The ISS as a Launch Platform for Phenomena of Interest

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ABSTRACT

The lower ionosphere/thermosphere (150 to 350 km) is the boundary between the sensible atmosphere of the Earth and space. This region receives energy and momentum from both the sun (ultra-violet light and electromagnetic energy coupled via the earth’s magnetosphere) and the lower atmosphere (turbulent waves). These processes act as system drivers and feedback elements, but are still poorly understood, making the weather in this region unpredictable. Few observations of these processes and their coupling are available and no validation of the predicted fundamental neutral or plasma processes has been accomplished. Although almost immeasurable themselves, these atmospheric changes have very measurable consequences to Earth’s inhabitants. The need to make long-term measurements is crucial, but is frustrated by factors such as cost, short orbit lifetimes, and infrequent launch opportunities. The International Space Station (ISS), orbiting just above this “inaccessible region”, is an ideal platform from which CubeSats can be deployed to study the region below. To prove the feasibility of such missions, students and engineers from Utah State University and the Space Dynamics Laboratory developed a conceptual design for an ISS CubeSat ejection system. Known as ICES, this responsive platform is capable of deploying multiple CubeSats to support missions studying the ionosphere/thermosphere region.

INTRODUCTION

The most significant advances in solar and space physics, or Heliophysics, over the next decade are most likely to derive from new observational techniques. The connection between advances in scientific understanding and technology has historically been demonstrated across many disciplines throughout time. There are clear ties between advances in our understanding of Heliophysics processes and the deployment of new sensing techniques that fuel new discoveries. The study of the Heliophysics system requires multipoint observations on a planetary scale to develop understanding of the coupling between disparate regions: magnetosphere, ionosphere, thermosphere, and mesosphere, as illustrated in Figure 1. Multipoint measurements are also needed to develop understanding of the various scalars or vector field signatures (i.e. gradients, divergence) that arise from coupling processes that occur across temporal and spatial scales and also within localized regions. The need for a better understanding of the ionosphere and thermosphere processes and their role in bridging the dynamics between the Earth’s atmosphere and geospace has been highlighted in the NASA 2009 Heliophysics science Roadmap1. Four inherently ionosphere and/or thermosphere -centric science question mission targets have been identified (i.e., ONEP, INCA, CISR, and DGC) as key future investigations. It is clear that the Heliophysics community now needs multi-point measurements from within the space environment to...
make progress on important scientific questions within the ionosphere-thermosphere- mesosphere region.

Figure 1: System View of Dynamics and Processes due to Coupling with Drivers from the Sun and Earth’s Atmosphere (Courtesy NASA).

Remote imaging is one source of multi-point measurements of Heliophysics systems, but not all measurement parameters of interest can be observed through remote sensing techniques. Some examples of phenomena that are not well observed by remote imaging are electric field patterns and currents flowing along magnetic field lines- both of which are important quantities for understanding the coupling of regions. The details of atmospheric composition are difficult to observe remotely, but are an important parameter for the chemical dynamics of the lower ionosphere/thermosphere. It is also clear that significant scientific advances can be made by placing remote imaging sensors at multiple points to make distributed observations of globally coherent phenomena such as atmospheric tides and auroral storms, or to improve the observations through advanced signal processing methods such as tomography or improved temporal/spatial resolutions.

The resources that will be available over the next decades for all areas of Heliophysics research have limits. It is, therefore, important that the scientific and technical community find ways to leverage the costs of developing new technologies to advance science. The high cost of access to space, at first review, is a serious impediment to making multipoint measurements within the space environment or, in other words, in deploying constellations of traditional satellites. It is therefore desirable to develop much smaller and lower-cost sensor/satellite systems such that the largest number of distributed measurements can be economically made in the space environment. The smaller the mass and volume of the sensor/satellite, the larger the number of sensors/satellites that can be deployed from a single launch vehicle will be. The prospect of creating miniaturized sensors and satellite systems is good, given the enormous investment in that field. Commercial, medical, and defense industries all have an interest in producing highly capable, portable, and low-power battery-operated consumer electronics, in-situ composition probes, and novel reconnaissance sensors. The advancement represented by these technologies has direct application in developing small sensor/satellite system for Heliophysics research.

Affordable constellations are not the only observational tool enabled by smaller and lower-cost sensor/satellite systems. It also becomes feasible to put "almost disposable" platforms into heretofore sparsely or unsampled locations or regions where it is currently not economical to place a larger, more expensive satellite. Deployed into very low Earth orbit, these small, low-cost platforms could carry instruments into the lower ionosphere/thermosphere, for example. The region between 150 km and about 350 km in the Earth’s atmosphere is not conducive to long orbit lifetimes, but could be monitored nearly continuously by periodically deploying small satellites from the International Space Station, for instance. This lower ionosphere/thermosphere is a region in which the full complexity of electro-dynamics and fluid-dynamics is exhibited, but where satellite drag ensures a quick end to satellite lifetimes. It has thus become known as the “inaccessible region”. Small, low-cost satellites can be placed into short lifetime trajectories lasting only a few weeks or months for scientific purposes which would not be feasible for larger, more-expensive satellites.

The CubeSat standard for picosatellites (see Figure 2) was developed in the late 1990’s for the use of the academic community in teaching space systems engineering to the next generation. It has since become widely accepted both internationally and locally by a broad spectrum of organizations due to the low-costs and relatively easy access to launch services which promoting the standard has engendered.
The distinguishing characteristic of CubeSats is the mechanical standard for containerized launch services and how the picosatellites are opportunistically paired with those launch vehicles that provide deployment containers. California Polytechnic State University’s (Cal Poly) “Poly Picosatellite Orbital Deployer (P-POD)”, shown in Figure 3, is an example of a widely accepted containerized launch system for secondary payloads. Most launch vehicles in the United States have designed support for multiple P-POD containers which each delivering a 3 liter (10 x 10 x 30 cm) volume weighing no more that 4 kg to orbit. The basic CubeSat (1U) form is 10 x 10 x 10 cm. Three 1U CubeSats fit within a P-POD, but other form factors have also been deployed, including 1.5U (10 x 10 x 15 cm) and 3U (10 x 10 x 30 cm). Other types of CubeSat deployers are also being developed to better suit specific missions and capabilities.

Current estimates place the number of CubeSat developers at over 100 worldwide including governments, industry and academia. The CubeSat is becoming recognized as a viable spacecraft for scientific investigations and a number of institutions are developing miniaturized space weather sensors for CubeSats.

**ICES**

In order to prove the feasibility of CubeSat constellation missions being deployed from the ISS, students at Utah State University (USU), with the support of Space Dynamics Laboratory engineers, developed a conceptual design for a system called ICES (ISS CubeSat Ejection System). ICES is a platform capable of deploying multiple CubeSats from the ISS over the duration of a two-year mission. It supports in-situ CubeSat battery charging, remotely commanded CubeSat deployment, and environmental control. The ICES concept of operations is presented in Figure 4. It illustrates the major events of a mission conducted using ICES. In parallel with the study of the ICES development system, several conceptual science missions were explored, including their required instrumentation. By approaching the study this way, the scientific payloads created realistic driving requirements for ICES, enhancing the confidence that real science missions in the “inaccessible region” could be accomplished.

**Conceptual Science Mission**

The characteristics of large portions of the Earth’s atmosphere are difficult to observe using traditional spacecraft (Figure 5).
Figure 5: Measurement Techniques and the Inaccessible Region

There are several mid- to low latitude ionospheric phenomena such as the Equatorial Ionization Anomaly, plasma bubbles, and the equatorial electro jet that are not fully understood in context of the wind and dynamics of the thermosphere. The lack of accurate composition information on the lower ionosphere and thermosphere is becoming a limitation to sophisticated computer modeling of this region\textsuperscript{10}. Empirical composition models such as MSIS (Mass Spectrometer Incoherent Scatter) have known limitations due to the scarcity of the data set from which they are derived.

In addition, many questions exist on the reaction rates of the various chemical species in the upper atmosphere and certainly on the relation of high-latitude Joule heating to the composition, density, and structure of the thermosphere. Other questions deal with how heavy metallic ions deposited in the 90 to 100 km range by meteors become transported to higher altitudes and how the sudden stratospheric warming events affect the thermosphere.

All of these scientific investigations require multi-point measurements over seasonal, daily, and positional variations.

One of the primary science missions driving the configuration of ICES was an objective to investigate the composition structure of the major ions and neutrals in the thermosphere from 100-350 km range, including seasonal and diurnal variations. A secondary objective was to investigate the global distribution of minor species, such as metallic ions, and their role as tracers of thermosphere dynamics.

The science requirements to accomplish these studies are:

- Measure E and B (Electric and Magnetic, respectively) fields and all the plasma and neutral parameters with
  - 1 km spatial resolution
  - 10 min temporal resolution
  - 200-350 km altitude
  - ±25° Magnetic Latitude coverage
  - 1700-500 hrs LT coverage
  - 2 year mission

The required science instruments for this conceptual science mission are shown in Table 1.

<table>
<thead>
<tr>
<th>Instrument</th>
<th>Measurements</th>
<th>Dimensions</th>
<th>Power</th>
<th>Telemetry</th>
</tr>
</thead>
<tbody>
<tr>
<td>TOFMS</td>
<td>(C_i(N_i, N_e), C_n(N_n))</td>
<td>Cylindrical, Length=15cm, Dia=3.5cm</td>
<td>5W</td>
<td>1-10KB/s from Science Board (includes all instrument s)</td>
</tr>
<tr>
<td>TTS</td>
<td>(N_n, V_n, T_n)</td>
<td>Parallelepiped, 7cm x 5cm x 9cm (will fly 2 TTS’s)</td>
<td>2 x 7 = 14W</td>
<td></td>
</tr>
<tr>
<td>OPAL</td>
<td>(T_n)</td>
<td>Parallelepiped, 9.8cm x 17.8cm x 7cm</td>
<td>2W avg &amp; 4W peak</td>
<td></td>
</tr>
<tr>
<td>WinDi</td>
<td>(V_n, T_n)</td>
<td>Parallelepiped, 9.8cm x 17.8cm x 7cm</td>
<td>5W avg &amp; 10W peak</td>
<td></td>
</tr>
</tbody>
</table>

The science from this conceptual mission has been a driver for all of the subsystems in the design. Requirements that have flown down from the science mission include the use of 3U CubeSats (to support large measurement instruments), the number of satellites required to perform the mission with the desired spatial and temporal resolutions (36 satellites), and the amount of power needed, among others.

**Constellation Design**

The conceptual science mission for ICES requires a constellation that can provide a minimum temporal resolution of 10 minutes and a spatial resolution of 1 km. In order to fully comply with NASA’s Jettison Policy (PDD1011), each item ejected from the ISS must be 200 m from the ISS within the 1st orbit. No ejections can occur 2 days prior to the arrival of a visiting vehicle or 2 days after the departure of a visiting vehicle. All deployed CubeSats must also conform to NASA’s de-orbit policy.
In order to conform to NASA’s ejection policy, all CubeSats will be ejected in a down and aft direction at < 0.5 m/s. The region of space into which the CubeSat will be ejected contains sufficiently dense atmosphere to produce substantial aerodynamics disturbance torques. In order to take advantage of this aerodynamic drag, an Aero-dart configuration has been selected for the CubeSats. An Aero-dart is a 3U CubeSat with solar panels deployed at an angle, as shown in Figure 6, to provide aerodynamic stability. A deployment angle of 40 degrees has been used for the conceptual mission. Using this Aero-dart configuration, orbital simulations and analyses were conducted to determine which constellations would conform to NASA’s policies and provide sufficient coverage to meet the conceptual science mission’s goals.

Figure 6: Aero-dart CubeSat Design

A study was performed to determine an appropriate ejection angle relative to the local horizontal to complete the aforementioned requirements. The ejection angle was varied between 0 degrees (aft) and 90 degrees (nadir). An angle of 70 degrees was selected due to structural constraints and was analyzed extensively. With a deployment angle of 70 degrees from the local horizontal (and aft) the minimum distance between the ISS and a deployed CubeSat is 28 times the required distance after a single orbit (as shown in Figure 7).

Figure 7: CubeSat Position Relative to ISS as a Function of Ejected Angle.

The minimum ejection delta-V required to clear the keep out sphere (200m) is 3.545 cm/s (shown in Figure 8). The maximum ejection delta-V permitted is 5 cm/s, so the ejection is achievable with a 70 degree ejection angle.

Figure 8: CubeSat Position Relative to ISS as a Function of Ejection Delta-V.

The orbital lifetime of an Aero-dart CubeSat ejected from the ISS with the appropriate angle and delta-V was examined. The lifetime analysis was simulated for deployments in 2013 and 2017, which correspond to projected maximum and intermediate altitudes for the
ISS, respectively. The 2013 time also corresponds with a projected solar maximum. Monte Carlo simulations were completed in which the mass, deployment velocity, ballistic coefficient, drag area, solar panel deployment angle, solar radiation area, and Earth albedo were varied. The Monte Carlo simulations were completed for both the 2013 and 2017 ejection dates.

For the 2013 ejections, the altitude of the ISS was varied between 410 and 420 km, according to Figure 9, and the Solar F10.7 value was varied between 130.3 and 150.3 based on predicted values reported by NOAA (National Oceanic and Atmospheric Administration).

![Figure 9: Predicted ISS Altitude in 2013](image)

The Monte-Carlo simulation results in Figure 10, showing that the lifespan of an Aero-Dart CubeSat ejected from the ISS in 2013 will be between 2.75 and 4 months.

![Figure 10: Monte Carlo Results of Aero-Dart Lifespan with 2013 Ejection](image)

CubeSats ejected in this configuration will have a minimum distance of 11.56 km between themselves and the ISS after the first orbit (as shown in Figure 11).

![Figure 11: Aero-Dart Range from ISS with 2013 Ejection](image)

The Monte Carlo simulation was repeated for an ejection date in 2017. In 2017, the ISS is predicted to be at a lower altitude- varying between 376 km and 384 km, as seen in Figure 12. The expected solar output is also expected to be lower with F10.7 values varying between 78.5 and 96.5.

![Figure 12: Predicted ISS Altitude in 2017](image)

The Monte-Carlo simulation shows that the lifespan of an Aero-Dart CubeSat ejected from the ISS in 2017 will be between 2.75 and 4.75 months (shown in Figure 13). Even though the CubeSats were ejected at a lower altitude, the decrease in solar output allowed for greater mission lifetimes than the higher altitude deployments.
CubeSats ejected in this configuration will have a minimum distance of 13.74 km between themselves and the ISS after the first orbit (shown in Figure 14).

Launch Systems

The ISS has the potential to play a major role in facilitating the studies of the lower ionosphere/thermosphere region and many of the science targets identified in the NASA 2009 Heliophysics Roadmap. The ISS is expediently located in the middle of the lower ionosphere/thermosphere region; hence, it could become a permanent launch platform for regular or responsive deployment of the small satellite fleet. For example, a group of satellites could be launched in response to a storm or stratospheric warming. It would also be the ideal location from which to routinely launch probes into the inaccessible region below or to maintain a long-term multi-point observational capability. The unique advantage of ISS is that deployments of these small satellites is not contingent on finding a suitable ground based launch opportunity, whose scheduling could never be triggered by a storm type scenario. The relatively high ISS orbit inclination also provides complete mid-latitude and equatorial coverage; during geomagnetic storms, the areas of interest are exactly these. The ability to deploy on command enables not only the scientific study of the ionosphere/thermosphere, but also benefits other areas such as astrophysics and Earth science.

The ability to regularly deploy satellites on weekly to monthly time scales or during a specific geophysical event is the distinguishing compelling capability for locating a large number of CubeSats on the ISS. This is a distinctly different approach than simply deploying CubeSats from various resupply vehicles just before their arrival at the ISS. While useful, that technique does not allow for long-term monitoring, constellation development, or launch in response to scientific phenomena of interest.

One concept for attaching ICES to the ISS is to make use of the Japanese Experiment Module (JEM), as illustrated in Figure 15. This element of the ISS, also known as KIBO, has the ability to attach Exposed Facility (EF) modules for user-defined payloads (see Figure 17). Each EF can weigh up to 500kg and is provided both low and high rate telemetry and up to 3 kW of power. KIBO has twelve attachment points for EF modules of which five lay in the wake side. These wake side locations are prime sites for ICES and current plans have them free of payloads for the 2013-2018 time frames. ICES and its load of CubeSats...
would be transported to the ISS using established resupply vehicles.

The Japanese H-II Transfer Vehicle (HTV), shown in Figure 16, would be used as the transport vehicle for ICES. All launch vehicles that are currently scheduled or in the planning stages to rendezvous with the ISS were considered as possible candidates to deliver ICES. At this time, however, the Japanese HTV is the only transfer vehicle capable of delivering EF payloads to KIBO and, as such, has been selected as the delivery method.

![Japanese H-II Transfer Vehicle](image)

**Figure 16: Japanese H-II Transfer Vehicle**

The ICES payload will conform to all requirements of a standard KIBO EF payload, as shown in Figure 17, and can be directly loaded onto an EF pallet for delivery aboard an HTV to the ISS. Once docked with the ISS, the EF pallet will be removed from the HTV by the Canadarm2 Robotic arm. The Canadarm2 will transfer the pallet to the KIBO Remote Manipulator System, which will attach the individual payloads to KIBO.

![An Exposed Facility Payload for the Japanese Experiment Module on the International Space Station](image)

**Figure 17: An Exposed Facility Payload for the Japanese Experiment Module on the International Space Station (Courtesy of JAXA).**

**Structural**

The structure of ICES is fairly simple and consists of three major parts: containment vessels to house and deploy CubeSats, the overall structure to which the containment vessels and supporting electronics are mounted, and KIBO module standard interface hardware.

ISS requirements state that fluid containers should have a safe-life against rupture or leakage when the release of the fluid would cause a catastrophic hazard. CubeSat batteries fall under the hazardous fluid-container category; therefore the containment of potential leakage and fragments is a necessity. Two options were considered: requiring individual CubeSats to provide containment for their batteries (potentially decreasing the mass available for other systems) or using the CubeSat deployers as containment vessels. The latter option was selected in order to minimize the requirements levied against potential ICES CubeSats. The CubeSat deployer under development by Planetary Systems Corporation, with minor modifications (such as adding an O-ring and a pressure release valve), fulfills this requirement. The Planetary Systems deployer shown in Figure 18 also offers the ability to pass power and data connections through the deployer to the contained CubeSats. By requiring CubeSats to have a mating connector, charging and state of health monitoring is enabled, increasing system safety. Charging and state of health monitoring is discussed in more detail in subsequent sections.

![A CubeSat Deployer (Courtesy of Planetary Systems Corporation)](image)

**Figure 18: A CubeSat Deployer (Courtesy of Planetary Systems Corporation).**

The ICES structure is envisioned as a simple, lightweight framework able to support containment vessels and their supporting electronics. Placement of
containment vessels is driven by orbital injection requirements, CubeSat quantity maximization, and the necessary size and shape of KIBO payloads. CubeSat containment vessels are pointed downward and aft at 70 degrees (with relation to the ISS). The deployer frame is made of lightweight extruded aluminum.

Figure 19: ICES deployer capable of carrying 38 3U (114 U’s volume) CubeSats.

The number of CubeSats able to be deployed from ICES is a function of both the mass and the volume of the contained CubeSats and their containment vessels. As a starting point, a configuration was developed to support the previously discussed conceptual science mission. It was determined that in order to complete the mission, at least 9 satellites must be in orbit at any given time. That would require a total 36 satellites per year, or 72 for the entire mission (plus a few extras for margin). The developed ICES configuration, as pictured in Figure 19, is able to deploy 38 3U CubeSats (114 U’s of CubeSat volume). After the first year of the mission, the depleted ICES module would be replaced with a full module in order to maintain the constellation.

In addition, an “Extended Capacity” configuration employing extra-long containment vessels was designed to be able to deploy 240 U’s of CubeSat volume. A mass analysis determined, however, that due to the 500 kg limit for KIBO payloads, the maximum number of CubeSats able to be deployed from the ISS using ICES would be on the order of 150-200 U’s. The design was revised in order to support the maximum reasonable capacity. The result, pictured in Figure 20, holds 198 U’s (66 3U CubeSats).

Figure 20: ICES deployer containing the maximum reasonable capacity (198 U’s of CubeSats)

Standard fixtures (illustrated in Figure 17) are used for KIBO interfaces. They include the Payload Interface Unit (PIU) used to attach to the External Facility (EF), the Grapple Fixture used by the KIBO robotic arm for module installation, and HTV Cargo Attachment Mechanisms (HCAM-P) to attach to palates during delivery and disposal to and from the ISS. In addition to an attachment point, the PIU also provides power, communications, and thermal connections to the EF.

**Thermal**

Thermal requirements (listed in Table 2) were generated to support ISS safety requirements and to provide a controlled, safe environment in which to store CubeSats. A number of options were considered for thermal management of the ICES payload. The first option considered was hot bias with iridite or black anodize. This would utilize the onboard cooling system, but is potentially a more complicated design.

The second option is cold bias utilizing white paint or a similar coating. This would require heaters throughout the ICES container.

The third and best option is to utilize thermal blankets, Multi-Layer Insulation (MLI), to control the temperature of ICES.

The ISS can accommodate heat rejection up to 3kW. There is cooling service at all KIBO attachment points except point 7. The EF cooling system rejects the heat from payloads by circulating the Fluorinert refrigerant. There are 2 cooling channels at attachment points 1 and 2 with a maximum 6 kW heat rejection capability. The temperature range of the Fluorinert is from 16 to 24 degrees Celsius and the flow rate is adjusted against the amount of the heat rejection.
Payloads can release their heat passively into deep space.

### Table 2: Thermal Requirements

<table>
<thead>
<tr>
<th>Short ID</th>
<th>Requirement Definition</th>
<th>Verification</th>
<th>Flow down</th>
</tr>
</thead>
<tbody>
<tr>
<td>Touch Temperature</td>
<td>ISS crew shall not be exposed to excessive high or low surface touch temperatures. 45°C max. surface temperature. -4°C min. surface temperature.</td>
<td>Analysis /Testing</td>
<td>ISS Safety</td>
</tr>
<tr>
<td>Component operational temperature</td>
<td>The ICES thermal subsystem shall maintain all ICES electrical and mechanical components within their operational temperature range</td>
<td>Analysis /Testing</td>
<td>ICES Functionality</td>
</tr>
<tr>
<td>Storage temperature</td>
<td>ICES shall provide a thermal environment such that the stored CubeSats and their components remain within a temperature range of -5° to 40°C</td>
<td>Analysis /Testing</td>
<td>CubeSat Storage and Functionality</td>
</tr>
<tr>
<td>Spacecraft operational temperature</td>
<td>The ICES thermal subsystem shall maintain all spacecraft components and structures within their operational temperature range</td>
<td>Analysis /Testing</td>
<td>CubeSat Storage and Functionality</td>
</tr>
</tbody>
</table>

### Power

There are several major requirements to which the ICES payload power subsystem must conform: ICES power consumption shall not exceed 3kW (120VDC); ground station operators shall have the ability to monitor the health and status of CubeSat batteries on ICES; ICES shall provide a 20W (10V) supply for charging each CubeSat Battery; and ICES shall comply with all safety requirements according to JSC-20793 (Crewed Space Vehicle Battery Safety Requirements). Each CubeSat in ICES shall have an internal battery charging circuit designed specifically for the type of battery chemistry used. The concept of operations for the power subsystem is shown in Figure 21.

![Figure 21: Power/Ejection ConOps](image)

The ICES payload converts 120VDC down to usable voltages for battery charging and system power. It will provide 10VDC to the CubeSat battery charging circuits for battery maintenance. Before a CubeSat is ejected from ICES, the power system will monitor a test discharge of its batteries and downlink this information to the ground. The health of the battery will be determined and it will be decided if a CubeSat is fit for ejection. If an anomaly is detected (e.g. a ruptured battery), CubeSat deployment will be halted and appropriate measures (such as disconnecting power to the charging circuit and preventing future ejection) will be taken. Otherwise, the battery will be recharged and the CubeSat ejected.

![Figure 22: Power Function Block Diagram](image)

An EMI filter will be directly connected to the 120V provided by KIBO to protect from current inrush, as shown in Figure 22. A DC-DC converter is used to provide 10VDC for charging and 5VDC for the system electronics. 10VDC is applied to each containment...
vessel. The containment vessel power controller will be capable of charging or discharging each CubeSat battery it contains. The battery status data and discharge curve for each CubeSat will be stored and passed to the communications system for ground evaluation.

**Figure 23: Connection Diagram**

The ICES power interface connects to KIBO through the equipment exchange unit, as shown in Figure 23. The ICES power interface connects to multiple containment vessel controllers through a serial port interface. The containment vessel controller connects to up to three CubeSats.

In charging mode, a charge signal will be received from the ICES power interface. Power will be applied to the CubeSat charging circuit and will be monitored. In test mode the discharge signal will be received from ICES power interface. The power supply will be disconnected from the charging circuit and a load will be applied directly to the CubeSat battery terminals. The power will be monitored as the battery drains to ensure the battery hasn’t failed. Power will be passed through all of the CubeSats to charge their batteries.

A battery waiver will need to be applied for in order for ICES to successfully operate. The current NASA limit for battery capacity is $2 \times 10^4$ Joules (SSP 52005). A standard Clyde Space CubeSat battery is $8.2 \text{ V} \times 10 \text{ Amp hours}$ or $2.95 \times 10^5$ Joules. This is 15 times more than the ISS requirement allows!

The CubeSats specified for this conceptual science mission would need more battery power (possibly 20 or 40 Amp-hours). A waiver would be required.

**Communications**

The communications requirements are shown in the following table (Table 3).

<table>
<thead>
<tr>
<th>Short ID</th>
<th>Requirement</th>
<th>Verification Method</th>
<th>Design Compliance</th>
</tr>
</thead>
<tbody>
<tr>
<td>Health and Status</td>
<td>Ground station operators shall have the ability to monitor the health and status of CubeSat batteries on ICES</td>
<td>Test Operational Power Consumption</td>
<td>Yes, by design</td>
</tr>
<tr>
<td>Remote Deployment Commanding</td>
<td>ICES shall receive commands from a ground station, and deploy a CubeSat when given the command</td>
<td>Ground Functional Test</td>
<td>Yes, by design</td>
</tr>
</tbody>
</table>

Raw data will be downlinked via ICS through DRTS (Data Relay Test Satellite), or NASA’s TDRSS. All data is sent to Tsukuba Space Center (TKSC). Users can get the engineering data at their own site. Commands for EF experiments are sent from Tsukuba Space Center.

No additional information is added as the data is passed from the containment vessels to the ICS DRTS, but the format of the data may change for compatibility. The containment vessels send and outgoing battery status and discharge curve for individual CubeSats and receive ejections commands as incoming messages.

**Risk and Safety**

ICES has the advantage of using several systems with high Technical Readiness Levels. This allows engineers the opportunity to focus their efforts on developing the newer technologies needed to perform the mission. Many of the heritage subsystems need little to no changes to be adapted for use on this project. The H-II launch vehicle, for example, has performed similar missions several times in the past. It has delivered KIBO payloads to orbit and aided in their disposal at end of life, as well. Those payloads all use standard, tested parts and systems to interface with the ISS. CubeSats (even those with deployable solar arrays and antennas) have also been flown many times with great success. In addition, ICES will also use heritage technologies for thermal management.

Several technologies must be significantly modified in pursuing missions with ICES. The containment vessel used to house the CubeSats during launch and while docked to the ISS is being developed by Planetary
Systems and will be flight tested by them. Slight modifications will be needed to make this a viable solution for this system, however. In order to classify the Planetary Systems CubeSat deployer as a containment vessel, an O-ring and a pressure relief valve must be added. In addition, it must be modified to allow for a single 3U CubeSat to be ejected at a time (it currently supports only the simultaneous launch of two 3U satellites). Also, although ground-based communications systems for satellite data transfer are well developed, communicating with constellations of satellites poses its own challenges. More effort will need to be invested in this capability.

A few new systems will need to be developed from the ground up for use with ICES. The power system, discussed previously, with its ability to charge CubeSat batteries with a variety of compositions and monitor satellite health through charging curves is already in the early stages of development at USU. Also, although KIBO payloads have been flown with standard interfaces in the past, the exact configuration of the ICES structure needs further investigation.

The TRL’s associated with subsystems in the ICES mission are shown in Table 4.

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Overall TRL</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Launch Services</td>
<td>9</td>
<td>Heritage LV</td>
</tr>
<tr>
<td>Structure</td>
<td>3</td>
<td>Parts are 9, configuration unknown</td>
</tr>
<tr>
<td>Containment Vessel</td>
<td>6</td>
<td>Planetary Systems with slight modifications</td>
</tr>
<tr>
<td>Thermal</td>
<td>9</td>
<td></td>
</tr>
<tr>
<td>Comm.</td>
<td>3</td>
<td>Multiple satellite communications</td>
</tr>
<tr>
<td>CubeSat</td>
<td>8</td>
<td>Deployable solar arrays</td>
</tr>
<tr>
<td>Power</td>
<td>4</td>
<td>Battery monitoring</td>
</tr>
<tr>
<td>Total System</td>
<td>3</td>
<td></td>
</tr>
</tbody>
</table>

In addition to the assignment of TRL’s, a hazard analysis was performed to identify potential points of failure for the designed system. Hazards were categorized according to the scope of the failure and then rated by the severity according to the consequence of the failure and its probability of occurrence (using the chart shown in Figure 24). Practices and procedures required to mitigate the risks posed by these occurrences were developed. These hazards, ratings, and required actions, shown in Table 5 and Table 6, are discussed below.

Figure 24: Hazard analysis chart (Courtesy of NASA)

The hazards that need the most attention are those that pose a risk to the ISS, visiting vehicles, and/or astronauts. Three main potential hazards were identified: a CubeSat battery rupture, a CubeSat contact with the ISS or other vehicle, and an uncontrolled ejection or ejection failure.

It was the possibility of a battery rupture that prompted the research to be done into the unique charging and state of health monitoring system at USU. A ruptured battery has the potential to contaminate or damage nearby systems, including the ISS. By determining if the battery, and thus the associated CubeSat, is fit to be ejected, this risk is mitigated. Unsafe CubeSats are stored safely in their containment vessels, never to be ejected.

Many regulations are already in place to prevent materials ejected from the ISS from recontacting the ISS or contacting vehicles approaching or leaving the ISS. Some of these regulations involve restricting the times at which objects may be ejected. Others measures include specifying ejection velocities and directions. ICES is compliant with all of these regulations. In addition, extensive modeling of CubeSat orbits has been performed. The minimum size of CubeSats (10 cm x 10 cm x 10 cm) also ensures that they may be tracked from the ground.

If a containment vessel door was to open, but the contained CubeSats were not deployed, an uncontrolled ejection situation may result. Redundant and proven ejection systems should be employed in addition to deployment sensors.
Hazards that pose only a risk to the mission include losing communications with ICES, losing contact with a CubeSat on orbit, or the malfunction of a containment vessel resulting in it not being able to open. Losing communications with ICES could cause a complete mission failure as additional CubeSats would not be able to be deployed. Redundant and well-tested systems should be employed. Due to the nature of CubeSats, however, the success of a mission does not ride on any one satellite. The loss of several CubeSats would have little to no effect on mission success.

### Table 5: Hazards to Others

<table>
<thead>
<tr>
<th>Potential Failure</th>
<th>Consequence</th>
<th>Likelihood</th>
<th>Action Required</th>
</tr>
</thead>
<tbody>
<tr>
<td>Battery rupture</td>
<td>3</td>
<td>2</td>
<td>Battery monitoring, containment vessel, don’t eject unsafe CubeSats</td>
</tr>
<tr>
<td>CubeSat contacts ISS, other vehicle</td>
<td>5</td>
<td>2</td>
<td>Simulation, orbit monitoring, deployment restrictions</td>
</tr>
<tr>
<td>CV door opens, CubeSat ejection fail</td>
<td>5</td>
<td>2</td>
<td>Redundant ejection systems, deployment sensor</td>
</tr>
</tbody>
</table>

The purpose of the effort is the verification that it is feasible to launch large numbers of small satellites from the ISS. In so doing, the stage will be set for a unique method to address outstanding space-weather issues as well as advance our knowledge of the ionosphere’s interactions with the mesosphere, thermosphere, and magnetosphere. Establishing the feasibility involves verifying: 1) that indeed the technology to deliver a canister to the International Space Station that contains many dozens of small satellites and which can be attached via an external attachment point is possible; 2) that the subsequent remote deployment operations is consistent with ISS safety protocol; and 3) that most significantly, the overall pro-rated cost per launch is defensively affordable. To address these questions, the team is meeting with the ISS stake holders as well as the broader technical and scientific community. The strategic location of the ISS makes it an ideal platform from which to address science questions that need in situ measurements to be made in both a difficult altitude range, 150 to 350 km, and in rapid response to space weather phenomena such as a Superstorm or a stratospheric warming.

The broader impacts of this effort lie in several directions: 1) new access to space capability; 2) an international openness due to the nature of ISS; and 3) a dramatically improved capability for a large number of student teams engaged in small satellite development to have their payloads launched. It is important to verify that a cost effective and technically feasible method to use the ISS external payload capability is to launch satellite constellations on demand. This will provide national agencies, as well as commercial and educational entities, with a new launch capability for LEO. This concept significantly enhances the recently announced NASA plans to establish a Flight Opportunities Program, which will reside within the NASA Office of the Chief Technologist and focus on the use of commercial reusable suborbital research (CRuSR) vehicles and the ISS to conduct future research and education. Although the proposed study focuses on the lower ionosphere and thermosphere and ultimately magnetosphere science, the new access capability is not so restricted and science and technology missions covering a broader range will be enabled. The ISS platform itself is already an

### Table 6: Hazards to Self

<table>
<thead>
<tr>
<th>Potential Failure</th>
<th>Consequence</th>
<th>Likelihood</th>
<th>Action Required</th>
</tr>
</thead>
<tbody>
<tr>
<td>Lose connection to ICES</td>
<td>3</td>
<td>2</td>
<td>Redundancy, testing</td>
</tr>
<tr>
<td>Lose contact with CubeSat on orbit</td>
<td>5</td>
<td>2</td>
<td>Redundancy due to nature of CubeSats</td>
</tr>
<tr>
<td>Containment vessel doesn’t open</td>
<td>5</td>
<td>2</td>
<td>Redundancy due to nature of CubeSats</td>
</tr>
</tbody>
</table>

SUMMARY

Utah State University – Space Dynamics Lab (USU-SDL), in close collaboration with multiple NASA institutions and the geospace community at large, is developing a detailed study targeting the viability and use of the ISS as a launch platform for phenomena of interest in the lower ionosphere/thermosphere region. The science and technology demonstration viability study is in collaboration from the Goddard Space Flight Center Helio-physics Division Space Weather laboratory. The study includes a launch vehicle and International Space Station integration and deployment process systems engineering design and trade study with collaboration from the Johnson Space Center Space Environments group and Payload Office. The ICES (e.g., CubeSat) storage container interface and on-orbit operations engineering design and trade study is in collaboration with the Ames Research Center Small Satellite Technology group. The study was also used as the design project for the Spring 2011 Space Systems Design course at Utah State University. Fourteen students enrolled in the course and participated in the study.
internationally accepted resource. Hence, use of this potential resource has few international limitations. This is most certainly true for the international educational pursuits of either core science or small satellite technology demonstrations. The successful outcome of the proposed work would result in a launch platform from which numerous satellites would be deployed with no recognition of payload or national priorities.

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REFERENCES


2. Puig-Suari, J., CubeSat Design Specification Rev. 12., The CubeSat Program, Cal Poly SLO. jpuigsua@calpoly.edu


