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LunarCube: A Deep Space 6U CubeSat with Mission Enabling Ion Propulsion Technology

Michael Tsay, John Frongillo, Kurt Hohman
Busek Co. Inc.
11 Tech Circle, Natick, MA 01760-1023; 508-655-5565
mtsay@busek.com

Benjamin Kevin Malphrus
Morehead State University
235 Martindale Drive, Morehead, KY 40351; 606-783-2212
b.malphrus@moreheadstate.edu

ABSTRACT

Busek, in partnership with Morehead State University (MSU), is developing a versatile 6U CubeSat platform nicknamed “LunarCube” that can undertake missions beyond LEO. The spacecraft can host a variety of science payloads, and its mission capability is highlighted by >3km/s of delta-V maneuverability with a groundbreaking ion propulsion system heretofore unavailable to CubeSats. Salient features of this propulsion system include innovative use of solid iodine propellant and a 60W class mini RF ion thruster that is capable of 1.3mN thrust and 3250sec specific impulse (Isp). The primary objective of the LunarCube program is to support a deep space CubeSat mission to the Moon from GEO or a translunar trajectory (such as the SLS/EM-1 drop-off) and carry out a lunar science campaign as a technology demonstration of the platform. A secondary objective is to showcase that much of the spacecraft’s miniaturized avionics and power system can survive the harsh radiation environment. The LunarCube concept, especially its ion propulsion element, has received significant attention from the CubeSat user community targeting near-term lunar flights. In fact, the platform has already morphed into an EM-1 CubeSat mission known as “Lunar IceCube”, selected for flight by NASA’s Next Space Technologies for Exploration Partnerships (NextSTEP) program.

INTRODUCTION

Due to the extremely rapid progress made in the miniaturization of electronics, small spacecraft such as CubeSats are increasing their utility and concomitant acceptance by the space communities that include NASA, DoD and commercial organizations. The next phase in the small spacecraft evolution is the inclusion of capable propulsion, as most of the interesting applications require mobility. Such applications include formation flying for telecommunication, stationkeeping of low-flying earth observation (EO) CubeSats, and low cost robotic precursor missions to the moon, asteroids and other inner planets. CubeSats have come a long way since its conception 15 years ago and missions beyond LEO are no longer just a dream. NASA has recently announced that the upcoming Space Launch System’s maiden flight (SLS EM-1 mission) will host up to eleven 6U CubeSats and deploy them on a lunar trajectory.[1] This development has prompted a race to bring online sophisticated, high delta-V micro propulsion systems to help the CubeSats either slowdown to enter lunar orbit or escape for an asteroid rendezvous. One of these propulsion solutions is Busek’s iodine-fueled, 3cm-grid-sized RF ion thruster known as BIT-3 that is the core of the LunarCube technology.[2] The NASA-funded LunarCube program aims to develop a standardized 6U CubeSat bus with propulsion which could carry a variety of science payloads as long as they are within a pre-determined design envelope (i.e. 1.3U/1.2kg from current design). The ultimate goal is to provide a COTS vehicle for science investigators, allowing them to focus on developing instruments rather than worrying about how to reach their destinations or transmitting telemetry data.

LunarCube’s BIT-3 RF ion thruster, shown in Figure 1, has genesis from NASA-sponsored R&D efforts for micro electric propulsion (EP).[3] RF ion thrusters are a type of plasma based EP device that is characterized by its scalability and high Isp. The thrust-producing ions are created by electrode-less, Inductively-Coupled Plasma (ICP) discharge before accelerating to high exhaust velocity via electrostatic grids. Isp performance varies with thruster size, but >3000sec is the norm as with other types of gridded ion thrusters.

Among all EP devices, RF ion thrusters have a unique advantage of being iodine-compatible, as their plasma-generation chambers are typically made of corrosion-resistant ceramic materials. The ability to use iodine as propellant is a true game-changer for CubeSat propulsion, because iodine can be stored as a dense solid (3x storage density than pressurized xenon) and its sub-Torr storage pressure is safe to launch while allowing for the use of very lightweight, even plastic tanks. Busek's BIT-3 thruster is regarded as the world's first gridded ion thruster to demonstrate with iodine propellant. This achievement is significant because the BIT-3 thruster, when augmented by iodine propellant, has size, thrust, power, volume, and contamination parameters that are mission-enabling for a very large class of CubeSat missions that are closely aligned with the past, current and future science goals of the space community.[2]

Figure 1: BIT-3 RF Ion Thruster Firing with Iodine Propellant at 60W Thruster Power

LUNARCUBE DESIGN OVERVIEW

Development of the LunarCube deep space CubeSat bus leverages MSU's CubeSat mission experience, utilizes systems with significant flight heritage (in LEO), and incorporates new NanoSat technologies to develop an evolved, radiation-tolerant 6U CubeSat that can support interplanetary investigator science. The 6U LunarCube is based on MSU's bus heritage and incorporates high power generation (84 Watts of continuous power in cislunar space), a radiation-tolerant, distributed multiple processor-based payload processor system, a highly-capable micronized Guidance, Navigation and Control (GNC) system designed for lunar missions. Several options are considered for communications including COTS systems and a high throughput X-band communication system designed by JPL for lunar CubeSat missions. The JPL X-ban radio, known as Iris, is currently the default communication system.

The 6U LunarCube bus was essentially derived from MSU's successful 2U Cosmic X-Ray Background NanoSatellite mission and 1U KySat-2 missions. Additionally, several of the subsystems have successfully flown on numerous NanoSat and MicroSat missions and most of the COTS subsystems will have flight heritage prior to the development of a complete LunarCube flight unit. Although the spacecraft design is evolving, an up-to-date specification overview is provided in Table 1, with Figure 2 and Figure 3 showing the subsystem layout. The combination of flight qualified hardware and innovative solutions to difficult engineering challenges provides for a robust spacecraft bus solution for the LunarCube program.

Table 1: Up-to-Date Design Specifications of LunarCube 6U Bus with Ion Propulsion

Parameter	Value
Launch Mass	12.0kg max
Bus Mass (w/o Payload or Propulsion)	7.6kg
Propulsion Dry Mass	1.7kg
Propellant Mass (Iodine)	1.5kg
Payload Mass Capability	1.2kg
Payload Volume Capability	1.3U
Pointing Accuracy	±.002˚
Orbit Knowledge	10m, 0.15m/s
Maneuver Rate	10˚/s
Payload Power Capability	5W (peak), 3.8W
Prime Power Generated	84W nominal
Voltages Available	12V, 5V, 3.3V
Downlink Data Rate	12 kbps
Spacecraft Op Lifetime	> 2 years
Performance of Ion Propulsion System	
Power Available for Cont. Thrusting	65W (at PPU input)
Nom. Power to Thruster Head	50W
Nom. Thrust	1.1mN
Nom. Isp (Including Neutralizer)	2500sec
Max Delta-V Capability	3.2km/s
Total Impulse Capability	37,000N-sec
Total Firing Time Before EOL	9,300hrs

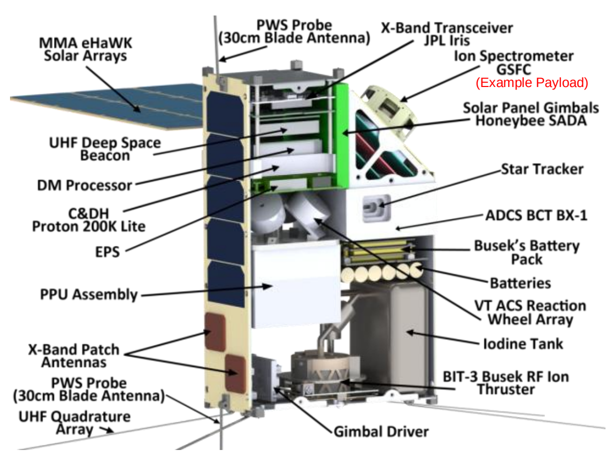

Figure 2: Up-to-Date Design Layout of the 6U Lunar Cube with Example Science Payload

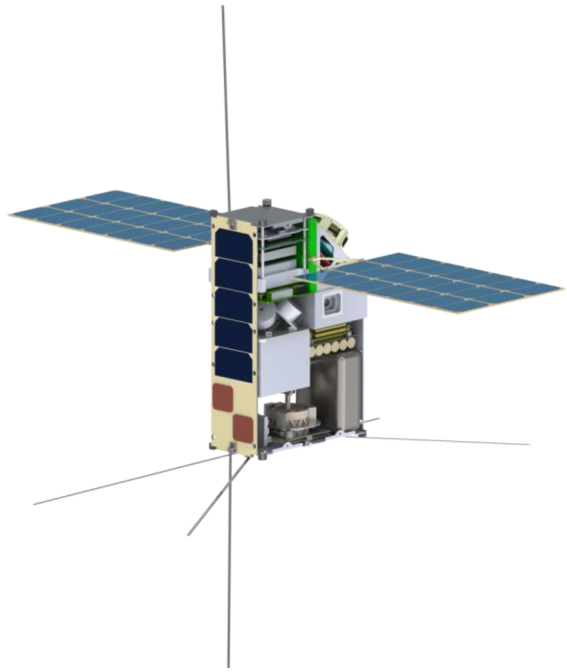

Figure 3: Complete View of the Baseline Design for the 6U LunarCube Spacecraft

A majority of resources to date has been allocated to investigating the most critical-for-success subsystems of LunarCube, which include the BIT-3 RF ion propulsion system, C&DH, communications, and power management. While attitude control system (ACS) was not proposed to be studied in the early phase, it became necessary to consider the ACS design and its impact on other systems. For example, design features of the main propulsion system were driven by the ACS concept, which utilizes a 4-wheel reaction wheel assembly (RWA) and a "gimbaled" primary thruster for wheel de-saturation/momentum dumping. As designed the propulsion system incorporates a BIT-3 ion thruster mounted on a 2-axis translational stage for thrust vectoring. Though this setup is under-actuated for roll-axis control, the thruster can theoretically offload the angular momentum through a 3-burn maneuver. Moreover, a redundant wheel (4th wheel) is proposed to alleviate the RPM requirement on the roll axis so the thruster's role in angular momentum dumping is minimized. Note that for thrust vectoring the 2-axis stage shown in this paper is just a preliminary concept; it will likely be replaced by a more traditional ball-joint type of gimbal during next design revision.

ION PROPULSION SUBSYSTEM

Iodine BIT-3 System Description

Figure 4 describes the concept of a complete iodine BIT-3 propulsion system with both the thruster and the neutralizer cathode operating on a single iodine reservoir tank. This diagram largely represents the experiments performed to date, except during testing the subminiature neutralizer was operated with xenon instead. In addition, the Power Processing Unit (PPU) designed to drive all the subsystem components was replaced by bench power supplies with manual open-loop controls.

During operation, the propellant reservoir is heated to produce sublimated iodine vapor which is then fed to the thruster. The transmission line and flow control valves are heated to above the tank temperature in order to prevent condensation. The vapor flow rate is measured in-situ and in real time with a calibrated pressure transducer upstream of the injector, via choked flow equation. Due to the accuracy of the pressure transducer, this measurement method has an inherent 15% error margin for the reported flow rate, which reflects on the Isp result.[2]

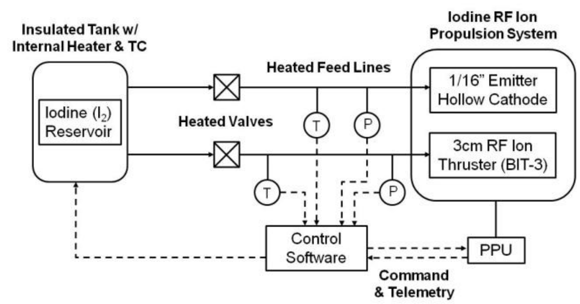

Figure 4: Iodine BIT-3 System Block Diagram

Why Use Iodine

The idea of utilizing iodine as an EP propellant was conceived over a decade ago and patented by the U.S. Air Force.[4,5,6] Busek pioneered the practical application of such propellant in 2010 with the development of an iodine-fueled 200W Hall Effect Thruster (HET) known as BHT-200.[7] This development has led to a patent involving the iodine feed system that was the departure point of all Busek's iodine EP technologies.[8] Owing to its mission-enabling properties for small spacecraft, iodine-fueled EP concepts have been receiving a lot of attention from NASA and industry alike.[9,10]

Iodine has many advantages over legacy EP propellant xenon from the perspective of storage and handling requirement (4.9g/cc solid vs. 1.6g/cc highly compressed gas), launch safety, and cost. Table 2 compares the physical properties between these two propellants.[2] Due to its extremely low storage vapor pressure (<250mTorr at room temperature), an iodine reservoir tank can be thin-wall, irregular-shaped and made of lightweight material such as thermoplastics. These attributes can help significantly reduce the overall physical budget impact of the propellant tank on an already volume and mass-constrained CubeSat platform.

Iodine appears naturally as solid crystal in diatomic form (I_2), with relatively-low dissociation energy at around 1.54eV.[12] It is true that in a plasma generator some of the I_2 molecules may become ionic dimers (I_2^+) without dissociating, but as Ref. 11 points out these species are relatively low in percentage (<10%) and most of the propellant exit as singly-charged monomer I^+. Having I_2^+ is actually advantageous in an electrostatic thruster like the BIT-3, as thrust and Isp scales linearly with the square root of the ejected ion mass.

Table 2: Property Comparison between Iodine and Xenon[12]

Element	I	Xe
Atomic Mass	126.9	131.3
Ionization Properties (monatomic)		
First Ionization Potential (eV)	10.5	12.1
Peak Cross Section (10^{-16} cm^2)	6.0	4.8
Storage and Handling Properties		
Storage density (gm/cm^3) near room temp.	4.9	1.6*
Melting Point (°C)	113.7	-112
Boiling Point at 10 Pa (°C)	9	-181

*14 Mpa, 50 C (NIST Database)

Iodine BIT-3 Test Data

Performance of the prototype BIT-3 thruster was first established with xenon and then compared to that with iodine. The result, reported in Ref. 2, shows strong correlation and iodine was determined as a suitable drop-in replacement for xenon. Figure 5 shows the thruster operating with iodine at its nominal power of 60W and at de-rated power of 30W. The total thruster power includes input power to the grids and the RF discharge power, but does not include power supply efficiency or neutralizer consumption.

Figure 6 and Figure 7 show a summary of thrust and Isp performance of the iodine BIT-3. The thruster is capable of varying thrust from 1.3mN down to 0.2mN, though Isp would tail off rapidly as the thruster power decreases. As designed, the nominal operating point of BIT-3 is at 60W thruster power and 42µg/s iodine flow, which produces 1.3mN thrust and 3250sec Isp. Since the thruster will likely operate at around 50W max in LunarCube according to Table 1, the performance will be limited to 1.1mN thrust and 2800sec Isp. When taking neutralizer's propellant consumption into account, the overall system Isp will end up being close to 2500sec (Table 1). Notice that the thrust and Isp values mentioned above were not directly measured from a thrust stand; they were calculated based on the measured ion beam current. Such thrust calculation is a tried-and-true method for gridded ion thrusters, and Busek has previously confirmed that this method has a typical accuracy of 95% when comparing to actual thrust stand measurements.

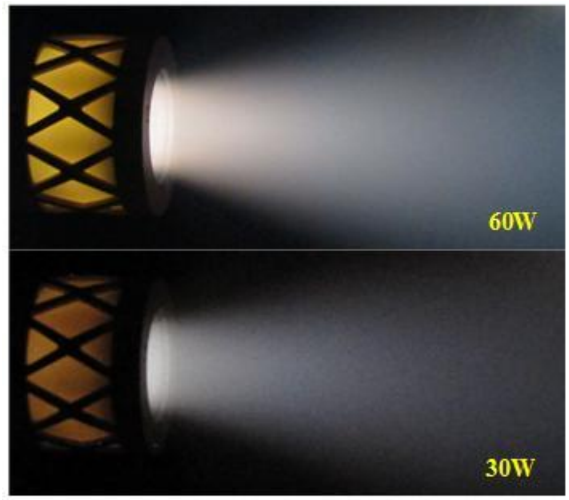

Figure 5: Iodine BIT-3 Operating at Different Total Thruster Power Levels

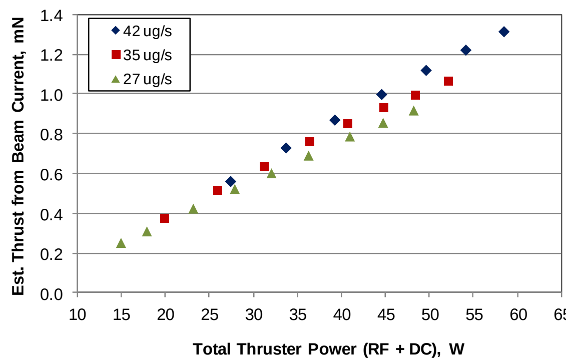

Figure 6: Iodine BIT-3 Estimated Thrust vs. Total Thruster Power

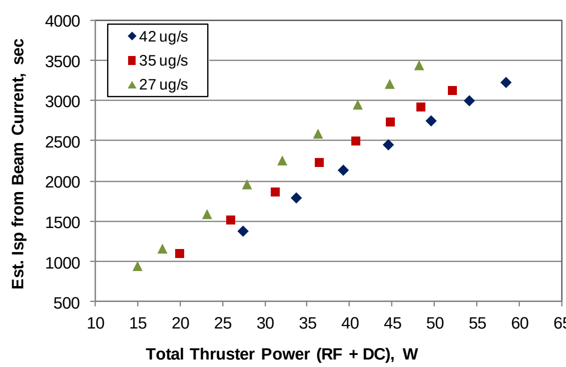

Figure 7: Iodine BIT-3 Estimated Isp vs. Total Thruster Power

Propulsion Power Processing Unit (PPU)

The BIT-3 ion propulsion subsystem will have a dedicated PPU that acts as an electrical interface between spacecraft bus and the thruster. It will provide all the functionalities to operate the thruster, the iodine feed system and the gimbal. The microcontroller-based PPU will provide telemetry and error messages to the spacecraft bus, as well as provide command and control of the various high-voltage converters within the PPU. The complete PPU assembly will be fabricated from high-reliability COTS components and contained within an approximately 1U-sized volume. An aluminum enclosure will provide some radiation shielding.

Though the BIT-3 PPU hardware does not exist yet, its design will be similar to the breadboard PPU recently developed for Busek's 1cm RF ion thruster "BIT-1" (pictured in Figure 8). Among the components found in the BIT-1 PPU, the development of the RF generator board was perhaps the most important one. This device, measured 3.5"×3.5"×1" and shown in Figure 9, employs a signal generator with adjustable frequency and a wide-band power amplifier circuit. The signal generator, known as Voltage Controlled Oscillator (VCO), requires only a simple square wave input and therefore can be made with a digitally-controlled clock generator chip. The amplifier utilizes a class E topology with some modifications to improve performance when driving a high current RF coil. This custom-designed circuitry provides several attractive features that include 1) up to 85% of DC-to-RF conversion efficiency, 2) low parts count and compatible with low voltage MOSFET or other switch devices, 3) automatic impedance matching with closed-loop control via integrated load power sensor.

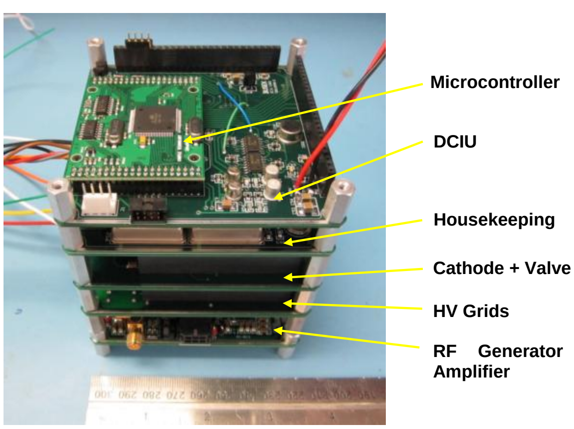

Figure 8: Complete, Breadboard Style BIT-1 PPU

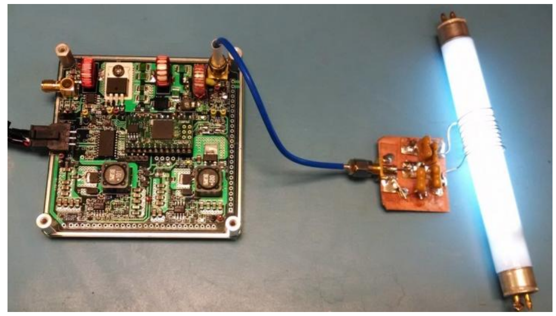

Figure 9: Demonstration of Novel RF Generator/Amplifier Board within the BIT-1 PPU

6U LUNARCUBE BUS DESIGN

Potential missions for the LunarCube platform define challenging requirements. The missions' relatively long duration in the extreme thermal and radiation environment will undoubtedly push the envelope of current CubeSat technologies. The LunarCube's 6U CubeSat system concept is designed to meet the stringent lunar mission requirements [13] while utilizing cost-effective COTS and modified COTS systems.

ADCS and Navigation

Requirements dictate that the Attitude Control System (ACS) provide 3-axis stabilization, as well as 1 degree of pointing knowledge and 2 degrees of pointing control for maintaining 1) the example science instrument 60 degrees from a normal to the lunar surface, 2) the solar array axis perpendicular to the sun, 3) the X-band communication antenna perpendicular to the Earth, and 3) thruster orientation as required for GNC (guidance, navigation, and tracking) and during operation of the main propulsion system. The ACS supports four primary modes, with heritage from flight-proven architecture:

Sun Acquisition Mode – initially at CubeSat deployment for power positive control; will use RWA and sun sensors, with a gyroless submode for fail safe operation.

Observing Mode – during science data taking and cruise operations; will use RWA, star trackers, and IMU to provide inertial, sun, and nadir attitude control as well as slew maneuvers.

Delta-V Mode – utilizes a gimbaled primary thruster to provide trajectory and orbit maneuvers.

Delta-H Mode – utilizes the gimbaled thruster on an infrequent basis for RWA momentum dumping.

Command and Data Handling (C&DH)

The C&DH system onboard the LunarCube spacecraft is responsible for commanding and monitoring all spacecraft subsystems and for communicating with the propulsion system processor to maintain proper trajectory and spacecraft orientation. Given that the spacecraft will be undergoing nearly constant thrust when there is positive power, effective GNC is crucial to the mission. As such, the selection of an appropriate flight computer is paramount.

The C&DH architecture is distributed between the flight computer (Space Micro Proton Lite 200k radiation tolerant processor)[14], avionics controller (Blue Canyon Technologies XB1) [15] , and the low-cost, radiation tolerant, high speed payload processor (Honeywell Dependable Multiprocessor or DM) [16] . Each of the three systems has the capability to control basic spacecraft functions providing redundancy for risk mitigation. The Space Micro flight computer has the potential to send unprocessed data to the ground in the event of failure or lock-up of the payload processor. The DM processor can host all spacecraft functions if required. The on-orbit reconfigurable XB1 includes an integrated ARM Xinc designed to control ACS and navigation functions, as well as all basic spacecraft functions, telecommunications and telemetry, as well as an ARM Xinc 400MHz processor, which accepts and stores real-time commands or macros. Its 4GB storage capacity is capable of storing up to 10 days of science data (based on the proposed science demonstration mission). The DM can preprocess raw science data, minimizing communication system requirements and minimizing required downlink rates.

Communication

The LunarCube vehicle while in deep space will require an uplink command rate of 9.6kbps and telemetry downlink at 115kbps. Additionally, since the spacecraft contains a propulsion system, security protocol and data encryption on the uplink side is required. AES-256 bit encryption is used for uplink commands and MD-5 check-sums are incorporated into command sequences.

Consistently closing the communications link from a CubeSat transmitting 1.5W Effective Isotropically Radiated Power (EIRP) at distances beyond Earth-Moon L2 to downlink significant volumes of science data represents a challenge. To solve this issue the LunarCube platform will utilize the JPL Iris X-band transceiver[17] coupled to broad-beam patch antennas and an evolved mission operations strategy that uses the high-gain Morehead State 21-meter ground station[18] for long stare times (up to 6 hour passes). This operation strategy would support a data throughput rate of 12kbps conservatively and ensure the acquisition of >400Mb science data per day assuming a 10 hour stare time.

The JPL Iris X-band CubeSat standard radio transceiver is state-of-the-art, rad-hard, DSN compatible and can support CCSDS standards and SLE data packet protocol. The system architecture is designed for Direct to Earth (DTE) link and Proximity Operations for nano-spacecraft on deep space missions. The radio supports a wide range of data rates (62.5-256kbps telemetry, subcarrier, low rate tones to 8Mbps available) needed for wide range of distances, and has full duplex capabilities appropriate for Doppler, ranging and Delta-DOR. Iris utilizes a Marina-2 FPGA Modem Processor and Virtex 5 microprocessor. The Iris stack includes an X-Band Receiver, X-Band Exciter and power supply board. In this design, the LunarCube spacecraft will couple Iris to 4 patch antennas that are independently selectable. The system uses 12.8W_{DC} for full communication or 6.4W receive only. It produces 25dBm transmit power with -130dBm receive sensitivity. The entire stack occupies <0.5U and weighs <500g. Table 3 shows a communication link model utilizing the JPL Iris radio coupled to MSU's 21-meter space tracking antenna.[19]

Table 3: Communications Link Model Utilizing JPL Iris and MSU 21-Meter Dish

Frequency X-band	7.1-7.6 GHz
RF Transmit Power	1 Watt (minimum)
Transmit Antenna Gain	6 dBi nominal (X-band patch antenna)
Transmit Distance	Lunar to Earth (410,000 km, nominal)
Receive Antenna	21 meter dish
Receive Antenna G/T	38.7 dB/k
Receive Antenna Gain	62 dBi
Link Margin	3 dB
Data Rate	12 kbps nominal

Bus Power Generation and Management

The LunarCube power system is comprised of 2 deployable MMA HaWK [20] solar panel arrays that generate 72W of continuous power, 4 sets of MMA fixed solar panels that generate nominally 12W of continuous power, 2 Honeybee Robotics solar array gimbals, a radiation tolerant version of the MSU Electrical Power System (EPS), and a Power Management and Distribution (PMAD) system based on the Morehead CXBN design.[21] The combination of these technologies delivers ~84W continuous power from high efficiency (28%) UTJ-S solar cells at the beginning of life, representing sufficient power generation for required propulsion events and significant power margin (>30%) for the science mission. The positive power margin more than accommodates solar cell degradation during the mission life. The PMAD is an innovative, expandable, direct energy transfer system that employs shunt regulation for charging and effective battery protection circuitry. It utilizes 8 Lithium Ion batteries (with significant flight heritage) to support and extend mission life.

The onboard BIT-3 ion propulsion system has a 65W constant power draw (at PPU input) when the spacecraft is in sunlight. The MSU EPS is configured with all necessary features to accommodate this constant power-on mode defined by the mission profile. The energy storage architecture will accommodate radio transmission during the highest required date downlink transmission which will occur when the propulsion system is not powered. The MSU CubeSat EPS is a versatile power system solution for CubeSats ranging from the 1U to the 6U form factors. The MSU EPS has significant capability, powered by a dedicated MSP-430 microprocessor that allows on-orbit programming.[22] With Direct Energy Transfer (DET) and Peak Power Tracking (PPT) capabilities, battery charge regulations, numerous protection systems including battery protection, a BSL system for in-flight software uploads, a high efficiency power stage and a dedicated microcontroller, the EPS offers a proven and robust power management solution for CubeSat systems. The MSU EPS has flight heritage on 3 successful CubeSat/MicroSat missions. Figure 10 shows one of the flight EPS models.

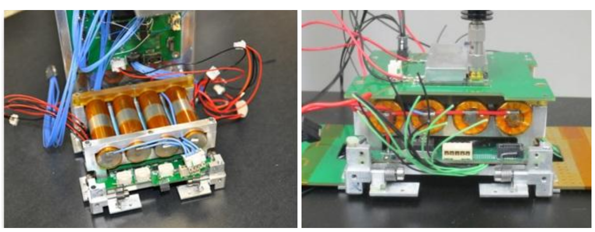

Figure 10: MSU Flight-Heritage EPS

CONCLUSION

Significant progress has been made toward the development of a deep space 6U CubeSat bus with mission-enabling ion propulsion technology. Initiated under a partnership between Busek and Morehead State University, the LunarCube project aims to bring the CubeSat user community a one-stop solution for ferrying a science instrument to lunar orbit and beyond. The core of LunarCube technology involves MSU's flight-heritage bus electronics, state-of-the-art communication and ADCS systems, and Busek's innovative iodine RF ion thruster BIT-3. Such propulsion system will provide up to 3.2km/s of delta-V capability for a 6U/12kg spacecraft, which will enable a variety of missions including asteroid rendezvous, inner planet flyby, as well as low flying Earth Observation (EO) CubeSats. The LunarCube platform has already shown being highly attractive to science payload developers targeting the upcoming SLS EM-1 lunar mission in 2018.

In fact, the Busek-MSU team was recently selected by NASA Headquarters for a mission called "Lunar IceCube" as part of EM-1 CubeSat flight programs.[23] The team will provide a flight-version of the LunarCube spacecraft and host a miniaturized IR spectrometer developed by Goddard Space Flight Center for searching ice/water in lunar regolith. This development will buy down the Non-Recurring Engineering (NRE) cost of the LunarCube platform, paving way for future opportunities in both the science community and commercial markets.

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REFERENCES

1. Cirtain, J. and Pelfrey, J., "Exploration Missions to Host Small Payloads," 28th AIAA/USU Conference on Small Satellites, North Logan, UT, August 2014, SSC14-IV-3.
2. Tsay, M., et al., "Iodine Fueled Mini RF Ion Thruster for CubeSat Applications," 34th International Electric Propulsion Conference, Kobe, Japan, July 2015, IEPC-2015-273.
3. Tsay, M., et al., "Micro Radio-Frequency Ion Propulsion System," 48th AIAA Joint Propulsion Conference and Exhibit, Atlanta, GA, July 2012, AIAA-2012-3947.
4. Tverdokhlebov, O.S., and Semenkin, A.V., "Iodine propellant for electric propulsion – To be or not to be," 37th AIAA Joint Propulsion Conference, Salt Lake City, UT, July 2011, AIAA-2001-3350.
5. Dressler, R., Chiu, Y.-H., and Levandier, D., "Propellant Alternatives for Ion and Hall Effect Thrusters," 28th Aerospace Sciences Meeting & Exhibit, Reno, NV, January 2000, AIAA-2000-0602.

6. Dressler, R., Levandier, D., and Chiu, Y.-H., U.S. Air Force, U.S. Patent for "Iodine electric propulsion thrusters," Publication number US6609363 B1, filed on 26 December, 2000.

7. Szabo, J, et al., "Performance Evaluation of an Iodine-Vapor Hall Thruster," *Journal of Propulsion and Power*, Vol. 28, No. 4, July-August 2012, pp. 848-857.

8. Szabo, J., Pote, B., Paintal, S., Robin, M., and Hruby, V., Busek Co. Inc., Natick, MA, U.S. Patent for "Iodine fueled plasma generator system," Publication number US8610356 B2, filed on 27 July, 2012.

9. Dankanich, J., et al., "Mission and System Advantages of Iodine Hall Thrusters," 50th AIAA Joint Propulsion Conference, Cleveland, OH, July 2014, AIAA-2015-3905.

10. Woeppel, E., et al., "The Near Earth Object (NEO) Scout Spacecraft: A low-cost approach to in-situ characterization of the NEO population," SpaceOps Conference, Pasadena, CA, May 2014, AIAA 2014-1703.

11. Szabo, J. and Robin, M., "Plasma Species Measurements in the Plume of an Iodine Fueled Hall Thruster," *Journal of Propulsion and Power*, Vol. 30, No. 5, 2014, pp. 1357-1367.

12. Szabo, J., et al., "Iodine Propellant Space Propulsion," 33rd International Electric Propulsion Conference, Washington, DC, October 2013, IEPC-2013-311.

13. Space Launch System (SLS) Spacecraft & Payload Integration Office (SPIO), Space Launch System Secondary Payload User's Guide, NASA Document No: SLS-SPIE-HDBK-005, CR SPIE-0121, PCN SP00813, December 16, 2014.

14. Pignol, M., "COTS-Based Applications in Space Avionics," Proceedings of the Conference on Design, Automation and Test in Europe, pp. 1213-1219, ISBN: 978-3-9810801-6-2, 2010.

15. Dumont, C., Blue Canyon Technologies, XB1 Spacecraft Bus Interface Control Document, Revision D, July 17, 2014.

16. Samson, J. and Malphrus, B., "Update on DM CubeSat Technology Development: ISS Flight Experiments," Institute for Electronic and Electrical Engineering, 978-1-4799-5380-6/15/$31.00 ©2015 IEEE.

17. Duncan, C., Iris for INSPIRE CubeSat Compatible, DSN Compatible Transponder Flight Communications Systems Section 337, Jet Propulsion Laboratory, California Institute of Technology, SmallSat Conference Presentation 2013.

18. Malphrus, B., et al., "University-Based Nanosatellite Missions and Ground Operations at Morehead State University," SpaceOps, American Institute of Aeronautics and Astronautics, SmallSat Journal, SSC12-VII-6, 2012.

19. Cheung, K., et al., "Next-Generation Ground Network Architecture for Communications and Tracking of Interplanetary SmallSats," Proceedings of the Interplanetary Small Satellite Conference, San Jose, CA April 2015.

20. Werner, D,, "Builders Packing More Capability into Small Satellites," Space News, August 13, 2012.

21. Malphrus, B., et al., "The Cosmic X-Ray Background NanoSat (CXBN): Measuring the Cosmic X-Ray Background Using the CubeSat Form Factor," American Institute of Aeronautics and Astronautics, AIAA-2012-1261128, 2012.

22. Molton, B., "KySat-2 Electrical Power System Design and Analysis," ProQuest Dissertations And Theses; Thesis (M.S.), Morehead State University, 2013; Publication Number: AAT 1548671; ISBN: 9781303585388; Source: Masters Abstracts International, Volume: 52-04.; 70 p.

23. Clark, S., "NASA Adding to List of CubeSats Flying on First SLS Mission, Spaceflight Now, April 8, 2015.