

# Space Weather and Radiation Multi-point Magnetometry (SWaRMM): Cube Satellite Mission Design Study

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**Abstract**—Emerging developments in CubeSat technology will allow multi-point measurements for various space weather, radiation, and plasma wave characterization. A mission such as SWaRMM will be able to conduct multi-point measurement science by deploying six 0.5U cube satellites in a slowly separating string of pearls constellation. By employing a simple mini fluxgate magnetometer for each CubeSat's only payload, SWaRMM will enable a low cost, low energy, and efficient mission able to conduct science over a variety of applications. The mission is presented and characterized through various subsystems and communications networks. Preliminary results indicate plausibility for mission success for a swarm constellation of 0.5U CubeSats, given further trade study and analysis to the EPS.

**Index Terms**—CubeSat Constellation String-of-Pearls Swarm

## ABBREVIATIONS

<b>SWaRMM</b>	Space Weather and Radiation Multi-point Magnetometry
<b>MSU</b>	Montana State University
<b>SSEL</b>	Space Science and Engineering Laboratory
<b>SMAD</b>	Space Mission Analysis and Design
<b>ICE</b>	Integrated CDH and EPS unit
<b>LEO</b>	Low Earth Orbit
<b>EPS</b>	Electrical Power System
<b>ADCS</b>	Attitude Determination and Control System
<b>CDH</b>	Command and Data Handling
<b>EMIC</b>	Electromagnetic Ion Cyclotron
<b>P-POD</b>	Poly Pico-satellite Orbital Deployer
<b>AEO</b>	Average Energy Output
<b>OAE</b>	Orbital Average Energy
<b>EOE</b>	End of Eclipse
<b>BOL</b>	Beginning of Life

## NOMENCLATURE

$R$	Vector from Earth center to satellite center of mass
$a$	Semi-major axis
$\mu$	Earth's gravitational parameter
$e$	Eccentricity
$f$	True Anomaly
$h$	Attitude
$R_{\oplus}$	Earth's Radius
$P$	Period
$P_{sa}$	Power required from solar array
$\theta$	Incidence angle
$G_s$	Solar flux

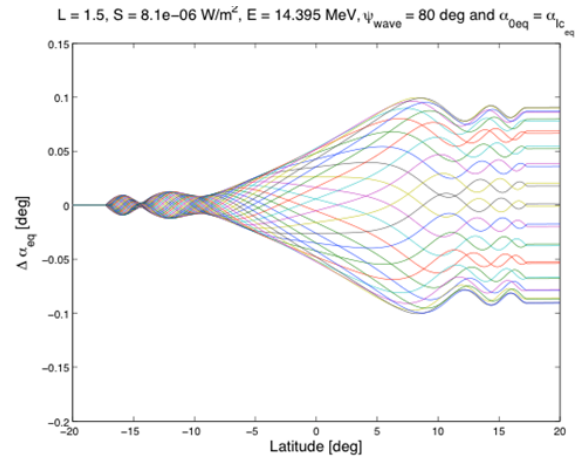


Fig. 1. Total pitch-angle scattering versus latitude for an oblique EMIC wave interacting with equatorially resonant protons ( $L=1.5$ ) [1]

$\eta$	Solar cell efficiency
$I_d$	Inherent degradation
$P_{BOL}$	Beginning-of-life solar array power output
$A_{sa}$	Required solar array area
$\omega$	Angular momentum
$\vec{L}_G$	Gravity gradient torque

## I. SCIENCE GOALS

The SWaRMM design currently utilizes a single mini fluxgate magnetometer per CubeSat on a deployed boom. Using this concept in a constellation of 6 CubeSats will allow each payload to collect magnetometry measurements over many spatial and temporal wave distributions in the Earth's magnetosphere and ionosphere. Two science goals that are being investigated to be integrated into a SWaRMM mission include EMIC wave characterization and the study of Birkeland currents. EMIC waves occur in the Earth's magnetosphere from plasma instabilities and propagate energy from anisotropic protons when the particles gyrofrequencies match the Doppler shifted wave frequency. An example of this can be seen in Figure 1, where EMIC pitch-angle scattering vs. latitude is observed [1]. Previously, these waves have been extensively characterized at low latitude inclinations where EMIC wave ion fluxes peak at minimums in the Earth's magnetic field lines. A high latitude mission would allow the

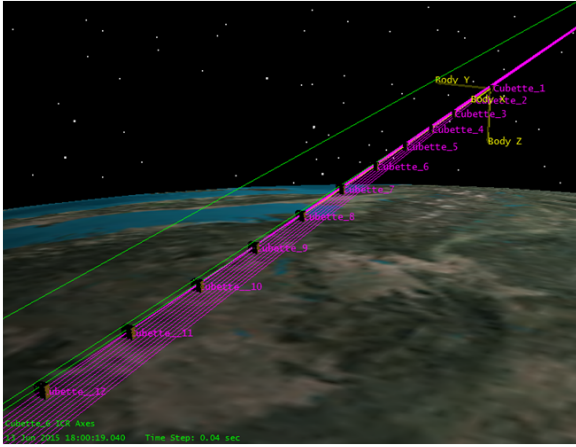


Fig. 2. STK String of Pearls Deployment

characterization of EMIC waves at various local minimums in order to understand the true nature of EMIC wave propagation. EMIC waves may play an important role in facilitating radiation and electron loss cones in the magnetosphere, which impact satellites and other space technologies in orbit [1]. Secondly, a mission investigating Birkeland currents in Earth's high latitude ionosphere may be considered. Birkeland currents flow along Earth's geomagnetic field lines by the bulk motion of plasmas and play an important role in accelerating magnetospheric electrons to produce the Auroras Borealis [2].

## II. SWARM PRELIMINARY DESIGN

The mission is in a zeroth-order preliminary design phase which aims to utilize flight heritage from MSU's FIREBIRD 3 and 4 and ICE design concepts, as well as SMAD principles [3]. The mission aims to deploy six 0.5U (5x10x10cm) size CubeSats in a slowly separating string of pearls constellation in LEO. The string of peals configuration is a linear constellation in which all satellites orbit within the same plane, as seen in Figure 2. This will allow for multi-point measurement space science at high latitude inclinations (60-80 deg). The mission life is dependent only on the length of orbit of each unit, as each CubeSat is designed to relay space network data individually, eliminating inter-satellite communication and dependency. The subsystems for structural, EPS, ADCS, and CDH will be analyzed and designed to meet the mission flow-down requirements and present current mission outcomes, given reasonable assumptions. The design will then be iterated and re-defined concurrently as trade studies within the design emerge. Current launch opportunities for proposal range from ejection from a P-POD to ISS on board deployment.

### A. Payload

The payload for each CubeSat will consist of a single mini fluxgate magnetometer extended from a deployed boom once in orbit. A magnetometer mission will enable a variety of missions ranging from EMIC wave characterization to analyzing Birkeland currents in the Earth's Ionosphere. Based on previous successful magnetometer missions, the single magnetometer will also allow for a low power design,

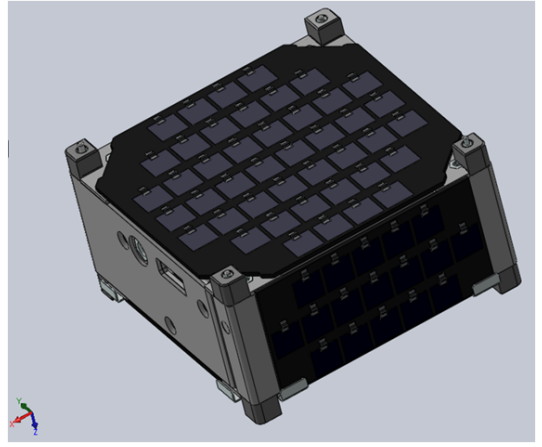


Fig. 3. Solid Works 0.5U CubeSat Rendering

allocating no more than 200 mW to the payload power budget. The deployed boom is to be designed to extend past the perceived Debye length, and may also be studied to act in conjunction with gravity gradient stabilization for the CubeSat. [4].

### B. Configuration and Structure

Different preliminary structure designs for the mission will be considered. Currently, all designs must meet a 0.5U CubeSat size with a weight of no more than 0.65kg per unit in order to be deployment compatible within one P-POD. One current design can be seen in Figure 3 with a configuration of CTJ SolAero cells, providing  $122cm^2$  of cell area to be analyzed for power output [5]. However, using flight heritage designs from MSU's FIREBIRD 3 and 4 mission is important, and a configuration of BTJM solar cells will first be considered, with the BTJM cell seen in Figure 5, and cell IV curve shown in Figure 4.

### C. Mass Budget

Currently each CubeSat is projected to have a positive mass budget even with calculated uncertainties accounting for the mass reserve. Utilizing the components seen in Table I, there is still mass left over to allocate to ADCS gravity boom and hysteresis rod components. Each component has been

Component	Mass (g)	% Uncertainty	Total (g)
Astro Dev Radio	78	10	85.8
VHF Antenna cable and feed point	15	30	19.5
UHF Antenna cable and feed point	9.7	20	11.64
Payload	18	10	19.8
Gravity Boom, housing, Hysteresis	30	30	39
ICE Board with Battery	160	10	176
Structure	102.6	10	112.86
Staking and Coating	10	30	13
M3 x 11mm standoffs	4.8	10	5.28
Cabling	16.6	30	21.58
Solar Panels Type A	108	30	140.4
Total	552.7		644.86

TABLE I  
SWARMM MASS BUDGET

System	Average Duty Factor (%)	Average Power (W)	Orbit Average Energy (watt-min)
Flight Computer/EPS	100	0.10	9.5
Command/Data Receiver	100	0.15	14.2
Telemetry/Data Transmitter	2.9	0.15	13.7
Payload	50	0.10	9.5
Total Output		0.50	46.8

TABLE II  
0.5U POWER BUDGET

taken from measurements and estimates from FRIEBIRD's mass budget and has been scaled down to a 0.5U structure accordingly. It is important to note that many of these numbers, such as the payload, is only a preliminary estimate as a specific component has not yet been selected. Including the worst case mass uncertainties, each unit comes out 5.14 grams below the 650g margin. This is promising, as the mass in the solar panel group is considered a high estimate for  $300\text{cm}^2$  of solar cell area, and will likely be lowered in future design iterations.

#### D. Electrical Power System

The EPS will need to be designed to provide a power source, energy storage, power distribution, and power regulation [3]. Each CubeSat will employ a primary input of solar power with a secondary backup of a single 4.2V rechargeable Lithium Ion battery to deliver 3.3-V of regulated power to the units micro-controllers. Each CubeSat will employ ICE's flight hardware for these micro-controllers. In order to reduce system complexity, a method of direct energy transfer will be employed to connect the solar cells directly to the battery, add-only a diode element for back-flow protection. This will eliminate energy loss that would otherwise go towards power point tracking circuitry. The major assumptions for the EPS is that a 5.6V can be achieved with a string of solar cells in an array. The EPS is first analyzed with a design based on FIREBIRD 3 and 4's successful BTJM solar cell flight heritage, with the BTJM cell seen in Figure 5. Currently, to be able to support a 200mW payload with a configuration of solar cells, 0.8W of average sunlit power must be produced, calling for an average solar current of 143 mA. A SWaRMM CubeSat will have two solar panels per (10x10cm) side for the +y and -y faces, while three (5x10cm) sides will have only one

Parameter	Assumed Value	Result	Equation
$h$	500 km		
$R_{\oplus}$	6378 km		
$a$	6878 km		
$\mu_{\oplus}$	$3.986 \times 10^5 \text{ km}^3 \text{ s}^{-2}$		
$P$		94.6 min	1
$P_{sa}$	0.8W	1.04W	$0.8W + 30\% \text{ reserve}$
$\theta$	45 deg		
$G_s$	$1353W\text{m}^{-2}$		
$\eta$	28%		
$I_d$	100%		
$P_{BOL}$		$267.8W\text{m}^{-2}$	2
$A_{sa}$		$38\text{cm}^2$	3

TABLE III  
POWER SUBSYSTEM SIZING

panel per face. This configuration is only valid assuming one panel can fit on the three (5x10cm) faces, allowing for one face to be left empty for boom deployment housing. Given the flight unit parameters in Table II, and by using current to user bus values for varying beta angles, an average power of 0.85W is achieved, producing a Solar Array AEO of 52.35 watt-min in comparison to the flight unit's output of 46.8 watt-min. This provides a net positive amount of energy available to recharge the battery after EOE, however this does not provide a standard 30% reserve that must meet 60.84 watt-min. It is also best to not use this estimate as using a single BTJM cell on the 0.5U face would require booster circuitry to provide a higher solar cell voltage to match the battery voltage output.

Alternatively, the power budget may be sized to estimate the principal solar array size needed to provide a minimum of 1.04W of average solar array power. This is the minimum 0.8W margin for a 200mW payload, plus a 30% reserve. Sizing for a principle solar array can be seen in Table III. Equation 1 is first used to calculate the orbital period of 94.6 min. If a sunlit orbit fraction of 0.65 is assumed, then there will be 61.5 min of sunlight per orbit. Equation 2 estimates a BOL power estimate of  $267.8W\text{m}^{-2}$ , which results in a need for  $38\text{cm}^2$  of solar cell area per face in order to power the CubeSat, given an assumed 28% cell efficiency and 45 deg sunlight incidence angle.

$$P = 2\pi \sqrt{\frac{a^3}{\mu_{\oplus}}} \quad (1)$$

$$P_{BOL} = G_s \eta I_d \cos \theta \quad (2)$$

$$A_{sa} = \frac{P_{sa}}{P_{BOL}} \quad (3)$$

Solar Cell Area = 26.6 cm<sup>2</sup>

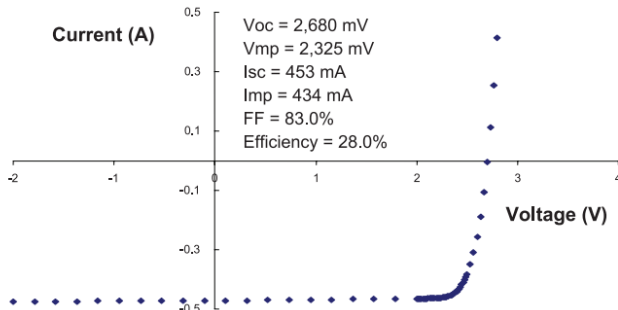


Fig. 4. BTJM Cell IV Curve

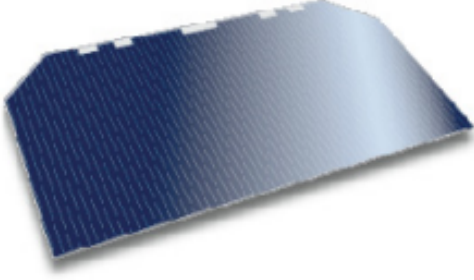


Fig. 5. BTJM Cell

In this approach, the key assumption is that the solar array strings are able to be attached in such a way as to match the battery voltage, plus 0.7V for the direct energy transfer diode drop. In order to accommodate the estimated cell area and array strings, miniaturized (1x1cm) cells, as seen in Figure 3, may be needed to power a 0.5U design. Further trade studies of specific solar cells will need to be conducted.

#### E. Attitude Determination and Control

Orbit determination is not required for a SWaRMM mission and most likely will not be investigated given the 0.5U size of each unit will not likely be able to support an added GPS component. However, CubeSat tumble control may be desired for ground link communications and overall unit stability. One feasible option for this may include utilizing the gravity gradient produced from the boom deployed magnetometer payload, or by incorporate a point mass. This will allow spacecraft nadir pointing as the unit will align with its maximum moment of inertia along the radius vector. Further assessment using a gravity gradient torque calculation is needed, starting with the magnitude of the position vector:

$$R = \frac{a(1 - e^2)}{1 + e \cos(f)} \quad (4)$$

If the orbital period is

$$p = 2\pi \sqrt{\frac{a^3}{\mu}} \quad (5)$$

The gravity gradient torque can be found using

$$\vec{L}_G = \left[ \frac{3\omega_{orbit}^2(1 + e \cos(f))^3}{(1 - e^2)^3} \right] \left( \frac{\vec{R}_b}{R} \right) \times I \left( \frac{\vec{R}_b}{R} \right) \quad (6)$$

where the full derivation can be found through Hall [7]. Simplifying for the worst-case disturbance

$$\vec{L}_{Gmax} = \frac{3\mu}{2R^3} |I_z - I_y| \sin 2\theta \quad (7)$$

can be used to find the maximum gravity torque [3]. In the simplest case of treating each 0.5U CubeSat as a solid rectangular prism, each mass moment of inertia can be found in Table V.

Using a worst case  $\theta = 1$  deg, and equation 7, the max gravity torque comes out to be  $\vec{L}_{Gmax} = 6.03 \times 10^{-10} N \cdot m$ .

Inertia axis	$kg \cdot m^2$
$I_x$	0.108
$I_y$	0.108
$I_z$	0.067

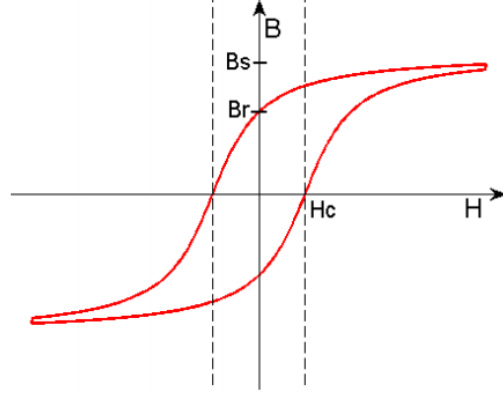
TABLE V  
MASS MOMENTS OF INERTIA

Fig. 6. Hysteresis loop

Given the torque, mass moments of inertia, and inertia tensors for the tip mass and boom, further simulation for gravity gradient control can be done outside of this paper's analysis [8].

In addition to gravity gradient control, hysteresis rods may be assessed to be implemented on the bottom of each spacecraft to dampen gyration motion. Hysteresis rods dampen a system by shifting polarities in response to magnetic field changes, converting rotational energy into heat. Figure 6 shows a hysteresis loop characterized by the magnetic hysteresis parameters  $H_c$ ,  $B_r$ , and  $B_s$ . The hysteresis loop provides information on the rod's induced magnetic flux density given the magnetic field strength, and ultimately describes the dampening per cycle per unit volume with its area [9]. The magnetic parameters vary with rod diameter, length, material, and external field strength and will need to be estimated in order to select a hysteresis rod design outside of the scope of this paper.

#### F. Command and Data Handling

In order to facilitate mission simplicity, the SWaRMM mission will incorporate independent satellite to ground link communications. Eliminating most autonomous operations is a specific goal for the mission, which will allow for the creation of simple systems in the flight hardware as well as in the software configurations. Autonomous command sequencing would still be desired. Also, utilizing two ground stations for communications for the mission may be desired. Utilizing two ground stations could produce several outcomes ranging from being able to communicate with 3 satellites at a time, per each 10 minute pass, or by being able to receive twice the amount of data for all 6 satellites at a time in each 10 minute pass.

The current assumption for the mission science is that the EMIC waves have a 5Hz frequency, therefore a minimum of a

System	Requirements	Outcomes
Top level	Perform multi-point magnetometer measurements for space weather and radiation events in the magnetosphere.	TBD
Secondary Payload	Demonstrate 0.5U CubeSat technology for a multi-unit mission Mini Fluxgate magnetometer	Plausible 200mW budget
Orbit Structure	Operate within a suitable orbit to characterize various space weather and plasma wave propagation Accommodate the payload at the lowest cost and size	400-600km Orbit at 60-80 deg. inclination 0.5U Less than 0.65kg
EPS	Power Source from Photovoltaic cells with a secondary LiIon battery source	ICE board design
C&DH	Based on a low-power microcontroller	ICE board design
ADCS	Gravity gradient boom will be investigated	TBD

TABLE IV  
SYSTEM REQUIREMENTS AND VERIFICATION

Telemetry	Bits/sec	MB/day	% Storage/day
Housekeeping	14	0.14	2.4
Burst Parameter	92	0.31	15.6
Context Channel	3	0.01	0.5
High Resolution	480	4.94	81.4
Total	590	5.4	100

TABLE VI  
PERCENT STORAGE SPACE DATA

Parameters	Outcomes
Baud	19200
Good passes per day	2
Time of good passes	10 min
Bad passes per day	3
Time of bad passes	3 min
Bitrate	590 bits/sec
Total down-link per day	0.12MB

TABLE VII  
TOTAL BITS AVAILABLE TO DOWN-LINK

10 Hz sample rate will be needed to characterize the waves. A Q 3 axis magnetometer producing a 16 bit value for each axis at 10 Hz will give a bit rate of 480 bits per second. When added to housekeeping data and context data, each satellite will produce about 5.4 MB a day, as seen in Table VI. Each satellite will be equipped with a minimum of 4 GB of NAND-Flash, therefore, each satellite would be able to run for 740 days without overwriting data. The full data of the mission will be able to be stored on the satellite with the available down-link capability.

### G. Down-link

With all six satellites gathering data with a 100 percent duty cycle, each ground station would be able to down-link about 2 percent of the data collected each day per CubeSat, as seen in Table VII. Therefore, a campaign mode would need to be used, gathering data for a time, then stopping to down-link. The other viable option, based off of FIREBIRD'S ability, is to only down-link low-res data and look for potential events. This way, only high-res data that is a viable candidate for an event is down-linked. With the high amount of storage on the satellite there is little risk of losing the data due to overwriting, so data can be down-linked long after the mission life is over.

## III. CONCLUSION

Thus far in the SWaRMM mission design and analysis, a constellation of six 0.5U Cube Satellites presents a feasible mission proposal. Multi-point magnetometry has proven to provide insight on many interesting space weather and radiation questions, and will easily incorporate into a 0.5U design. Although the current results are promising, further characterization of the EPS and ADCS will need to be done, and trade studies for system components within each system may also be desired. Preliminary estimates in this paper currently do not foresee any major CDH or link analysis issues when based on FIREBIRD's flight heritage. In the continuation of the mission, SWaRMM will prove to be an unprecedented constellation mission with its unique 0.5U design and low power, low cost technology demonstration.

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## REFERENCES

- [1] Maria de Soria-Santacruz, *Electromagnetic Ion Cyclotron (EMIC) Waves for Radiation Belt Remediation Applications*, Space Propulsion Laboratory, MIT, 2015.
- [2] Wiley, *Observing the Birkeland currents*, ScienceDaily. ScienceDaily, 6 October 2014.
- [3] Wertz, J.R., and Larson, W.J., *Space Mission Analysis and Design*, 3rd ed., Microcosm, El Segundo, CA, 1999, pp. 353-518.
- [4] Electdom Matandirotya, and Robert R. Van Zyl, *Evaluation of a Commercial-Off-the-Shelf Fluxgate Magnetometer for CubeSat Space Magnetometry*, JoSS, Vol.2, No.1, pp. 133-146.
- [5] SolAero Technologies, *CTJ Photovoltaic Cells*, Datasheet, REV 2015.03.
- [6] Emcore Corporation, *BTJM Photovoltaic Cell*, Datasheet, REV 2012.09.
- [7] Hall, Chris *Satellite Attitude Dynamics*, <http://www.dept.aoe.vt.edu/cd-hall/courses/aoe4140/SatDy.pdf>, 2002, pp. 1-12.
- [8] Erich Bender, *An Analysis of Stabilizing 3U CubeSats Using Gravity Gradient Techniques*, California Polytechnic State University, San Luis Obispo, 2011, pp. 1-24.
- [9] David T. Gerhardt, *Passive Magnetic Attitude Control for CubeSat Spacecraft*, University of Colorado, Boulder, CO, 24th Annual AIAA/USU SSC10.