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

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October 2017

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LIST OF ACRONYMS, ABBREVIATIONS, DESIGNATORS, AND SYMBOLS

ACO	Advanced Concepts Office
ADCS	Attitude Determination and Control System
AFM	abrasive flow machining
AIAA	American Institute of Aeronautics and Astronautics
AM	additive manufacturing
C&DH	command and data handling
COTS	commerical off-the-shelf
DRO	distant retrograde orbit
EBM	electron beam melting
ED04	Advanced Concepts Office
EELV	evolved expendable launch vehicle
EM	exploration mission
EM32	Metal Joining and Processes Branch
ER23	Spacecraft & Auxiliary Propulsion Systems Branch
ES21	Structural & Mechanical Design Branch
ES22	Thermal & Mechanical Analysis Branch
ESPA	EELV secondary payload adapter
FOI	Swedish Defense Research Agency
GBM	gamma-ray burst monitor
GEVS	General Environmental Verification Specification
GN&C	guidance, navigation, and control
GPS	global positioning system

LIST OF ACRONYMS, ABBREVIATIONS, DESIGNATORS, AND SYMBOLS (Continued)

GR&A	ground rules and assumptions
HAN	hydroxylammonium nitrate
HEHN	hydroxyethylhydrazie
ICPS	interim cryogenic propulsion system
IMU	inertial measurement unit
I/O	input/output
ISIS	Innovative Solutions in Space
ISO	isolation
iSAT	iodine satellite
JPL	Jet Propulsion Laboratory
L2	second Lagrange point designator
LBB	leak before burst
LDRO	lunar distant retrograde orbit
LEO	low Earth orbit
LFPS	Lunar Flashlight Propulsion System
MDP	maximum design pressure
MEL	master equipment list
MoonBEAM	Moon Burst Energetics All-Sky Monitor
MPCV	multipurpose crew vehicle
MSA	MPCV stage adapter
MSC	MacNeal-Schwendler Corporation
MSFC	Marshall Space Flight Center
NEA	Near Earth Asteroid

LIST OF ACRONYMS, ABBREVIATIONS, DESIGNATORS, AND SYMBOLS (Continued)

PM	photomultiplier
PRV	pressure regulation valve
RCS	Reaction Control System
RF	radio frequency
RMS	root mean square
SBC	single board computer
SBIR	Small Business Innovative Research
SCAPE	Self-Contained Atmospheric Protectve Ensemble (facility)
SiPM	silicone photomultiplier
SLS	Space Launch System
SP	secondary payload
SPDS	secondary payload deployment system
SSC	Swedish Space Corporation
TE	Technical Excellence (award)
Ti	titanium
TM	Technical Memorandum
TRL	Technology Readiness Level
XP50	SSL, Spacecraft/Payload Integration and Evolution (SPIE) Office
UHF	ultra-high-frequency
UTX	U-band transceiver
μCAT	microcathode arc thruster

NOMENCLATURE

g	gravitational acceleration
I_{sp}	specific impulse
m_{prop}	propellant mass
t	thickness
V	volume

TECHNICAL MEMORANDUM

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

1. INTRODUCTION

During early 2017, the Advanced Concepts Office (ACO) at NASA Marshall Space Flight Center (MSFC) completed 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 (AM) can provide to CubeSats. In that spirit, this design study has much in common with the show cars of the 1960's, generating interest and showing what may be possible in the future. For example, many features of General Motors' 1958 show car, the XP 700, made their way into the Corvettes of the early 1960's, including the ground-breaking 1963 Stingray™. Similarly, many aspects of this proposed CubeSat design will find their way into small satellites of the near future. In addition to using the additively manufactured tanks as part of the spacecraft structure, the system uses a green propellant, AF-M315E high-performance monopropellant, which is denser than hydrazine, saving precious volume. The design also incorporates microcathode arc thruster (μCAT) electric microthrusters for momentum unloading, eliminating the need to carry additional propellant for unloading the reaction wheels, as well as eliminating the plumbing necessary for routing propellant.

This Technical Memorandum (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. Overall, it 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 change in volume (ΔV) requirements such as lunar and interplanetary. Additive manufacturing may indeed be a game changer for the 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 a 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 follow-up observations searching for the electromagnetic counterpart of gravitational waves.

3. SCIENCE INSTRUMENTS AND OPERATION

The description of the instruments presented below is to the level required for the concept 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 1 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, respectitvely. While four detectors were allocated in the final master equipment list (MEL) for the study, the configuration shows five detectors. This was done to show that the volume exists for the additional detector, though the mass of the other subsystems would need to be reduced in order to meet the overall mass limit. In addition, the instruments require electronics cards in the spacecraft. These elements are considered part of the avionics subsystem and are detailed in that section of this TM.

Table 1. Basic science instrument dimensions and requirements.

Instrument	Dimensions or Area	Mass (kg)	Power (W)	Temperature (°C)
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 2 and 3 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 only uses the minimum number of detectors. Electronics boards for the instruments are not included in the MELs here, but are part of the avionics subsystem, detailed later in this TM.

Table 2. 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 3. 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

Since the goal of this study was focused on the benefits of AM in relation to CubeSats, rather than the science mission, the approach was different than is typical for concept studies. Usually, the science customer is interested in determining the feasibility and cost of a proposed mission. However, for this concept study, the customer hoped to see what breakthrough performance could be enabled by using AM for the design of the propulsion system, and structure, of a small CubeSat. In addition, the employment of green propellants with higher densities were also part of the trade space, with the goal being to reduce the propellant volume.

Because CubeSats are, by definition, small, they are very limited in volume. The subsystem components for a typical 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 and 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 be the example case for demonstrating the benefits of AM. A previous study (in 2016) 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 results of the 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. These goals, determined by subsystem and AM experts to be achievable, will be put into practice this year, as one of the outcomes of the study is a roadmap and set of performance goals that the AM experts will try to meet by actually printing these components.

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 ACO 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 the 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 provide a path forward, and goals that the AM propulsion and structural elements would have to meet.

Table 4 lists the requirements that guided the design study. The trajectory requirements were few regarding science. 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-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's (SLS's) exploration missions (EMs), the design team carried over the previous study baseline orbit of the lunar distant retrograde orbit (LDRO), with a rideshare on SLS EM-1. (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 4. 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 × days before download + margin
Event data transfer	100 kb per trigger, 10 triggers per day to ground within 60 minutes
Pointing control (driven by space-craft, 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 × 6 mm MicroFC-60035-SMT (36 mm^2); 2 × 2 array 14.2 × 14.2 mm^2 ArrayC-60035-4P-BG
No. of SiPM's	Cover scintillator crystal surface area, 75% fill factor using 2 × 2 arrays
No. of detectors	Four minimum
Operating temperature	–40 °C 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. This assumption provided the outbound trajectory, mass limitations, and transfer trajectory requirements necessary to place the spacecraft into its operational orbit, which is LDRO. LDRO is very stable, requiring little or no orbit maintenance. However, getting to LDRO is challenging[1] from a CubeSat perspective because of a rather large ΔV budget necessary for the transfer. This was actually a reason to select this option, for it provided a challenging goal for the design team to meet. If successful, the design team would be showing a substantial increase in CubeSat capability with AM.

A plot of LDRO is shown in figure 1. This is a highly stable orbit, requiring a minimum of station keeping to maintain. 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.

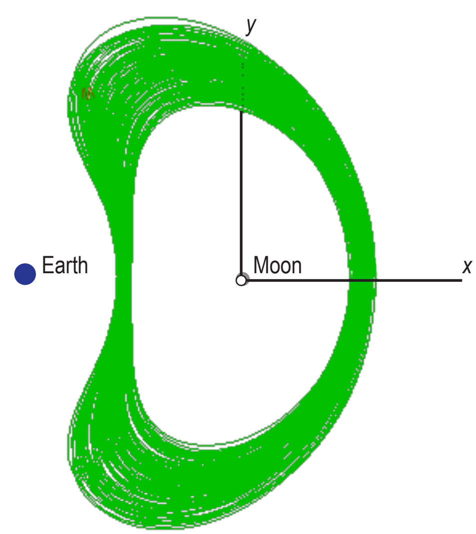

Figure 1. Lunar distant retrograde orbit in the rotating frame. Total $\Delta V = 368.5$ m/s.

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 5. 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 7.1. Also shown in table 5 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.

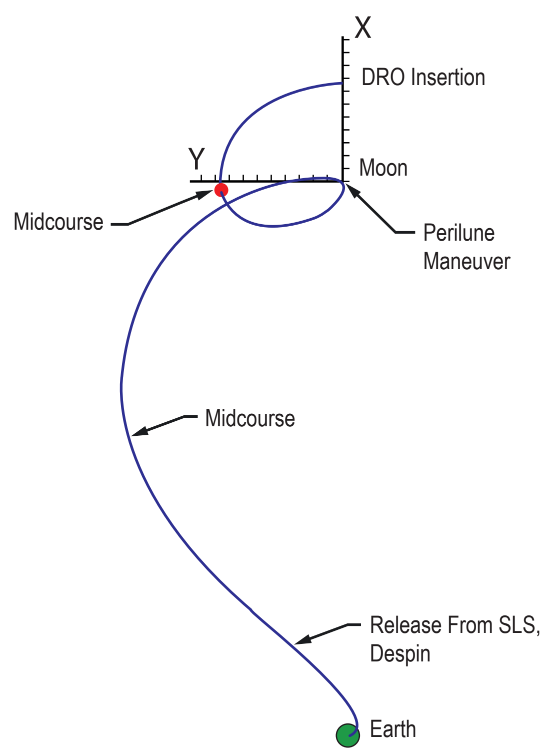

Figure 2. Transfer trajectory to LDRO. Trajectory work was performed on a previous study by Dan Thomas (ED04), and is representative of a possible MoonBEAM transfer trajectory.

Table 5. Maneuver (ΔV) budget for the MoonBEAM mission.

Maneuver/Category	Value
Despin	0 m/s
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	0 m/s
Orbit maintenance	0 m/s
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 can be seen in figure 3. Sitting between the interim cryogenic propulsion system (ICPS) 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 the 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.

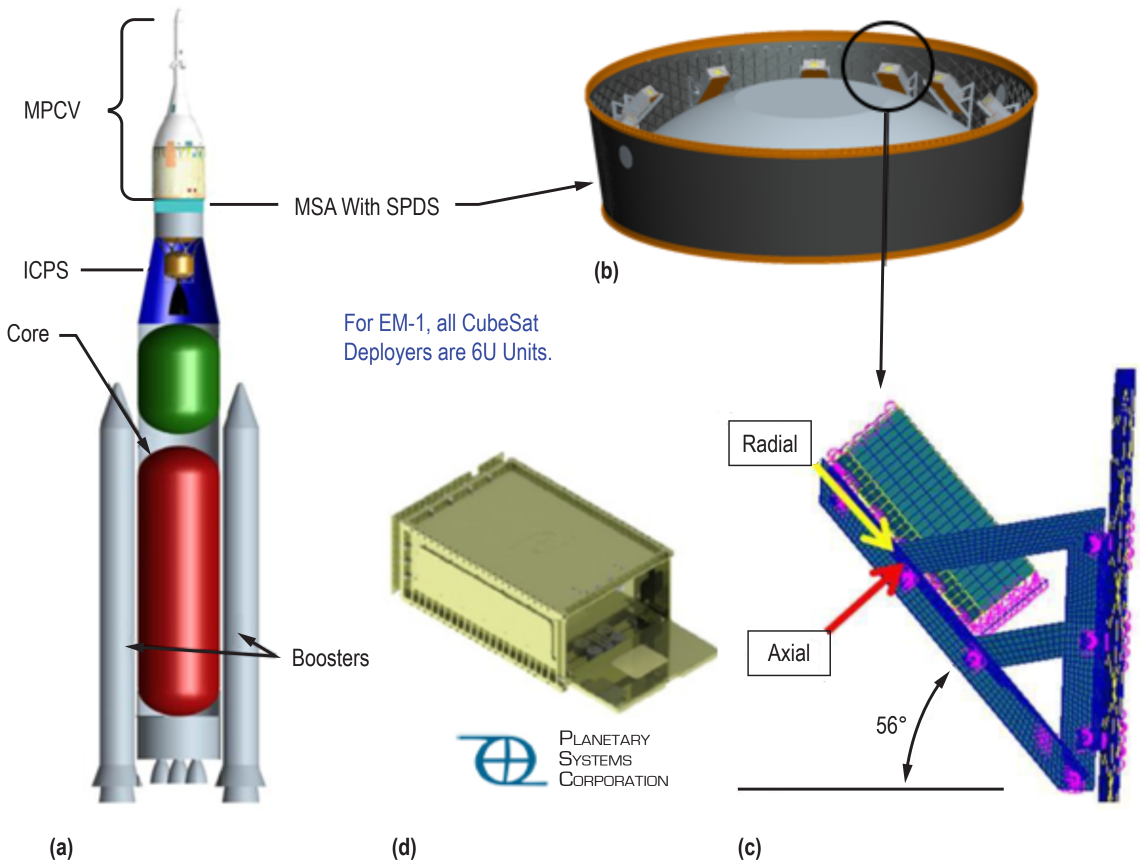

Figure 3. Rideshare accommodations for MoonBEAM: (a) SLS EM-1 configuration, (b) SPDS showing placement of the CubeSat dispensers, (c) mounting angle for the dispenser, and (d) Planetary Systems Corporation dispenser that is being used on SLS EM-1.

7. SPACECRAFT DESIGN

Details for the spacecraft design are presented below, and include descriptions of the overall configuration as well as subsystems. Each subsystem section contains a description of ground rules and assumptions (GR&A) 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 (GBM) 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 as it required more volume than any other subsystem. The propulsion tank size and shape was 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 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.

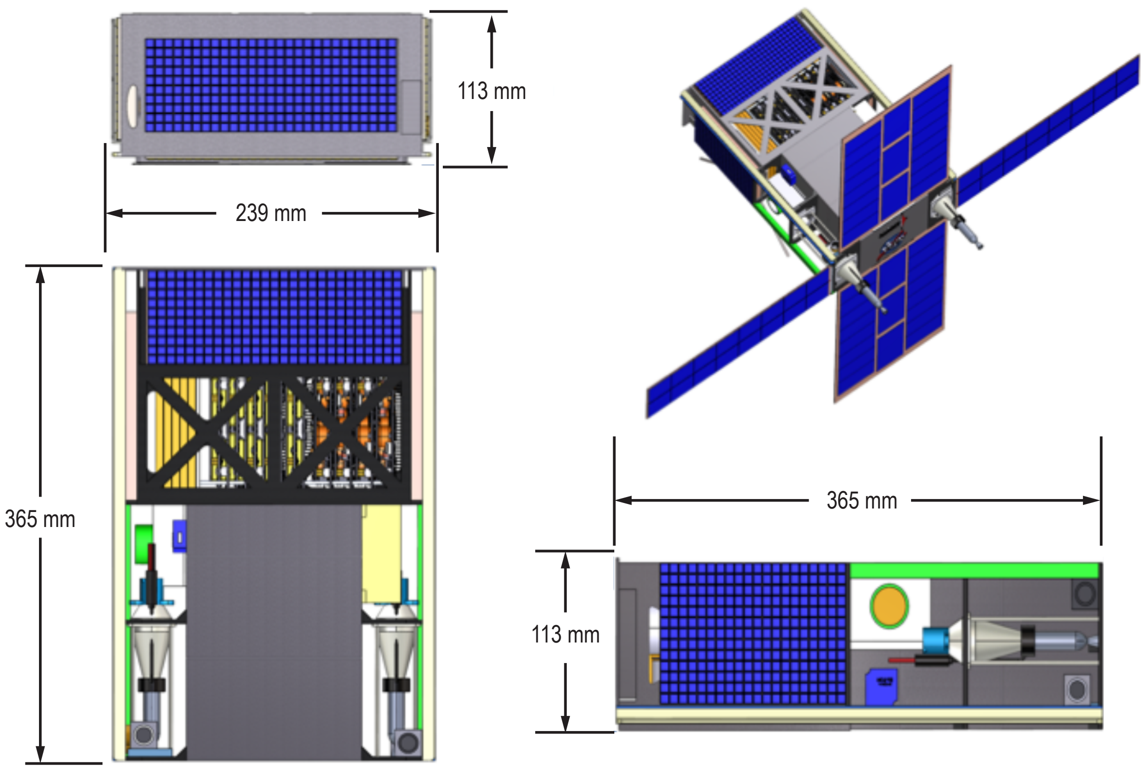

Figure 4. MoonBEAM spacecraft layout showing the stowed and deployed configurations.

7.2 Mass Properties

The MoonBEAM mass rollup is listed in table 6, 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 American Institute of Aeronautics and Astronautics (AIAA) standsards[2] for mass contingency, the total predicted mass of 13.99 kg just squeezes under the 14 kg limit. Most components used in the design have a high technology readiness level 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. Regarding power, the spacecraft was designed with a 20% margin, a value that is slightly lower than would normally be desired at this level of design analysis. Perhaps additional mass can be saved from other subsystems, such as structures, that could be used for additional battery mass. More detailed discussions of power and propulsion subsystem design and recommendations are in the appropriate subsystem sections below.

Table 6. Master equipment list.

Subsystem	Basic Mass (kg)	Growth (%)	Predicted Mass (kg)
Structures	3.88	14	4.41
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	0.38	13	0.43
Total dry mass	10.09	16	11.73
Propellant	–	–	2.26
Total mass	12.35	13	13.99

Nevertheless, 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 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 ($10 \times 20 \times 30$ 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 GRAs used in the ADCS design are listed in table 7. 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 7. 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
Communications 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 10°/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 (SP) 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[3] 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 8).

Table 8. μCAT detumble maneuver calculations.

Axis	Momentum (Nms)	Moment Arm (m)	Required Total Impulse (Ns)	Maneuver Time (hr)	Power Required (A-hr)*	
					25 V	15 V
Roll	0.0099	0.12	0.082	38.157	15.263	25.438
Pitch	0.0204	0.12	0.17	78.558	31.423	52.372
Yaw	0.0265	0.12	0.221	102.125	40.85	68.083
Totals	0.057	–	0.473	–	87.536	145.893

* 50-Hz operation requiring 10 W per thruster module; shown for 15 V 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°–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 oblate a nickel cathode (fig. 5), after which the ablated material is accelerated away through the use of a magnetic field. The advantage of this 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.

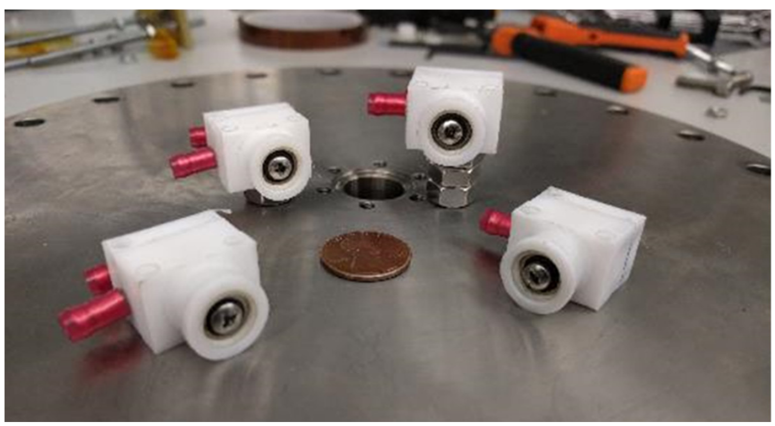

Figure 5. Microcathode arc thruster.

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 RMS) 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 (GPS) receiver was not deemed necessary at this time. The GN&C MEL for the current phase of the MoonBEAM study is shown in table 9.

Table 9. 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

Mission analysis results indicated that the MoonBEAM propulsion system must provide 369 m/s of ΔV to achieve and maintain LDRO. Due to the large ΔV budget required, LDRO proves to be a challenging target for such a small spacecraft. By reaching it, the design team demonstrates that AM and green propulsion technologies are key enablers to more capable CubeSats that can reach further into cislunar space and beyond.

7.4.1 Ground Rules and Assumptions

The propulsion system ground rules and assumptions are listed in table 10.

Table 10. Propulsion system ground rules and assumptions.

Property	Value
Total impulse	Minimum of 4,650 Ns
Mass allocation	No greater than 5 kg
Propulsive capability	Three-axis attitude control
Modular propulsion system	Design and packaged as a integrated, self-contained, propulsion module
Initial ullage volume (percentage)	2.5%–3.5%
Unusable propellant (percentage)	No PMD: 7%–10%; simple PMD: 5%–7%; cellular/lattice structure PMD: 2.5%–5%; piston tank: >1%–2.5%
Inadvertent fluid leakage	Two fault tolerance
Fabrication techniques available	Traditional methods, AM of Ti-6-4 and Inconel 718, AM of refractory metals, joining refractory metals to titanium and inconel 718
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	Approximately 15 minutes firing maximum
Thruster impulse bit	No greater than 125 mN-s
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

A previous 2016 study concluded that the mission was infeasible with conventional approaches and 6U volume limitations. The preliminary concept study baselined a 6.88-kg wet mass hybrid hydrazine (ΔV) and cold gas (RCS) propulsion system. The mission design closed from a mass perspective, but there was insufficient volume to package the conventionally fabricated components and the rather large requirement of hydrazine propellant. The concept shown of the 2016 design in figure 6 is incomplete, but does include the major propulsion system elements as well as the solar arrays, detectors, and a few avionics cards. The design exceeded the target 6U envelope by over 40%. This was primarily due to packaging spherical propellant tanks into cuboid structures resulting in inefficient utilization of available volume.

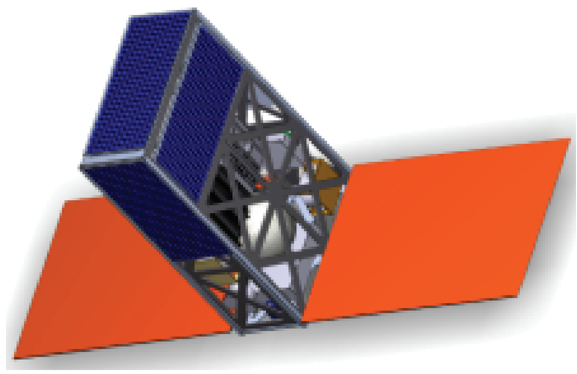

Figure 6. MoonBEAM concept 2016.

ER23 performed a followup analysis to the official ACO study in late 2016. The objective of that analysis was to determine if AM and the high-performance green propulsion system could make the mission work. The evolved system's wet mass was estimated to be 5.3 kg, carrying 20% performance margin, and fit within the allocated volume. With this new data, ER23 and EM32 proposed and won a NASA Marshall Space Flight Center (MSFC) Technical Excellence (TE) award to revisit the MoonBEAM concept study.

Departing from typical studies, the follow-up study focused on the technology enablers, rather than the science mission. Mission analysis concluded that MoonBEAM's propulsion system must provide 369 m/s of ΔV to achieve and maintain LDRO. LDRO proves to be a challenging target for such a small spacecraft. By reaching it, the design team demonstrates that AM and green propulsion technologies are key enablers to more capable CubeSats that can reach further into cis-lunar space and beyond.

This study leveraged new experience garnered by ER23 from managing the propulsion systems for the Lunar Flashlight and Near Earth Asteroid (NEA) Scout CubeSat, which are manifested on SLS EM-1. CubeSat are often more volume constrained that mass constraint. To get around this, other propulsive CubeSat spacecraft are breaking with traditional spherical propellant tank design, and moving to cuboidal shapes to maximize volume utilization. NASA's Lunar Flashlight Propulsion System (LFPS) (shown in fig. 7) and NEA Scout RCS modules are also cuboidal in design.

Figure 7. Lunar Flashlight Propulsion System.

Another objective of this study was to challenge how CubeSats with propulsion systems are integrated. One of the TE award objectives was to explore subsystem configuration and integration onto a ‘strong back’ propulsion system, much like how payloads integrate onto an evolved expendable launch vehicle (EELV) secondary payload adapter (ESPA) ring as shown in figure 8. Subsystems of the spacecraft would mount onto the propulsion system, which would form the core structure or ‘strong back’ of the spacecraft. This, in theory, would reduce the weight of the CubeSat and increase volume utilization for the propulsion system by eliminating the traditional chassis enclosure of a CubeSat. While in theory this idea has merit, its implementation was difficult for designers to adopt. Additionally, full implementation of the concept may be negated by the constraints and requirements of using commercial off-the-shelf (COTS) CubeSat subsystem cards, designed to fit a 1U platform. The concept was not fully explored during this trade study, but was discussed in detail with the study team, and agreed to as a next logical step. The follow-on work to this TM will explore this approach in detail.

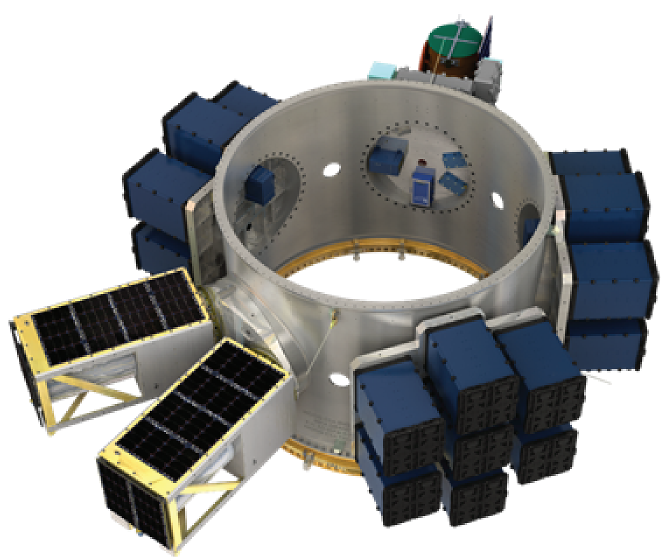

Figure 8. ESPA ring.

7.4.2 Green Propulsion

Several low toxicity, high-performance ionic liquid ‘green’ monopropellant formulas have been introduced in the last decade as advantageous alternatives to an in-space propulsion system trades space of predominantly hydrazine bases systems. The two primary ionic liquid formulas of interest are AF-M315E and LMP-103S. Both propellants exhibit greater density specific impulse (I_{sp}), improved storability and stability, and can be handled safely without Self-Contained Atmospheric Protective Ensemble (SCAPE) facilities. Despite demonstrating higher performance and lower toxicity than hydrazine, infusion of these promising propellants has been slow.

7.4.3 AF-M315E

AF-M315E is a member of a series of storable hydroxylammonium nitrate (HAN) and hydroxyethylhydrazine (HEHN) 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.

7.4.4 LMP-103S

LMP-103S is a long-term storable monopropellant blend based on a mixture of ammonium dinitramide, water, methanol, and ammonia, and has a higher performance than hydrazine. Its development began in 1997 as a cooperation between the Swedish Space Corporation (SSC) and the Swedish Defense Research Agency (FOI). The propellant can be rapidly decomposed over suitable catalysts, and the reaction products can be used to generate motive gas.

7.4.5 Propellant Tank Sizing

MoonBEAM’s propulsion system must provide 369 m/s of ΔV to achieve and maintain LDRO. Assuming a spacecraft initial mass of 14 kg and a thruster performance of 220 s, the propellant inventory of both AF-M315E and LMP-103S was calculated. It was quickly evident that AF-M315E was the logical choice for this propulsion system due to the volume requirements. For various unusable propellant values, a thermal operation regime, initial ullage volume, and thruster I_{sp} valves, it was estimated that the propellant tank needed to hold 2.26 kg of AF-M315E to meet requirements. The estimates are shown in table 11.

Table 11. Required AF-M315E propellant calculations results tabulated for various I_{sp} and ΔV values.

I_{sp}		ΔV (m/s)						
Unusable (%)	Usable (g)	200 s	205 s	210 s	215 s	220 s	225 s	230 s
10	2,034	308	316	323	331	339	346	354
7.5	2,091	317	325	333	341	349	357	365
5	2,147	327	335	343	351	359	367	376
2.5	2,204	336	344	353	361	369	378	386
0	2,260	345	354	363	371	380	388	397

7.4.6 Printing a Propellant Tank

In the fall of 2016, ER23 built and hydrostatically tested several additively manufactured cuboidal propellant tanks (fig. 9). The tanks were designed for a maximum design pressure (MDP) of 400 psia, and a burst factor of 4.

Figure 9. Printed cuboidal propellant tanks.

The analysis predicted burst pressure of the propellant tank was 1,600 psia. All of the tanks that were hydrostatically tested exceeded 1,600 psia. Most failed at over 2,000 psia, and the failure exhibited itself as a leak before burst rather than an outright failure, as illustrated in figure 10. The orientation of the failures was alight with the primary strain direction, regardless of the print orientation. This result far exceeded expectations. The measured burst pressure provides a safety factor of 5 for the propellant tank before burst. These results show the propellant tank will be safe to use at the designed operating pressure.

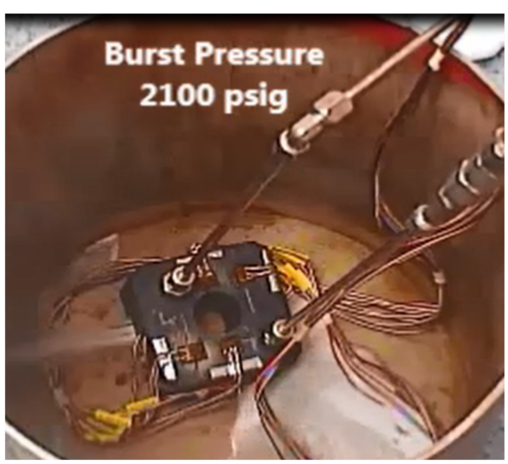

Figure 10. Printed titanium tank bursting.

This work only served to demonstrate the potential for AM applied to propellant tanks. The tank was designed to be incorporated into a 3U CubeSat propulsion module and is still being used for ground testing applications. Future iterations of this tank concept will incorporate many new features, including improved integration means, internal flow passages, improved printability, and mass optimization techniques.

The LFPS, another green propellant micropropulsion system, pushes on the boundary of the CubeSat propulsive capability and the limit of traditional manufacturing methods to achieve that capability. In place of fuel lines and ducts, passages are drilled into the thick walls of the prop module. Whenever a passage is required to change direction, it is necessary to drill intercepting perpendicular holes and welding a plug in to place to seal the passage. Figure 11 illustrates this process. Additive manufacturing allows passages to be printed into the tank walls. Furthermore, it allows for optimization of the wall thickness, reducing structure mass, pressure drop, and unusable propellant. This example illustrates the benefits of AM to small spacecraft.

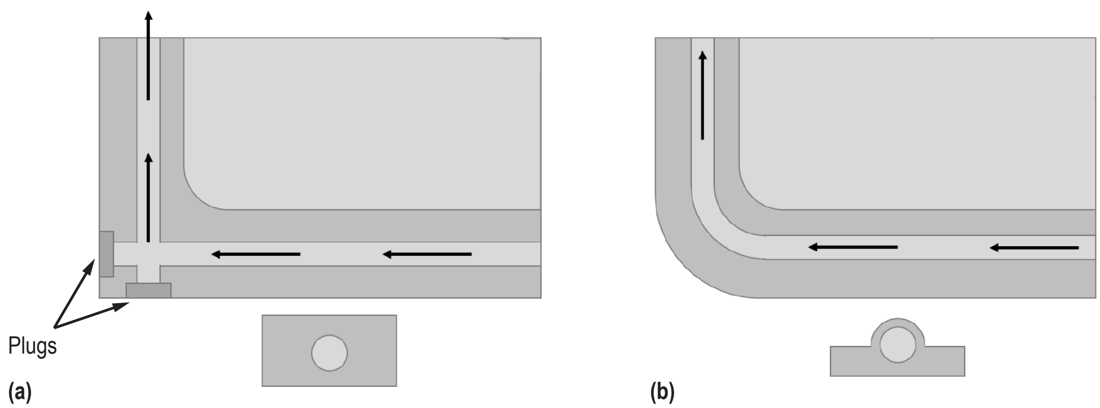

Figure 11. Integral tank wall propellant passage bends: (a) Traditional manufacturing and (b) AM.

Further exploring the ‘printability’ of small integral flow passages, ER23 designed and printed several representative builds on an EOS M 290. The one of relevance to this section was a propellant tank wall segment with integral flow passages. Shown in figure 12, the passage diameters varied from 2 to 0.3 mm in diameter. The objective was to determine the minimum flow passage diameter that can be reliably printed and cleared of powder, and determine the maximum flow passage diameter that can be printed without requiring support material. Only circular flow passages were considered in the build, though the freedom of AM allows for unique geometries to facilitate printability and performance.

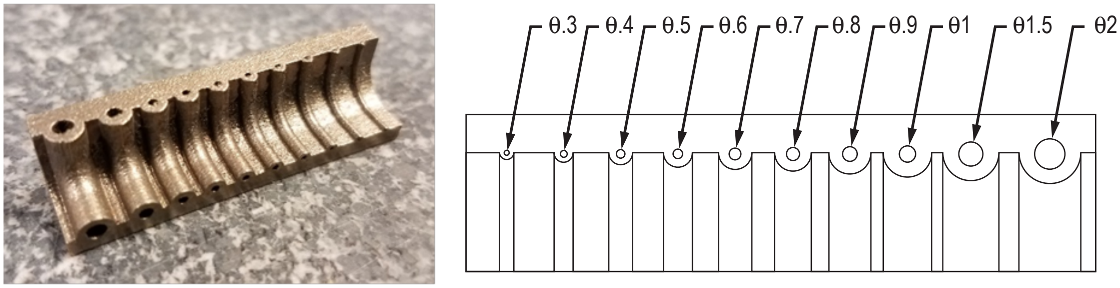

Figure 12. Integrated flow passage—wall segment build.

Printed feature tolerances can be maintained up to 0.001 to 0.002 inch, but features below 0.030 inch become an issue. It was observed that holes ‘shrink,’ and the tolerance on these small features can vary from 0.002 to 0.008 inch. But, these features can be cleaned up, if accessible. Flow passages 4 mm in diameter and smaller can be printed without requiring support material. However, flow passages smaller than 0.7 mm in diameter become an issue to clear. Flow passage diameters between 2 and 0.7 mm were achieved without issues. Furthermore, several processes are available to improve the flow passage interior surface smoothness, since the roughness around these geometries will not be ideal for fluid flow. Abrasive flow machining (AFM), also known as abrasive flow deburring or extrude honing, is an interior surface finishing process characterized by flowing an abrasive-laden fluid through a work piece. AFM smooths and finishes rough surfaces, and is specifically used to remove burrs, polish surfaces, and form radii. The nature of AFM makes it ideal for interior surfaces, slots, holes, cavities, and other areas that may be difficult to reach with other finishing methods.

The MoonBEAM propulsion system will utilize AM to build the propellant tank, incorporating internal flow passages to maximize the utility of the structure. The design approach is simple. The propellant tank will first be optimized for volume. Recall that CubeSat are often more volume constrained than mass. Then, the tank will be topology optimized to exclude mass where it is not needed to manage expected loads. Figure 13 shows how the original part file was optimized for mass. It was assumed that the propellant tank would have an MDP of 400 psia. The designer applied an MDP ×2 to arrive at a burst pressure, and optimized the design to survive 800 psia.

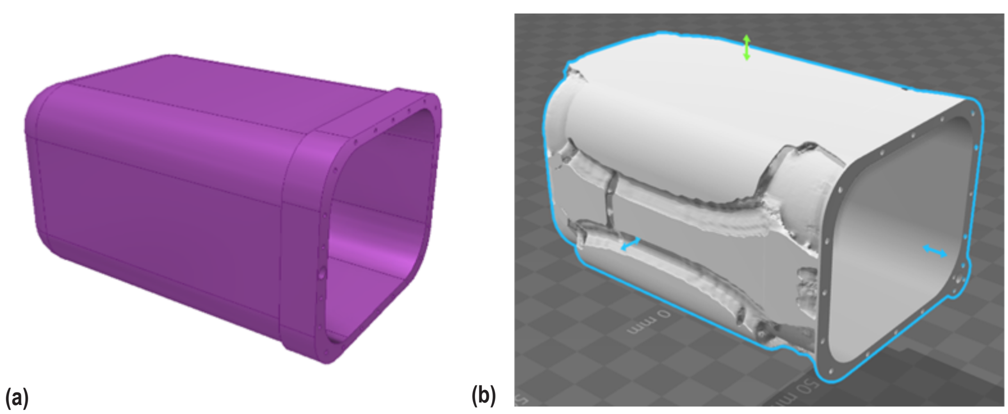

Figure 13. MoonBEAM propellant tank: (a) Original part and (b) topology optimized part.

This optimized part was 40% of the mass of the original part. Though this part above is representative of the volume of the MoonBEAM propellant tank, it did not host the other design elements that are expected to be incorporated into the final design. But, it does highlight the potential that these tools have to evolve how propellant tanks are envisioned, designed, and built now.

7.4.7 Thruster

The thrusters will use a green propellant, AF-M315E high-performance monopropellant, which is denser than hydrazine. AF-M315E, a low toxicity ionic liquid sometimes called a 'green propellant,' has nearly >50% density-I_{sp} than hydrazine, leading to smaller, more capable propulsion systems. Despite demonstrating higher performance and lower toxicity than hydrazine, infusion of these promising propellants has been slow. These benefits come at a price. Both AF-M315E and LMP-103S have much higher combustion temperatures that hydrazine, and as a result, traditional materials are not well suited. Designers must now consider using refractory or other exotic high-temperature alloys that are better suited for green propellants.

The refractory metal group, commonly defined as niobium, molybdenum, tantalum, tungsten, and rhenium, are a classification of extraordinarily creep and wear resistant metals with melting points above 2,000 °C. Because of their high melting points, there are limited practical fabrication methods. Powder metallurgy is the most prevalent, but AM techniques are gaining in prominence. MSFC is investing in parameter development for 'printing' refractory metal components. In the last year, several Small Business Innovative Research proposals were received, proposing to mature refractor metal printing and apply it to thruster component fabrication.

Three thrust levels were considered for MoonBEAM: 100 mN, 1 N, and 5 N. The notional thrusters were traded based on power required versus power available, heat produced, and estimated life based on throughput, and whether or not attitude control and momentum wheel desaturation could be achieved by other means and the thrusters alone.

Based on these trades, it was determined that two 1-N thrusters, as shown in figure 14, was the appropriate solution. Each 1-N thruster was estimated to required 15 W to preheat, and would produce close to 60 W of radiated heat and 5 W of conducted heat into the spacecraft. Broken down by maneuver, the longest propulsive maneuver for a 1-N thruster was estimated to be 18 minutes. That is a metric achievable by a few existing thrusters today. The thruster burn times per maneuver are shown in table 12.

Figure 14. Notional 1-N thruster.

Table 12. Thruster burn times by maneuver.

Spacecraft Mass (kg)	ΔV (m/s)	I_{sp} (s)	Δm_{prop} (kg)	Impulse (Ns)	Thrust (N)	Time (s)	Time (m)
14	5	220	0.03	70	2	35	0.58
13.97	5	220	0.03	70	2	35	0.58
13.94	162	220	1.101	2,175	2	1,067	18.12
12.93	155	220	0.9	1,933	2	967	16.11
12.03	5	220	0.03	90	2	30	0.5
12	10	220	0.06	120	2	60	1
11.95	–	–	–	–	–	–	–

The new baseline MoonBEAM propulsion system configuration, illustrated in figure 15, includes two 1-N AF-M315E thrusters to execute propulsive maneuvers, an additively manufactured propellant tank that stores 2.26 kg of AF-M315E propellant, a postlaunch pressurization system or 'gas generator,' and may use either a traditional propellant management device, a diaphragm, or a piston. The system level schematic is shown in figure 16. The design also incorporates μCAT electric microthrusters (not covered in this section) for momentum unloading, eliminating the need to carry additional propellant for unloading the reaction wheels, as well as eliminating the plumbing necessary for routing propellant.

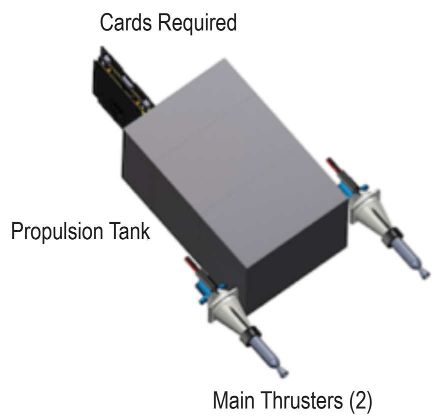

Figure 15. MoonBEAM propulsion system major elements.

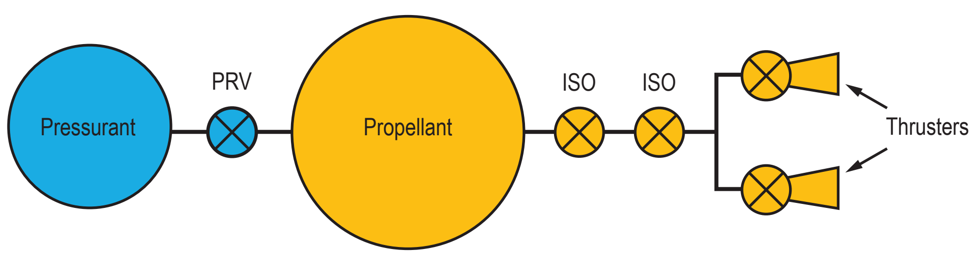

Figure 16. MoonBEAM propulsion system schematic.

Per SLS requirements, all secondary payloads are required to guard the launch vehicle and the adjacent hardware within the MSA, against inadvertent fluid release from a pressurized system inside of a closed volume. The SLS Program Safety & Mission Assurance confirmed that, as a general rule, pressurized systems that are two fault tolerant to the release of fluid through controlled release devices do not require additional analysis beyond the analysis done for pressure system requirements. For MoonBEAM, the two thruster valves and two redundant isolation valves accomplish dual fault tolerance to inadvertent fluid release.

The propulsion mass estimation list, show in table 13, shows the estimated mass for each major component at the applied mass contingency per AIAA S-120.[4]

Table 13. Propulsion system MEL.

Component	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
1-N AF-M315E thruster	2	0.1	0.2	25	0.25
Thruster/valve driver board	1	0.05	0.05	25	0.06
μCAT thruster module	4	0.02	0.08	30	0.1
μCAT controller board	1	0.05	0.05	30	0.07
Propellant tank (included in structures MEL)	1	–	–	–	–
Total	–	–	0.38	13	0.43
AF-M315E propellant	1	2.26	2.26	0 (accounted for in estimate)	2.26

7.4.8 Future Work

These goals, determined by subsystem and AM experts to be achievable, will be put into practice this year, as one of the outcomes of the study is a roadmap and set of performance goals that the AM experts will go and try to meet by actually printing these components.

A notional piston tank concept is shown in figure 17. A piston would allow for very high usable propellant fraction by ensuring that all propellant is consolidated in one volume and not allow dispersement through the tank volume. The ullage space on the back side of the piston is also usable volume for packaging components such as valves, sensors, and controller electronics. ER23 has experience with the design, manufacturing, and testing of piston tanks on this scale. The testing series was very successful, with good expulsion demonstrated and low friction between the piston and the tank wall. It was observed that the pistons' length-to-diameter ratio would be more ideal if it exceeded 1.5:1. The use of a piston drives design and manufacturing requirements. The contacting surfaces of the piston and the tank wall must be very smooth to seal and to slide. Recommended surface finishes are √32 or better. The piston must be installed inside the tank, and the surfaces must be improved over the expected printed surface finish, usually greater than √300. So, the tank must be designed with a bolted or welded flange to complete the pressure structure.

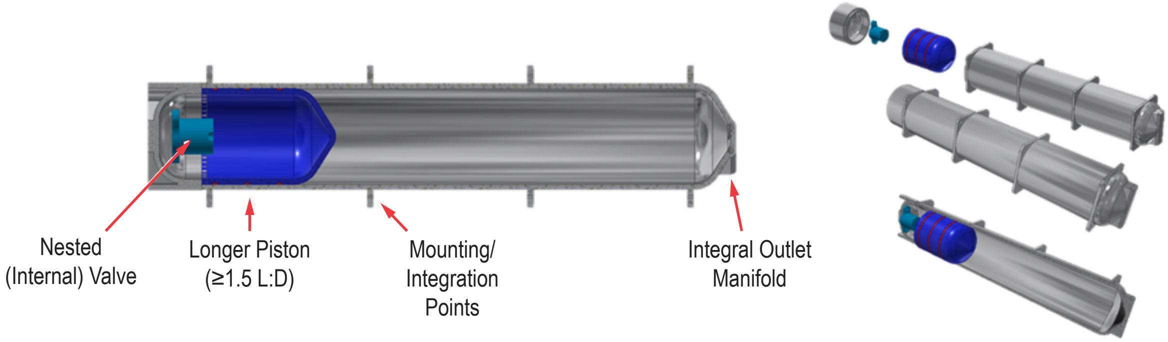

Figure 17. Notional piston tank concept.

ER23 is currently building an AM 1-N thruster in-house (fig. 18), as part of a NASA Innovative Kick Starter award and the TE award that funded this follow-on study. The detailed design and development process of that thruster will be captured in a separate NASA TM being drafted in parallel to this one.

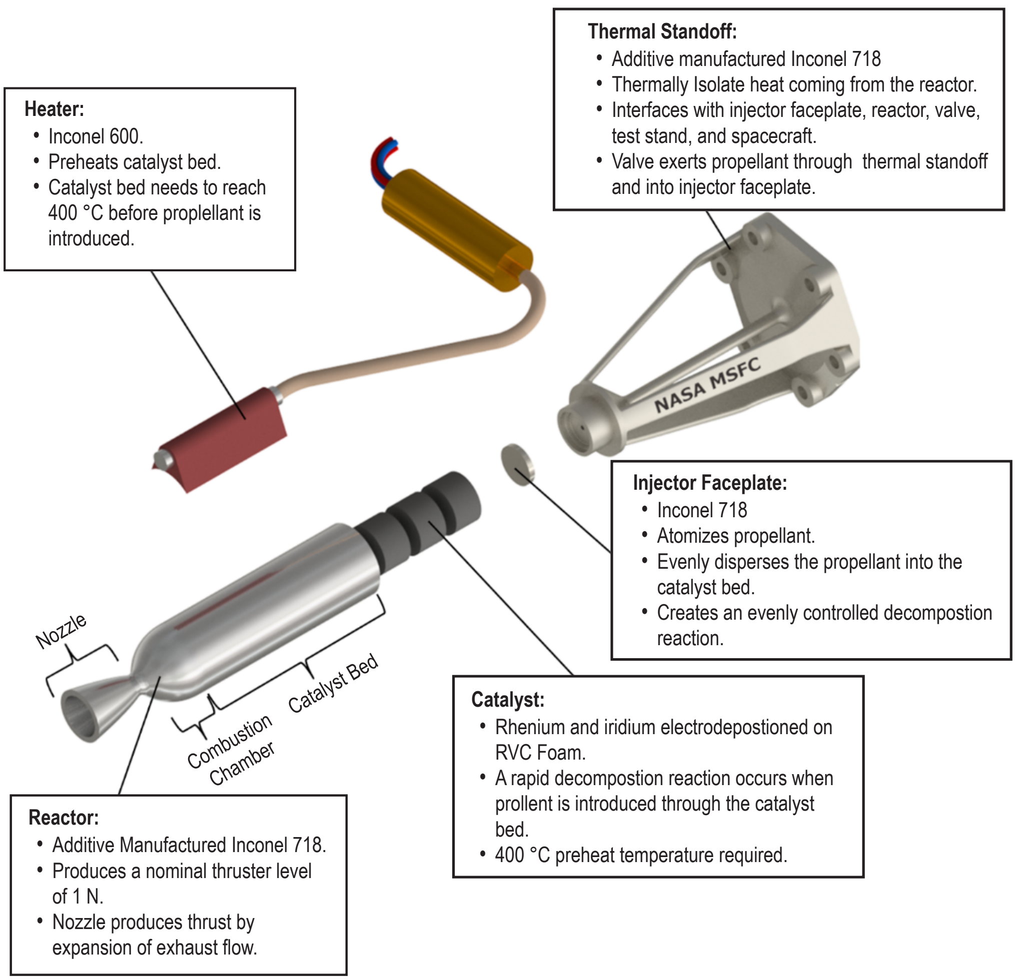

Figure 18. ER23 printed 1-N green prop thruster.

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 14.

Table 14. Avionics subsystem ground rules and assumptions.

Category	Value
C&DH	
Instrument data	Bused to processor or grouped into zones on backplane
Data bus protocols	I2C, RS232, RS422
Redundancy	$2\times$ 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. 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. In addition, the stack will include GN&C and propulsion boards as needed. See those sections of this TM for information on the GN&C and propulsion boards.

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 COTS boards are available and are used in unmanned aerial vehicles, drones, and missiles, some with flight heritage (see fig. 19).

The spacecraft SBC was baselined on Innovative Solutions in Space (ISIS) iOCB 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 pre-phase A capability, mass, and power. They are ruggedized to MIL-STD-202 with a high operating temperature range and high shock and vibration levels.[5] 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 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 a 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-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 (fig. 19). The phase 1 link budget analysis was found to be still applicable for 250 MB/day from an LDRO orbit. It assumed a UHF band (430–440 MHz) uplink and downlink, at a distance of 1,000,000 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. Locations of the avionics components within the CubeSat are shown in figure 20.

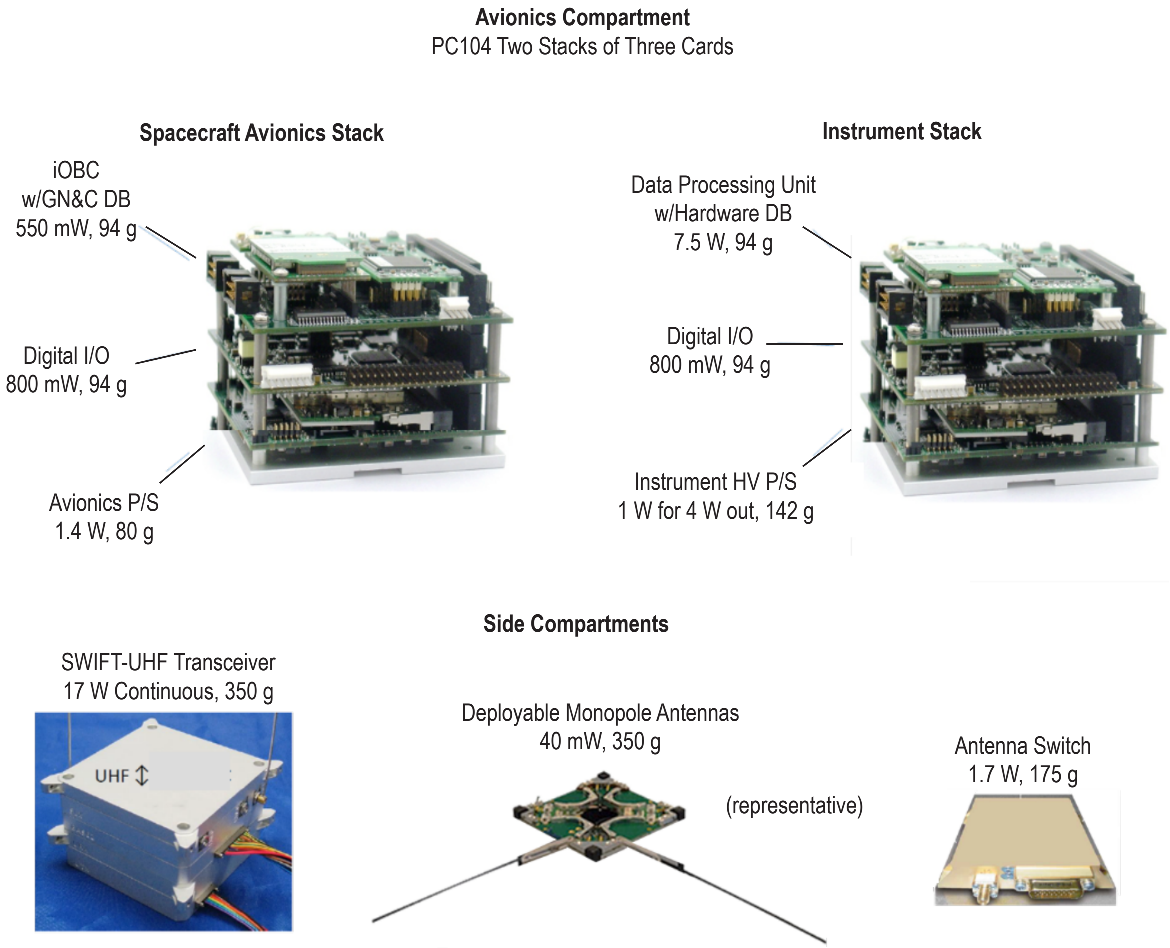

Figure 19. Spacecraft avionics stacks and communications components.

For a continuous 4-kbps transmission to ground, it was assumed crowd-source ground stations would be utilized for data collection, as was assumed in the 2016 MoonBEAM study. That would be a consortium of NASA facilities, universities, and interested amateurs worldwide. It is further assumed 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.).

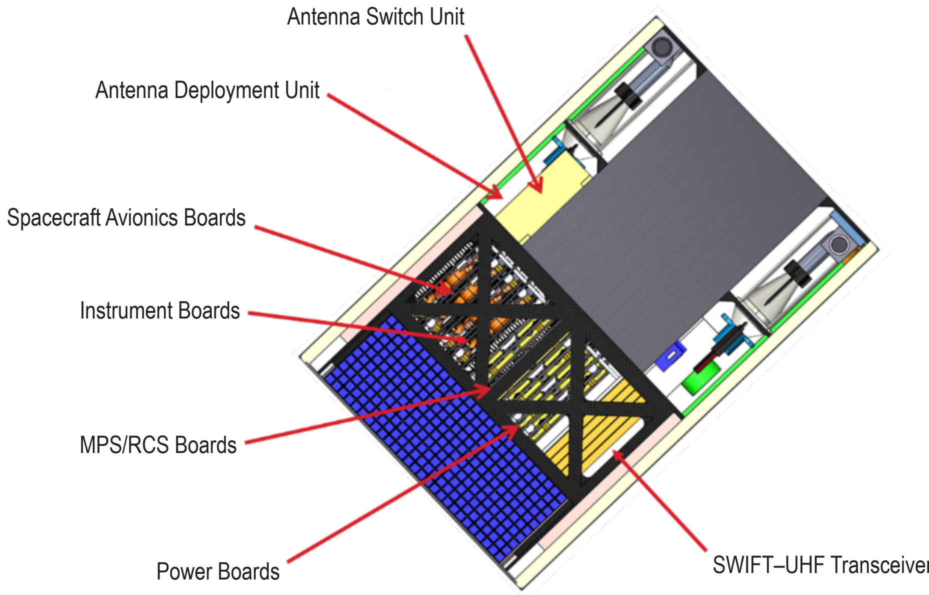

Figure 20. Avionics component locations within the bus (total stack size: 10 boards, PC104 cards).

With the spacecraft Sun pointing for solar array power and the gamma array detectors deep space pointing, it was necessary to included two sets of antennas (fig. 21). For when the Earth is on the solar array side of the spacecraft, two one-half-wave dipole antennas are 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 (fig. 18). A deployment mechanism from ISIS CubeSat is suggested for the monopole antenna deployment. 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.

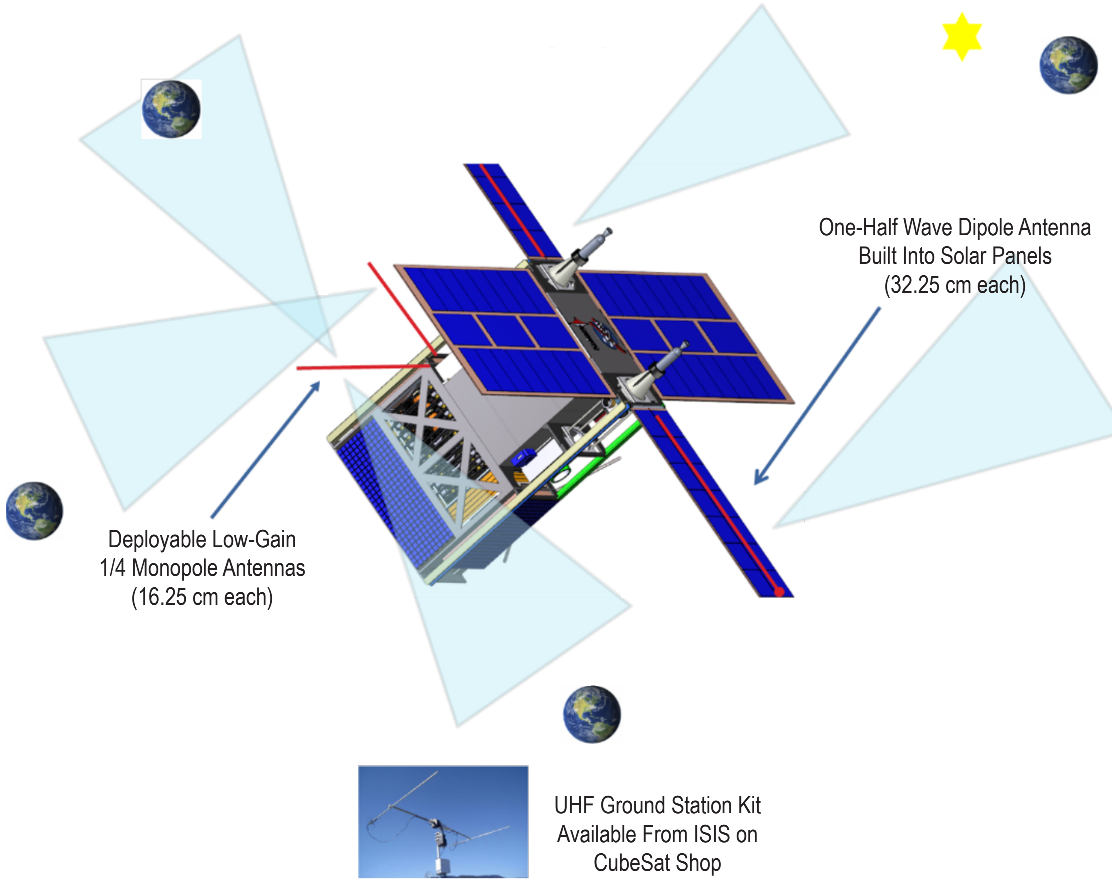

Figure 21. Spacecraft avionics antenna locations and Earth field-of-view layout.

Figure 22 shows a block diagram of the Avionics and Science Instrument systems of the spacecraft. The diagram is baseline off the Fermi gamma-ray monitor spacecraft block diagram, where the instrument side is very similar. 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 spacecraft power and GN&C components for completeness; refer to those sections of this TM for details. Table 15 is the Avionics MEL for the study.

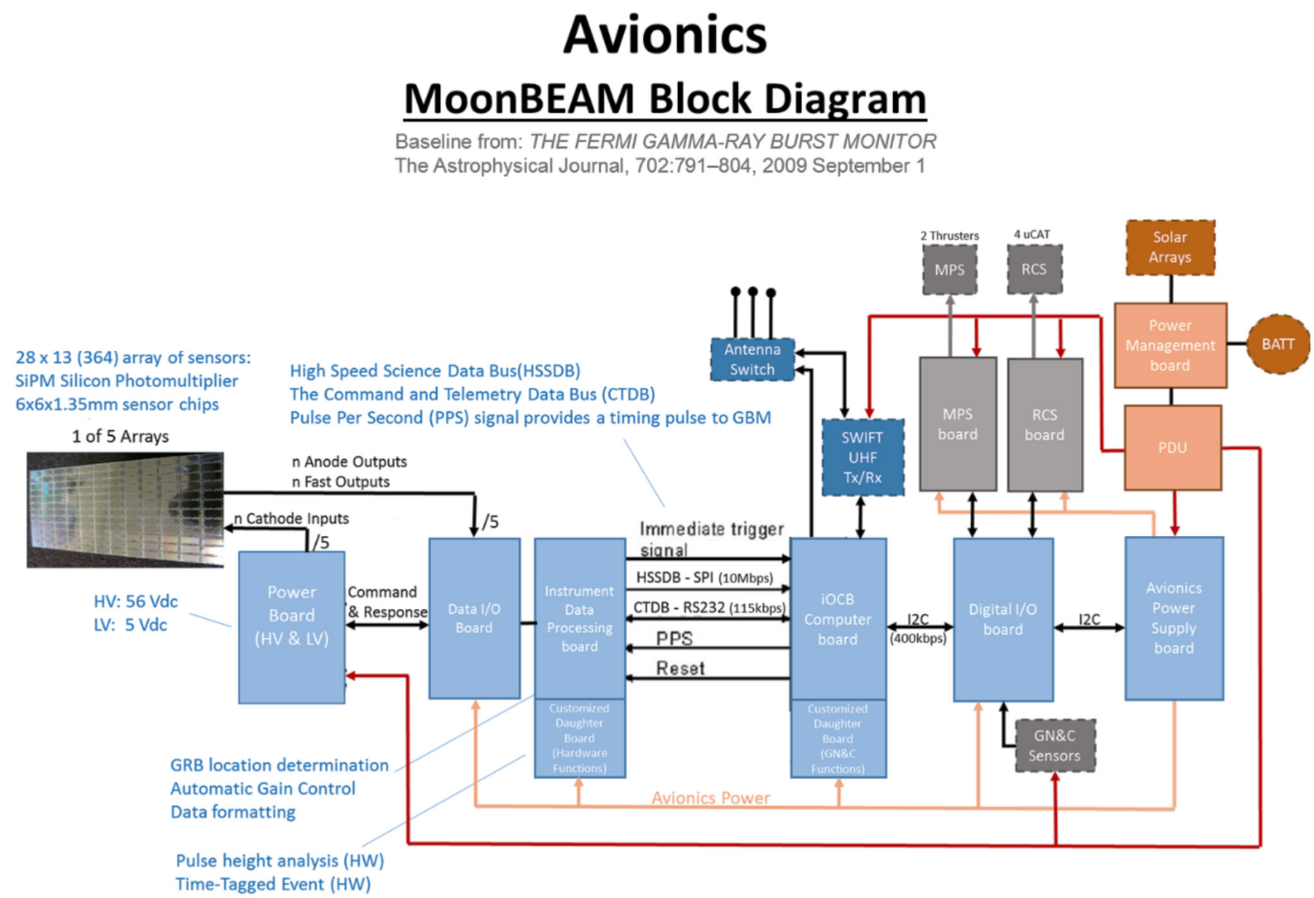

Figure 22. Spacecraft avionics and science instrument system MoonBeam block diagram.

Table 15. 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 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 a finite element model.
- MSC Nastran—Solver used for analysis of all nonprinted primary structures.
- Simmulia Abaqus—Solver used for analysis of all printed primary structures.
- Collier Research Hypersizer—Used in conjunction with MSC Nastran and Simmulia Abaqus to optimize all MoonBEAM primary structures.

Structural analysis and optimization of the MoonBEAM 6U CubeSat was accomplished in two parts due to varying requirements for pressurized and unpressurized structures.

7.6.1 Analysis Phase I

The first analysis phase focuses on the printed titanium fuel and pressurant tank walls, which share a common bulkhead wall. Both the fuel and pressurant chambers are required to show positive strength margins with an internal pressure of 400 psi and an ultimate strength factor of safety of 2.5. Figure 23 depicts a half-symmetric representation of the MoonBEAM printed tank structure. This structure was analyzed using Simmulia Abaqus with large displacements enabled. Due to nonlinear behaviors exhibited by the flat tank walls when pressurized, a large displacement, nonlinear analysis approach is required.

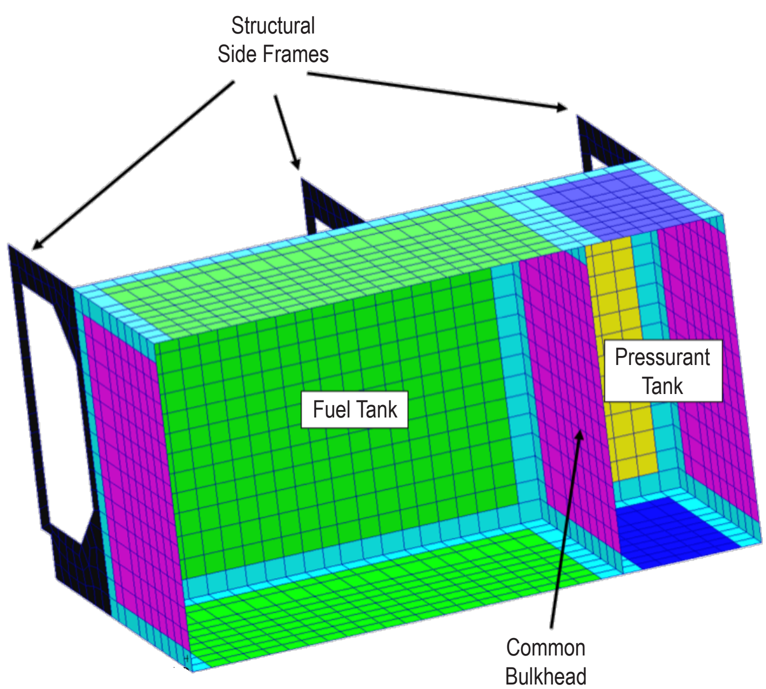

Figure 23. MoonBEAM printed tank strutures (half symmetric).

Three load conditions were run to optimize tank sizing with a trivial constraint set applied to the overall model. Figures 24 and 25 depict the full MoonBEAM finite element model and a section cut of the pressurized tank structure:

- Load case 1—Fuel tank pressurized to 400 psia, pressurant tank at 0 psia.
- Load case 2—Fuel tank at 0 psia, pressurant tank pressurized to 400 psia.
- Load case 3—Fuel and pressurant tanks pressurized to 400 psia.

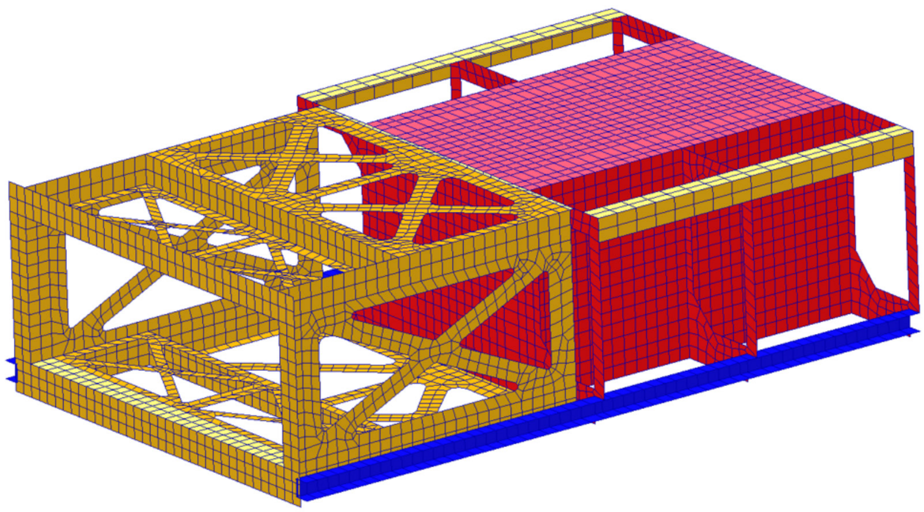

Figure 24. MoonBEAM primary structure (tank structure shown in red).

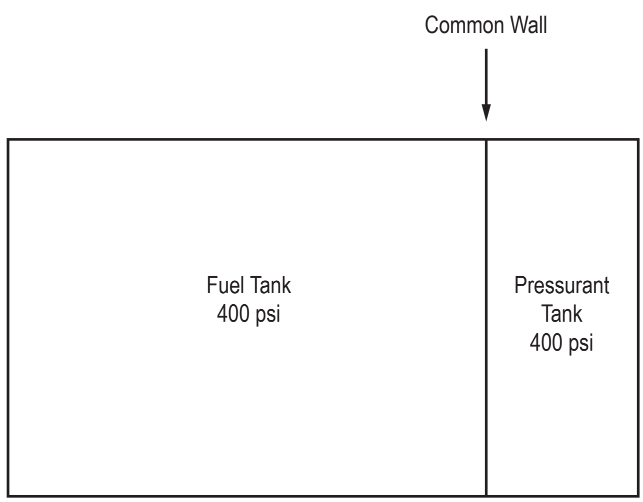

Figure 25. Fuel/pressurant tank pressurization (load case 3 shown).

Tank properties assume printed titanium 6AL-4V. They were provided by the MSFC Materials Group (Doc. Ref. ESSSA-FY13-2015) and are shown in table 16 (see electron beam melting (EBM) tested groups mean):

- Tensile strength = 154.94 ksi
- Yield strength = 148.57 ksi
- Modulus of elasticity = 17.05 Msi
- Fracture elongation = 3.97%.

Table 16. EBM properties of Ti-6Al-4V as tested.

	UTS (ksi)	YS (ksi)	Elongation (%)	Reduction of Area (%)
EBM tested groups mean	154.75	148.57	3.97	5.17
MIL-HDBK-5J Tensile Properties	135	125	10	25

An interesting characteristic of the printed titanium is that it has an ultimate and yield strength that exceeds that of typical Ti-6AL-4V, as found in MIL-HDBK-5J.[6] The down side for printed titanium is a less ductile reduced elongation allowable of 3.97% as compared to the more robust 10% for standard Ti-6AL-4V in MIL-HDBK-5J.

Model sizing after several analysis iterations produced results as seen in figure 26. The flat faceted faces of the tank structure yield thick titanium walls as would be expected. Note that the side frames are subjected to further sizing optimization using inertial loading in phase II of the analysis process.

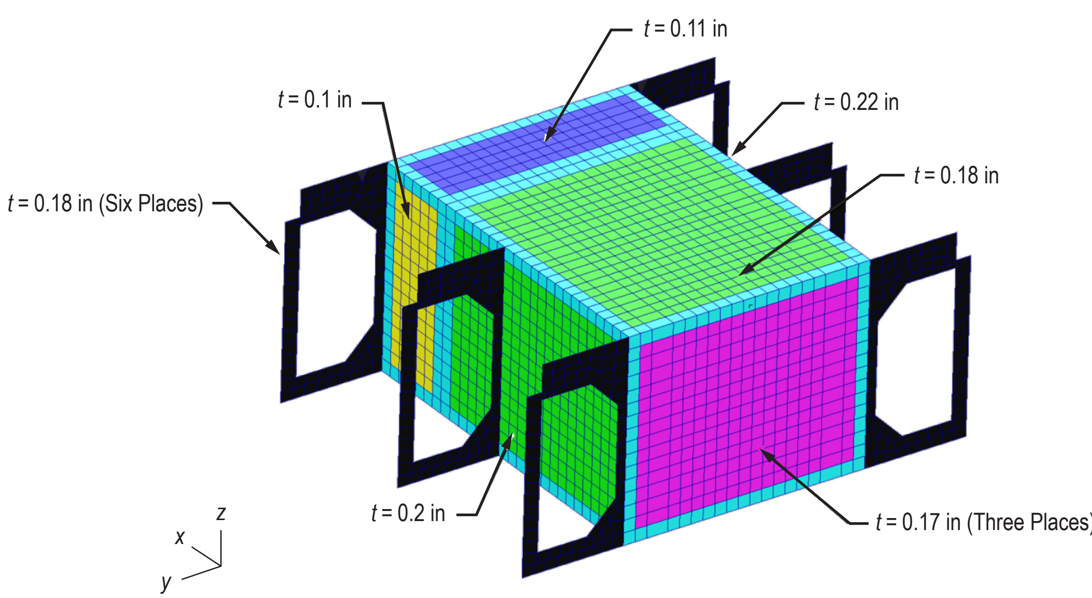

Figure 26. Optimized tank and side frame final sizing.

7.6.2 Analysis Phase II

The second analysis phase focuses on the remaining MoonBEAM structures which include side frames, mounting rails, upper stiffening L-sections, and a subsystem mounting box. Figure 27 depicts the remaining structure sized in this section. Analysis is performed using MSC NASTRAN SOL 101 as a solver with HyperSizer as an optimization tool.

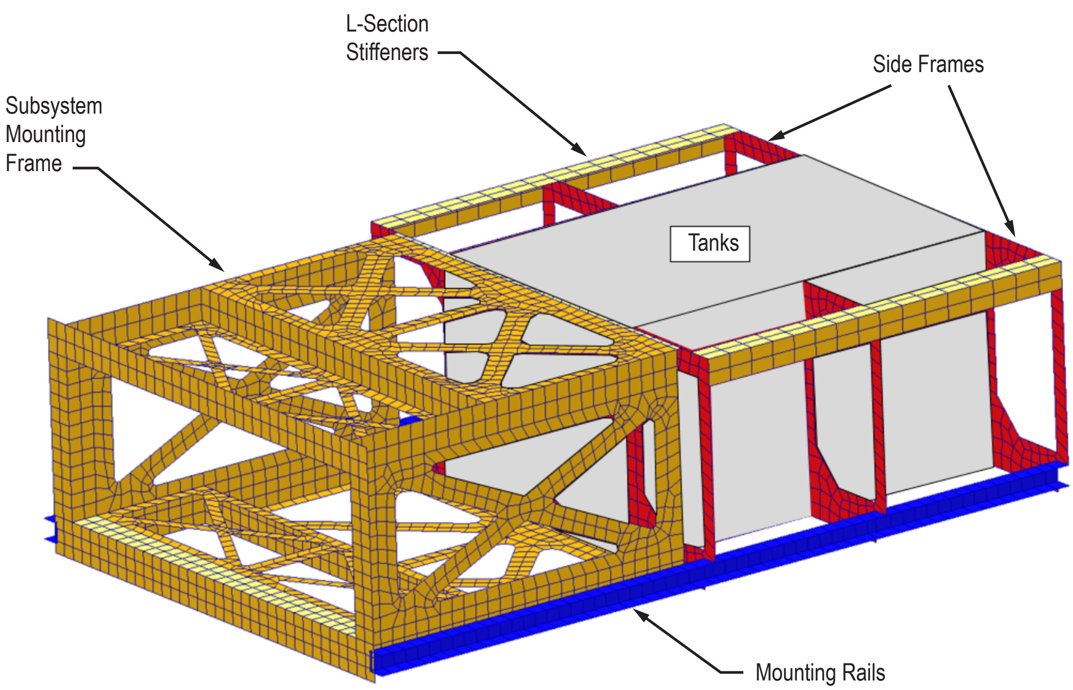

Figure 27. MoonBEAM phase II structural analysis components.

All structures sized in phase II are aluminum with the exception of the tank side frames, which are titanium. The following are material specifications for the various components:

• 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 were fabricated according to specifications dictated by the CubeSat Deployment Box vendor. Assumed material properties follow:

• 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 for the phase II analysis are shown in figure 28. 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 Environments Verification Specification load graphs. Note that inertial loads are very high due to harsh dynamic environments on the walls of the SLS mass stage adapter. The following 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).

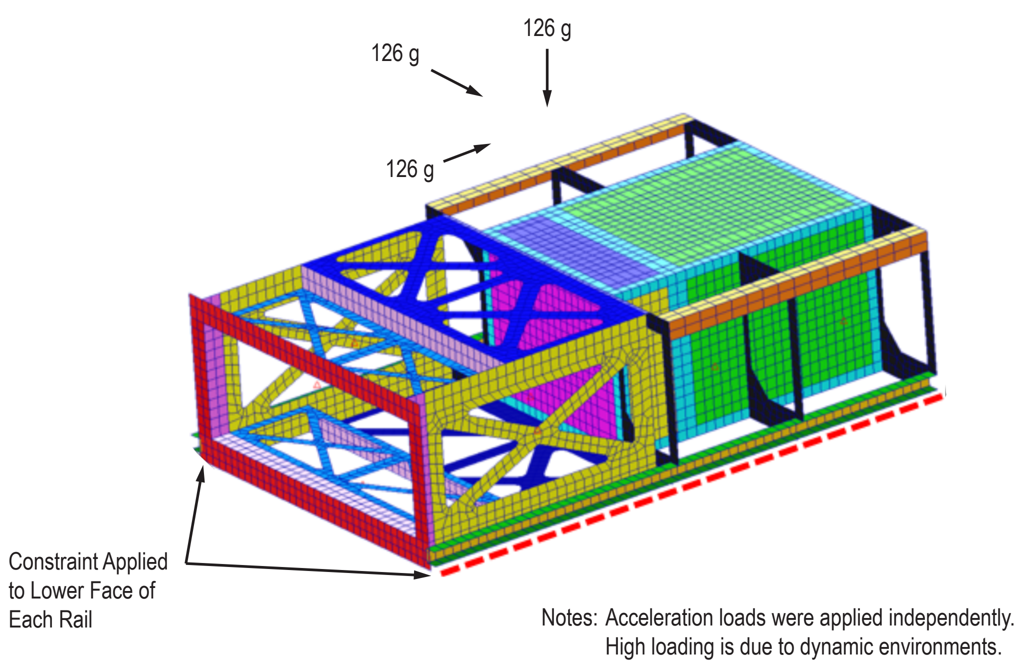

Figure 28. MoonBEAM structural optimization (phase II analysis).

Final sizing for structures optimized in the phase II analysis are shown in figure 29.

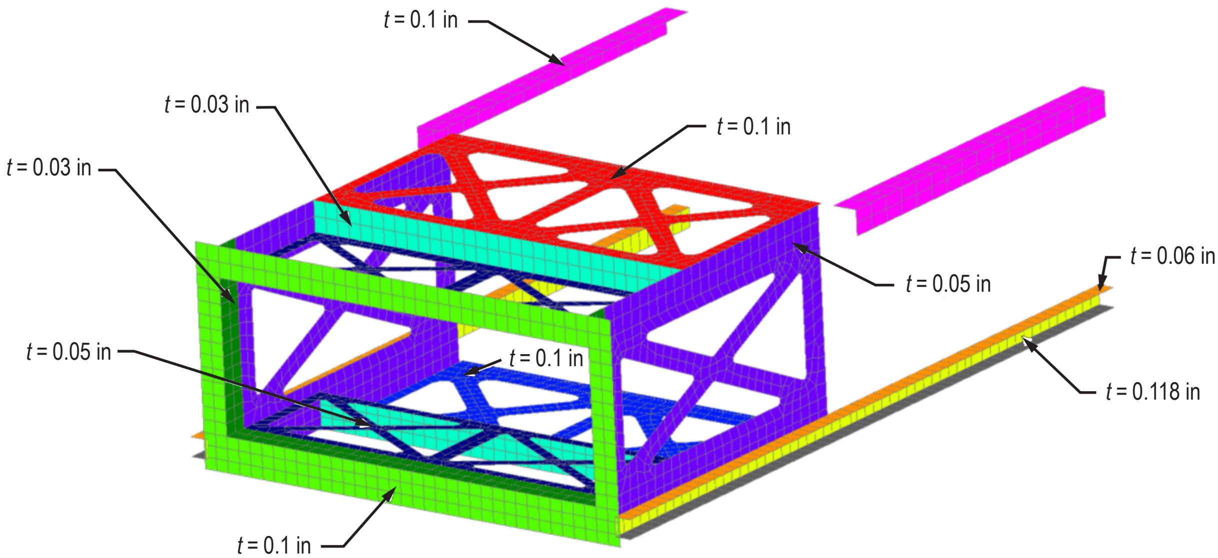

Figure 29. Phase II analysis sizing results (aluminum structure).

The ground rules and assumptions used in the structural assessment are listed in table 17. 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.[7] Additionally, a requirement for pressurized tank structures was included that stipulates a factor of safety of 2.5 for the fuel tank and pressurant tank structures.

Table 17. Structures subsystem ground rules and assumptions.

Category	Value
Dispenser interface	Continuous rail tab as defined in Planetary Systems Corps. ICD for 6U CubeSat dispenser
General	Primary structure will be designed to meet minimum strength requirements as stated in NASA-STD-5001B
Load cases	Basic CubeSat structure will be designed using GEVS ascent acceleration loads for small SLS payloads Pressurized tank structures will be designed to withstand 400 psi with an ultimate factor of safety of 2.5
Factor of safety for metallic materials	Ultimate factor of safety = 1.4 Yield factor of safety = 1.25 Tank pressurization proof load factor = 1.5
Secondary structures	Secondary structure mass is assumed to be 10% of the combined subsystem mass

The final structural masses are shown in the MEL in table 18.

Table 18. Structures subsystem MEL.

Component	Qty.	Unit Mass (kg)	Total Mass (kg)	Contingency (%)	Predicted Mass (kg)
Tank structure (printed Ti)	1	2.908	2.908	15	3.34
Other structure (aluminum)	1	0.616	0.616	15	0.71
Secondary structures	1	0.352	0.352	0	0.352
Total			3.88	14	4.41

7.7 Power

The power system performs three functions for the spacecraft: generation of electrical power from sunlight, storage of electrical energy for those time periods when there is no sunlight, and conditioning and power switching to individual loads (turning each load on and off). The ground rules and assumptions that guided the design of the power system are listed in table 19.

Table 19. Power subsystem ground rules and assumptions.

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	Lunar distant retrograde orbit (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. They are:

(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 20. 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 21.

Table 20. Power requirements by operation mode and system.

Load	Standby (W)	Initialization (W)	Science (W)	Preburn (W)	Burn (W)
Avionics	29	29	29	29	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

Table 21. Power operations sequence.

Operation	Power Required	Power Generated	Balance	Comment
Initialization	41.3	–	–41.3	Battery provided (37 minutes)
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	–	Powered by battery (24 minutes 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 of 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, 59.3 W of power will be required. 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 and 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 30 shows the major elements of the power subsystem. The solar arrays and power electronics are taken from an iSat, a CubeSat with an electric thruster using iodine as the propellant. The solar arrays are layed out on the same substrate as those for iSat, but are configured differently. Instead of the three 20×30 cm panels on iSat, the iSat body-mounted panel is divided into two 10×30 cm panels (making four panels altogether) and arranged as shown.

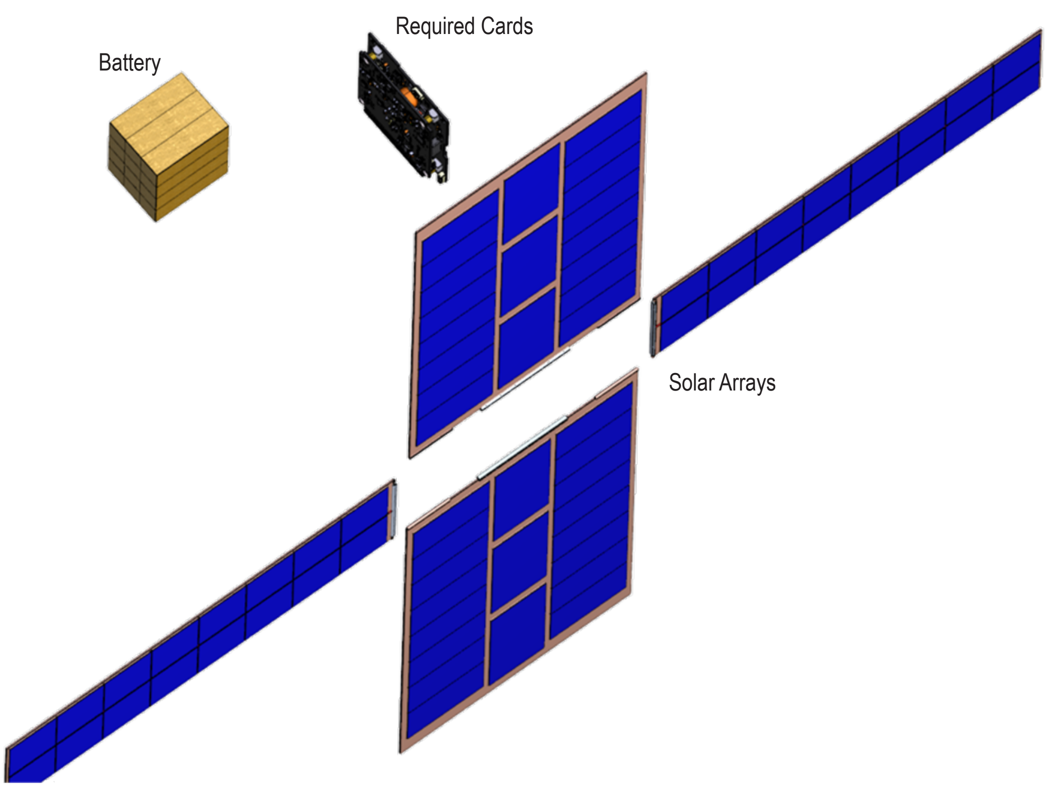

Figure 30. Power subsystem elements.

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 consists of two boards—a power management board which regulates the solar array and provides charge control for the secondary battery and a power distribution board which switches all loads. These are taken directly from the iSat design as well. Table 22 details the mass breakdown of the power system.

Table 22. Power subsystem elements.

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 to use high Technology Readiness Level technologies like multilayer insulation 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 23. 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 23. 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 24. The masses for coatings and multilayer insulation 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 24. Mass estimate for MoonBEAM thermal control system.

	Qty	CBE Unit (kg)	CBE Total (kg)	Contingency (%)	MEV (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

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.

Mitigation of the heat conducted from the engine into the spacecraft (the engine 'soak-back') is of particular concern. Estimates from the propulsion system designer state that the thruster temperature could exceed 1,600 °C, and it will be difficult to direct the heat away from the sensitive components just centimeters away from the thruster. Using AM to 'print' the propulsion system may enable designers to develop a custom set of nozzles and supports that could mitigate the soak-back. The nozzles and supports could be printed with additional radiative surfaces or printed to minimize conduction. The use of topological optimization to design a structural component that minimized component mass has been explored previously at NASA by Hull[8] (fig. 31). It is conceivable that a similar approach could be developed to design the propulsion system to meet structural and thermal requirements.

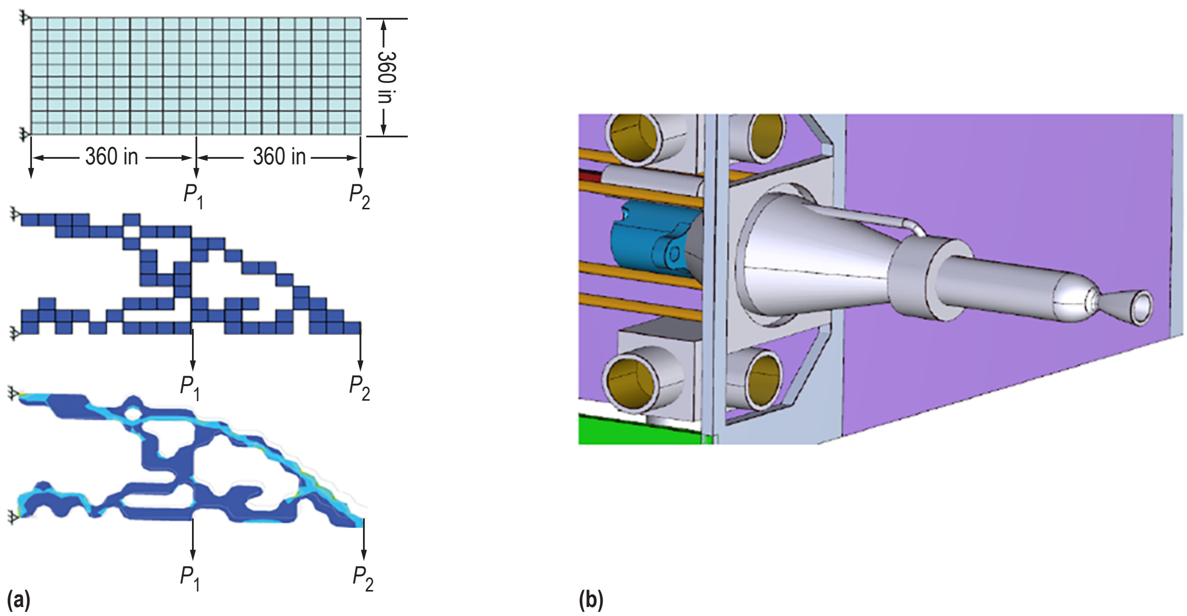

Figure 31. MoonBEAM: (a) Example of topological optimization (from ref. 8) and (b) conceptual drawing of the thruster.

8. CONCLUSIONS AND FUTURE WORK

Key points of the design are listed in table 25. With the assumptions made by the propulsion analyst for ullage, I_{sp}, and propellant density of the green propellant option allowed the design to close. 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 25. Brief summary of spacecraft design.

MoonBEAM dry mass	11.73 kg
Dry mass growth allowance	16%
Propellant mass	2.26 kg
MoonBEAM wet mass	13.99 kg
ΔV capability	368.5 m/s
ΔV margin	10%
Format (volume)	6U
Propellant	AF-M315E high-performance monopropellant
I_{sp}	220 s

Power requirements during preburn were satisfied by turning off the radio, saving 17 W, and providing a total of 32 W to the propulsion system for preheating the thrusters. Keeping the radio operating during this time is not necessary and would have forced the power system to be too large for the configuration. In addition, the 32 W for preheating the thrusters is most likely a conservative number, with the actual requirements being lower.

The next step in the design process is to prove the feasibility of the propulsion system design. Using AM methods to create the propellant tank and associated elements to meet the ullage and volume requirements determined by the design study will show the design feasible. These efforts are currently underway, and should be completed by the end of 2017. While the original MoonBEAM science mission may eventually be proposed on a more conventional CubeSat platform, the results of this latest study may result in more capable and volumetrically efficient CubeSat 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.

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14. ABSTRACT
The Advanced Concepts Office at the NASA Marshall Space Flight Center completed 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. In addition to using the additively manufactured tanks as part of the spacecraft structure, the main propulsion system uses a 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 described.

15. SUBJECT TERMS
additive manufacturing, CubeSat, conceptual spacecraft design, green propellant, gamma-ray burst, gravitational wave

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