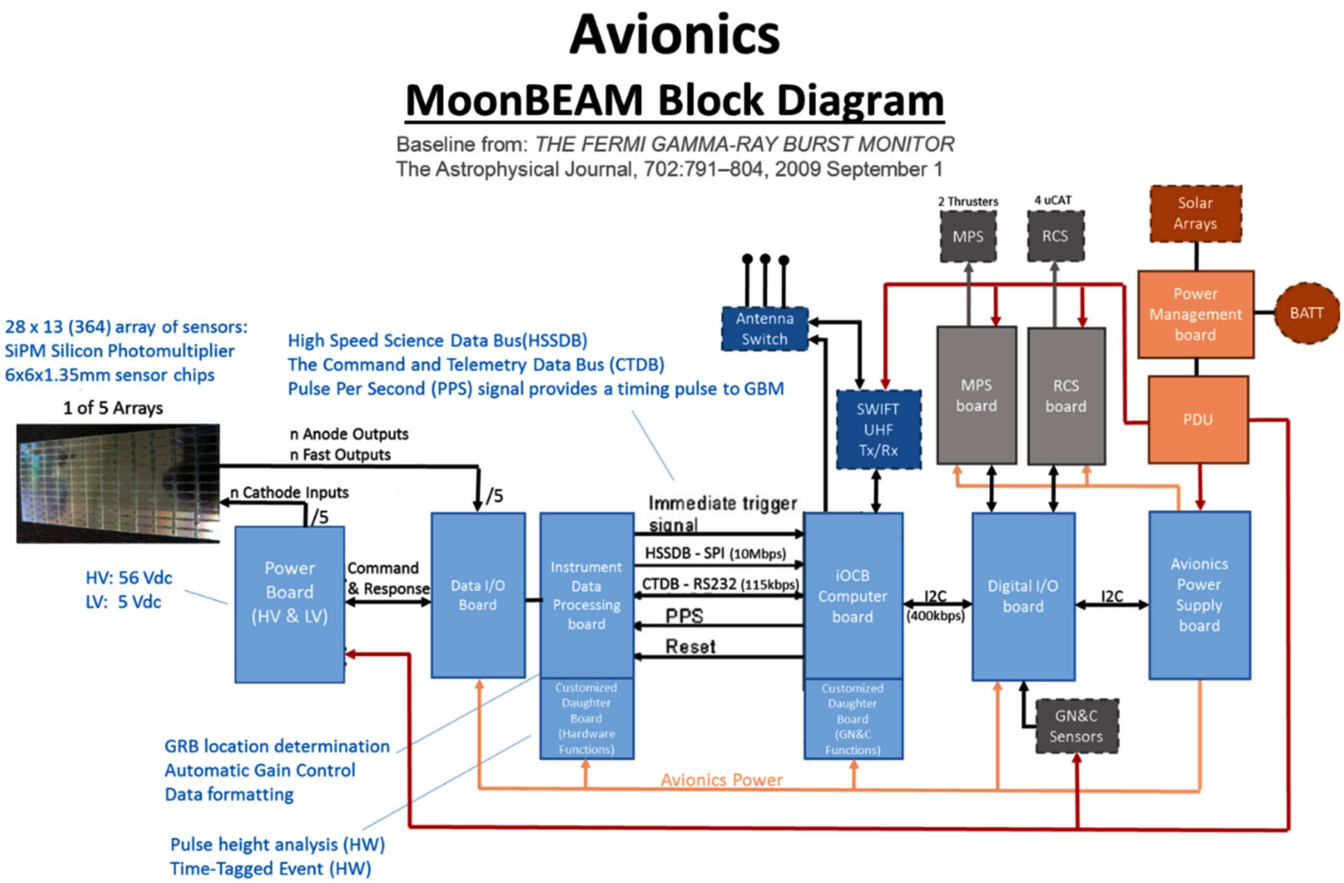

Figure 16. Spacecraft avionics and science instrument system block diagram.

Table 14. Avionics subsystem MEL.

Component	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
Swift-UTX transceiver	1	0.35	0.35	10	0.385
UHF antenna	1	0.03	0.03	20	0.036
Spacecraft avionics board stack	1	0.268	0.268	20	0.322
Instrument board stack	1	0.33	0.33	20	0.396
Antenna deployment	1	0.1	0.1	20	0.12
Antenna switch	1	0.175	0.175	20	0.21
Total			1.25	17	1.47

7.6 Structures

MoonBEAM structural models were created and analyzed using the following software tools:

- Pro/ENGINEER (Pro E) computer-aided design—used step file created in Pro E to create a finite element model of the MoonBEAM primary structures.
- MacNeal-Schwendler Corporation (MSC) Patran™—used for creation of finite element model.
- MSC Nastran™—solver used for analysis of all nonprinted primary structures.
- Collier Research Hypersizer—used in conjunction with MSC Nastran and Simmulia Abaqus to optimize all MoonBEAM primary structures.

The GR&As used in the structural assessment are listed in table 15. Since it is assumed that there will not be a dedicated test article to verify the structural adequacy, the yield factor of safety for metallic materials was set to 1.25 (protoflight) in accordance with NASA-STD-5001B.[6]

Table 15. Structures subsystem GR&As.

Category	Value
Dispenser interface	Continuous rail tab as defined in Planetary Systems Corps. Interface Control Document for 6U CubeSat dispenser
General	Primary structure will be designed to meet minimum strength requirements as stated in NASA-STD-5001B[6]
Load cases	Basic CubeSat structure will be designed using General Environmental Verification Specification ascent acceleration loads for small SLS payloads
Factor of safety for metallic materials	Ultimate factor of safety = 1.4 Yield factor of safety = 1.25
Secondary structures	Secondary structure mass is assumed to be 10% of the combined subsystem mass

7.6.1 Analysis

This section describes analysis of all nonprinted structural MoonBEAM components. Components analyzed include the side frames, mounting rails, upper stiffening L-sections, and a subsystem mounting frame. Figure 17 depicts the structure sized in this section. Analysis is performed using MSC Nastran SOL 101 as a solver with Hypersizer as an optimization tool.

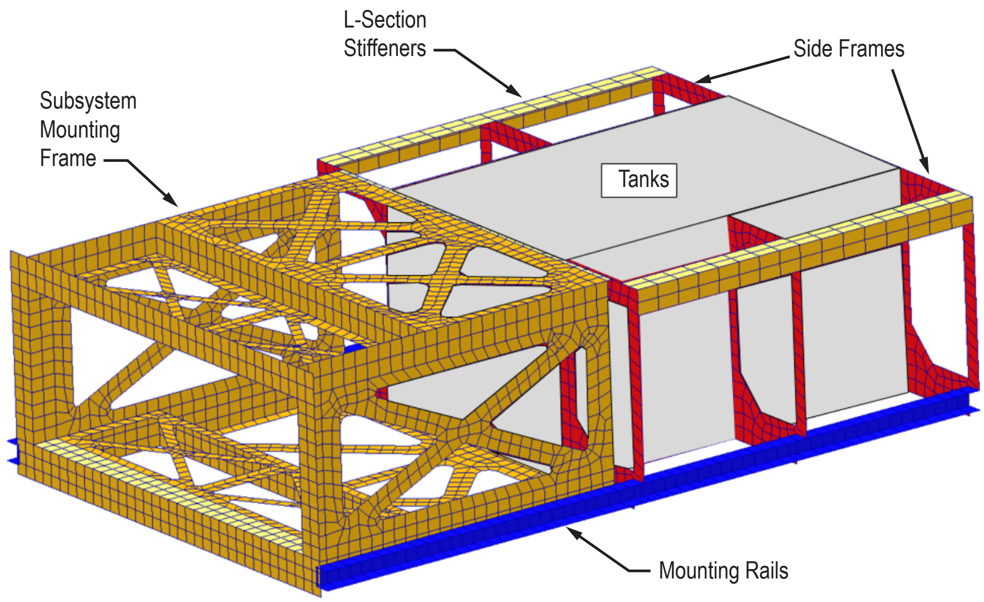

Figure 17. MoonBEAM step 2 structural analysis components.

All structure sized assumes aluminum construction with the exception of the tank side frames, which are Ti. Material specifications for the various components follows:

• Subsystem mounting frame—aluminum 2219-T851.
• L-section stiffeners—aluminum 2219-T851.
• Mounting rails—aluminum 7075-T6 (vendor requirement).

Mounting rail elements shown in blue are fabricated according to specifications dictated by the CubeSat deployment box vendor. Assumed material properties were as follows:

• Aluminum 2219-T851 (yellow elements)
  – Tensile strength = 61 ksi
  – Yield strength = 47 ksi
  – Modulus of elasticity = 10.5 Msi.

• Aluminum 7075-T6 (blue elements)
  – Tensile strength = 78 ksi
  – Yield strength = 70 ksi
  – Modulus of elasticity = 10.4 Msi.

Loads and constraints are shown in figure 18. The model is constrained along the mounting rails and loads consist of inertial accelerations applied in each primary axis of the model for a total of three load cases. The inertial acceleration magnitude was obtained from SLS General Environmental Verification Specification load graphs. Note that inertial loads are very high due to harsh dynamic environments on the walls of the SLS MSA. These load values are considered a good starting point for small SLS payloads such as the 6U CubeSat:

• Load case 1—126 g (applied in the x-axis).
• Load case 2—126 g (applied in the y-axis).
• Load case 3—126 g (applied in the z-axis).

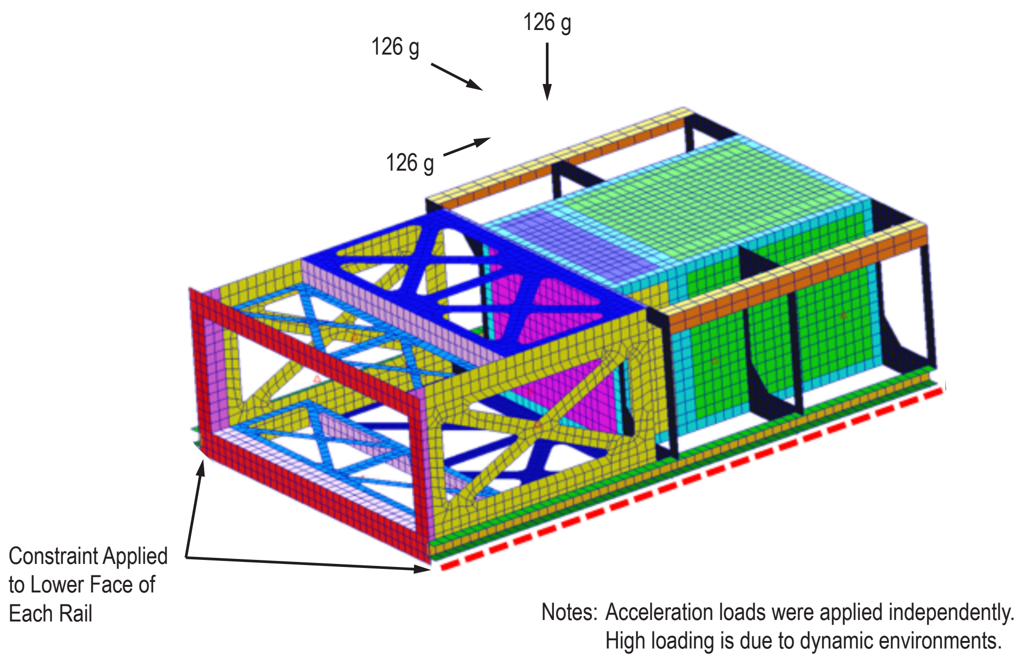

Figure 18. MoonBEAM structural optimization.

Final structural sizing for all nonprinted structural components is given in figure 19.

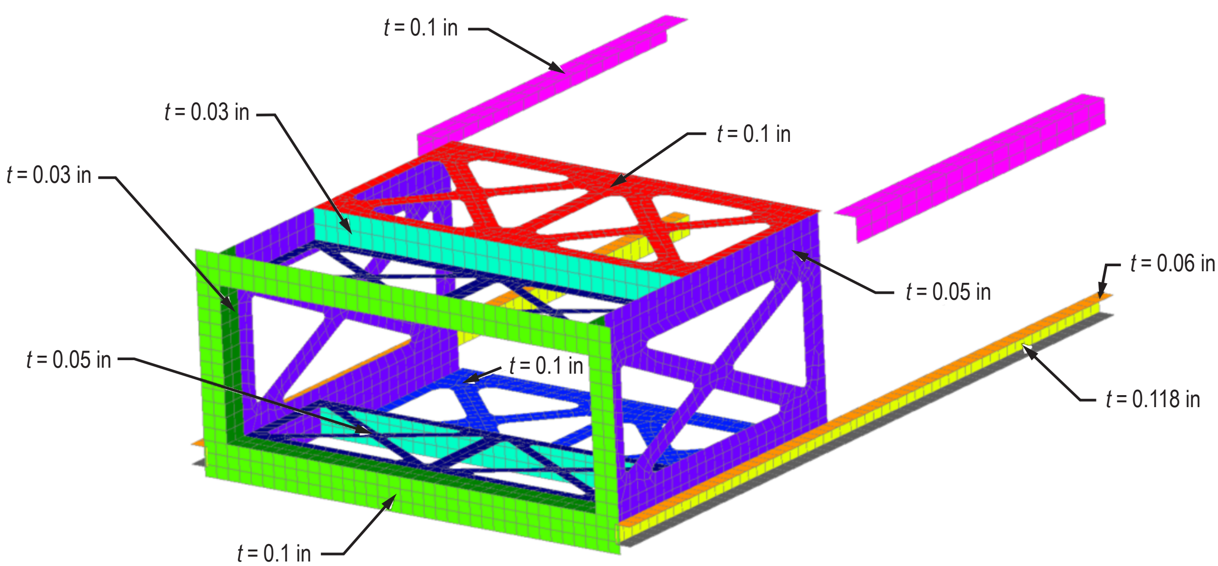

Figure 19. Analysis sizing results for all nonprinted (aluminum) structure.

The final structural masses are shown in the MEL in table 16. When comparing to the results shown in the phase 1 TM,[1] note that the propellant tank is considered part of the propulsion MEL for this study, while it was considered part of structures in phase 1.

Table 16. Structures subsystem MEL.

Component	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
Other structure (aluminum)	1	0.616	0.616	15	0.71
Secondary structures	1	0.352	0.352	–	0.352
Total			0.97	10	1.06

7.7 Power

The power system performs three functions for the spacecraft: (1) Generation of electrical power from sunlight, (2) storage of electrical energy for those time periods when there is no sunlight, and (3) conditioning and power switching to individual loads (turning each load on and off). The GR&As that guided the design of the power system are listed in table 17.

Table 17. Power subsystem GR&As.

Category	Value
Power subsystem required to provide power for all spacecraft elements plus payload power	Vehicle will provide capability to store, generate, manage/condition, and distribute power to all subsystems and payloads on the vehicle
Operation orbit	LDRO (500,000 km max)
Bus voltage	28 V nominal
Power during initial checkout/solar array deployment	Power will be provided to all attached architecture elements during initial checkout and solar array deployment (24 min)
Overload protection will be provided	For all critical functions (should consider resettable fuses)
Fault tolerance	Single string
Ground reference	A common ground reference will be provided across all subsystems
Secondary battery charge/discharge efficiency	95%
Secondary battery maximum depth of discharge	60%

There are five distinct power operation modes for this mission, each with a separate power requirement:

(1) Standby—The spacecraft maintains attitude with only avionics (computers, attitude control, and communications) powered. No science instruments are powered, and there is no propulsion.

(2) Initialization—The spacecraft has just been released from its carrier. Attitude control is asserted to stabilize the craft and eliminate spin; spacecraft systems are in test and solar arrays are in the process of being deployed. Power must come from batteries charged before launch.

(3) Science—The spacecraft is performing its normal science operations. Attitude control is asserted to point the spacecraft; avionics are powered but no propulsive maneuvers are ongoing.

(4) Preburn—The propulsion system is performing initialization in preparation for a propulsive maneuver, but propulsion has not yet begun. Science instruments are not powered, and the radio is off.

(5) Burn—Propulsion is accelerating the spacecraft while attitude control is pointing it into the direction of travel. No science instrumentation is powered. Power for this mode must be provided by batteries since the arrays may not be pointing toward the Sun.

Power requirements for each power operation mode are detailed in table 18.

Table 18. Power requirements by operation mode and system.

Load	Standby (W)	Initialization (W)	Science (W)	Preburn (W)	Burn (W)
Avionics	29	29	29	12	29
GN&C	0.9	5.4	5.4	5.4	5.4
Propulsion	–	–	–	32	9
Science	–	–	4	–	–
Total	30	34.4	38.4	49.4	43.4
Total with 20% margin	36	41.3	46.1	59.3	52.1

Note that the power design margin is 20% instead of the AIAA minimum of 30%. Because the maximum power available from the arrays is limited to 60 W total, 30% margin for the preburn mode is not possible. This represents a source of risk to the project at this stage of development.

The operations sequence details the power balance in table 19.

Table 19. Power operations sequence.

Operation	Power Required (W)	Power Generated (W)	Balance (W)	Comment
Initialization	41.3	–	–41.3	Battery provided (37 min)
Powered coast	36	60	24	Balance used to charge battery (1:48 charge time). Arrays must be pointed directly toward Sun
Preburn warmup	59.3	60	0.7	Arrays must be pointed directly toward Sun
Burn	52.1	0–60	N/A	Powered by battery (24 m Max)
Powered coast	36	60	24	Balance used to charge battery (1:48 charge time)
Science	46.1	52	5.9	Assumes 30° off-point

During initialization, the spacecraft requires 41.7 W power but is not producing any power at all. The battery is therefore providing all power for the vehicle. After 37 minutes, the battery will be at the limit of discharge (40% charged).

While coasting in standby mode with arrays deployed and pointing directly at the Sun, the arrays provide 60 W of power while the spacecraft requires only 36 W. That leaves 24 W with which to charge the battery. At this rate, the battery will be fully charged after 1.8 hours.

During preburn, the power required will be 59.3 W. With the arrays pointed directly toward the Sun, they will produce at least 60 W—just enough to meet the requirement.

In the burn mode, the spacecraft must be pointed so that the thrust propels the craft in the required direction of travel. The arrays, then, may not be pointed toward the Sun, thus may not produce enough (or any) power. All power must be assumed to come from the battery. At 52.1 W, the battery can power the craft for 30 minutes before reaching the minimum charge state of 40%. The battery must be charged in standby mode for at least 1.8 hours afterward in order to have enough energy for another 30-minute burn.

Figure 20 shows the major elements of the power subsystem. Regarding the power system design, the solar arrays and power electronics are taken from the iSat (a CubeSat with an electric thruster using iodine as the propellant) project. The solar arrays are layed out on the same substrate as those for iSat, but are configured differently. Instead of the three 20 cm by 30 cm panels on iSat, the iSat body-mounted panel is divided into two 10 cm by 30 cm panels (making four panels altogether) and arranged as shown.

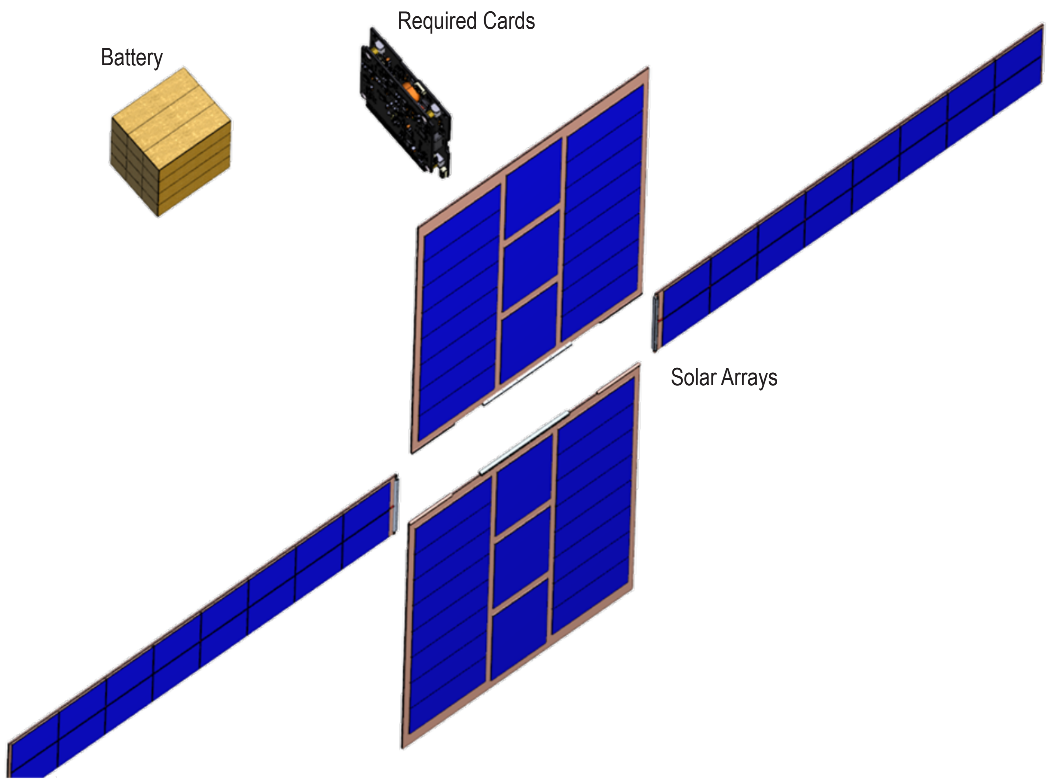

Figure 20. Power subsystem elements.

The resulting array is nevertheless the same in area (0.18 m^2) and generates the same amount of power (60 W at end of life).

The power electronics consist of two boards—a power management board that regulates the solar array and provides charge control for the secondary battery and a power distribution board that switches all loads. These are taken directly from the iSat design as well. Table 20 details the mass breakdown of the power system.

Table 20. Power subsystem MEL.

Component	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
Solar panel—full size	2	0.266	0.53	20	0.64
Solar panel—half size	2	0.133	0.27	20	0.32
Power management board	1	0.06	0.06	20	0.07
Power distribution board	1	0.06	0.06	20	0.07
Battery	1	0.44	0.44	20	0.53
Total			1.36	20	1.63

7.8 Thermal

The thermal control system for MoonBEAM will be a critical aspect of the final design. In addition to the typical task of ensuring that the avionics and power system components remain within each of their operational temperature ranges, the thermal control system is further tasked with mitigating the conduction of heat generated by the propulsion system into the spacecraft structure.

The general approach to the design of the thermal control system for MoonBEAM is to conductively isolate the science instruments from the payload, use thermal straps to conduct heat from the avionics into the fuel tank, and use high TRL technologies like multilayer insulation (MLI) and thermal coatings as needed to reflect radiation away from sensitive components.

Operating temperature ranges for the major components of MoonBEAM have been assumed and are tabulated in table 21. In addition to these values, it is noted that the fuel tank needs to be maintained at a minimum temperature of 10 °C, and that the thruster will need to be heated to 400 °C prior to firing in order to prime the catalyst. The design and estimated mass of the heating system required for these tasks are included in the propulsion system design.

Table 21. Assumed survival and operating temperatures for MoonBEAM components.

Subsystem	Component	Survival Min (°C)	Survival Max (°C)	Operating Min (°C)	Operating Max (°C)	Power (W)
Power	Power management board	–20	60	–20	60	–
Power	Power distribution board	–20	60	–20	60	–
Power	Battery	–20	50	–10	40	–
Power	Solar cells	–	110	–	100	–
Avionics	Avionics board stack	–40	95	–25	65	2.75
Avionics	Instrument board stack	–40	95	–25	65	9.3
Avionics	Swift transceiver	–40	95	–25	65	17
GN&C	Reaction wheel	–	–	–40	70	1
GN&C	Star tracker	–40	95	–40	50	1
GN&C	IMU	–	–	–40	85	0.1
GN&C	Sun sensor	–	–	–25	50	0.1

Mass properties for the components of the MoonBEAM thermal control system are listed in table 22. The masses for coatings and MLI were assumed based on a percentage of the total surface area of the spacecraft. Two thermal straps were sized to allow heat from avionics components to be conducted into the fuel tank. No thin-film heaters are included in this mass estimate. Thermal modeling of the spacecraft during each mission phase will be required to size these heaters.

Table 22. Mass estimate for MoonBEAM thermal control system.

	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
MLI	1	0.1	0.1	30	0.13
Coatings	1	0.01	0.01	30	0.013
Thermal straps	2	0.039	0.078	15	0.09
Totals			0.188	24	0.233

Heat conducted from the thruster into the spacecraft is of particular concern. An initial, 1-D transient analysis estimated minimal heat conducted into the thruster; however, this analysis should be confirmed via test.

A complete thermal model of the spacecraft should be developed as the design matures. The model would be exercised for the different mission phases and can be used to improve the mass estimate for the thermal control system.

8. STRONGBACK CONCEPT

Another design option that the team assessed was a 'strongback' concept in which the AM propellant tank would act as the primary structure for attachment of the subsystems and science instruments. For this design, the propellant tank would extend nearly the length of the spacecraft, with the purpose being to package more efficiently and save mass (or increase propellant volume if mass constraints allowed). The design team did no subsystem analysis for this design except for propulsion, which laid out the tank and thruster design, and configuration, which placed the other subsystems and necessary structure in the remaining volume.

Figure 21 shows the strongback design. Note that the number of main thrusters is reduced to one, as opposed to two in the baseline design. While this saves volume, it also requires longer burn times for maneuvers. Unfortunately, the strongback configuration did not package well enough to allow both thrusters to carry over. The layout also required the avionics card stack to be split, requiring three separate stacks as opposed to a single stack in the baseline design. While this is a minor issue, it does result in more cabling.

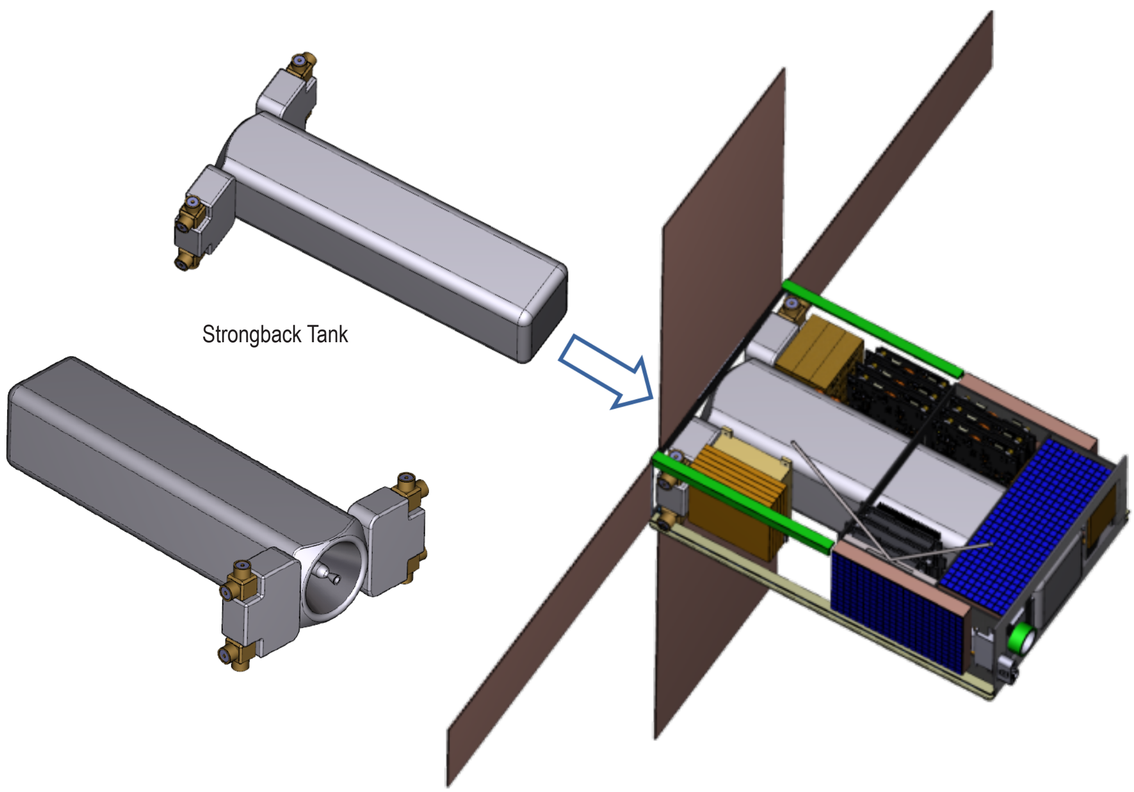

Figure 21. Strongback configuration showing the revised propellant tank and thruster layout. Note the split configuration of the avionics cards necessary to allow room for the tank.

Figure 22 compares the two configurations. Note that the strongback is not as structurally efficient as was anticipated, as structure must still be supplied for mounting the gamma-ray detectors (shown in blue) and the solar arrays (not shown). In addition, the deployment rails, which must be aluminum, are also required. Even if the structure for mounting the subsystems could be printed with the tank, these other structural elements would still be required.

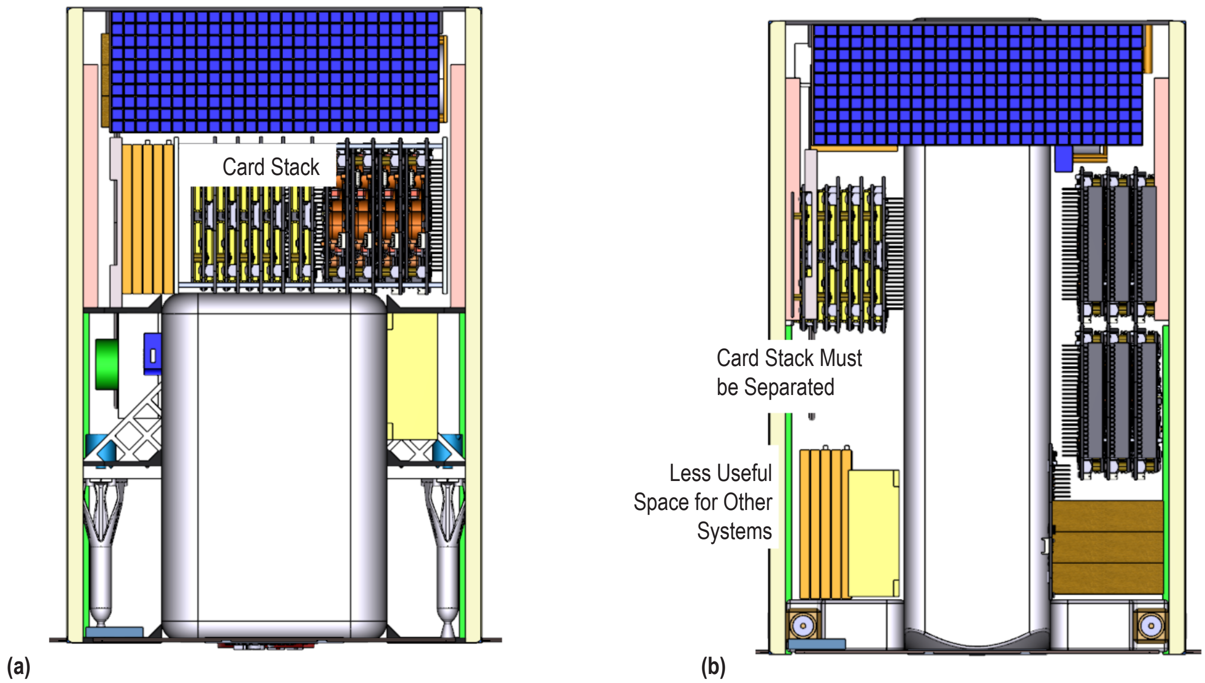

Figure 22. Comparison of the (a) baseline tank and (b) strongback tank.

In conclusion, the design team could see no advantage to the strongback design, even though a complete subsystem analysis should be done to allow a complete assessment. The packaging just does not offer any advantages. In fact, it underscores how well the baseline design was configured.

9. GAS GENERATOR CONCEPT

Another option that the design team considered was a gas generator concept. A gas generator is a propulsion system element that generates pressurant gas on orbit, rather than requiring a pressurized volume at launch. Advantages of this system are as follows: (1) A reduced potential threat to the launch vehicle, (2) avoid fracture critical classification, (3) easily ignited and burns rapidly, (4) long-term storable, (5) low reactivity, and (6) the maximum expected operating pressure (MEOP) for the propulsion system can be much lower, meaning that the propulsion system structure can be thinner and lighter. While no other subsystem analysis was completed for this concept, the propulsion analyst did design a propulsion system that employed a gas generator. Lack of time prevented the design from being incorporated into the spacecraft configuration, but the team was able to determine the impact of the gas generator on mass and volume.

Funded by this TE award, gas generator testing was conducted at MSFC within the Spacecraft & Auxiliary Propulsion Systems Branch (ER23) to investigate potential gas-generating compounds and characterize a chemical reaction that could be utilized for a postlaunch pressurization system (PLPS). A test setup, as shown in figure 23, was fabricated and initial experiments were performed to obtain critical measurements, evaluate pressurizing capabilities, and understand the reaction behavior of a gas-generating compound. The test setup was designed to be iterated upon and simulate the PLPS. In addition to characterizing gas-generating compounds, experimental results will inform additional compound criteria as well as future PLPS design requirements.

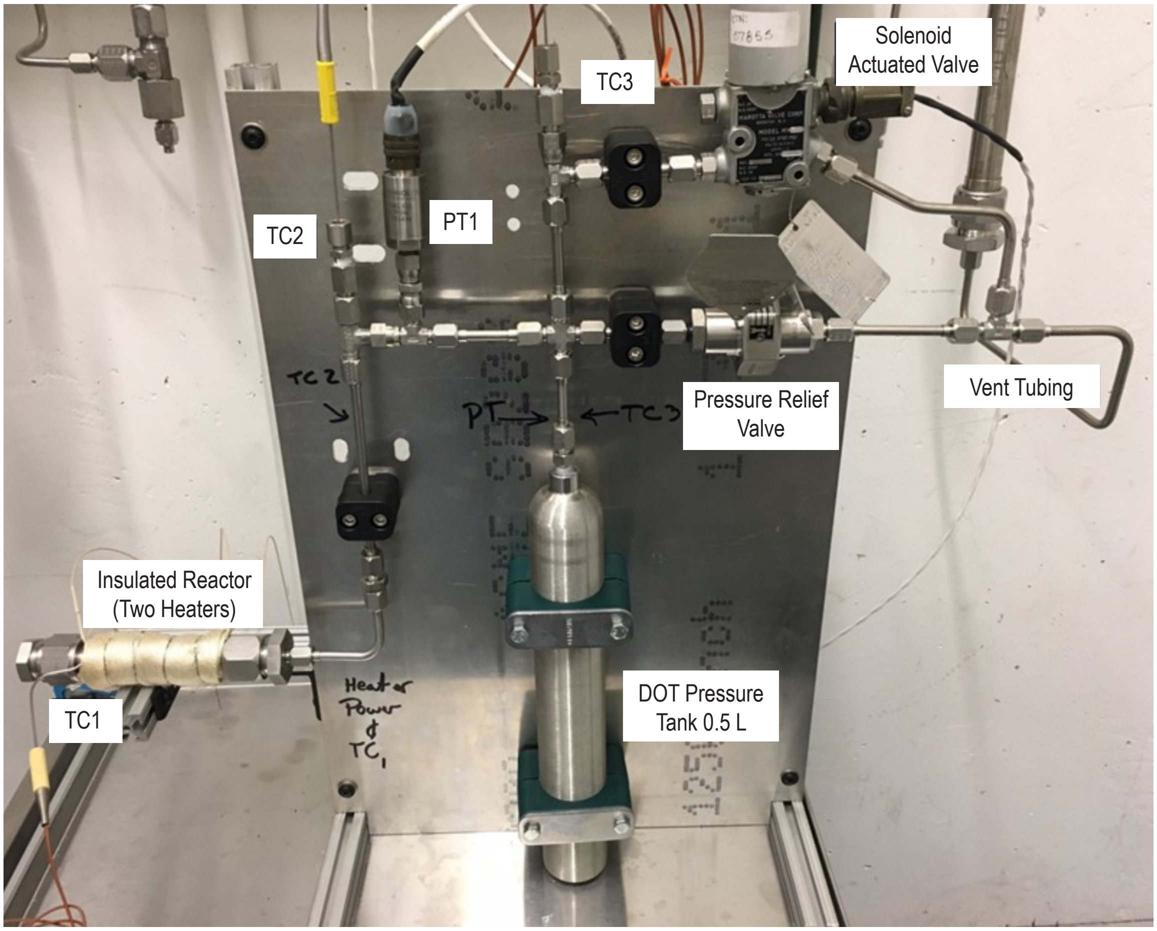

Figure 23. PLPS testing configuration.

Consistent pressure yields for varying quantities of the tested compound were achieved. However, higher pressure yields would be desirable and additional gas-generating compounds or packing methods should be investigated. Further characterization and improved accuracy of postreaction product mass measurements is also required to precisely determine product weight percentages. The test setup, procedures, and chemical reaction analysis may serve as a framework for the investigation for future potential gas-generating compounds and development of the PLPS.

A PLPS is designed to generate the gasses required to drive secondary payload propulsion systems. The gas generated by the PLPS will feed into a pressure tank, followed by the propellant tank, and then the thruster chamber. The PLPS would most likely be used for small CubeSat propulsion systems. A high level schematic of a monopropellant propulsion system utilizing a PLPS is shown in figure 24.

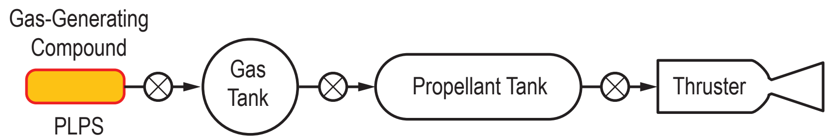

Figure 24. Generic propulsion system gas generator system schematic.

Over two dozen tests were performed in the experiment, with pressure of greater than 500 psia being generated by the compound. The compound tested generated a mix of dry inert gases. Thanks to the data gathered from the experimental trials, it was possible to make some educated estimates of the mass and volume of a PLPS for MoonBEAM.

The PLPS would occupy a volume of 304 to 432 cm^3 and have an estimated mass of 0.65 to 0.86 kg. This estimate assumes a 20% mass growth allowance. Additional information can be found in a recently published NASA TM.[7] Given those estimates, a gas generator concept was formulated and is shown in figure 25.

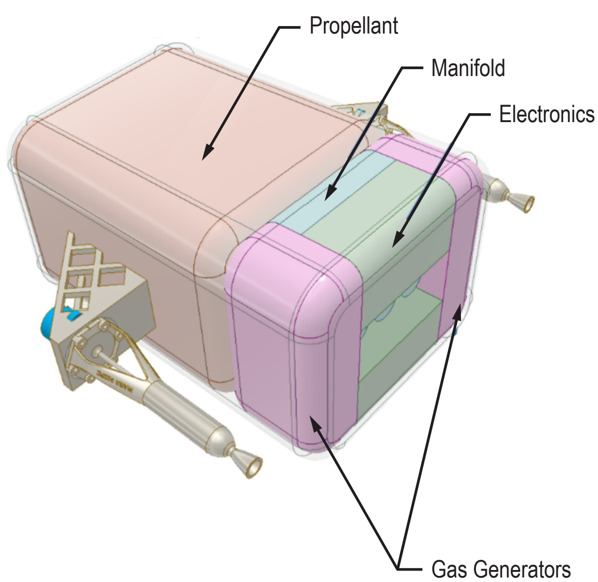

Figure 25. MoonBEAM gas generator configuration concept.

The gas generator could be designed such that it allowed for successive gas-generating events over the course of the mission, which would greatly lower the MEOP. This would subsequently allow for a thinner walled, lighter propellant tank structure. The study period did not allow for a reoptimization for the propulsion system structure to account for a lower MEOP.

The effort has brought the gas generator system to a TRL of 3. A follow-on effort is planned for spring 2018 to design a higher fidelity system that will allow for expulsion and repressurization of a bread board system in a relevant environment. This will, if successful, bring the PLPS's TRL to 4.

Table 23 shows the mass comparison of the baseline and gas generator concepts. Since no subsystem analysis could be completed for the other subsystems, the estimated mass of the gas generator concept assumes all other subsystem masses remain constant. This assumption may not hold, as power and thermal subsystems could be affected. Nevertheless, one can see that the overall mass of the spacecraft is expected to increase. The actual mass increase of the propulsion system is unknown, as the gas generator allows for thinner-walled tanks, which would save some mass, though not enough to negate the 0.92-kg mass increase shown in the table.

Table 23. Mass summary comparing the baseline propulsion system design and the gas generator option.

Element	Baseline Design	Gas Generator Option
Tank struture (printed Ti)	3.11	3.11
1N AF-M315E	0.2	0.2
Thruster/valve driver board	0.06	0.06
μCAT thruster module	0.1	0.1
μCAT controller board	0.07	0.07
Valves	0.2	0.2
Manifold	0.23	0.23
Gas generator	–	0.92
Mass total	3.97	4.89

In summary, it appears that the primary benefits of the gas generator concept would be safety for other payloads (if the spacecraft is a ride share) and packaging, and not mass savings.

10. CONCLUSIONS AND FUTURE WORK

Key points of the design are listed in table 24. During the phase 1 study, the original 8.75U spacecraft volume was reduced to 6U with the help of AM. Between the phase 1 and phase 2 studies, the propulsion designers took the volume and packaging requirements from the phase 1 study, performed some AM work to prove feasibility, and redesigned the propulsion system, which was incorporated into the phase 2 spacecraft. With the assumptions made by the phase 2 propulsion analyst for ullage, I_{sp}, and propellant density of the green propellant, the resulting design with a full propellant load is slightly over the mass limit of the 6U deployer, though a propellant offload of 0.19 kg allows the design to meet the mass requirement while still achieving an orbit that meets the science requirements. The packaging efficiency afforded by the AM propellant tank is the largest single contributor to the reduction in volume, with the propellant density and I_{sp} also being important, but to a lesser extent. These three items allowed the previous 8.75U design to be repackaged into a 6U volume.

Table 24. Brief sumary of spacecraft design.

	Baseline Design	**Baseline With 0.19 kg Propellant Offload**
MoonBEAM dry mass	11.93 kg	11.93 kg
Dry mass growth allowance	16%	16%
Propellant mass	2.26 kg	2.07 kg
MoonBEAM wet mass	14.19 kg	14 kg
ΔV capability	334 m/s	304 m/s
Format (volume)	6U	
Propellant	AF-M315E High-Performance Monopropellant	
I_{sp}	220 s	

While the original MoonBEAM science mission may eventually be proposed on a more conventional CubeSat platform, the results of this study clearly show the benefits of AM to CubeSats, resulting in more capable and volumetrically efficient designs that greatly exceed the propulsive capability of the current state-of-the-art, enabling CubeSats to extend their reach into cis-lunar and interplanetary space with chemical propulsion. Even though the resulting design did not meet the original maneuver budget required to reach LDRO, it still provides sufficient capability with chemical propulsion to achieve several other orbits that meet the science requirements, such as an EM-L3 halo and a TESS-type orbit if ride sharing on SLS EM-1, which is a substantial increase in 6U CubeSat capability.

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14. ABSTRACT

The Advanced Concepts Office at NASA Marshall Space Flight Center completed the second phase of a mission concept study for the Moon Burst Energetics All-sky Monitor (MoonBEAM). The goal of the concept study was to show the enabling aspects that additive manufacturing can provide to CubeSats, and to implement propulsion elements that were demonstrated to be feasible based on requirements from the first phase of the study. The spacecraft uses additively manufactured components for much of the main propulsion system, and green propellant, which is denser than hydrazine. Momentum unloading is achieved with electric microthrusters, eliminating much of the propellant plumbing. The science mission, requirements, and spacecraft design are detailed in this Technical Memorandum (TM), which is a follow-on to the TM published at the end of phase 1.

15. SUBJECT TERMS

additive manufacturing, CubeSat, conceptual spacecraft design, green propellant, gamma-ray burst, gravitational wave

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