Lessons Learned for Deploying a Microsatellite from the International Space Station

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The growth of the small satellite industry necessitates increased access to space through rideshare and secondary launch opportunities. The International Space Station (ISS), or Station, has been recently used to deploy several CubeSats, and it is gaining interest in the small satellite community as a burgeoning launch platform for an expanding suite of payload classes. We are working in conjunction with other space entrepreneurs and enthusiasts to utilize Station as the first true Space Port. We have manifested the first-of-its-kind deployment of a 50 kg microsatellite from the ISS in 2015 and are sharing our journey to encourage others to follow and blaze new trails to space access.

Our lessons learned are applicable to any future payload developers that would potentially use the ISS as a platform for space access, and we describe pitfalls and opportunities to consider when manifesting a launch. We describe our experiences in both defining and complying with requirements imposed by NASA, the Launch Vehicle Service Provider, and the Secondary Launch Integrator. Early understanding of the requirements is critical as they directly impact the satellite design, capability, concept of operations, and expected lifetime. We discuss the most challenging requirements we encountered and our approaches for addressing them. We explain the timelines for gaining the necessary approvals and our personal experience in navigating the various approval bodies.

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1. INTRODUCTION

The International Space Station (ISS) is an emerging deployment platform for a new class of small spacecraft. Recently, the ISS has been used to deploy myriad CubeSats,[1] and we have manifested the first-of-its-kind deployment of a 50-kg microsatellite. We plan to use this unique opportunity to foster the ISS as a reliable space-access platform by leveraging the existing Station infrastructure and regular cargo resupply services. Our goals are to share our experiences, encourage others to continue development of these international assets, and expand access to low-Earth orbit.

Often, the most uncertain part of a space mission is launch. Regular and reliable access to space is exacerbated for microsatellite-class spacecraft (i.e. 10–100 kg) because they are not big enough to justify purchase of a dedicated medium- or heavy-lift vehicle,[2] there are no operational launch vehicles specifically sized for microsatellites,[3] and they are too large to be containerized and easily wedged into spare volume as secondary payloads (as has been done with numerous CubeSats).[4]

The current launch solution is rideshare, which matches multiple non-homogeneous secondary payloads with a primary going to a close-enough orbit on a rocket with excess lift capacity. For government-purchased launches, rideshare is facilitated by the United States Air Force (USAF) Space Test Program (STP),[5] but for obvious reasons tend to favor military, civil, and academic payloads—and competition amongst these groups is already fierce. Commercial entities such as Spaceflight Services[6]

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[1] It's a March of the CubeSats as Space Station Deployment Continues, http://www.nasa.gov/mission_pages/station/research/news/cubesat_deployment/#.VEh7LoeL428

[2] A large, dedicated rocket might be feasible for a manifest of several microsatellites launched together (perhaps to populate an entire orbital plane), but this is not currently a widespread practice, nor is it applicable to this situation, as we are launching a single spacecraft.

[3] See *Comparison of Orbital Launch Systems*, Wikipedia (http://en.wikipedia.org/wiki/Comparison_of_orbital_launch_systems). Filter to operational systems then sort by "Mass to LEO". The smallest two (US) domestic vehicles are both Orbital Sciences rockets: the Pegasus, at 443 kg, and the Minotaur I at 580 kg, which have four to five times the capacity required to launch the heaviest of microsatellites. The only vehicle at or below 100 kg was the North Korean Unha, which is not considered.

[4] See link to a photo of a set of three 1U NASA CubeSats containerized in an ISIPod dispenser and its mounting location below the Cygnus Mass Simulator prior to the maiden launch of the Antares launch vehicle. https://twitter.com/SpaceflightInc/status/306880665624924161/photo/1

[5] USAF STP Mission Statement, http://www.kirtland.af.mil/library/factsheets/factsheet_print.asp?fsID=6878&page=1

[6] Disclosure: Spaceflight Services is a sister company of Andrews Space.

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perform a similar function for payloads that do not qualify for government-owned launches or that qualify but don't make the cut. Rideshare, while better than no launch at all, has significant disadvantages: a limited number of primary launches each year (especially when filtered by orbit),[1] and secondary payloads are (understandably) subject to the primary payload's timeline.

An ideal long-term solution is a diversity of dedicated, affordable, small launch vehicles, several of which are in development but not currently operational.[2] Until then, the only launch platform with regular (i.e. quarterly) launch opportunities to a consistent orbit is the Station and its heterogeneous fleet of cargo resupply vehicles.

ISS deployment differs from traditional launch and presents unique considerations that impact design standards, fault tolerance, environments, and the concept of operations. For example, ISS deployment involves launching the payload containerized as pressurized cargo in a resupply vehicle and then deploying it from Station via an airlock. Obviously, this process involves astronauts physically interacting with the spacecraft once aboard Station, which invokes a set of concomitant safety and human-factor considerations. Because of these unique challenges, and because the ISS has only recently been used a launch platform in any capacity, ISS deployment of a microsatellite has never been performed.

This paper presents the lessons learned from planning this mission, specifically:

- defining and complying with requirements imposed by NASA, the Launch Vehicle Service Provider, and Secondary Launch Integrator
- discussing the most challenging of these requirements and our approach to satisfying them
- explaining timelines for gaining the necessary approvals and our personal experience in navigating the various approval bodies

2. MISSION AND PROCESS OVERVIEW

Andrews Space developed and produced the spacecraft: a 3-axis stabilized imaging microsatellite. Our customer contracted for launch with our sister company, Spaceflight, Inc., who manifested the mission, provides mission management services, and interfaces with the designated ISS Secondary Launch Integrator. The Secondary Launch Integrator is primarily responsible for working with the NASA safety boards and other ISS stakeholders to facilitate the approval process and deployment logistics.

Unlike most satellite deployments, which are automated from lift-off, Station-deployed satellites pass through astronaut hands. Many of the required processes are not yet formally documented, or even informally codified as tribal knowledge. Instead, engineers leverage their familiarity with human-rated spaceflight and ISS deployment processes for CubeSats as precedent.

A good example of this approach is how the *Space Shuttle Program Payload Verification Document (NSTS 14046)* [1] was leveraged; specifically, the use of Phase I, II, and III Payload Safety Reviews, which are defined in *NSTS 13830* [2] and in *SSP 30599* [3] for International Space Station Program (ISSP) cargo elements:

> The Phase I safety assessment report must identify payload systems having catastrophic hazard potential and reflect the verification approach proposed to confirm intended system performance.
>
> The Phase II safety assessment report must contain a verification plan(s) which identifies the test and analytical efforts required to verify intended hardware performance for all systems, with operational hazard potential. The plan(s) must identify the basic content of the test and/or analysis effort along with a summary of the pass/fail criteria and simplified end-to-end schematics/diagrams depicting electrical, mechanical, fluid and software controlled interfaces with clear and consistent nomenclature.
>
> The Phase III safety assessment report shall summarize the results achieved by the verification activity and compare the results from all independent verification activities.

These safety reviews, and the overall mission planning and approval processes, take significant time and evaluate increasingly detailed spacecraft requirements, design implementation, and verification activities. Table 1 shows the safety review milestones in the context of spacecraft delivery, launch, and deployment.

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[1] http://spaceflightservices.com/manifest-schedule/

[2] For example: DARPA's ALASA (http://www.darpa.mil/Our_Work/TTO/Programs/Airborne_Launch_Assist_Space_Access_(ALASA).aspx), Virgin Galactic's LauncherOne (http://www.virgingalactic.com/LauncherOne), US Army SMDC's SWORDS (http://www.smdc.army.mil/FactSheets/SWORDS.pdf), the Super Strypi, Rocket Lab (http://www.rocketlabusa.com/) and Firefly (http://www.fireflyspace.com/vehicles/firefly-a). Disclosure: the author has either worked with principle members of or worked on most of these vehicles.

Table 1. Nominal Mission Planning Timeline

Event	Date (in months)
Phase II data submittal	PII – 2
Phase III data submittal	PIII – 2
Human-factors Test	PIII – 0 to 2
Delivery to SLI	PIII – 0 to 2
Phase I review (PI)	L – 6 to 12
Launch contract signed	L – 12
Phase II review (PII)	L – 3 to 6
Phase III review (PIII)	L – 1 to 2
Delivery to NASA	L – 1 to 2
Launch (L)	L – 0
Deployment	L + 1 to 3

Phase I is the least official review; it determines a spacecraft's suitability for ISS deployment before a launch contract is signed. Data is exchanged to facilitate a catastrophic hazards assessment and identify feasible mitigation plans. If NASA verifies that all catastrophic hazards have been identified and acceptable mitigation plans have been proposed, and if the payload is compatible with the ISS deployment process, a contract can be signed. This process can occur as long as 12 months before launch or just prior to the Phase II review.

Around L–8 months, the Phase II information is submitted to the Safety Review Panel, which begins an iterative process where information gaps are identified and filled in. The Phase II review can be as early as L–6 months, but for some CubeSat missions has been as late as L–3 months.

Between the Phase II and III reviews the spacecraft is delivered to the Secondary Launch Integrator and integrated with the separation system and launch container.[1] At this stage (as a deviation from the CubeSat process) a Human-Factors Test (HFIT) is performed. CubeSats are entirely encapsulated in deployers and approved to go through Station, but our spacecraft is large enough it has to be removed from the launch container by astronauts to fit in the airlock. This leads to additional requirements for sharp edges and other handling concerns.

Acceptance of the containerized spacecraft by NASA is linked to passing the Phase III review. The most important Phase III spacecraft deliverables are environmental test reports that demonstrate compliance with the hazard mitigation approaches. As with Phase II, the Phase III data package is submitted up to two months before the review, with the Phase III review occurring one to two months before launch.

After satellites pass Phase III, the containerized spacecraft is delivered to NASA for launch vehicle integration. After launch, the containerized spacecraft is unloaded from the cargo delivery capsule and onto Station. Time between unloading and deployment is one to three months, where the duration and uncertainly stem from planning a deployment's complex logistics around the spacewalk and visiting vehicle schedule.

All of these activities are discussed below in greater detail, but for reasons that will become obvious we start with the safety reviews.

3. PHASE I – A CAUTIOUS PARADIGM

It is prudent, given that human safety is involved, that the safety review and approval process for this mission has been exceedingly cautious. Those interested in future ISS missions should understand this perspective at the beginning of their design process because it affects fundamental design decisions in profound ways. This section provides a high-level overview of a reductive but useful paradigm for understanding ISS-unique requirements, outlines the specific methodology for identification of potential hazards, and describes several design choices that should be considered non-starters from an approval perspective.

Start with a cautious paradigm, because this framework makes sense of many otherwise inexplicably onerous requirements; more specifically, assume that NASA's top priority is human safety, that the next is Station safety, and that there is no third priority. All parties value successful deployment of the payload, but NASA's charter with respect to this mission is protection of its astronauts and Station—they have approval authority and their priorities are non-negotiable. It is from this perspective that the first step in assessing a vehicle's suitability for ISS deployment begins: the Phase I hazard assessment.

Catastrophic Hazard Categories

The primary purpose of the Phase I safety review is to identify potential sources of catastrophic hazard and review the proposed mitigation plans. Mitigations must be dual-fault tolerant (i.e. two inhibits can fail and the hazard will still not manifest). We were never provided with a comprehensive list of potential hazard categories. Instead, a series of discussions about the spacecraft design identified several areas that needed to be addressed by the Phase I package. We have grouped these items into several categories:

Intentional RF emissions—Transmitters from 14 kHz (Very Low Frequency, VLF) to 15.2 GHz (J-band) are potential hazards.[2] The NASA *OE/Manager, ISS Safety Review Panels* (SRP*)* distributed a letter on February 4, 2014 clarifying the Panel's policy that defines intentional Radio Frequency (RF) hazards.[4] The technical details of this memo are not reproduced here, as they are subject to change

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[1] As of October 2014, we are between the Phase II and Phase III reviews.

[2] Frequencies above J-band are not specifically mentioned in the referenced memo; however, this should not be interpreted as implying no hazards exists but rather that no specific hazard criteria have been identified.

and should be independently verified as current, but the letter provides thresholds of electric field strength (e.g. V/m), power density (e.g. W/m^2), maximum radiated power (e.g. W), and contact current (e.g. mA) as a function of several RF bands. Separate criteria are provided for ISS crew radiation and ISS hardware, but both sets of criteria must be met.

Lasers—Our spacecraft does not contain any laser emissions, so this potential hazard was not discussed, but a consistent application of our cautious paradigm, and knowing that warning labels are printed even on low-power laser pointers used for presentations, suggests any laser source would be considered a hazard.

Heaters—No quantitative guidance was given on heaters, but any temperature that could burn human flesh after extended contact was considered a hazard.

Fluids— For fluids that are widely recognized as hazardous and that require peculiar support equipment and suits to handle safely (e.g. hydrazine), see the subsection below: *Design Non-starters*. But even seemingly benign fluids can be hazards: high-pressure inert gasses (or sources of high-pressure gasses like solid gas generators) are an explosion risk, low-pressure inert gasses are still potential asphyxiants, and even water was theorized as a hazard (for reasons that remain unclear) in sufficient volume (in our case several liters).

As we will see in the next section, all of the aforementioned hazard categories are best mitigated with a dual-fault tolerate power system isolating scheme. Fluids are the first special case of hazards because in addition to power system isolation the fluid containment system much also be assessed for mechanical failure.

For example, if properly constrained against becoming a projectile, a commercial CO2 cartridge would likely be acceptable.[1] But a propulsion system with valves, manifolds, and tubing runs would require proof pressure and leak testing, and any valves require two-fault mechanical tolerance (e.g. fill and drain valves staked closed, serially plumbed solenoid and/or latching valves).

In short, any attempt to fly a propulsion system should involve close coordination with all parties as early in the design process as possible and will require continual review from the level of propulsion system schematics and conceptual design through acceptance testing of the flight unit.

Deployables—Common examples of deployable mechanisms are solar panels, payload covers, and antennas. The hazard is unintentional deployment that could hit and injure an astronaut, hit and damage something within the ISS or generate debris, or deploy when the spacecraft is in the airlock or outside the Station but as not yet been released. The last case is a hazard if the accidental deployment would prevent the vehicle from being brought back inside Station because it no longer fits inside the airlock.

Deployables are also an issue if the mechanism is not easily reset (i.e. if reset requires specialized tools not available on Station or if reset cannot easily be done in a zero-g environment by untrained personnel, such as replacing burn wire). Resetability is important because even if unintentional deployment inside Station did no proximate harm it would likely halt the mission and effectively strand the vehicle inside Station.

Deployable systems, like fluids, will likely require mechanical testing to verify robustness in addition to the mitigation against accidental deployment that is provided by a power isolation scheme.

Batteries—Batteries are a unique hazard because compared to the previously mentioned hazards the requirements are more thoroughly documented. The Propulsion and Power Division of the Engineering Directorate at NASA Johnson Space Center published *EP-WI-032 Statement of Work Engineering Evaluation, Qualification and Flight Acceptance Tests for Lithium-ion Cells and Battery Packs for Small Satellite Systems* (henceforth: *Battery SOW*).[5] This document lays out a comprehensive test program for lithium-ion rechargeable batteries—a very common solution for small satellites because Li-ion batteries have high energy density, desirable recharge, and long-life when compared to other battery technologies. These features have also made them ubiquitous in consumer electronics, which has created a market for low-cost and small-form-factor products and a robust industrial base of cell manufacturers.

Fundamentally, the purpose of the *Battery SOW* is to demonstrate that the battery and/or individual cells contain protection circuitry that prevents various off-nominal conditions from damaging the battery (leak, fire, and explosion[2] are potential failures). In addition to an engineering review of the protection circuitry design, the *Battery SOW* prescribes a set of qualification tests[3] and acceptance tests.[4] The electrical tests are:

- battery overcharge: demonstrates the protection circuit prevents charging when a fully charged cell is exposed to a charging voltage
- battery overdischarge: demonstrates the protection circuit prevents discharge to 0 V and the battery from going into reversal

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[1] The Southern Stars Sky Cube 1U cubesat contained a commercial-grade CO2 cartridge to inflate a deorbit balloon; Skycube was deployed from the ISS in January of 2014. http://www.southernstars.com/skycube/

[2] http://xkcd.com/651/

[3] tests performed on flight-like hardware that is not intended for flight

[4] tests performed on flight hardware

- external short: demonstrates the protection circuit prevents an external short from damaging the battery (note: if unsuccessful, this test can damage the battery and be dangerous; the *Battery SOW* prudently recommends this test and others be performed in an abuse chamber)
- a set of cell tests to be performed on units from the same lot as the cells used in the flight battery
  - cell external short
  - cell overcharge
  - cell over-discharge
  - cell (simulated) internal short
  - cell heat-to-vent
  - cell vent and burst pressure test

To pass, the battery / cells must exhibit nominal electrical characteristics (i.e. open-circuit voltage, closed-circuit voltage, and temperature) before and after each test. The stress tests are also interspersed with a set of nominal charge and discharge cycles. Lastly, the *Battery SOW* identifies a set of environmental qualification and acceptance tests.

Thermal cycle qualification testing performs charge and discharge cycles at the worst-case hot, worst-case cold, and ambient temperature. Thermal cycle acceptance testing is a 1 hour cold soak at 0°C, 2°C/minute ramp to 65°C, and 1 hour hot soak, performed twice.

Random vibration qualification and acceptance tests are specified at slightly different test levels and durations, as shown in Table 2 and Table 3 below.

Table 2. Qualification Random Vibe Levels

Freq (Hz)	ASD (G^2/Hz)	dB/Oct	Grms
20	0.05760	n/a	n/a
40	0.05760	0.00	1.07
70	0.14400	4.93	2.02
700	0.14400	0.00	9.74
2000	0.03744	-3.86	13.65

Table 3. Acceptance Random Vibe Levels

Freq (Hz)	ASD (G^2/Hz)	dB/Oct	Grms
20	0.02280	n/a	n/a
40	0.02280	0.00	0.76
70	0.07200	4.93	1.43
700	0.07200	0.00	6.89
2000	0.01872	-3.86	9.65

Thermal vacuum (TVAC) testing is specified in two representative environments: Intra-Vehicular Activity (IVA) and Extra-Vehicular Activity (EVA). The IVA test pumps down[1] a battery and maintains it at a vacuum of better than 0.1 psia for 6 hours. The battery is weighed to ensure no leaking or venting has occurred and visually inspected for damage such as bulging or deformation. A charge-discharge cycle is also performed to confirm nominal post-test electrical performance. The EVA test is identical except it is performed to a hard vacuum level of better than 1e-5 Torr. For TVAC, the qualification and acceptance test criteria are identical.

Operational Hazards—The hazards above have focused on the spacecraft's physical configuration, but the concept of operations is also considered. We have received verbal guidance that the spacecraft should be non-operational for at least 30 minutes post-deployment. Ultimately, the timer value will be assessed by the safety panel based on the expected separation rate, expected maneuvers or deployment events that occur after start-up, and the radiated power spectrum and density.

If there is any propulsion capability with a ΔV larger than the separation velocity, also expect scrutiny of the how and when the propulsion system can be operated.

Design Non-starters

There are several categories of hazards we intentionally avoided in our spacecraft design, even before we began detailed coordination for an ISS deployment, because we assumed it would not be worth the (potentially futile) effort to attempt to get them approved by any launch vehicle (as a secondary payload). The items below clearly violate the prime directive of this section to value safety above all else.

Hazardous fluids—hydrazine is a common hazardous fluid in aerospace applications; it was used on the Shuttle, so obviously NASA has experience in its use in human-rated vehicles (although not in the habitable volume). However, that does not mean NASA will let you use it on your microsatellite. Highly flammable, hypergolic, poisonous, or carcinogenic fluids should be avoided.

Pyrotechnics or other explosives—a best practice is to use non-explosive actuators (NEA). The rationale is two-fold. First, NEAs are not an explosive hazard and don't generate high-energy projectiles that require active capture (like explosive bolts often do). Second, most NEAs are resettable, which enables the mechanism to be demonstrated and then reset. (Note: burn-wire is a good non-explosive option, but is not easily resettable.)

4. PHASE I – SATELLITE HAZARD ASSESSMENT

Our satellite was originally designed to be manifest as a secondary payload on an unmanned vehicle and was past

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[1] The TVAC tests specify a pump down rate of 8 psi/min, which is strange because vacuum chambers pump down non-linearly. Initially the rate may be 8 psi/min, but as the chamber achieves higher levels of vacuum the pump-down rate asymptotically approaches zero.

critical design review maturity before we began close coordination with the Secondary Launch Integrator. As a result, we were initially non-compliant with some of the ISS-specific design requirements described above. This section describes the hazards we identified and advice for those starting with a blanker slate.

First, some high-level lessons learned:

- Avoid design non-starters.
- Obtain early design guidance for any propellants, pressurized gasses, or gas generators.
- Select batteries and/or cells with previous NASA spaceflight heritage. The list is long and changes, so our best advice is to obtain a current list from a launch service provider who works with NASA. Once ISS launch has become more established there will also be a base of vendors and suppliers whose hardware flew on previous ISS missions.

Hazard Assessment

The Phase I submittal included a hazard assessment, organized below in the categories presented above in § 3.

Intentional RF emissions—Our spacecraft has X-band and UHF transmitters. The 2 W UHF transmitter was not a hazard. The 8 W X-band transmitter exceeded the threshold of several requirements by at least an order of magnitude. Our Phase II package included an engineering memo that documented various communication system parameters, calculated the emission environments, and compared them to the hazard levels criteria.

Heaters—The propulsion system uses several resistive heaters. The nominal operational temperature is over 200°C.

Fluids—The baseline propellant is butane. The system is self-pressurized to a few atmospheres at the hottest expected operating temperature, so large factors of safety in the pressure vessels were easy to demonstrate. However in the oxygenated Shuttle environment butane is flammable[1] and an asphyxiant. The design did not have dual-fault tolerance against mechanical failure of the electrically actuated non-latching solenoid valve that releases propellant. The mechanical fill/drain and vent valves are torqued and staked to provide two locking features.

Deployables—The spacecraft has a payload cover and UHF antenna that deploy as a single event. The Phase II submittal includes figures depicting the mechanical design, specifications of the spring force, and the part number for the commercial-off-the-shelf pin-puller (a type of NEA) mechanism.

Propulsion system—Propulsion represents a hazard beyond the previously mentioned heaters and fluid because it could be used to boost the spacecraft above the ISS orbit or cause collision if not properly controlled.

Battery—The power system contains a 216 W-Hr Li-Ion battery. Schematics were provided showing the battery protection circuitry and the qualification and acceptance test plans.

Human Factors Implementation Team (HFIT)

HFIT is concerned with personnel who handle the spacecraft and considers issues such as sharp edges, hold points, and frangible materials.

Our standard drawing notes were provided to demonstrate that standard practice was to break and de-burr sharp edges and identify any potentially sharp surfaces. In our case, these were the knife-edges on the payload and star tracker baffles and the edges of the solar cells.

3. PHASE II – A DESIGN APPROACH

The purpose of the Phase II review is to present the detailed design approach to eliminate or mitigate the hazards presented at Phase I.

Power System Isolation

A significant number of the hazards are mitigated by a single design change: implementing dual-fault tolerant isolation of the solar arrays and battery from the avionics. This approach mitigates all hazards that require electrical power to manifest, which in our case are: premature deployment from errant pin puller activation, RF emission, heater activation, and propellant release.

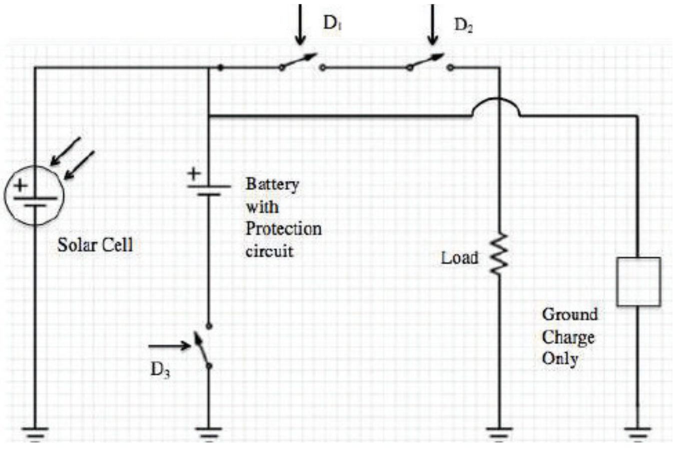

Figure 1 – Power Isolation Block Diagram (image courtesy of NanoRacks; source: NR-SRD-029)

Figure 1 is a reference implementation of two high-side and one low-side inhibits that isolate the batteries and solar arrays from the rest of the electronics. The system is dual-fault tolerant because two of the three switches can fail (i.e. close when they are not supposed to close) and the system

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[1] Once released, butane quickly dissipates to concentrations too sparse to ignite, but as demonstrated by a cigarette lighter it can sustain a flame at a leak point. Also, most cigarette lights don't carry 8 liters of butane.

will remain unpowered. We encountered several challenges with a naïve implementation of this approach:

- Our spacecraft consumes over 100 W at peak power draw. For a 7.8 VDC battery, this is over 12 A of current. This is far beyond the steady-state current rating of the switches in the flyaway portion of the separation system.
- The avionics are roughly in the geometric center of the spacecraft, while the separation system (and aforementioned switches) are at the base. Round-trip harness runs from the battery and solar panels to the switches and back to the avionics would exceed 100 cm, which adds excessive harness mass and voltage drop in the power supply lines.
- An obvious downside of the reference implementation is that each switch (D1, D2, and D3) represents a single point of failure. The cautious paradigm tacitly trades mission assurance for safety.

Our approach was to implement the spirit of the schematic in a new Battery and Array Isolation Box (see Figure 2). This box mounts above and interfaces directly with the existing battery enclosure and connector. Additionally, the Isolation Box has six leads that connect to the positive and negative terminals of three separation switches at the deployment interface; it also connects to all the solar array strings and provides power to the rest of the electronics.

High-power MOSFETS are used as switches within the Isolation Box. Each array and battery input is isolated from the avionics supply by two strings of two MOSFETS on the high side and two parallel MOSFETS on the low to provide redundancy against MOSFET failure. Each MOSFET is controlled by one of the three separation switches so that power is not connected until three serial switches close.

Heaters

In addition to the protection provided by the power system isolation, we also showed via finite-element thermal analysis that the heaters were buried deep inside the vehicle and thermally isolated such that even if operating at the maximum operational temperature no exposed surfaces would be dangerously hot.

Propulsion Capability and Butane Propellant

A schematic of the microsatellite propulsion system is shown in Figure 3, which identifies a single electro-mechanical valve that controls propellant release.

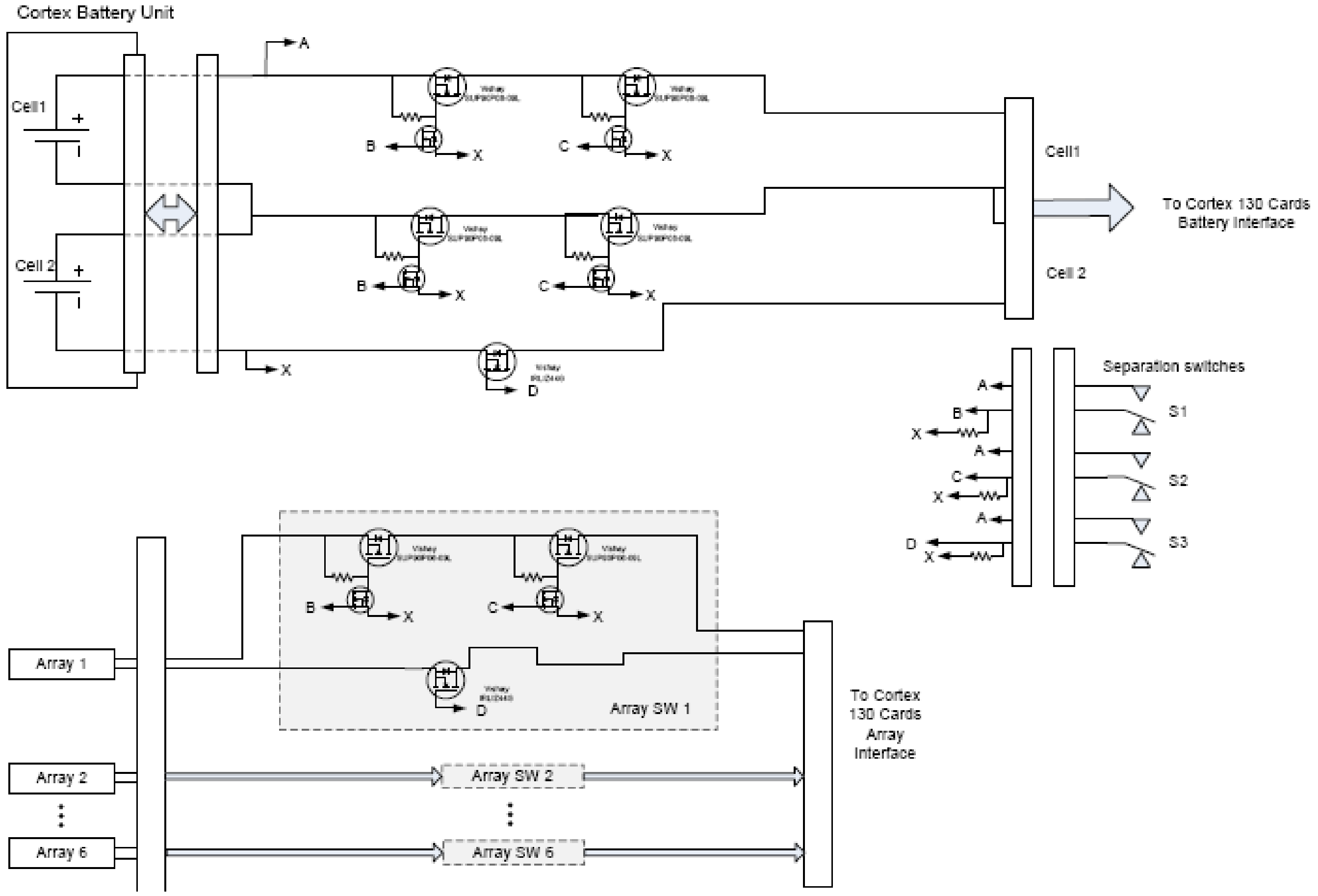

Figure 2. Power Isolation System Electrical Schematic

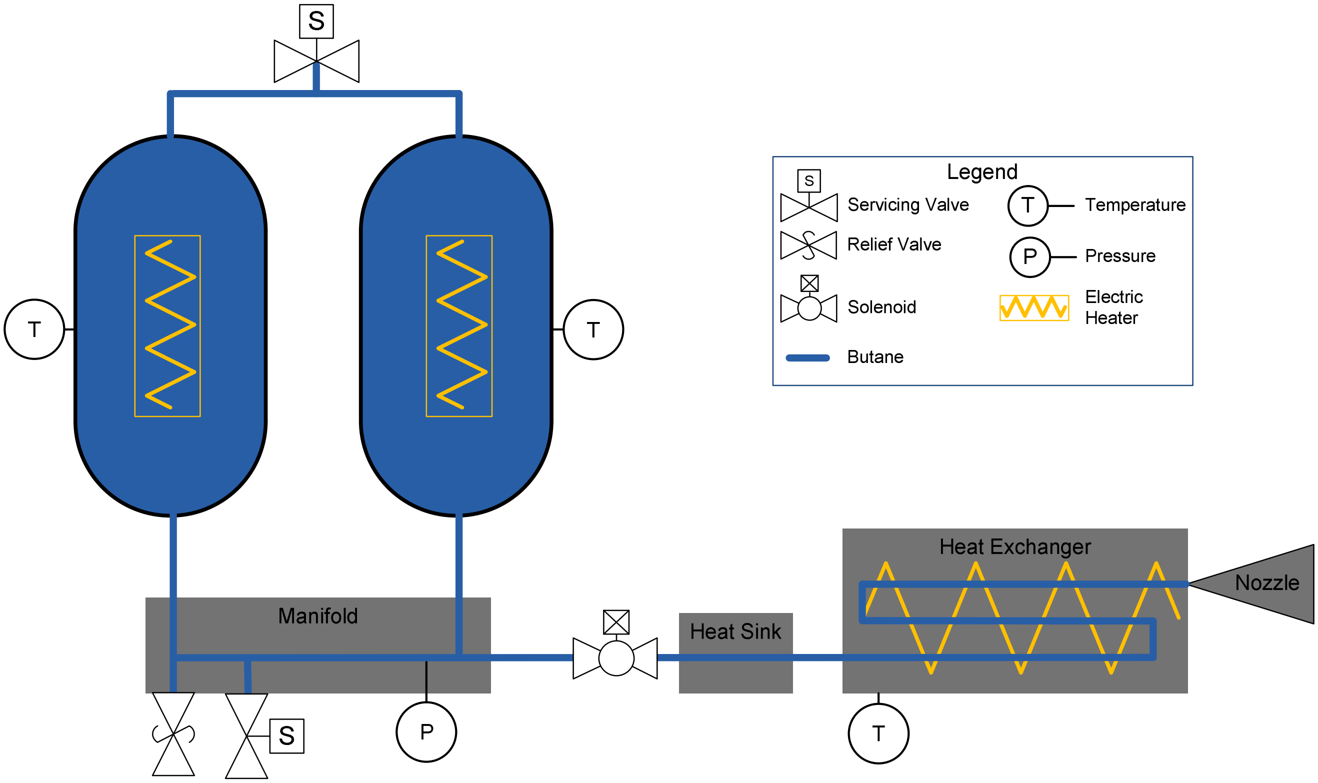

Figure 3. Propulsion System Schematic

Based on early coordination it was determined there was insufficient time to work with the safety panel and obtain approval to use propulsion, much less with butane propellant. That said, this early coordination did not identify any showstoppers, there was simply insufficient time to fully partner a solution before the manifested launch date. Future mission planners should take note to allow extra time to obtain approval for propulsive satellites.

There were two unresolved concerns: (1) providing dual-fault tolerance against mechanical leaks in the electrically actuated valve, and (2) providing suitable assurances that the propulsion system could not accidently put the spacecraft on a collision trajectory with the ISS.

The current approach is to forego loading propellant on the maiden ISS deployment and continue partnering resolution of outstanding issues so propulsion can be used on future missions. We continue to work propulsion approval and are pursuing two complementary mitigation strategies to address the current open concerns.

First, we propose addressing incorrect pointing during propulsion maneuvers by ensuring the spacecraft cannot autonomously plan or execute propulsive maneuvers. Second, we propose performing pre-maneuver analysis to show the resulting trajectory would be safe for a worst-case thrust orientation, similar to the type of range safety analysis performed by launch vehicle providers.

Adding additional valves can provide dual-fault tolerance of the electro-mechanical elements. Another option is to advocate that the power isolation system provides dual-fault tolerance against electrical activation and that leak and environmental testing demonstrates mechanical robustness.

Environmental Testing

Qualification and acceptance testing of the design in the relevant quasi-static load and random vibration environments serves several purposes: it qualifies the mechanical design to the launch environments, demonstrates the deployable mechanisms are unlikely to experience a non-electrical failure causing them to deploy during launch, and provides additional confidence in the battery-level acceptance vibration testing described in the previous section.

Battery Protection

The baseline battery was space-qualified[1] and used space-proven cells,[2] but it did not include native protection against over-current, over-charge, or over-discharge. Protection against those failures was allocated to the avionics (or electronic ground-support equipment) responsible for cell charging, discharging, and balancing.

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[1] http://andrews-space.com/news-blog/2013/6/12/cortex-battery-unit-completes-space-qualification

[2] http://yardney.com/lithium-ion-2/ncp25-1/

This approach was rejected during initial partnering of the Phase II package because some of the protections were implemented in software and/or FPGA logic, and relying on this solution would have required an independent audit of all the flight software and embedded logic (i.e. it was actually less expensive and less time-consuming to design analog protection circuits directly into the battery than audit all of the code).

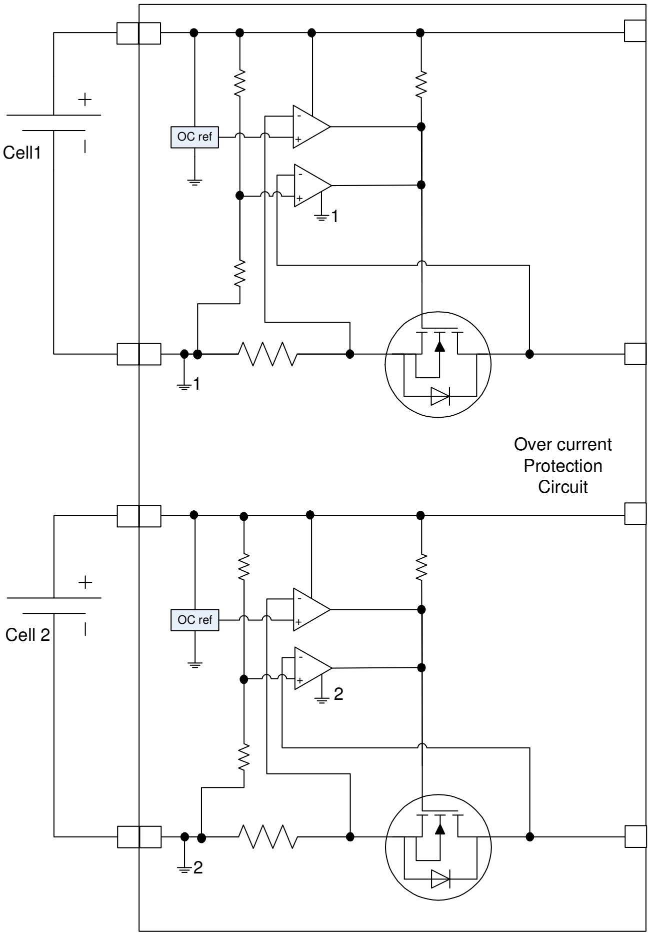

Figure 4. Battery Protection Circuit

Our alternative approach was to modify the battery design to include a protection circuit, which performs three functions:

- detects over-current conditions and disconnects the cells from load; reconnects cells if/when short-circuit is cleared
- detects under-voltage conditions and disconnects the cells from load; reconnects cells if/when voltage recovers to a specified level
- detects over-voltage conditions and disconnects the cells from the charging source; reconnects cells if/when charge voltage recovers to a specified level

The set points are established by the circuit design and resistor values, and the switching and power isolation is provided by high-power MOSFETS (see Figure 4).

This new "smart" battery unit is being put through a delta-qualification program to verify compliance with the requirements in § 2 (i.e. charge, discharge, over-current, over-voltage, under-voltage, vibration, thermal cycle, and vacuum).

Battery Testing

The battery test approach described in § 3 was written from the perspective of approving AA-class Li-ion cells (typical for CubeSats).[3] Our microsatellite also uses Li-ion battery technology, but in a much larger form-factor that is specifically designed for space applications. As such, some of the identified tests were either not applicable or impractical to perform.

Our approach interpreted the intent of the *Battery SOW* and used that understanding to generate a qualification and acceptance test plan more appropriate for our cells and battery, for example:

- the cells have been previously space-qualified and used in numerous NASA missions; as such, pre-existing safety data was available on these cells, which obviates the need for cell-level testing
- the nominal charge and discharge profiles specified in the Battery SOW were replaced with the profiles recommended by the cell manufacturer
- the separate IVA and EVA thermal vacuum test were combined into a single test at the more stringent vacuum levels

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Several approaches were used to address handling:

- We provided a document that describes the vehicle hold points and lifting and handling procedures.
- A complete materials list helps identify frangible materials, which in our case are glass in the star tracker lenses (recessed), payload telescope mirrors (covered), and over the solar cells (exposed). The list is also used to assess materials used for outgassing compliance (see *Outgassing* below).
- The shipping container, designed by the Secondary Launch Provider, was outfitted with handles so that while containerized it can be maneuvered by hand during pre-launch and on-orbit operations.

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[3] http://www.cubesatshop.com/ → Power Systems →Power Supplies

Other Design Considerations

The above subsections focused on hazard identification, but several additional design considerations were discussed in the Phase II package.

Launch environments—compared to typical launch vehicle environments, the journey to Station as pressurized cargo on a commercial resupply vessel is a relatively benign dynamic environment. This is because instead of being hard-mounted to a rigid surface and thus directly exposed to acoustic, shock, and random vibration environments, these environments are attenuated through the cargo vessel and container housing the spacecraft.[4] That said, it is still recommended to design to typical secondary spacecraft environments, such as those described in *GSFC-STD-7000, General Environmental Verification Standard (GEVS).*[6]. Certification that your satellite is compatible to launch environments such as GEVS or those specified for a given launch vehicle is a typical deliverable for most Launch Service Providers; this is not unique to an ISS deployment.

From delivery to deployment—it can often be months between when a spacecraft is encapsulated in the launch container (which obscures ground support equipment connections and the ability to charge batteries) and when the satellite is deployed. Satellite delivery is nominally two months before launch, launches slip, and ISS deployment could occur months after the satellite arrives. Quiescent power draw in the storage and launch configuration needs to be exceedingly small and batteries sized accordingly. Analysis documenting battery encapsulation time is a typical deliverable for most Launch Service Providers; this is not unique to ISS deployment. However, expect rigorous review of associated analysis for this deliverable due to the extended time a satellite can remain on Station prior to ISS deployment and as a design practice ensure ample margin.

Outgassing—a complete materials list supports an independent out-gassing assessment. The list includes all materials used, total mass, total surface area, and, where applicable, manufacturer and surface treatment. A materials list as described is a routine deliverable for most Launch Service Providers; this is not unique to an ISS deployment. The most straightforward way to generate this information is from a CAD model, but it can be time consuming to compile even if the model is complete and thorough. For materials selection during the design process we use < 1% total mass loss (TML) and < 0.1% Collected Volatile Condensable Materials (CVCM) per ASTM E595.[10] Generally, materials can be selected from data sheets or widely published parameters that meet these criteria to obviate the need for and cost of independent testing and measurement.

Orientation during deployment—components (e.g. telescopes) that are sensitive to direct solar illumination should have covers or other means of protecting them while the ISS Robotic Arm[5] is positioning the spacecraft for deployment. This operation could take hours, and given a 90 minute nominal orbit the spacecraft could be sunlit in an arbitrary orientation prior to deployment (and post deployment until the tip-off rates are under control).

Our design incorporates a deployable cover to protect the payload. The star tracker apertures are not covered. Instead, an analysis was performed to estimate the likelihood, duration, and incidence of potential solar exclusion angle violation. Separate testing will be used to demonstrate robustness of the star trackers while they are in the pre-deployment (i.e. unpowered) state under the predicted conditions. Note this is a mission assurance and risk reduction test, not safety related.

Tip-off rates—the spacecraft is mated to a separation system designed to egress Station via an airlock. Once outside, the entire assembly is grappled by a robotic arm and maneuvered for deployment. The stack-up of the flexibility of the arm, alignment of the separation force with the spacecraft center-of-mass, and spacecraft center-of-mass uncertainty results in estimated tip-off rates of several degrees per second. If three-axis controlled, the spacecraft must be able to null these rates in an acceptable timeframe. Several degrees per second is at the high end of typical micro-satellite agility, so de-saturating this much momentum likely requires use of magnetic torque rods or another momentum desaturation device, which depending on how they are sized could take several orbits (during which the spacecraft may be collecting less than typical solar power).

We performed a Monte Carlo analysis using a 6 degree-of-freedom simulation to demonstrate that the vehicle control system would be able to recover and null body rates while maintaining a healthy margin on the battery state-of-charge. The stochastic variables in the simulation were the position of the spacecraft in the orbit[6], initial vehicle attitude, and tip-off rates. Over 1,000 simulation runs, the results showed the spacecraft body rates recovered to within 0.1 deg/sec (our criteria for cover deployment) within two hours of deployment. The worst case resulted in less than 10 percent discharge of the battery.

A key lesson for other spacecraft is to perform tip-off analysis early and incorporate the results into the design. As

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[4] For launch, SCOUT will be housed in a custom-design aluminum enclosure that is cantilevered from the separation system and completely encloses the spacecraft. This rigid box is then strapped down to the inside of the pressurized cargo hold. This was done because were no hard-points on the interior surfaces of the cargo vessel to which SCOUT could bolted, as it would be on a traditional launch vehicle. The box is cantilevered (rather than being supported by the spacecraft proper) because SCOUT was designed to carry launch loads through it's based and not through other parts of the structure.

[5] In our case the robotic arm built by Canada.

[6] It was important to evaluate a diversity of orbits locations (i.e. true anomalies) because the relative orientation of the vehicle's angular momentum and the local magnetic field vectors determine the direction and magnitude of the external torque generated by the torque rods

our spacecraft was already designed, the analysis made us glad for robust margins on the effectivity of our torque rods and capacity of our battery, and indicated no design changes were necessary.

Post-deployment timeline—not mentioned during the Phase I review was a requirement that the spacecraft should have a timer that prevents power-up for so some period after deployment (nominally 30 minutes). During this window the vehicle should not initiate attitude control maneuvers, deploy anything, vent, radiate, etc. For Phase II, we documented an existing feature of our avionics that delays power-up of the flight computer until a user-adjustable time after the array and battery isolation is removed (i.e. a cold-start timer). Note that batteries should be sized for any quiescent power draw during this cold-start waiting period.

Orbital life—this requirement is not specific to ISS deployment, but as with any other launch an orbital debris assessment must be performed in accordance with NASA's Process for Limiting Orbital Debris.[8] Because of its relatively low altitude, on-orbit life is one of the easiest requirements to meet. A spacecraft with a ballistic coefficient between 25–60 kg/m² can expect to re-enter in 3 months to 3 years due to drag alone. This analysis was performed for solar min and solar max conditions based on historical data from 1957 to 2007.

3. PHASE III – TRUST BUT VERIFY

The purpose of the Phase III review is to present data verifying the Phase II mitigation approaches have been implemented. The Phase III package is not yet completed for our mission at the time of writing, and some of the supporting design and test activities are ongoing. This section presents the expected content of the Phase III package.

Power System Isolation

As-built schematic of the Power Isolation System, the qualification report, and the flight unit acceptance report. The test reports will include results of the electrical and environmental testing.

Battery Protection

As-built schematic of the modified battery design, the qualification report, and the flight unit acceptance report. The test reports will include results of the electrical and environmental testing.

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Completion of an HFIT review. This review will encompass a full review of the handling procedures during launch integration, unloading of cargo, integrating the spacecraft with the separation system, and egress from Station via airlock.

Spacecraft Acceptance

Spacecraft acceptance test report, which includes the results from spacecraft-level environmental testing and results of a day-in-the-life simulation performed with the as-built satellite to demonstrate proper operation of the integrated power system (i.e. Power Isolation System, smart battery, and spacecraft avionics), the cold-start timer, and cover deployment.

Final review of as-built BOM to check for consistency with previously provided materials list to assess frangibles and outgassing compliance.

4. PATH FORWARD

Timelines and Approvals

The Phase I content was submitted to the NASA Safety Panel several weeks prior to the review. The submitted documentation included our complete hazard assessment and schematics of our proposed hazard mitigation approaches.

Phase II Content was submitted a month prior to the review in fall of 2014. It included a complete set of hazard mitigation approaching included electrical schematics, design information like a materials lists, results of tip-off and other analyses, and documentation of the post-deployment concept of operations.

As of October 2014, the Phase III content is being prepared to submit 1–2 months prior to launch.

Hardware Delivery and Integration

We plan to deliver our satellite to the Secondary Launch Integrator two months prior to launch. Integration into the launch and shipping container will occur at our production facility and then ship directly to the launch integration facility. The container containing our satellite will be loaded and unloaded with the other pressurized cargo as part of normal cargo resupply operations. The satellite may spend weeks or months on orbit after launch before deployment, based on the scheduling of other Station events and availability of personnel. At the time of writing, deployment is anticipated in the second half of 2015.

4. CONCLUSIONS

For the foreseeable future, launch options for small satellites will remain sparse. Deployment from the ISS represents a tantalizingly frequent and reliable opportunity, but like any new space endeavor bears significant challenges that must be addressed and overcome. These challenges primarily stem from human-rated safety considerations that are to be taken seriously and merit the requisite effort to demonstrate compliance to the associated requirements and safety precautions. The timeline for an ISS deployment is similar to a traditional launch, up to the point of launch, and then diverges, as the satellite remains aboard Station for an additional period of time until it is unpackaged and deployed.

We categorized the hazards that should be considered early in the design process to avoid costly design changes approaching launch. Our satellite design was complete at the time we manifested an ISS deployment, which drove our hazard mitigation and verification approaches to focus on low-schedule-risk solutions, and the balance of risk favored—by necessity—ISS safety assurance over satellite mission assurance. Case-in-point, our decision to forego the use of the existing propulsion system for the maiden deployment. We also focused on Power System Isolation because if improperly implemented it can drive costly rework, and be a source for a variety of hazards such as RF emission, deployables, and heaters. If smartly implemented, Power System Isolation can be a panacea for addressing the very same hazards and your salvation with NASA safety boards.

Other important considerations include analyses to improve mission assurance, such as considering the design implications of sun incidence, high tip-off rates, and the post-deployment timeline. These factors can size batteries and attitude control effectors, drive incorporation of design features like covers, and levy requirements on the initialization sequence and flight software.

Future work includes completing the third phase of the safety review process, the HFIT review, and navigating the challenges we're sure to encounter during final launch integration. To support future missions we continue to work on gaining approval to operate a propulsion system.

Hopefully, our lessons learned from this experience will help future spacecraft developers to design-in success early and more smoothly navigate the nascent processes to leverage this international asset. As a body of evidence and is built by our experience and others, solutions and generally accepted best practices will emerge for deployments from the ISS as they have for other traditional launch vehicles.

REFERENCES

[1] NSTS 14046, Space Shuttle Program Payload Verification Document, Rev E, dated March 2000.

[2] NSTS/ISS 13830, Payload Safety Review and Data Submittal Requirements, Rev C, dated July 1998.

[3] SSP 30599, Safety Review Process, Rev D, Dated September 2006.

[4] OE-14-002, Documentation of Intentional Radio Frequency (RF) Hazards, memorandum dated 2014-02-04.

[5] EP-WI-032, Engineering Evaluation, Qualification and Flight Acceptance Tests for Lithium-ion Cells and Battery Packs for Small Satellite Systems, February 20, 2014.

[6] GSFC-STD-7000, General Environmental Verification Standard (GEVS), dated April 2005.

[7] ASTM E595, Standard Test Method for Total Mass Loss and Collected Volatile Condensable Materials from Outgassing in a Vacuum Environment.

[8] NASA-STD-8719.14, Process for Limiting Orbital Debris, with Change 1, dated 2012-05-25.

BIOGRAPHY

***Adam Wuerl** received a B.S. in Aerospace Engineering from the University of Washington in 2003 and a M.S. in Aerospace Engineering from Stanford University in 2006. He has been with Andrews Space since 2012 (and also interned as an undergraduate). In between, he spent 9 years at Lockheed Martin Space Systems Company in Sunnyvale, CA and Huntsville, AL. He is currently the Program Manager for the SCOUT imaging microsatellites and Kestrel Eye Block II, and the Andrews Space Systems Engineering, Integration, and Test lead. Previously, he has been a lead and systems engineer on various missile defense and satellite programs including THAAD, the Multiple Kill Vehicle, and various restricted remote sensing programs, where he specialized in mission analysis, modeling and simulation, system architecture, and concept of operations.*