RECOVERY OF IN-SPACE CUBESAT EXPERIMENTS (RICE) PROJECT

Bryan Chan[(1)], Nicole Bauer[(1)], Jessica R. Juneau[(1)], Stephanie Stout[(1)], Kento Masuyama[(1)], Dave Spencer[(1)]

[(1)]Georgia Tech SSDL, 270 Ferst Dr., Atlanta, GA 30332, USA, bchan@gatech.edu, nbauer3@gatech.edu, jjuneau3@gatech.edu, sstout3@gatech.edu, kmasuyama@gatech.edu, david.spencer@aerospace.gatech.edu:

ABSTRACT

The RICE project is a university project in conjunction with Eloret Corporation and NASA Ames Research Center that seeks to develop a low-cost, low-mass flight system that is capable of exposing scientific CubeSat experiments to the Low-Earth Orbit (LEO) environment and safely recovering them for laboratory analysis on the ground. A CubeSat sized payload was selected because it is a standard interface system that has become widely popular with university programs and the aerospace industry. The proceeding paper will describe the research and trade studies that have been completed in order to define mission architecture possibilities and science payload requirements for the RICE mission.

1. MISSION ARCHITECTURE DESIGN

The intent of this section is to describe the current mission architecture for the Recovery of In-Space CubeSat Experiments (RICE) mission. The mission objectives and overview are first described, with a basic explanation of the mission concept and flight system configuration. Next, the baseline architecture is outlined within the Grand Menu along with all architecture possibilities. The overarching baseline architecture selections are then explained, incorporating the trade studies and discussions that went into those decisions. The goal is to present an up-to-date description of the mission architecture for the RICE mission.

1.1 Mission Concept Overview

The RICE project seeks to develop a low-cost, low-mass spacecraft that is capable of exposing scientific CubeSat experiments to the LEO space environment and safely recovering them for laboratory analysis on the ground. A CubeSat sized payload was selected because it is a standard interface system that has become widely popular with university programs and the aerospace industry. The overall mission objectives of RICE are as follows:

1) The RICE mission and flight system will be designed in order to guarantee payload recovery and survivability
2) The RICE system shall expose the science payload to the microgravity environment of LEO
3) The RICE system shall re-enter the Earth's atmosphere and return the science payload to Earth
4) The RICE mission design will enable rapid payload recovery after re-entry and landing
5) The RICE system design shall favor simplicity.

In order to meet the detailed mission objectives, the RICE flight system will consist of two main systems. The first is the entry vehicle, which will house the payload and will be capable of fully surviving atmospheric entry, descent, and landing. The second system will be a service module that will support the power, propulsion, communications, and command and data handling needs of the entry vehicle and payload. A model of the RICE flight system can be seen in Figure 1 below.

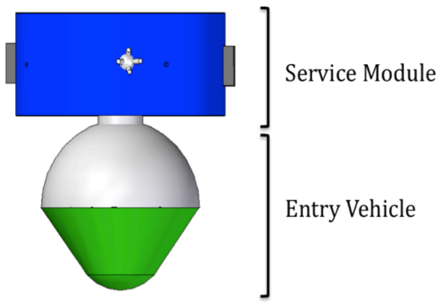
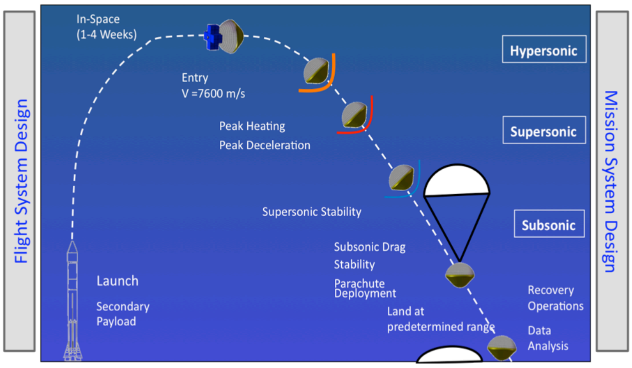


Figure 1. Visual depiction of RICE flight system.

The RICE mission concept will consist of four stages: launch; on-orbit operations; entry, descent, and landing; and recovery. As can be seen in Figure 2 below, the RICE flight system will be launched as a secondary payload, from which it will be injected into a low-earth orbit. After approximately 1 to 4 weeks on-orbit, the vehicle will then perform a de-orbit maneuver, where the service module will separate from the entry vehicle. The entry vehicle will follow a ballistic trajectory until a subsonic parachute is deployed. The capsule will then be quickly recovered upon ground impact, where it can be transported to the lab for sample analysis.



1.2 Grand Menu

In order to examine all possible design paths for the RICE mission, a Grad Menu was developed and is shown in Table 1 in the Appendix. The mission was divided into four sections: Launch, On-Orbit, Re-Entry, and Recovery.

The launch section covers all driving design considerations related to the launch, including the launch vehicle selection, payload class, launch adapter, and the launch priority. The on-orbit section largely covers the service module and payload design. Within the service module design, considerations exist for each subsystem, while the payload design focuses on payload requirements that affect the RICE interface with the CubeSat. The on-orbit section also includes the orbit for the RICE mission, including the range of altitudes and eccentricities considered in the mission design. The re-entry section focuses on all design considerations associated with the entry, descent, and landing portion of the mission. This encompasses the deceleration method (parachute vs. impact sphere), the aeroshell geometry, landing footprint, TPS, and stabilization method for the hypersonic portion of re-entry. Finally, the recovery section covers the entry vehicle landing and recovery portion of the mission and all design considerations that affect it. This includes the baselined landing site, the recovery time set by the payload requirements, the recovery method, and the tracking method.

Within the Grand Menu, certain design considerations have already been eliminated (shown in red) or baselined (shown in green). Design option elimination and baseline selections will be explained in the following section for the design decisions that have the largest effect on the overall mission and flight system architecture.

1.3 Description of Baseline Launch Decisions

The primary driver of the launch vehicle decisions is the spacecraft mass and the total mission cost, because one of the mission objectives of RICE is to keep the mission as simple as possible. Therefore, the RICE launch priority was selected to be secondary, because of the large reduction in mission cost. As a baseline, the payload class was selected to be the NanoSat, 30 kg class, because the estimated flight system mass was within this range. The NanoSat class requires the RSA launch interface, so it too was chosen as part of the baseline design. The FalconSat class was eliminated because the maximum launch payload mass was lower than the RICE estimated mass.

This largely affects the mass and volume constraints placed on the flight system. As a NanoSat class payload, the maximum mass must not exceed 30kg and the volume must not exceed a cylinder with a diameter and height of 18.7 inches. The Ride-Share Adapter (RSA) also requires that the spacecraft be interfaced using a Lightband attachment and that the payload not interfere with the survival of the primary payload. Figure 3 shows the RSA adapter in which the RICE flight system would occupy the central, octagonal volume.

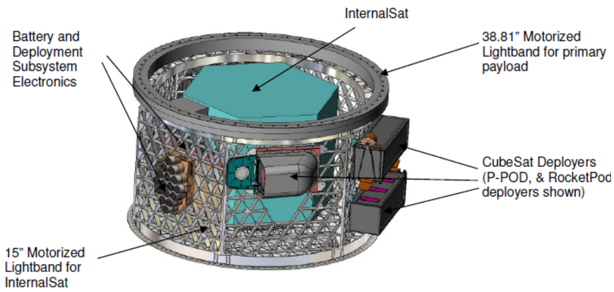


Figure 3. RSA with payload envelope [1]

1.4 Description of Baseline Orbit Decisions

For the baseline mission orbit, the simplest case was selected: low-earth, circular orbit. The range of altitudes examined was determined by a study of the de-orbit burn and its effect on the amount of propellant required, the landing footprint, and the re-entry heat load. From a study of the de-orbit delta V and the required propellant (shown in Figure 4), the propellant mass for the range of -3 to -4 degrees was found to be relatively constant for orbits between 300km and 1000km. Therefore, that range of altitudes was chosen as the baseline to be examined in all future analyses.

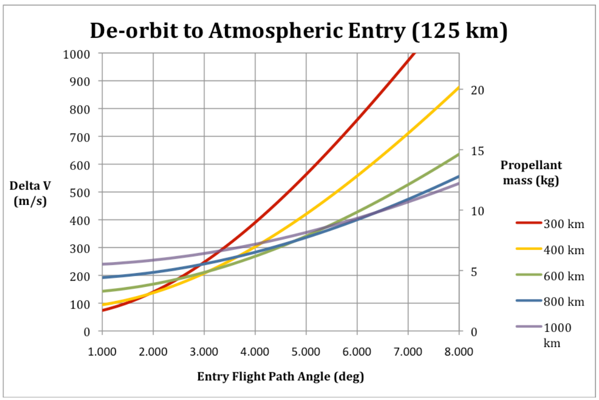


Figure 4. Deorbit characteristics for varying altitudes

1.5 Service Module Architecture

For the spacecraft attitude, spin-stabilization was selected as a baseline. By spin stabilizing about the minimum moment of inertia axis, the spacecraft would be more resilient to disturbance torques. The satellite could also be continually pointed along the solar vector, allowing for body-mounted solar panels along the top surface of the service module and maintaining the entry vehicle within the shadow of the service module at all times, thus minimizing excessive heating from the sun. Finally, spin stabilization minimizes the number of attitude control components required, thus greatly lowering the overall mass and cost.

For the spacecraft propulsion system, a series of six micro-thrusters, two per axis, were selected as a baseline. Two clusters of the thrusters will be placed on opposite sides of the service module curved walls and aligned as closely as possible with the vehicle's center of mass. This configuration was selected in order to minimize the amount of internal tubing and to simplify the attitude control and de-orbit propulsion system. However, losses exist with the system because of the offset from the spacecraft's vertical axis during the re-entry burn. Further analysis will be done in order to justify this design choice.

Body-mounted solar panels were baselined for RICE's electrical power system for various reasons. First, using body-mounted rather than deployable solar cells reduces the risk, complexity, and overall cost of the system. In addition, because the spacecraft is spin-stabilized, the panels can be placed largely on the top of the service module and oriented to along the solar vector at all times.

Finally, for the communications subsystem, omni-directional capabilities were selected for the baseline architecture. Because the spacecraft will be spin-stabilized, it will maintain a fixed position in the inertial frame, and therefore no strict pointing requirements can be maintained with respect to the Earth. For now, three omni-directional antennas were selected, with one placed on the lower surface of the service module and two placed 180 degrees apart on the curved service module walls.
Figure 5 below captures the main architecture decisions that drive the baseline service module design.

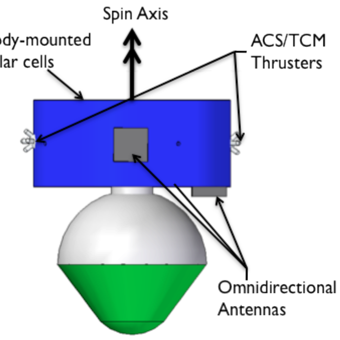


Figure 5. Service module architecture for RICE.

1.6 Re-entry Deceleration Method

In order to safely recover the science payload, two methods of re-entry deceleration were examined. The first involved an impact sphere, made of carbon foam that would aid in absorbing the accelerations experienced during ground impact. The second option involved using a subsonic parachute. The chute was selected to be subsonic, because previous POST analysis showed that the entry vehicle would reach subsonic speeds during the entry trajectory with reasonable time to deploy a chute. The driving factor in the elimination of the impact sphere option was due to the high accelerations experienced in landing, which would not be acceptable for most science payloads. The impact loadings were approximated using Meyer's theory, from which maximum accelerations can be calculated as a function of impact velocity, maximum capsule diameter, and mass. The expected acceleration experienced by the science payload upon landing was estimated to be between 161 and 281 G's.

In contrast, using a 1kg flare parachute with a drag area of 7.5 m^2, the maximum acceleration expected upon ground impact was calculated to be around 8.43 G's. Further analysis will be done in order to predict the deployment shock, which will depend upon the chute reefing, packaging, and deployment dynamic pressure. Certain problems exist with the selection of a parachute as the deceleration mechanism, including the question of how the chute will be deployed without overcomplicating the entry vehicle design. Also, when a flare parachute was investigated, the oblong parachute canister forced the entry capsule to be oversized, which added unnecessary mass and volume.

As a result, alternative parachute housings are being investigated.

1.7 Aeroshell Geometry

In order to select the proper geometry for the entry vehicle, or aeroshell, a quantitative survey of four possible geometries was completed. Several factors were taken into account, including the vehicle drag coefficient, the expected heating range, the initial orientation requirements, the stability in all flight Mach regimes, the terminal descent architecture, overall complexity, and finally the flight heritage. After all considerations, the Mars Microprobe 45 degree spherecone geometry was selected because of its excellent stability, its flight heritage, and its overall complexity. However, because the Microprobe geometry has a larger volume distribution in its spherical portion, difficulties will arise when trying to move the center of gravity forward. This will be accounted for in the entry vehicle packaging. Table 2 in the appendix shows the factors considered for the four geometries considered (Mars Microprobe, Sphere, CEV, and Stardust). Figure 6 below shows a packaging model of the current entry vehicle design. The parachute is modeled within a canister, which will change when a suitable alternative is found. The insulating shell exists in order to insulate the science payload from the heat dissipated from the Thermal Protection System (TPS).

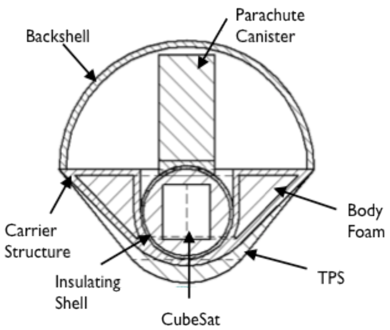


Figure 6. Current packaging model of entry vehicle

1.8 TPS Selection

A preliminary trade study of TPS material selection has been performed using the initial mass and geometry estimates for the parachute reference configuration and the -3 to -4 flight path angle. The TRAJ trajectory tool was used to define the ballistic trajectories, which were then fed into the FIAT thermal response model. Two TPS materials were considered for the heat shield including PICA and SIRCA. Three materials were considered for the backshell including PICA, SIRCA and LI-2200 (shuttle tile). Table 3 summarizes the results and assumptions of the case using a -3 deg flight path angle. Based on these results SIRCA is the best candidate material for the heatshield and LI-2200 is the best material for the backshell based on both mass and volume.

Table 3. Initial TPS sizing results.

Un-margined Thickness (in)			
	PICA	SIRCA	LI2200
Stagnation	0.664	0.315	x
Frustum	0.548	0.254	x
Back shell	0.310	0.134	0.114
Assumptions			
Nose Radius	0.1	m	
Velocity	7.6	km/s	
FPA	-3	deg	
Probe Mass	6.88	kg	
Initial Temp	70	F	
Base Radius	0.267	m	
Cone Angle	45	deg	
Reference Info			
Peak Heating (stag)	191	W/cm^2	
Peak Pressure (stag)	6.26	kPa	
Heat Load (stag)	9650	J/cm^2	

2. SCIENCE REQUIREMENTS

The motivation behind the RICE mission is to build a framework for cost-effective, recoverable space missions. While many fields, including materials science, stand to benefit from the availability of such a system, the focus is currently on space biology missions. Results of such experiments will lead to enhanced understanding of the effects of microgravity or radiation on biological systems. RICE is not designed to a single specific mission, but rather is meant to be compatible with many missions within an acceptable range of complexity and requirements. The goal of this section is to define a set of requirements that the payload will constrain the spacecraft system to in order to support some envelope of possible missions.

2.1 Science Motivation

There are several fields of science that could benefit from an inexpensive, flexible platform designed to expose experiments to aspects of the space environment and then safely return the experiment to Earth. One of the most important areas of research that could benefit from RICE is radiation and microgravity exposure for biological systems. Understanding of the effects of both long term exposure to radiation and microgravity is crucial to the further human exploration of the solar system and the RICE platform is a unique capability that would fill gaps in the current suite of space biology research laboratories.

Existing biological research platforms that have sample return capability include the Russian Foton/Bion series spacecraft and the International Space Station. Both of these laboratories come with considerable constraints including: (a) relatively high mission costs that prevent a access to space from a large community of researchers, (b) large gaps in mission opportunities and long lead times for missions typically result in outdated science objectives that may be years out of sync with current research priorities, (c) for the case of the ISS, human safety constraints limit the environmental exposure and the place a multitude of requirements on even the simplest of science experiments. The RICE platform is specifically designed to reduce the cost of performing biological research, increase the number of mission opportunities, and enable access to space environments that are otherwise unreachable to other platforms such as high altitude and high inclination orbits which have a more harsh radiation environment [2].

In 2007, the Ames Research Center hosted a workshop to develop concepts for small astrobiology science missions. The results of the workshop produced several mission concepts, two of which were small payload sample return missions [3]. This finding supports the need for the RICE platform.

2.2 Microgravity and Radiobiology

The combination of long-term exposure to microgravity and radiation environments cannot be simulated anywhere on the surface of the Earth. Space based biological experiments have indicated that the combination of radiation and microgravity create a synergy that has the potential to be more destructive and disable natural repair mechanisms in biological systems [2]. In addition the type of radiation encountered in space is unique and difficult to duplicate in the laboratory. It is well known that different types of radiation cause different types of damage to biological systems [4] which is one reason space based research laboratories are necessary for radiation research.

2.3 Science Advisory Board

Although RICE is intended to be compatible with a range of science missions, a set of maximum desirable capabilities was needed to serve as enveloping requirements. In order to set these values, a panel of scientists was put together to brainstorm reference missions specific to RICE. A charter was put together explaining the motivation and then-current design of RICE and distributed to several scientists at Ames, Georgia Tech, and other universities. Through a series of telephone conferences, meetings, and surveys, several possible payloads and associated missions were identified. These missions fell under several topics, including biology, materials science, and atmospheric sciences. However, in order to remain within the original RICE motivation, the focus has been kept on biology-related missions.

Several constraints were given to the science panel as determined by the then-current design of the RICE spacecraft. These included a 1-U volume and mass constraint for the science payload, an orbit in Low-Earth Orbit (LEO), and survivability of the biology payload during expected waits for integration with the system as well as with the launch vehicle. Several common themes arose in response to these restrictions. One was that many science teams want 2-U allocated to the payload. Another comment was that Geosynchronous Transfer Orbit (GTO) offered a more interesting radiation environment. These are interesting possibilities whose feasibility will be studied in possible future generations of RICE. However, for this first iteration, as explained in the Grand Menu section, simplicity was a major driver. Table 4 (in the appendix) shows three reference missions identified with input from the science panel, which are believed to be feasible within the current design of the RICE system and will serve as the starting point for putting together the RICE science payload requirements. Other biological payloads and science concepts discussed include: fruit flys, fish, plants, rodents, material science, astrobiology, and atmospheric science.

3. REFERENCES

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2. Nelson, G. A. (1994). *Radiation In Microgravity.* Pasadena, CA: Jet Propulsion Laboratories.

3. Yost, B., Fishman, J. L., & Fonda, M. (2007). *Astrobiology Small Payloads Workshop Report.* Moffett Field, CA: NASA Ames Research Center.

4. Nelson, G. A. (2003). Fundamental Space Radiobiology. *Gravitational and Space Biology Bulletin* .

Table 1. RICE Grand Menu

	DESIGN CONSIDERATION	OPTION 1	OPTION 2	OPTION 3	OPTION 4	OPTION 5
Launch	Adapter	RSA	ESPA	Lightband (AFUNP)		
	Launch Priority	Primary	Secondary			
	Payload Class	NanoSat: 30kg, 18.7in diam/height	FalconSat: 20kg, 14m^3	ESPA Class: 180kg, 35.5in x 28.24in	AF NanoSat Class: 50kg, 50x50x60cm	
	Launch Vehicle	Falcon 1e	Atlas V	Delta IV	Minotaur	Dnepr (Russian)
On-Orbit	Orbit					
	Altitude Range	LEO (300-1000km)	GTO			
	Orbital Shape	Circular	High Eccentricity			
	Service Module Architecture					
	ADACS	3-axis stabilized	Spin stabilized			
	Attitude Control	Thrusters only	Magnetic torquers+ thrusters	Reaction wheels+ thrusters		
	Propulsion System	Cold Gas	Monoprop hydrazine	Bi-prop hydrazine		
	Propulsion Mechanism	ACS thrusters	ACS/TCM thruster cluster	ACS thrusters+ main thruster		
	EPS					
	Solar Cell Location	Body Mounted	Deployable			
	Power Storage (Battery)	NiCd	LiON			
	Thermal	Active	Passive			
	Structures					
	Bus Material	Aluminum	Titanium	Composite		
	Communications					
	Directionality	Omnidirectional: Patch (3)	Unidirectional: Parabolic			
	Data Return	Beacon	Telemetry	Telemetry + Data		
	Data Type	Payload Monitoring	Spacecraft Health	Spacecraft Health + Payload Monitoring		
	Licensing	Amateur	FCC Experimental			
	Frequency	UHF	S-Band	C-Band	L-Band	
	Payload					
	Payload Class	1U	2U			

	Thermal Control Tolerance (Maintained by Payload)	+/- 1C	+/- 2.5C	+/- 5C		
	Thermal Control Baseline Temperature Range	4-40C				
	Power	1W	4W	5W		
Re-Entry	Deceleration	Parachute	Impact Sphere			
	Max Loading	<10G	<15G			
	Parachute					
	Mach Regimes	Subsonic	Supersonic			
	Impact Sphere					
	Material	Carbon Foam	Rohacell	Polyurethane		
	Geometry	Sphere	Hemisphere	Hybrid		
	Hypersonic Entry Stabilization	Spin Stabilization	Passively Stable			
	Entry Type	Zero-lift	Lifted entry			
	TPS					
	Forebody Material	PICA	LI900	SIRCA	LI2200	Avcoat
	Aftbody Material	PICA	LI900	LI2200	None	
	Aeroshell Geometry	Microprobe	Sphere	CEV	Stardust	
	Landing Footprint	<10km	<50km	<100km		
Recovery	Landing Sites	Land	Water			
	Recovery Method	Impacted Landing	Helicopter Snatch			
	Recovery Time (PL Reqt's)	<12 hr	<2 hr			
	Tracking Method	Optical Tracking	Beacon			

Table 2. Summary of aeroshell geometry study.

	Microprobe	Sphere	CEV	Stardust
Drag Coefficient	1.05	1	1.75	1.52
Heating (W/cm2) *	150-250	100-200	45-100	80-180
Peak Deceleration *	8-15 g	8-15 g	8-15 g	8-15 g
Initial Orientation	Mono-stable	Omnidirectional	Spin-Stabilized	Spin-Stabilized
Hypersonic Stability	Excellent	Unstable	Sufficient	Sufficient
Supersonic Stability	Stable	Unstable	Marginal Stability (C.G. Dependant)	Marginal Stability (C.G. Dependant)
Subsonic Stability	Stable	Unstable	Unstable (requires chute)	Unstable (requires chute)
Possible Terminal Descent	1, 2, 3	1, 2, 3	1, 2, 3	1, 2, 3
Preferred Terminal Descent	3	3	1, 2	1, 2
Overall Complexity (1-10)	1	1	5	5
Heritage	DS-2 / PV / Galileo	Mirka	Apollo / CEV	Stardust / Genesis

* Estimates for Vrel=7.6 km/s, FPA=1-5, Entry Mass=10-20 kg

D-Chute+M-Chute	1
D-Chute+Parafoil+MAR	2
Energy Absorbers	3

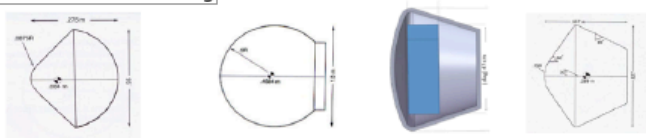

Table 4. Reference requirements for RICE biological payload.

Field of Science	***Space Biology***	***Space Biology***	***Space Biology***
Reference Mission	**Microorganisms in Microgravity/ Radiation**	**Snails in Microgravity/ Radiation**	**Human Tissue in Microgravity/ Radiation**
Science Objective			
Science Objective	***Determine the effects of microgravity on live animal development.***	***Determine the effects of microgravity on live animal development.***	***Determine the effects of microgravity on live animal development.***
Science Priority	***Informs decisions on the design of exploration mission systems.***	***Informs decisions on the design of exploration mission systems.***	***Informs decisions on the design of exploration mission systems.***
Data Sources	***See ASGSB (http://asgsb.org/index.php) for links to relevant pubs.***	***See ASGSB (http://asgsb.org/index.php) for links to relevant pubs.***	***See ASGSB (http://asgsb.org/index.php) for links to relevant pubs.***

Requirement Area			
Volume	*Minimum: 1 U CubeSat volume; 2U greatly increases capability*	*Minimum: 2U*	*Minimum: 1 U CubeSat volume; 2U greatly increases capability*
Mass	*Minimum: 1 kg, 3-4 kg greatly increases capability*	*Minimum: 3-4 kg*	*Minimum: 1 kg, 3-4 kg greatly increases capability*
Thermal Management	*20-25 C thermal control provided to CubeSat surface*	*10-40 C thermal control provided to CubeSat surface*	*20-25 C thermal control provided to CubeSat surface (must have 37 +/- 0.1 deg C at sample)*
Environmental Exposure	*Microgravity, high inclination orbits will have high radiation exposure*	*Microgravity, high inclination orbits will have high radiation exposure*	*Microgravity, high inclination orbits will have high radiation exposure*
On-Orbit Mission Life	*20-60 days or more is desirable to increase radiation exposure*	*20-60 days or more is desirable to increase radiation exposure [est.]*	*20-60 days or more is desirable to increase radiation exposure*
Max Recovery Time	*6 hours*	*6 hours*	*6 hours*
Static Inertial Loading	*At least Bion flight profile*	*At least Bion flight profile*	*At least Bion flight profile*
Dynamic Inertial Loading	*<6.0 - 7.5 g rms (below 30 Hz). Above 30Hz, there is very little coupling to biological systems*	*<6.0 - 7.5 g rms (below 30 Hz). Above 30Hz, there is very little coupling to biological systems*	*<6.0 - 7.5 g rms (below 30 Hz). Above 30Hz, there is very little coupling to biological systems*
Total Electrical Energy	*4 W of power for thermocouples and fine tuning thermal control, Up to 20 W peak during rapid heating cycle*	*4-10 W of power [est.]*	*4-10 W of power [est.]*

Data Storage	***500 MB of monitoring environmental data***	***500 MB of monitoring environmental data***	***500 MB of monitoring environmental data***
Communications	***Real time temperature monitoring (updates at least every 6 hrs)***	***Real time temperature monitoring (updates at least every 6 hrs)***	***Real time temperature monitoring (updates at least every 6 hrs)***
Launch Integration Time	***3 weeks***	***3 weeks***	***3 weeks***