TEAM ALPHA CUBESAT

GROUND TOURNAMENT 2 PRELIMINARY DESIGN REVIEW REPORT

CQDP

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TABLE OF CONTENTS

ACS PRELIMINARY CUBE QUEST DESIGN REVIEW PACKAGE

The material presented here is the preliminary design elements as they are currently defined for Alpha CubeSat. The material is organized in the following fashion:

A. OVERALL SYSTEM DESIGN

MISSION GOALS

Team ACS intends to complete in both the Cube Quest Challenge Deep Space Derby and the Lunar Derby for all prizes.

For the Deep Space Derby ACS the Judges must verify that ACS has reached the minimum required distance from Earth (4,000,000 kilometers, as defined in the Rules). While maintaining at least this distance for prize eligibility, ACS will then seek to accomplish the communications and longevity achievements.

Judges score will score the Competitor Team performances and NASA will award the following Deep Space Derby Prizes (details and constraints are given in the Rules):

1. BEST BURST DATA RATE: $225,000 will be awarded to the Competitor Team that receives the largest, and $25,000 will be awarded to the Competitor Team that receives the second largest volume of error-free data from their CubeSat over a 30-minute period.

2. LARGEST AGGREGATE DATA VOLUME SUSTAINED OVER TIME: $675,000 will be awarded to the Competitor Team that receives the largest, and $75,000 will be awarded to the Competitor Team that receives the second largest, cumulative volume of error free data from their CubeSat over a continuous 28-day (calendar days) period.

3. SPACECRAFT LONGEVITY: $225,000 will be awarded to the Competitor Team with the longest elapsed number of calendar days, and $25,000 will be awarded to the Competitor Team with the second longest elapsed number of calendar days between the first and the last confirmed reception of data from their CubeSat.

4. FARTHEST COMMUNICATION DISTANCE FROM EARTH: $225,000 will be awarded to the Competitor Team that receives at least one, error-free, CubeSat generated data block from the greatest distance and $25,000 will be awarded to the Competitor Team with the second greatest distance.

Distance must also meet minimum Challenge requirement.

For the Lunar Derby Prizes, the Judges must verify that ACS has achieved a verifiable lunar orbit (as defined in the Rules) to win an equal share of the Lunar Derby Prize. While maintaining a verifiable lunar orbit, ACS will acquire as much error-free data within single continuous 30-minute periods, and as much error-free data within any 28-day (calendar day) period.

Judges will score ACS performances according to the Rules. NASA will award the following Lunar Derby Prizes (refer to the Rules for details and constraints):

1. LUNAR PROPULSION: $1,500,000 will be divided equally between all Competitor Teams that achieve at least one verifiable lunar orbit, with a maximum of $1,000,000 to any one Competitor Team.

2. BEST BURST DATA RATE: $225,000 will be awarded to the Competitor Team that receives the largest, and $25,000 will be awarded to the Competitor Team that receives the second largest, cumulative volume of error-free data from their CubeSat over a 30-minute period.

3. LARGEST AGGREGATE DATA VOLUME SUSTAINED OVER TIME: $675,000 will be awarded to the Competitor Team that receives the largest, and $75,000 will be awarded to the Competitor Team that receives the second largest, cumulative volume of error free data from their CubeSat over a contiguous 28-day (calendar) period.

4. SPACECRAFT LONGEVITY: $450,000 will be awarded to the Competitor Team that achieves the longest elapsed number of calendar days, and $50,000 will be awarded to the Competitor Team that achieves the second longest elapsed number of calendar days, between the first and last confirmed reception of data from their CubeSat.

The ACS winning tactics/capabilities for each derby and corresponding prize challenge are as follows:

Deep Space Derby - alternate launch options, propulsion options, ballistic escape and capture minimum energy trajectories

- **Burst Rate:** Ka Band, Available Power & CPU Cycles, NASA DSN
- **Aggregate Data Volume:** Ka Band, Available Power & CPU Cycles, NASA DSN
- **Spacecraft Longevity:** Simplicity of design elements, redundancy, fault tolerance
- **Farthest Comm Distance:** Driven by return trajectory requirements therefore TBD

Lunar Derby - alternate launch options, propulsion options, ballistic escape and capture minimum energy trajectories

- **Lunar Orbit:** minimum energy resonance orbits
- **Burst Rate:** Ka Band, Available Power & CPU Cycles, NASA DSN
- **Aggregate Data Volume:** Ka Band, Available Power & CPU Cycles, NASA DSN
- **Spacecraft Longevity:** Simplicity of design elements, redundancy, fault tolerance

SYSTEM-LEVEL REQUIREMENTS

Team ACS and the ACS Spacecraft must meet the following spacecraft and/or system-level requirements:

1. Abide by the prevailing Cube Quest Challenge rules as defined in Document No.: CCP-CQ-OPSRUL-001 Cube Quest Challenge Ground Tournaments, Deep Space Derby, and Lunar Derby Operations and Rules December 4, 2014 Revision C, December 30, 2015 and subsequent revisions as made applicable.
2. ACS Spacecraft Requirements Matrix has been abstracted from Document No.: CCP-CQ-OPSRUL-001, and have been flowed into Table X.X.
3. All abstracted rules are classified as either administrative or technical requirements.
4. All technical requirements are further classified as either spacecraft and/or system level requirements applicable to one or more systems/subsystems.
5. All technical requirements have been flowed into the spacecraft system/subsystem design development and analysis process.

SYSTEM-LEVEL BLOCK DIAGRAMS/DESIGN DESCRIPTION

System-level block diagrams (e.g., CubeSat, ground systems including ground stations, mission operations center, data center, communications networks, ground operators, etc.) have been prepared for all defined ACS Systems. The diagrams provided are as follows:

- Alpha CubeSat Spacecraft
- Communications System (COMM)
- Electrical Power System (EPS)
- Data Management System (DMS)
- Guidance, Navigation & Control (GN&C)
- Structures & Mechanisms System (S&Mech)
- Propulsion System (PROP)
- Thermal Control System (TCS)
- Payload Systems (PS)
- Ground Systems
- Launch Service Provider (LSP) Systems

IDENTIFICATION OF ALL REQUIRED ENVIRONMENTS FOR ACS

The ACS spacecraft is anticipated to be transported by motor vehicle in a shock mounted case until it is delivered to the Launch Service Provider integration facility.

The ACS Launch Service Provider will be keep ACS in a thermally stable clean room/storage environment until integrated for launch.

The ACS spacecraft may be shipped to the International Space Station (ISS) as pressurized or unpressurized cargo in consultation with the Launch Service Provider based on flight space availability and NASA flight safety guidance.

The ACS spacecraft in-space operating environments are still being characterized. Initial analysis suggests that a combination of sunpointing and occasional use of the Ka transceiver should help prevent inordinately low temperatures. High periods of use of the Ka transceiver likely will require thoughtful planning to mitigate the potential for thermal throttling.

REQUIREMENTS ANALYSIS

The ACS requirements analysis to date has started with the following design considerations outlined in the introductory sections. Additional details can be found the system/subsystem write-ups which follow.:

TECHNOLOGY READINESS LEVEL

The estimated Technology Readiness Level (TRL) for each ACS System/Subsystem has been flowed into Table X.X ACS Technology Readiness Level.

TRL definitions used are as defined in NASA/SP-2007-6105 Rev 1 pg 296.

A rational for each stated TRL is provided.

The logical construct used is that Commercial Off The Shelf (COTS) services/components available from multiple vendors are by definition TRL 9. Services/components flying on the ACS spacecraft as technology development missions are by definition no higher than TRL 7.

SUMMARY OF SYSTEM LEVEL MARGINS

ACS has established the following system level margins to be tracked and refined as we proceed with mission development.

- ACS Spacecraft Volume Budget
  - The ACS spacecraft volume budget allocated to the system/subsystem level closes with a positive margin of XX%.
  - It is anticipated that further optimization of the ACS spacecraft volument budget can be accomplished by repacking COTS systems/subsystems if necessary.
  - The ACS spacecraft volume budget is presented in Appendix Table XX ACS Spacecraft Volume Budget.
- ACS Spacecraft Mass Budget
  - The ACS spacecraft mass budget allocated to the system/subsystem level closes with a positive margin of XX%.

    - It is anticipated that further optimization of the ACS spacecraft mass budget can be accomplished by repacking COTS systems/subsystems if necessary.
    - The ACS spacecraft mass budget is presented in Appendix Table XX ACS Spacecraft Mass Budget.
- ACS Spacecraft Power Budget
    - The ACS spacecraft power budget allocated to the system/subsystem level closes with a positive margin.
    - It is anticipated that further optimization of the ACS spacecraft power budget can be accomplished by a combination of load management rules if necessary.
    - The ACS spacecraft power budget is presented in Table Appendix XX ACS Spacecraft Power Budget.
- ACS Spacecraft Trajectory Delta-V budget.
    - The ACS spacecraft trajectory Delta-V budget allocated to the system/subsystem level closes with a positive margin.
    - It is anticipated that further optimization of the ACS spacecraft trajectory Delta-V budget can be accomplished by a combination of optimization of the High Thrust Short Duration (HTSD), Low Thrust Long Duration (LTLD) propulsion capabilities as well as the ballistic escape and capture trajectories to be used.
    - The ACS spacecraft trajectory Delta-V budget is presented in Appendix Table XX ACS spacecraft trajectory Delta-V budget.
- ACS Spacecraft Communications Link Budget
    - The ACS Spacecraft Communications Link budget allocated to the system/subsystem level closes with a positive margin.
    - It is anticipated that further optimization of the ACS Spacecraft Communications Link budget can be accomplished by a combination of optimization of the High Thrust Short Duration (HTSD), Low Thrust Long Duration (LTLD) propulsion capabilities as well as the ballistic escape and capture trajectories to be used.
    - The ACS Spacecraft Communications Link budget is presented in Appendix Table XX ACS Spacecraft Communications Link budget.

SUMMARY OF KEY MISSION RISKS AND MITIGATION STRATEGIES

- Be sure to include trajectories, ranges, velocities, orbital mechanics and propulsive maneuvers analysis that support communications range and directional elements (antennas, solar arrays, pointing requirements, etc).

B. Implementation Plan

C. Ground Systems and Mission Operations Designs

D. Systems/Subsystems Design

- Spacecraft Architecture
  - CAD Model
  - Systems Block Diagram
  - Interfaces
  - Schedule
- Systems Overview
- System Designs
  - Electrical Power System (EPS)
    - Power Management and Distribution
    - Solar Arrays (conformal exterior)
    - Batteries (conformal propulsion tank corners)
  - Communications System (COMM)
    - Ka Band Radio
    - Antenna (TX+RX integrated w/solar arrays)
  - Data Management System (DMS)
    - On Board Computer
  - Structures & Mechanisms
  - Attitude Determination & Control System (ADCS)
  - Guidance, Navigation & Control System (GN&C)
  - Propulsion System (PROP)
    - Hybrid Trajectory Injection Motor Core
    - Hybrid Trajectory Injection Motor Fuel Tank
    - Ion Thrusters
    - Ion Propellant Tanks
  - Thermal System
  - Primary Payload - Encoded Bit Stream
  - Scar for Secondary Payload (future)
- System Budgets
  - Volume Budget
  - Mass Budget
  - Power Budget

SPACECRAFT ARCHITECTURE

The Alpha CubeSat spacecraft design is driven by the preceding considerations and is reflected in the Computer Aided Design (CAD) models, the overall systems block diagram, individual Systems block diagrams, individual System/Discipline Consideration Models (e.g., spreadsheet calculations to the STK software suite) and the interface models to be developed based on the outlined flow taxonomy which follows. Due to file size considerations these materials have not been interleaved into this report.

Systems Integration

- **CAD Model (*.pdf)**
    - Alpha CubeSat Spacecraft Cover
    - Alpha CubeSat Exploded View
    - Alpha CubeSat Exploded View w/ Annotations
    - Alpha CubeSat Stowed View
    - Alpha CubeSat Deployed View from Aft
    - Alpha CubeSat Deployed View from Forward
- **Systems Block Diagrams**
    - Unified Systems Block Diagrams v5.pdf
- **Spacecraft Mass, Power, and Volume Budgets & Misc. Tables**
    - Baseline Budget Cross Check.xlsx
    - Conceptual Engineering Review Workbook v5.xlsx/.pdf
        - Team Alpha CubeSat Roster
        - Mode – State Transitions
        - Milestone Schedule (embedded)
        - Dimensions
        - Table of Contents (embedded)
        - Spacecraft Configuration Summary Table
        - Systems Active in Modes-States
        - Mass and Volume Budgets
        - Power Budget
        - Phase 0 Safety Review Readiness
    - Team Alpha CubeSat Roster
        - Team Alpha CubeSat Organization-V5.pdf

System/Discipline Consideration Models

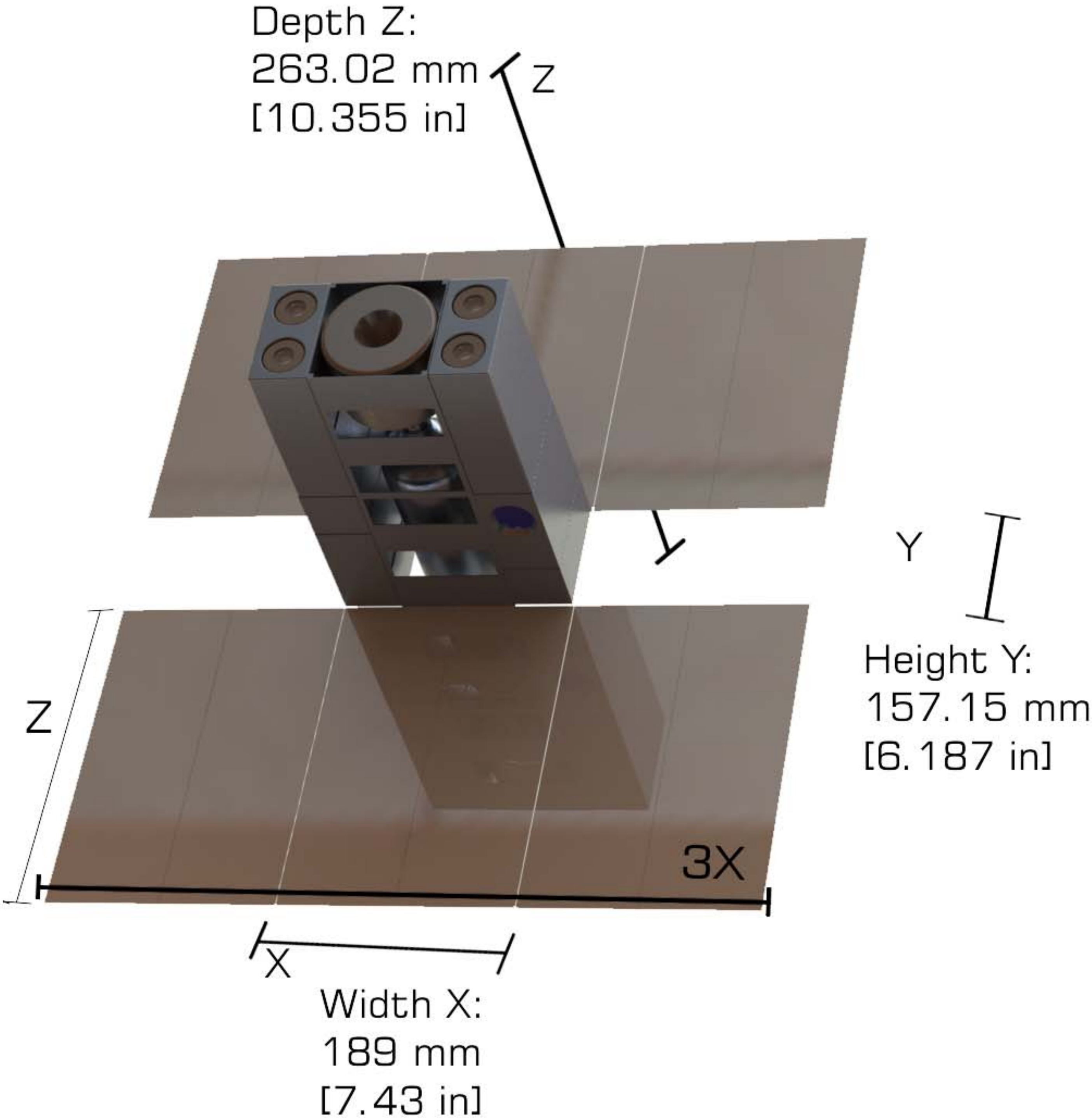
- Models unless noted are located in the Team Alpha CubeSat Conceptual Engineering Review Workbook Set which is supplied as a formatted appendix to this report.
    - Communications System (COMM)

    - Link Budget Worksheet.xlsx
- Guidance, Navigation & Control System (GN&C) / Trajectories
    - AlphaCubesat_ThrustCalc01_CRC.xlsx
    - WSB_lunar-capture.pdf
    - Dahlstrom – ISDC Halfway.pdf
    - STK Astrogater Model under development
- Propulsion System (PROP)
    - ACS Delta-V Propulsion Calculations.xlsx
- Thermal System
    - Solar Panel Heat Rejection.xlsx
    - Energy balance –CubeSat.xlsx

Interfaces

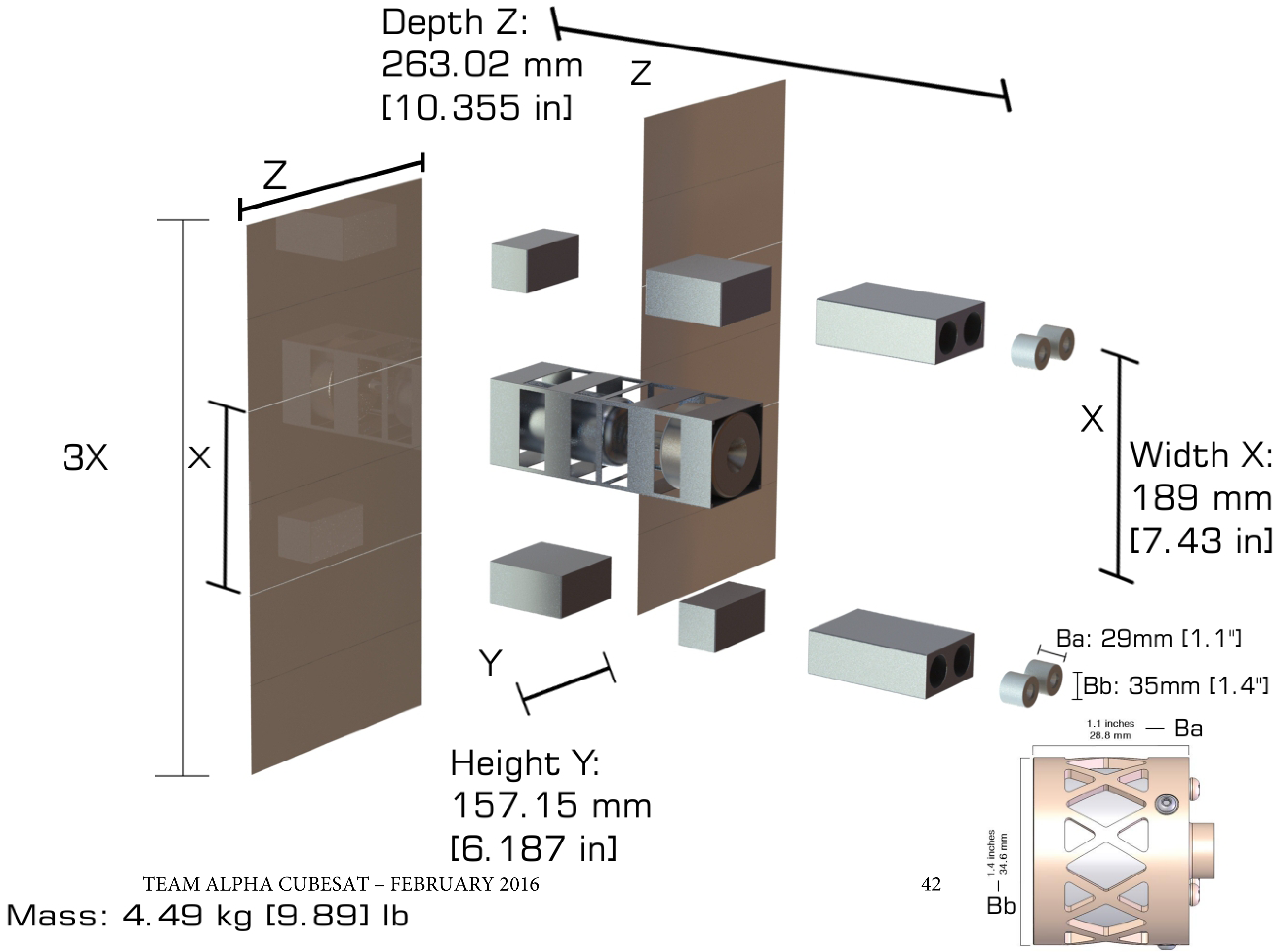
- Flow Taxonomy
  - Mass
    - Solid
    - Liquid
    - Gas
  - Information
    - Commands
    - Data
    - Telemetry
  - Energy
    - Kinetic
    - Magnetic
    - Electrical
    - Thermal
    - Light
    - Radiation

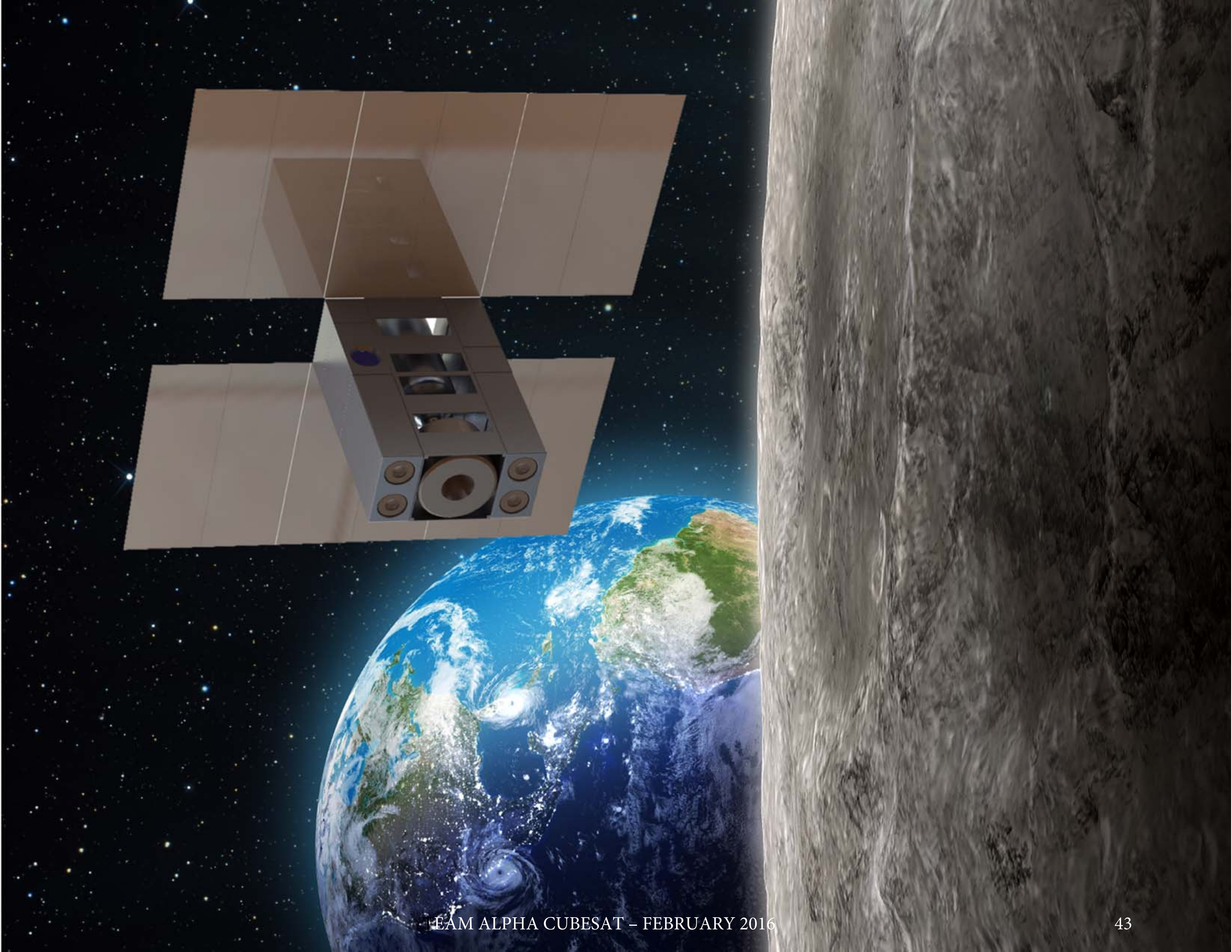
Team AlphaCubeSat
6U Dimensional Drawing

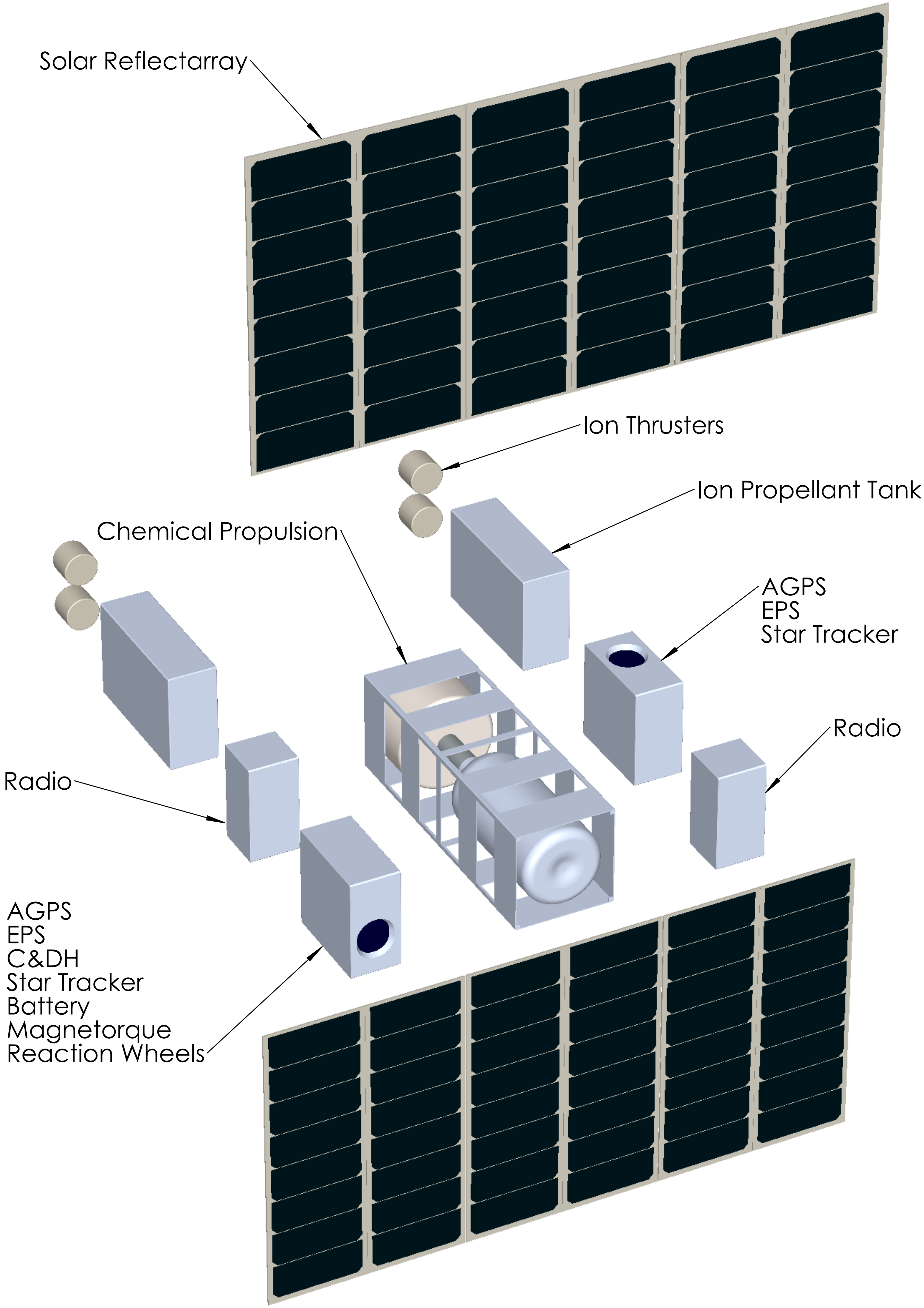


Mass: 4.49 kg [9.89] lb

Team AlphaCubeSat
6U Dimensional Drawing






Solar Reflectarray
Ion Thrusters
Ion Propellant Tank
Chemical Propulsion
AGPS
EPS
Star Tracker
Radio
Radio
AGPS
EPS
C&DH
Star Tracker
Battery
Magnetorque
Reaction Wheels

Team Alpha CubeSat Schedule

The architectural and engineering development of the Alpha CubeSat spacecraft is tracking to the following external Cube Quest Challenge schedule show below.

A detailed internal engineering development and program schedule is being assembled but has not been completed because the architecture of the system, is being driven by the COTS first strategy adopted. Accordingly, is likely to remain fluid until the make versus buy decisions are firmed up in the Preliminary Design Phase.

Team Alpha CubeSat Schedule as of February 5, 2016			
Milestone	**Date**	**Applicability**	**Status**
Cube Quest Challenge Team Registration Opens	November 24, 2014	Yes	Challenge Announced
In-Space Competition; non-EM-1 launches	November 24, 2014	Yes	Competition Begins
Cube Quest Summit	January 7, 2015	Yes	Attended
Notice of Intent to Form Team Alpha CubeSat	January 1, 2015	Yes	Submitted & Confirmed
Formal Registration Acceptence	March 2, 2015	Yes	Confirmed
Notice of Intent of Team Alpha Cubesat to Compete	March 2, 2015	Yes	Submitted & Confirmed
Mission Concept Registration Data Package	April 30, 2015	Yes	Submitted
Monthly Report Team Inception through March 2015	April 30, 2015	Yes	Submitted
Monthly Report - April 2015	May 7, 2015	Yes	Submitted
Monthly Report - May 2015	June 7, 2015	Yes	Submitted
Cube Quest Challenge Townhall	June 11, 2015	Yes	Attended
Monthly Report - June 2015	July 7, 2015	Yes	Submitted
Alpha CubeSat Conceptual Design Review Process			
GT1 Data Submission	July 3, 2015	Yes	Submitted
GT1 Tournament	August 3, 2015	Yes	Submitted
Monthly Report - July 2015	August 7, 2015	Yes	Submitted
ACS Conceptual Design Review	August - October	Yes	Team agreed press to PDR
Cube Quest Summit II	October 21, 2015	Yes	Attended
Cumulative Monthly Report - January 2015 - January 2016	February 2, 2016	Yes	Submitted
Alpha CubeSat Preliminary Design Review Process			
GT2 Data Submission	February 5, 2016	Yes	Pending
GT2 Tournament	March 1, 2016	Yes	Pending
ACS Preliminary Design Review (PDR)	March - April	Yes	Team Vote
Alpha CubeSat Critical Design Review Process			
GT3 Data Submission	August 5, 2016	Yes	Future Event
GT3 Tournament	September 7, 2016	Yes	Future Event
ACS Critical Design Review (CDR)	September - October	Yes	Team+LSP+NASA Vote
Alpha CubeSat Flight Readiness Review Process			
GT4 Data Submission	February 3, 2017	Yes	Future Event
GT4 Tournament	March 1, 2017	Yes	Future Event
ACS Flight Readiness Review (FRR)	March - December	Yes	Team+LSP+NASA Vote
ACS Delivery to Launch Service Provider (LSP)	FRR Complete + 1 month	Yes	Future Event
ACS Delivery to Deep Space Trajectory Insertion Point	FRR Complete + 3/6 months	Yes	Future Event
In-Space Competition; EM-1 scheduled launch date	EM-1 Launch (early 2018)	Reference	Sliped to Late 2018
End of Competition	EM-1 Launch + 365 days	Yes	Future Event

SYSTEMS OVERVIEW

Team Alpha CubeSat has organized the narrative material for each defined System in the following manner:

- **Purpose/Responsibility** – The purpose of each System and its assigned responsibilities are defined.
- **Driving requirements** - The requirements which a given System must meet that are most constraining and/or the most difficult to accommodate (e.g., the tall poles in the tent).
- **Trade space** - The set of potentially viable design solutions for each System is bounded by some combination of first principles physics, driving requirements, as well as cost (i.e., commercial off the shelf ➔ new product), schedule (i.e., availability of product, orchestration of component builds/testing/mandatory design and flight safety reviews/final assembly/integration/launch), and technical (i.e., Technology Readiness Level (TRL), flight heritage, performance/redundancy/availability/margin adequacy) risk.
- **Analysis** - The qualitative and quantitative processes used to evaluate the trade space to draw out the design solutions that are both satisfactory and sufficient.
- **Baseline** – Each System has a baseline architecture which defines the set of subsystems/components which are considered part the System in question for the purposes of the mission.
- **Block diagram** - Each system has a block diagram which shows the delineated subsystems/components, the physical interfaces, augmentations under consideration, and special considerations of note.
- **Design Alternatives under consideration** – These design alternatives come into play where there is either a known System deficiency requiring an augmentation, an area of risk which could require a major design change, and/or an opportunity to enhance System performance that is sufficiently compelling to warrant consideration
- **Identified cost, schedule, and technical risks** – The baseline design choices selected for each System have some identified cost, schedule, and technical risk which the flight project is buying off on mitigating prior to launch..
- **Other related tournament questions** - The tournament workbook, and other Cube Quest Challenge technical documentation raises some number of specific questions which for convenience should explicitly reference where elsewhere in the design document they are or will be addressed, or alternatively addressed in this section.

ELECTRICAL POWER SYSTEM (EPS)

Purpose / Responsibility

The purposes of the Electrical Power System, in order of priority, is to:

1. Accept current from solar panels to operate loads and charge batteries
2. Power loads using stored electrical power when power from solar panels is insufficient or unavailable
3. Measure and report battery condition as well as temperatures to environmental control subsystem
4. Measure and report current draw from discrete subsystems and busses
5. Provide some level of solar output data to GNC which will be used to sanity check sun sensor position data
6. Provide power to secondary payloads as appropriate

It is the responsibility of the power system to maintain the batteries within their nominal envelope in terms of charge / discharge currents, state of charge, and temperature while providing power to all subsystems.

Driving Requirements

All system requirements are driven by the most severe test of the system. Physically, this is the high vibration environment of launch. Thermally, this is deep space solar exposure under high load or fast charge. Maintaining the electrical storage, generation, and load management will be critical in all phases of the mission. Full on load numbers are calculated at 66.5 Watts. The battery system must be able to provide power to necessary systems during the lunar derby while passing through the moon's shadow without suffering damage from an excessively deep discharge or forcing a shutdown of critical systems.

Trade Space

Total load is calculated to be 66.5 Watts, so with the six 3u solar assemblies providing an estimated 96 Watts, there is a sizable margin for both charging concurrent with operation as well as non-optimal off-axis charging which may be necessary to maintain a communication link or stabilize the internal temperatures of the craft. It should not be necessary to expand the panels, but it may become necessary to articulate the panels on one axis should later mission analysis reveal this as a requirement.

Internal battery storage is specified as a 12VDC Li-Ion 7.8 Ah unit if volume and mass budget permit. This should require little in the way of charging on the pad with the craft in a powered down configuration and would provide power during portions of the mission where pointing the communications equipment precludes directing the panels directly sunward or while in lunar orbit in the moon's shadow. Battery-only run time from fully charged is estimated to be 2 hours and 49 minutes at 50% load.

Analysis

While the solar panels in full sun provide plenty of power, there are mission parameters which preclude directing the craft so that the panels are perpendicular to the sun. To keep things simple and reliable the goal was to avoid unnecessary articulation, but depending upon other mission parameters it may become impossible to maintain the battery charge while accomplishing other mission objectives. This may make it necessary to articulate the panels on one axis (180 degrees on pitch axis).

Battery storage is almost excessive for the deep space leg of the mission, but in lunar orbit, up to half of the orbital path will be in lunar shadow. Depending on mission parameters unknown at this time, the battery capacity will need to be reassessed. Should additional capacity be required in the same or less space, other battery configurations or chemistries may be necessary.

The power system will need to be able to remove power from a malfunctioning subsystem temporarily to prevent damage and potentially bring this system back online (hard reset), as well as drop power from less critical systems to conserve power, bringing them back online once current flow is back under control. This is relatively simple to accomplish with hall effect current sensing.

Baseline

The use of commonly available cubesat solar panels in a fixed dual 2x3u stowed, 6x3u deployed configuration providing 96 Watts of power in full sun is baselined.

The use of a 12VDC Li-Ion 7.8 Ah battery is baselined and will be reevaluated once the lunar orbit period is known.

Current control will either be a thermal circuit breaker for loads or hall effect current sensors with solid state relays (SSRs) to interrupt current when necessary. The system using sensors and SSRs is baselined due to the flexibility of this approach.

Some communication between GNC and EPS to verify sensor data (sun position sensor) is baselined.

Block diagram

See Unified Systems Block Diagrams v5.pdf in appendix.

Alternatives under consideration

- No other power generation methods besides solar panels are being considered
- Other power storage methods and battery chemistries are being considered to include LiFe and supercapacitors.
- Some subsystem power control channels may be grouped to simplify the circuit and reduce the size of the power control subsystem.

Identified cost, schedule, and technical risks

Specific off the shelf solution and cost TBA. A partner organization is independently developing a EPS system for a variety of applications including cube- and nanosats.

They have indicated that the product is open source hardware and would be able to adapt the design to our specific requirements. It is unclear if their development schedule will occur in time for use this product in the competition. In the event that they cannot meet this schedule, off the shelf components are available for all major components.

There is a chance that some of the subsystem power estimates are off. In this event, we do have an adequate margin to allow for it without having to rework the system.

Some battery formulations, most notably Lithium Polymer (LiPo), become unstable in the event of physical damage, excessive temperature or charge / discharge rates. These formulations of not being considered.

Other related tournament questions not already addressed

None at this time (TBA)

[reference COTS solar array vendor materials on pages 228-232]

COMMUNICATIONS SYSTEM (COMM)

Purpose / Responsibility

The purposes of the Communications System, in order of priority, is to:

1. Receive and validate commands (CMD)
2. Relay commands to the appropriate subsystem or bus
3. Transmit telemetry including vehicle and subsystem status information (TLM)
4. Transmit the required data for competition packets
5. Transmit and receive data as required for secondary payloads

It is the responsibility of the Communications System to perform the above tasks while staying within legal limits in terms of frequency allocation and power. To this end, licensed radio operators are on staff, both as engineers and advisors.

Driving Requirements

The requirements are driven by the most severe test of the system. Physically, this is the high vibration environment of launch. Thermally, this is deep space solar exposure. Legally Alpha Cubesat's communication system must stay within regulatory bounds in terms of frequency allocation and output power levels. After these requirements have been satisfied, the functional requirement is to provide high speed communications over the 4,000,000 km distance required by the Deep Space Derby portion of the competition.

The maximum distance of the Lunar Derby is under 10% of the distance required by the deep space derby, so a communication system designed to operate in the latter environment will exceed the requirements of the former.

Trade Space

Two frequency bands are currently under consideration, though the team remains open to the use of others. Ka-band (32GHz) is highest on the list followed by UHF (460MHz). Other bands under consideration are L-band (915MHz), C-band (5.7GHz), X-band (10GHz), and Ku-band (12-18GHz).

System Requirements

List all subsystem requirements, duplicating the requirements in the System Design Chapter that are relevant to the communications subsystem. Show how they are derived from, and their relationships to, the system-level requirements that are listed in the System Design Chapter.

Power requirement 35W, actual calculated is 33.3.

Thermal dissipation 30W, actual calculated is 28.3

System Design

Describe and illustrate the subsystem design of the communications subsystem. Show how the subsystem design, once fully implemented, will satisfy all subsystem requirements. Include Interfaces to other subsystems, relevant COTS parts cut sheets or specifications and any other documentation necessary to fully describe the communications subsystem.

In particular, the communications subsystem design description should include:

Alpha will use a Tethers Unlimited SWIFT-KTX programmable SDR transceiver with both a KA band transmitter and an X band receiver on board. The solar panels on the craft double as the antenna arrays thanks to integrated reflectarray antennas similar to that used on ISARA. These arrays have a pencil beam pattern for Ka band, and will also include a region of small antennas for X band reception.

- Complete descriptions of the ground station(s) including locations, transmitters, receivers and antenna patterns

The use of NASA DSN resources is baselined for uplink and downlink, primarily DSN-25 (Goldstone), DSN-34 (Canberra), and DSN-54 (Madrid). The capabilities of these stations are well documented in NASA records, available here: http://deepspace.jpl.nasa.gov/dsndocs/810-005/104/104H.pdf Other ground stations may be used in a backup or contest role including the equipment of HAM radio operators.

- Planned RF frequency bands, or, for optical communications, wavelengths

Uplink (command and control) activity will occur on X band at or around 7.145 GHz. The high speed downlink for telemetry, contest data packets, and payload will occur on Ka band at or around 32 GHz

- Planned transmission powers, modulation methods and coding approaches

The uplink (command and control) activity will use standard QPSK modulation at 30-50W to the dish feed, yielding a link margin of at least 19dB. Higher power transmissions are not a problem. Command and control data security will follow standard practice.

The Ka band high speed downlink will use 16QAM modulation with Reed Solomon forward Error Correction (FEC) at 5W or less. Other power settings, modulation, and FEC methods may be tried should the link fail, as these may be implemented via software commands.

- Include supporting analysis. Analysis should include environmental conditions, margins, uncertainties, assumptions, and operating states, modes and phases.

The supporting analysis is available in the included link budget. The links close, but there may be insufficient margin to achieve a reliable link in the event the receiving station(s) are occluded with heavy cloud cover. Should such conditions occur, it may still be possible to participate in the contest by increasing the transmitter power to a full 5W (intermittently and subject to thermal management) and/or slow the data rate. All of these changes may be triggered by commands on the X band system, which has a substantial margin and is largely unaffected by weather.

System Analysis

Please refer to the included link budget. The analysis tool used is mature and well documented within the spreadsheet. TRL data is available in the included Alpha Cubesat Technology Readiness Level (TRL) document.

DATA MANAGEMENT SYSTEM (DMS)

Purpose/Responsibility

The purposes of the Data Management System, in order of priority, is to:

1. Provide reliable data paths between all spacecraft Systems, subsystems, and/or buses.
2. Provide satisfactory and sufficient computational capacity to process all received command scripts as needed.
3. Provide satisfactory and sufficient computational capacity to process all telemetry including vehicle and subsystem status information (TLM) as needed for transmission.
4. Provide satisfactory and sufficient computational capacity to process and execute all required mode/state transitions.
5. Provide satisfactory and sufficient computational capacity to generate the required encoded bit stream for competition packets.
6. Provide satisfactory and sufficient computational capacity to support secondary payload requirements

It is the responsibility of the Data Management System to perform the above tasks meeting all defined quality of service requirements (i.e., performance, availability, and security) without exceeding the prevailing power and thermal limits for any given operational mode/state as well as not endangering its own ability to function

Driving requirements

The requirements are driven by the most severe test of the system which is anticipated to be the vibration environment at launch and maintaining operational stability in a long duration enhanced radiation environment subject to significant thermal cycling.

The quality of service requirements:

Performance: The DMS must have sufficient computational capacity (Central Processing Unit cycles, cache memory, main memory, and bulk addressable data storage space) to maintain all required code accessible, perform required housekeeping, calculated the encoded bit stream, and ensure that transmit buffer is kept filled to capacity when required to do so. The DMS must throttle its functions as necessary to not exceed the prevailing power and thermal limits for any given operational mode/state as well as not endangering its own ability to function due to high or low temperature conditions.

Availability: The DMS must routinely deal with multiple single event memory upsets without reboot or restart, recover from known cascading multiple event/unanticipated processing conflicts without restart, as well as recover from unknown cascading faults by restart. While the time to recover to a normal operational state is not a quantified requirement at this time, it is anticipated that it will be bounded by a watchdog timer to maximize the probability of recovery in the event of an uncharacterized failure.

Security: The DMS must be able to authenticate the source and validate the integrity of any command scripts received. The DMS must only allow the execution of authenticated and validated command scripts.

It is not anticipated the computational requirements to generate the required encoded bit stream for competition packets will stress the available capacity.

There are no secondary payload computational requirements defined at this time.

Trade space

The set of potentially viable design solutions for the Data Management System is bounded by some combination of first principles physics, driving requirements, as well as cost (i.e., Commercial-Off-The-Shelf (COTS) ➔ new product), schedule (i.e., availability of product, orchestration of component builds/testing/mandatory design and flight safety reviews/final assembly/integration/launch), and technical (i.e., Technology Readiness Level (TRL), flight heritage, performance/redundancy/availability/margin adequacy) risk.

There exist multiple space qualified and potentially space qualifyable Data Management System components and integrated Systems which are available on a COTS basis that could meet or exceed the Alpha CubeSat Data Management System requirements.

Analysis

Current analysis level is qualitative assessment of vendor specification sheets, ongoing technical discussions with other cubesat System developers as well as cubesat users concerning their selections/available products.

In the event that mass, volume, power, and/or other requirements end up driving the Alpha CubeSat to an alternate COTS or semi-custom Data Management System it is anticipated that all elements of defined risk are manageable if not mitigateable.

A near realtime state model of the system is planned to be built using the open source Mission Control Technology suite (a.k.a. WARP) as it is being augmented by the Team Alpha CubeSat founding sponsor (XISP-Inc). This will provide a simulation/operations support environment for interface verification and validation as well as ongoing assessment of system performance, availability, and security. This augmented tool kit is anticipated to be used throughout the development, testing, integration, and operations of the flight system.

Baseline

For the purposes of establishing a conceptual engineering baseline for the Data Management System, and allied systems we have chosen Blue Canyon Technologies XB1 complete CubeSat bus solution as a COTS solution readily adaptable to our design (it is designed to be split into two .5U packages) that meets or exceeds our defined requirements.

The XB1 is a highly integrated, precision spacecraft platform including:

- Ultra high-performance pointing accuracy,
- robust power system,
- command and data handling,
- RF communications,
- propulsion interfaces, and
- multiple flexible payload interfaces.
- Precision stellar-based attitude determination & control provided by dual star trackers.
- Supports precision orbit propagation of multiple target objects with flexible pointing commands to enable a wide range of missions.
- The XB1 Flight Software and simulation environment supports user-developed flight applications.

Block diagram

See Unified Systems Block Diagrams v5.pdf in appendix.

Design Alternatives under consideration

There are no currently known design System deficiencies with the baseline Data Management System solution.

A simulation and operations support environment is being developed to test the efficacy of the system on both a qualitative and quantitative basis.

In the event a System deficiency requiring an augmentation surfaces, an area of risk which requires a major design change is identified, and/or an opportunity to enhance System performance that is sufficiently compelling to warrant consideration emerges it is anticipated that the design to interfaces will defined as to allow plug-in/plug-out replacement.

Identified cost, schedule, and technical risks

There are no currently identified cost, schedule, and/or technical risks associated with the Data Management System baseline design choice that have been flagged as an issue.

However, since the baseline Data Management System is a highly integrated solution if a significant deintegration/repackaging of subsystem components emerges as a requirement the baseline choice will most likely need to change.

Other related tournament questions not already addressed

None at this time (TBA)

STRUCTURES & MECHANISMS

Alpha CubeSat Structures Chapter

Dimensions and Mass Properties of ACS Structure

The structural layout is defined to be a 1Ux1Ux3U center stack with tandem 0.5Ux1Ux3U volumes on either side. This configuration is to position the main propulsive system thrust through the center of gravity of the spacecraft. Deployable trifold solar panels will be attached to the 2Ux3U sides of the spacecraft. Our size is constrained by the SLS Payload User's Guide (SLS-SPIE-HDBK-005) as defined in table 5-1 on page 22, our maximum stowed dimensions cannot exceed:
Width: 239.00mm
Length: 366.00mm
Depth: 113.00mm
Mass: 14 kg.
The outer chassis will bear a significant portion of the design loads and will be modeled in a finite element analysis to prove structural integrity.
The Alpha CubeSat chassis outer mold line dimensions and mass follow the SLS Payload constraints.

<u>Alpha CubeSat chassis outer dimensions and mass properties:</u>
Width: 239.00mm
Length: 366.00mm
Depth: 98.00mm
Maximum Mass: 1 kg
Internal Volume: 6,302 cubic centimeters

The internal volume was calculated assuming similar chassis thickness (approximately 17 mm) as Pumpkin CubeSat products. For example, the Pumpkin 6U CubeSat (SUPERNOVA-Rev00_20140925.doc) states outer length of their spacecraft as 365 mm and inner dimension as 329.2 mm bringing the internal volume to 7000 cc. ACS internal volume is 9.2% smaller due to less depth as a result of folded solar panels.

<u>ACS Inner dimensions:</u>
Width: 206 mm
Length: 329 mm
Depth: 93 mm

These body outer and inner mold line dimensions do not include deployables in their stowed configuration such as the solar panels (each panel is 2.5mm thick per ClydeSpace information) and antenna. The plan is to use three 6U sized panels from ClydeSpace per solar panel array totaling six panels total. With trifold panels, the solar panels in their stowed configuration are expected to be 7.5mm thick in a triple stack and will be faced against the two 2Ux3U faces of the 6U body.

The Alpha CubeSat outer stowed dimensions including all deployables vary from the chassis outer dimensions by 15 mm (symbolizing the 7.5mm thick folded solar panels on either side of the spacecraft) in the depth dimension bringing the Depth to 113.00 mm total. The solar panel mass will not exceed 2.346 kg taking into account a 15% structural mass reserve.

The center of mass envelope is defined in the table below from the CubeQuest Challenge requirements:

Parameters	Units	6U	
		Min.	Max.
Center of Mass, X	in. (mm)	-1.57 (-40)	+1.57 (+40)
Center of Mass Y	in. (mm)	+0.39 (+10)	+2.76 (+70)
Center of Mass Z	in. (mm)	+5.24 (+133)	+9.17 (+233)

Construction

Two options exist for the construction of the outer chassis of the ACS. It is most economical to obtain materials as off-the-shelf, space ready cubesat pieces from Pumpkin and custom machine the pieces to fit our configuration. The materials used for the chassis will be primarily AL7071 and Al6065.

It is also possible we will find a vendor motivated by demonstrating their machining technology that will 3-D print our primary structure using identical aluminum alloys as are commonly used in cubesat construction.

The chassis of the ACS spacecraft will undergo optimization iterations to acquire the lowest mass possible. For the structural analysis, the factors of safety planned to be used are 1.1 for Yield Strength and 1.5 for Ultimate strength as taken from NASA Payload Flight Equipment Requirements and Guidelines for Safety–Critical Structures (SSP 52005 Rev D) Table 5.1.2-1 Minimum Safety Factors For Payload Flight Structures Mounted to Primary and Secondary Structure.

The critical deployable mechanisms on ACS are the two solar panel arrays. Attachment points for the solar panels are constructed as follows. Each wing panel of the trifold are attached to the central panel by leaf-springs from tape measure strips to provide attachment and a mechanism to spring them open. The central panel is attached to the forward face (opposite of the engine exhaust) by a wire coil spring that allows the folded trifold 90 degrees of articulation to fold the stowed panel against the cubesat's 6U body faces. It also provides a mechanism to spring the arrays into their fully-deployed position and a mast attachment point from the array to the satellite body that can be articulated by rotation around the mast's axis to point the array towards the sun

The following section describes the design loads applicable to structure design.

DESIGN LOADS

Launch Loads

The maximum structural loads on the ACS spacecraft will occur during launch.
Launch vibrations have been summarized as x, y, z directional loads in g's as seen in the table below. A finite element analysis is planned for the chassis design and the launch loads will be applied as forces on the satellite located at the contact points of the deployment mechanism and moments around the center of gravity.

ACS will be designed to structural standards as defined in the DESIGN LOADS section of the NASA SECONDARY PAYLOAD INTERFACE DEFINITION AND REQUIREMENTS DOCUMENT (SLS-SPIE-RQMT-018).
Table 3-7 Secondary Payload Component Loads Due to Random Vibration from the Secondary Payload IDRD states:

Configuration 1a – 41lb Payload		
Axial	**Lateral**	**Radial**
±28.2g	±15.6g	±18.0g
Configuration 1b – 60lb Payload		
Axial	**Lateral**	**Radial**
±18.0	±14.3	±18.0
Configuration 2 – Sequencer		
Axial	**Lateral**	**Radial**
±28.2g	±15.6g	±18.0g

The above loads are the maximum load case scenario to be experienced by ACS and correspond to attaining the SLS EM-1 launch.
These loads will be applied to a finite element model of the ACS chassis to prove the design will have sufficient structural integrity.

Temperature Loads

It is also stated in the Secondary Payload IDRD (SLS-SPIE-RQMT-018) that the thermal environment range for spacecrafts is -143 degrees F to +200 degrees F. A finite element model of the ACS structure will undergo a transient thermal analysis to simulate rapid temperature change characteristic of the extreme space environment.

Propulsion Loads

The propulsion loads are planned to not exceed an acceleration higher than 1g. This will be accomplished by designing the HTSD propulsion system to have the appropriate limited thrust. At current, at the fully-loaded mass of 14kg, the thrust maximum can be 137.2N. This

maximum thrust will have to be reduced as the vehicle expends mass in propellants and deployed payloads over the mission.
At this time, COTS solutions for cubesat propulsion have demonstrated thrust that is below this maximum. The exception is the N2O-40% Aluminized Paraffin Hybrid Motor that will exert 10.204gs at 14kg.

However, it is expected that with a proper redesign of the propulsion system to have a throttle, an adjusted chamber pressure, throat area and engine bell expansion ratio, the thrust maximum limit can be achieved.

For more details on propellant amounts, including the total mass of propellant for the GT-2 baselined combination HTSD & LTLD propulsion system that respectively uses a N2O-40% Aluminized Paraffin Hybrid Motor and 4 Busek BIT-1 electric ion thrusters fueled by Iodine, see the Propulsion Chapter of this document [reference page number 62 and 63]. The propellant masses were developed using the original DeltaVs of the GT-1-level trajectory and propulsion system analysis that were required to complete the ACS mission and meet the vehicle mass and volume requirements.
The maximum loads produced by propulsion on the ACS will be applied to the flight configuration (with solar panels deployed) to assure structural integrity of the solar panel deployment mechanism. A finite element model will be created of the ACS and deployed solar panels to test the attachment points specifically and prove they will withstand propulsion loads.

ATTITUDE DETERMINATION & CONTROL SYSTEM (ADCS)

Purpose/Responsibility

The purposes of the Attitude Determination & Control System (ACDS), in order of priority, is to:

1. Provide the necessary, satisfactory, and sufficient sensors to support attitude determination.
2. Provide the necessary, satisfactory, and sufficient actuators to support attitude control.
3. Provide the executable control law logic to read the sensor data and command the actuators to achieve any commanded attitude within a reasonable time frame.

It is the responsibility of the Attitude Determination & Control System to perform the above tasks meeting all defined quality of service requirements (i.e., precision, speed, and parsimonious use of resources both consumable and renewable) without exceeding the prevailing power and thermal limits for any given operational mode/state as well as not endangering its own ability to function

Driving requirements

The requirements are driven by the most severe test of the system which is anticipated to be the vibration environment at launch and maintaining operational stability in a long duration enhanced radiation environment subject to significant thermal cycling and wear due to use.

The quality of service requirements:

<u>Precision</u>: ACDS must meet the attitude determination precision necessary to live within the error bounds of the initial Guidance, Navigation & Control (GN&C) orbital trajectory insertion requirements and any subsequent maneuver requirements. In addition, the ACDS control authority must be satisfactory and sufficient both in total and in usable increments to maintain sun pointing and/or Earth pointing attitudes as needed.

<u>Speed</u>: ACDS must be able to control attitude to a defined point within a reasonable time frame as defined by the mission operations timeline and the available resources.

<u>Parsimonious use of resources both consumable and renewable</u>: ACDS must provide optimized solutions for any control actions to insure the parsimonious use of all resources (e.g., consumable and renewable).

There are no secondary payload ACDS requirements defined at this time.

Trade space

The set of potentially viable design solutions for the Attitude Determination & Control System is bounded by some combination of first principles physics, driving requirements, as well as cost (i.e., Commercial-Off-The-Shelf (COTS) → new product), schedule (i.e., availability of product, orchestration of component builds/testing/mandatory design and flight safety reviews/final assembly/integration/launch), and technical (i.e., Technology Readiness Level (TRL), flight heritage, performance/redundancy/availability/margin adequacy) risk.

There exist multiple space qualified and potentially space qualifyable Attitude Determination & Control System components and integrated Systems which are available on a COTS basis that could meet or exceed the Alpha CubeSat Attitude Determination & Control System requirements.

Analysis

Current analysis level is qualitative assessment of vendor specification sheets, ongoing technical discussions with other cubesat System developers as well as cubesat users concerning their selections/available products.

In the event that mass, volume, power, and/or other requirements end up driving the Alpha CubeSat to an alternate COTS or semi-custom Attitude Determination & Control System it is anticipated that all elements of defined risk are manageable if not mitigateable.

A near realtime state model of the system is planned to be built using the open source Mission Control Technology suite (a.k.a. WARP) as it is being augmented by the Team Alpha CubeSat founding sponsor (XISP-Inc). This will provide a simulation/operations support environment for interface verification and validation as well as ongoing assessment of system performance, availability, and security. This augmented tool kit is anticipated to be used throughout the development, testing, integration, and operations of the flight system.

Baseline

For the purposes of establishing a conceptual engineering baseline for the Data Management System, and allied systems we have chosen Blue Canyon Technologies XB1 complete CubeSat bus solution as a COTS solution readily adaptable to our design (it is designed to be split into two .5U packages) that meets or exceeds our defined requirements.

The XB1 is a highly integrated, precision spacecraft platform including:

- Ultra high-performance pointing accuracy,
- robust power system,
- command and data handling,
- RF communications,
- propulsion interfaces, and
- multiple flexible payload interfaces.

- Precision stellar-based attitude determination & control provided by dual star trackers.
- Supports precision orbit propagation of multiple target objects with flexible pointing commands to enable a wide range of missions.
- The XB1 Flight Software and simulation environment supports user-developed flight applications.

Block diagram

See Unified Systems Block Diagrams v5.pdf in appendix.

Design Alternatives under consideration

There are no currently known design System deficiencies with the baseline Attitude Determination & Control System solution.

A simulation and operations support environment is being developed to test the efficacy of the system on both a qualitative and quantitative basis.

In the event a System deficiency requiring an augmentation surfaces, an area of risk which requires a major design change is identified, and/or an opportunity to enhance System performance that is sufficiently compelling to warrant consideration emerges it is anticipated that the design to interfaces will defined as to allow plug-in/plug-out replacement.

Identified cost, schedule, and technical risks

There are no currently identified cost, schedule, and/or technical risks associated with the Attitude Determination & Control System baseline design choice that have been flagged as an issue.

However, since the baseline Attitude Determination & Control System is a highly integrated solution if a significant deintegration/repackaging of subsystem components emerges as a requirement the baseline choice will most likely need to change.

Other related tournament questions not already addressed

None at this time (TBA)

GUIDANCE, NAVIGATION & CONTROL SYSTEM (GN&C)

Purpose/Responsibility

The purposes of the Guidance, Navigation & Control System (GN&C), in order of priority, is to:

1. Provide the necessary, satisfactory, and sufficient sensors *i.e., Sun Sensor, Star Trackers) to support guidance and navigation (i.e., position and trajectory determination).
2. Provide the executable control law logic to read the sun sensor data and make it available to support Attitude Determination and Control System Sun and Earth pointing solutions as needed..
3. Provide the executable control law logic to read the Star Tracker data and calculate delta trajectory solutions from uploaded baseline.

It is the responsibility of the Guidance, Navigation & Control System to perform the above tasks meeting all defined quality of service requirements (i.e., precision, speed, and parsimonious use of resources both consumable and renewable) without exceeding the prevailing power and thermal limits for any given operational mode/state as well as not endangering its own ability to function

Driving requirements

The requirements are driven by the most severe test of the system which is anticipated to be the vibration environment at launch and maintaining operational stability in a long duration enhanced radiation environment subject to significant thermal cycling and degradation of optical surfaces.

The quality of service requirements:

<u>Precision</u>: GN&C must meet the position and trajectory determination precision necessary to live within the error bounds of the uploaded baseline trajectory at each phase of the mission. In addition, the GN&C must be able to provide position and trajectory determination to enable the ACDS to maintain sun pointing and/or Earth pointing attitudes as needed.

<u>Speed</u>: GN&C must be able to calculate the spacecraft position and make trajectory determination (based on deltas from uploaded baseline trajectory solutions) within a reasonable time frame as defined by the mission operations timeline and the available resources.

<u>Parsimonious use of resources both consumable and renewable</u>: GN&C must provide position and trajectory determination capabilities sufficient to allow uploaded navigation solutions to be optimized to insure the parsimonious use of all resources (e.g., consumable and renewable).

There are no secondary payload GN&C requirements defined at this time.

Trade space

The set of potentially viable design solutions for the Guidance, Navigation & Control System is bounded by some combination of first principles physics, driving requirements, as well as cost (i.e., Commercial-Off-The-Shelf (COTS) ➔ new product), schedule (i.e., availability of product, orchestration of component builds/testing/mandatory design and flight safety reviews/final assembly/integration/launch), and technical (i.e., Technology Readiness Level (TRL), flight heritage, performance/redundancy/availability/margin adequacy) risk.

There exist multiple space qualified and potentially space qualifyable Guidance, Navigation & Control System components and integrated Systems which are available on a COTS basis that could meet or exceed the Alpha CubeSat Attitude Determination & Control System requirements.

Analysis

Current analysis level is qualitative assessment of vendor specification sheets, ongoing technical discussions with other cubesat System developers as well as cubesat users concerning their selections/available products.

In the event that mass, volume, power, and/or other requirements end up driving the Alpha CubeSat to an alternate COTS or semi-custom Attitude Determination & Control System it is anticipated that all elements of defined risk are manageable if not mitigateable.

A near realtime state model of the system is planned to be built using the open source Mission Control Technology suite (a.k.a. WARP) as it is being augmented by the Team Alpha CubeSat founding sponsor (XISP-Inc). This will provide a simulation/operations support environment for interface verification and validation as well as ongoing assessment of system performance, availability, and security. This augmented tool kit is anticipated to be used throughout the development, testing, integration, and operations of the flight system.

Baseline

For the purposes of establishing a conceptual engineering baseline for the Data Management System, and allied systems we have chosen Blue Canyon Technologies XB1 complete CubeSat bus solution as a COTS solution readily adaptable to our design (it is designed to be split into two .5U packages) that meets or exceeds our defined requirements.

The XB1 is a highly integrated, precision spacecraft platform including:

- Ultra high-performance pointing accuracy,
- robust power system,
- command and data handling,

- RF communications,
- propulsion interfaces, and
- multiple flexible payload interfaces.
- Precision stellar-based attitude determination & control provided by dual star trackers.
- Supports precision orbit propagation of multiple target objects with flexible pointing commands to enable a wide range of missions.
- The XB1 Flight Software and simulation environment supports user-developed flight applications.

Block diagram

See Unified Systems Block Diagrams v5.pdf in appendix.

Design Alternatives under consideration

There are no currently known design System deficiencies with the baseline Guidance, Navigation & Control System solution.

A simulation and operations support environment is being developed to test the efficacy of the system on both a qualitative and quantitative basis.

In the event a System deficiency requiring an augmentation surfaces, an area of risk which requires a major design change is identified, and/or an opportunity to enhance System performance that is sufficiently compelling to warrant consideration emerges it is anticipated that the design to interfaces will defined as to allow plug-in/plug-out replacement.

Identified cost, schedule, and technical risks

There are no currently identified cost, schedule, and/or technical risks associated with the Guidance, Navigation & Control System baseline design choice that have been flagged as an issue.

However, since the baseline Guidance, Navigation & Control System is a highly integrated solution if a significant deintegration/repackaging of subsystem components emerges as a requirement the baseline choice will most likely need to change.

Other related tournament questions not already addressed

None at this time (TBA)

PROPULSION SYSTEM (PROP)

DEVELOPMENT SUMMARY:

The GT-2-level propulsion system development has evaluated several candidate COTS options and determined a few that can deliver on the DeltaV requirements of the ACS mission trajectory. As of GT-2, we have updated our trajectory analysis to take advantage of low-energy and Weak Stability Boundary characteristics of the EM system, we have greatly reduced our DeltaV requirement from the GT-1 total of ~1.3km/s to 180m/s (CITATION: Trajectory Report for GT-2 and Belbruno Trajectory [reference page 148]).

The main benefit of this reduced DeltaV requirement is that we can now consider several COTS propulsion systems and propellants for evaluation. We first determined those who had, by the manufacturer's specifications, were able to provide a DeltaV for a 6U cubesat that exceeded our required DeltaV in stock configuration. We estimated by scaling the DeltaVs of these systems to a 6U vehicle. The propulsion system candidates, their scaled DeltaVs and TRLs are as follows (compared to our GT-1 Baseline configuration):

System	Propellant(s)	Scaled DeltaV for 6U (m/s)	TRL
Phase 4 CAT (P4-50) Ambipolar Thruster	Iodine	989.5 (1,979 for 3U)	5
	Water	744 (1,499 for 3U)	5
Tethers Unlimited HYDROS	Water	150 (Scalable to >2km/s)	5
BASELINE: Busek BIT-1	Iodine	1,333.6 (GT-1 Calculation)	5
BASELINE: Hybrid Motor	N2O-40% Aluminized Paraffin	228.0 (GT-1 Calculation)	5

CITATION: Manufacturer's Specifications on propulsion systems [are in GT1 and GT2 Reports].

From here, the candidates will be evaluated by their following qualities:

Quality	Purpose
Propellant Safety	Compatibility to NASA Cabin Standards to allow vehicle operations in the ISS [See applicable safety requirements documents on page 293].
System Mass & Volume	Determination of fit of propulsion system into 6U mass and form factor.
Total Runtime Required	As several of the candidate systems have a total thrust of 1N or less, it is expected runtimes will need to be extended to impart the required total impulse for a given DeltaV for a specific trajectory maneuver.
Maximum Thrust Does Not Exceed 1g Acceleration	Due to ACS's deployables having a structural limit of 1g (9.8 m/s^2 of acceleration for in-space maneuvers.

These qualities can be quantified as so:

- Propellant Safety – Per the NASA Cabin Safety Standards [See applicable safety requirements documents on page 293]. , we are not permitted to use propellants that are inherently reactive, unstable or toxic to life. Propellants and individual components must be inert on their own when unprovoked by any external energetic force and in safed configuration.
- System Mass & Volume – Prior GT-1 propulsion development work had placed a goal limit of less than 3,000 cm^3 volume and 10 kg mass for the propulsion system and propellants to allow reservation for other systems. The propulsion system must meet or exceed the same requirements.

- Total Runtime Required – In orbital mechanical analysis, maneuvers are approximated as instantaneous accelerations given that the propulsion system burn time is sufficiently short compared to the trajectory's transit time. Also, propulsion systems have an upper limit on the operation time. Hence, to allow accuracy to the trajectory analysis, the runtime should not exceed more than 1% of a given flight leg. Also the runtime should not exceed the manufacturer's lifetime limit. To enable this, the propulsion system needs to have sufficient thrust and runtime to impart sufficient impulse for a given DeltaV maneuver.
- Maximum Thrust Does Not Exceed 1g Acceleration – Propulsion system, for the 6U mass of 14kg, must not have a thrust that exceeds 137.2N so that acceleration on the vehicle does not exceed 1g (9.8 m/s^2). This is due to the defined structural limit of deployable systems in the Structures & Mechanisms section of the ACS GT-2 report [reference S&Mech section on page 57].

ANALYSIS:

The candidate propulsion systems were analyzed using classical propulsion theory and information on the DeltaV of the specified GT-2-level trajectory. Manufacturer's specifications on the propulsion systems' I_{sp}, Thrust, Propellants were used to develop quantifications of the propulsion system's mass and volume and total runtime required within, if applicable, the above maximum thrust limit.

The following information was gathered. More can be seen in the attached Propulsion Analysis Workbook [reference Propulsion System calculations pages 153 and 154].

System	Propellant(s)	Propellant Mass (kg)	Propellant Volume (cm^3,U)	Total Runtime (days, % of Total)
Phase 4 CAT (P4-50) Ambipolar Thruster	Iodine	0.51	102.85, 0.10	10.71 (3.40%)
	Water	0.16	158.36, 0.16	66.26 (21.03%)
Tethers Unlimited HYDROS	Water	0.85	854.85, 0.85	0.04 (0.01%)
BASELINE: Busek BIT-1	Iodine	0.21	43.41, 0.04	72.87 (23.13%)
BASELINE: Hybrid Motor	N2O-40% Aluminized Paraffin	1.28	1042.05, 1.04	0.0002 (0.00%)

CONCLUSIONS:

The reduced DeltaV of the Belbruno trajectory allows us to eliminate the combination HTSD-LTLD propulsion system. All propulsion systems meet the mass and volume limitations established.

The only two propulsion systems that have been elimiated are the Phase 4 CAT (P4-50) Ambipolar Thruster using Water and Busek BIT-1 using Iodine have overly long propulsion runtimes required to impart the required impulse for the required DeltaV.

The remaining candidates that meet requirements are the N2O-40% Aluminized Paraffin HTSD motor, HYDROS and Phase 4 CAT (P4-50) Ambipolar Thruster using Iodine.

FUTURE DESIGN METHODOLOGY:

At this time, the most important determinant to select a propulsion system for HTSD system that meets mission requirements is I_{sp} as it determines the DeltaV capable. The baselined propulsion system using N2O-40% Aluminized Paraffin for HTSD have an expected and demonstrated I_{sp} of 200s. With these values, the propulsion system has sufficient DeltaV to meet the predicted DeltaV required by the GT-2-level trajectory analysis. For this reason, any other propulsion system candidate needs to meet or exceed this I_{sp} minimum.

Also, from the structural requirements, the propulsion system design is required to not have the vehicle at any time and at any loaded mass under HTSD propulsion experience an acceleration higher than 1g. This is the structural limit of deployable systems. It is intended that this will be accomplished by designing the system to have limited thrust by an appropriate sizing of the elements and operating conditions of the rocket nozzle and combustion chamber.

For this reason, there is a strong need to understand the math and physics-based relationship between Thrust and I_{sp} for HTSD propulsion. For HTSD propulsion, the Thrust and I_{sp} are related to the design of the propulsion system's combustion chamber dimensions, chamber pressure, throat area and nozzle expansion ratio. For this reason, a unique combustion chamber and nozzle will be sized and baselined that fits into the 6U form factor and produces the appropriate thrust at or higher than the required I_{sp}. From this, a variety of propulsion systems and propellant configurations can be evaluated.

THERMAL CONTROL SYSTEM (TCS)

Purpose/Responsibility

The purpose of the Thermal Control System is to dissipate System heat loads:

1. Electrical Power System Passive Thermal Dissipation
    - Solar Array Subsystem Passive Dissipation
    - Power Management and Distribution Subsystem Passive Dissipation
    - Battery Subsystem Passive Thermal dissipation
2. Data Management System Passive Thermal Dissipation
3. Propulsion System Passive Thermal Dissipation
4. Communications System Passive Thermal Dissipation
5. Guidance Navigation and Control System Passive Thermal Dissipation
6. Attitude Determination and Control System Passive Thermal Dissipation
7. Structures & Mechanisms Passive Thermal Dissipation

It is the responsibility of the Thermal Control System to assure that the spacecraft neither becomes too hot and sustains damage or becomes too cold and sustains damage.

Driving requirements

The Thermal Control System must maintain the heat balance in at least three challenging modes.

1. During the use of the hybrid propulsion system
2. During extended flight with either the ion thrusters on or off
3. During competition communications tests

Trade space

The set of potentially viable design solutions for the Thermal Control System is bounded by some combination of first principles physics, driving requirements, as well as cost (i.e., commercial off the shelf new product), schedule (i.e., availability of product, orchestration of component builds/testing/mandatory design and flight safety reviews/final assembly/integration/launch), and technical (i.e., Technology Readiness Level (TRL), flight heritage, performance/redundancy/availability/margin adequacy) risk.

The Thermal System uses some combination of tools to move heat:

1. Heat Pipes (baseline)
2. Peltier Effect Tiles (potential augment 1)
3. Phase Change Materials (Single) (potential augment 2)
4. Phase Change Materials (Dual) (potential augment 2)

The Thermal System uses some combination of tools to mitigate and/or reject heat to the environment:

1. Attitude Precision (Sun Pointing)
2. Radiator (Passive)
3. Temperature Sensors
4. Thermal Management Controllers
5. Spacial Adjacency of Equipment
6. Distribution of Equipment in Spacecraft
7. Power Cycling of Equipment

It is anticipated that all identified tools and strategies will be used with the exception of the three identified augments. The augments will be used if the passive tools to move heat are deemed insufficient.

Analysis

The qualitative and quantitative processes used to evaluate the trade space to draw out the design solutions that are both satisfactory and sufficient.

We have completed a thermal dissipation calculation for a solar panel.

We have created a spreadsheet based heat balance model

We need to verify the accuracy of the Emissivity values for all radiating surfaces or surfaces with solar load (earth load or moon load).

We are maintaining calculation workbook book with scanned notes and sketches.

The cognizant thermal engineer has outlined 5 different internal load cases:

I1 through I5.

1. All systems off
2. Full Power, Everything turned on, absolute worst case
3. Standby Mode
4. Transmit only
5. Normal Operation

And 5 different positional based external load cases

1. Ex1) LEO Day
2. Ex2) LEO Night
3. Ex3) Moon Orbit Day
4. Ex4) Moon Orbit, Dark Side
5. Ex5) Deep Space (i.e. far enough away from large objects there is only a solar load)

This makes for 25 load cases.

We are starting the analysis with I5-EX5. Deep space-normal operation and will then continue to develop I5-EX1 LEO Day, normal operation. Once the template is setup, the other 23 cases will be generated as time permits.

Energy Balance Assumptions.

- Also assumed no power scenario in LEO.
- Exented surfaces used were minimal. Approximately .1 meters squared of surface area for rejected heat to space.
- Standard concept of conducting the system waste heat to the back side of the satellite, located away from the solar load,

The Energy Balance spreadsheet assumes the Ion thruster would have 50% of its surface area exposed and radiating to space. This helped reduce the size of additional heat rejecting surfaces we have to consider as part of the design. We may be able to get away without such features (exposed heat pipe surfaces), but it will mean less radiative power to emit unwanted energy, and higher operating temperatures for the onboard systems. Looks like in LEO we will be on the order of 330 K (57 C) external surface temperatures when running at full power and Ion Thrusters turned on.

Baseline

The heat loads, tools, and strategies for dissipation, movement, and overall management have been identified on a qualitative basis and the quantitative analysis has begun.

Based on the available mass, volume, and power only passive systems are baselined.

If subsequent analysis determines active systems are required several options have been identified and will be actively tracked as resources that can be added to the design if required.

Block diagram

See Unified Systems Block Diagrams v5.pdf in appendix.

Design Alternatives under consideration

There are no currently known design System deficiencies with the baseline Thermal Control System solution.

A simulation and operations support environment is being developed to test the efficacy of the system on both a qualitative and quantitative basis.

In the event a System deficiency requiring an augmentation surfaces, an area of risk which requires a major design change is identified, and/or an opportunity to enhance System performance that is sufficiently compelling to warrant consideration emerges it is anticipated that the design to interfaces will defined as to allow plug-in/plug-out replacement.

Identified cost, schedule, and technical risks

There are no currently identified cost, schedule, and/or technical risks associated with the Thermal Control System baseline design choice that have been flagged as an issue.

Other related tournament questions

None at this time (TBA)

PRIMARY PAYLOAD

Purpose/Responsibility

The primary payload for Alpha CubeSat is the Cube Quest Challenge encoded bit stream generator.

Driving requirements

Deep Space Derby Prizes:

• **Best Burst Data Rate:** $225,000 will be awarded to the competitor team (as defined in challenge rules) that receives the largest volume of error-free data from their CubeSat over a 30-minute period from greater than 4 million kilometers; $25,000 will be awarded to the competitor team that receives the second largest volume of error-free data.

• **Largest Aggregate Data Volume Sustained Over Time:** $675,000 will be awarded to the competitor team that receives the largest cumulative volume of error-free data from their CubeSat over a continuous 28-day period from greater than 4 million kilometers; $75,000 will be awarded to the Competitor team that receives the second largest volume of error-free data.

• **Spacecraft Longevity:** $225,000 will be awarded to the competitor team with the longest elapsed number of days between the first and the last confirmed reception of error-free data from their CubeSat from greater than 4 million kilometers; $25,000 will be awarded to the competitor team with the second longest elapsed number of days between the first and the last confirmed reception of error-free data.

• **Farthest Communication Distance from Earth:** $225,000 will be awarded to the competitor team that receives at least one, error-free, CubeSat-generated data block from the greatest distance beyond a minimum of 4 million kilometers; $25,000 will be awarded to the competitor team with the second greatest distance.

NASA will award the following Lunar Derby Prizes:

• **Lunar Propulsion:** $1,500,000 will be divided equally between all competitor teams that achieve at least one verifiable lunar orbit, with a maximum of $1,000,000 to any one competitor team.

• **Best Burst Data Rate:** $225,000 will be awarded to the competitor team that receives the largest cumulative volume of error-free data from their CubeSat over a 30-minute period while in lunar orbit; $25,000 will be awarded to the competitor team that receives the second largest volume of error-free data.

• **Largest Aggregate Data Volume Sustained Over Time:** $675,000 will be awarded to the Competitor team that receives the largest cumulative volume of error-free data from their CubeSat over a continuous 28-day period while in lunar orbit; $75,000 will be awarded to the competitor team that receives the second largest volume of error-free data.

• **Spacecraft Longevity:** $450,000 will be awarded to the competitor team that achieves the longest elapsed number of days between the first and last confirmed reception of error-free data from their CubeSat while in lunar orbit; $50,000 will be awarded to the competitor team that achieves the second longest elapsed number of days between the first and last confirmed reception of error-free data.

Trade space

The only trade space with respect to the primary payload is determining which competitions your team will compete in. In the case of Team Alpha CubeSat we have chosen to compete in both the Deep Space Derby and the Lunar Derby, and will attempt to design to win all challenges.

Analysis

We will develop both qualitative and quantitative models to evaluate the efficacy of the Team Alpha CubeSat design.

The current level of analysis shows that the:

- communication link budget closes with positive margin for both the Deep Space Derby and the Lunar Derby.
- The first order trajectory calculation based on SLS launch closes for the combined mission. The ISS trajectory calculation requires further work.
- The first order propulsion calculations based on SLS launch closes for the combined mission. The ISS trajectory calculation requires further work.
- The first order volume, mass, and power budgets based on SLS launch closes for the combined mission. The ISS trajectory calculation requires further work

Baseline

This report defines a baseline architecture for each System that appears tractable for SLS launch baseline. The ISS alternative requires further work.

Block diagram

Each system has a block diagram which shows the delineated subsystems/components, the physical interfaces, augmentations under consideration, and special considerations of note.

See Unified Systems Block Diagrams v5.pdf in appendix.

Design Alternatives under consideration

There are no primary payload design alternatives that have been defined or are anticipated.

Identified cost, schedule, and technical risks

The choice to baseline participation in both the Deep Space Derby and the Lunar Derby as well as all competitions has some elements of increased risk. However, the baseline design choices selected for each System appear to have resulted in a more robust spacecraft design which likely may prove more capable of meeting the competition performance objectives. Team Alpha CubeSat will rely on both qualitative and quantitative analysis to determine if the aggregated cost, schedule, and technical risk which the flight project is buying off can be practically mitigated prior to launch.

Other related tournament questions

SCAR FOR SECONDARY PAYLOAD

Not applicable at the present state of the design. Multiple commercial opportunities have been identified and will be defined to a level that would allow them to be accommodated if the design margin is determined to be available.

OPERATIONAL MODES AND TRANSITIONS

A block diagram showing the anticipated Alpha CubeSat Mode/State Transitions is attached in the System Block Diagram Package. This diagram was extrapolated from an existing 3U communication spacecraft design (BitSat, by Deep Space Industries, Inc) with unique extensions to accommodate additional modes and allow for a more deterministic transition flow.

Based on our qualitative assessment it is anticipated that a simplified control logic flow is possible for Alpha CubeSat focused on three primary flight regimes:

1. Prepare for operations
2. Achieve a Navigation Milestone
3. Achieve a Communication Milestone

A conventional Alpha CubeSat Mode/Transitions table is also attached System Block Diagram Package.

SPACECRAFT SYSTEMS	Volume without Contingency (U)	Contingency %	Contingency (U)	Volume with Contingency (U)
Electrical Power System (EPS)				
Power Management and Distribution	see XB1			see XB1
Solar Arrays (conformal exterior)	0.720			0.720
Batteries (conformal propulsion tank corners)	see XB1			see XB1
Communications System (COMM)				
Ka Band Radio	0.330			0.330
UHF Radio	see XB1			see XB1
Antenna (TX+RX integrated w/solar arrays)	see solar			see solar
Data Management System (DMS)				
On Board Computer	see XB1			see XB1
Structures & Mechanisms				
Integrated with each system	0.000			0.000
Attitude Determination & Control System (ADCS)				
Subsystems	see XB1			see XB1
Guidance, Navigation & Control System (GN&C)				
Subsystems	see XB1			see XB1
Propulsion System				
Hybrid Trajectory Injection Motor Core	2.000			2.000
Hybrid Trajectory Injection Motor Fuel Tank	1.000			1.000
Ion Thrusters (Four Total)	0.500			0.500
Ion Propellant Tanks (Two Total)	1.000			1.000
Thermal System				
Integrated with each system	0.000			0.000
Primary Payload Encoded Bit Stream				
Allocated to Data System	0.000			0.000
Scar for Secondary Payload (future)	0.000			0.000
CubeSat Bus	1.000			1.000
Estimated Spacecraft Total Volume	5.550	8.11%	0.450	6.000
Total Allowable Spacecraft Volume (U)	6.000			6.000

SPACECRAFT SYSTEMS	MASS without Contingency (kg)	Contingency %	Contingency (kg)	MASS with Contingency (kg)
Electrical Power System (EPS)				
Power Management and Distribution	see XB1			see XB1
Solar Arrays (conformal exterior)	0.300			0.300
Batteries (conformal propulsion tank corners)	see XB1			see XB1
Communications System (COMM)				
Ka Band Radio	0.375			0.375
UHF Radio	see XB1			see XB1
Antenna (TX+RX integrated w/solar arrays)	see solar			see solar
Data Management System (DMS)				
On Board Computer	see XB1			see XB1
Structures & Mechanisms				
Integrated with each system	0.000			0.000
Attitude Determination & Control System (ADCS)				
Subsystems	see XB1			see XB1
Guidance, Navigation & Control System (GN&C)				
Subsystems	see XB1			see XB1
Propulsion System				
Hybrid Trajectory Injection Motor Core	3.000			3.000
Hybrid Trajectory Injection Motor Fuel Tank	5.000			5.000
Ion Thrusters (Four Total)	0.352			0.352
Ion Propellant Tanks (Two Total)	3.000			3.000
Thermal System				
Integrated with each system	0.000			0.000
Primary Payload Encoded Bit Stream				
Allocated to Data System	0.000			0.000
Scar for Secondary Payload (future)	0.000			0.000
CubeSat Bus	1.500			1.500
Estimated Spacecraft Total Mass	13.527	3.50%	0.473	14.000
Total Allowable Spacecraft Mass (kg)	14.000			14.000

Power Budget

SPACECRAFT SYSTEMS	Power without Contingency (w)	Contingency %	Contingency (w)	Power with Contingency (w)
Electrical Power System (EPS)	90.000			90.000
Power Management and Distribution	0.000			0.000
Solar Arrays (conformal exterior)	0.000			0.000
Batteries (conformal propulsion tank corners)	0.000			0.000
Communications System (COMM)	0.000			
Ka Band Radio	0.000			0.000
Antenna (TX+RX integrated w/solar arrays)	0.000			0.000
Data Management System (DMS)	0.000			
On Board Computer	0.000			0.000
Structures & Mechanisms	0.000			
Integrated with each system	0.000			0.000
Attitude Determination & Control System (ADCS)	0.000			
Subsystems	0.000			0.000
Guidance, Navigation & Control System (GN&C)	0.000			
Subsystems	0.000			0.000
Propulsion System	0.000			
Hybrid Trajectory Injection Motor Core	0.000			0.000
Hybrid Trajectory Injection Motor Fuel Tank	0.000			0.000
Ion Thrusters (Four Total)	0.000			0.000
Thermal System	0.000			
Integrated with each system	0.000			0.000
Primary Payload Encoded Bit Stream	0.000			
Allocated to Data System	0.000			0.000
Scar for Secondary Payload (future)	0.000			0.000
Estimated Spacecraft Total Power	66.460			0.000
Total Spacecraft Power Margin* (w)	23.540			0.000
*Assumes solar array as source, battery can supplement and/or make up for non-optimal pointing.				

Baseline Design Correlation/Cross Check

Alpha CubeSat Mass Budget Correlation

System	Subsystem	Part Name	Description	Vendor	Quantity	Mass	Total Mass
						gram	gram
Power	Solar Array	3U CubeSat Solar Panel	Solar Reflectenna Array	Pumpkin	12	170	2040
Electronics	Bus	XB1 Cubesat Bus	AGPS, C&DH, EPS and Battery Pack	GomSpace	1	1150	1150
	Ka Transceiver	SWIFT-KTX	Ka Band Transciever	Tethers Unlimited	1	500	500
Propulsion	Ion Thrusters	BIT-1	Ion Thruster	Busek	4	53	212
	Ion Tank		Ion Iodine Propellant and Tank		2	3129.09	6258.188
	Ion Feed Valve		Ion Feed Valve	Busek	4	35	140
	Chemical	Chemical Propulsion	Aerojet unit with propellant as reference	Aerojet	1	3200	3200
Total				**Maximum Consumption**			**13500.188**

Estimated Baseline Mass Consumption	13500.19
Total Mass Budget	14000
Estimated Spacecraft Level Mass Margin (kg)	499.81
Estimated Spacecraft Level Mass Margin (%)	3.57%

Alpha CubeSat Power Budget Correlation

System	Subsystem	Part Name	Description	Vendor	Quantity	Power	Total Power
						watts	watts
Power	Solar Array	3U CubeSat Solar Panel	Solar Reflectenna Array	Pumpkin	12	8.00	96.00
Total				**Maximum Production**			**96.00**
Electronics	Bus	XB1 Cubesat Bus	AGPS, C&DH, EPS and Battery Pack	GomSpace	1	6.30	6.30
	Ka Transceiver	SWIFT-KTX	Ka Band Transciever	Tethers Unlimited	1	16.00	16.00
Propulsion	Ion Thrusters	BIT-1	Ion Thruster	Busek	4	10.00	40.00
	Ion Tank		Ion Iodine Propellant and Tank		2	0.00	0.00
	Ion Feed Valve		Ion Feed Valve	Busek	4	0.04	0.16
	Chemical	Chemical Propulsion	Aerojet unit with propellant as reference	Aerojet	1	4.00	4.00
Total				**Maximum Consumption**			**66.46**

Estimated Baseline Power Consumption	66.46
Total Power Budget (watts)	96.00
Estimated Spacecraft Level Power Margin (watts)	29.54
Estimated Spacecraft Level Power Margin (%)	30.77%

Alpha CubeSat Volume Budget Correlation

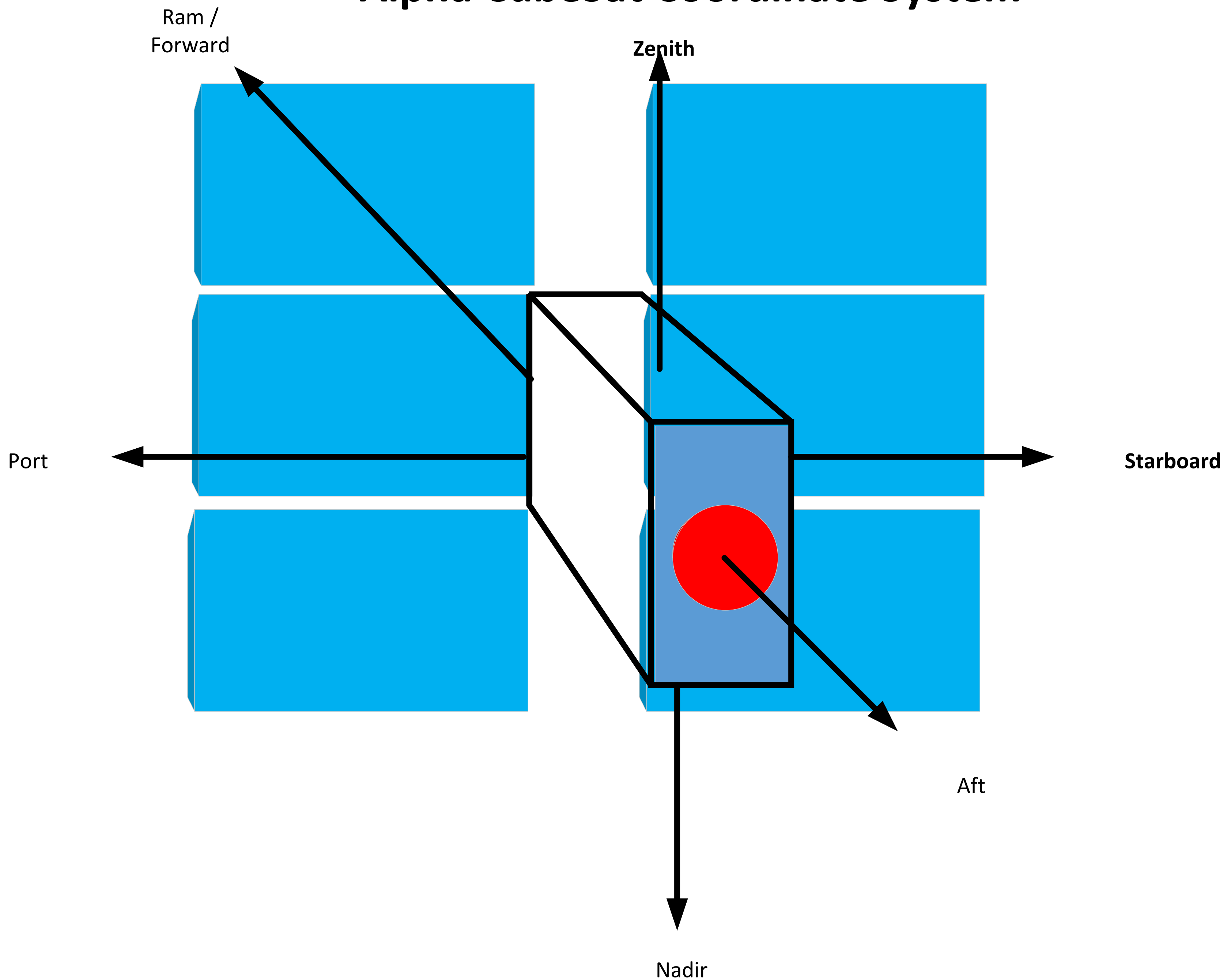
System	Subsystem	Part Name	Description	Vendor	Quantity	Length	Width	Height	Volume	Total Volume
						mm	mm	mm	U	U
Power	Solar Array	3U CubeSat Solar Panel	Solar Reflectenna Array	Pumpkin	12	100.00	100.00	2.00	0.02	0.24
Electronics	Bus	XB1 Cubesat Bus	AGPS, C&DH, EPS and Battery Pack	GomSpace	1	200.00	100.00	50.00	1.00	1.00
	Ka Transceiver	SWIFT-KTX	Ka Band Transciever	Tethers Unlimited	1	86.00	86.00	45.00	0.33	0.33
Propulsion	Ion Thrusters	BIT-1	Ion Thruster	Busek	4	34.60	dia.	28.80	0.11	0.43
	Ion Tank		Ion Iodine Propellant and Tank		2	150.00	100.00	50.00	0.75	1.50
	Ion Feed Valve		Ion Feed Valve	Busek	4	20.00	20.00	25.00	0.01	0.04
	Chemical	Chemical Propulsion	Aerojet unit with propellant as reference	Aerojet	1	227.00	100.00	100.00	2.27	2.27
Total						**Maximum Consumption**				**5.82**

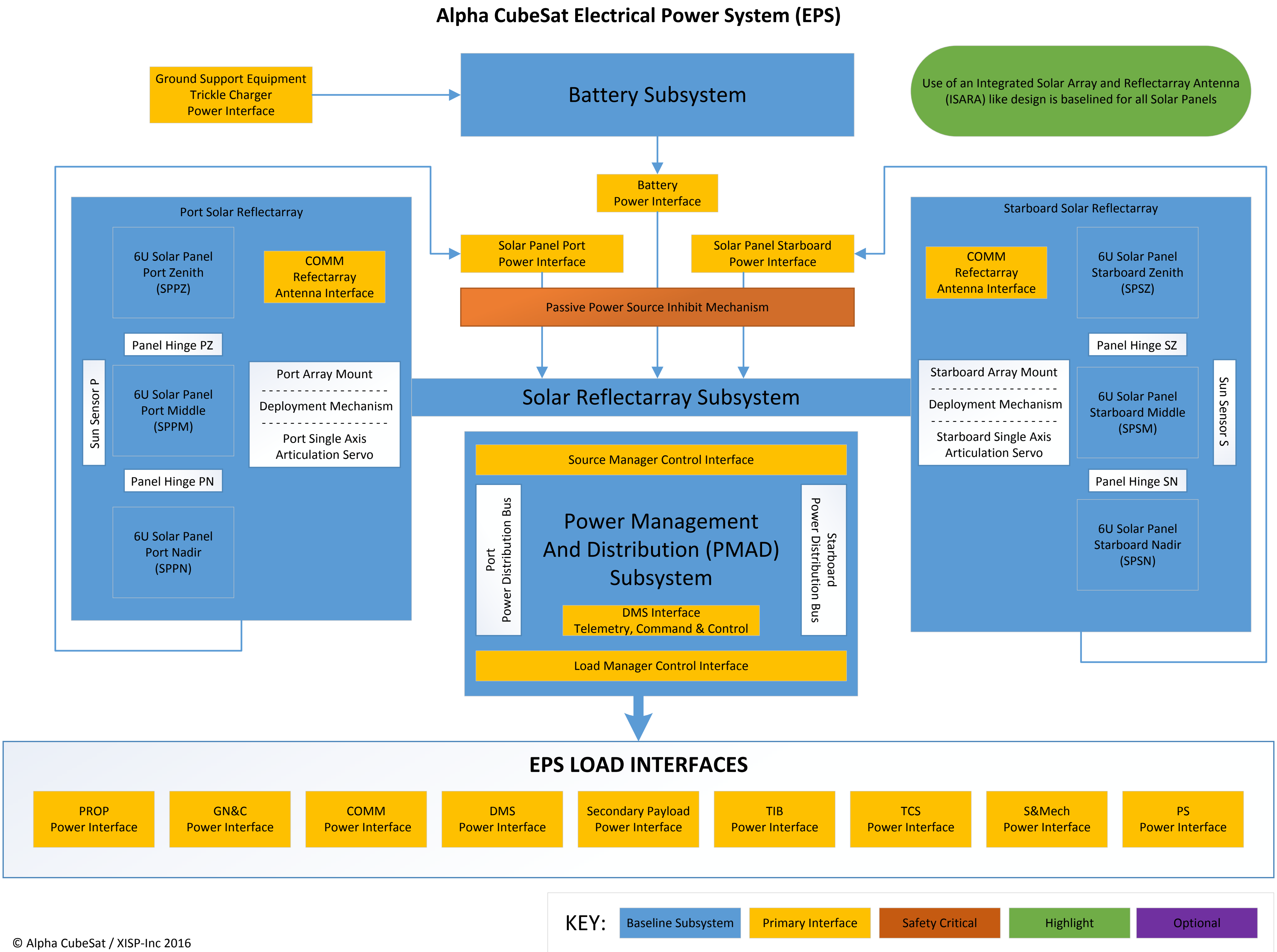
Estimated Baseline Volume Consumption	5.82
Total Volume Budget	6
Estimated Spacecraft Level Volume Margin (U)	0.18
Estimated Spacecraft Level Volume Margin (%)	3.07%

Based on the available mass, power and volume budgets as well as the current baseline component assessments there are positive spacecraft margins for mass, power and volume.

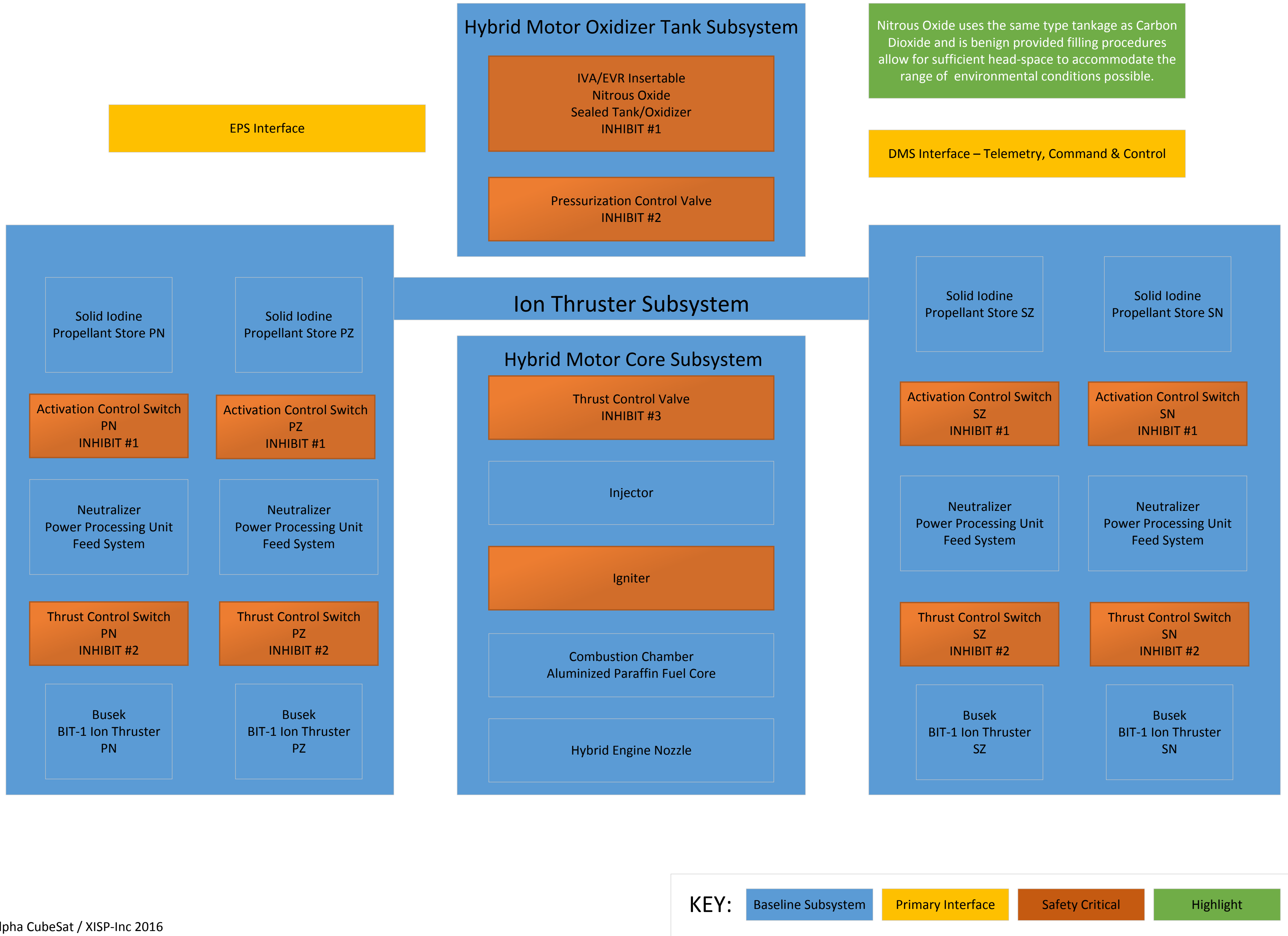
Based on the calculated values some reoptimization of the System level design may be warranted to allow for System and subsystem margin allocation as part of the preliminary design process.

Alpha CubeSat Coordinate System

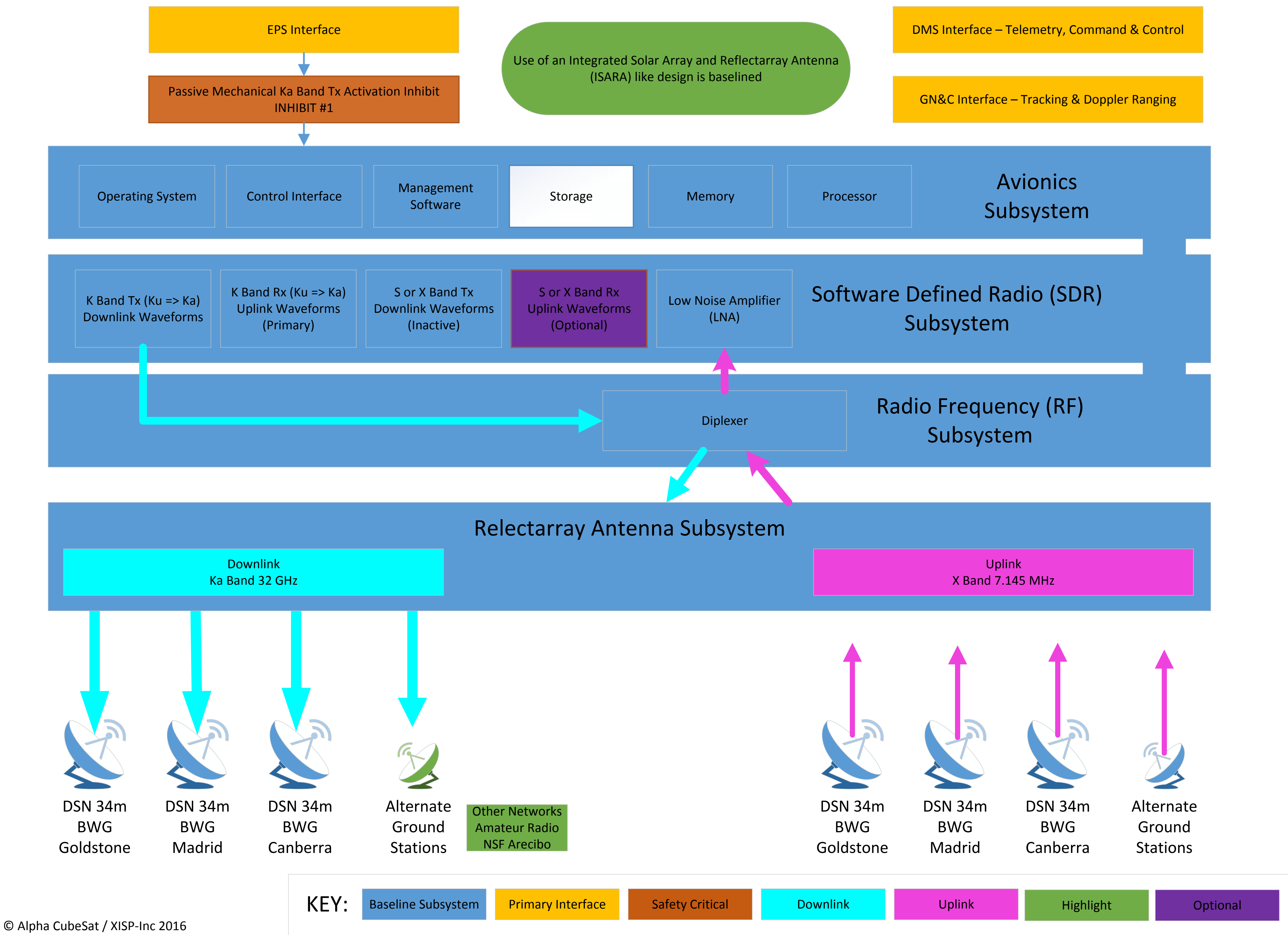



Alpha CubeSat Electrical Power System (EPS)
Ground Support Equipment Trickle Charger Power Interface
Battery Subsystem
Use of an Integrated Solar Array and Reflectarray Antenna (ISARA) like design is baselined for all Solar Panels
Battery Power Interface
Solar Panel Port Power Interface
Solar Panel Starboard Power Interface
Passive Power Source Inhibit Mechanism
Port Solar Reflectarray
6U Solar Panel Port Zenith (SPPZ)
COMM Refectarray Antenna Interface
Panel Hinge PZ
Sun Sensor P
6U Solar Panel Port Middle (SPPM)
Port Array Mount
Deployment Mechanism
Port Single Axis Articulation Servo
Panel Hinge PN
6U Solar Panel Port Nadir (SPPN)
Solar Reflectarray Subsystem
Starboard Solar Reflectarray
COMM Refectarray Antenna Interface
6U Solar Panel Starboard Zenith (SPSZ)
Panel Hinge SZ
Starboard Array Mount
Deployment Mechanism
Starboard Single Axis Articulation Servo
6U Solar Panel Starboard Middle (SPSM)
Sun Sensor S
Panel Hinge SN
6U Solar Panel Starboard Nadir (SPSN)
Source Manager Control Interface
Port Power Distribution Bus
Power Management And Distribution (PMAD) Subsystem
Starboard Power Distribution Bus
DMS Interface Telemetry, Command & Control
Load Manager Control Interface
EPS LOAD INTERFACES
PROP Power Interface
GN&C Power Interface
COMM Power Interface
DMS Power Interface
Secondary Payload Power Interface
TIB Power Interface
TCS Power Interface
S&Mech Power Interface
PS Power Interface
KEY:
Baseline Subsystem
Primary Interface
Safety Critical
Highlight
Optional

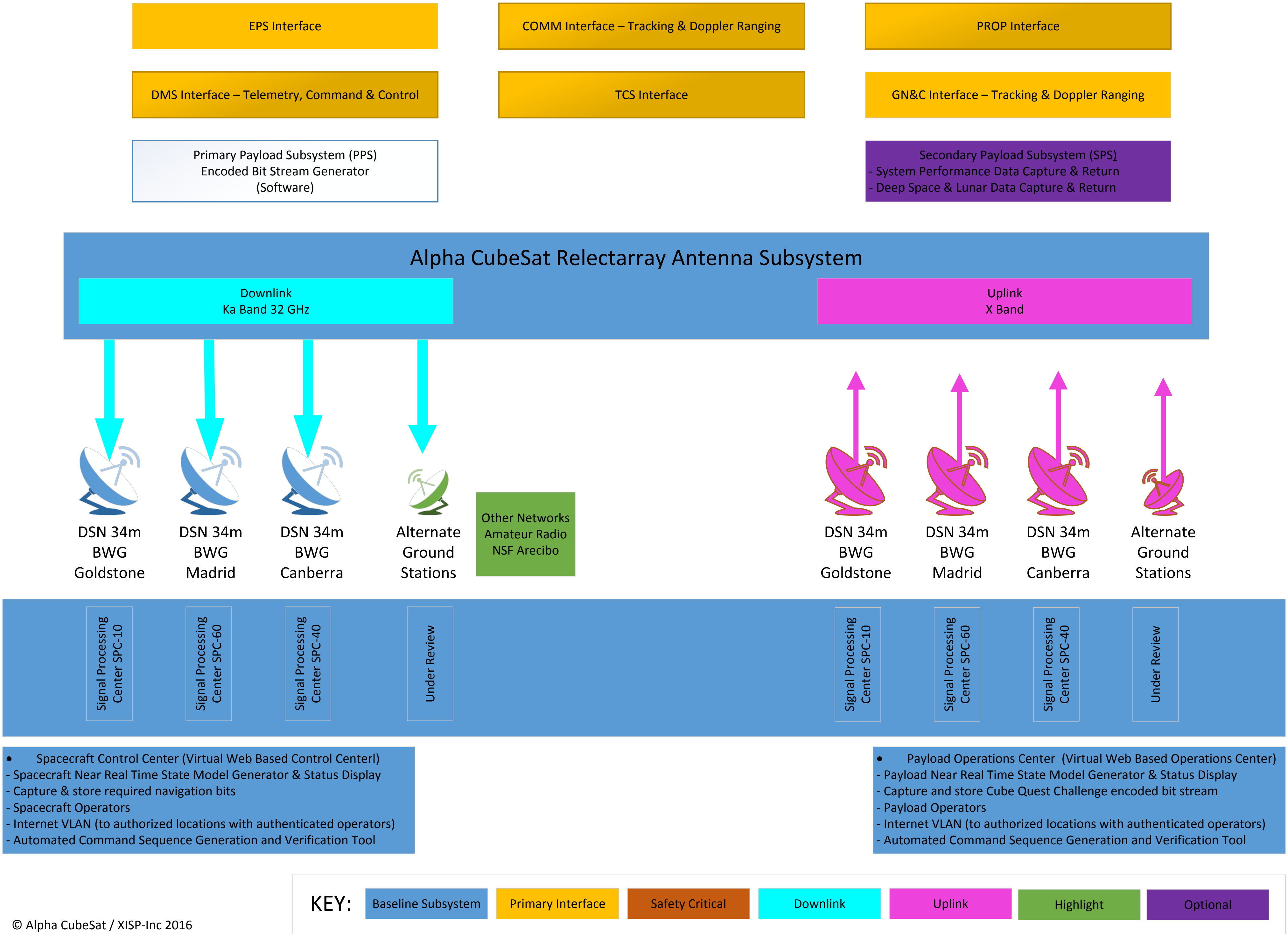
Alpha CubeSat Propulsion System (PROP)



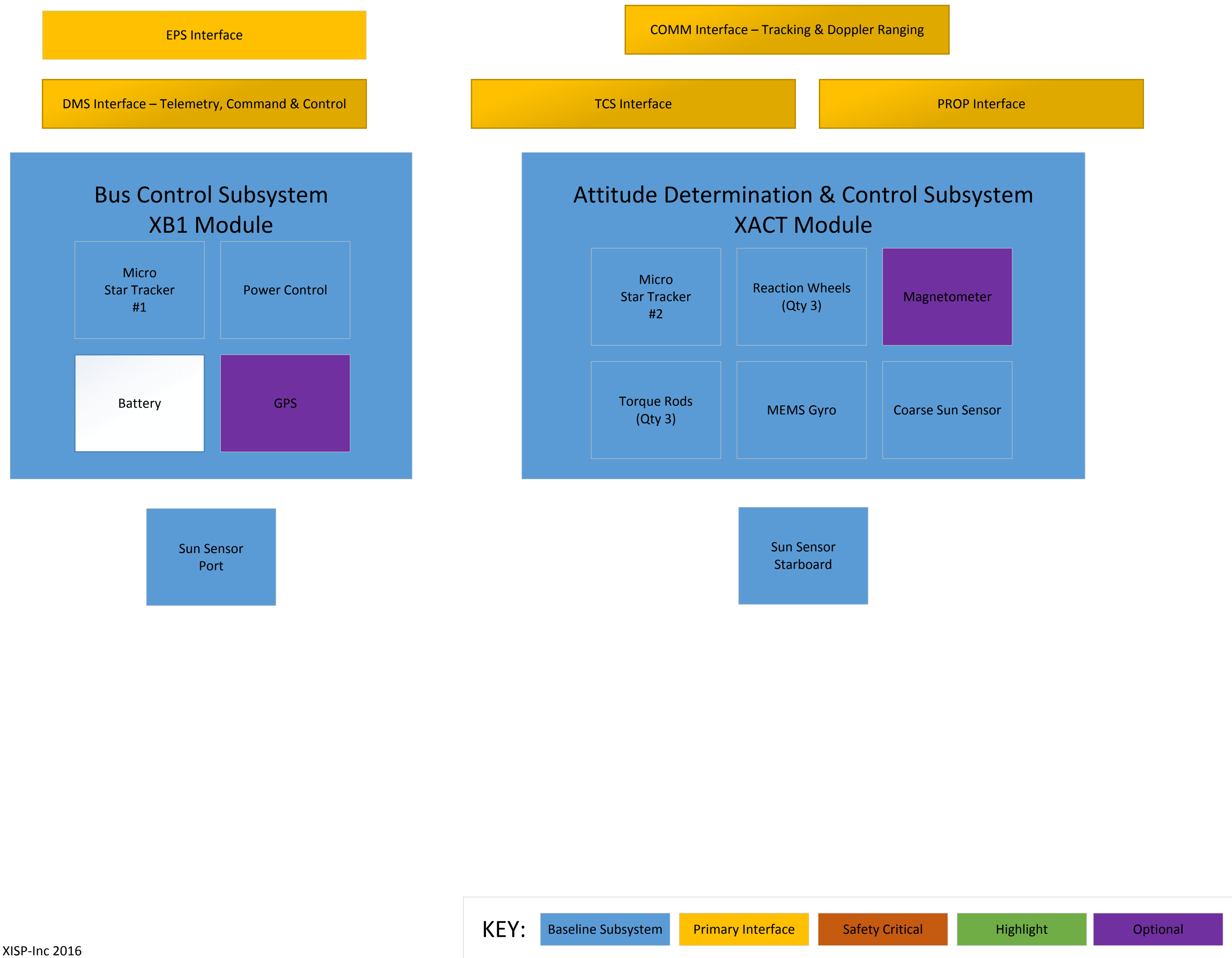
Alpha CubeSat Communications System (COMM)



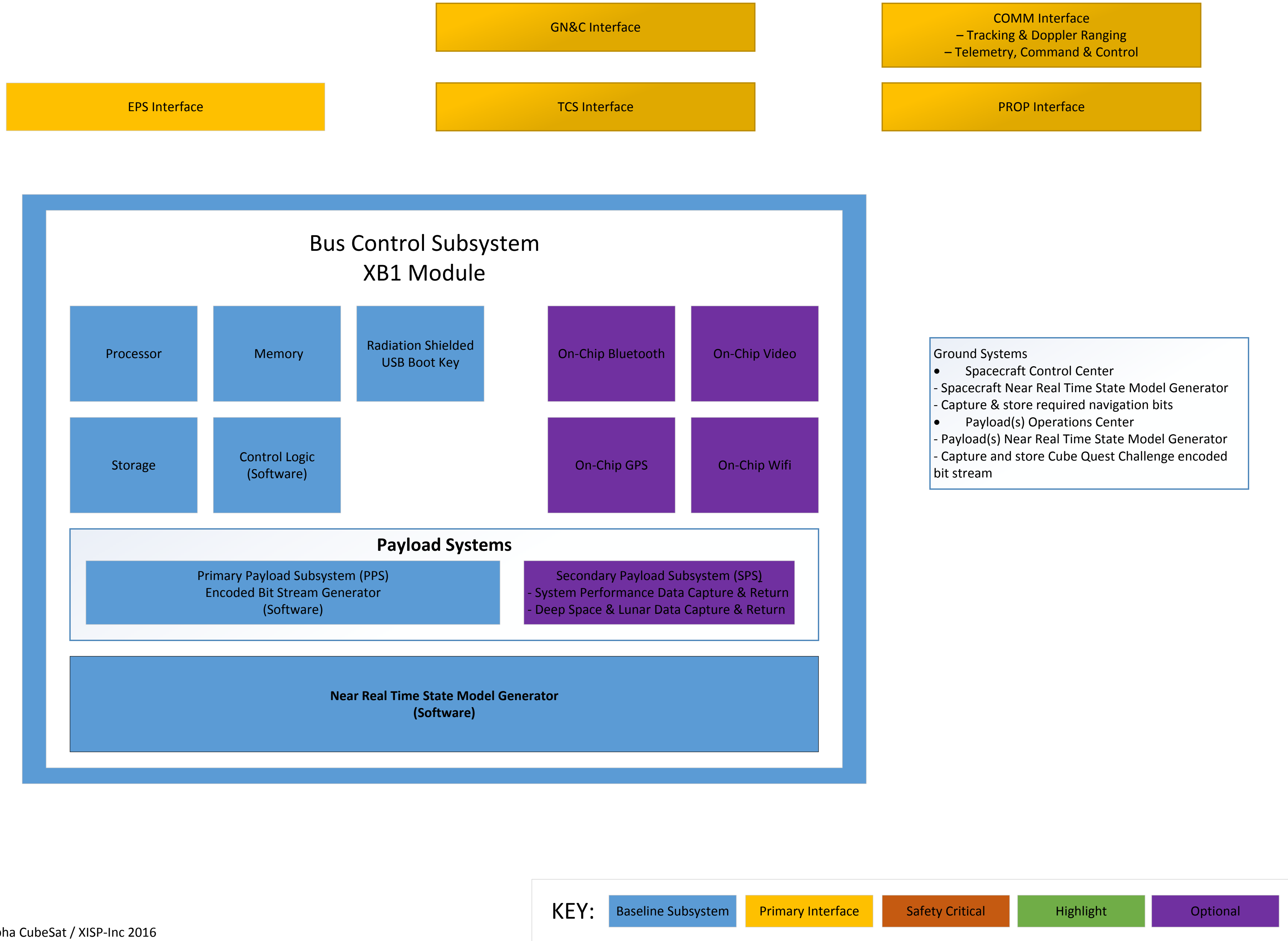
Alpha CubeSat Ground Systems



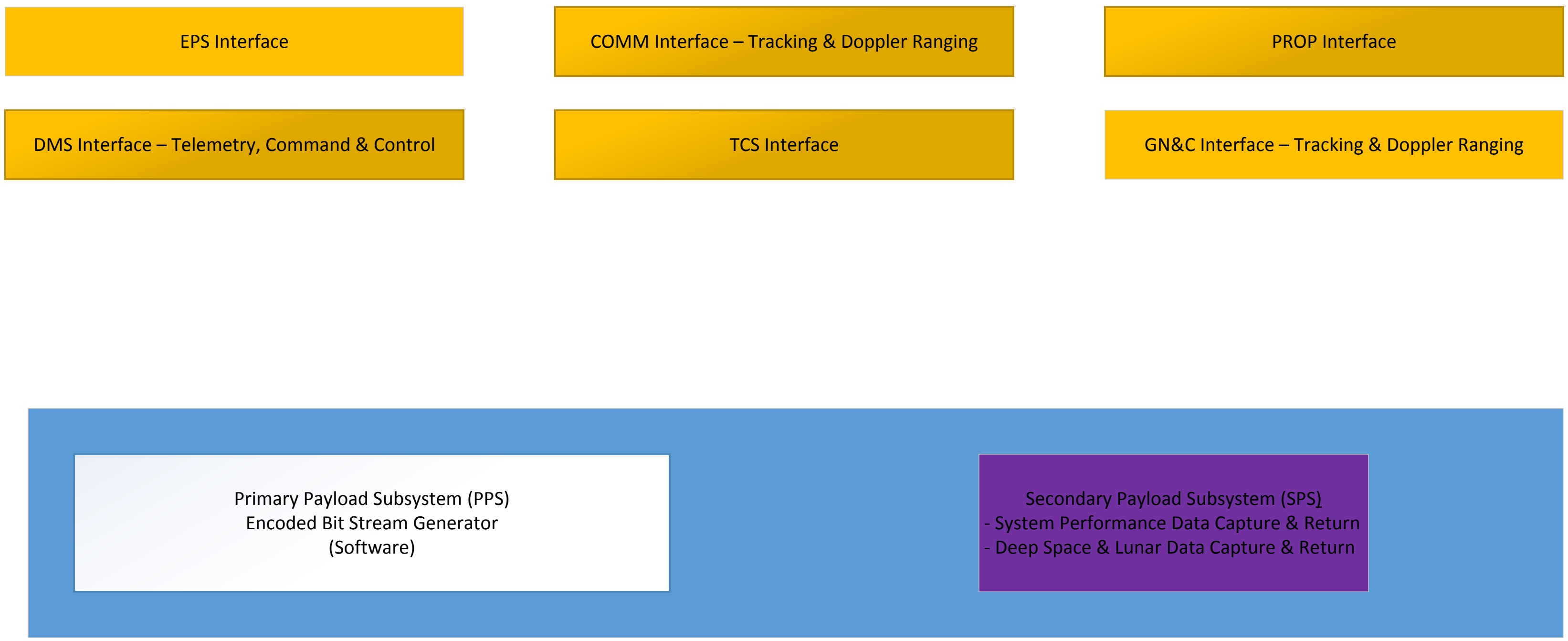
Alpha CubeSat Guidance, Navigation, & Control System (GN&C)

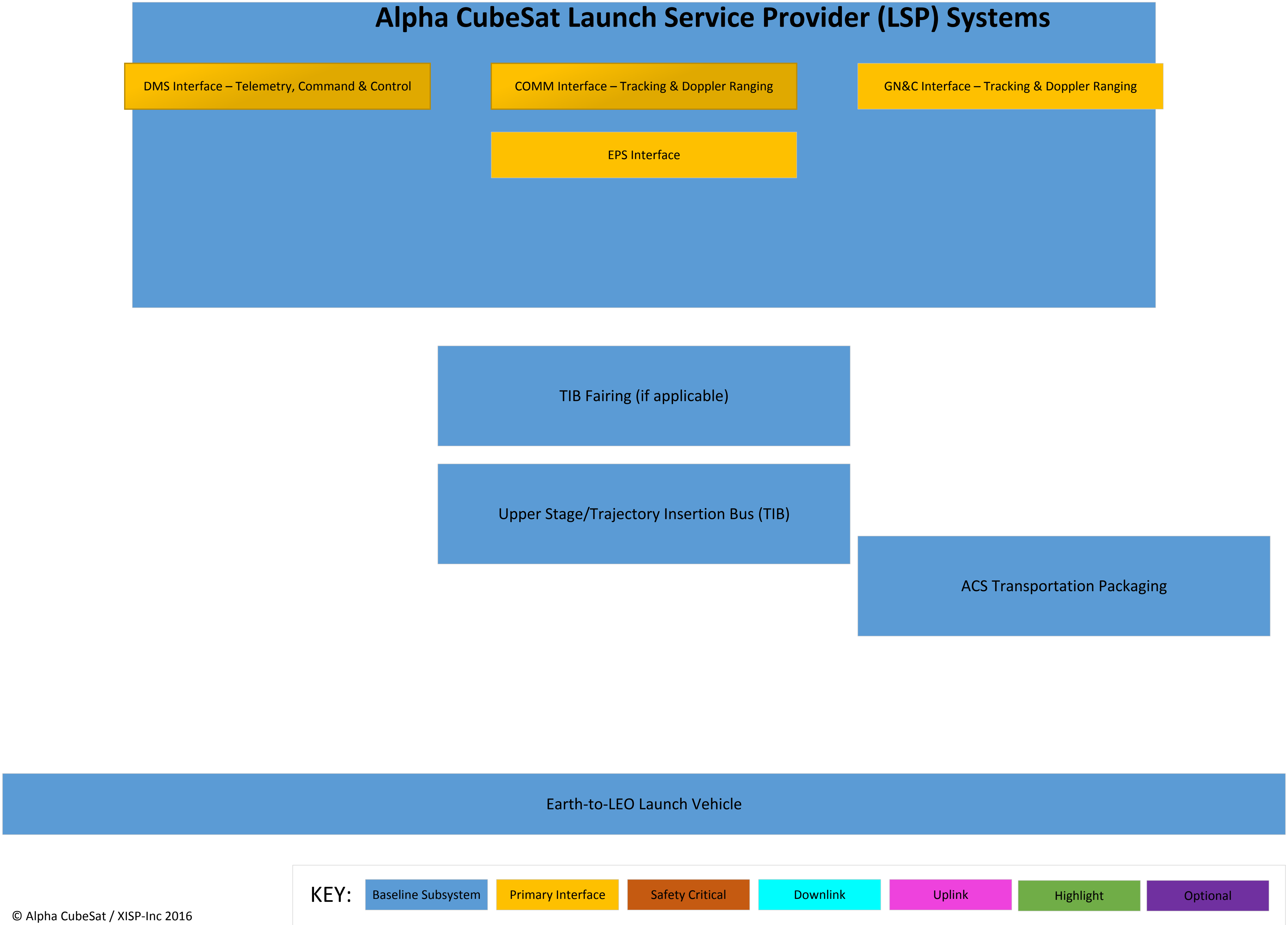


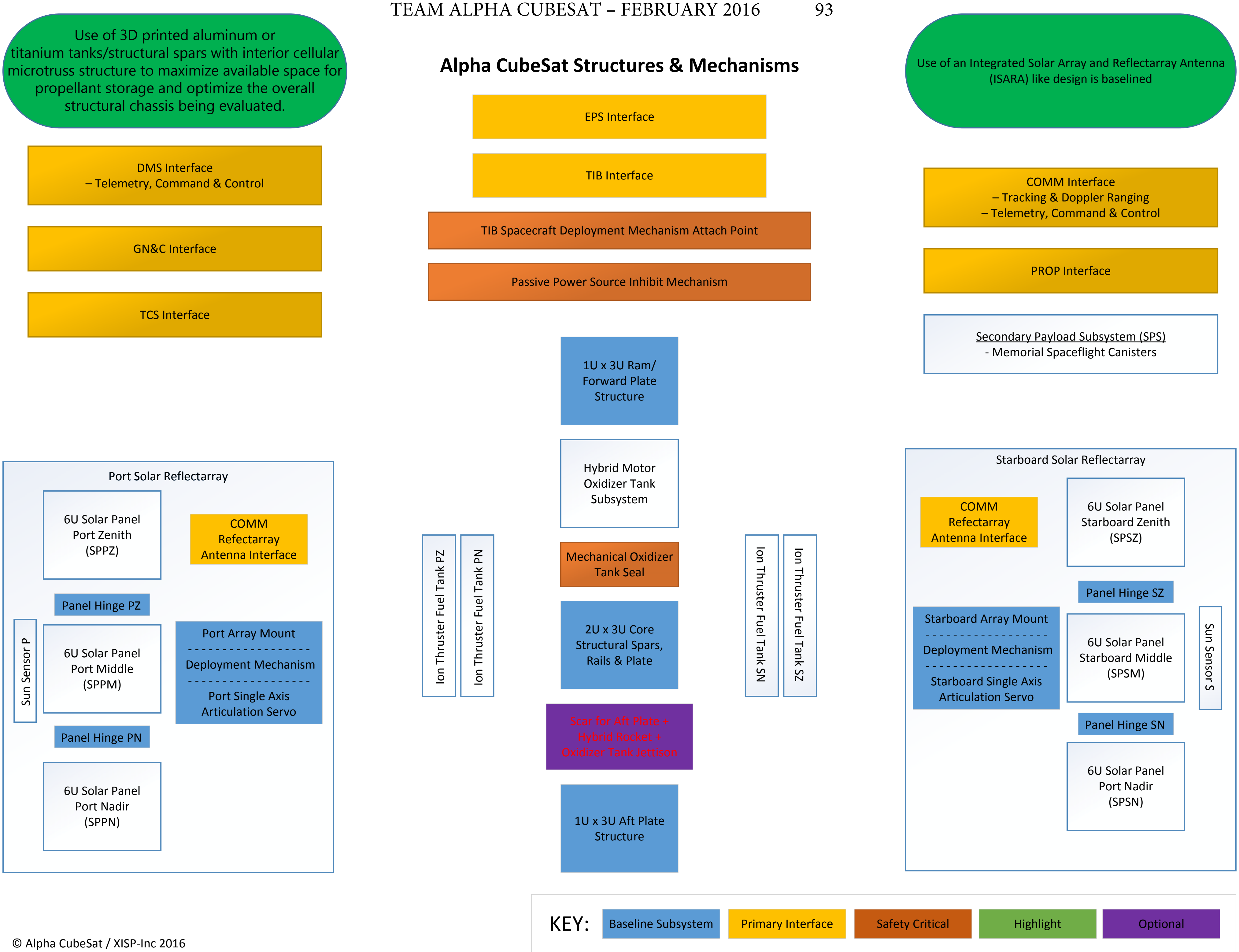
Alpha CubeSat Data Management System



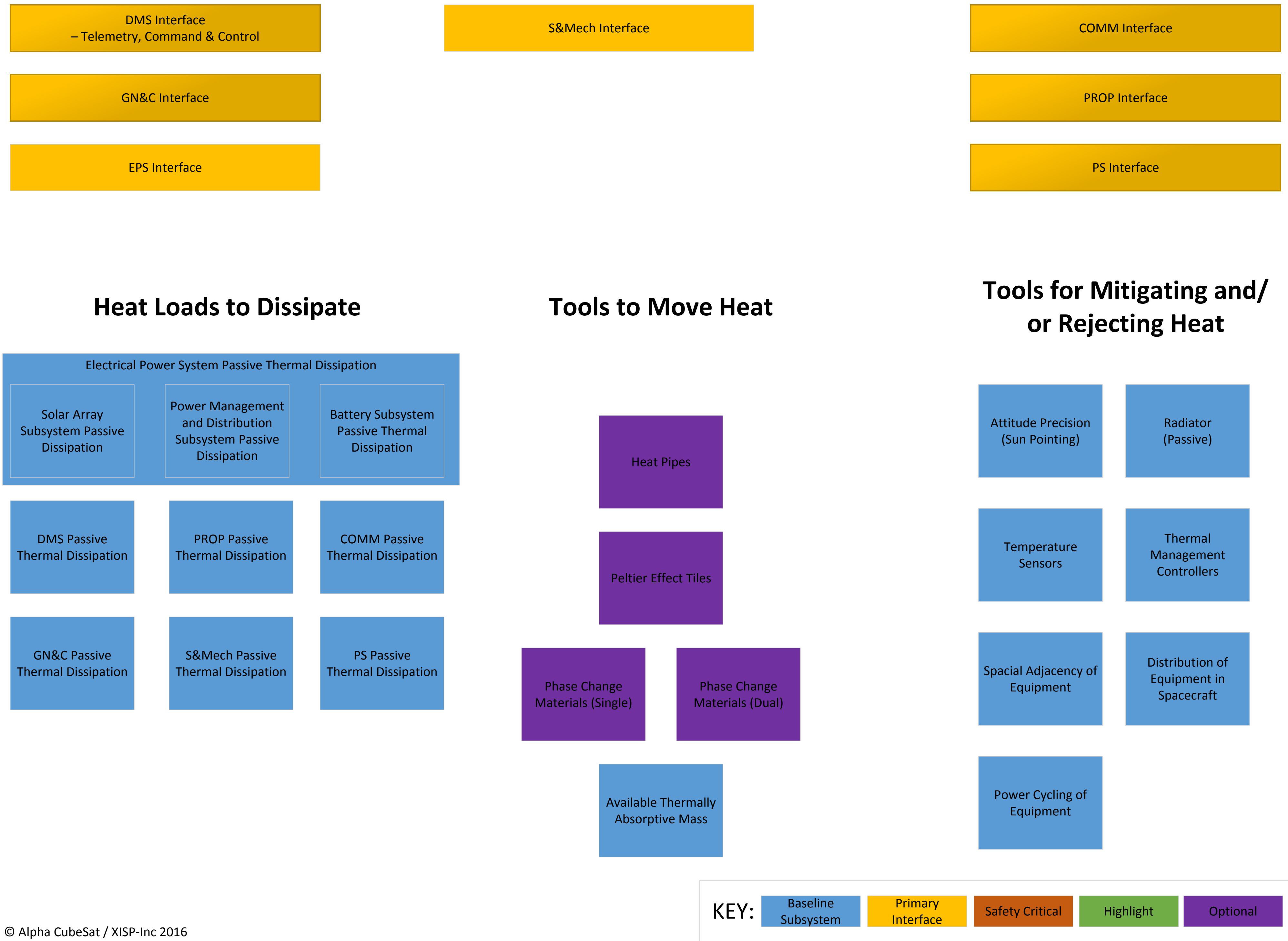
Alpha CubeSat Payload System



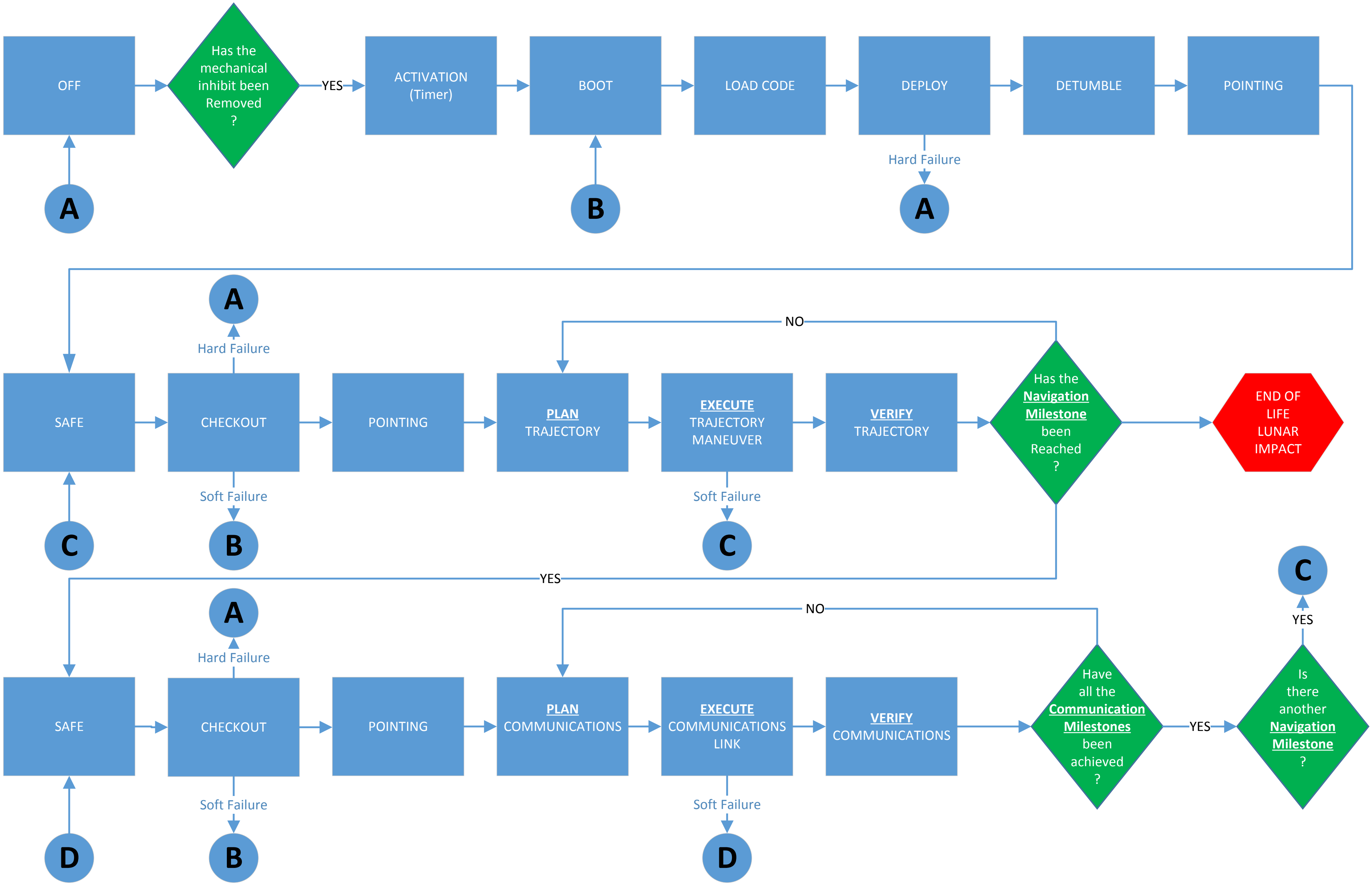

Alpha CubeSat Launch Service Provider (LSP) Systems
DMS Interface – Telemetry, Command & Control
COMM Interface – Tracking & Doppler Ranging
GN&C Interface – Tracking & Doppler Ranging
EPS Interface
TIB Fairing (if applicable)
Upper Stage/Trajectory Insertion Bus (TIB)
ACS Transportation Packaging
Earth-to-LEO Launch Vehicle
KEY:
Baseline Subsystem
Primary Interface
Safety Critical
Downlink
Uplink
Highlight
Optional


Alpha CubeSat Structures & Mechanisms
Use of 3D printed aluminum or titanium tanks/structural spars with interior cellular microtruss structure to maximize available space for propellant storage and optimize the overall structural chassis being evaluated.
DMS Interface
– Telemetry, Command & Control
GN&C Interface
TCS Interface
EPS Interface
TIB Interface
TIB Spacecraft Deployment Mechanism Attach Point
Passive Power Source Inhibit Mechanism
Use of an Integrated Solar Array and Reflectarray Antenna (ISARA) like design is baselined
COMM Interface
– Tracking & Doppler Ranging
– Telemetry, Command & Control
PROP Interface
Secondary Payload Subsystem (SPS)
- Memorial Spaceflight Canisters
1U x 3U Ram/ Forward Plate Structure
Hybrid Motor Oxidizer Tank Subsystem
Mechanical Oxidizer Tank Seal
2U x 3U Core Structural Spars, Rails & Plate
Scar for Aft Plate + Hybrid Rocket + Oxidizer Tank Jettison
1U x 3U Aft Plate Structure
Ion Thruster Fuel Tank PZ
Ion Thruster Fuel Tank PN
Ion Thruster Fuel Tank SN
Ion Thruster Fuel Tank SZ
Port Solar Reflectarray
6U Solar Panel Port Zenith (SPPZ)
COMM Refectarray Antenna Interface
Panel Hinge PZ
Sun Sensor P
6U Solar Panel Port Middle (SPPM)
Port Array Mount
Deployment Mechanism
Port Single Axis Articulation Servo
Panel Hinge PN
6U Solar Panel Port Nadir (SPPN)
Starboard Solar Reflectarray
COMM Refectarray Antenna Interface
6U Solar Panel Starboard Zenith (SPSZ)
Panel Hinge SZ
Starboard Array Mount
Deployment Mechanism
Starboard Single Axis Articulation Servo
6U Solar Panel Starboard Middle (SPSM)
Sun Sensor S
Panel Hinge SN
6U Solar Panel Port Nadir (SPSN)
KEY:
Baseline Subsystem
Primary Interface
Safety Critical
Highlight
Optional

Alpha CubeSat Thermal Control System (TCS)



Alpha CubeSat Mode / State Transitions



Alpha CubeSat Mode / State Transitions

TO FROM	OFF	ACTIVATION	BOOT	LOAD CODE	DEPLOY	DETUMBLE	POINTING	SAFE	CHECKOUT	PLAN TRAJECTORY	EXECUTE TRAJECTORY MANEUVER	VERIFY TRAJECTORY	PLAN COMMUNICATIONS	EXECUTE COMMUNICATIONS LINK	VERIFY COMMUNICATIONS
OFF	-	Timer	-	-	-	-	-	-	-	-	-	-	-	-	-
ACTIVATION	-	-	Scripted	-	-	-	-	-	-	-	-	-	-	-	-
BOOT	H-Reset	-	-	Scripted	-	-	-	-	-	-	-	-	-	-	-
LOAD CODE	-	-	-	-	Scripted	-	-	-	-	-	-	-	-	-	-
DEPLOY	H-Reset	-	-	-	-	Scripted	-	-	-	-	-	-	-	-	-
DETUMBLE	-	-	-	-	-	-	Scripted	-	-	-	-	-	-	-	-
POINTING	-	-	-	-	-	-	-	Scripted	-	Scripted	-	-	Scripted	-	-
SAFE	-	-	-	-	-	-	-	-	Scripted	-	-	-	-	-	-
CHECKOUT	H-Reset	-	S-Reset	-	-	-	Scripted	-	-	-	-	-	-	-	-
PLAN TRAJECTORY	-	-	-	-	-	-	-	-	-	-	Scripted	-	-	-	-
EXECUTE TRAJECTORY MANEUVER	-	-	-	-	-	-	-	Timer	-	-	-	Scripted	-	-	-
VERIFY TRAJECTORY	-	-	-	-	-	-	-	Scripted	-	Scripted	-	-	-	-	-
PLAN COMMUNICATIONS	-	-	-	-	-	-	-	-	-	-	-	-	-	Scripted	-
EXECUTE COMMUNICATIONS LINK	-	-	-	-	-	-	-	Timer	-	-	-	Scripted	-	-	-
VERIFY COMMUNICATIONS	-	-	-	-	-	-	-	Scripted	-	-	-	-	Scripted	-	-

H-Reset = Hard Reset turns the all systems off and restarts the activation timer (i.e., cold boots the spacecraft).
S-Reset = Soft Reset restarts all systems (i.e., warm boots the spacecraft).
Scripted = Command scripts are programmed sequences of commands which can be executed by scheduled time triggers and/or sensed event triggers.
Timer = Watch dog timer which forces a defined mode/state transition if an intended event does not occur within a specified timeframe.

Alpha Cubesat Systems on During Modes / States

FROM / TO	Electrical Power System (EPS)	Communications System (COMM)	Data Management System (DMS)	Stuctures & Mechanisms	Attitude Determination & Control System (ADCS)	Guidance, Navigation & Control System (GN&C)	Propoulsion System	Thermal System	Primary Payload Encoded Bit Stream	Scar for Secondary Payload
OFF										
ACTIVATION										
BOOT										
LOAD CODE										
DEPLOY										
DETUMBLE										
POINTING										
SAFE										
CHECKOUT										
PLAN TRAJECTORY										
EXECUTE TRAJECTORY MANEUVER										
VERIFY TRAJECTORY										
PLAN COMMUNICATIONS										
EXECUTE COMMUNICATIONS LINK										
VERIFY COMMUNICATIONS										

Alpha CubeSat Systems, Subsystems & Required Services

Propulsion System (PROP)
- Hybrid Motor Oxidizer Tank Subsystem

- Custom, Aerojet MPS-120XL CubeSat High-Impulse Adaptable Modular Propulsion System (CHAMPS) a 2U x 1U hydrazine propulsion system used to scale
- IVA/EVR Insertable Nitrous Oxide Sealed Tank/Oxidizer
- Pressurization Control Valve
- Hybrid Motor Core Subsystem

- Custom, Aerojet MPS-120XL CubeSat High-Impulse Adaptable Modular Propulsion System (CHAMPS) a 2U x 1U hydrazine propulsion system used to scale
- Thrust Control Valve
- Injector
- Igniter
- Combustion Chamber Aluminized Paraffin Fuel Core
- Hybrid Engine Nozzle
- Ion Thruster Subsystem

- Solid Iodine Propellant Store (Qty=4)
- Activation Control Switch (Qty=4)
- Neutralizer Power Processing Unit Feed System (Qty=4)
- Thrust Control Switch (Qty=4)
- Busek BIT-1 Ion Thrusters (Qty=4)

Data Management System (DMS)
- Bus Control Subsystem

- Blue Canyon Technologies XB1 Module
- Processor
- Memory
- Storage
- Control Logic (Software)
- Radiation Shielded USB Boot Key
- Near Real Time State Model Generator (Software)

Ground Systems (GS)
- Spacecraft Control Center (Virtual Web Based Control Center)

- Spacecraft Near Real Time State Model Generator & Status Display
- Capture & store required navigation bits
- Spacecraft Operators
- Internet VLAN (to authorized locations with authenticated operators)
- Automated Command Sequence Generation and Verification Tool
- Payload Operations Center (Virtual Web Based Operations Center)

- Payload Near Real Time State Model Generator & Status Display
- Capture and store Cube Quest Challenge encoded bit stream
- Payload Operators
- Internet VLAN (to authorized locations with authenticated operators)
- Automated Command Sequence Generation and Verification Tool

Guidance, Navigation & Control System (GN&C)
- Blue Canyon Technologies XB1, integrated
- Bus Control Subsystem

- Blue Canyon Technologies XB1 Module
- Bus functionality for GN&C, EPS, TCS, DMS, COMM, and Solid State Relays
- Interfaces and control provided for Payloads, PROP, and EPS Solar Array Subsystem
- Micro Star Tracker
- Power Controller
- Battery
- GPS
- Attitude Determination & Control Subsystem

- Blue Canyon Technologies XACT Module
- Micro Star Tracker
- Reaction Wheels (Qty=3)
- Mangetomer
- Torque Rods (Qty=3)
- Course Sun Sensor
- MEMS Gyro

Structures & Mechanisms (S&Mech)
- Post Solar Reflectarray Panel Hinge PZ
- Post Solar Reflectarray Panel Hinge PN
- Post Solar Reflectarray Single Axis Articulation Servo
- Post Solar Reflectarray Deployment Mechanism
- Post Solar Reflectarray Mount
- TIB Spacecraft Deployment Mechanism Attach Point
- Passive Power Source Inhibit Mechanism (EPS)
- 1U x 3U Ram/Forward Plate Structure
- Mechanical Oxidizer Tank Seal
- 2U x 3U Core Structural Spars, Rails & Plate
- Scar for Partial Aft Plate + Hybrid Rocket Ejection
- Starboard Solar Reflectarray Panel Hinge SZ
- Starboard Solar Reflectarray Panel Hinge SN
- Starboard Solar Reflectarray Single Axis Articulation Servo
- Starboard Solar Reflectarray Deployment Mechanism
- Starboard Solar Reflectarray Mount

Payload Systems (PS)
- Primary Payload Subsystem (PPS)

- CubeQuest Challenge Encoded Bit Stream Generator
- Secondary Payload Subsystem (SPS)

- System Performance Data Capture & Return
- Deep Space & Lunar Data Capture & Return
- Memorial Spaceflight Canisters

Communications System (COMM)
- Tethers Unlimited SWIFT-KTX Programmable K Band Transceiver
- Avionics Subsystem
- Software Defined Radio Subsystem
- Reflectarray Antenna Subsystem

- Clyde Space 6U CubeSat SIDE Solar Panels (6) or equivalent with integrated reflectarray antenna
- Ground Stations

- NASA DSN 34m BWG Ka Band 32 GHz Downlink Standard Service Baseline
- Alternate Ground Station Ka Band 32 GHz Uplink is baselined
- NASA DSN 34m BWG S or X Band Uplink and corresponding alternate ground station services are a defined option if required

Electrical Power System (EPS)
- Battery Subsystem

- Blue Canyon technologies XB1 Module Battery
- Solar Reflectarray Subsystem

- Clydespace 6U Solar Panels (Qty=6) with Reflectarray Antenna added
- Power Management And Distribution Subsystem

- Port Power Distribution Bus linked to BCT XB1 Module
- Starboard Power Distribution Bus linked to BCT XB1 Module
- Source Manager Control Interface
- Load Manager Control Interface

Thermal Control System (TCS)
- Heat Loads to Dissipate

- EPS Passive Dissipation
- PROP Passive Dissipation
- GN&C Passive Dissipation
- COMM Passive Dissipation
- S&Mech Passive Dissipation
- DMS Passive Dissipation
- PS Passive Dissipation
- Tools to Move Heat
- Tools for Mitigating and/or Rejecting Heat

Launch Service Provider (LSP) Systems
- Earth-to-LEO Launch Vehicle
- Upper Stage/Trajectory Insertion Bus (TIB)
- TIB Fairing (if applicable)
- ACS Transportation Packaging

KEY:	Baseline Subsystem	Primary Interface	Safety Critical	Highlight	Optional

Alpha CubeSat Technology Readiness Level (TRL)*

	System/Subsystem Name	TRL@GT-2	TRL@GT-3	TRL@GT-4	Rational for Stated TRL
		PDR	CDR	FRR	
0	**Alpha CubeSat Spacecraft**	**5**	6	7	**Alpha CubeSat is a technology demonstration satellite**
1	**Communications System (COMM)**				
	Tethers Unlimited SWIFT-KTX Programmable K Band Transceiver (Avionics+ Software Defined Radio)	5	6	7	COTS K band product being upgraded by vendor to Ka
	Clyde Space, Pumpkin, or equivalent 6U CubeSat Integrated Reflectarray Antenna	7	8	9	Reflectarray antennas are now a COTS product from multiple vendors
	NASA DSN 34m BWG Ka Band 32 GHz Downlink Standard Service Baseline	9	9	9	Available DSN Standard Service
	Alternate Ground Station Ka Band 32 GHz Uplink is baselined	7	8	9	Alternate Ka Band Ground Stations are currently operational
	NASA DSN 34m BWG S or X Band Uplink	9	9	9	Available DSN Standard Service
	Corresponding alternate S or X Band Uplink ground station services (option)	9	9	9	Alternate S or X Band Ground Stations are currently operational
2	**Electrical Power System (EPS)**				
	Blue Canyon technologies XB1 Module Battery	7	8	9	COTS product
	Clyde Space 6U CubeSat SIDE Solar Panels	7	8	9	COTS product
	Power Management And Distribution BCT XB1 Module	7	8	9	COTS product
3	**Data Management System (DMS)**				
	Bus Control Subsystem - Blue Canyon Technologies XB1 Module	7	8	9	COTS product
4	**Guidance, Navigation & Control (GN&C)**				
	Blue Canyon Technologies XACT Module	7	8	9	COTS product
	Blue Canyon Technologies XB1 Module	7	8	9	COTS product
5	**Structures & Mechanisms System (S&Mech)**				
	Solar Reflectarray Panel Hinge (Qty=4)	7	7	7	COTS products are now available from multiple vendors
	Solar Reflectarray Single Axis Articulation Servo (Qty=2)	7	7	7	COTS products are now available from multiple vendors
	Solar Reflectarray Deployment Mechanism (Qty=2)	7	7	7	COTS products are now available from multiple vendors
	Solar Reflectarray Mount (Qty=2)	7	7	7	COTS products are now available from multiple vendors
	TIB Spacecraft Deployment Mechanism Attach Point	7	7	7	COTS products are now available from multiple vendors
	Passive Power Source Inhibit Mechanism (EPS)	7	7	7	COTS products are now available from multiple vendors
	1U x 3U Ram/Forward Plate Structure	7	7	7	COTS products are now available from multiple vendors
	Mechanical Oxidizer Tank Seal	7	7	7	COTS products are now available from multiple vendors
	2U x 3U Core Structural Spars, Rails & Plate	7	7	7	COTS products are now available from multiple vendors
	Scar for Partial Aft Plate + Hybrid Rocket Ejection	2	TBD	TBD	Potential option to recover margin
6	**Propulsion System (PROP)**				
	Hybrid Motor Oxidizer Tank Subsystem	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Hybrid Motor Core Subsystem	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Busek BIT-1 Ion Thrusters (Qty=4)	5	6	7	Multiple vendors have tested prototypes, integration challenge
7	**Thermal Control System (TCS)**				
	EPS Passive Dissipation	5	6	7	Multiple vendors have tested prototypes, integration challenge
	PROP Passive Dissipation	5	6	7	Multiple vendors have tested prototypes, integration challenge
	GN&C Passive Dissipation	5	6	7	Multiple vendors have tested prototypes, integration challenge
	COMM Passive Dissipation	5	6	7	Multiple vendors have tested prototypes, integration challenge
	S&Mech Passive Dissipation	5	6	7	Multiple vendors have tested prototypes, integration challenge
	DMS Passive Dissipation	5	6	7	Multiple vendors have tested prototypes, integration challenge
	PS Passive Dissipation	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Tools to Move Heat	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Tools for Mitigating and/or Rejecting Heat	5	6	7	Multiple vendors have tested prototypes, integration challenge

8	**Payload Systems (PS)**				
	CubeQuest Challenge Encoded BIT Stream Generator	5	6	7	Multiple vendors have tested prototypes, integration challenge
	System Performance Data Capture & Return	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Deep Space & Lunar Data Capture & Return	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Memorial Spaceflight Canisters	5	6	7	Multiple vendors have tested prototypes, integration challenge
9	**Ground Systems**				
	Spacecraft Control Center	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Spacecraft Near Real Time State Model Generator	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Capture & Store required navigation bits	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Payload(s) Operations Center	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Payload(S) Near Real Time State Model Generator	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Capture and Store Cube Quest Challenge Encoded BIT Stream	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Internet VLAN	5	6	7	Multiple vendors have tested prototypes, integration challenge
	Automated Command Sequence Generation and Verification Tool	5	6	7	Multiple vendors have tested prototypes, integration challenge
10	**Launch Service Provider (LSP) Systems**				
	Earth-to-LEO Launch Vehicle	9	9	9	COTS service is available from multiple vendors
	Upper Stage/Trajectory Insertion Bus (TIB)	5	6	7	COTS service is available or planned by multiple vendos
	TIB Fairing (if applicable)	5	6	7	COTS service is available or planned by multiple vendos
	ACS Transportation Packaging	9	9	9	Commerical Cargo is a COTS service
	NOTES:				
	*As defined in NASA/SP-2007-6105 Rev 1 pg 296. Include rational for stated TRL.				

SLS SECONDARY PAYLOAD USERS GUIDE QUESTIONNAIRE – GT1

The SLS Secondary Payload Users Guide Questionnaire Alpha CubeSat GT1 response is attached as a separate appendix.

SAFETY PHASE 0 PRESENTATION

The Alpha CubeSat Safety Phase 0 Presentation development status is outlined on the following table Phase 0 Safety Review Readiness Assessment.

Phase 0 Safety Review Readiness Assessment		
Descriptive Element Name	**Location of Content in Document**	**Status**
Phase 0 Cover Page	Boiler plate	Available
Agenda	This outline	Available
Spacecraft Programmatics		
§ Payload Objectives	Section I - Mission Statement	Provided
§ Payload Team Roster	Section I - Team Roster	Provided
§ Payload Concept of Operations	Section II - Concept of Operations	Provided
§ Space Operational Sequences	Section II - Concept of Operations	Provided
§ Launch Related Activities	Section II - Concept of Operations	Provided
§ Schedule	Section V - Team Alpha CubeSat Schedule	Provided
Flight System Overview		
§ CAD Model	Section V - Spacecraft Architecture	Provided
§ Spacecraft Block Diagram	Section V - Spacecraft Architecture	Provided
§ Interfaces	Section V - Spacecraft Architecture	Provided
System Designs		
Electrical Power System (EPS)	Section V - Systems Overview EPS	Provided
§ System Block Diagram	Section VIII - Engineering Workbook	Provided
§ Power Management and Distribution	Section V - Systems Overview EPS	Provided
Grounding/Bonding		Under Development
Separation Switches		Under Development
§ Solar Arrays (conformal exterior)	Section V - Systems Overview EPS	Provided
§ Batteries (conformal propulsion tank corners)	Section V - Systems Overview EPS	Provided
Battery Concepts		Under Development
Battery System Diagram		Under Development
Compliance with Proposed Battery Charging Requirements		Under Development
Communications System (COMM)	Section V - Systems Overview COMM	Provided
§ System Block Diagram	Section VIII - Engineering Workbook	Provided
§ Ka Band Radio	Section V - Systems Overview COMM	Provided
§ Antenna (TX+RX integrated w/solar arrays)	Section V - Systems Overview COMM	Provided
Data Management System (DMS)	Section V - Systems Overview DMS	Provided
§ System Block Diagram	Section VIII - Engineering Workbook	Provided
§ On Board Computer	Section V - Systems Overview DMS	Provided
Structures & Mechanisms (S&M)	Section V - Systems Overview S&M	Provided
§ System Block Diagram	Section VIII - Engineering Workbook	Provided
Attitude Determination & Control System (ADCS)	Section V - Systems Overview ADCS	Provided
§ System Block Diagram	Section VIII - Engineering Workbook	Provided
Guidance, Navigation & Control System (GN&C)	Section V - Systems Overview GN&C	Provided
§ System Block Diagram	Section VIII - Engineering Workbook	Provided
Propulsion System (PROP)	Section V - Systems Overview PROP	Provided
§ System Block Diagram	Section VIII - Engineering Workbook	Provided
§ Hybrid Trajectory Injection Motor Core	Section V - Systems Overview PROP	Provided
§ Hybrid Trajectory Injection Motor Fuel Tank	Section V - Systems Overview PROP	Provided
§ Ion Thrusters	Section V - Systems Overview PROP	Provided
§ Ion Propellant Tanks	Section V - Systems Overview PROP	Provided
Propellant Safety		Under Development
Thermal System (TCS)	Section V - Systems Overview TCS	Provided
§ System Block Diagram	Section VIII - Engineering Workbook	Provided
Primary Payload - Encoded Bit Stream	Section V - Primary Payload	Provided
§ System Block Diagram	Section VIII - Engineering Workbook	Provided
Scar for Secondary Payload (future)	Section V - Secondary Payload	Future
§ System Block Diagram	Section VIII - Engineering Workbook	Future
Preliminary Safety Assessment	Section III - Safety & Quality Assurance	Provided
§ Standard Hazards	Section III - Safety & Quality Assurance	Provided (Introduction)
§ Unique Hazards	Section III - Safety & Quality Assurance	Provided (Introduction)
§ Approach to Meeting IDRD Safety Requirements	Section III - Safety & Quality Assurance	Provided (Introduction)
§ Anticipated Hazards	Section III - Safety & Quality Assurance	Provided (Introduction)
§ Design Options to Be Assessed	Section III - Safety & Quality Assurance	Provided (Introduction)
§ Payload and SPDS Battery Charging Requirements	Section III - Safety & Quality Assurance	Provided (Introduction)

ENGINEERING WORKBOOKS, VENDOR DATA & OTHER REFERENCES

All specification sheets and referenced papers is available as compendium of source documents.

Trajectory Workbook
Launch Services Provider Workbook

Communications System (COMM)
- COMM Engineering Workbook
- COMM Vendor Data
- COMM Other References

Electrical Power System (EPS)
- EPS Engineering Workbook
- EPS Vendor Data
- EPS Other References

Data Management System (DMS)
- DMS Engineering Workbook
- DMS Vendor Data
- DMS Other References

Guidance, Navigation & Control (GN&C)
- GN&C Engineering Workbook
- GN&C Vendor Data
- GN&C Other References

Structures & Mechanisms System (S&Mech)
- S&Mech Engineering Workbook
- S&Mech Vendor Data
- S&Mech Other References

Propulsion System (PROP)
- PROP Engineering Workbook
- PROP Vendor Data
- PROP Other References

Thermal Control System (TCS)
- TCS Engineering Workbook
- TCS Vendor Data
- TCS Other References

Payload System (PPS)
- TCS Engineering Workbook
- TCS Vendor Data
- TCS Other References

Ground Systems
- GRDS Engineering Workbook
- GRDS Vendor Data
- GRDS Other References

COMM SYSTEM

Subsystem Requirements

List all subsystem requirements, duplicating the requirements in the System Design Chapter that are relevant to the communications subsystem. Show how they are derived from, and their relationships to, the system-level requirements that are listed in the System Design Chapter.

Power requirement 35W, actual calculated is 33.3.

Thermal dissipation 30W, actual calculated is 28.3

Subsystem Design

Describe and illustrate the subsystem design of the communications subsystem. Show how the subsystem design, once fully implemented, will satisfy all subsystem requirements. Include Interfaces to other subsystems, relevant COTS parts cut sheets or specifications and any other documentation necessary to fully describe the communications subsystem.

In particular, the communications subsystem design description should include:

Alpha will use a Tethers Unlimited SWIFT-KTX programmable SDR transceiver with both a KA band transmitter and an X band receiver on board. The solar panels on the craft double as the antenna arrays thanks to integrated reflectarray antennas similar to that used on ISARA. These arrays have a pencil beam pattern for Ka band, and will also include a region of small antennas for X band reception.

- Complete descriptions of the ground station(s) including locations, transmitters, receivers and antenna patterns

The use of NASA DSN resources is baselined for uplink and downlink, primarily DSN-25 (Goldstone), DSN-34 (Canberra), and DSN-54 (Madrid). The capabilities of these stations are well documented in NASA records, available here: http://deepspace.jpl.nasa.gov/dsndocs/810-005/104/104H.pdf Other ground stations may be used in a backup or contest role including the equipment of HAM radio operators.

- Planned RF frequency bands, or, for optical communications, wavelengths

Uplink (command and control) activity will occur on X band at or around 7.145 GHz. The high speed downlink for telemetry, contest data packets, and payload will occur on Ka band at or around 32 GHz

- Planned transmission powers, modulation methods and coding approaches

The uplink (command and control) activity will use standard QPSK modulation at 30-50W to the dish feed, yielding a link margin of at least 19dB. Higher power transmissions are not a problem. Command and control data security will follow standard practice.

The Ka band high speed downlink will use 16QAM modulation with Reed Solomon forward Error Correction (FEC) at 5W or less. Other power settings, modulation, and FEC methods may be tried should the link fail, as these may be implemented via software commands.

- Include supporting analysis. Analysis should include environmental conditions, margins, uncertainties, assumptions, and operating states, modes and phases.

The supporting analysis is available in the included link budget. The links close, but there may be insufficient margin to achieve a reliable link in the event the receiving station(s) are occluded with heavy cloud cover. Should such conditions occur, it may still be possible to participate in the contest by increasing the transmitter power to a full 5W (intermittently and subject to thermal management) and/or slow the data rate. All of these changes may be triggered by commands on the X band system, which has a substantial margin and is largely unaffected by weather.

Subsystem Analysis

Please refer to the included link budget. The analysis tool used is mature and well documented within the spreadsheet. TRL data is available in the included Alpha Cubesat Technology Readiness Level (TRL) document.

β TEST Version -Revised	AMSAT / IARU Annotated Link Model System	Version: 2.5.3

Organization: XISP-Inc

Project: Alpha CubeSat

Developed by: Jan A. King, W3GEY/VK4GEY With Editorial Assistance and Support from Ralph Wallio, WΦRPK; Ignacio Mas; Lou McFadin, W5DID; Jeff Capehart, W4UFL; Michelle Denise, W5NYV

Approvals:

NOTE #1:	Com. System Engineer:	Enter Name of Communications Engineer: Aaron D. Harper	X	NOTE #7: Com. Eng.
NOTE #2:	Project Manager:	Enter Name of Project Manager: Gary Barnhard		P.M.
NOTE #3:	Orbit Type:	NASA Cube Quest Challenge Deep Space Derby Altitude: 4 million KM		
NOTE #4:	Model Under Investigation:	6u Cubesat for DSD and LD competition Communications		
NOTE #5:	Model/Case No./Rev No.:	TBD/TBD/2.1		Config. Control
	Date Data Last Modified:	2016 February 05	LINK MODEL STATUS	Document Not Released
NOTE #6	Date W/S Formulas Last Modified:	2008 December 17		NOTE #8:

Eb/No Method:
S/N Method:

Eb/No Method:
S/N Method:

Introduction, Instructions for Use, References, Revisions: Alpha CubeSat 2016 February 05

Introduction:

This spreadsheet system is an attempt to provide a new kind of learning tool. It is intended, clearly, to be a working link model in order to allow satellite system designers to design and then document fully the RF radio links associated with Command (uplink) and Telemetry (downlink) equipment. It is, however, also intended to be a tutorial on the RF portion of a satellite system. The model makes liberal use of "pop-up" notes and "tools" to enhance the understanding (and hopefully the knowledge) of the Link Model Operator (that's you). After you use the model for awhile, let me know if I have been successful. - Jan A. King, W3GEY and VK4GEY; w3gey@amsat.org

Instructions for Use:

Colors: Colors are used in the link model to make it easier to find data and to protect the link model from crashing. Many of the worksheets are interconnected in that equations in one W/S refer forward or back to data located in other worksheets. Loss of this connection could be critical. Also, the cells are not yet protected (and may never be) as the system has not yet been finalized. Color can be used to provide "coded" messages to the link model operator's brain, once it has been used for awhile. This has been found by the designer to be fairly effective (at least with his brain). Color is used for both the text and the cell background. Some colors have been picked for large field areas where it is not so nice to have the Excel cell grid structure showing. Typically, light grey light green light yellow or white are used this way. These colors have been found by our staff psychologist to have a relaxing effect on the operator. Now let's look at the important uses of color:

NOTE: This is a "pop-up" note. You will see a lot of single cells throughout the model that look like this. Using your mouse, place your cursor on the cell. You don't need to click. A note will pop up. These are either local instructions on how to enter data or use data or some form of training note. You will find that some notes are somewhat larger than the screen. I've tried hard to avoid this, but I haven't been entirely successful. The problem with this is that if you scroll to see the rest of the note and if the yellow cell scrolls off of the screen then the note will close. Frustration will ensue. There are two solutions: 1) Reduce the scale of the viewing page from 100% (the ususal setting) to 75% or 85%. This should allow you to see all of the note. 2) Alternatively, using the mouse, select from the upper toolbar, "View", "Toolbars", and select the one called "Reviewing". There should now be a checkmark to the left of that option. Now, you should find a new toolbar up above the text area of Excel. The far left icon will say "new comment" if you are making a new one. But, if you move the curser over the far left icon you will notice the pop-up prompt now says "edit cell." Now, move the curser over the "NOTE:" cell and left click then left click on the same far left icon. This will allow you to edit the cell BUT it will also FREEZE the cell in the ON condition. Now, you can move the note around by using the slide bars on the side and bottom of the screen to see all of the note. It's probably a good idea not to modify the note. You can close the note by just moving the cursor to an empty cell somewhere and left clicking. It is suggested that you try this process now with the test note above at Cell [D23]. It's been set up to frustrate you in just such a way as the real notes might do later on.

X.XX This is a data entry cell. The link model operator is expected to enter data. The blue background means it is a critcal data entry cell. It is anticipated that your system's selected value is quite likely to be different than the default value used in the cell when you received this link model.

X.XX This is also a data entry cell. This type of cell may not need to be changed as the value you are likely to use may be the same as the default value.

X.XX This is a cell containing an equation or a constant that should not be changed. **The operator should not modify these cells.** A majority of the link model contains this type of cell.

X.XX or X.XX These are cells containing important but, intermediate results. Two colors were used to provide a slight gradation of importance. The orange color is considered to be a result having slightly more significance than the lighter yellow cell.

X.XX This is a key "bottom line" result. It is a primary output of a particular W/S.

X or **X** or **X** A few cells use conditional fomatting which allow the cell colors to change depending on the outcome of the preceeding calculations. Typically a RED box means the result was not successful in achieving the desired performance. A GREEN box means the result did meet or exceed the desired performance. A YELLOW box means the result achieved the performance threshold but, is considered marginal.

Sub-Title Box A pink box like this is simply a sub-title for a sub-worksheet.

X.XX An olive green box is a location where data has been transferred to this worksheet from another and may be transferred to yet another. No action need be taken here. It's purpose is only so that the operator is aware that the data is being transferred from and to other locations.

Frequency Sometimes an olive green cell will be used to re-emphasize a frequency selection as in the "System Performance Summary" W/S.

Non-Coherent FSK Sometimes a tan color cell is used to denote a selected system condition that is non-numeric.

Gains and Losses: A positive gain or directivity is always experssed as a positive number. Sometimes the value may be seen to have a + in front of it. Gains can also be negative (remember, the gain of an antenna is expressed as 10log(P/P $_{isotropic}$). So, if the gain in a particular direction, is below that of an isotropic radiator, then the gain will be expressed as a negative number in dBi.

Losses in link budgets are commonly found as either positive or negative. A loss, by it's nature, is a negative quantity but, some believe that if the loss is clearly referred to as such in the budget *parameter* column, it can have a positive sign. That is the case in this link budget. All losses are shown as being a positive value. The argument is symantic. The question could be asked, "Is a positive loss a negative? And is a negative loss, positive? The important thing for the link model operator to know when using this modeling system is that the losses are show as positive values **BUT**, in the equations that sum the gains and losses to yield the result, the gains are *added* and the losses are *subtracted*. For example, see the equation in Cell [B11] of the "Uplink" W/S.

Speciality W/S vs. Tools: The first 13 W/Ss are all interconnected, in that they all have equations that make use of data contained in one or more of the other W/Ss. These worksheets, taken together, constitute the link model. The next 5 W/Ss are supplementary to the model and are considered to be *tools*. The important distinction is, that tools *never* produce results that are automatically linked into the model itself, whereas within the first 13 W/Ss there is lots of interlinking going on. The primary process is one where data calculated or selected in one of the S*peciality* W/Ss (e.g., "Receivers") becomes just one entry in either the Uplink or the Downlink budget. The usefulness of a tool is to be able to explore a specific tradeoff without having to worry about that data winding up in the formal

Uplink or Downlink pages.

There is one additional and imporatant comment about tools. Within the Speciality W/Ss, there are some embedded tools. The best example of this is in the "Receivers" W/S. Contained in separate sub-tables is a *Noise Figure/Noise Temperature Calculator (Tool)* and a *Ground Station, Antenna or Sky Noise Temperature Calculation Tool* .

Proceeding Through the Model: Starting with the "Title Page" W/S, proceed through each Speciality W/S, adding data, in sequence. Then select the next tab at the bottom of the W/S. The "Uplink", "Downlink" and "System Performance Summary" W/Ss contain the final results of the model. The Tools W/Ss are located beyond the "System Perfomance Summary" W/S and may be explored and used as they may be helpful to you. Any comments you may have on this model will be greatfully received by me. **Thanks!** Jan, VK4GEY.

References: The following references were used to prepare this link model:

1 A.R.R.L., *The ARRL Antenna Handbook,* American Radio Relay League, 1974, pp. 153-155.

2 Deloraine, E.M., Westman, H.P., Edie, L.C. *Reference Data for Radio Engineers, 3rd Edition* , Federal Telephone & Radio Corp., 1949, pp. 362-396.

3 Feher, Dr. Kamilo, *Digital Communications, Satellite/Earth Station Engineering* , Prentice-Hall Books, 1983, Chapter 4.

4 Ippolito, L.J.Jr., *Radiowave Propagation in Satellite Communications,* Van Norstrand Reinhold Co., 1986, Chapters 3 and 7.

5 Jordan, E.C. (Edit.), *Reference Data for Engineers: Radio, Electronics, Computer, and Communications, 7th Edition* , Howard W. Sams & Co., 1985, pp. 29-26 - 29-37 and pp. 30-03 - 30-11.

6 Martin, W.L., *AMMOS and DSN Support of Earth Orbiting and Deep Space Missions,* Jet Propulsion Laboratory, TMOD Directorate, 1996, p.44-46.

7 Morgan, W.L. and Gordon, G.D., *Principles of Communicaitons Satellites* , John Wiley & Sons, Inc., 1993, Chapter 2 and pp.140-143.

8 Van Wie, D.G. and Roark, R.C., *A New Alert Protocol,* Blue Water Design, LLC*, 2003,* pp. 18-23.

9 Jackson, R.B., *The Canted Turnstile as an Omnidirectional Spacecraft Antenna,* X-712-67-441, NASA/Goddard Space Flight Center, 1967, Entire Document.

Revisions: The following formal revisons have been made to this Link Model System:

Version:	*Date:*	*Adjustments and/or Modifications Made:*
2.0	1/30/2005	NEW; β-Test Version
2.1	2/7/2005	Revised All "Pop-up" Notes; Corrected some cell colors to improve consistancy; Added reference 9; Corrected cells A19 & D19 in "Uplink" W/S.
2.1.1	2/12/2005	Revised Equation at Cell [B15] of "Uplink Budget" W/S. Index function should use column H values not column C values.
2.1.2	2/21/2005	Modified Data for Monopole Antenna Pattern in Monopole Table in "Antenna Patterns" W/S. Added 3 dB to all Values (0° to 90°)
2.1.3	2/26/2005	Modified "Receviers" W/S. Added loss value for cable D. Modified 2nd Stage to "Communications Receiver" at Ground Station.
2.1.4	2/27/2005	Added Tubo Code Option to "Modulation-Demodulation Method" W/S.
2.2	2/27/2005	Added EZNEC+ and Chart Wizzard Antenna Plots to "Antenna Pattern" W/S.
2.2.1	5/15/2005	Edited Notes in I.I.R.R W/S.
2.2.2	6/23/2005	Edited More Notes Throughout Link Model.
2.3	7/16/2005	Revised Antenna Gain and Antenna Pointing Losses W/Ss to Include a High Gain (Parabolic Reflector) S/C Antenna Option & Iso. Radiator Option.
2.3.1	9/28/2005	Modified Notes at Cells [P135] and [V52] of "Receivers" W/S. Added **To** reference temperature "readout" at Cell [U56] of "Receivers" W/S.
2.3.2	10/4/2005	Modified Equation at Q62 of "Antenna Gain" W/S. Equation was "=21/(F55/1000)*H62" and now is "=21/((F55/1000)*H62)." TNX Ignacio Mas.
2.4	10/22/2006	Changed "Downlink" to "Uplink" at D22 in "Antenna Gain" W/S. Changed hard coded cells in "Ant. Pointing Losses" W/S for referenced cells. Fixed errors in downlink portion of worksheet. There were several incorrect references. Added NOTEs at Line 57 of the "Uplink" W/S and Line 56 of the "Downlink W/S" to remind user about S/N when using coding. TNX Jeff Capehart W4DFU.
2.5	Not Released	Added HEO, GEO and Deep Space Orbit Capability. Link Model Operator selects options. Separted Orbit and Frequency into two separate pages.
2.5.1	3/6/2008	Repaird Bugs in User #2, Delta Longitude, Range, Azimuth and Earth Central Angle; Thank to Michelle Denise, W5NYV
2.5.2	3/18/2008	Repaired Import of Frequency Values to "Transmitters" and "Receivers" Worksheets; Thanks to Michelle Denise, W5NYV
2.5.3	12/17/2008	In "Atmos. & Ionos. Losses" W/S; temporarily made Atmos. Loss dependent on Manually Set Elevation Angle. This needs more work.

System Orbit Characteristics:

Alpha CubeSat **2016 February 05** **Version: 2.5.3**

Orbit Option to be Used in Link Model (LEO, HEO, GEO, Deep Space)

Select Orbit Option: 4 **Deep Space**

Slant Range: 3,999,000 km Used in Path Loss Calculation

Option No.:	Orbit Type:	Slant Range:	
1	LEO	2783.9	km
2	HEO	41126.8	km
3	GEO	38097.0	km
4	Deep Space	3.999E+06	km

Blue = User Data Entry Values
Black = Computed Values (**No Data Entry**)
Red = Key Results
Blue =Critical User Data Entry Values
NOTE: Cells Not Yet Protected

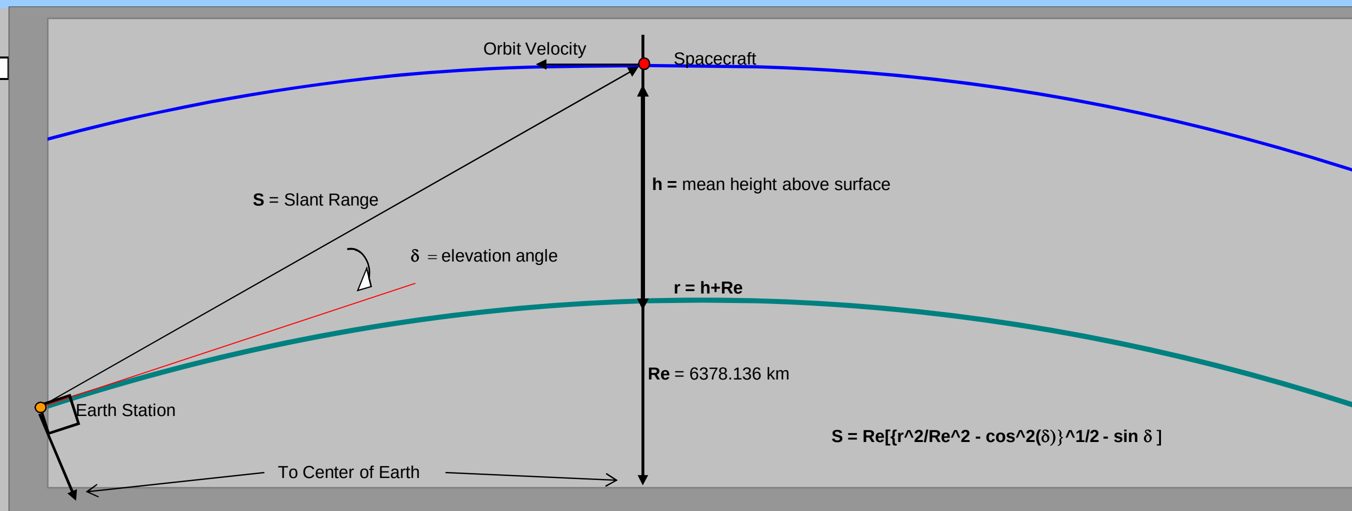
Element Reference Epoch: 2005, 87.50000

LEO Orbit - Option #1 NOTE:

Low Earth Orbit Properties

Slant Range to Spacecraft vs. Elevation Angle

Parameter:	*Value:*	*Unit:*
Earth Radius:	6,378.14	km
Height of Apogee (ha):	805.0	km
Height of Perigee (hp):	795.0	km
Semi-Major Axis (a):	7,178.1	km
Eccentricity (e):	0.000697	
Inclination (I):	98.61	°
Argument of Perigee (ω):	180.0	°
R.A.A.N. (Ω):	123.70	°
Mean Anomaly (M):	0.00	°
Period:	100.874	minutes
dω/dt:	-2.9241	deg./day
dΩ/dt:	0.9860	deg./day
dM/dt:	Not Implemented	deg./day
Mean Orbit Altitude:	800.00	km
Mean Orbit Radius:	7,178.14	km
Sun Synchronous Inclination:	98.61	°
Elevation Angle (δ):	5.0	°
Slant Range (**S**):	2,783.9	km.



LEO Orbit Geometry

High Earth Orbit (HEO) - Option #2 NOTE:

HEO Orbit Properties

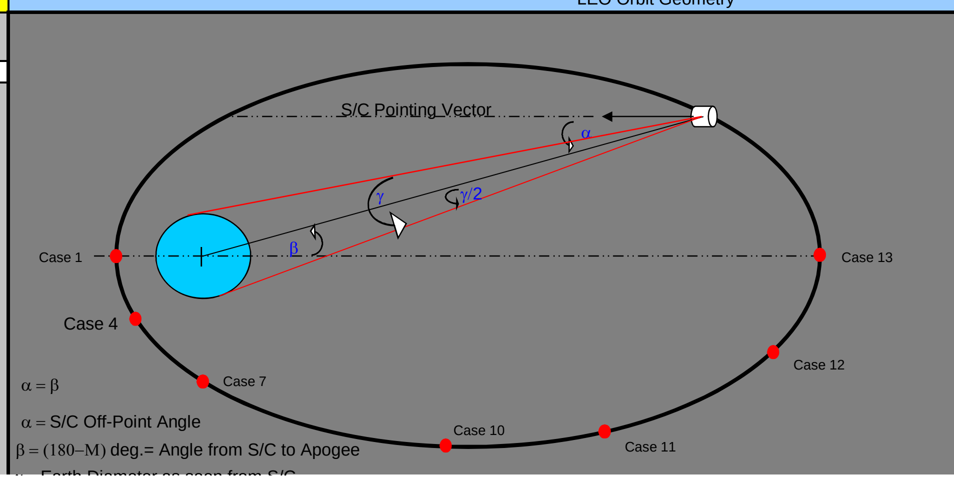
S/C Spinning and NADIR-Pointing at Apogee

Parameter:	*Value:*	*Unit:*
Earth Radius:	6,378.14	km
Height of Apogee:	35,786	km
Height of Perigee:	500	km
Semi-Major Axis (a):	24,521.14	km
Eccentricity (e):	0.719502	
Inclination (I):	7.00	degrees
Argument of Perigee (ω):	180.0	degrees
R.A.A.N. (Ω):	0.00	degrees
Mean Anomaly (M):	180.00	degrees
Period:	636.90	minutes
dω/dt:	0.7542	deg./day
dΩ/dt:	-0.3814	deg./day

1) To Change Orbit Keplarians Modify **ONLY** Blue Values Above.

2) Choose Case No. and Enter Here. Proceed to "Uplink & Downlink

HEO Orbit



Choices" Below.

Geometry

γ = Earth Diameter as seen from S/C
$\alpha + \gamma/2$ = Worst Case Squint Angle

CASE NO. SELECTED: 13 | 35,786.0 km Altitude | Elevation Angle: 5.0 ° | Slant Range (S): 41,126.8 km

CASE:		R(km):	M(deg.):	altitude (km):	S/C off-point angle:	S/C rcvr. ant. temp.(K)
1	perigee	6878.1	0	500.0	180.0 deg.	35
2		6977.6	15	599.5	165.0 deg.	35
3		7286.6	30	908.5	150.0 deg.	35
4		7838.8	45	1,460.7	135.0 deg.	35
5		8697.9	60	2,319.8	120.0 deg.	35
6		9970.3	75	3,592.2	105.0 deg.	35
7		11827.0	90	5,448.8	90.0 deg.	35
8		14533.4	105	8,155.2	75.0 deg.	35
9		18472.4	120	12,094.3	60.0 deg.	35
10		24076.0	135	17,697.8	45.0 deg.	40
11		31380.2	150	25,002.0	30.0 deg.	50
12		38775.1	165	32,396.9	15.0 deg.	90
13	apogee	42164.1	180	35,786.0	0.0 deg.	170
14		41756.6	175	35,378.4	5.0 deg.	160

User Defined Case:

SOME KEY ORBIT & LINK PARAMETERS	
EARTH ANGULAR DIAMETER (γ):	17.4 °
S/C POINTING VECTOR (α):	10.0 °
WORST CASE SQUINT ANGLE:	18.7 °
RX ANTENNA POINTING LOSS:	0.00 dB
TX ANTENNA POINTING LOSS:	0.00 dB
GROUND RCVR Eb/No:	10.1 dB
S/C RCVR Eb/No	36.5 dB

3) If CASE No. 14 is Selected, Choose Mean Anomaly Value and S/C Rcvr Antenna Temp. and Enter Here.

Geostationary Earth Orbit (GEO) - Option #3

NOTE:

Path Length to User Terminal from Spacecraft

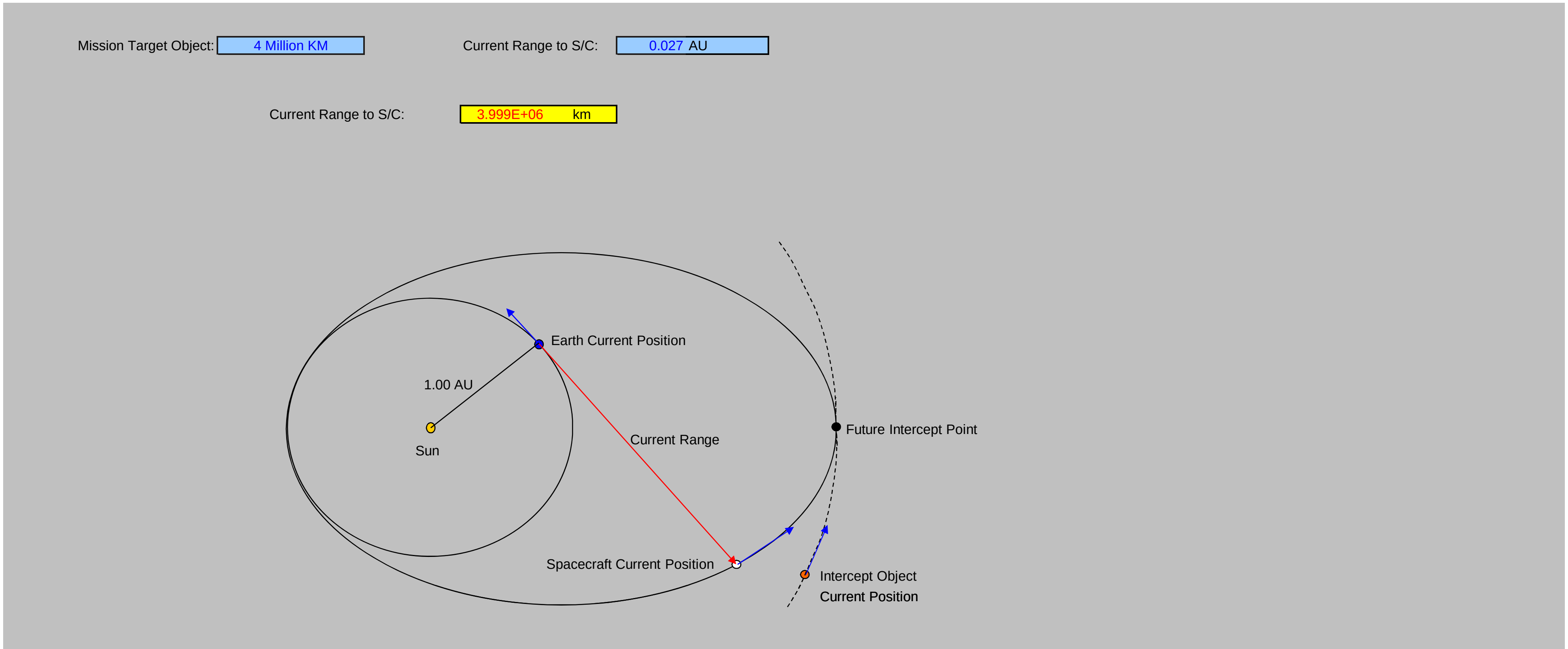
Parameter:	*Value:*	*Unit:*	*Comment(s):*
Geostationary Altitude:	35,786.019	km	Height Above Geoid
Equatorial Radius of Earth (Re):	6,378.137	km	
Geostationary semi major axis	42,164.156	km	Accurate to 1/10 meter
Typical Path Length:	37,410.000	km	User at typical Longitude difference from satellite and at mean latitude.
Shortest Path Length:	35,786.019	km	User at same longitude as satellite and at the equator
Longest Path Length:	41,678.957	km	User at max. longitude difference from satellite and at max. latitude (0.0° User Elevation Angle).

UPLINK: → S/C → **DOWNLINK:**

	User #1:		Comment		User #2: NOTE:	
User Latitude:	40.000	°	+ = North Latitude; - = South Latitude	User Latitude:	40.000	°
User Longitude:	-105.000	°	+ = East Longitude; - = West Longitude	User Longitude:	-116.000	°
Spacecraft Slot (Longitude):	-132.000	°	Enter Slot Postion in Degrees East Longitude (NOTE: Δ Longitude ≤ 81.3°) [- = W. Long.; + = E. Long.]	S/C Slot Longitude:	-132.000	°
Slant Range to User:	38097.0	km	The distance from the GEO satellite to the user. This Value used in Link Budget Path Loss Calculation.	Slant Range to User:	37715.2	km
User Elevation Angle:	36.015	°	This is the Elevation Angle to the GEO spacecraft from the User (latitude and longitude) site.	User Elevation Angle:	40.853	°
User Azimuth Angle:	218.403	°	This is the azimuth angle to the GEO spacecraft from the User (latitude and longitude) site.	User Azimuth Angle:	204.041	°
Earth Central Angle:	46.957	°	The angle measured from Earth center between the sub-satellite point and the ground station location.	Earth Central Angle:	42.577	°

Deep Space Mission - Option #4: Range Expressed in Astronomical Units (AU)

NOTE:



Heliocentric Transfer Mission (Example)

UPLINK & DOWNLINK Frequency Choices:

Orbit Type Selected: **Deep Space**

Path Loss = 22.0 + 20 log (S/λ)

Slant Range for Orbit Option Selected: **3,999,000 km**

NOTE:

	Option:	Frequency:	Wavelength (λ):	Path Loss:	
Uplink:	#1:	145.800 MHz	2.056 meters	207.8	dB
	#2:	437.500 MHz	0.685 meters	217.3	dB
	#3:	1269.900 MHz	0.236 meters	226.6	dB
Operator Selected Option: →	#4:	7145.000 MHz	0.042 meters	241.6	dB
Downlink:	#1:	145.800 MHz	2.056 meters	207.8	dB
	#2:	437.450 MHz	0.685 meters	217.3	dB
	#3:	2405.000 MHz	0.125 meters	232.1	dB
Operator Selected Option: →	#4:	32000.000 MHz	0.009 meters	254.6	dB

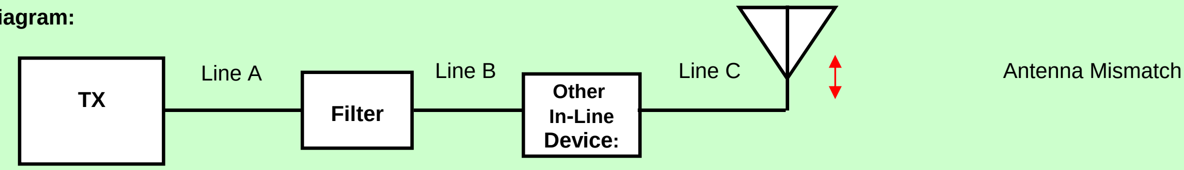
Uplink Frequency Choice:	4	**7145.000 MHz**
Path Loss for Orbit Selected:		**241.6 dB**
Downlink Frequency Choice:	4	**32000.000 MHz**
Path Loss for Orbit Selected:		**254.6 dB**

System Transmitters & Line Losses: Alpha CubeSat 2016 February 05

Uplink Transmitter System (At Ground Station):

NOTE:

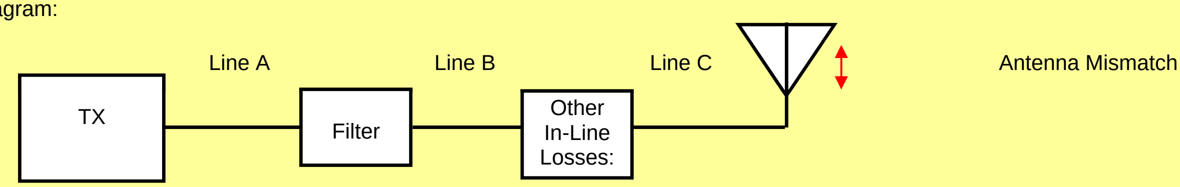
Block Diagram:



Transmitter Power:	50.00	Watts =	17.0	dBW =	46.99	dBm
Cable or Waveguide ("Line") Losses:						
Line A Length:					1.0	meters
Line B Length:					0.3	meters
Line C Length:					25.0	meters
Total Line Length (Line A+B+C):					26.3	meters
Cable/W. Guide Type:					Belden 9913	cable
Cable/W.Guide Loss/meter:	0.05 dB	At (freq.)	7145	MHz =	1.315	dB
Other Components in Line:						
No. of In-Line Connectors:	6	Connectors X 0.05 dB/Con. =			0.3	dB
Filter Insertion Losses:					1.0	dB
Other In-Line Losses:	Device:	Directional Coupler			0.5	dB
Antenna Mismatch Losses:		(See "VSWR Loss Tool" W/S)			0.5	dB
Total Line Losses:					3.62	dB
Total Power Delivered to Antenna:					13.37	dBW

Downlink Transmitter System (At Spacecraft):

Block Diagram:



Transmitter Power:	3.0	Watts =	4.8	dBW =	34.77	dBm
Cable or Waveguide Loss:						
Line A Length:					0	meters
Line B Length:					0	meters
Line C Length:					0.3	meters
Total Line Length (Lines A+B+C):					0.3	meters
Cable/Guide Type:					MicroCoax MCJ185A	cable
Cable/Guide Loss/meter:	0.49 dB	At (freq.)	32000	MHz =	0.147	dB
Other Components in Line:						
No. of In-Line Connectors:	2	Connectors X 0.05 dB =			0.1	dB
Filter Insertion Losses:					0.0	dB
Other In-Line Losses:	Device:	N/A			0	dB

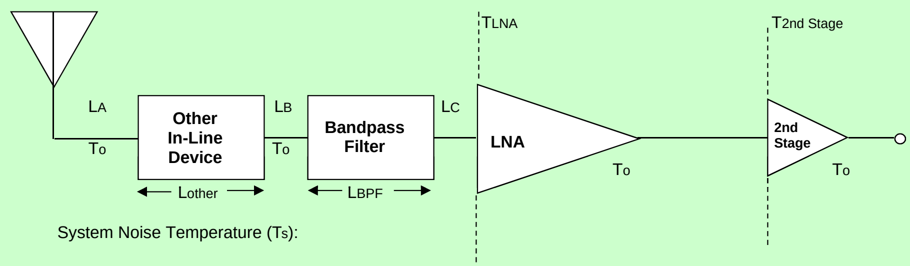
Antenna Mismatch Losses:	(See "VSWR Loss Tool" W/S)	0.240	dB
Total Line Losses:		**0.49**	dB
Total RF Power Delivered to Antenna:		4.28	dBW

System Receivers and Line Losses: Alpha CubeSat 2016 February 05

Uplink Receiver System (At Spacecraft):

NOTE:

Block Diagram:



System Noise Temperature (T_s):

$T_s = (\alpha)T_a + (1-\alpha)T_o + T_{LNA} + T_{2ndStage}/G_{LNA}$

Where:

$T_a \equiv$ Antenna Temperature or Sky Temperature (°K)

$T_o \equiv$ System Line Temperature (Physical Temperature) (°K) $\equiv$ System Reference Temperature

$T_{LNA} \equiv$ Noise Temperature of the Low Noise Amplifier (°K)

$T_{2nd\ Stage} \equiv$ Noise Temperature of Next Stage Amplifier or Mixer (°K)

$G_{LNA} \equiv$ The gain of the LNA in linear (non-dB) units NOTE:

$\alpha \equiv$ Feed Line Coefficient = 10^ -((L_A/10)+(L_B/10)+(L_C/10)+(L_{BPF}/10)+(L_{other}/10))

Where:

L_A, L_B, $L_C \equiv$ All Cable or Waveguide Losses (expressed in dB)

$L_{BPF} \equiv$ Insertion Loss of any bandpass fiter used in front of LNA (expressed in dB)

$L_{other} \equiv$ Insertion Loss of any other In-Line device in front of LNA (expressed in dB)

Cable or Waveguide "Line" Losses:

Line A Length:		0.3	meters
Line B Length:		0	meters
Line C Length:		0	meters
Cable/Guide Type:		MicroCoax MCJ185A cable	
Cable/Guide Loss/meter:	0.1 dB at frequency	7145.0	MHz
Line A Loss:	L_A =	0.03	dB
Line B Loss:	L_B =	0	dB
Line C Loss:	L_C =	0	dB
Bandpass Filter Insertion Loss:	L_{BPF} =	0.0	dB
Insertion Loss of Other In-Line Devices:	L_{other} =	0	dB
No. of In-Line Connectors: 2	X .05 dB/Con.=	0.1	dB
Other In-Line Device Type:		none	
Total In-Line Losses from Antenna to LNA:		0.13	dB
Transmission Line Coefficient:	α =	0.9705	
Antenna or "Sky" Temperature: NOTE:	T_a =	290	K
Spacecraft Temperature:	T_o =	3.9	K
LNA Temperature:	T_{LNA} =	0.8	K
LNA Gain: 40.0 dB	G_{LNA} =	10000.0	

Noise Temperature/Noise Figure Calculator (Tool): NOTE:

$NF_{dB} = 10\ LOG_{10}[1+(T/T_o)]$

or

$T = T_o[10^\wedge(NF_{dB}/10)-1]$

T_o = 3.9 K

NF_{dB} = 0.8 dB T = 0.8 K

OR

T = 200.0 K NF_{dB} = 17.18 dB

2nd Stage Temperature: $T_{2ndStage}$ = 0 K

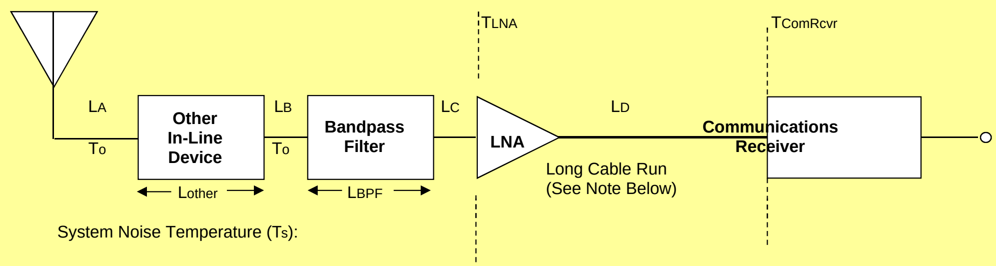
System Noise Temperature: T_s = 282.4 K

Enter Data Here:

Result is Here

Downlink Receiver System (At Ground Station):

Block Diagram:



System Noise Temperature (T_s):

$T_s = (\alpha)T_a + (1-\alpha)T_o + T_{LNA} + T_{ComRcvr}/(G_{LNA}/L_D)$

Where:

$T_a \equiv$ Antenna Temperature or Sky Temperature (°K)

$T_o \equiv$ System Line Temperature (Physical Temperature) (°K)

$T_{LNA} \equiv$ Noise Temperature of the Low Noise Amplifier (°K)

$T_{ComRcvr} \equiv$ Noise Temperature of Communications Receiver Front End (°K)

$G_{LNA} \equiv$ The gain of the LNA in linear (non-dB) units NOTE:

$\alpha \equiv$ Feed Line Coefficient = 10^ -((L_A/10)+(L_B/10)+(L_C/10)+(L_{BPF}/10)+(L_{other}/10))

Where:

L_A, L_B, $L_C \equiv$ All Cable or Waveguide Losses (expressed in dB)

$L_{BPF} \equiv$ Insertion Loss of any bandpass fiter used in front of LNA (expressed in dB)

$L_{other} \equiv$ Insertion Loss of any other In-Line device in front of LNA (expressed in dB)

Cable or Waveguide "Line" Losses: NOTE:

Line A Length:		2.5	meters
Line B Length:		0.3	meters
Line C Length:		0.3	meters
Cable/Guide Type:		Belden 9913 cable	
Cable/Guide Loss/meter:	0.092 dB (at freq.)	32000.0	MHz
Line A Loss:	L_A =	0.23	dB
Line B Loss:	L_B =	0.0276	dB
Line C Loss:	L_C =	0.0276	dB
Bandpass Filter Insertion Loss:	L_{BPF} =	0.0	dB
Insertion Loss of Other In-Line Devices:	L_{other} =	0.0	dB
No. of In-Line Connectors:	4 X 0.05 dB/con.=	0.2	dB
Other In-Line Device Type:		none	
Total In-Line Losses from Antenna to LNA:		0.49	dB
Transmission Line Coefficient:	α =	0.8943	

Ground Station, Antenna or Sky Noise Temperature Calculation Tool:

Galactic Noise Component:

Receiver Frequency: 32000 MHz

Coldest Galactic Noise Temp.: 3 K

Antenna or "Sky" Temperature: NOTE:	T_a =	126	K
Ground Station Feedline Temperature:	T_o =	280	K
LNA Temperature:	T_{LNA} =	31.41	K
LNA Gain: 60.0 dB	G_{LNA} =	1000000.0	
Cable/Waveguide D Length: NOTE:		25.0	meters
Cable/Waveguide D Type:		Belden 9913 cable	
Cable/Waveguide D Loss/meter:		0.092	dB/m
Cable/Waveguide D Loss:		2.3	dB
Communications Receiver Front End Temperature	$T_{ComRcvr}$ =	1000	K
System Noise Temperature:	T_s =	174	K

Warmest Galactic Noise Temp:	3	K
Terrestrial Noise Component:		
Receiver Bandwidth:	80.0	KHz
NOTE: Estimated or Measured Noise Level:	-132.4	dBm
Noise Source Effective Temperature:	52	K
Minimum Sky Noise Temp:	55	K
Maximum Sky Noise Temp:	55	K

System Antenna Gains (Directivities): Alpha CubeSat 2016 February 05

Uplink Antenna System:

NOTE:

Ground Station:

Uplink Frequency: 7145 MHz — Wavelength: 0.0420 meters

Operator Selects Option 1 to 4 Here

OPTION: 3 Parabolic Reflector — Polarization: RHCP

1	Yagi	Boom Length (λ):	3.2	Optimum Elements (n):	12	per Plane (in V and in H)		Maximum Gain:	16.3	dBiC	Beamwidth:	30.6 °	Antenna Length:	0.134 meters
2	Helix	Turns (n):	10.5	Turn Spacing (λ):	0.25	Circumference (λ):	1.0	Gain:	16.0	dBiC	Beamwidth:	32.2 °	Antenna Length:	0.110 meters
3	Parabolic Reflector	DSS-25		Diameter: 34.0 m		Aperture Efficiency:	56%	Gain:	94.7	dBiC	Beamwidth:	0.1 °		
4	User Defined	KLM (22x22 Element)	Yagi	(Example)				Gain:	18.5	dBiC	Beamwidth:	24.0 °	Antenna Length:	X.XX meters

Spacecraft:

Uplink Frequency: 7145 MHz — Wavelength: 0.0420 meters

Operator Selects Option 1 to 7 Here

OPTION: 7 Other (User Defined) — Polarization: RHCP

1	Monopole		Gain:	2.15	dBiL	Beamwidth:	156.2 °	No Radiation in Back Hemisphere **AND** Null on Axis ("Tip Null")
2	Dipole		Gain:	2.15	dBiL	Beamwidth:	156.2 °	Null On Axis; Both Poles
3	Canted Turnstyle		Gain:	2.0	dBiC (typical)	Beamwidth:	180 °	Circular Pol. On Axis; RHCP one pole, LHCP Opposite Pole, Linear in Equatorial Plane
4	Quadrifilar Helix	Loop (λ): 1/2	Gain:	4.0	dBiC	Beamwidth:	150 °	No Radiation in Back Hemisphere; Excellent Axial Ratio Performance Off-Axis
5	Patch		Gain:	6.0	dBi (L or C)	Beamwidth:	90 °	Low Radiation in Back Hemisphere; High On-Axis Gain; Can be Maded Linear or Circularly Polarized
6	Parabolic Reflector		Gain:	49.5	dBi (L or C)	Beamwidth:	0.5 °	To Be Used if a High Gain Antenna is Required on S/C. Dish Diameter: 5.4 m; Dish Aperture Efficiency: 55%
7	Other (User Defined)	reflectenna array	Gain:	8.0	dBi	Beamwidth:	4.8 °	Gain, Beamwidth and Roll-Off Equation To Be Provided By Link Model Operator

UPLINK ↑ DOWNLINK ↓

Downlink Antenna System:

Spacecraft:

Downlink Frequency: 32000 MHz — Wavelength: 0.0094 meters

Operator Selects Option 1 to 5 Here

7 Other (User Defined) — Polarization: RHCP

OPTION:									
1	Monopole		Gain:	2.15	dBiL	Beamwidth:	156.2	°	No Radiation in Back Hemisphere & Null on Axis ("Tip Null")
2	Dipole		Gain:	2.15	dBiL	Beamwidth:	156.2	°	Null On Axis; Both Poles
3	Canted Turnstyle		Gain:	2.0	dBiC (typical)	Beamwidth:	180	°	Circular Pol. On Axis; RHCP one pole, LHCP Opposite Pole, Linear in Equatorial Plane
4	Quadrifilar Helix	Loop (λ): 1/2	Gain:	4.0	dBiC	Beamwidth:	150	°	No Radiation in Back Hemisphere
5	Other (User Defined)	Patch (Example)	Gain:	6.0	dBi (L or C)	Beamwidth:	90	°	No Radiation in Back Hemisphere
6	Parabolic Reflector		Gain:	53.9	dBi (L or C)	Beamwidth:	0.3	°	To Be Used if a High Gain Antenna is Required on S/C. Dish Diameter: 2.0 m; Dish Aperture Efficiency: 55%
7	Other (User Defined)	reflectenna array	Gain:	32.0	dBi	Beamwidth:	3	°	Gain, Beamwidth and Roll-Off Equation To Be Provided By Link Model Operator

Ground Station:

Downlink Frequency: 32000 MHz   Wavelength: 0.0094 meters

Operator Selects Option 1 to 4 Here

3 Parabolic Reflector   Polarization: RHCP

OPTION:											
1	Yagi	Boom Length (λ): 2.0	Optimum Elements (n): 8	per Plane (in V and in H)	Maximum Gain:	14.1	dBiC	Beamwidth:	39.7 °	Antenna Length:	0.019 meters
2	Helix	Turns (n): 10.5	Turn Spacing (λ): 0.25	Circumference (λ): 1.0	Gain:	16.0	dBiC	Beamwidth:	32.2 °	Antenna Length:	0.025 meters
3	Parabolic Reflector	DSS-25	Diameter: 34.0 m	Aperture Efficiency: 56%	Gain:	79.0	dBiC	Beamwidth:	0.0 °		
4	User Defined	KLM (22x22 Element) Yagi	(Example)		Gain:	18.5	dBiC	Beamwidth:	24.0 °	Antenna Length:	X.XX meters

Look-Up Table

Optimum Yagi Antenna Performance:

Boom Length (λ):	Optimum No. Elements (n):	Maximum Gain (dBi):
0.35	3	9.65
0.55	4	10.86
0.80	5	11.85
1.15	6	12.45
1.45	7	13.35
1.80	8	14.05
2.10	9	14.40

2.45	10	15.25
2.80	11	15.95
3.15	12	16.30
3.55	13	16.95
4.00	14	17.45
4.40	15	18.15
4.75	16	18.65
5.20	17	19.35
5.55	18	19.85
6.00	19	20.25
6.50	20	20.75
7.00	21	21.35
7.50	22	21.65

Data Taken from *ARRL Antenna Book*

System Antenna Pointing Losses: Alpha CubeSat 2016 February 05

NOTE:

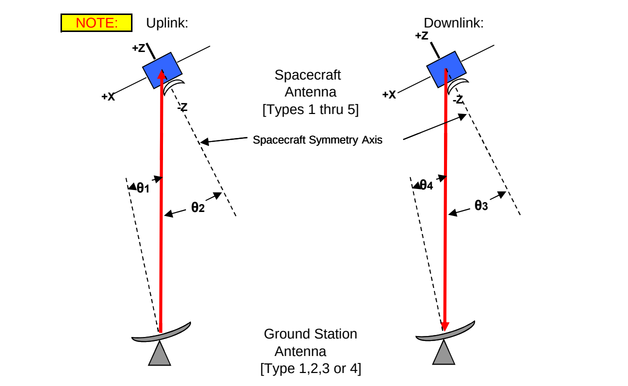


Figure 1 Figure 2

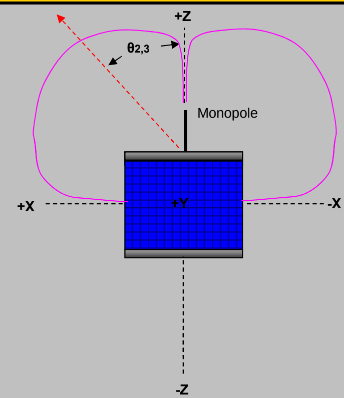


Figure 3

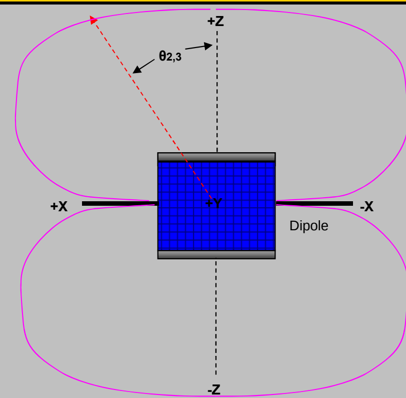


Figure 4

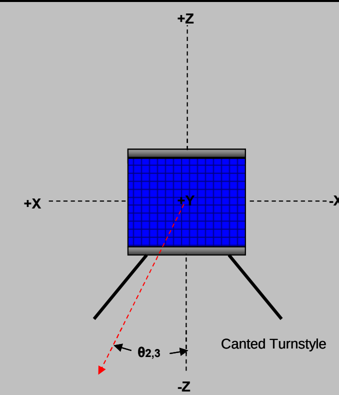


Figure 5

Antenna Loss Determination:

(See Also Figure 8)

Uplink Antenna System:

NOTE:

Ground Station:

Uplink Frequency: 7145 MHz Wavelength: 0.0420 meters

This Option was Selected on the Previous Page

3 Parabolic Reflector Polarization: RHCP

1	Yagi	Maximum Gain:	16.3	dBiC	Beamwidth:	30.6 °
2	Helix	Gain:	16.0	dBiC	Beamwidth:	32.2 °
3	Parabolic Reflector	Gain:	94.7	dBiC	Beamwidth:	0.1 °
4	User Defined	Gain:	18.5	dBiC	Beamwidth:	24.0 °

Esimated Pointing Error (θ1): 0.0015 ° **Approx. Antenna Pointing Loss:** 0.0 dB

NOTE:

2.77

Antenna Roll-Off Calculation Formulas

Spacecraft:

Uplink Frequency: 7145 MHz Wavelength: 0.0420 meters

This Option was Selected on the Previous Page

7 Other (User Defined) Polarization: RHCP

						Antenna Coordinate System:		Antenna Roll-Off Calculation Formulas
1	Monopole	Gain:	2.15 dBiL	Beamwidth:	156.2 °	See Figures 1and 3	monopole	12.8 dB
2	Dipole	Gain:	2.15 dBiL	Beamwidth:	156.2 °	See Figures 1and 4	dipole	0.0 dB
3	Canted Turnstyle	Gain:	2.0 dBiC	Beamwidth:	180 °	See Figures 1, 5 & 8	canted turnstyle	0.0 dB

4	Quadrifilar Helix	Loop (λ): 1/2	Gain:	4.0 dBiC	Beamwidth:	150 °	See Figures 1and 6	quadrifilar helix	0.0 dB
5	Patch		Gain:	6.0 dBi (C or L)	Beamwidth:	90 °	See Figures 1and 7	patch antenna	0.0 dB
6	Parabolic Reflector	[For S/C Hi Gain Option]	Gain:	49.5 dBi (C or L)	Beamwidth:	0.5 °	Dish Boresight Aligned with +Z Axis	parabolic reflector	32.7 dB
7	Other (User Defined)	Reflectenna	Gain:	8.0 dBi	Beamwidth:	4.8 °	Link Model Operator to Provide	user defined	0.0 dB

Angle between S/C antenna symmetry axis and vector from S/C to gnd. station (θ2): 3 °   **Approx. Antenna Pointing Loss:** 0.0 dB

879.25 — Intermediate Calculation - Please Ignore This Value.

Link Model operator enter equation for functional behaviorof user defined antenna here.

UPLINK ↑   **DOWNLINK ↓**

Downlink Antenna System:

Spacecraft:

Downlink Frequency: 32000 MHz   Wavelength: 0.0094 meters

This Option was Selected on the Previous Page

7 Other (User Defined)   Polarization: RHCP

Antenna Coordinate System:

↑ Antenna Roll-Off Calculation Formulas ↓

1	Monopole		Gain:	2.15 dBiL	Beamwidth:	156.2 °	See Figures 2 and 3	monopole	12.8 dB
2	Dipole		Gain:	2.15 dBiL	Beamwidth:	156.2 °	See Figures 2 and 4	dipole	0.0 dB
3	Canted Turnstyle		Gain:	2.0 dBiC	Beamwidth:	180 °	See Figures 2, 5 & 8	canted turnstyle	0.0 dB
4	Quadrifilar Helix	Loop (λ): 1/2	Gain:	4.0 dBiC	Beamwidth:	150 °	See Figures 2 and 6	quadrifilar helix	0.0 dB
5	Other (User Defined)	Patch (Example)	Gain:	6.0 dBi	Beamwidth:	90 °	See Figures 2 and 7	Patch (Example)	0.0 dB
6	Parabolic Reflector	[For S/C Hi Gain Option]	Gain:	53.9 dBi (C or L)	Beamwidth:	0.3 °	Dish Boresight Aligned with +Z Axis	parabolic reflector	38.1 dB
7	Other (User Defined)	Reflectenna	Gain:	32.0 dBi	Beamwidth:	3 °	Link Model Operator to Provide	user defined	0.0 dB

1458.47 — Intermediate Calculation - Please Ignore This Value.

Enter functional behavior of user defined antenna here.

Angle between S/C antenna symmetry axis and vector from S/C to gnd. station (θ3): 3 °   **Approx. Antenna Pointing Loss:** 0.0 dB

Ground Station:

Downlink Frequency: 32000 MHz   Wavelength: 0.0094 meters

This Option was Selected on the Previous Page

3 Parabolic Reflector   Polarization: RHCP

1	Yagi	Maximum Gain:	0.0	dBiC	Beamwidth:	39.7 °
2	Helix	Gain:	0.0	dBiC	Beamwidth:	32.2 °
3	Parabolic Reflector	Gain:	0.0	dBiC	Beamwidth:	0.0 °
4	User Defined	Gain:	18.5	dBiC	Beamwidth:	24.0 °

12.40

Esimated Pointing Error (θ4): 0.0015 °   **Approx. Antenna Pointing Loss:** 0.1 dB

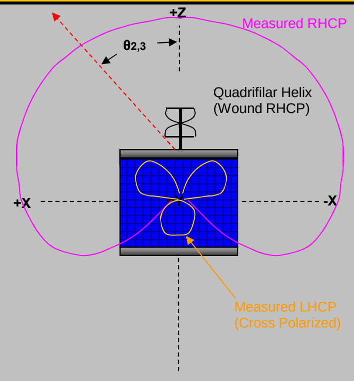


Figure 6

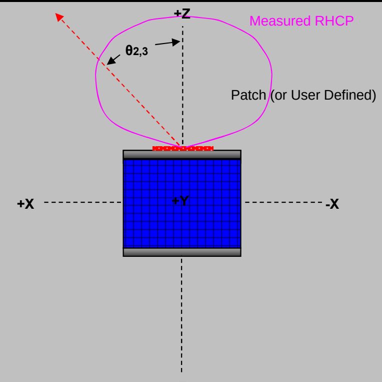


Figure 7

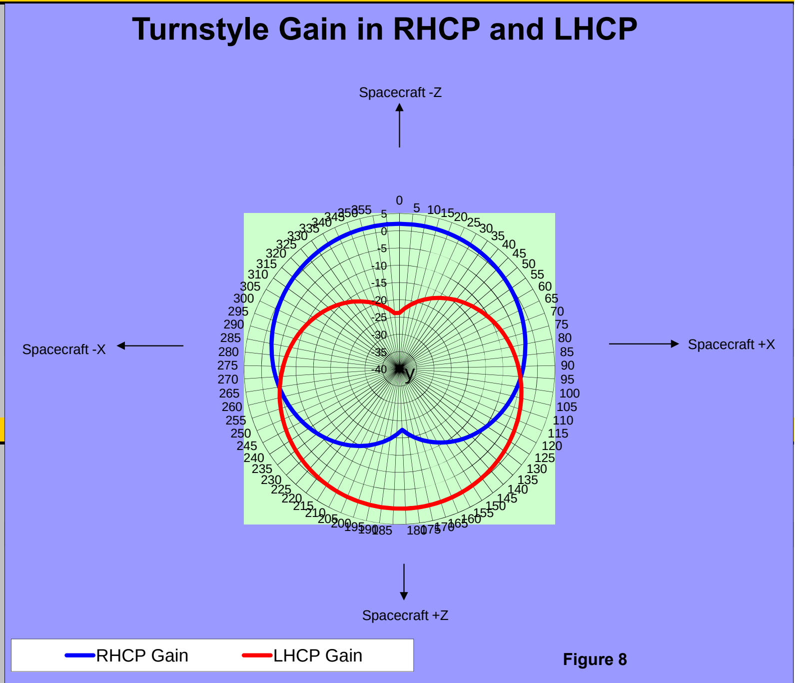


Figure 8

System Polarization Loss and Cross Polarization Isolation: Alpha CubeSat 2016 February 05

Axial Ratio ≡ 10*LOG [Power Emitted (or Received) with Antenna Aligned with Major Axis / Power Emitted (or Received) with Antenna Aligned with Minor Axis]

NOTE:

Power when Aligned with Minor Axis

Power when Aligned with Major Axis

	Circular Right Hand or Left Hand	Elliptical Right Hand or Left Hand	Linear Vertical or Horizontal
Transmit Antenna	Axial Ratio = 1.0 = 0.0 dB	Axial Ratio = 2.0 = 3.0 dB	Axial Ratio = ∞
Receive Antenna	Axial Ratio = 1.0 = 0.0 dB	Axial Ratio = 2.0 = 3.0 dB	Axial Ratio = ∞

Power when Aligned with Major Axis

NOTE:

UPLINK: Operator selects uplink antenna characteristics in blue boxes.

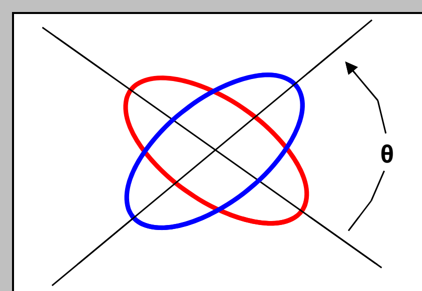
Polarization Loss Calculation:

Co-Polarization Loss:

Axial ratio of Tx Antenna (Ant. #1) in dB =	1.00	[dB]
Axial ratio (Ant. #1) =	1.26	[ ]
Axial ratio of Rx Antenna (Ant. #2) in dB =	1.00	[dB]
Axial ratio (Ant. #2) =	1.26	[ ]
Polarization Angle θ between antennas =	5.0	[degrees]
Polarization Angle θ between antennas =	0.087266	[Radians]
Polarization Loss =	0.99961	[ ]
Polarization Loss =	0.00	[dB]

Cross Polarization Coupling/Isolation:

Cross Pol. Power Fraction =	0.00039	
Cross Pol. Power Fraction =	-34.10	[dB]
Cross Polarization Isolation =	34.10	[dB]



Polarization Angle (θ) ≡ Angle between transmit and receive major axes.

Polarization Loss Equation:

PL = 0.5*(1+((1-r_1^2)*(1-r_2^2)*COS(2*θ)+4*r_1*r_2)/((1+r_1^2)*(1+r_2^2)))

DOWNLINK: Operator selects downlink antenna characteristics in blue boxes.

Polarization Loss Calculation:

Co-Polarization Loss:

Axial ratio of Tx Antenna (Ant. #1) in dB =	1.00	[dB]
Axial ratio (Ant. #1) =	1.26	[ ]
Axial ratio of Rx Antenna (Ant. #2) in dB =	1.00	[dB]
Axial ratio (Ant. #2) =	1.26	[ ]
Polarization Angle θ between antennas =	5.0	[degrees]
Polarization Angle θ between antennas =	0.087266	[Radians]
Polarization Loss =	0.99961	[ ]
Polarization Loss =	0.00	[dB]

Cross Polarization Coupling/Isolation:

Cross Pol. Power Fraction =	0.00039	
Cross Pol. Power Fraction =	-34.10	[dB]
Cross Polarization Isolation =	34.10	[dB]

Example Calculations:

	Tx Ant. A.R. #1: (dB)	Rx Ant. A.R. #2: (dB)	θ (degrees)	Pol. Loss (dB)
Tx Circular,	0.0	0.0	90.0	0.0
Rx Variable:	0.0	1.0	90.0	-0.1
	0.0	2.0	90.0	-0.2
	0.0	3.0	90.0	-0.5
	0.0	6.0	90.0	-1.3
	0.0	10.0	90.0	-2.2
	0.0	30.0	90.0	-3.0
	0.0	30.0	0.0	-3.0
Tx & Rx Elliptical:	3.0	3.0	0.0	0.0
	3.0	3.0	45.0	-0.9
	3.0	3.0	90.0	-1.9
Tx & Rx Linear:	30.0	30.0	0.0	0.0
	30.0	30.0	30.0	-1.3
	30.0	30.0	45.0	-3.0
	30.0	30.0	60.0	-6.0
	30.0	30.0	90.0	-54.0
Tx Elliptical,	2.0	30.0	0.0	-1.5
Rx Linear	2.0	30.0	45.0	-3.0
	2.0	30.0	90.0	-4.0

NOTE: A linearly polarized antenna may be represented by an Axial Ratio value of 30 dB.

NOTE: This is a typical small satellite case.

NOTE: This is also a typical small satellite case.

Atmospheric and Ionospheric Losses: Alpha CubeSat 2016 February 05

Loss due to Atmospheric Gases:			
Uplink and Downlink:			
Elevation Angle:		*Loss:*	*Unit:*
0 °		10.2	dB
2.5 °		4.6	dB
5 °		2.1	dB
10 °		1.1	dB
30 °		0.4	dB
45 °		0.3	dB
90 °		0.0	dB
Min. Elev. Angle:		5	deg.
Loss Determined:		2.1	dB

NOTE:

Link Losses Resulting from Signals Passing Through Atmospheric Gases:

Losses due to atmospheric gases (Nitrogen, Oxygen, Carbon Dioxide, Hydrogen, etc.) are nearly independent of atmospheric temperature, mean density and relative humidity at frequencies below 2 GHz. Atmospheric absorption depends strongly upon the total number of molecules distributed along the path between the spacecraft and the ground station. This, in turn, means that the losses from or to the satellite are elevation angle dependent.

The table to the left is a look-up table. The minimum elevation angle selected in the "Orbit" worksheet is matched against the closest fit from the table and the result is given at Cell [D23] and is automatically inserted into the uplink and downlink budgets.

The data used here is taken from "*Radiowave Propagation in Satellite Communications*" by Louis J. Ippolito, Jr., Van Nostrand-Reinhold, 1986, pp. 33-34, Tables 3-3a-c.

One additional interpolated value is added at 2.5° elevation angle. This was not taken from Ippolito's text.

If you are using uplink or downlink frequencies above 2 GHz, refer to the referenced text given above to determine the appropriate atmosperic losses. At millimeter wave frequencies the losses can be much higher.

Loss due to Ionosphere:			
Uplink:	Loss Determined:		0.0 dB
Frequency:	*Unit:*	*Loss:*	*Unit:*
146	MHz	0.7	dB
438	MHz	0.4	dB
2410	MHz	0.1	dB
7145	MHz	0.04	dB

Link Model Operator Estimate Inserted Here.

Loss due to Ionosphere:			
Downlink:	Loss Determined:		0.0 dB
Frequency:	*Unit:*	*Loss:*	*Unit:*
146	MHz	0.7	dB
438	MHz	0.4	dB
2410	MHz	0.1	dB
32000	MHz	0.008	dB

Link Model Operator Estimate Inserted Here.
If the "Link Model Operator" has selected a user option for the frequency, then an estimate of the ionospheric losses must be provided by the operator.

Link Losses Resulting from Signals Passing Through the Ionosphere:

Radio waves passing through the ionosphere at VHF, UHF and Microwave frequencies are influenced far less by this layer of ionized particles than at frequencies in the HF, MF and LF portions of the radio spectrum. While there is certainly some correlation between the elevation angle to a satellite and the signal absorption or scintillation experienced, this dependency is nearly masked out by the time variability of effects.

There is, however, a frequency depencency that can be quantified, on average. As transmitter frequencies go below 100 MHz there are times when the attenuation can increase to as much as tens of dB, especially at low elevation angles. The ionosphere certainly limits the lowest frequency at which satellite communications is feasible. Below 20 MHz, during solar maximum space signals are usually fully absorbed or reflected by the layers of the ionosphere (D, E, F_1 and F_2).

The values provided in this table are approximate mean values for low earth station elevation angles. It is proposed that these values can be conservatively used in satellite link analyses. The higher order statistics of these loss parameters would be interesting to review, however, this effort is more than is necessary for the development of an effective link budget.

The losses determined here for the uplink and downlink are based on the operator-selected frequency choice made in the "Orbit" worksheet. If the "User Defined" option is selected by the link model operator, then the operator must estimate the appropriate ionospheric loss value and manually insert it in either Cell [D34] or Cell [D47] accordingly.

Proceed to the "Modulation-Demodulation Method" W/S.

Modulation/Demodulation Method: **Alpha CubeSat** **2016 February 05**

NOTE:			Select Here:	Choice Made:	Result:
UPLINK:	**Modulation, Coding & BER Option:**		**18**	QPSKw.FEC	**Eb/No: Threshold**
Command Link					**10.6** dB

Option:	Modulation Type:	Coding:	Bit Error Rate Spec:	Required Eb/No (dB):
1	AFSK/FM	None	1.00E-04	21.0
2	AFSK/FM	None	1.00E-05	23.2
3	G3RUH FSK	None	1.00E-04	16.7
4	G3RUH FSK	None	1.00E-05	18.0
5	Non-Coherent FSK	None	1.00E-04	13.4
6	Non-Coherent FSK	None	1.00E-05	13.8
7	Coherent FSK	None	1.00E-04	10.5
8	Coherent FSK	None	1.00E-05	11.9
9	GMSK	None	1.00E-04	8.4
10	GMSK	None	1.00E-05	9.6
11	BPSK	None	1.00E-05	9.6
12	BPSK	None	1.00E-06	10.5
13	QPSK	None	1.00E-05	9.6
14	QPSK	None	1.00E-06	10.5
15	BPSK	Convolutional R=1/2, K=7	1.00E-06	4.8
16	BPSK	Conv. R=1/2,K=7 & R.S. (255,223)	1.00E-06	2.5
17	BPSK	Conv. R=1/6,K=15 & R.S. (255,223)	1.00E-07	0.8
18	QPSKw.FEC	Reed Solomon FEC	1.00E-06	9.6

Operator Estimate of Implementation Loss

NOTE:	**Implementation Loss Estimate:**	**1.0**	dB

UPLINK: ↑ **DOWNLINK:** ↓

			Select Here:	Choice Made:	Result:
DOWNLINK:	**Modulation, Coding & BER Option:**		**19**	16QAM	**Eb/No: Threshold**
Telemetry Link:					**0.9** dB

Option:	Modulation Type:	Coding:	Bit Error Rate Spec:	Required Eb/No (dB):
1	AFSK/FM	None	1.00E-04	21.0
2	AFSK/FM	None	1.00E-05	23.2
3	G3RUH FSK	None	1.00E-04	16.7
4	G3RUH FSK	None	1.00E-05	18.0
5	Non-Coherent FSK	None	1.00E-04	13.4
6	Non-Coherent FSK	None	1.00E-05	13.8
7	Coherent FSK	None	1.00E-04	10.5
8	Coherent FSK	None	1.00E-05	11.9
9	GMSK	None	1.00E-04	8.4
10	GMSK	None	1.00E-05	9.6
11	BPSK	None	1.00E-05	9.6
12	BPSK	None	1.00E-06	10.5
13	QPSK	None	1.00E-05	9.6
14	QPSK	None	1.00E-06	10.5
15	BPSK	Convolutional R=1/2, K=7	1.00E-06	4.8
16	BPSK	Conv. R=1/2,K=7 & R.S. (255,223)	1.00E-06	2.5
17	BPSK	Conv. R=1/6,K=15 & R.S. (255,223)	1.00E-07	0.8
18	BPSK	Turbo Code (Parallel w. Interleaver)	1.00E-06	0.75
19	16QAM	Reed Solomon FEC	1.00E-07	0.9

Operator Estimate of Implementation Loss

	Implementation Loss Estimate:	**0.0**	dB

NOTE:

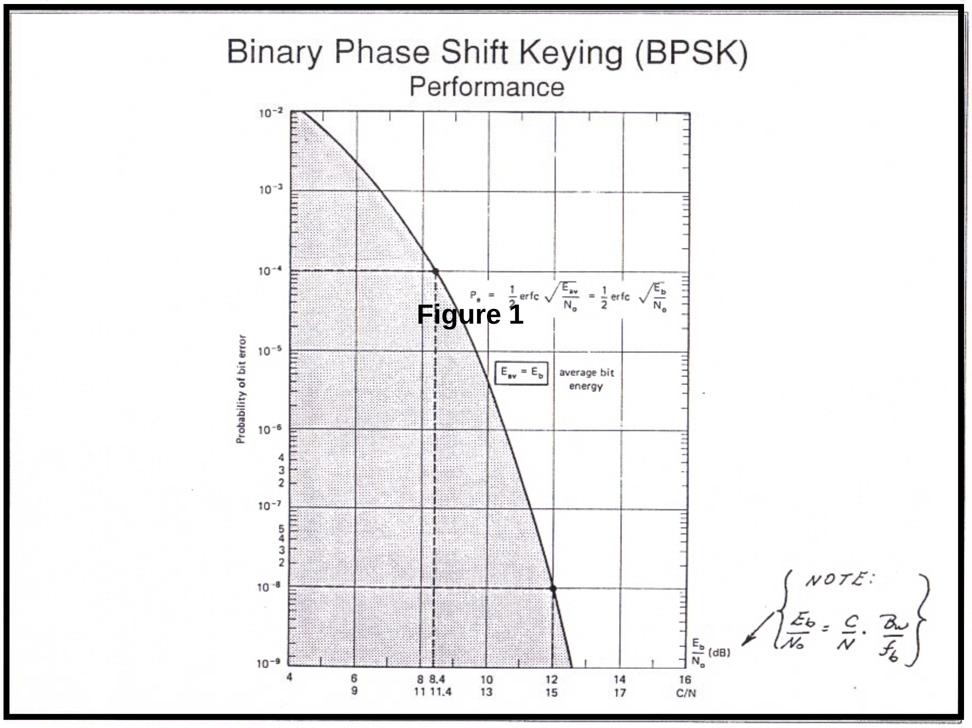

Binary Phase Shift Keying (BPSK)
Performance
Probability of bit error
$P_e = \frac{1}{2} \operatorname{erfc} \sqrt{\frac{E_{av}}{N_o}} = \frac{1}{2} \operatorname{erfc} \sqrt{\frac{E_b}{N_o}}$
$\boxed{E_{av} = E_b}$ average bit energy
$\frac{E_b}{N_o}$ (dB)
C/N
NOTE:
$\frac{E_b}{N_o} = \frac{C}{N} \cdot \frac{B_w}{f_b}$

Figure 1

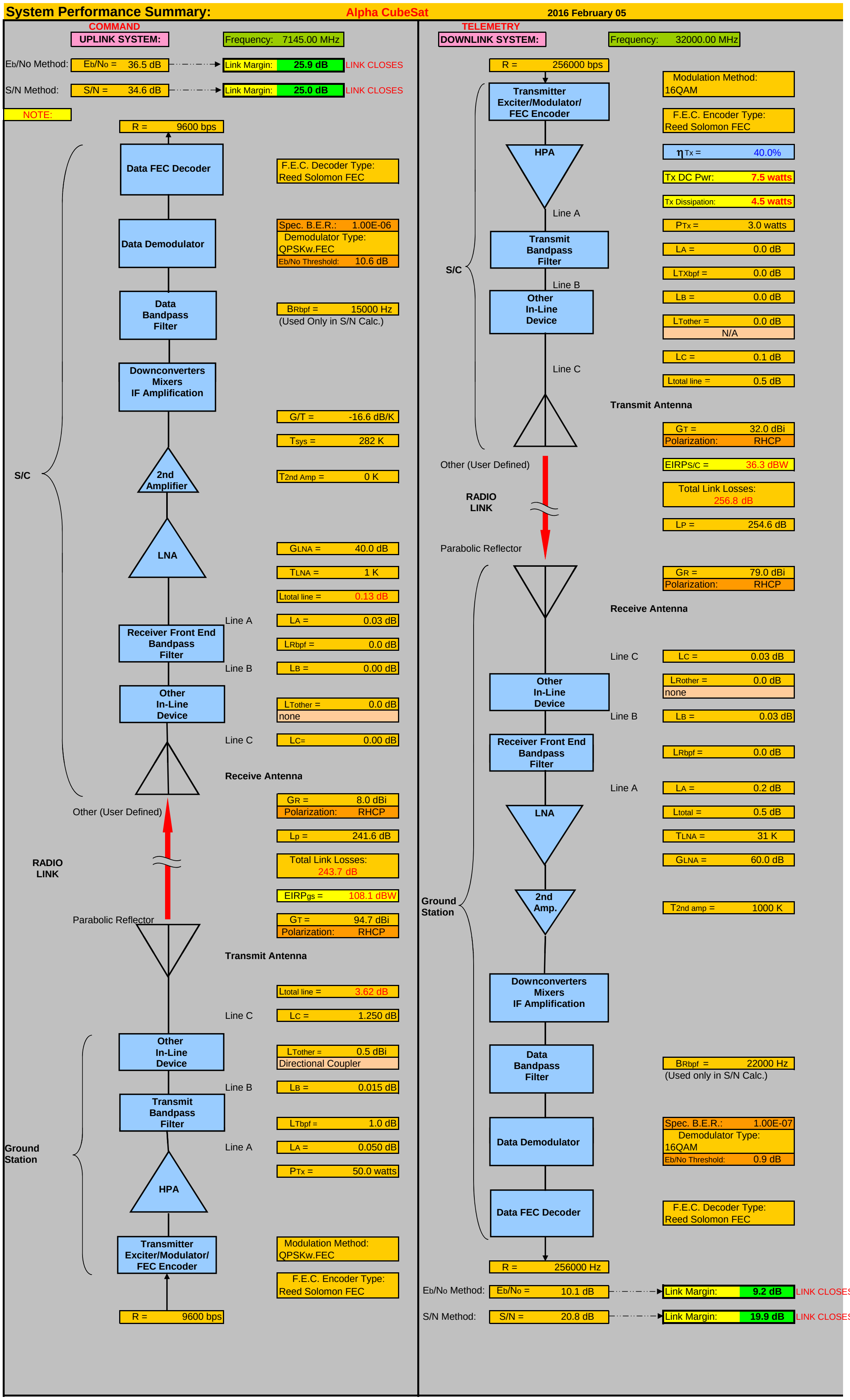
Alpha CubeSat **Uplink Command Budget:**	NOTE:		Alpha CubeSat **Version: 2.5.3**	**Date Data Last Modified:** **2016 February 05**
Parameter:	*Value:*	*Units:*	*Comments:*	
Ground Station:				
Ground Station Transmitter Power Output:	50.0	watts	This value is transferred from "Transmitters" W/S, Cell [E15].	
In dBW:	17.0	dBW	Transmitter power expressed in dB above one watt	
In dBm:	47.0	dBm	Transmitter power expressed in dB above one milliwatt	
Ground Stn. Total Transmission Line Losses:	3.6	dB	This value is transferred from "Transmitters" W/S, Cell [I33]	
Antenna Gain:	94.7	dBi	This value is selected at "Antenna Gain" W/S, Cell [E11]	
Ground Station EIRP:	108.1	dBW	Ground Station Effective Isotropic Radiated Power (EIRP) [EIRP=Pt x Ltl x Ga]	
Uplink Path:				
Ground Station Antenna Pointing Loss:	0.0	dB	This value is calculated in the "Antenna Pointing Losses" W/S, and transferred from Cell [K43]	
Gnd-to-S/C Antenna Polarization Losses:	0.0	dB	This value is calculated in the "Polarization Loss" W/S and is transferred from Cell [F40].	
Path Loss:	241.6	dB	Lp = 22 + 20LOG(D/λ); Transferred from "Frequency" W/S	
Atmospheric Losses:	2.1	dB	This value is transferred from "Atmos. & Ionos. Losses" W/S, Cell [D23]	
Ionospheric Losses:	0.0	dB	This value is transferred from "Atmos. & Ionos. Losses" W/S, Cell [D47:D50]	
Rain Losses:	0.0	dB	This value should be estimated by the link model operator and place into Cell [B18]	
Isotropic Signal Level at Spacecraft:	-135.6	dBW	This is the signal level received in space in the vacinity of the spacecraft using an omnidirectional antenna.	
Spacecraft (Eb/No Method):				
------- Eb/No Method -------				
Spacecraft Antenna Pointing Loss:	0.0	dB	This value is transferred from "Antenna Pointing Losses" W/S, Cell [K63]	
Spacecraft Antenna Gain:	8.0	dBi	This value is selected at "Antenna Gain" W/S, Cell [E24]	
Spacecraft Total Transmission Line Losses:	0.1	dB	This value is transferred from the "Receivers" W/S, Cell [J52]	
Spacecraft Effective Noise Temperature:	282	K	This value is calculated in the "Receivers" W/S and Transferred from Cell [J67]	
Spacecraft Figure of Merrit (G/T):	-16.6	dB/K	$G/T = G_a - L_{tl} - 10\log(T_s)$. This is the uptimate measure of the receiver's performance.	
S/C Signal-to-Noise Power Density (S/No):	76.3	dBHz	Boltzman's Constant: -228.6 dBW/K/Hz	
System Desired Data Rate:	9600	bps	Operator selects this value. Be Careful! This is the data rate, not the symbol rate.	
In dBHz:	39.8	dBHz	This is simply = 10log(R); R= data rate	
Command System Eb/No:	36.5	dB		
Demodulation Method Seleted:	**QPSKw.FEC**		Values selected in "Modulation-Demodulation W/S, Cell [E3]	
Forward Error Correction Coding Used:	**Reed Solomon FEC**		Value selected in "Modulation-Demodulation" W/S, also Cell [E3]	
System Allowed or Specified Bit-Error-Rate:	1.0E-06		The selected value is transferred from the "Modulation-Demodulation W/S, Cells [E6:E23]	
Demodulator Implementation Loss:	1.0	dB	This value is transferred from the "Modulation-Demodulation W/S, Cell[E25]	
Telemetry System Required Eb/No:	9.6	dB	The selected value is transferred from the "Modulation-Demodulation W/S, Cells [F6:F23]	
Eb/No Threshold:	10.6	dB	This is the result of the "Modulation-Demodulation" W/S and is transferred from Cell [H32]	
System Link Margin:	**25.9**	dB		
Spacecraft Alternative Signal Analysis Method (SNR Computation):			← **NOTE:**	

---------- *SNR Method* ------------

Spacecraft Antenna Pointing Loss:	0.0	dB	This value is transferred from "Antenna Pointing Losses" W/S, Cell [K63]
Spacecraft Antenna Gain:	8.0	dBi	This value is selected at "Antenna Gain" W/S, Cell [E24]
Spacecraft Total Transmission Line Losses:	0.1	dB	This value is transferred from the "Receivers" W/S, Cell [J52]
Spacecraft Effective Noise Temperature:	282	K	This value is calculated in the "Receivers" W/S and Transferred from Cell [J67]
Spacecraft Figure of Merrit (G/T):	-16.6	dB/K	$G/T = G_a - L_{tl} - 10\log(T_s)$. This is the ultimate measure of the receiver's performance.
Signal Power at Spacecraft LNA Input:	-127.8	dBW	$P_s = P_{iso} + G_a - L_{pl} - L_{tl}$; This is the signal power that has arrived at the ground station receiver.
Spacecraft Receiver Bandwidth:	15,000	Hz	Signal Spectrum Must Pass Through This Data Filter. **NOTE:**
Spacecraft Receiver Noise Power (Pn = kTB)	-162.3	dBW	Pn = K + 10log(Ts) + 10log(B). This is the total noise power arriving at the ground station receiver.
Signal-to-Noise Power Ratio at G.S. Rcvr:	34.6	dB	Ps/Pn = Ps(in dBW) - Pn(in dBW)
Analog or Digital System Required S/N:	9.6	dB	If system is digital, use values from "Modulation-Demodulation" W/S. If analog, use appropriate value from text book.
System Link Margin	**25.0**	dB	

Alpha CubeSat **Downlink Telemetry Budget:**	NOTE:		Alpha CubeSat **Version: 2.5.3**	**Date Data Last Modified:** **2016 February 05**
Parameter:	*Value:*	*Units:*	*Comments:*	
Spacecraft:				
Spacecraft Transmitter Power Output:	3.0	watts	This value is transferred from "Transmitters" W/S, Cell [E50]	
In dBW:	4.8	dBW	Transmitter power expressed in dB above one watt	
In dBm:	34.8	dBm	Transmitter power expressed in dB above one milliwatt	
Spacecraft Total Transmission Line Losses:	0.5	dB	This value is transferred from "Transmitters" W/S, Cell [I68]	
Spacecraft Antenna Gain:	32.0	dBi	This value is selected at "Antenna Gain" W/S, Cell [E41]	
Spacecraft EIRP:	36.3	dBW	Spacecraft Effective Isotropic Radiated Power (EIRP) [EIRP=Pt x Ltl x Ga]	
Downlink Path:				
Spacecraft Antenna Pointing Loss:	0.0	dB	This value is calculated in the "Antenna Pointing Losses" W/S, and trasferred from Cell [K85]	
S/C-to-Ground Antenna Polarization Loss:	0.0	dB	This value is calculated in the "Polarization Loss" W/S and is transferred from Cell [F60].	
Path Loss:	254.6	dB	Lp = 22 + 20LOG(D/λ); Transferred from "Frequency" W/S	
Atmospheric Loss:	2.1	dB	This value is transferred from "Atmos. & Ionos. Losses" W/S, Cell [D23]	
Ionospheric Loss:	0.0	dB	This value is transferred from "Atmos. & Ionos. Losses" W/S, Cell [D47:D50]	
Rain Loss:	0.0	dB	This value should be estimated by the link model operator and place into Cell [B18]	
Isotropic Signal Level at Ground Station:	-220.4	dBW	This is the signal level received at the Earth in the vacinity of the ground station using an omnidirectional antenna.	
Ground Station (EbNo Method): ***------- Eb/No Method -------***				
Ground Station Antenna Pointing Loss:	0.1	dB	This value is transferred from "Antenna Pointing Losses" W/S, Cell [K102]	
Ground Station Antenna Gain:	79.0	dBi	This value is selected at "Antenna Gain" W/S, Cell [E58]	
Ground Station Total Transmission Line Losses:	0.5	dB	This value is transferred from the "Receivers" W/S, Cell [J123]	
Ground Station Effective Noise Temperature:	174	K	This value is calculated in the "Receivers" W/S and Transferred from Cell [J138]	
Ground Station Figure of Merrit (G/T):	56.1	dB/K	G/T = Ga-L_{tl}-10log(T_s). This is the uptimate measure of the receiver's performance.	
G.S. Signal-to-Noise Power Density (S/No):	64.2	dBHz	Boltzman's Constant: -228.6 dBW/K/Hz	
System Desired Data Rate:	256000	bps	Operator selects this value. Be Careful! This is the data rate, not the symbol rate.	
In dBHz:	54.1	dBHz	This is simply = 10log(R); R= data rate	
Telemetry System Eb/No for the Downlink:	10.1	dB		
Demodulation Method Seleted:	**16QAM**		Values selected in "Modulation-Demodulation W/S, Cell [E30]	
Forward Error Correction Coding Used:	**Reed Solomon FEC**		Value selected in "Modulation-Demodulation" W/S, also Cell [E30]	
System Allowed or Specified Bit-Error-Rate:	1.0E-07		The selected value is transferred from the "Modulation-Demodulation W/S, Cells [E33:E50]	
Demodulator Implementation Loss:	0	dB	This value is transferred from the "Modulation-Demodulation W/S, Cell[E52]	
Telemetry System Required Eb/No:	0.9	dB	The selected value is transferred from the "Modulation-Demodulation W/S, Cells [F33:F50]	
Eb/No Threshold:	0.9	dB	This is the result of the "Modulation-Demodulation" W/S and is transferred from Cell [H32]	
System Link Margin:	**9.2**	dB		
Ground Station Alternative Signal Analysis Method (SNR Computation): ***---------- SNR Method ------------***				
Ground Station Antenna Pointing Loss:	0.1	dB	This value is transferred from "Antenna Pointing Losses" W/S, Cell [K102]	
Ground Station Antenna Gain:	79.0	dBi	This value is selected at "Antenna Gain" W/S, Cell [E58]	

Ground Station Total Transmission Line Losses:	0.5	dB	This value is transferred from the "Receivers" W/S, Cell [J123]
Ground Station Effective Noise Temperature:	174	K	This value is calculated in the "Receivers" W/S and Transferred from Cell [J138]
Ground Station Figure of Merrit (G/T):	56.1	dB/K	G/T = Ga-Ltl-10log(Ts). This is the ultimate measure of the receiver's performance.
Signal Power at Ground Station LNA Input:	-142.0	dBW	Ps = Piso+Ga-Lpl-Ltl; This is the signal power that has arrived at the ground station receiver.
Ground Station Receiver Bandwidth (B):	22,000	Hz	Signal Spectrum Must Pass Through This Data Filter **NOTE:**
G.S. Receiver Noise Power (Pn = kTB)	-162.8	dBW	Pn = K + 10log(Ts) + 10log(B). This is the total noise power arriving at the ground station receiver.
Signal-to-Noise Power Ratio at G.S. Rcvr:	20.8	dB	Ps/Pn = Ps(in dBW) - Pn(in dBW)
Analog or Digital System Required S/N:	0.9	dB	If system is digital, use values from "Modulation-Demodulation" W/S. If analog, use appropriate value from text book.
System Link Margin	**19.9**	dB	



System Performance Summary:
Alpha CubeSat
2016 February 05
COMMAND
UPLINK SYSTEM:
Frequency: 7145.00 MHz
Eb/No Method:
Eb/No = 36.5 dB
Link Margin: 25.9 dB
LINK CLOSES
S/N Method:
S/N = 34.6 dB
Link Margin: 25.0 dB
LINK CLOSES
NOTE:
R = 9600 bps
Data FEC Decoder
F.E.C. Decoder Type: Reed Solomon FEC
Data Demodulator
Spec. B.E.R.: 1.00E-06
Demodulator Type: QPSKw.FEC
Eb/No Threshold: 10.6 dB
Data Bandpass Filter
BRbpf = 15000 Hz
(Used Only in S/N Calc.)
Downconverters Mixers IF Amplification
G/T = -16.6 dB/K
Tsys = 282 K
S/C
2nd Amplifier
T2nd Amp = 0 K
LNA
GLNA = 40.0 dB
TLNA = 1 K
Ltotal line = 0.13 dB
Line A
LA = 0.03 dB
Receiver Front End Bandpass Filter
LRbpf = 0.0 dB
Line B
LB = 0.00 dB
Other In-Line Device
LTother = 0.0 dB
none
Line C
LC= 0.00 dB
Receive Antenna
GR = 8.0 dBi
Polarization: RHCP
Other (User Defined)
Lp = 241.6 dB
RADIO LINK
Total Link Losses: 243.7 dB
EIRPgs = 108.1 dBW
Parabolic Reflector
GT = 94.7 dBi
Polarization: RHCP
Transmit Antenna
Ltotal line = 3.62 dB
Line C
LC = 1.250 dB
Other In-Line Device
LTother = 0.5 dBi
Directional Coupler
Line B
LB = 0.015 dB
Transmit Bandpass Filter
LTbpf = 1.0 dB
Ground Station
Line A
LA = 0.050 dB
HPA
PTx = 50.0 watts
Transmitter Exciter/Modulator/ FEC Encoder
Modulation Method: QPSKw.FEC
F.E.C. Encoder Type: Reed Solomon FEC
R = 9600 bps
TELEMETRY
DOWNLINK SYSTEM:
Frequency: 32000.00 MHz
R = 256000 bps
Transmitter Exciter/Modulator/ FEC Encoder
Modulation Method: 16QAM
F.E.C. Encoder Type: Reed Solomon FEC
HPA
ηTx = 40.0%
Tx DC Pwr: 7.5 watts
Tx Dissipation: 4.5 watts
Line A
PTx = 3.0 watts
Transmit Bandpass Filter
LA = 0.0 dB
S/C
LTXbpf = 0.0 dB
Line B
LB = 0.0 dB
Other In-Line Device
LTother = 0.0 dB
N/A
Line C
LC = 0.1 dB
Ltotal line = 0.5 dB
Transmit Antenna
GT = 32.0 dBi
Polarization: RHCP
Other (User Defined)
EIRPS/C = 36.3 dBW
RADIO LINK
Total Link Losses: 256.8 dB
LP = 254.6 dB
Parabolic Reflector
GR = 79.0 dBi
Polarization: RHCP
Receive Antenna
Line C
LC = 0.03 dB
Other In-Line Device
LRother = 0.0 dB
none
Line B
LB = 0.03 dB
Receiver Front End Bandpass Filter
LRbpf = 0.0 dB
Line A
LA = 0.2 dB
LNA
Ltotal = 0.5 dB
TLNA = 31 K
GLNA = 60.0 dB
Ground Station
2nd Amp.
T2nd amp = 1000 K
Downconverters Mixers IF Amplification
Data Bandpass Filter
BRbpf = 22000 Hz
(Used only in S/N Calc.)
Data Demodulator
Spec. B.E.R.: 1.00E-07
Demodulator Type: 16QAM
Eb/No Threshold: 0.9 dB
Data FEC Decoder
F.E.C. Decoder Type: Reed Solomon FEC
R = 256000 Hz
Eb/No Method:
Eb/No = 10.1 dB
Link Margin: 9.2 dB
LINK CLOSES
S/N Method:
S/N = 20.8 dB
Link Margin: 19.9 dB
LINK CLOSES

KA BAND SATELLITE COMMUNICATIONS DESIGN ANALYSIS AND OPTIMISATION

LEONG See Chuan, SUN Ru-Tian, YIP Peng Hon

ABSTRACT

Ka band satellite communications (SATCOM) frequencies provide new opportunities to meet high bandwidth demands, especially for small aerial, maritime and mobile land platforms supporting beyond line of sight requirements for network-centric operations. This is possible due to the availability of 3.5GHz of bandwidth, and also because Ka ground segment components are typically smaller in dimension compared to those of Ku band. However, Ka band links experience much higher rain fades in tropical regions as compared to Ku band and C band. In this article, various factors in the link budget are explored to determine their impact on overall signal strength. These factors can be traded off and optimised to enable the realisation of a Ka band solution for SATCOM.

Keywords: Ka band, satellite communications, link budget, trade-off analysis, mitigation technique

INTRODUCTION

Various types of satellites, including Geosynchronous Earth Orbit (GEO), Medium Earth Orbit and Low Earth Orbit support beyond line of sight communications. The link budget analysis in this article is based on GEO satellites. A GEO satellite orbits at a fixed longitudinal location at an altitude of about 36,000km above the equator. The transponders on the satellite provide a signal boost and frequency translation of signals for the ground terminals. The antennas on the satellite are designed to provide the required communications coverage to the terminals on the ground. The ground segment comprises the hub and remote terminals of different sizes and transmission powers. The remote terminals can be hosted on different static or mobile platforms.

Operating in the Ka band offers some significant advantages over conventional satellite networks operating in the Ku band and lower frequencies. Not only is more bandwidth available at the higher Ka band frequencies, Ka band antennas have higher gain than antennas of comparable size operating at lower frequencies. However, the disadvantage of using the Ka band is that adverse weather conditions impact the Ka band much more than at lower frequencies. It is therefore important that there is appropriate planning for the implementation of well-designed ground systems, network links reliability and resources so as to mitigate these adverse weather effects (Petranovichl, 2012) (Abayomi Isiaka Yussuff, & Nor Hisham Khamis, 2012) (Brunnenmeyer, Milis & Kung, 2012).

This article presents a design approach and analysis of key satellite communications (SATCOM) network parameters for a Ka band network. Various trade-offs and optimisation between operational parameters (e.g. link availability), ground segment (e.g. power amplifier ratings and antenna sizes) and space segment (e.g. transponder power and bandwidth) will be considered. In addition, mitigation techniques such as hub site diversity, adaptive coding and modulation (ACM) and uplink power control are explored to mitigate the increased rain fades at Ka band and improve the overall link availability. This analysis demonstrates that it is feasible to use the Ka band to support mission critical SATCOM operations in our region.

KA BAND DRIVERS

The Ka band is attractive as a SATCOM solution due to a few reasons.

Availability of Spectrum and Higher Throughput

Substantially more spectrum bandwidth is available at the Ka band than at the Ku band and other lower frequencies. For example, Ku band allocation is around 2GHz for uplink and 1.3GHz for downlink with actual contiguous bandwidth allocation of less than 0.5GHz per satellite. In comparison, the Ka band SATCOM has a bandwidth of 3.5GHz for both uplink and downlink. Table 1 illustrates the military and civilian frequency allocation. With the wider spectrum availability at the Ka band, higher traffic throughput can be supported. Full motion video for example, has been identified as a key driver in the demand for bandwidth that can be realised by Ka band satellites (Northern Sky Research [NSR], 2012). In addition, as the Ka band has commercial and military bands adjacent to each other, commercial services can also complement the military band's capacity.

Band	Receive (GHz)	Transmit (GHz)
Military	20.2 - 21.2	30.0 - 31.0
Civilian	17.7 - 20.2	27.5 - 30.0

Table 1. Frequency allocation within the Ka band

Greater Cost Efficiency

Ka band satellites feature narrow spot beams (0.5° to 1.5° at 3dB beam width) which support greater frequency reuse in geographically isolated spots. With larger allocation and frequency reuse capabilities, using the Ka band translates to at least a 1 to 2 order magnitude increase in transponder throughput, therefore reducing leasing cost per unit bandwidth.

Smaller Terminals

At higher frequencies, wavelengths are smaller, allowing proportionally smaller, lighter weight and probably less expensive terminals to be realised. The reduction of physical dimensions therefore allows Ka band SATCOM to be made available for new markets such as manpacks and mobile platforms. The use of more focused and narrow Ka band spot beams provides higher equivalent isotropic radiated power (EIRP), signal gain (G/T) and therefore better signal link quality or higher data rates for these smaller terminals. Comparing the Ka band to the Ku band, the improvement in overall link quality with the use of narrow spot beams is in the range of 6dB to 10dB.

Greater Resiliency to Interference

With wider Ka band bandwidth, better inherent anti-interference properties can be achieved (e.g. frequency hopping or direct sequence spread spectrum). With Ka band transponder sizes of 125MHz or more over 54MHz at Ku band, the additional interference margin with twice the spreading can be improved by at least 3dB.

KA BAND CHALLENGES

With the introduction of smaller mobile terminals for Ka band SATCOM, more stringent link requirements will need to be met. The design challenges are as follows:

Meeting Adjacent Satellite Interference Regulations

The regulatory bodies governing satellite communications include the International Telecommunication Union (ITU) and the Federal Communications Commission. With the high density of satellites in orbit and many more Ka band satellites planned for launch, adjacent satellite interference (ASI) will be a key concern. Satellite terminals that wish to transmit must meet the emission regulations. ASI is more challenging for small terminals where the antenna side lobe powers are large with respect to their main lobes, thereby limiting the maximum power they are allowed to transmit. When these terminals are on the move, allowable emissions are constrained further as the mechanical antenna pointing accuracy experienced during shock and vibration needs to be accounted for during movement through land, various sea states or air turbulence.

Large Rain Attenuation

The SATCOM link that passes through the atmosphere is degraded by rain, fog, cloud, ice, snow and hail. The biggest challenge in using the Ka band is the high rain attenuation compared with the Ku band and higher rainfall rates in the tropics. Since the electromagnetic wave absorption component is increased at Ka band, the amount of attenuation per unit length is also increased (see Figure 1). Additional

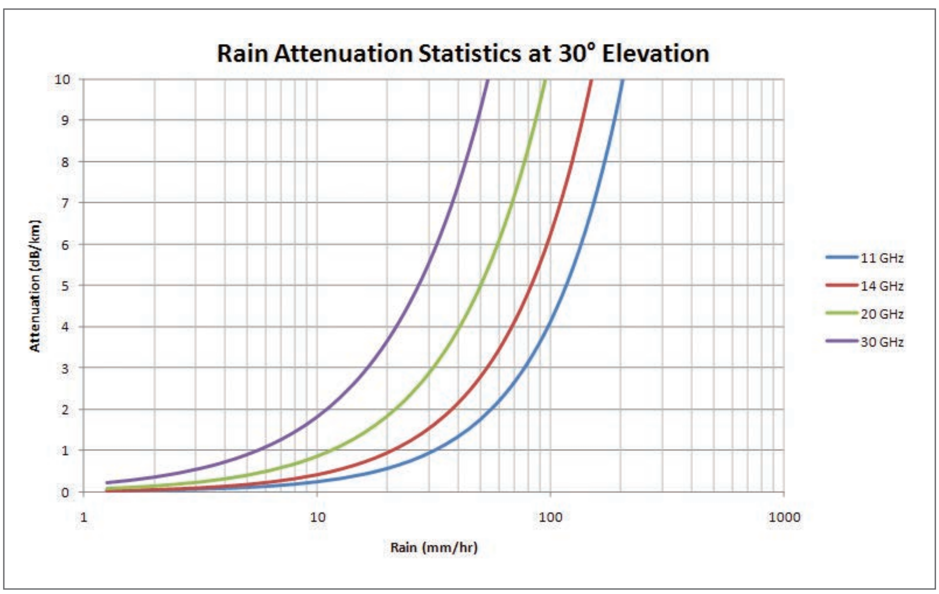


Figure 1. Rain attenuation statistics at 30 degrees elevation

margin is needed to ensure high system availability or trade-off in link availability. However, adding an additional margin may be impractical for remote terminals with small antenna and low power amplifier that operates in high rainfall regions. For example, collected rain statistics in Singapore generated by Leong and Foo (2007) show a higher rain rate than ITU specifications (International Telecommunications Union – Radiocommunications Sector [ITU-R], 2012). This results in a downlink rain loss of 12dB at the Ka band versus 2.6dB at the Ku band to achieve 99% link availability. In addition to higher attenuation, the rain fade rate at the Ka band will be very much higher than at the Ku band. The high rain fade rate will impact the operation of mitigation measures such as ACM algorithms built into the satellite modem.

MITIGATION TECHNIQUES

The large rain attenuation at the Ka band may not be compensated fully by the improvement in Ka band narrow spot beams and better interference environment. Degradations in link quality can be further mitigated by employing three main techniques.

Hub Site Diversity

Site diversity is a fade mitigation measure that involves two or more hub terminals set up to transmit or receive the signal in real time by using an algorithm to choose the least amount of link degradation among all the hub sites at any one instance. When one hub experiences rain and detects that the link may be cut, the algorithm calls for a switchover to the other hub where there are clear skies (see Figure 2).

For site diversity to be useful, there are two main considerations. First, hub sites must be sufficiently separated to achieve the required diversity gain or diversity improvement factor. It is shown that diversity gain improves with distance but the gain tapers off at distances more than 11km as it can be treated as a single site fade event (Leong, Loh, Chen, Yip, & Koh, 2012). Table 2 shows that the diversity gain is not just a function of distance but also the orientation of the line connecting the two sites. The diversity gain for Sentosa-Woodlands (South-North direction) is almost equivalent to the Tuas-Changi (West-East) site combination although the distance between each pair of sites is quite different. Second, when a site diversity decision is made, the downtime incurred from the hub switchover and the predicted duration of rain outage must both be taken into account. Due to the complexity of site diversity and the resulting cost of implementation, it will be more cost effective to use Ka band satellite networks.

The hub diversity concept can similarly be extended to remote terminals. In a bent pipe link, when the transmitter and receivers are located at a distance apart, the two sites may not experience the same amount of rainfall but the rainfall at the sites may be correlated. Therefore, in a typical link budget planning, the dual rain fade conditions for both the uplink and downlink are considered when the distance between the transmitter and receiver is less than 3km. For distances greater than 50km, a single rain fade condition, usually on the uplink side, is considered. In these two planning methods, the range of rain attenuation at 99% total link availability at the Ka band varies from 12dB to 39dB. Due to this large attenuation range, it is therefore important to plan the attenuation value accurately so as to meet the end user service level agreement while optimising the entire ground and space resources (Leong, 2012).

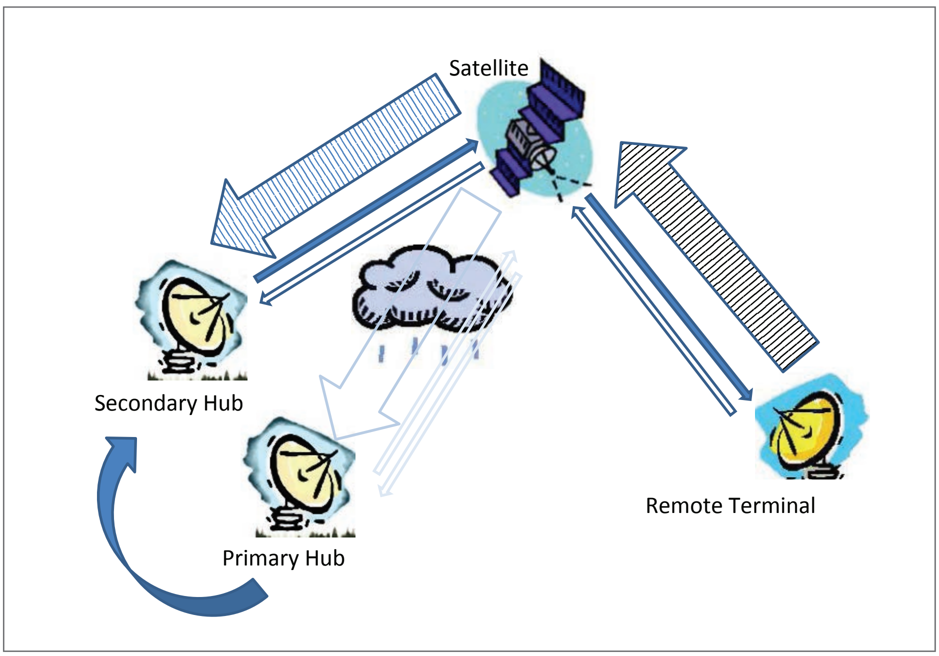


Figure 2. Illustration of hub site diversity

Selection Combination	Div Gain / dB	Dist / km
Tuas-Sentosa	11.2	22.72
Tuas-Woodlands	10.1	24.40
Sentosa-Woodlands	13.9	23.62
Sentosa-Changi	8.8	23.13
Woodlands-Changi	12.0	27.49
Tuas-Changi	14.8	42.44

Table 2. Diversity gain improvement over a single site

Adaptive Coding and Modulation

In ACM, the modulation and coding (MODCOD) of the carrier is altered within the modem in step sizes to increase the survivability of the transmission link. By decreasing the data rate, the signal to noise ratio required for a lower MODCOD is reduced and therefore the carrier becomes more resilient to rain fade. To support a varying data rate transmission during dynamic rain conditions, the video codec running in the application layer should allow a seamless reduction in video quality or resolution to ensure that the recipient is able to receive it. In other words, by adjusting the MODCOD, it is possible to optimise the trade-off between performance and survivability. Applications therefore need to be designed and tested accordingly to take full advantage of the ACM capability. ACM typically provides 15dB of margin across the full range of MODCODs.

Automatic Uplink Power Control

Automatic Uplink Power Control (AUPC) is implemented by increasing carrier power at the transmit end to ensure link survivability. When a rain fade event is encountered, more power is drawn from the high power amplifier (HPA) to maintain the carrier to noise ratio. Due to the need for additional equipment, AUPC is usually employed only at larger hub stations since the smaller remote terminals' HPA may already be operating with negligible backoff during clear sky. AUPC at hub stations typically provide 15dB of power control margin.

DESIGN ANALYSIS AND OPTIMISATION

Taking into consideration space segment parameters; ground segment mitigation techniques that improve the link quality; environment factors that decrease the link quality significantly; and the increased use of high bandwidth demand video application, a more stringent design analysis approach for link budget calculations is required. The approach will also require a sensitivity analysis, where various trade-offs between operational parameters (e.g. desired link availability for control and mission links), ground segment (e.g. power amplifier ratings and antenna sizes) and space segment (e.g. transponder power and bandwidth) can be analysed and optimised. Through these trade-off analyses, the feasibility of using the Ka band to support mission critical military aeronautical, maritime and land SATCOM operations can be determined.

There are many parameters to consider in the link budget. The primary parameters are as shown in Figure 3.

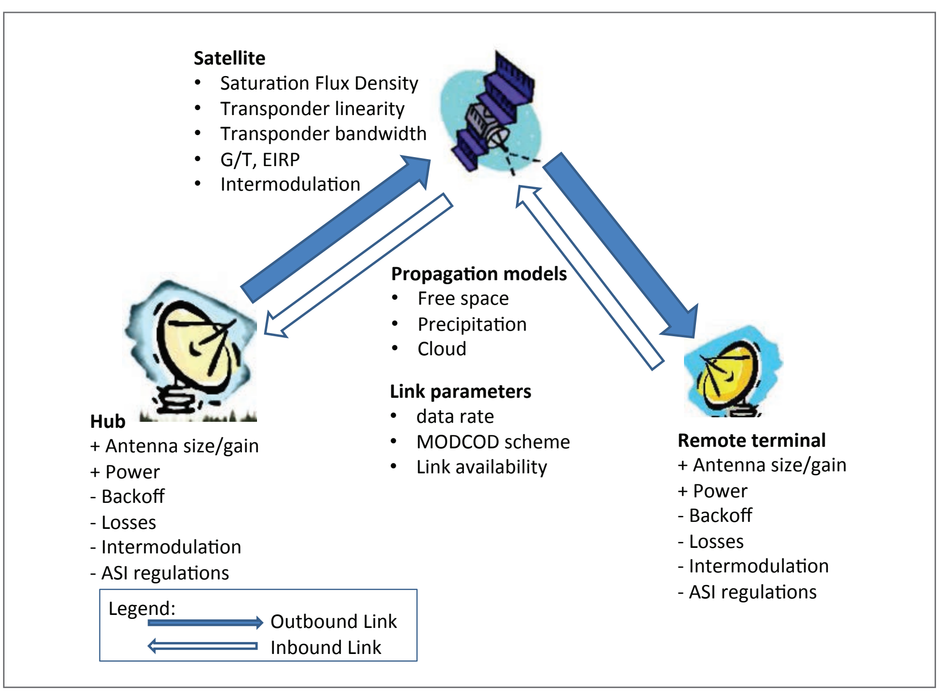


Figure 3. Major link parameters used in link budget analysis

It is recommended to start the satellite network design by first identifying the design boundaries – which are the most constraining factor(s) and which are the parameters that are within and outside of the designers' control. The typical constraints are as follows:

Satellites

Usually, the area of operations will define the choice of satellites. If two or more satellites are able to provide the required coverage, then parameters such as the available power and bandwidth on the transponder, receiver G/T, saturation points of the receivers and saturation flux density (SFD) can be used for the trade-off analysis. The linearity of the transponders is also an indicator of their performance. The more linear they are, the lower the intermodulation noise relative to the carrier will be produced, and therefore the better the output signal which can be achieved.

Remote Terminals and Hub

Constraints for remote terminals include the infrastructure or platform they will be hosted in. If the terminals are to be used on the move, the platform will very likely limit the antenna size/weight, position, minimum/maximum elevation angles and/or power amplifier size. If the hub has been implemented, its fixed infrastructure such as antenna size and power amplifier size may be constraining factors. Transmit power back-off (reduction in the transmit power level) and intermodulation noise should be catered for if multiple frequency carriers are transmitted from a common power amplifier. Losses due to cables and interconnectors as well as inaccuracies in antenna pointing should also be taken into account.

Besides these technical parameters, the satellite network designer should also take market availability of the products into consideration.

Communication Links

a) Outbound Link - The outbound link is the overall communications link from the hub to the terminal. It consists of the hub uplink and the terminal downlink. The outbound link is generally engineered so that the terminal downlink dominates performance. Since the hub services many terminals, it is generally cost effective to make the hub antenna large enough to provide extra transmit power margin on the hub uplink.

b) Inbound Link - The inbound link is the overall communications link from the terminal to the hub. It consists of the terminal uplink and the hub downlink. The inbound link is also generally engineered so that the terminal uplink dominates performance, since the large hub antenna provides extra receive gain on the hub downlink.

c) MODCOD Scheme - The choice of MODCOD is related to the signal to noise ratio required by the modem to demodulate the signal successfully as well as the carrier bandwidth required. These parameters are usually referenced from the modem specifications. The available transmit power or the receiver sensitivity may limit the choice of MODCOD scheme.

Operational Inputs

The operational inputs consist of the information exchange requirements, data rates and link availability required for the mission. Depending on the application and mission, the end user may have minimum data rate and link availability requirements. These would then be set as design targets and inputs to the link budget analysis. They impact the satellite transponder resources directly such as power and bandwidth required to support the link.

CASE STUDY: SATCOM ON THE MOVE

A remote terminal antenna size of 0.45m or 0.6m, power amplifier of up to 20W and an inbound link of up to 5Mbps are used as the input parameters in this case study. If the choice of satellite is still open, the designer should look for one with high G/T and high linearity transponder in order to meet the desired link availability for the mission and minimise the resources required.

Sensitivity Analysis

With numerous link budget parameters, sensitivity analysis is needed to determine the critical trade-offs between size, power, bandwidth and link availability. The key findings are highlighted as follows:

a) Increasing remote terminal antenna size from 0.45m to 0.60m allows a reduction in the required transponder power equivalent bandwidth (PEB) by 20% to 40% per 64Kbps link, leading to long-term savings in operating expenses. At the same time, it allows the required power on the hub to be reduced by 30% to 40%. Both directly contribute to an increase in the number of remote terminals that can be supported.

b) It is estimated that a single transponder can support about 9 x 5Mbps or 16 x 3Mbps mission links. For the mission link, satellite SFD – a parameter controlled by the satellite service provider – and the EIRP contour in which the hub is located, are the major factors influencing the number of links which can be supported per satellite transponder. Increasing the SFD sensitivity level by 6dBW/m^2 reduces the transponder PEB required by 60% to 70%, leading to significant savings in operating expenses. It is therefore important to choose, negotiate and establish a service level with the satellite service provider which meet user requirements.

c) For a mission link with high data rate (3Mbps to 5Mbps) but small antenna (0.45m to 0.6m) and limited power (up to 20W), the maximum link availability is only 96% to 97%. With lower data rates (below 1Mbps), a higher link availability of at least 98% can be achieved.

Application of Mitigation Techniques

Hub Site Diversity

Hub site diversity provides a means to overcome rain fade on the path between the hub and the satellite. Consequently, when there is no rain attenuation, the number of links that can be supported per transponder/hub increases. In essence, this increases the total capacity of the satellite network in terms of increasing the number of remote terminals that can be supported per satellite transponder. For remote terminals equipped with 0.45m antenna and up to 20W power, hub site diversity can increase the number of remote terminals supported per transponder by up to 18%.

Adaptive Coding and Modulation

The mission link availability will be improved if ACM is applied. During rain events when the link functions in degraded mode, for example at a lower data rate, videos are transmitted at a lower resolution. By decreasing the data rate from 1Mbps to 512Kbps or 256Kbps, the link availability is increased from 98% to 98.5%. This translates to a reduction in downtime of 43.8 hours per year. Commercial-off-the-shelf satellite modems are usually equipped with ACM that enable the link to be sustained as link conditions deteriorate.

Operational Considerations

Besides designing a network with the required link availability, data rates and power, it is necessary to address operational concerns and plan for contingencies.

Impact of Loss of Mission Link and Mitigation

A link of 64Kbps could be lost in rain exceeding approximately 20mm/hr. The impact to the mission depends on factors such as the period of link outage and latency requirements of the data. Mitigating measures for link outage can include a store-and-forward method whereby the data is stored on board the platform until a communications link is re-established.

Link Resiliency

The links should be designed to be robust against intentional or unintentional interferences. The communications security and transmission security features of the SATCOM link depend to a large extent on the modem capabilities and waveform. The accuracy of tracking and pointing as well as the design of the SATCOM antennas, especially on side lobe emissions, also play a part in reducing interferences in the network.

CONCLUSION

The use of the Ka band in SATCOM has allowed for new and smaller mobile terminals that utilise high throughput applications as compared to the Ku band to be feasible options in operations. However, with significantly larger rain attenuation to overcome, the Ka band link budget design analysis is more complex than in lower frequency bands to achieve comparable link availability. The use of sensitivity and trade-off analysis in the illustrated SATCOM on the move case study demonstrates the feasibility of Ka band SATCOM in our region. Other Ka band operational considerations – such as the possibility of fallback to lower frequency band during severe fade conditions and change in transmission plans required when crossing over multiple spot beams to cover the area of operation – may also be included as part of the design analysis upon future exploration.

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BIOGRAPHY

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Configuration and the Calculation of Link Budget for a Connection via a Geostationary Satellite for Multimedia Application in the Ka band

M.A.Mebrek, L.H.Abderrahmane, A.Himeur, S.Bendoukha

***Abstract*—**In this article, we are going to do a study that consist in the configuration of a link between an earth station to broadcast multimedia service and a user of this service via a geostationary satellite in Ka- band and the set up of the different components of this link and then to make the calculation of the link budget for this system. The application carried out in this work, allows us to calculate the link budget in both directions: the uplink and downlink, as well as all parameters used in the calculation and the development of a link budget. Finally, we will try to verify using the application developed the feasibility of implementation of this system.

***Keywords*—**Geostationary satellite, Ground station, Ka band, Link budget, Telecommunication

I. Introduction

IN the context of future satellite communications systems, the deployment of the Ka band is a requires, particularly because of the saturation of the L, C and Ku bands. This operation will provide the advantage of wider channels that support a greater number of users; it also allows reducing the size of the user terminal and antenna [1].

Adding to this that, the realization of a satellite meets a need which results in the definition of the objectives of the space mission. Thus for example a communications satellite is the product of needs expressed by users working in fields varied such as mobile telephony, television and internet by satellite, radio navigation, and systems of localization...Etc [2]. For this, and given the complexity and the cost of space projects, their implementation is divided into phases to have a good understanding and good control on the project.

The work presented in this article between in the first phase of the design of a satellite and which consists in the contribution to the analysis of mission of a telecommunications satellite for the internet or mobile phone by satellite for example, and that in geostationary orbit [3].

We will in what will follow, do the configuration of a system which consists of a link between a broadcast station and a user through a satellite in geostationary orbit, and then do the calculation of the link budget the latter in the order to see if this link can be achieved in the future.

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II. Problem Statement

The calculation of the link budget is a very important step in the design phase of any satellite in order to ensure the proper functioning of the latter up after the launch, our work is within this context, or we will set a link in Ka-band between a station of emission of service and a receiver (user) via a geostationary satellite and to ensure that the system normally works with these parameters we should do the calculation of the link budget in the end leaving a margin of error sufficient as a guarantee for the proper functioning of the system. We cannot take this margin large because this causes additional costs and an over-sizing of the system and a lesser margin can lead to an excessive error rate which may caused the loss of the bond, so it must adjust the parameters of entry until a margin, at least 8 dB [4] greater than the value of the quality of the link estimated, the calculation of the link budget consists in the determination of the ratio of signal to noise at the level of the satellite for the uplink and at the level of the reception station for the downlink, this report is given by the following equations [5]:

A. For the uplink

$$\left(\frac{C}{N_0}\right)_U = \frac{\dfrac{\left(\dfrac{P_{out,b}}{L_{feed,b}}G_{t,b}\right)G_{r,s}\big/L_{feed,s}}{L_f}}{kT_{s,s}} = \frac{\dfrac{\left(P_{t,b}G_{t,b}\right)G_{r,s}\big/L_{feed,s}}{L_f}}{kT_{s,s}} \quad (1)$$

$$= \frac{\dfrac{EIRP_b \cdot G_{r,s}\big/L_{feed,s}}{L_f}}{kT_{s,s}} = \frac{EIRP_b}{L_f}\left(\frac{G_{r,s}\big/L_{feed,s}}{T_{s,s}}\right)\frac{1}{k}$$

$$\begin{aligned}(C/N)_U &= P_{out,b} - L_{feed,b} + G_{t,b} - L_f + G_{r,s} - T_{s,s} - k - Ls \\ &= EIRP_b - L_f + G_{r,s} - T_{s,s} - Ls + 228.6 \quad dBHz\end{aligned} \quad (2)$$

B. For the downlink

$$\left(\frac{C}{N_0}\right)_D = \frac{\dfrac{\left(\dfrac{P_{out,s}}{L_{feed,s}}G_{t,s}\right)G_{r,b}\big/L_{feed,b}}{L_f}}{kT_{s,b}} = \frac{\dfrac{\left(P_{t,s}G_{t,s}\right)G_{r,b}\big/L_{feed,b}}{L_f}}{kT_{s,b}} \quad (3)$$

$$= \frac{\dfrac{EIRP_s \cdot G_{r,b}\big/L_{feed,b}}{L_f}}{kT_{s,b}} = \frac{EIRP_s}{L_f}\left(\frac{G_{r,b}\big/L_{feed,b}}{T_{s,b}}\right)\frac{1}{k}$$

$$\begin{aligned}(C/N)_D &= P_{out,s} - L_{feed,s} + G_{t,s} - L_f + G_{r,b} - T_{s,b} - k - Ls \\ &= EIRP_s - L_f + G_{r,b} - T_{s,b} - Ls + 228.6 \quad dBHz\end{aligned} \tag{4}$$

For the calculation of Lf losses mentioned in (2) and (4), say the losses in the free space that is a basic step in the calculation of a link of communication especially satellite in geostationary orbit because of the large distance between the satellite to Earth. Losses in the free space can be expressed by the following report:

$$L_f = \left(\frac{4\pi d}{\lambda}\right)^2 \text{(dB)} \tag{5}$$

At the same time, it is necessary also to take into account all sources of losses that can cause degradation of the link budget. Thus, it is affected by a set of losses that will degrade it, all sources of degradation are accumulated in the term Ls, mentioned in the equations (2) and (4), and it is defined as follows:

$$Ls = L_{Em}.L_{Atm}.L_{Pol}.L_{Poin}.L_{feed} \tag{6}$$

Among these sources of degradation, we find the losses due to the depointing antenna [6], noted by Lpoin and defined by the following equations:

$$L_{\theta_T} = 12.\left(\frac{\theta_T}{\theta_{-3dB}}\right)^2 \quad \text{(dB)} \tag{7}$$

$$L_{\theta_R} = 12.\left(\frac{\theta_R}{\theta_{-3dB}}\right)^2 \quad \text{(dB)} \tag{8}$$

We have the atmospheric losses (LAtm) due to the diverse atmospheric phenomena, we have [7]:

A. Absorption by oxygen molecules γo (dB/km):

$$\begin{cases} \left[\dfrac{6.6}{f^2+0.33} + \dfrac{0.19}{(f-118.7)^2+2}\right] f^2 10 & \text{For } f = 57\text{GHz} \\ 14.9 & \text{For } 57 \le f \le 63\text{GHz} \\ \left[2.10^{-4} r_t^{1.5}(1-1.2\times10^{-5} f^{1.5}) + \dfrac{4}{(f-63)^2+1.5 r_p^2 r_t^5} + \dfrac{0.28\ r_t^2}{(f-118.75)^2+2.84 r_p^2 r_t^2}\right] & \text{For } 63 \le f \le 350\text{GHz} \end{cases} \tag{9}$$

Where: f: frequency (GHz),
rp= p / 1013,
rt = 288/ (273 + t),
p: pressure (hPa),
t: temperature (°c).

B. Absorption by water vapor:

$$\gamma_W = \left[3.27.10^{-2} r_t + 1.67.10^{-3}\frac{\rho}{r_p} + 7.7\times10^4 f^{0.5} + \frac{3.79}{(f-22.23)^2+9.81 r_p^2 r_t} \frac{11.73 r_t}{(f-183.31)^2+11.85 r_p^2 r_t} + \frac{4.01.r_t}{(f-325.153)^2+10.44 r_p^2 r_t}\right] f^2 \rho r_t r_p 10^{-4} \tag{10}$$

Where: ρ is the density of the water vapor (g/cm3).

C. Attenuation due to the rain:

$$\gamma_R = k.R^{\alpha} \tag{11}$$

Where: K and α are coefficients which depend on the frequency and polarization.
R is the intensity of rainfall in mm/h.

D. Attenuation due to clouds and fog:

$$\gamma = \text{Af2M} \tag{12}$$

Where: γ : The weakening in dB/Km,
F: The frequency in GhZ,
M: The water content in g/m3,
A: Coefficient which depends on temperature.

We also have other sources of loss, such as:

1) LEM: Corresponds to the losses between the output of the transmitter and the antenna (line, duplexers, filters...).
2) Lfeed: Corresponds to the losses between the receiving antenna and the input of the receiver.
3) Lpol: Corresponds to the polarization losses from a bad adaptation of polarization between two antennas.

III. Context Of Study

For our study which consists in the configuration of a communication link for a multimedia service such as the internet by satellite for example between an transmit earth station (the one that offers the service) and receiver (user service) via a telecommunications satellite on geostationary orbit, we will take the case that is shown in figure 1 which consist in a link between a ground station equipped with a fixed parabolic antenna installed on the site of Arzew (Oran, Algeria) and a user found anywhere in Algeria via a satellite in geostationary orbit telecommunication at a defined position [8].

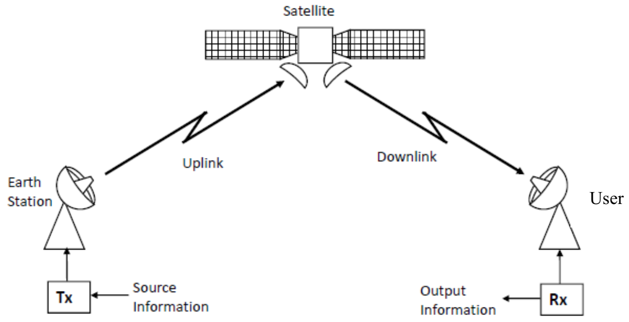

Fig. 1 Connection between ground station and user via geostationary satellite (Case of study).

IV. Working Methodology

A procedure for the design of a satellite link is given by the following steps [9]:

1) Choice of carrier frequency based on the availability and allocation of spectrum by the ITU.
2) Selection of the transmission powers.
3) Estimation of losses between the transmitter and the antenna.
4) Estimation the maximal depointing angle.
5) Calculating the gain of the antennas.
6) Calculation of free space losses.
7) Estimation of atmospheric absorption.
8) Estimation of the noise temperature of the system (clear sky).
9) Calculation of Eb/N0 for the data rate required.
10) Report search Eb/N0 required to satisfy the BER based on the type of modulation and coding.
11) Adding 1 or 2dB to compensate the errors of implementation. Calculation of the margin error of the link.
12) Calculation of the margin error of the link.
13) Adjustment of the input parameters until a margin of at least 8 dB greater than that estimated with degradation due to rain.

The margin of the system is given by:

$$M = \left(\frac{E_b}{N_0}\right)_{Calc} - \left(\frac{E_b}{N_0}\right)_{Req} \qquad (13)$$

With

$$\left(\frac{E_b}{N_0}\right)_{Calc} = \frac{1}{R} * \left(\frac{C}{N_0}\right) \qquad (14)$$

Where: R is the bit rate

If this margin is respected then the transmission may be made, otherwise it must either change the settings or resize some essential parameters to improve the quality of the bond.

V. Developed Software

The software developed in our study is designed under the environment Matlab 7.8; it is structured in four main parties (Figure 2), calculation of the UPLINK budget (Figure 3), and the calculation of the DOWNLINK budget (Figure 4), calculation of the depointing angle of satellite antenna, calculation of depointing losses. In addition, it has a menu that contains all the different calculations that fit into the development of the link budget as antenna settings, the Earth satellite distance, atmospheric losses, and the orientation of the antenna of the station … etc.

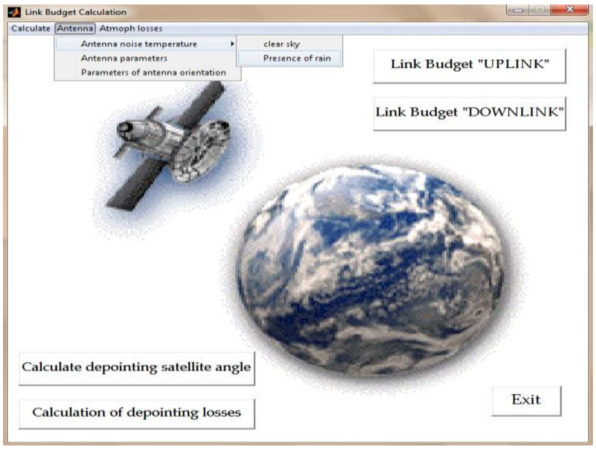

Fig. 2 Principal window of the application

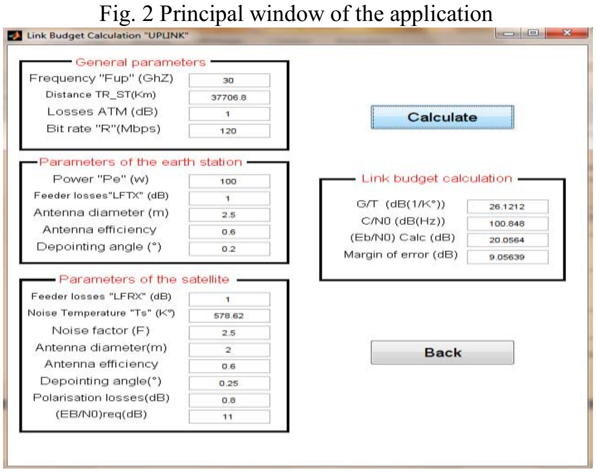

Fig. 3 Calculation of the uplink budget

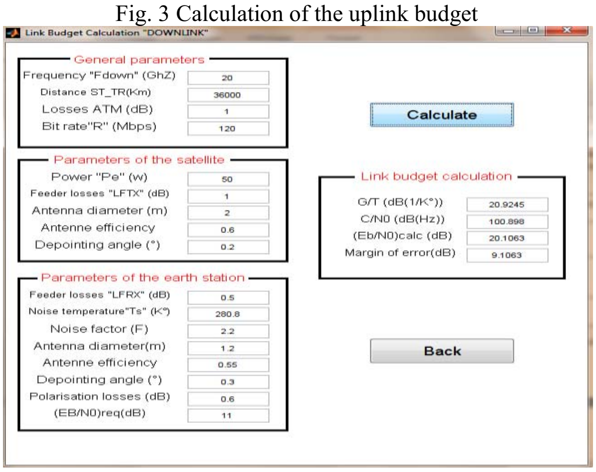

Fig. 4 Calculation of the downlink budget

VI. Simulation And Results

For any satellite link, we have a set of parameters that characterize it, for our case, we will configure a connection in Ka-band between a transmit earth station that is installed at the level of the city of Arzew (Oran, Algeria) and a user found anywhere in Algeria via a geostationary satellite and this for a multimedia application (Internet or Television by satellite, mobile phone).

Among these parameters, there are already presets such as: the coordinates of the station and the longitude of the satellite [9] to calculate the exact distance between the Earth and the satellite (this calculation is integrated in the application), the frequency…etc.

And we have parameters that we will define: the diameters of antennas, their gains, bit rate….etc.

Our contribution is to find an optimal combination between these different settings in the goal to establish a link with a margin of error quite sufficient (>8dB) [4], to ensure the proper functioning of our system. After several trials during what we have tried to take all the constraints into consideration (size, power, cost, access), we have arrived to the data summarized in the following tables:

A. For the uplink

TABLE I
PARAMETERS OF THE UPLINK CONNECTION

Parameters	Value
General data	
Frequency	30 GhZ
Longitude of the satellite	35.867° N
Latitude of the station	0.321° O
Longitude of the satellite	24.8°
Atmospheric losses	1 dB
Binary rate	120 Mbits/s
Polarization losses	0.8 dB
(Eb/No) req	11 dB
Earth station data	
Power	100 W
Feeder losses	1 dB
Diameter antenna	2.5 m
Antenna efficiency	0.65
Max depointing angle	0.25°
Satellite data	
Feeder losses	1 dB
Noise factor	2.5 dB
Diameter antenna	2 m
Antenna efficiency	0.6
Max depointing angle	0.2
T° noise of the system	578

B. For the downlink

TABLE II
PARAMETERS OF THE DOWNLINK CONNECTION

Parameters	Value
General data	
Frequency	20 GhZ
Distance earth satellite	40000 Km
Longitude of the	24.8°
satellite	1 dB
Atmospheric losses	120 Mbits/s
Binary rate	0.8 dB
Polarization losses	11 dB
(Eb/No) req	
Satellite data	
Power	50 W
Feeder losses	1 dB
Diameter antenna	2 m
Antenna efficiency	0.6
Max depointing angle	0.2°
Earth station data	
Feeder losses	0.5 dB
Noise factor	2.2 dB
Diameter antenna	1.2 m
Antenna efficiency	0.55
Max depointing angle	0.3
T° noise of the system	280

For both cases, we have taken a bit error rate “BER=10^{-7}” [10] and a Viterbie coding with QPSK modulation. With the parameters detailed in the table before and using our software, we have obtained the following results:

TABLE III
RESULTS OBTAINED WITH THE CALCULATION SOFTWARE

	UPLINK	DOWNLINK
Figure of Merit G/T (dB/K°)	26.12	20.92
Signal to noise ratio C/N0 (dBHz)	99.96	99.99
Margin of error (dB)	8.17	8.2

From this, we can see that we can set up a system consisting of a link between an Earth station broadcasting and a receiver via a geostationary satellite with the parameters of the table (1) and (2), because we can guarantee and ensure the proper functioning of our system and this through the margin of error that we have left, despite the fact that it is costly in terms of weight and power but it is essential to know the sensitivity of the link in band Ka to atmospheric disturbances especially rain.

Trajectory Design for Alpha CubeSat

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January 5, 2016

Summary

A trajectory design is described for the Alpha CubeSat mission. It satisfies the mission constraints of flight time, ΔV, and final lunar capture orbits. This design can be refined with numerical simulations. The total ΔV needed is 180 m/s, well within the capability of the mission. The total flight time is 315 days.

Description of Trajectory Design

Propulsion Capability

The spacecraft for Alpha CubeSat , we label SC, is assumed to have two types of propulsion systems. One is HTSD with available ΔV = .228 km/s and a LTLD with available ΔV = 1.334 km/s. This totals 1.562 km/s. (The LTLD is a Busek BIT-1 ion thruster using iodine – Isp = 1,200 s, Th = .4 micro Newtons. This yields **2.5 m/s per day** of continuous thrusting. It can produce **1.334 km/s** The HTSD thruster has an Isp =200s and Th = 1,400 Newtons. It uses N2O and aluminized paraffin. This yields **228 m/s)**

Starting Conditions (Earth Centered)

Time = 0 (a starting epoch)

Radial distance, rp(E) = 45,000 km

Velocity, vp = 4.19 km/s. This is provided by the launch vehicle for a piggyback payload and not by SC.

Apoapsis Condition(Earth Centered)

rp and vp place SC on a highly eccentric nearly parabolic escape trajectory, TE, from the Earth to an apoapsis distance, ra(E) = 4,000,000 km. This ellipse has an eccentricity at the start of e = .98. TE will be perturbed once SC reaches ra due to solar perturbations, but only slightly.

When SC reaches ra(E), it will be at an approximate apoapsis of an approximate highly eccentric ellipse where the radial velocity is approximately 0. At this location, the trajectory is turning around to return to the Earth.

The time of flight from rp(E) to ra(E) is approximately **166 days.** (see Figure 1)

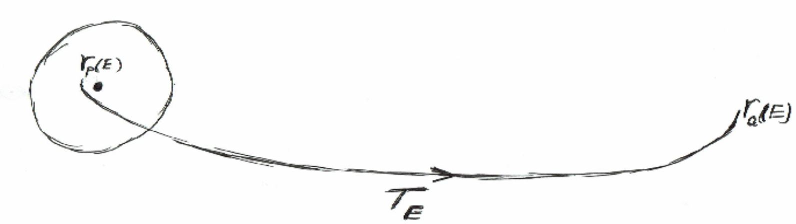

Figure 1. Escape Transfer to 4,000,000 km (inertial coordinates)

Targeting at ra(E) to the Moon

When SC is at ra(E) a maneuver is performed to target the trajectory to reach the Moon on a low energy trajectory that passes near the Earth-Sun L2 point. The energy is adjusted so that upon arrival near L2, SC lies near a stable manifold (a cylindrical tube in position-velocity space) that allows SC to move towards L2 vicinity with minimal energy(velocity), and then exit the L2 neighborhood with minimal velocity near an unstable manifold(another cylindrical tube in position velocity space). These tubes are connected at a halo orbit about L2. This allows the trajectory to move to the Moon with minimal energy and arrive near the Moon with the correct timing. In fact, when SC arrives near the Moon, it does so on a stable manifold to a region about the Moon where ballistic capture occurs – called a Weak Stability Boundary (WSB)[1,3].

The targeting at ra(E) is also done so that upon arrival at lunar periapsis the periapsis altitude, rp(M) is 500 km. The targeting maneuver at ra(E) is estimated to be **ΔV(ra) ≅ 12 m/s**. This achieves both the required plane change and lunar arrival conditions. The fact this maneuver is small is due to the large distance to the Moon and the fact that Earth-Sun L2 region and the lunar arrival state are in the WSB regions of the Earth and Moon, respectively. (see Figure 2)

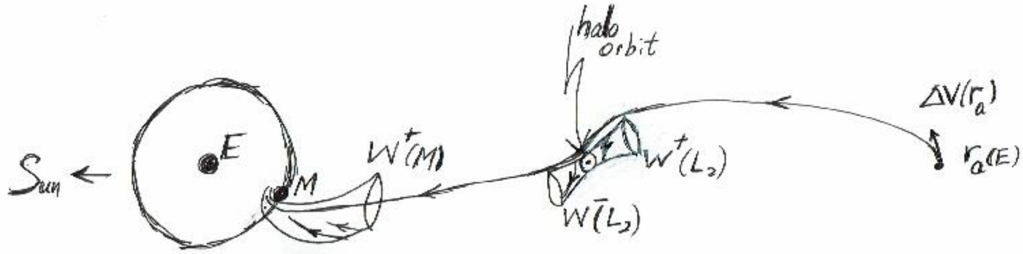

Figure 2. Trajectory from ra(E), passing near the stable manifold (W^+) and unstable manifold(W^-) of L_2 . These manifolds exist in position-velocity space and are shown here projected into position space, as an illustration. The trajectory is then guided to the Moon via another stable manifold, W^+, of the lunar WSB where lunar capture occurs for 0 ΔV. (Earth-Sun rotating coordinates)

As is described in [2], there exists a special family of orbits about the Moon in the WSB at this altitude, with an apoapsis altitude of ra(M) = 40,000 km. The initial osculating eccentricity is .89. These orbits, which are 500 x 40000 km in altitude are shown to be stable in [2] for at least one month where the orbital elements change by very little.

The remarkable thing about these orbits is that their periapsis exists in the WSB. This means that ballistic capture can occur, so that no ΔV is needed when the trajectory from ra(E) arrives at rp(M). That is, **ΔV1(rp(M)) = 0**. This condition is included when targeting from ra(E).

To satisfy the constraints of the Alpha CubeSat mission, ra(M) is lowered to 10,000 km by performing a maneuver at rp(M) of **ΔV2(rp(M)) = 118 m/s**. (see Figure 3)

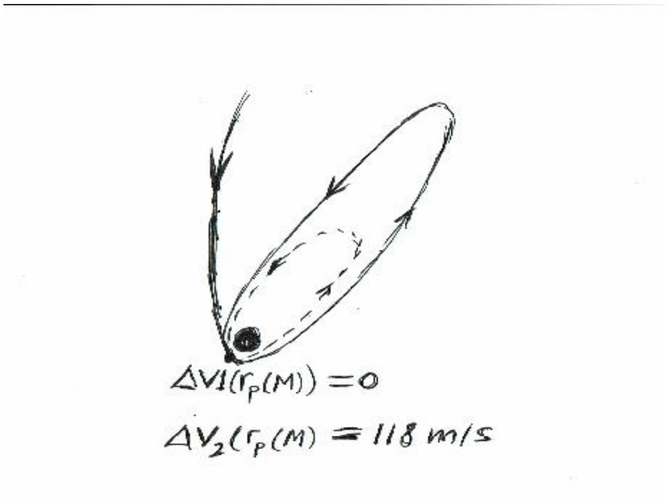

Figure 3. Arrival of trajectory into ballistic capture at rp(M) so that ΔV1(rp(M)) = 0. The initial osculating elliptical orbit has ra(M) = 40,000 km. A maneuver of ΔV2 = 118 m/s reduces ra to ra(M) = 10,000 km (dashed ellipse).

The time of flight from ra(E) to rp(M) is approximately **149 days.**

Course correction maneuvers may need to be made from the Earth to the Moon. The allocation for these is **ΔV(Corr) = 50 m/s.**

Summary

Total ΔV = ΔV(ra(E)) + ΔV1(rp(M)) + ΔV2(rp(M)) + ΔV(Corr) = 180 m/s

Total Flight Time = 315 days

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Belbruno Trajectory & Propulsion Capabilities Analysis

Developed by Ethan Chew (shinen.chew@gmail.com) for Alpha CubeSat GT-2 Propulsion Report on 2/5/16

Universe Constants

			Notes
g_0	9.8	m/s^2	For I_sp, Surface of Earth gravitational acceleration.

ACS Vehicle Definitions

			Notes
Vehicle Mass	14	kg	Per the competition requirements on maximum vehicle wet mass.
Max Accel Lim	9.8	m/s^2	Per the structural limitation of the ACS vehicle deployables for in-space manuevers.

Belbruno Trajectory Definition

From LEO-4Mkm			Notes
Vp	*3200*	*m/s*	Gray indicates provision by Launch Vehicle
LEO-4Mkm Leg Flight Time	166	days	
From 4Mkm-Lunar Intercept			
DeltaV(ra)	12	m/s	Black indicates provision by ACS Vehicle.
Entry to Lunar Orbit			
DeltaV1(rp(M))	0	m/s	Initial Entry into Lunar Orbit.
DeltaV2(rp(M))	118	m/s	Shaping of Lunar Orbit to meet NASA CubeQuest Competition Requirements.
4Mkm-Lunar Orbit Leg Flight Time	149	days	
Contingency			
DeltaV(Corr)	50	m/s	Reservation for correction maneuvers.
Total DeltaV	180	m/s	Required of ACS Vehicle.
Total Flight Time	315	days	

Propulsion System Requirements

For Propulsion Systems Operating at Maximum Thrust Limit

Propulsion System	Baseline Hybrid Propulsion System,	Phase 4 CAT (P4-50)	Ambip Tethers Unlim	Busek BIT-1 Ion Thruster		
Propellant	N2O-40% Aluminized Paraffin	Iodine	Water	Water	Iodine	
Thruster Quantity	1	1	1	1	4	
I_sp	200	506	1623	300	1200	s
Vehicle m_0	14	14	14	14	14	kg
From 4Mkm-Lunar Intercept						
Impulse(ra)	168	168	168	168	168	N-s
Maneuver Total Max Thrust	137.2	0.00272	0.00044	0.8	0.0004	N
Maneuver Individual Max Thru	137.2	0.00272	0.00044	0.8	0.0001	N
TimeThrust(ra)	1.224489796	61764.7059	381818.182	210	420000	s
Mass Propellant Used	0.085714286	0.03387916	0.01056245	0.05714286	0.01428571	kg
Vehicle End Mass	13.91428571	13.9661208	13.9894375	13.9428571	13.9857143	kg
% Leg Time	0.00%	0.43%	2.66%	0.00%	2.93%	
Entry to Lunar Orbit						
Impulse1(rp(M))	0	0	0	0	0	N-s
Maneuver Total Max Thrust	136.36	0.00272	0.00044	0.8	0.0004	N
Maneuver Individual Max Thru	136.36	0.00272	0.00044	0.8	0.0001	N

TimeThrust1(rp(M))	0	0	0	0	0	s
Mass Propellant Used	0	0	0	0	0	kg
Vehicle End Mass	13.91428571	13.9661208	13.9894375	13.9428571	13.9857143	kg
Impulse2(rp(M))	1641.885714	1648.00226	1650.75363	1645.25714	1650.31429	N-s
Maneuver Total Max Thrust	136.36	0.00272	0.00044	0.8	0.0004	N
Maneuver Individual Max Thru:	136.36	0.00272	0.00044	0.8	0.0001	N
TimeThrust2(rp(M))	12.04081633	605883.183	3751712.8	2056.57143	4125785.71	s
Mass Propellant Used	0.837696793	0.33233892	0.10378574	0.55961127	0.14033285	kg
Vehicle End Mass	13.07658892	13.6337819	13.8856518	13.3832459	13.8453814	kg
% Leg Time	0.00%	4.71%	29.14%	0.02%	32.05%	
Contingency						
Impulse(Corr)	700	700	700	700	700	N-s
TimeThrust(Corr)	5.102040816	257352.941	1590909.09	875	1750000	s
Mass Propellant Used	0.357142857	0.14116318	0.04401021	0.23809524	0.05952381	kg
Totals						
Impulse	2509.885714	2516.00226	2518.75363	2513.25714	2518.31429	N-s
Mass Propellant Used	1.280553936	0.50738127	0.1583584	0.85484937	0.21414237	kg
Burn Time (s)	18.36734694	925000.83	5724440.07	3141.57143	6295785.71	s
Burn Time (days)	0.000212585	10.7060281	66.2550934	0.03636078	72.8678902	days
% Total Flight Time	0.00%	3.40%	21.03%	0.01%	23.13%	
Volume & Mass Breakdown						
O/F Ratio	3	N/A	N/A	N/A	N/A	
Propellant Density						
Oxidizer	1222	N/A	N/A	N/A	N/A	kg/m^3
Fuel	1250	4933	1000	1000	4933	kg/m^3
Propellant Mass						
Oxidizer	0.960415452	N/A	N/A	N/A	N/A	kg
Fuel	0.320138484	0.50738127	0.1583584	0.85484937	0.21414237	kg
Propellant Volume						
Oxidizer (cm^3)	785.9373583	N/A	N/A	N/A	N/A	cm^3
Fuel (cm^3)	256.1107872	102.854505	158.358396	854.849368	43.4101705	cm^3
Oxidizer (U)	0.785937358	N/A	N/A	N/A	N/A	U
Fuel (U)	0.256110787	0.10285451	0.1583584	0.85484937	0.04341017	U
Total Propellant Volume (cm^3)	1042.048146	102.854505	158.358396	854.849368	43.4101705	cm^3
Total Propellant Volume (U)	1.042048146	0.10285451	0.1583584	0.85484937	0.04341017	U

“Delivering CubeSats to the Moon and Beyond with Electric Propulsion”

Interplanetary Small Satellite Conference

Pasadena, CA

28-29 April 2014

Busek Co. Inc.

BUSEK
Space Propulsion and Systems

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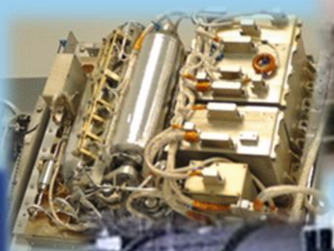
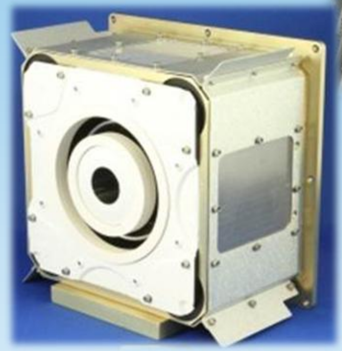
Busek Co. Inc.

Busek Co. Inc. is a leader in space propulsion systems development and manufacturing

- Core expertise begins with electric propulsion thrusters for military, government, and commercial satellites
- Expertise extends to space electronics, propellant feedsystems, and systems integration and testing
- Propulsion Technologies (thruster types) include:
    - *Hall*
    - *Electrospray (colloid)*
    - *Micro pulsed plasma*
    - *RF Ion*
    - *Microresistojet*
    - *Cold gas*
    - *Chemical (green monoprop)*



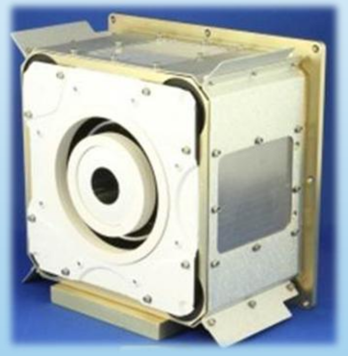
Overview

- Chemical Propulsion vs. Electric Propulsion
- Small spacecraft benefits, and limits and capability of propulsion
- CubeSat-scale spacecraft for a Lunar mission
- Propulsion-enabled ESPA-type spacecraft for Lunar and Mars CubeSat delivery
- Exposition of Busek propulsion offerings suitable for small spacecraft and ESPA missions

Chemical Propulsion vs. Electric Propulsion

Small chemical thruster (22N from AMPAC-ISP)

- Electric propulsion is much more fuel efficient than chemical propulsion
- EP has Specific impulse ~ 30X larger
- EP Results in significant spacecraft mass reduction or increase in capability



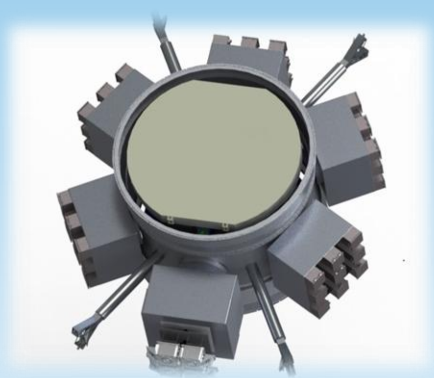
BHT-1500 Hall Thruster

Chemical Propulsion		Electric Propulsion
High thrust, low I_{sp}	vs.	Low thrust, high I_{sp}
T = Newtons and higher, typ.	vs.	T= microNewtons thru Newtons *only limited by available power*
Specific Impulse = $I_{sp} \leq 320$ sec	vs.	I_{sp} range from 500 - 10,000 sec
High propellant mass flow & low velocity	vs.	Low propellant mass flow & high velocity

BUSEK Space Propulsion and Systems

Why Small Satellites?

- Lower launch costs. Launch costs typically on a per kg basis
- Miniaturization of components and lower power requirements allow equal capability in a smaller platform
- Technological advancement allows lower cost capability, e.g. processors, solar panels
- Cheaper satellites allow for increased risk tolerance (reduced cost of losses), reduced redundancy, lowering costs further
- Lower cost = more missions.



NASA

TDRS

Small Satellites and Propulsion

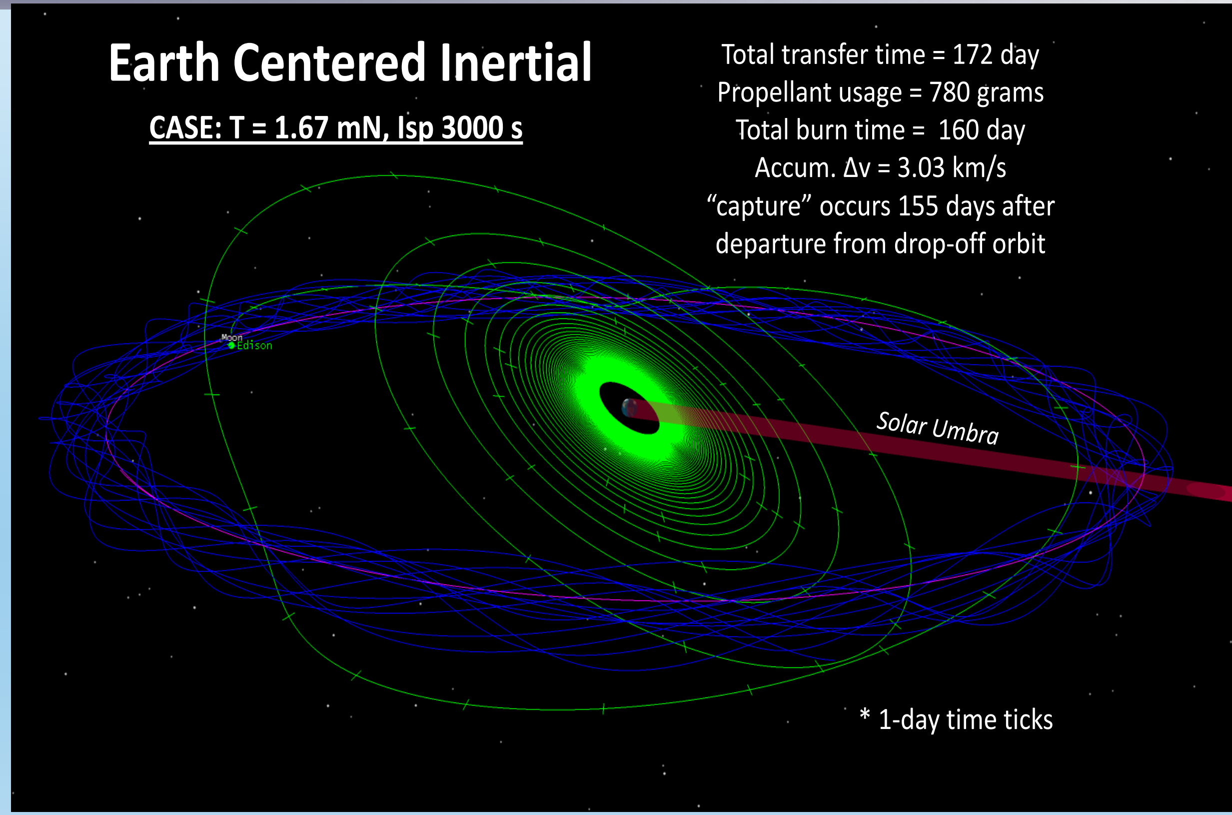
- While many satellite technologies scale favorably for small satellites, propulsion capability is limited by physics:
  - Propellant loading capacity is severely reduced
  - Mass fraction of propellant is relatively low
  - Propellant system dry mass is relatively high
  - Many thrusters cannot operate, or perform poorly, when scaled down
  - Power demands may exceed small satellite power availability
  - Inefficiencies may exacerbate thermal management challenges

Fewer propulsion technologies are suitable for small spacecraft, and selection drops off rapidly with decreasing size: Most chemical and electric propulsion limited by large dry mass. Chemical propulsion further limited by low I_{sp}, and electric propulsion often further limited by power demands.

Lunar Cubesat Mission

- ≈ 3km/s required to get to the moon
- Note propellant mass and I_{sp}
- Similarly, a 3kg (3U) spacecraft requires 300g propellant

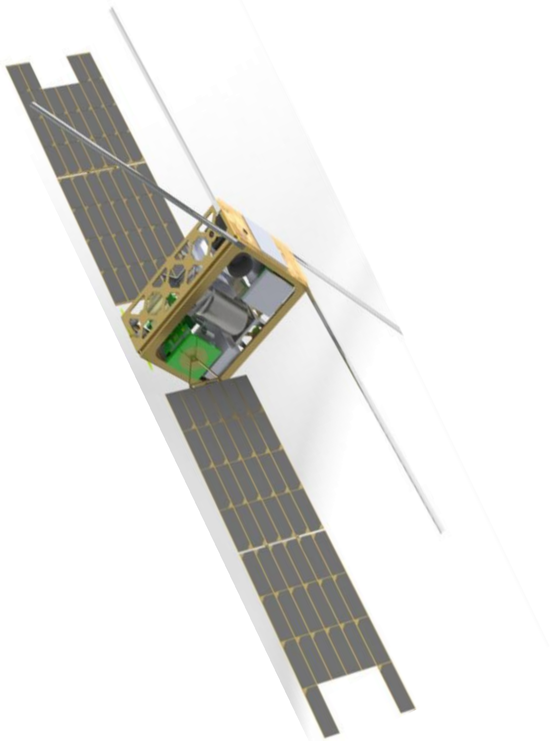
Lunar missions are possible with multiple propulsion technologies with appropriate system mass vs. I_{sp} tradeoffs



Lunar Cube trajectory from MEO to lunar intercept (green trace) and lunar capture/orbit (blue trace). (≈ 8kg s/c wet mass) Courtesy of JPL.

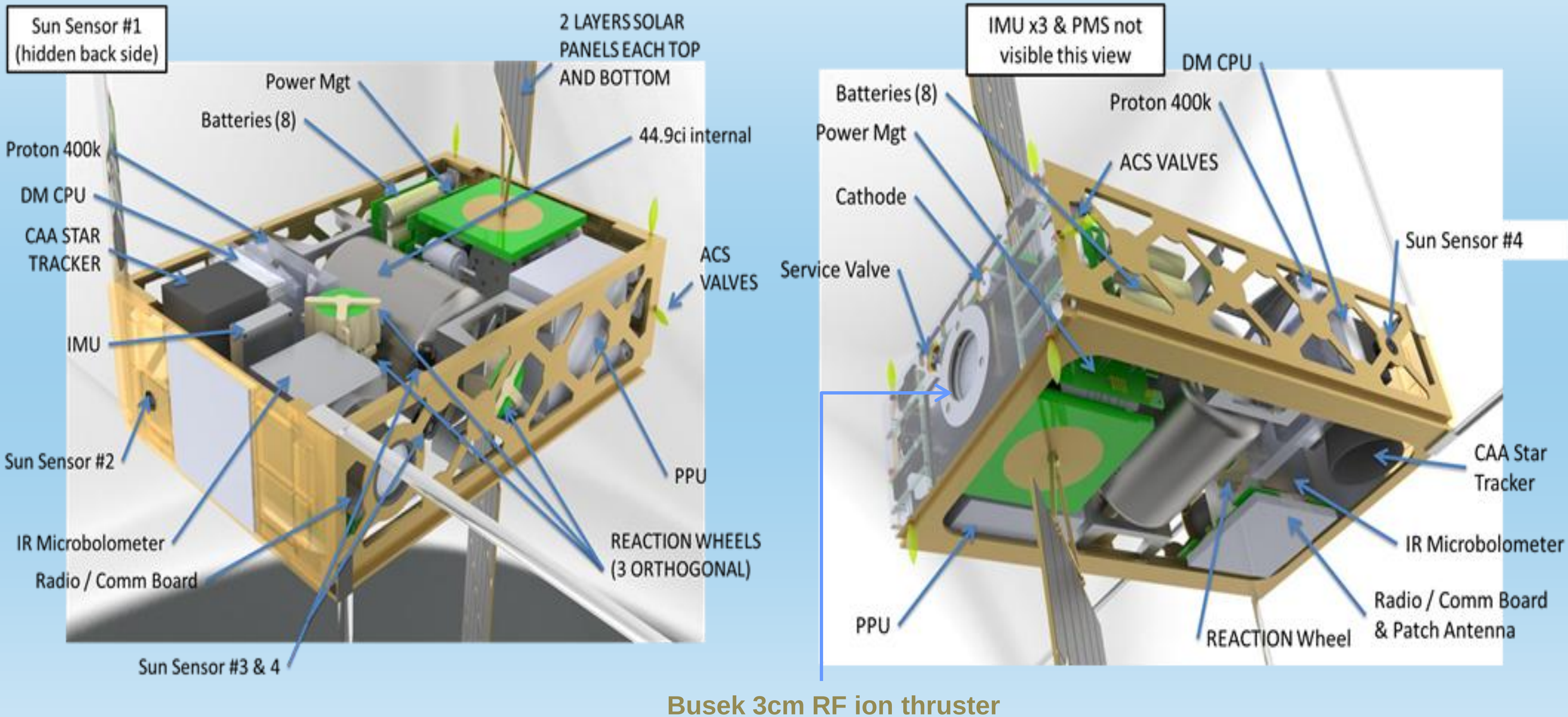
Prospective Lunar Cubesat System

Without the use of a larger platform as a carrier, CubeSats can go from Earth to Lunar orbit using on-board propulsion and still perform valuable science when they get there



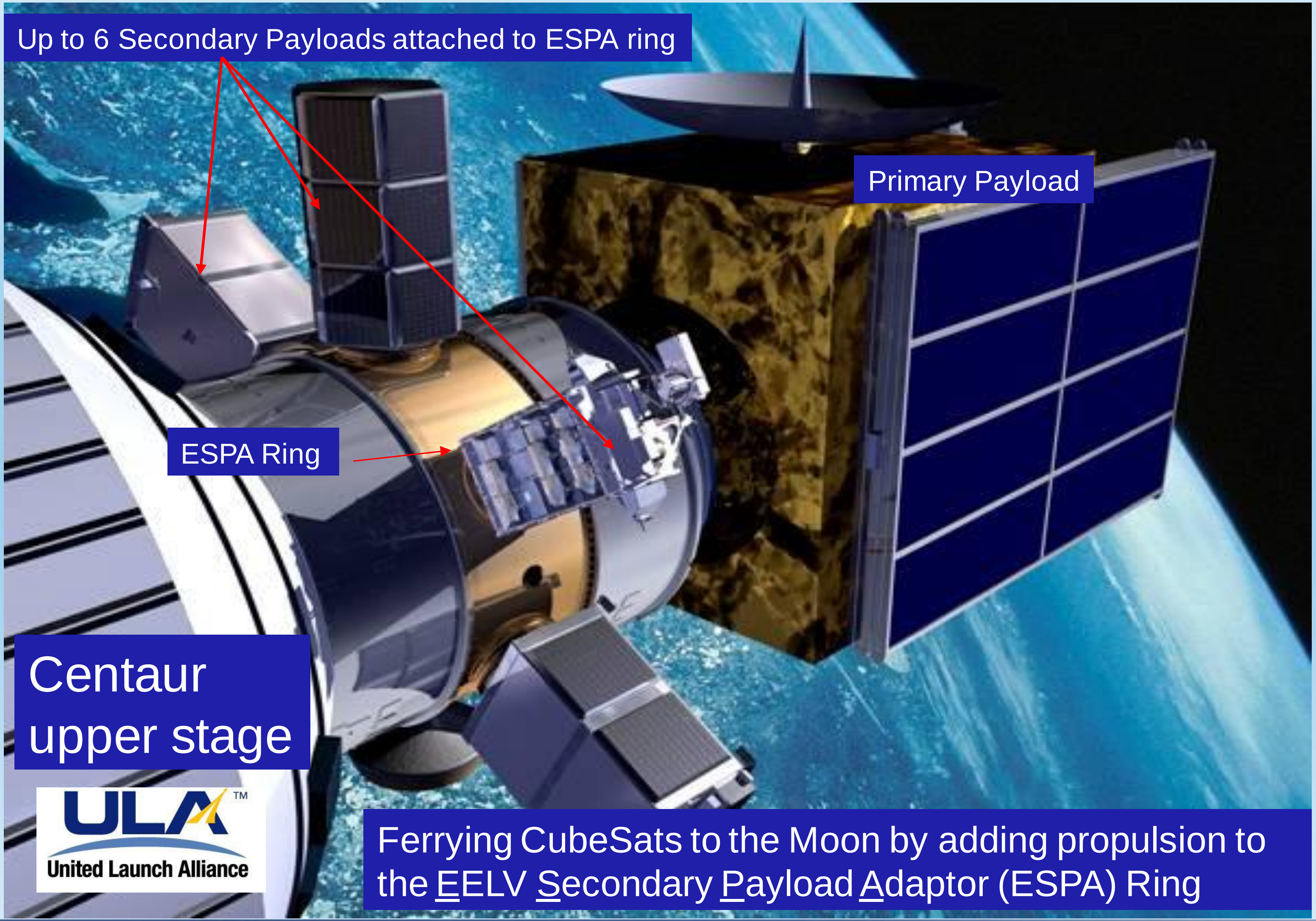
Property	Value
Mission	Demonstration of Lunar CubeSat
Initial Orbit	GPS (~20,000km)
Final Orbit	Lunar
S/C	6U CubeSat
S/C Mass	8kg
Peak Power	~96W
Propulsion	3cm RF Ion Thruster
deltaV	3.03km/s
Total transit time	~170days
Payload	Science Camera and Radiation Tolerant Computing

Lunar Cubesat Design Details



The 6U LunarCube concept is partially contributed by Morehead State University.

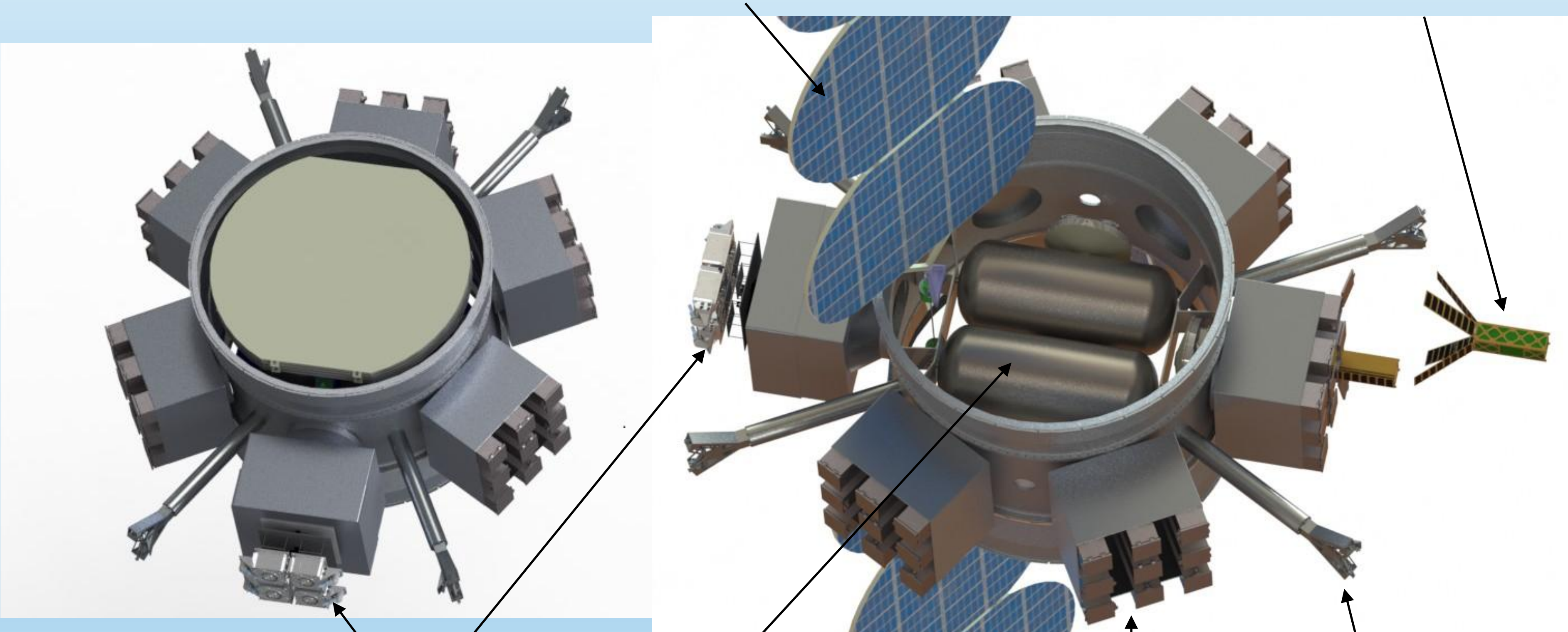
CubeSat "Lunar Ferry" via Propulsive ESPA



Propulsive ESPA Details

4kW Solar Array at BOL

Deployed 3U CubeSat



Propulsion Modules
Cluster of 4 BHT-1500, gimbal, PPUs, and flow control

Xenon tanks

5 Secondary Payloads
Each with 9 standard P-Pods (total 45x 3U CubeSats)

Cold-gas ACS Thrusters

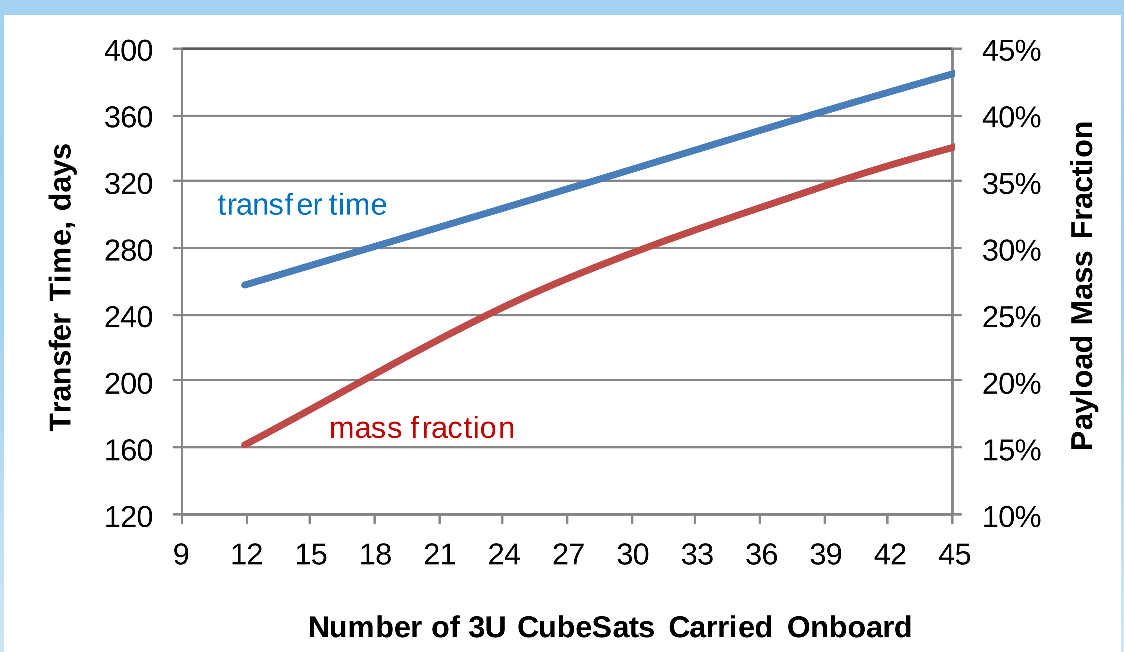
Propulsive ESPA Transfer Time

Mission:

- GTO (27º, 0.74 eccentricity) to lunar capture orbit
- ~3.7 km/s delta-V required

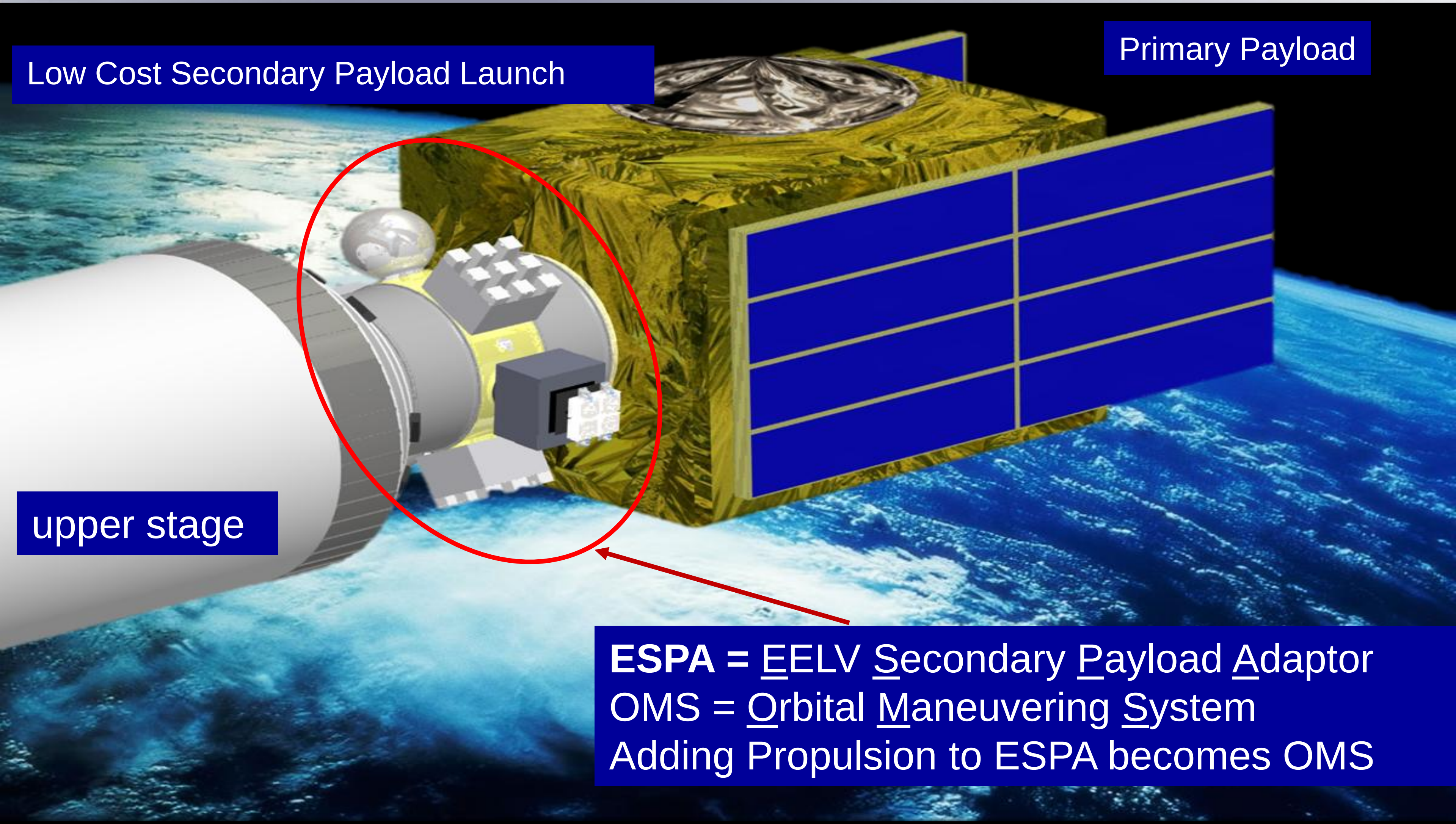
Propulsion:

- 4 Busek BHT-1500 Hall Effect Thrusters
- 237mN total thrust at 1640sec Isp



Transfer time as function of payload mass

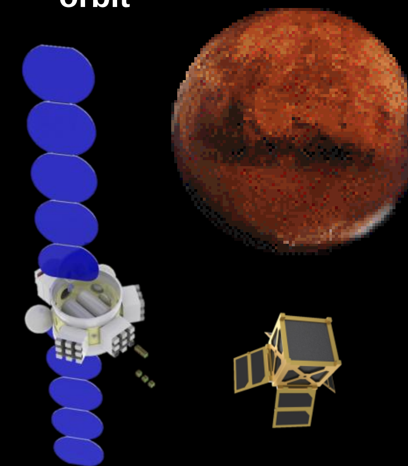
CubeSats to Mars carried by ESPA-OMS Carrier

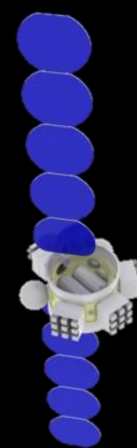


Mission Concept

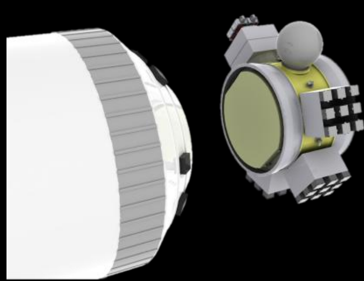
Solar panels are deployed and carrier begins the journey to Mars

CubeSats are deployed after entering Mars orbit

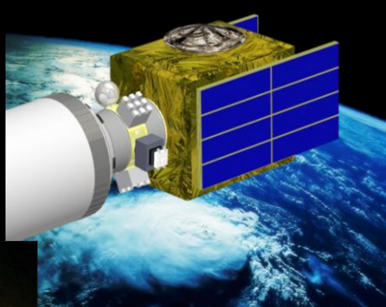




After primary payload release the CubeSat carrier released from the second stage



as secondary payload with GEO primary payload



Carrier for CubeSats launched

The CubeSat carrier or ESPA OMS using high efficiency propulsion can carry up to 27 – 3U CubeSats to Mars.

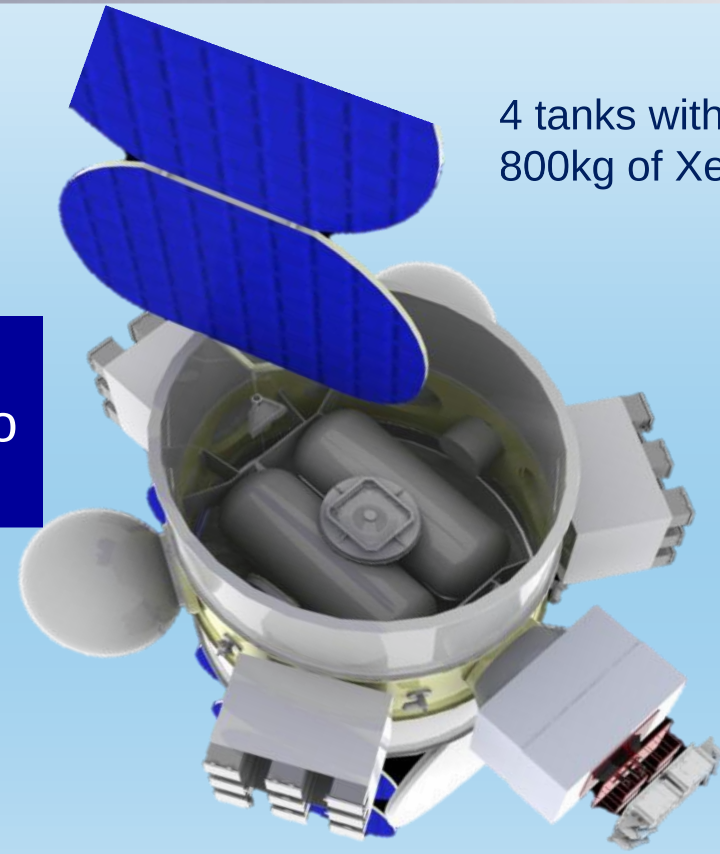
ESPA OMS Carrier delivers ~27 of 3U Cubesats to Mars and then serves as a communications relay back to earth

4kW array at BOL

HGA Antenna

27 P-Pods positions
Each can house up to
5U CubeSat

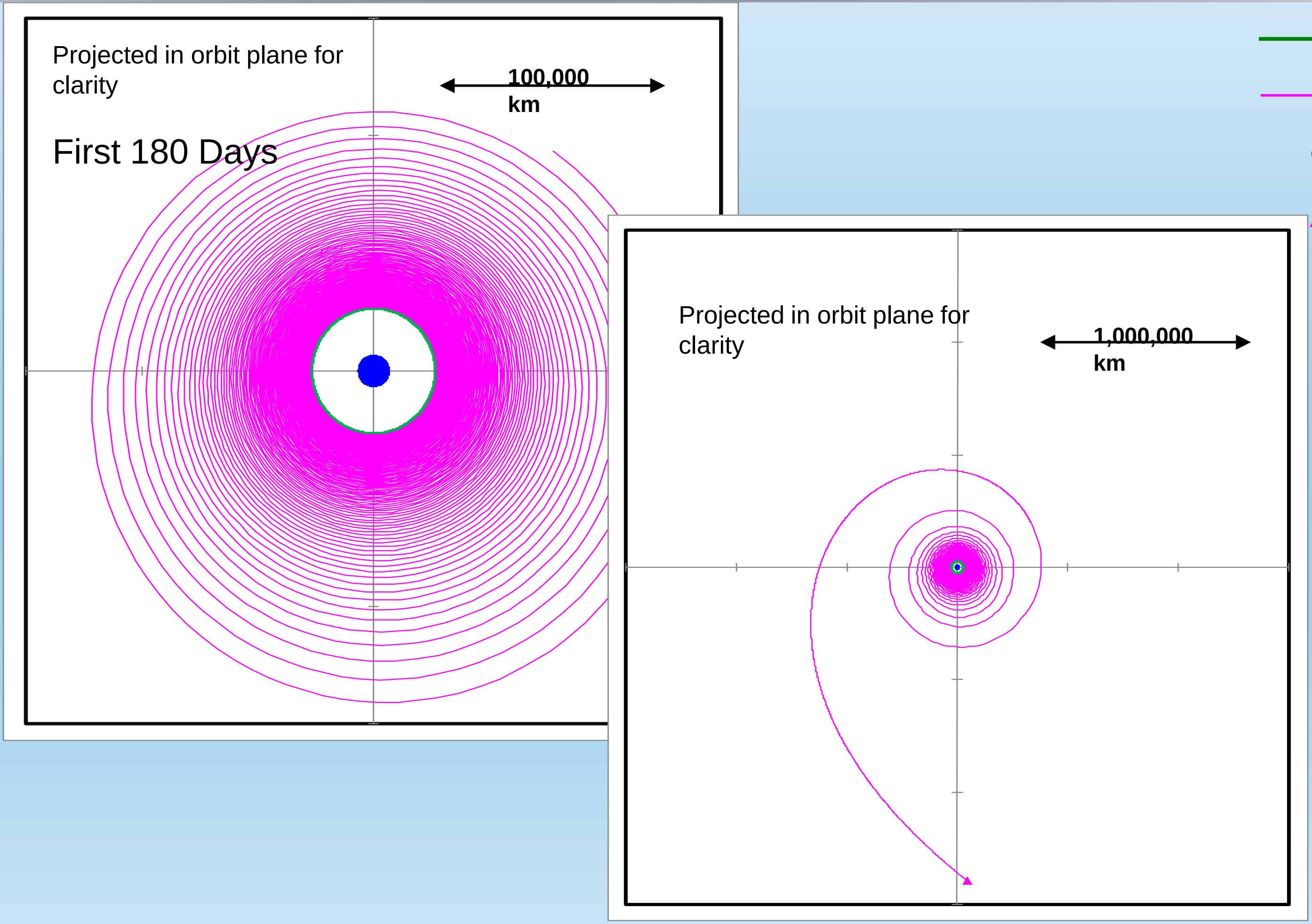
4 tanks with 800kg of Xe



Gimbaled Propulsion Module - Cluster of 4 Hall Effect Thrusters

Stimulating broad international participation, nations fly their own Cubesats to Mars

Earth Departure



GPS Orbit (plane B)

Powered Flight

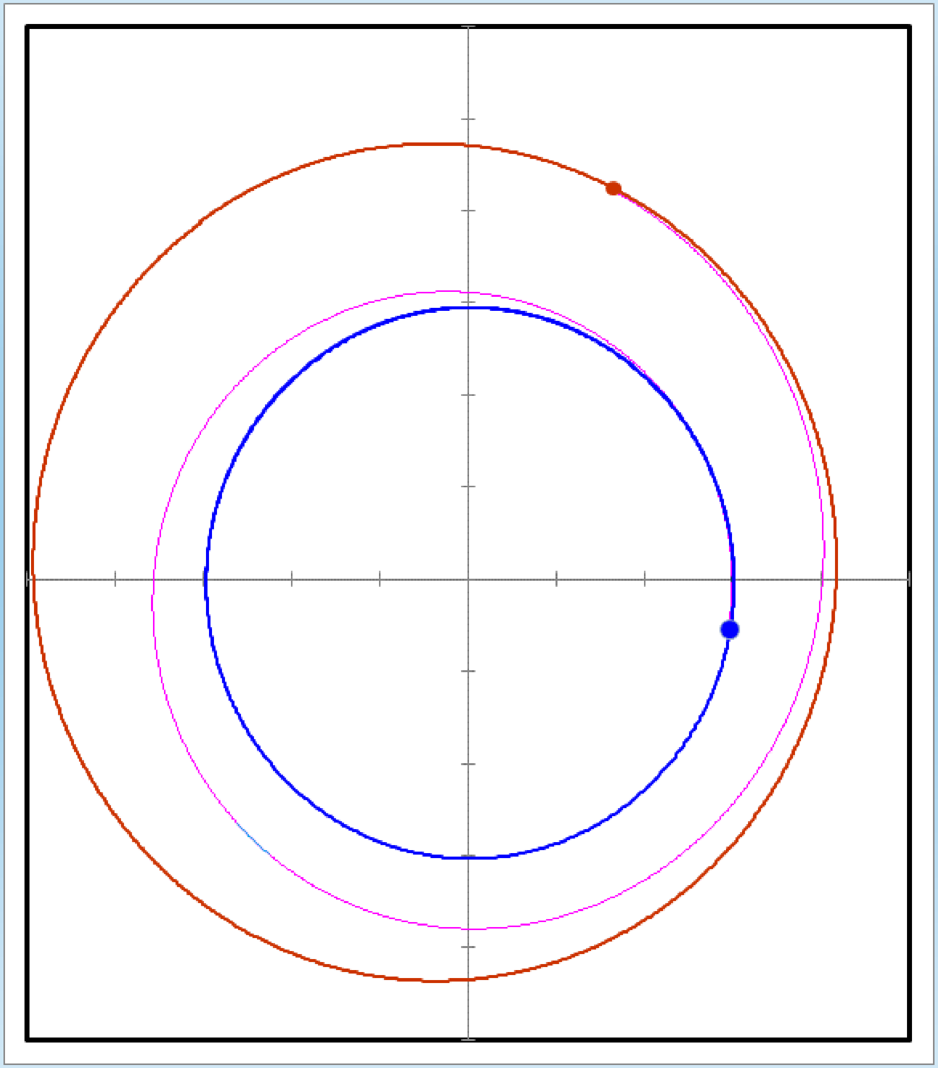
Earth

Escape Point (C_3 = 0)

- 4.35 km/s ΔV over 294 days
- Continuous low-thrust spiral orbit raise, concluding at C_3 = 0

294.1 days to Earth escape, 4.35 km/s ΔV, GPS parking orbit to C_3=0 escape

Interplanetary Trajectory



Earth Orbit

Mars Orbit

Powered Flight

Coast Flight

Earth @ Escape

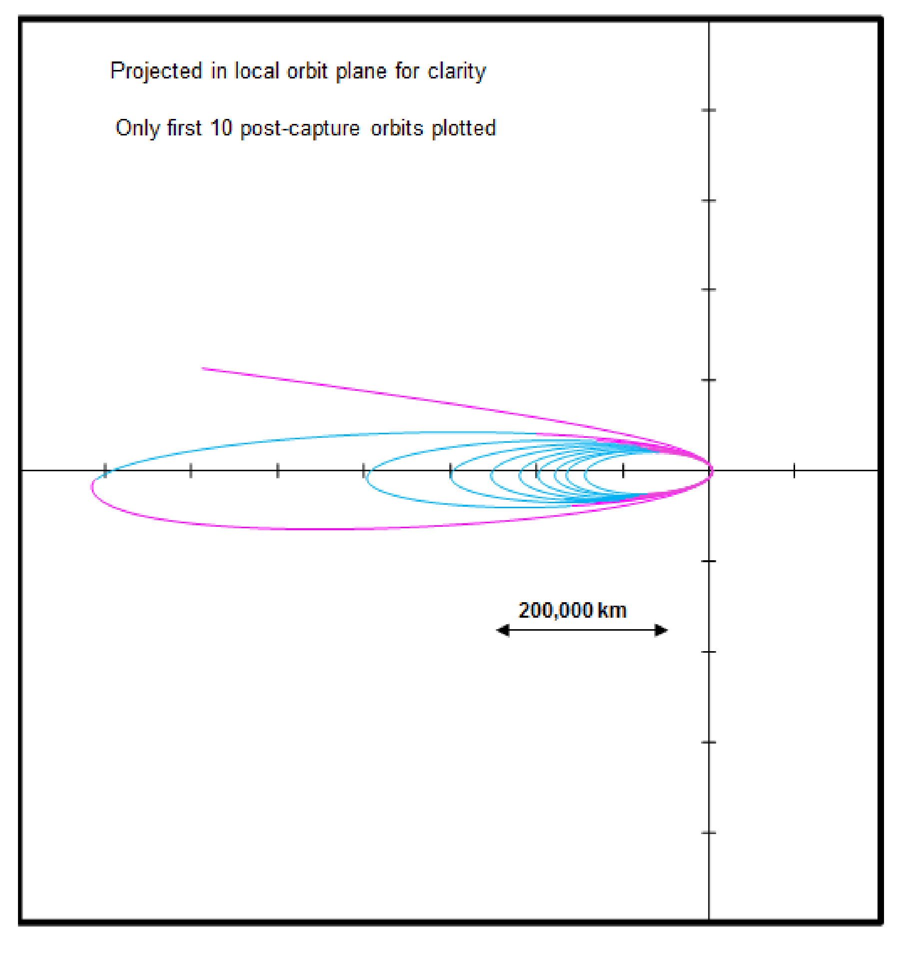
Mars @ Capture

- **604.1 days interplanetary cruise**
- **6.46 km/s ΔV**
- **Earth escape to Mars capture**

Powered flight uses paired thrusters @ 90% overall duty cycle:

- 10.8 hours thrusters 1/3 on
- 1.2 hours coast
- 10.8 hours thrusters 2/4 on
- 1.2 hours coast

Mars Capture



- **10 m/s cold-gas capture "burn"**
  (30.5 kg xenon expended)
- **341.3 days orbit lowering**
- **0.83 km/s low-thrust ΔV**
- **Apogee reduced to 50,000 km**

Phase 6: Mars Aerobraking

Apoapsis reduced to 650 km over 500 days

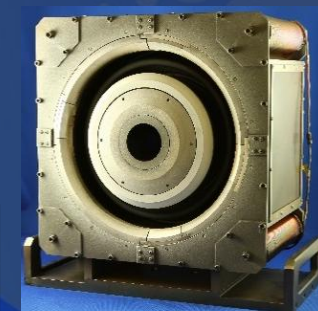
Phase 7: Circularization

0.15 km/s ΔV over 16 days
400 km circular orbit

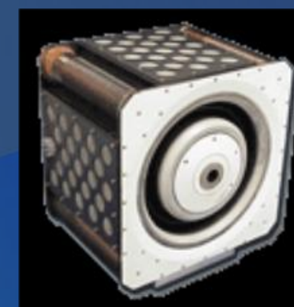
Busek Hall Thruster Technology

Busek is the Leader in Hall Effect thruster design and development technology with solutions from 100W to 20kW.

- All US Hall thrusters flown to date (BHT-200 to BPT-4000) are based on Busek technology
- Flight hardware provided for TacSat-2, FalconSat-3, LISA Pathfinder, FalconSat-5 and FalconSat-6 (current)
- Over 25 years of cutting-edge research, development and manufacture for government, academic and private customers



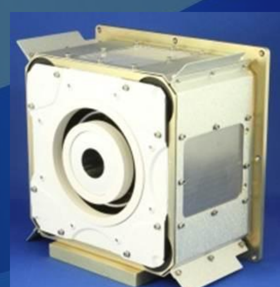
BHT-20K
Under development for NASA's Asteroid Redirect Mission



BHT-8000
Large GEO ComSats

BPT-4000 (Licensed technology)
GEO Comsats,

Hall Effect Thrusters – The ideal propulsion for orbit raising, station keeping, and de-orbit maneuvers.



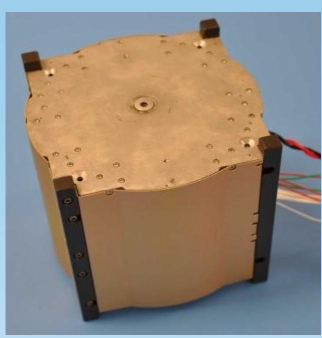
BHT-1500
Medium GEO ComSats

BHT-200
First US Hall Thruster to fly in space. TacSat-2.

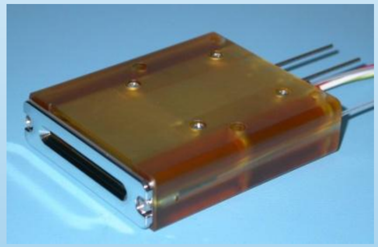
Busek's CubeSat Electric Propulsion Summary

Available 1U Package, <10W system power, ideal for missions at lunar orbit

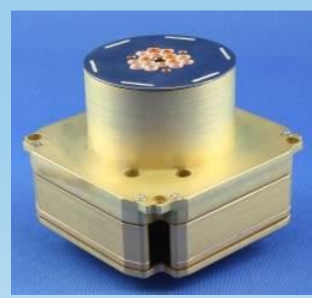
Electrospray Thruster
- ✓ High Efficiency
- ✓ Multi-emitter
- ✓ Low Risk/Technically Mature



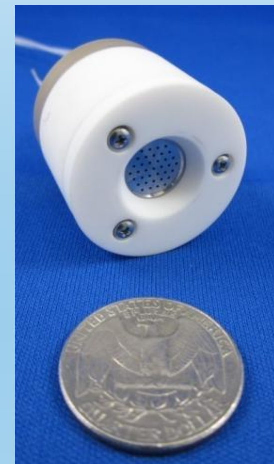
Micro Resistojet
- ✓ Simple, ideal for prox-ops
- ✓ Higher thrust
- ✓ Integrated Primary / ACS



Passive Electrospray Thruster
- ✓ No moving parts, valves
- ✓ No pressure vessel
- ✓ Low Power, high Isp



Micro Pulsed Plasma Thruster
- ✓ No moving parts, valves
- ✓ No pressure vessel
- ✓ Low Power
- ✓ Integrated Primary / ACS
- ✓ Prior version flown on FalconSat3



1 cm Micro RF Ion Thruster
- ✓ No internal cathode
- ✓ >2000s Isp
- ✓ FE Neutralizer is space qualified

50-100W system power, Capable of earth-moon transfer for a 6U s/c

3 cm Micro RF Ion Thruster
- ✓ No internal cathode
- ✓ Tested up to 3,000s Isp
- ✓ Higher thrust
- ✓ Thermionic Neutralizer is space qualified

Summary

- ✓ Small spacecraft deltaV limited relative to larger spacecraft, but Earth-to-Lunar missions feasible with ≈ 6U scale Cubesats with electric propulsion.
- ✓ Propulsive ESPA provides lower cost Lunar delivery of large quantities of Cubesats
- ✓ Propulsive ESPA provides interesting solution to Mars delivery of Cubesats by adding communications relay capability.
- ✓ Busek electric propulsion technologies are demonstrated capable of supporting such missions

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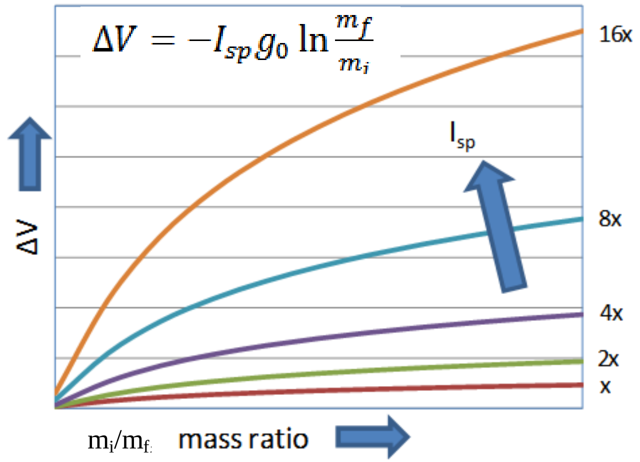
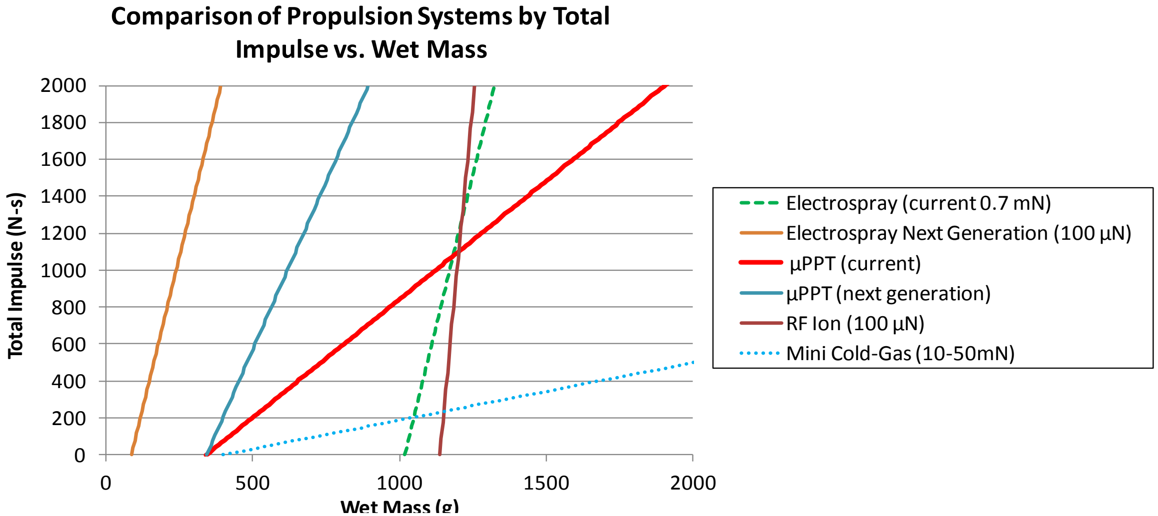
Backup Slides

Interplanetary Small Satellites Propulsion

The physical delta-V limits of small spacecraft are driven primarily by the increasingly unfavorable propellant mass fractions of small spacecraft, and secondarily by the more traditional metric of specific impulse (I_{sp}):





- While a large spacecraft may have a mass ratio of 5 or greater, total wet mass of a small spacecraft propulsion *system* will typically be less than 1/3 of total spacecraft mass.
- *Benefits of increased I_{sp} are often lost due to decreased mass ratio 'cost' of achieving said I_{sp}…*

(system requirements, valves, pressurized tanks, magnetics, thermal management, etc.)

Precision Proportional Flow Control Valve

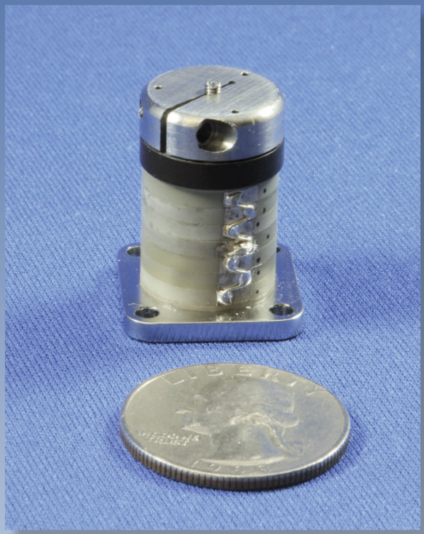
BUSEK
Space Propulsion and Systems

Low size, mass, power, sub-miniature PFCV for precision propellant management and space applications

Busek's 2 x 2 x 2.5cm miniature valves are the result of over 10 years of pioneering research and development, enabling new classes of CubeSat and NanoSat missions. These precision valves are next-generation versions of valves developed for the ST7 flight program.

The miniature valve has been designed to work with ionic liquids, but it is capable of precisely metering many other common propellants, including cold gas.

- Leak rate < 10e^{-5} mbar-L/s
- Components are designed to withstand > 500 PSI input pressure
- Design heritage from Lisa Pathfinder mission (delivered 2008)
- Low power: < 40mW
- Low mass and volume: 35 g and <10cc
- TRL 6



Busek's Normally Closed Proportional Flow Control Valve

Busek Co. Inc specializes in providing complete electric space propulsion systems including but not limited to a wide range of thrusters, propellant management systems, power processing units and digital control interface units. Busek provides analytical, computational, experimental and product services to government and industry.

Precision PFCV
Technical Specifications

Electrical

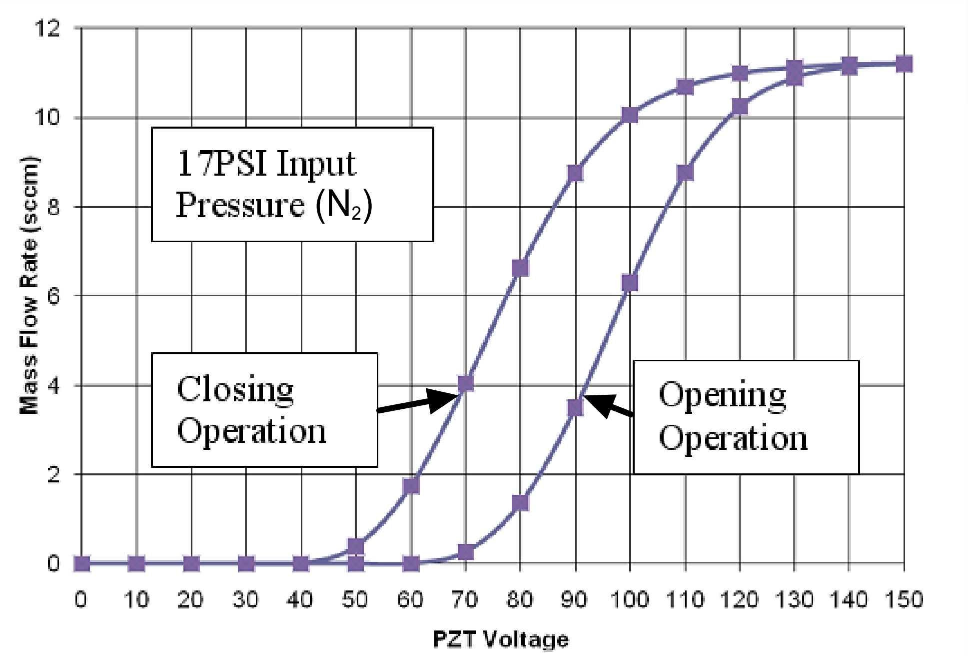
Valve Power	40 mW
Input Voltage	0 to + 200 VDC

Mechanical

Valve Mass	35 g
Valve Dimensions	2.0 x 2.0 x 2.5 cm

Performance

Control Resolution	Measured 2.5 pL/s resolution (μ = 0.0175 cP, Δp = 15 PSI), which translates to better than 0.005% resolution.
Heritage	LISA Pathfinder Disturbance Reduction System (ST7-DRS), design heritage
TRL	6



Hysteresis is inherent to the piezo actuator and is repeatable. Mass flow rate is calculated as a function of valve excitation and valve direction.

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70008501G

GPIM AF-M315E Propulsion System

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Aerojet Rocketdyne, Redmond, WA, 98073

Chris McLean[4]
Ball Aerospace and Technologies Corporation, Boulder, Co, 80301

The NASA Space Technology mission Directorate's (STMD) Green Propellant Infusion Mission (GPIM) Technology Demonstration Mission (TDM) will demonstrate an operational AF-M315E green propellant propulsion system. Aerojet-Rocketdyne is responsible for the development of the propulsion system payload. This paper statuses the propulsion system module development, including thruster design, system design and system component materials compatibility testing. Major system components of the propulsion system module include: propellant tank, latch valve, service valve and thruster valve. All system components, except the thruster valve, are flight proven (TRL 9) for hydrazine propellant; Status is given on modifications of these components to ensure that all internal wetted surfaces are compatible with the AF-M315E propellant.

The culmination of this program will be high-performance, green AF-M315E propulsion system technology at TRL 7+, with components demonstrated to TRL 9, ready for direct infusion to a wide range of applications for the space user community.

Nomenclature

EM	=	Engineering model
ESPA	=	EELV secondary payload adapter
GPIM	=	Green Propellant Infusion Mission
HAN	=	Hydroxyl ammonium nitrate
I_{sp}	=	Specific Impulse
IHPRPT	=	Integrated High Payoff Rocket Propulsion Technology
SCAPE	=	Self-Contained Atmospheric Protection Ensemble
TRL	=	Technology Readiness Level

I. Introduction

For four decades, monopropellant hydrazine systems have been the dominant propulsion technology for low-total-impulse applications; however, expensive storage, handling, and disposal procedures are required to address the propellant toxicity and flammability hazards, which, though well established, continue to hinder efforts to reduce mission integration costs and schedule. While traditional green alternatives such as cold gas and electric propulsion may reduce schedule and cost impacts, their limited specific impulse and thrust respectively preclude their application to missions requiring high total impulse and/or thrust. As such, the last decade has seen a growing awareness that the development of a low-toxicity alternative offering performance better than hydrazine would yield substantial crosscutting benefits to NASA and all space users. Toward this objective, the NASA Space Technology Mission Directorate (STMD) has initiated the Green Propellant Infusion Mission (GPIM) program with the objective of completing the first on-orbit demonstration of a complete AF-M315E high-performance (+50% density–I_{sp} compared to traditional hydrazine) green propellant propulsion system by the end of 2015. Hosted on a Ball Aerospace BCP-100 ESPA-class spacecraft bus, the GPIM Technology Demonstration Mission (TDM) will employ an Aerojet-developed advanced monopropellant payload module as the sole means of on-board propulsion,

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[1] Manager of Programs, Advanced Development, Aerojet-Rocketdyne, Redmond, WA.
[2] Chief Engineer, Advanced Development, Aerojet-Rocketdyne, Redmond, WA.
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performing a comprehensive battery of performance characterization and capabilities assessment maneuvers using both 1N and 22N thrusters[1,2,3,4,5]. The 1N and 22N thrust classes representing the largest segments of the monopropellant thruster market, see Figure 1). Although current planning calls for the on-orbit segment of the TDM to be completed within three months, the specific intent of the GPIM program is to advance AF-M315E technology to a readiness level suitable for immediate infusion in both short-duration and extended near-future applications. The propulsion system under development incorporates principally heritage hydrazine system components selected for the long-duration compatibility of their materials of construction with the new propellant.

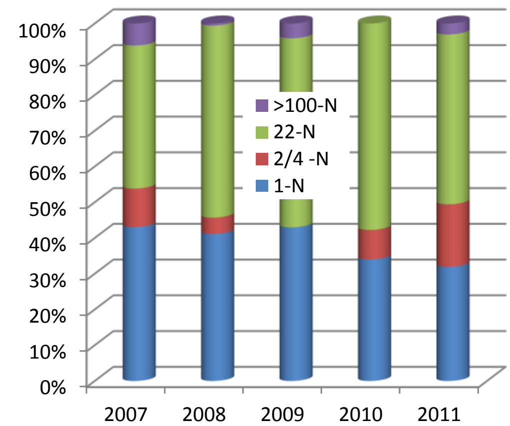

Figure 1 Market share by thrust level, 2007-2011

Aerojet Rocketdyne's commitment to green propulsion has spanned two decades and a wide range of propellant options. Initial experience was gained with HAN/glycine and HAN/methanol formulations[6]. Shifting focus to AFRL-developed AF-M315E ionic liquid advanced monopropellant in 2001, Aerojet Rocketdyne's green thruster technologies had matured to TRL5 by 2011, meeting the IHPRPT Phase II objective of 50% increased density-Isp over conventional hydrazine equivalents. Unique among a number of hydrazine alternatives that have emerged in recent years, AF-M315E is sufficiently green to enable safe handling in open containers for unlimited durations, whereas the properties and/or handling hazards (such as super-atmospheric vapor pressure or necessary stabilizers which may evaporate) of other current low-toxicity candidates preclude this. The summation of numerous development efforts and programs over many years, 2011 saw the first successful demonstration of more than 11.5 hrs firing life by an AF-M315E thruster employing a breakthrough patent-pending high-temperature catalyst (operated at near full thrust throughout), heralding readiness for infusion into a wide range of NASA, DoD, and commercial missions.

II. Payoff to NASA, Commercial and DoD Missions

NASA science missions place a special premium on performance, cost, robustness, and thermal requirements, all of which are enhanced by the use of GPIM's AF-M315E propulsion technology. AF-M315E offers higher performance than hydrazine, yields 12% higher I_{sp} (257 vs. 235 sec), and is 45% more dense (1.47 vs. 1.00 g/cc), affecting both reduced propellant and tank mass. A recent study showed significant benefits could be realized by using a high-performance, long-life hydrazine replacement for all of the three principal mission recommendations of the New Worlds, New Horizons in Astronomy and Astrophysics Decadal Survey (WFIRST, LISA, and IXO).[7] The study found that an AF-M315E system would reduce the propellant mass of WFIRST by >160 kg, about 10%, with a corresponding reduction in system dry mass (due to reduced tankage) of >30%. Other case studies in the report illustrate similar percent-wise benefits for missions in lower energy HEO and LEO orbits. Aerojet Rocketdyne estimates that an AF-M315E-based descent stage on the Mars Science Laboratory would have enabled 58 kg increased landed mass for the 930-kg rover compared to the hydrazine system that was flown. In addition to reduced test and loading costs owed to its low toxicity, AF-M315E simplifies the safe design and development of propulsion systems compared to hydrazine. Since leakage of AF-M315E is rated as a critical rather than catastrophic failure, only single-fault-tolerance is required for safety in handling flight systems. This alone accounts for significant savings, as redundant components are eliminated, yielding simpler architectures. Further, simpler and much less expensive design and verification criteria govern flight-qualification of fracture-critical hardware (e.g., propellant tanks) for non-hazardous propellants such as AF-M315E compared to hydrazine. The aggregate potential impact of these and increased performance-related cost savings is highly mission-dependent, but has been evaluated to tens of millions of dollars for large space missions such as JUNO, MSL, and Europa; and to several million for more modest missions such as GRAIL and MRO[8].

With its lower minimum temperature threshold, AF-M315E yields an additional advantage of mitigating operational concerns related to long-duration system thermal management. Whereas hydrazine space tanks and lines must be heated at all times to prevent freezing, AF-M315E cannot freeze (it has a glass transition). During long coast periods an AF-M315E propulsion system may be allowed to fall to very low temperatures and later reheated for operation without risk of line rupture by phase-change-induced expansion. This can be particularly beneficial to planetary spacecraft and planetary ascent vehicles, which can call for years of propellant storage in cold environments. For >1 AU interplanetary exploration missions, solar power is naturally more limited than for Earth-orbiting satellites; Equivalent solar power generation designs in Mars (e.g., MRO), Vesta (e.g., Dawn), and Jupiter (e.g., JUNO) orbits produce roughly 43%, 16%, and 3.7% of the electrical power they yield in Earth orbit, respectively. Tests also have demonstrated AF-M315E to has a significantly reduced sensitivity to adiabatic compression than hydrazine.

AF-M315E also offers comparable performance (density$-I_{sp}$) to traditional storable bipropellants for low ΔV missions while employing roughly half the number of components, thereby retaining the well-established increased reliability and reduced cost of traditional monopropellants. Many design issues and failure modes associated with long-duration interplanetary missions (e.g. control of mixture ratio, of propellant vapor diffusion and reaction, oxidizer flow decay) do not apply to an equally capable AF-M315E system.

The cost savings of green propellants associated with simplified range operations are quantifiable. The average contractual cost to load a NASA mission with conventional propellants is $135,000[8]. The cost for loading with AF-M315E will be a small fraction of this, and the associated schedule significantly expedited. Per current conventions, propellant loading operations require one shift for setup in SCAPE, a second shift waiting for propellant test confirmations, a third shift or more for actual loading, and a final additional shift to break down the setup, during which all remaining launch processing staff must wait at costs exceeding $100k/day for a typical Class B NASA mission. Thus elimination of the interruption of launch processing associated toxic propellant loading can save more than $100k per launch and two shifts of schedule. Naturally, it follows that simplified range operations would equally benefit commercial users through lower launch costs. An early Aerojet Rocketdyne study evaluating replacement of hydrazine with a HAN-based advanced monopropellant for Centaur RCS on an Atlas launch vehicle concluded ground support costs of fueling could be reduced by two-thirds[9].

III. GPIM Propulsion System

Under development as a self-contained module to allow independent assembly at Aerojet Rocketdyne for subsequent integration into the bus, the GPIM demonstration payload, illustrated in Figure 3 and shown in schematic in Figure 2, will deliver 50% more impulse than a comparably-packaged hydrazine system. Designed to attach to the Ball Aerospace BCP-100 bus via its standard payload interface plate (PIP), the GPIM demonstration payload comprises a simple, single-string, blow-down AF-M315E advanced green monopropellant propulsion system employing four 1N attitude-control thrusters and a single 22N primary divert thruster. The propellant feed manifold's principal components, consisting of a standard diaphragm propellant tank, latch valve, and service valves, represent all flight-proven (TRL 9 with hydrazine propellant) designs selected specifically for the long-term compatibility of their materials of construction with AF-M315E. Redundant pressure transducers monitor gas-side propellant tank pressure (and hence propellant consumption). Thrusters are mounted on the upper deck of a box-like payload primary structure. The 22N primary divert thruster is mounted on the spacecraft centerline with the thrust axis pointed through the PIP-mounted propellant tank and spacecraft centers of mass. The four 1N thrusters are

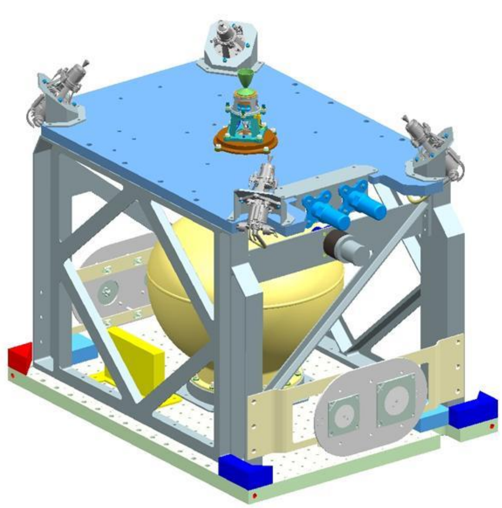

Figure 3 AF-M315E Propulsion System

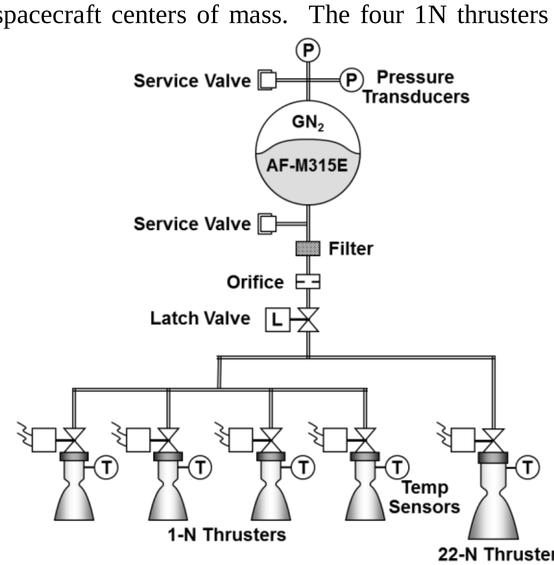

Figure 2 Propulsion System Schematic

canted on brackets at the corners of the upper deck to maximize the moment arm to the spacecraft center of mass, and thereby control authority and resolution of impulse measurement by the bus attitude and orbit determination and control (AODC) sensors. The remaining propulsion system components are consolidated on a component panel attached to the underside of the upper deck, except for the two service valves, which mount to a separate bracket positioned for easy access during fueling and range operations.

Design considerations for the AF-M315E propulsion system are mostly similar to a traditional hydrazine system, with a few special considerations. Principally, all system components must be compatible with AF-M315E, and therefore system component selections must strongly take compatibility into account, especially for longer duration missions. As the general schematic layout is identical to single string blow-down hydrazine systems commonly employed on small spacecraft, many of the same general design guidelines apply. The AF-M315E system however, is far less hazardous than a traditional hydrazine system when considering range safety requirements. The propellant is far less prone to leakage (due to higher viscosity), is non-toxic if leaked, and the thrusters cannot inadvertently fire without having first preheated catalyst beds. Initial discussions with KSC range safety personnel have consequently indicated the likely eventual acceptance of a reduced hazard severity classification of "critical" and possibly even "marginal" per MIL-STD-882E (Standard Practice for System Safety). In contrast, hydrazine external leakage is ranked a "catastrophic" hazard rating. Per Range Safety AFSPCMAN 91-710 requirements, a classification of "critical" or less only requires a two-seal inhibits to external leakage; hence no additional latch valves other isolation device are required in the feed system despite the fact that the advanced monopropellant thrusters employ only single-seat valves (for reasons that will be explained in Section IV). This approach reduces the complexity, power, and mass of the thruster valve, while simplifying electrical interfaces, all without sacrificing mission reliability.

Other differentiating design considerations arise principally from differences in the thermal characteristics of AF-M315E vs. conventional thrusters. Due to the advanced monopropellant thrusters' elevated minimum start temperature, catalyst bed preheat power requirements are higher compared to a conventional hydrazine system. This increase is partially offset, however, by the reduced power needs of the thrusters' single seat valves, as well as much lower power required for system thermal management during non-operating periods enabled by the propellant's demonstrated storage stability very low temperatures (although current CONOPS for the GPIM mission call for the propellant to be maintained within nominal system operating range). Radiation and conduction from the advanced monopropellant thrusters' high temperature chambers also impart a moderate increase in the thermal load to the system mounting interface.

IV. AF-M315E Green Advanced Monopropellant Thrusters

The Aerojet Rocketdyne 1N (GR-1) and 22N (GR-22) advanced monopropellant thrusters to be employed on GPIM represent the culmination of over two decades of research, spanning the development of enabling high-temperature test and data acquisition techniques applied to testing of a number of candidate propellants, extensive evaluation and test of numerous material systems for structural components and catalysts, and thruster performance characterization ranging from less than one up to 670 N (150 lbf) thrust in both sea-level and vacuum environments. Throughout a large portion of over two decades of research, inherently high reaction temperatures associated with ionic liquid propellants, coupled with poorly understood ionic-liquid thruster stability dynamics, constrained both thruster life and operational duty cycle capabilities. The last several years, however, have yielded significant breakthroughs related to both materials and a fundamental understanding of the governing mechanics of ionic liquid thrusters necessary to design and fabricate robust, practical (duty-cycle-unlimited) thrusters with sufficient life capability to meet real mission needs. A key, albeit by no means exclusive, contributor to the rapid acceleration in maturation of AF-M315E thruster technology seen in recent times has been the advent of Aerojet's patent-pending LCH-240 high-temperature long-life catalyst, demonstrating sufficient endurance within the propellant's decomposition/combustion environment to extend thruster life over 15× compared to the prior state-of-the-art.

The GR-1 and GR-22 advanced monopropellant thrusters implement a common design strategy whereby the use of refractory alloys (to accommodate the flame temperature of the AF-M315E propellant) is confined to the thrust chamber, nozzle and an upper thermal isolation structure, such that much of the thruster can be fabricated with conventional alloys in common use on hydrazine thrusters today. Trade studies indicate this hybrid approach yields significant respective cost and power savings compared to evaluated alternatives entailing either all-refractory or bulkier, heavily-insulated conventional alloy construction. The resulting flight thruster designs, shown side-by-side for comparison in Figure 4, comprise a series-assembled valve, injector, catalyst-containing chamber, and nozzle bearing general resemblance to conventional catalytic hydrazine thrusters of corresponding thrust classes, with two readily notable differences.

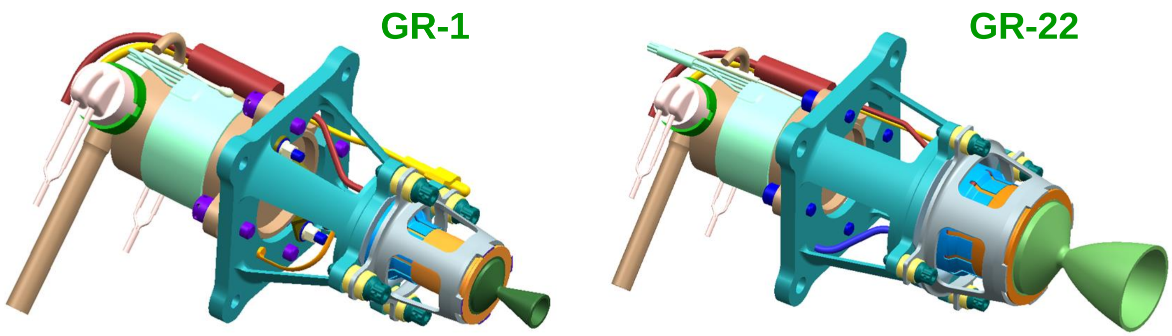

Figure 4 Aerojet GR 1 and GR 22 Thrusters

Most immediately apparent are the extended two-piece stand-off structures employed by both designs. These provide additional thermal isolation serving the dual roles of preventing overheating of the spacecraft interface by heat soak-back from the chambers during and following extended thruster firings, as well as limiting heat loss from the catalyst bed during thruster preheating, thereby minimizing power necessary to preheat the catalyst bed to the nominal start temperature. The stand-off structure employs a bolted mechanical joint as the primary interface between refractory and lower-temperature-capable conventional alloys, wherein a series of thermal spacers provide an efficient means to achieve the high temperature step-down necessary to implement a compact, highly thermally isolating, assembly. In accordance with engineering best practices, the GR-1 and GR-22 thruster designs incorporate redundancy on all fracture-critical structural elements, including both portions of the mounting structure and thermal stand-off and their conjoining fasteners (as well as at the control valve-to-thruster, and thruster-to-spacecraft mechanical interfaces). As dynamic load specifications imposed for both thrusters comprise up-to-date composite spectra developed by Aerojet to ensure broad utility of new/upgraded hydrazine thrusters designs, the GR-1 and GR-22 will be readily infusible into most applications likely to employ conventional monopropellants.

The GR-1 and GR-22 thrusters also employ notably smaller, single-seat valves with higher net reliability than the two-seat scheme generally favored for comparable hydrazine thrusters. This results from an inadvertent benefit inherent to specific properties of the ionic liquid propellant. Being more viscous than hydrazine, AF-M315E is intrinsically far less prone to leakage, such that the doubled risk of a thruster becoming inoperable in the event of either of two valve stages becoming inoperable is not justified. Moreover, having essentially no vapor pressure, AF-M315E will not self-pressurize or evaporate through small fissures such as a flaw in a valve seat, such that, in the very unlikely event that thruster valve leakage should occur, isolation of the downstream feed system by closing the upstream system latch valve would fully prevent any loss of propellant. Likewise for launch range operations, the innate safety of the propellant, accounting for its low vapor toxicity, and inability to activate un-preheated thrusters or react with external system and immediate work environment materials (unlike hydrazine), obviates the conventional rationale for the use of dual seat thruster valves. Thus, single seat valves provide higher mission assurance at lower mass, power (partially offsetting added preheat power requirements), and cost solution for the GPIM and future missions. Further, the added compactness of the GR-1 and GR-22 designs realized through the selection of single-seat valves has proven substantially facilitating in the close packaging of the GPIM demonstration system module, portending similar benefits to future ESPA-class spacecraft. Note that single seat valves have been used on many hydrazine-propelled spacecraft, and particularly prior NASA missions such as Cassini, Deep Impact, New Horizons, and Voyager (still successfully operating since its launch in 1977).

Technically, it is possible to complete the GPIM demonstration's planned three-month on-orbit life using conventional hydrazine thruster valves. Nevertheless, with a view to maximizing immediate infusability of the technology into both short-duration and extended missions, AF-M315E-specific material compatibility requirements (which differ from hydrazine) have been addressed in the selection of control valves for the GR-1 and GR-22 thrusters. Unlike for the upstream GPIM propellant feed system, where it was possible to simply select flight-heritage hydrazine components readily usable with AF-M315E with little or no modification, no such option exists for these new thrusters. In particular, as a mild acid, AF-M315E demonstrates long-term compatibility with a limited set of metals, none of which are ferromagnetic. Thus, the GR-1 and GR-22 thrusters employ largely new valve designs incorporating AF-M315E compatible wetted surfaces. The valves still derive considerable design and manufacturing process heritage from flight-proven products. Indeed, the GR-1 and GR-22 valve designs leverage existing process capabilities developed specifically for other applications necessitating isolation of valve ferromagnetics from working fluids.

The ongoing GPIM flight thruster development effort is structured in three overlapping phases. The first will execute early (June 2013) sea-level testing of heavyweight hardware derived from parallel preliminary flight thruster design activities. This testing will first perform duty cycle mapping of (principally the 22-N) thruster over a comprehensive range to verify broad functional stability, thereafter to anchor thruster life models as operated at duty cycles and simulated feed pressure blow-down ratio closely approximating projected mission performance requirements. Extensive thermal instrumentation will also yield detailed data to be used to anchor thermal models and optimize flight-thruster designs. Guided by test results, flight thruster designs will be completed in Phase 2. Engineering models (EM) of both the 1N and 22N thrusters will be fabricated and incorporated into a breadboard feed system functionally equivalent to the GPIM flight propulsion module for high-altitude protoflight testing. In Phase 3, flight designs will be finalized and flight (one each) qualification units fabricated. All thrusters will undergo standardized acceptance testing, comprising shock, vibration, and a check-out hot-fire. Qualification units will thereafter be subjected to qualification-level shock and vibration loads, followed by a mission-representative life test. On orbit, the thrusters will perform a series of maneuvers designed to both fully characterize thrust, Ibit, specific impulse, and thermal performance over a variety of duty cycles intended to encompass the full needs of near-future space applications.

Thruster Performance

Designed as functional alternatives to Aerojet Rocketdyne's 1N class MR-103G and 22N class MR-106L, thrust vs. feed pressure characteristics for the GR-1 and GR-22 are presented in Figure 5, with key operating metrics summarized in Table 1.

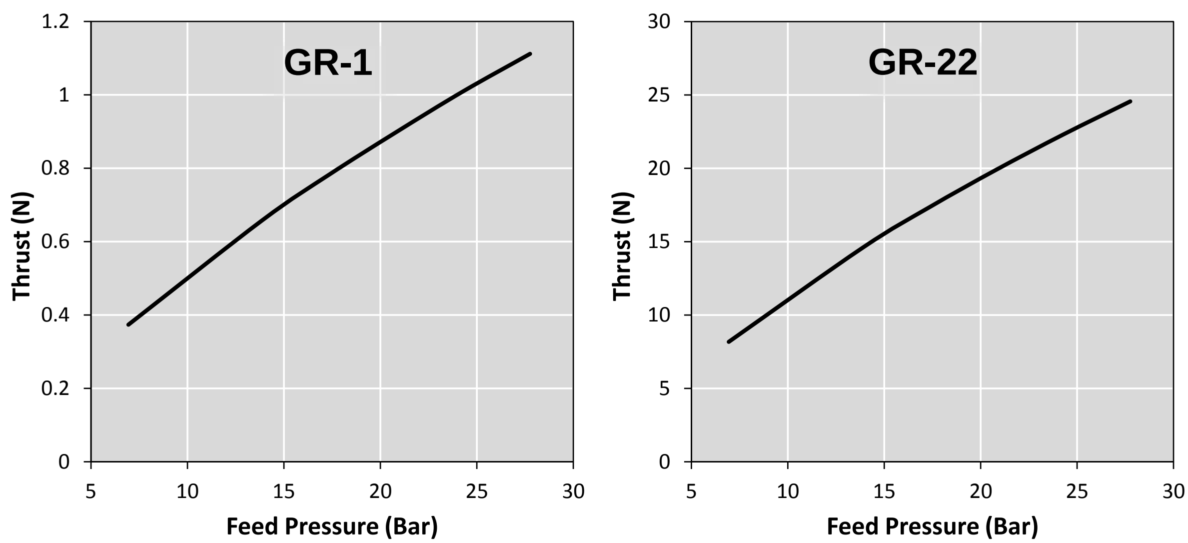

Figure 5 Aerojet Rocketdyne GR-1 and GR-22 Thrust vs. Feed Pressure

Table 1 Thruster Predicted Performance Summary

	GR-1	GR-22
Thrust (N)	0.4 - 1.1	8 - 25
Feed Pressure (bar)	6.8 - 27.6	6.8 - 27.6
Nozzle Expansion Ratio	100:1	50:1
Valve Power (W)	12	28
Preheat Power (W)	10	30
Specific Impulse (s)	235	250
Total Impulse (N-s)	23,000	74,000
Minimum Impulse Bit (mN-s)	8.0	116

V. Materials Compatibility Testing

AF-M315E propellant is acidic which can result in leaching of some common aerospace materials with long term propellant exposure. In addition, this fuel can act as both a reducing agent or as an oxidizing agent, so establishing metal passivation is more difficult than for pure reducing (hydrazine) or pure oxidizing (nitrogen tetroxide) propellants. Laboratory studies of this propellant inevitably show that for some test materials, it leaches metal ions. Nonetheless, safe, long-term storage of AF-M315E propellant in metallic and non-metallic tanks has been demonstrated[10]. For service components – valves, filters, elastomers, and lubricants – there is a small, but growing set of materials where laboratory testing indicates sufficient compatibility for at least 3-5 year missions[11].

A major effort of the GPIM program is to mature and qualify all AF-M315E propulsion system components for this mission, and for infusion on future space missions. An extensive materials compatibility test campaign is currently underway to confirm that all materials in system components that are wetted with the AF-M315E propellant are fully compatible, or material replacement of known incompatibles. The thruster valve requires the most extensive modifications to ensure it is AF-M315E compatible. All wetted surfaces for the thruster valve, service valve, latch valve and propellant filter will be manufactured from materials which are fully compatible with this propellant. The service valves being updated requires minor changes to all the sealing subcomponents. The latch valve is being evaluated to determine if any modifications to its materials is required. The system filter is the only system component that does not require any changes since its propellant wetted surfaces are already compatible.

Preliminary tests at elevated temperature revealed that the propellant tank elastomeric material met AMS-R-83412A specification requirements for compatibility. A longer term exposure test is currently being performed to determine any decrease in material functional properties and metal leaching profile over time. Latch valve components in test are: poppet seal, spring, and torque tube. For the service valve, the ball seal material, and back-up-ring are in test. The thruster valve seal elastomer material, is likewise in evaluation. In the very near future, common component materials, O-ring material and lubricants will be tested.

VI. Technology Maturation Status

As can be seen in the propulsion system schematic of Section III, AF-M315E-based advanced monopropellant systems are functionally equivalent to hydrazine systems, comprising the same number and type of components, but are distinct in that the different propellants have different material compatibilities. Historically AF-M315E and similar propellants have suggested only short duration compatibility with many common aerospace materials[12]. However, more recent accelerated aging tests performed under contract on Aerojet's Post-Boost program indicate AF-M315E to have similarly good long-term compatibility with a wider range of common aerospace materials, such that a large portion of existing flight-proven components are suitable for use with AF-M315E, although some elastomers (e.g. valve seats) may still require substitution (Component TRL status and required modifications are tabulated in Table 2). The available data provide high confidence that appropriately-selected flight-proven hydrazine components represent a low-risk option for the proposed TDM, and likely for future missions of at least five years and potentially longer. Feed systems similar to that planned for the proposed TDM are currently regularly flown in monopropellant and bipropellant applications where contamination by conventional propellants from propellant lines and components represents an unacceptable mission risk, such as A2100 and the Solar Dynamics Observatory spacecraft recently completed at NASA GSFC.

Liquid propellants similar to AF-M315E have been studied for over twenty years. Aerojet Rocketdyne has been a partner in this work and has participated in many material compatibility and propellant characterization studies. Aerojet Rocketdyne's assessment of propellant compatibility is based on long-standing experience of hydrazine compatibility testing. The topic of material compatibility immediately branches into two sub-topics: 1) the effect of the material on the propellant and 2) the effect of the propellant on the material.

- **Propellant Tank**

The propellant tank maturation approach is designed to maximize future mission infusion potential by emphasizing proven components and processes, while focusing only upon those areas required to achieve the GPIM goals to minimize cost and schedule risk. The program has shown that the shell material of the selected tank has long-term compatibility with AF-M315E. Recent compatibility testing of the bladder material has like-wise shown acceptable performance for multi-year missions, and hence made a wide variety of existing tanks applicable for the GPIM demonstration and future missions. This revelation is a major benefit for the infusion of the technology as it enables the use of simpler and lower cost elastomeric diaphragm tanks instead of more complex propellant management device (PMD) style tanks or metal diaphragm tanks. A PMD tank approach is possible for longer duration missions, however it would require an updated design, analysis and delta-qualification of the PMD for use with AF-M315E. Even with hydrazine, a PMD design usually has to be re-analyzed for each mission application, whereas a positive expulsion diaphragm provides a more robust and less sensitive propellant expulsion approach.

No delta-qualification of the tank is expected for the GPIM mission, as a qualification-by-similarity and analysis approach should be sufficient to meet mission goals. However, close attention will be paid to the fracture behavior of the tank material with AF-M315E and must be confirmed to comply with the fracture mechanics requirements of AFSPCMAN 91-710 for safe operation of a pressure vessel containing a non-hazardous fluid.

The application of the new propellant in the qualified design reduces the technical readiness of this tank from TRL9 when operating with hydrazine, to ~TRL6 with AF-M315E. The TRL 6 rating is based on the facts that (a) the tank is already qualified with a positive expulsion diaphragm that has shown acceptable compatibility for missions up to several years, and (b) the tank is already qualified for leak-before-burst at a higher proof pressure than required for this demonstration, and (c) the tank shell material has been shown to have long-term compatibility with AF-M315E.

Table 2 Propulsion System Component Summary

Component	Design Adaptation	TRL w/ AFM-315E	TRL w/ Hydrazine
Thruster Valve	Change wetted surface material	5	9 (similar N_2H_4 valve)
Latch Valve	No Change	6	9
System Transducer	No Change	N/A (gas side)	9
Filter	No Change	6	9 (similar N_2H_4 filter)
Service Valves	Change sealing ball material	5	9 (similar N_2H_4 valve)

- **Components**

Table 2 summarizes component selections, respective mission readiness, and modifications required for the TDM green propulsion system. Existing hydrazine system components (TRL9, but evaluated at TRL6 for use with AF-M315E) comprise a nearly complete compatible set, with several components requiring straightforward modifications. The thruster valve will require the interior wetted surfaces to be lined with fully tested compatible material.

For the GPIM mission, the pressure transducers are remaining on the nitrogen pressurant side of the propellant feed system. With the use of a diaphragm tank as a fuel barrier, no changes in the original pressure transducer were required for the GPIM mission. Future component development can be completed at relatively low risk to provide an AF-M315E compatible material version of the pressure transducer for more flexibility in pressure monitoring for future systems. The filter is based off of an existing flight proven design and requires no changes. Similarly, an existing latch valve was deemed acceptable for use on the GPIM program (although longer duration missions would likely need to replace the small valve springs). Lastly, an existing hydrazine flight-qualified service valve will be used commonly throughout the system, except that seal ball comprising one of the three redundant seals will be replaced with more acceptable material, and has already demonstrated compatible for this purpose on the AFRL funded LEAP-DP program.

VII. Propulsion System Payload Module Development Schedule

The overall propulsion effort can principally be divided into two major efforts, development of the thrusters and manufacturing of the propulsion system, Figure 6. Immediate system tasks included assessment of which system components to employ, and understanding of the scope of modifications needed to TRL9 hydrazine components for use with AF-M315E propellant. A complimentary effort was also initiated at the beginning to test the compatibility of all unknown materials with this green propellant. The flight system design effort has two phases, 1) system design up through PDR and 2) final system design up to CDR. The thruster development is divided into three phases: 1) Lab model 22N thruster development, 2) Engineering model (EM) thruster design and then 3) the final flight design activity. Flight thruster designs are expected to be only minor modifications to the EM model based on lessons learned from the EM system bench testing. Testing is also principally divided into three tasks: 1) initial lab model testing of the 22N thruster, 2) EM system bench level testing which includes assessment of both the 1N and 22N EM thruster designs as well as performance evaluation of the complete propulsion system with EM level components and 3) acceptance and qualification testing. Propulsion system delivery to Ball Aerospace Corporation is in November 2014.

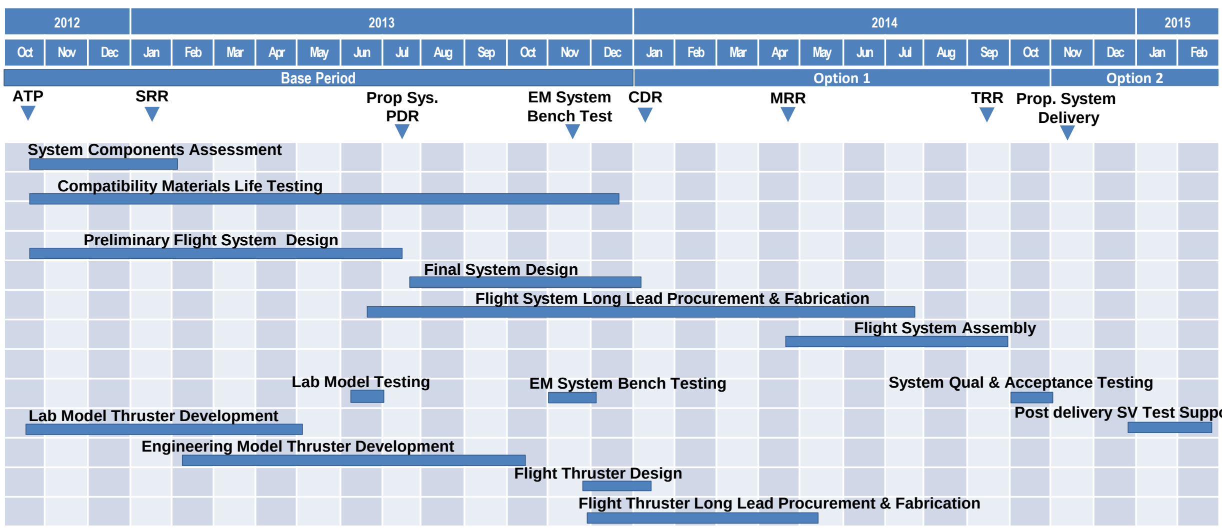

Figure 6 Propulsion System Schedule

VIII. Conclusion

The culmination of this program will be high-performance, green AF-M315E propulsion system technology at TRL 7+ that is ready for direct infusion to a wide range of applications for the space user community.

The combined benefits of low toxicity, easy open-container handling, and high performance of AF-M315E offer a strong alternative to hydrazine for dramatically reducing the cost of access to space for the small vehicles being developed by NASA, DoD and the commercial sector.

AF-M315E propulsion systems will enable spacecraft designers to accommodate significantly more propulsive performance than hydrazine, especially where volume is limited. Some differences in design considerations are needed over hydrazine systems, but in general the approaches are very similar. The GPIM demonstration program will show that these considerations are manageable, especially when compared to the significant benefits of AF-M315E propulsion systems.

Acknowledgments

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MPS-120XL™ CubeSat High-Impulse Adaptable

Overview

MPS-120XL™ **C**ubeSat **H**igh-Impulse **A**daptable **M**odular **P**ropulsion **S**ystem (CHAMPS) is a 2U x 1U hydrazine propulsion system that provides both primary propulsion and 3-axis control capabilities in a single package. The system is designed for CubeSat customers needing significant ΔV capabilities including constellation deployment, orbit maintenance, attitude control, momentum management, and de-orbit.

Specifications and Performance

- Dimensions: 10 cm x 10 cm x 22.7 cm
- Mass: <2.4 kg Dry, <3.2 kg Wet
- Operational Temperature Range: +5°C to +50°C
- Command Method: Digital or Discreet Analog 5 V
- Power Consumption: <4 W Startup, <1 W Operation
- Operational Voltage: 5 V Nominal
- BOL Thrust: 2.79 N (high thrust) to 0.26 N (low thrust) per thruster
- Minimum Impulse Bit (at blowdown-averaged feed pressure): 0.00484 to 0.000467 N-sec per thruster

22 CFR 125.4(b)(13) applicable

Development Status

In Development

Publications

Coming Soon

AND-ENERGY-SOLUTIONS)

CubeSat Modular Propulsion Systems Product Line Development Status and Mission Applications

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The CubeSat platform has greatly reduced the barrier to entry for space missions, resulting in significant market growth. Due to a lack of propulsive capabilities, CubeSat missions are confined to their dispersal orbits. Without propulsion the CubeSat platform cannot realize its total addressable market and the current market will stagnate. Propulsive capabilities enable the CubeSat platform to access the wider range of missions that will strengthen the value proposition of the platform and ensure continued growth in the market. The Aerojet CubeSat Modular Propulsion Systems Product Line satisfies the propulsive needs of the CubeSat community. The product line includes four products: MPS-110 cold gas system, MPS-120 hydrazine monopropellant system, MPS-130 AF-M315E monopropellant system, and MPS-160 solar electric power / solar electric propulsion (SEP^2) system. Systems range in size from 0.5U to 2U with designs generally scalable up to 180 kg class space vehicles such as ESPA node satellites. The CubeSat platform and community have created an environment of rapid development and flight with streamlined processes, Aerojet has therefore incorporated new manufacturing and component technologies that streamline manufacturing and test processes in order to realize aggressive mission schedule and cost thresholds. The configurations, development status, and mission applications of each product are discussed as well as the enabling manufacturing and component technologies that are incorporated into their designs.

Introduction

THE relative simplicity, low development cost, and wide range of available low-cost launch options (as secondary payloads) enabled by the CubeSat platform have opened space access to new classes of users and missions for whom barriers-to-entry of traditional approaches are an order of magnitude or more too high. As the fastest growing Aerospace market segment, the rate of CubeSat launches has increased steadily over the past decade, reaching a current total of 146 nanosatellites as of 2012 tracing their origins to twenty different nations (Canada, Columbia, Denmark, Estonia, France, Germany, Hungary, India, Italy, Japan, Korea, The Netherlands, Norway, Poland, Romania, Spain, Switzerland, Turkey, USA, Vietnam, etc.). That even traditional space users have embraced the cost and schedule advantages realizable through the CubeSat model of using COTS parts with standard interfaces is exemplified in a number of NSF- and NASA-funded missions (CSSWE, Firefly, CINEMA; GeneSat-1, PharmaSat, etc.), and particularly, PhoneSat (NASA), where the total cost of components was less than $7,000.

Due to a lack of propulsive capabilities, CubeSat missions are confined to their dispersal orbits. Without propulsion the CubeSat platform cannot realize its total addressable market and the current market will stagnate. Propulsive capabilities enable the CubeSat platform to access the wider range of missions that will strengthen the value proposition of the platform and ensure continued growth in the market. Propulsive capabilities ranging from ~10m/s for small dispersal maneuvers to >200m/s for large apogee maneuvers are required. The Aerojet CubeSat Modular Propulsion Systems (MPS) Product Line satisfies the propulsive needs of the CubeSat community. The product line simplifies propulsion mission planning and integration so that any level of CubeSat builder can consider a propulsive mission.

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Product Line Overview

In 2011, Aerojet began development of a 1U modular propulsion system call the CubeSat High-impulse Adaptable Modular Propulsion System (CHAMPS) designated “MRS-142” to address the emerging need for CubeSat propulsion systems.[i,ii] Leveraging designs and components developed for the MRS-142 along with key new technologies enabled Aerojet to develop the CubeSat Modular Propulsion Systems product line shown in Figure 1. The systems leverage common parts and designs in order to reduce non-recurring engineering and to achieve economies of scale that will enable reduced cost and lead times as product line production rates increase.

The objective of the CubeSat Modular Propulsion Systems product line is to simplify mission planning, system selection, and satellite integration to the point that any level of CubeSat builder can consider a propulsive mission. This objective is accomplished through the following features:

- Catalog of standard systems with clear propulsive capabilities listed
- “U” based form factor that enables simple mechanical interfacing
- Elimination of requirement for fluidic connections typically required of the tightly integrated propulsion systems found on larger satellites
- Propulsion system control unit with a single power and data connection that simplifies electrical and software integration

Product Image	Product Number	Description	ΔV for 3U 4kg BOL	ΔV for 6U 10kg BOL
	MPS-110	• System Mass: Varies depending on selected size • Propellant: Inert gas • Propulsion: 1 to 4 cold gas thrusters	10 m/s	N/A
	MPS-120	• System Mass: <1.3kg dry, <1.6kg wet • Propellant: Hydrazine • Propulsion: Four 0.26—2.8 N (BOL) rocket engines	209 m/s	81 m/s
	MPS-130	• System Mass: <1.3kg dry, <1.6kg wet • Propellant: AF-M315E • Propulsion: Four TBD—1 N (BOL) rocket engines	340 m/s	130 m/s
	MPS-120XW	• System Mass: <2.4kg dry, <3.2kg wet • Propellant: Hydrazine • Propulsion: Four 0.26—2.8 N (BOL) rocket engines	440 m/s	166 m/s
	MPS-120XL	• System Mass: <2.4kg dry, <3.2kg wet • Propellant: Hydrazine • Propulsion: Four 0.26—2.8 N (BOL) rocket engines	539 m/s	200 m/s
Image Coming Soon	MPS-160	• System Mass: TBD • Propellant: Xenon • Propulsion: 80W Solar Electric Power/Solar Electric Propulsion System (SEP^2)	N/A	>2,000 m/s

Figure 1: CubeSat Modular Propulsion Systems Product Line

Enabling Technological Innovations

A. Miniaturized Rocket Engine Technology

Aerojet investments to commercialize technologies stemming from small form factor missile defense applications has enabled miniature rocket engines and valves capable of supporting CubeSat missions. The resulting MR-14X series of engines realizes a ~4× reduction in engine size as shown in Figure 2. Aerojet's efforts to adapt miniature rocket engine technology for AF-M315E propellants enables both hydrazine and AF-M315E solutions.

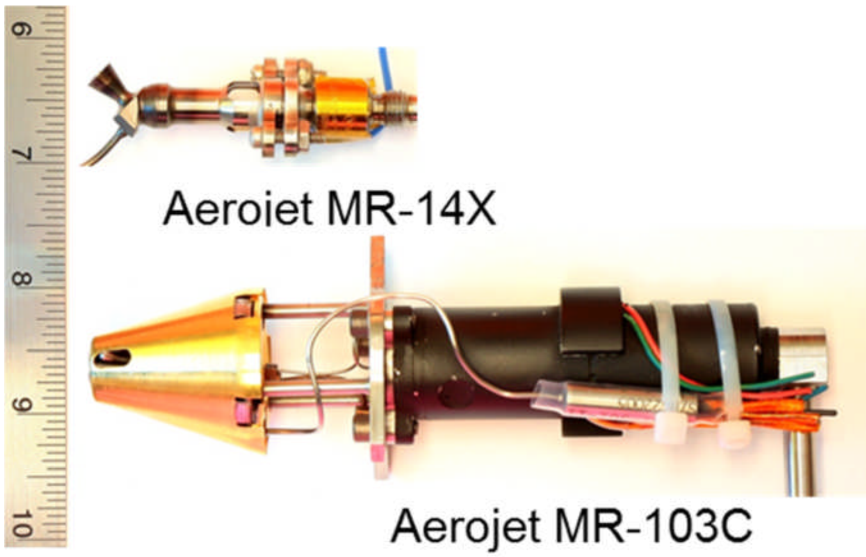


Figure 2: Aerojet Miniature Rocket Engine Compared with a Standard Rocket Engine

B. Additive Manufacturing Process Infusion

Subtractive manufacturing is a generic term used to describe a manufacturing process that removes material from a piece of stock in order to fabricate a part. Examples of subtractive manufacturing processes include: milling, turning, cutting, and drilling. In contrast, Additive manufacturing is a generic term used to describe a manufacturing process that deposits and bonds material together to fabricate a part. Additive manufacturing processes produce parts directly from a digital design. Additive manufactured parts typically require little or no tooling, significantly reducing the cost and lead time of designing, manufacturing, and maintaining tools. If fixtures or tooling are needed they can typically be fabricated during the build process, minimizing the need to create tools ahead of the build or maintain them after the build. The reduced requirement for tooling significantly reduces setup time and cost as well as inventory costs. Additive manufacturing processes typically consume only the material needed to make the part. Typically, most residual material used during the process is re-usable for fabrication of future batches of parts. Additive manufacturing eliminates the need for cutting fluids that are required in subtractive manufacturing processes. The combination of efficient use of material and elimination of support fluids results in significant reductions in material cost and waste. Overall, additive manufacturing process benefits can realize significant reductions in fabrication time and cost. These benefits enable opportunities for more design iterations than traditionally possible, enabling lower cost development programs with higher quality design outputs that are typically ready for direct transition to low volume production. These characteristics are of high importance to the typically long duration, high cost development programs and ultimately low volume production of spacecraft systems.

Current additive manufacturing machines are constrained to build envelopes of ~30 cm^3. The MPS-100 product line includes propulsion systems that fit the standard 1U CubeSat envelope of ~10 cm × 10 cm × 10 cm, making these systems ideal candidates for demonstration and infusion of additive manufacturing process technology. Aerojet has embraced the use of additive manufacturing methods and has begun infusion of new design philosophies and manufacturing processes to develop more affordable propulsion systems. The MPS-120 and MPS-130 liquid propulsion systems utilize a piston tank that includes a piston, propellant tank, and pressurant tank. Some components include internal flow passages that were identified as opportunities for improvements with additive manufacturing. Figure 3 shows how design for additive manufacturing enables improvements that reduce component count and eliminate potential leak paths in the system. Figure 4 demonstrates how additive manufacturing removes costly weld/inspection processes. These are just some examples of the benefits offered by additive manufacturing for propulsion systems. Aerojet is working to demonstrate that many types of additive manufacturing processes can be applied to the MPS-100 product line including: Electroforming (EL-Form®), Selective Laser Melting (SLM), Electron Beam Melting (EBM), and Laser Engineered Net Shaping (LENS™).

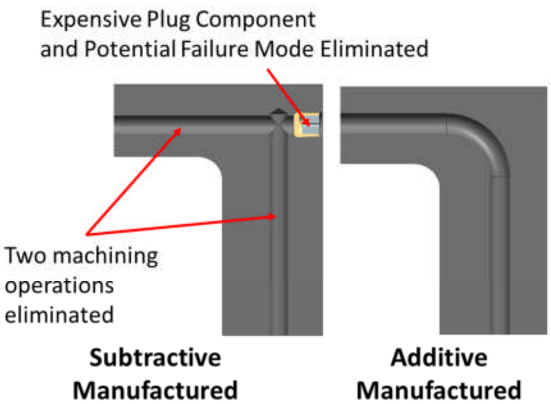


Figure 3: Internal Passages Enable Elimination of Components

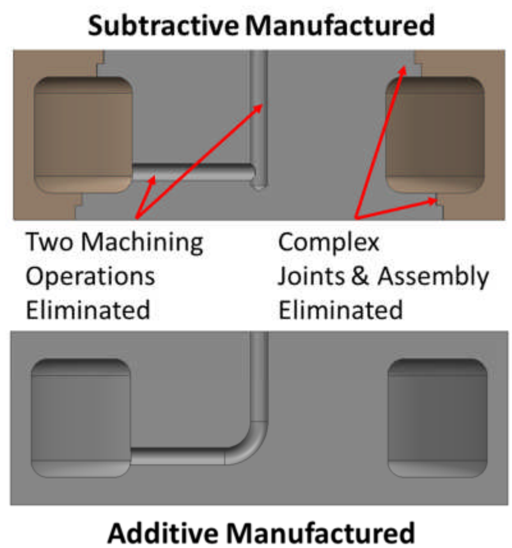


Figure 4: Internal Passages Enable Elimination of Processes

The EL-Form® process uses molten salt electrolytes, instead of the aqueous solutions of standard electroplating processes, to enable electrodeposition of compact metal layers onto a mandrel. EL-Form® enables refractory metals to be formed into dense, non-porous and crack-free layers. The EL-Form® process can create component structures on mandrels and/or dense coatings applied existing parts. The EL-Form® process was used to produce the Ir/Re chamber and nozzle for MR-143 engines in MPS-130 system. An operational demonstration of these components is planned for 2013.

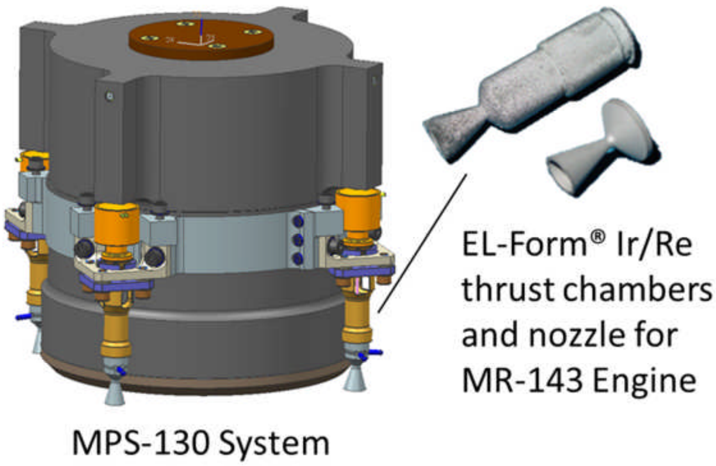


Figure 5: EL-Form® Components

The SLM and EBM processes deposit powder in layered fashion and apply laser (SLM) or electron beam (EBM) to sinter powder. Figure 6 are examples of Inconel and titanium components produced by SLM. Figure 7 presents as-printed propellant tank components manufactured by EBM. Operational demonstrations with these components is planned for 2013.

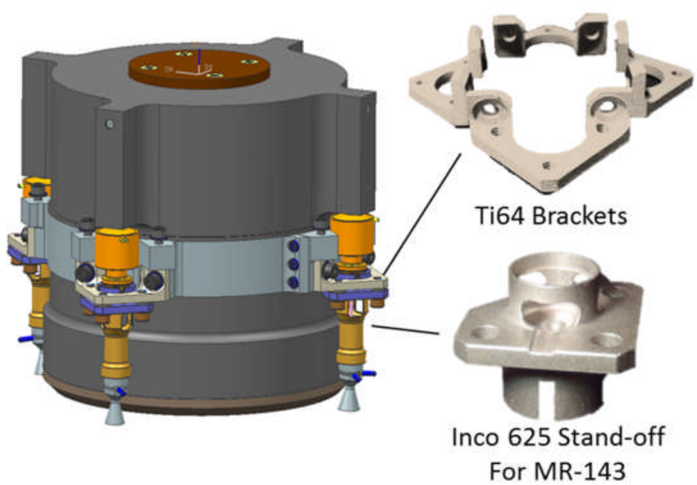


Figure 6: SLM Additive Manufactured Components

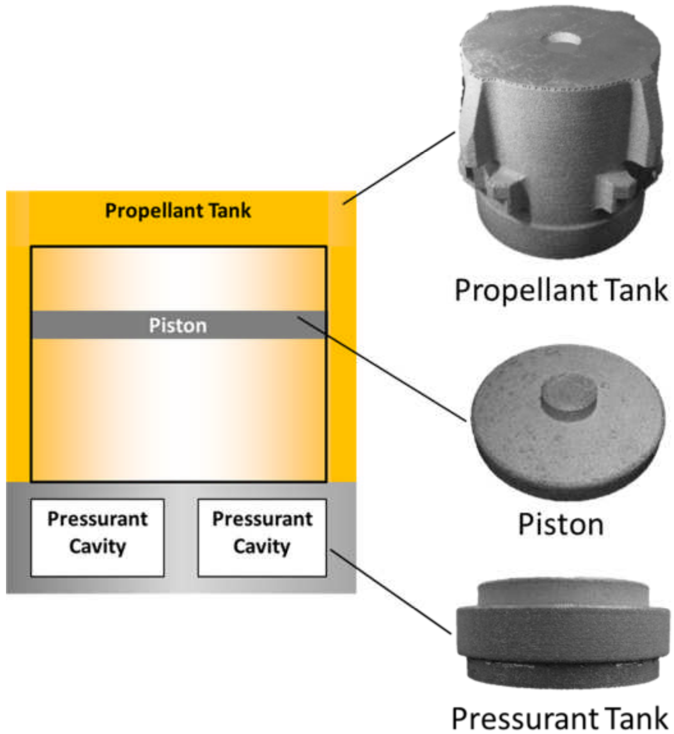


Figure 7: As-Printed EBM Additive Manufactured Piston Tank Components

Laser Engineered Net Shaping (LENS™) is a new manufacturing technology that simultaneously sprays and sinters powder, reducing or eliminating the need for powder removal required by SLM and EBM. Work is ongoing to demonstrate a LENS™ version of the common piston tank. An operational demonstration of the LENS™ tank is planned for 2013.

Demonstration of additive manufacturing production capabilities enables product line development, production, scaling, and tailoring at substantially lower cost and schedules than subtractive manufacturing processes alone. While the objective of the product line is to offer standardized parts, it is recognized that some customers will require non-standard sizes and geometries to fit within available space or to maximize use of available space. The use of additive manufacturing in the standard products enables Aerojet to offer non-standard configurations that do not necessarily require full re-qualification of the system. As an example, 1U and 2U variants of the MPS-120 will be standard, however it is possible to quickly develop and produce a 1.5U version if required by a customer.

C. Solar Electric Power/Solar Electric Propulsion (SEP^2) System Architecture

Several companies have offered electric propulsion systems for CubeSats capable of low ΔV and attitude control; however these systems have realized little mission utility. In order to truly benefit from electric propulsion, an apogee solar electric propulsion (SEP) system is desired that can provide significantly more ΔV than chemical systems. However, the cost and mass of electronics in typical apogee electric propulsion solutions are prohibitive on such a small scale. In order for an electric propulsion system to be effective on a platform as small and low cost as a CubeSat, a different approach is required compared with larger satellites.

For several years, Aerojet has been working on a technology called Direct Drive which operates electric thrusters directly from high voltage solar arrays in an attempt to boost efficiency, reduce components, and reduce waste heat. Previous Direct Drive development activities have focused on multi-kilowatt systems.[iii] However, the same technology applied to the CubeSat platform significantly reduces the mass and cost of power electronics to the point that primary electric propulsion on CubeSats becomes feasible. An integrated solar power system and direct drive solar electric propulsion control unit enabled Solar Electric Power and Solar Electric Propulsion (SEP^2) system enables electric propulsion apogee systems for CubeSats. Figure 8 is an example comparison of a traditional solar electric propulsion system with Aerojet's SEP^2 system concept.

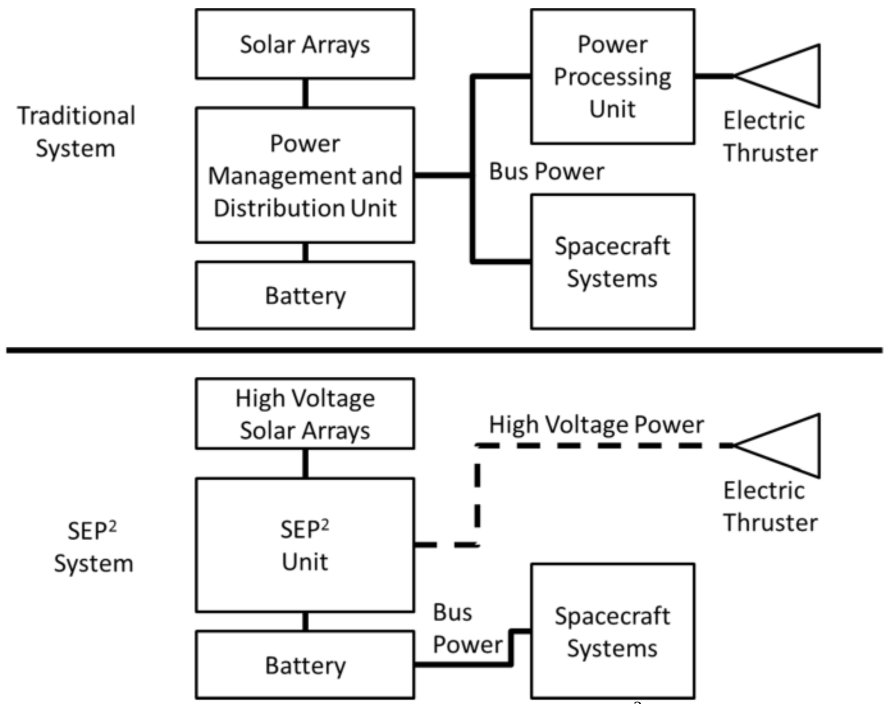


Figure 8: Comparison of Traditional and SEP^2 Systems

Modular Propulsion System Product Descriptions

D. MPS-120 Hydrazine Monopropellant Propulsion System

The MPS-120 maintains much of the original MRS-142 design with some significant changes to align with the overall product line approach. The system has been simplified with the new fluidic schematic shown in Figure 9. An additive manufactured titanium piston tank replaces the previous machined aluminum tank design of the MRS-142. While the aluminum tank is still an optional variant of the new MPS-120 product, the new baseline titanium version provides comparable ΔV and enables more commonality within the product line, reducing system costs. MPS-120 designs are complete and fabrication is currently under-way with MR-142 engines and additive manufactured titanium piston tank nearing completion and readiness for integrated testing.

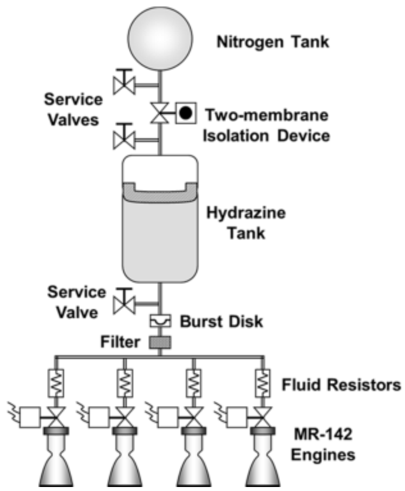


Figure 9: MPS-120 System Schematic

E. MPS-130 AF-M315E Monopropellant Propulsion System

The MPS-130 is a new product offering derived from the MPS-120. Figure 10 presents the fluid schematic for the MPS-130 which is almost identical to the MPS-120 except that a burst disk is not required for the AF-M315E green monopropellant and the system employs new MR-143 engines capable of operating on AF-M315E green monopropellant. The MR-143 engines are of similar size to the MR-142, but utilize rhenium chambers that survive the high combustion temperatures of AF-M315E propellant. At the time of this writing, the MPS-130 design and drawings are complete, and fabrication is currently under-way with MR-143 engine components produced and ready for engine assembly.

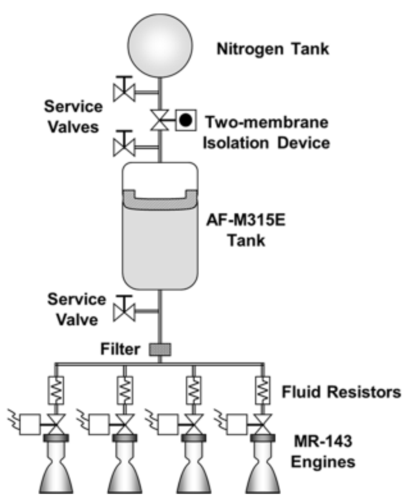


Figure 10: MPS-130 System Schematic

F. MPS-110 Cold Gas System

The MPS-110 Cold Gas system is being developed to provide a propulsive capability for missions on small platforms that need minimal ΔV to achieve their mission objectives. Applications would primarily be initial dispersion, minor orbit adjustments, or attitude control. The MPS-110 system derives valves, filter, and tank design from the MPS-120 system mentioned previously. Figure 11 is the fluidic schematic of the MPS-110. The system is capable of operating with a variety of pressurants such as GN2 or condensables enabling significant mission tailoring. MPS-110 pressurants have been selected and operational behaviors are well understood.

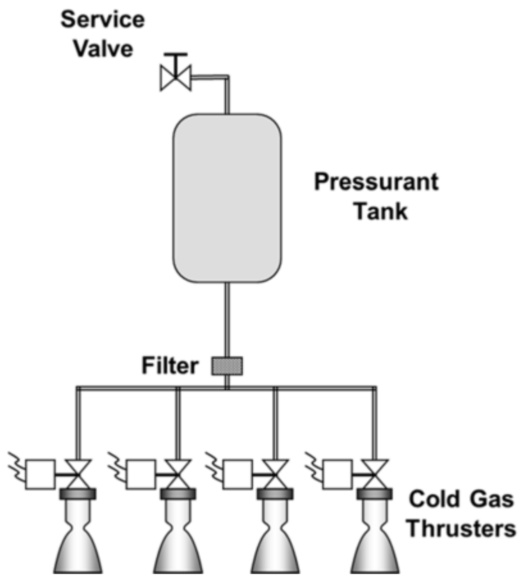


Figure 11: MPS-110 System Schematic

G. MPS-160 Electric Propulsion System

The MPS-160 is a concept system that differs significantly from the systems presented thus far in that it is a 2U system that includes both power and propulsion using the aforementioned SEP[2] system architecture. The MPS-160 concept development is aimed at developing such a system that would ultimately be capable of providing >2,000m/s to a 6U CubeSat from a 2U propulsion and power package. Figure 12 presents the MPS-160 system schematic. A Hall thruster is used to represent the apogee propulsion; however multiple types of electric thrusters are applicable. Hall thrusters, gridded ion thrusters, and other types of thrusters are in development at the power, voltage, and specific impulse levels required by the MPS-160 system enabling the system to support a wide range of missions.

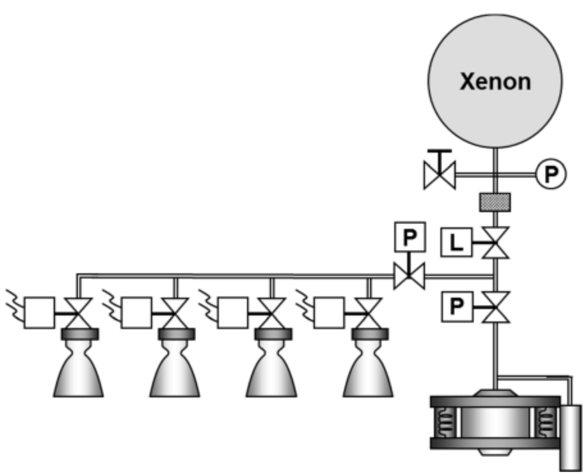


Figure 12: MPS-160 System Schematic

Mission Applications

H. Missions Requiring Dispersal

Every satellite begins its mission life with a deployment event from the launch vehicle upper stage, and to prevent re-contact after a number of orbits if the upper stage is not actively de-orbited, propulsive maneuvers are typically employed by the satellite to assure that collision does not occur with the upper stage. Alternatively, some satellite missions may desire to conduct propulsive maneuvers to “scatter” away from the larger upper stage, which can easily be tracked by amateur radio operators and launch trackers. Secondary payloads to date typically reserve any minimal ΔV capability found with cold gas systems for utmost critical mission events like attitude control or end-of-life de-orbit requirements. High-impulse propulsion systems, such as the MPS-120 CHAMPS, can provide secondary payloads with the tactical advantages that larger satellites have enjoyed for decades. Figure 13 shows the dispersal capabilities of Aerojet’s CubeSat Modular Propulsion Systems product line to impart 5 m/sec of ΔV to the maximum satellite mass that is achievable. This amount of ΔV is considered the minimum needed to achieve safe and tactical deployment, and also matches the typical 5 m/sec achieved from a CubeSat P-POD jettison event. Two observations can be made from this figure: the MPS-110 cold gas system is adequate in providing enough ΔV for most 3U CubeSats and some 6U CubeSats for dispersal applications, and the MPS-120 and MPS-130 can be integrated on satellites much larger than CubeSats to gain tactical dispersal capability for low cost compared to custom propulsion system solutions. This is very compelling for missions for smallsats in the range of 50-300 kg that are designed for simple mission capability and low-cost and where modularity is emphasized or required. Similarly, the MPS-120 and MPS-130 can be used as a modular addition to a deployable ESPA node to create a dedicated stage to capable of delivering multiple CubeSats to a desired orbit and/or phasing.

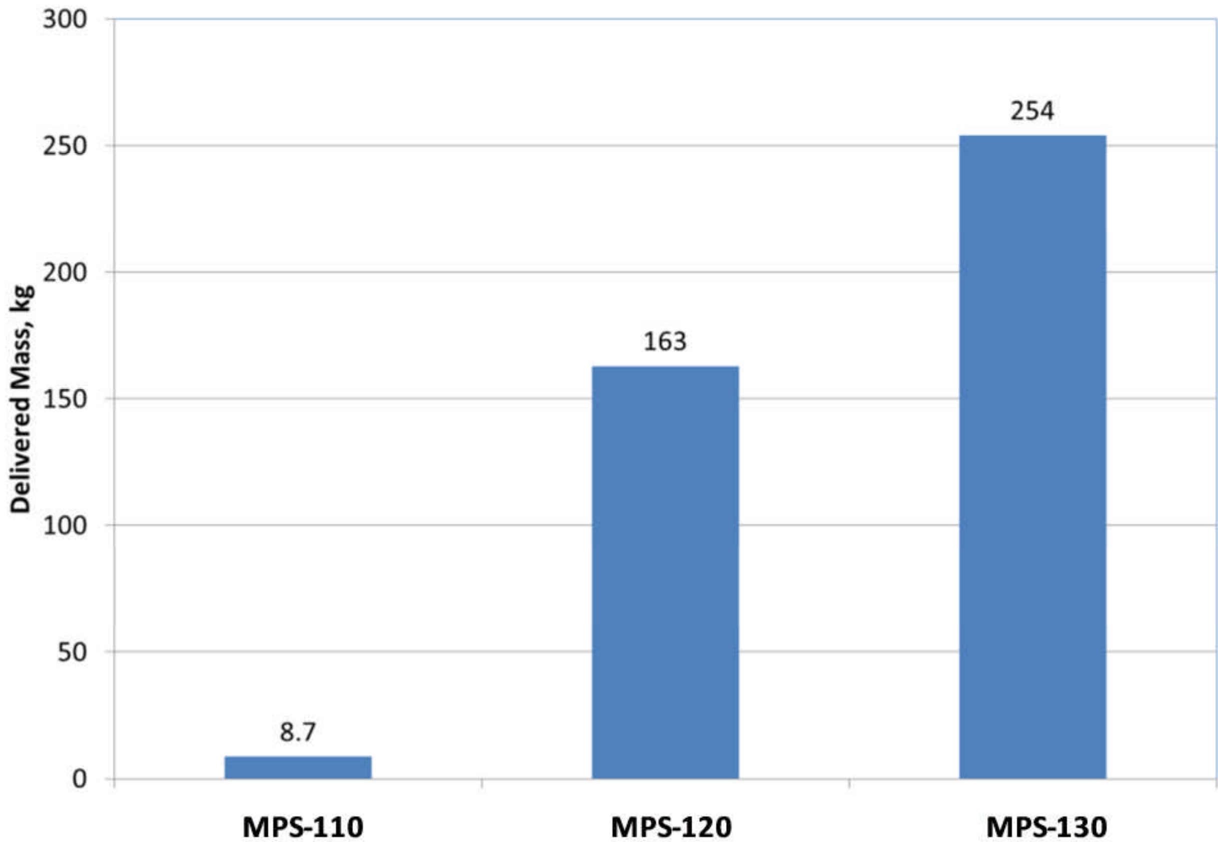


Figure 13: MPS Product Line Mass Dispersal Capability at 5 m/sec ΔV.

I. Missions Requiring Low Flight

Another significant area of interest in the CubeSat community is using low-cost imaging-capable CubeSats to fly at low altitudes to augment the resolution capability of COTS-based imaging systems. This can be employed to support responsive disaster monitoring, localized weather monitoring, and other situations where data from a particular area of interest becomes valuable for a temporary period. To make this concept compelling, significant ΔV is required to counteract drag and extend the lifetime of the satellite to the point where enough data is mined over the life of the satellite to be regarded as worth the cost of an otherwise expendable satellite. This evaluation should also factor in the responsive capability of the CubeSat form factor; a 6-12U imaging CubeSat that is small enough to be integrated with dedicated small satellite launch vehicles or tactical small satellite air-launched platforms could trump the logistical cost of maintaining a constellation of higher-value imaging satellites over longer mission lifecycles which do not necessarily guarantee fast image-capture over a new area of interest. Packageable within a 20 cm x 20 cm x 30 cm volume, these types of CubeSats could be pre-integrated with smaller, dedicated, on-demand launch vehicles sized to deliver spacecraft weighing less than 50 kg to LEO, to be used when other space-based assets are either not accessible or too expensive to utilize. This on-demand capability lends immediate tracking resources to organizations responsible for monitoring disasters like tornados, oil spills, forest fires, etc.

To assure frequent image updates over an area of interest, a low-altitude, repeating ground track orbit can be utilized to provide up to two revisits per day per satellite. Figure 14 below shows such an orbit at 262 km circular, which can provide up to 1.7 m resolution with a COTS type optical system that provides a 9 cm aperture and 1.25 m focal length. Revisit sites over areas of interest for repeating ground track orbits can be easily selected by calculating the required orbital injection site and inclination of the launch vehicle, with the satellite propulsion system conducting the final orbit “cleanup” burns. Image acquisition over multiple areas of interest can potentially be achieved with this system, as Figure 14 demonstrates, to support short and long-term change detection for global map data, crop management, climate monitoring, etc.

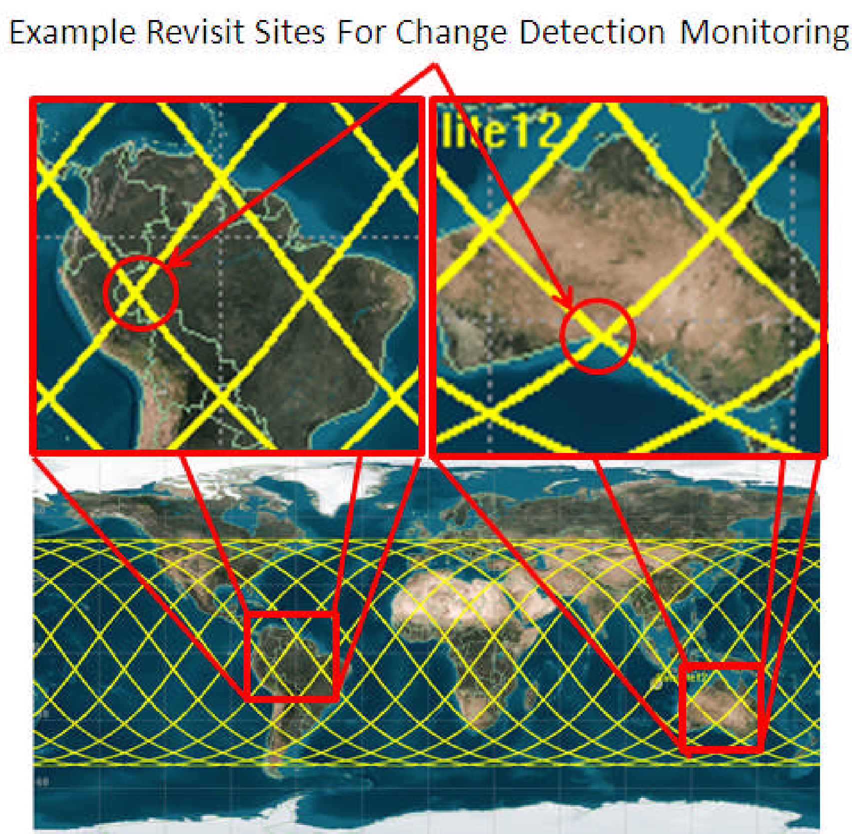


Figure 14: Low Altitude Repeating Ground Track Orbit Enables High Revisit Rate per Satellite.

At the altitude of the repeating ground track orbit in Figure 14, the CubeSat Modular Propulsion Systems product line can extend life of 6U CubeSats (baselined weighing 10 kg) with varying ballistic coefficients to the values shown in Table 1 below. This life augmentation capability provides the end user with frequent and persistent data to support many operational situations that required dedicated imaging assets over longer time periods.

Table 1: CHAMPS Lifetime Extension at 262 km Circular Orbit.

Lifetime (days) for 6U (10kg S/C) at 262 km			
	MPS-110	MPS-120	MPS-130
	Ballistic Coefficient = 50 kg/m^2		
Solar Max	4.5	43.0	66.0
Solar Nom	11.1	183.4	286.9
Solar Min	27.5	402.0	626.9
	Ballistic Coefficient = 50 kg/m^2		
Solar Max	19.0	169.3	259.9
Solar Nom	44.0	776.0	1215.9
Solar Min	109.4	1712.5	2675.1

Several COTS imaging systems have been identified[ivv] that can be retrofitted for structural and thermal stability as well as some optical aberrations to provide this resolution capability, while taking up less than 2U of payload space on a CubeSat. Such an optical system that employs a Maksutov-Cassegrain telescope mirror system is shown below in Figure 15 for visual comparison to the overall CubeSat form factor.

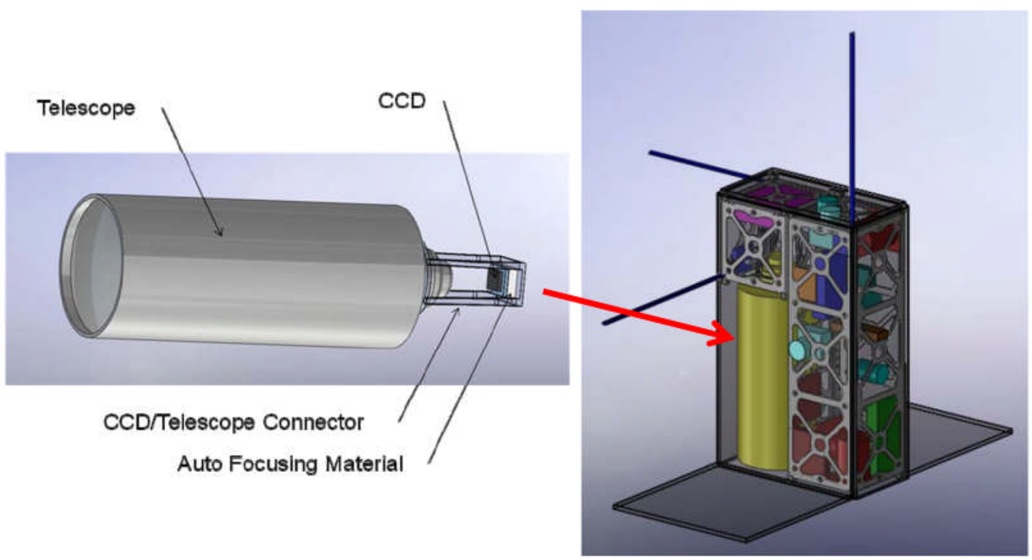
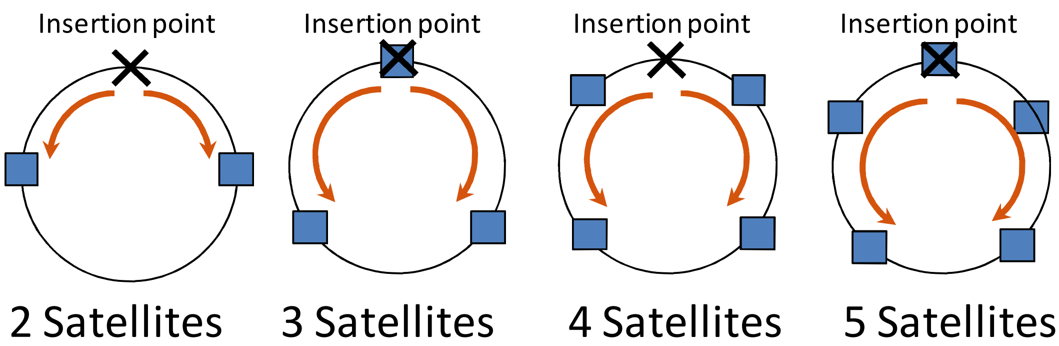


Figure 15: COTS imaging optics can package within CubeSat volumes.

Tasking, Processing, Exploitation, and Dissemination (TPED) has historically been problematic for these types of CubeSat missions due to difficulty of communicating with available ground stations to guarantee that high-value imaging data is collected and delivered to the end user with acceptable latency. However, recent CubeSat missions have employed deployable high gain antennas to communicate with ground assets with low RF power. Specifically, the AENEAS mission launched a 3U CubeSat that deployed a 0.5m parabolic antenna for communication on WiFi frequencies to ground assets that boasted a gain of 18dB[vi]. Other entities are currently developing 2m deployable antennas for S-band communication that occupy only 1U. Advancements in deployables technology continue to mature the possibility of achieving a link from LEO to a dedicated or mobile ground station using burst transmission mode, as well as the possibility of achieving a link to a higher altitude satellite communication network (i.e. TDRSS, etc.) to support high rate data transfer.

J. Constellation Deployment Missions

Another capability that can enable tactical satellite missions is the ability perform relatively fast phasing maneuvers to quickly deploy a constellation, or "scatter" it. This always comes at a cost impact in the form of propellant consumption, and thus less ΔV remaining for additional necessary maneuvers. Figure 16 below describes the phasing capability for MPS products for a variety of constellations at an orbital altitude of 500km.



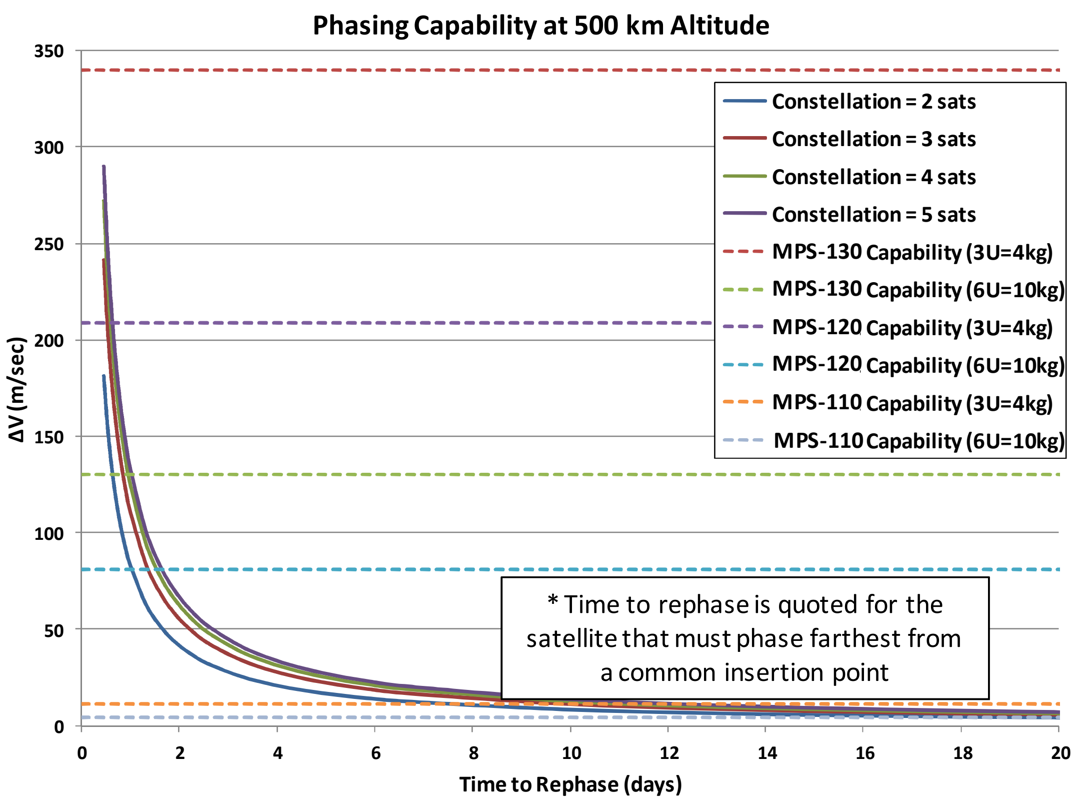


Figure 16: Product Line Phase/Rephase Capability at 500km Altitude.

K. Low Thrust Missions

The MPS-160 provides low thrust apogee propulsion for a wide range of missions. A small pressurant tank stores the xenon propellant at supercritical conditions and the ΔV capability is a function of tank size and storage pressure. Figure 17 plots the MPS-160 notional ΔV capabilities as a function of beginning-of-life storage pressure, tank size, and thruster specific impulse. A propellant storage temperature of 70°C was used to bound a worst-case estimate. ΔV requirements for various missions of interest are overlaid in the figure to show mission capability thresholds. It can be seen from the graph that the 1.5U, 3000s Isp case provides significant capabilities at a relatively low storage pressures. Further study is needed to ensure reasonable trip times and payload masses, however this preliminary assessment demonstrates that the MPS-160 could enable rideshare CubeSats to access missions to GEO and the Moon.

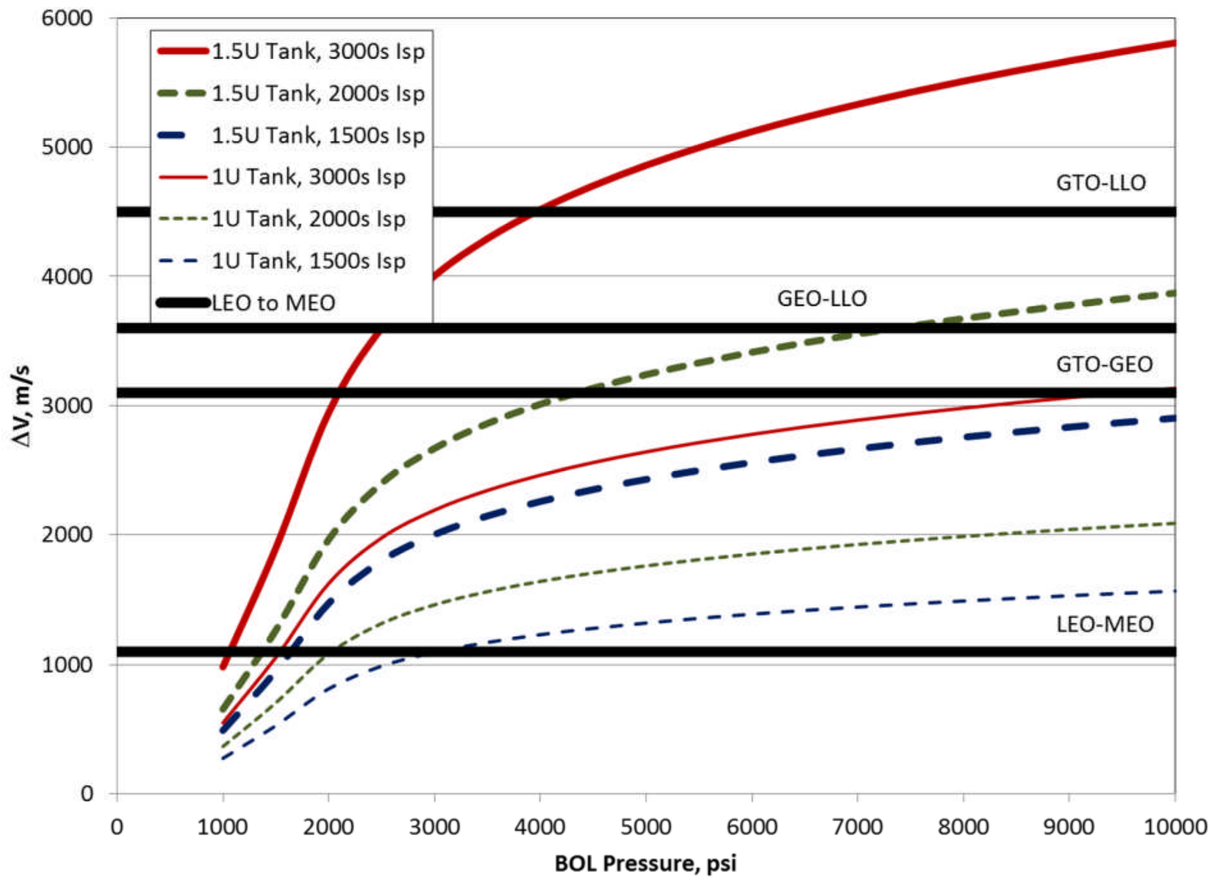


Figure 17: MPS-160 Notional ΔV Capabilities as a Function of Xenon Storage Pressure

Conclusion

As with traditional space applications, propulsion options providing means cost-effective of dispersal, constellation deployment, orbit management, and angular momentum dissipation will greatly augment the range of missions CubeSats can perform. In so doing, these expanded mission capabilities will strengthen the value proposition of the platform and further stimulate current market growth trends. The large reduction in launch costs potentially offered by CubeSats makes propulsion even more pivotal for their future, however, in that the full advantage of substantially increased multi-manifesting (stemming directly from the small CubeSat form factor) can only be realized if co-launched CubeSats possess a practical means of post-deployment orbit differentiation. To meet this growing need, Aerojet is developing the CubeSat Modular Propulsion Systems product line to simplify mission planning, system selection, and satellite integration to the point that any level of CubeSat builder can carry out a successful propulsive mission. Four products are in development with MPS-110 Cold Gas and MPS-120 Hydrazine Monopropellant systems on track to be flight-ready by 2014, to be followed by the MPS-130 and MPS-160 advanced (AF-M315E) monopropellant and solar electric propulsion[2] systems by 2015 and 2016, respectively.

References

[i] Schmuland, D. T., et. al. 2011 "Hydrazine Propulsion Module for CubeSats," *Proceedings of the AIAA/USU Conference on Small Satellites*, Mission Enabling Technologies, Paper No. SSC11-X-4, http://digitalcommons.usu.edu/smallsat/2011/all2011/69/.

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1cm RF Ion Thruster BIT-1

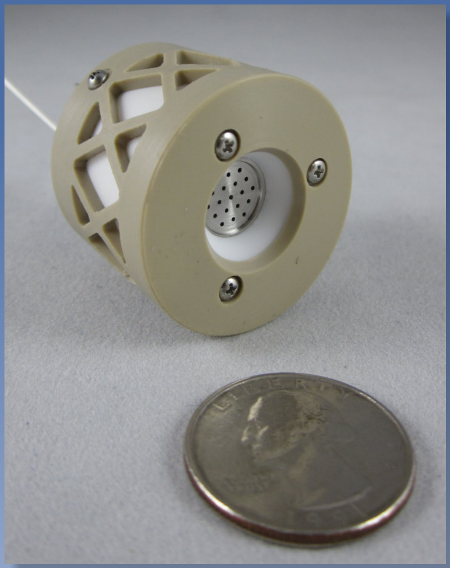
BUSEK
Space Propulsion and Systems

World's Smallest Ion Thruster

Busek's BIT-1 RF ion thruster is an ultra-compact, high-performance ion propulsion device designed with nano-satellite users in mind. Weighing just 53 grams and having a size close to a U.S. quarter, the BIT-1 thruster can produce 100 μN thrust and 2150 second I_{sp} with just 10W of power. When higher power is available, the thruster's performance can easily exceed 180 μN thrust and 3200 second Isp.

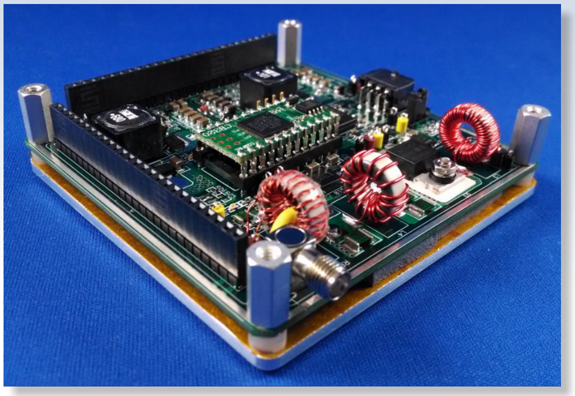
As with other Busek RF ion thrusters, BIT-1 employs inductively-coupled plasma (ICP) discharge to generate its ion source. The utilization of RF discharge eliminates the need for internal hot cathode and thus increases overall lifetime while enabling extreme miniaturization. Thruster life is dominated by grid erosion, which by simulation exceeds 20,000 hours. BIT-1 by default is paired with Busek's subminiature hollow cathode BHC-50E for ion beam neutralization.

In addition to its small size and low power, BIT-1 is designed to be compatible with the solid-storable propellant iodine. Such unique properties make the BIT-1 system extremely favorable for nano-satellites such as CubeSats, where volume and mass are highly constrained. Miniaturized, microcontroller-based Power Processing Unit (PPU) for BIT-1 also exists in the CubeSat form factor. The PPU contains an innovative RF generator/amplifier board with integrated load power sensor and automatic frequency matching. Based on a modified Class E RF amplifier topology, the RF board has a proven 80% DC-to-RF power conversion efficiency for BIT-1 operation.



BIT-1 RF Ion Thruster

BIT-1 Operating with Xenon at 13W

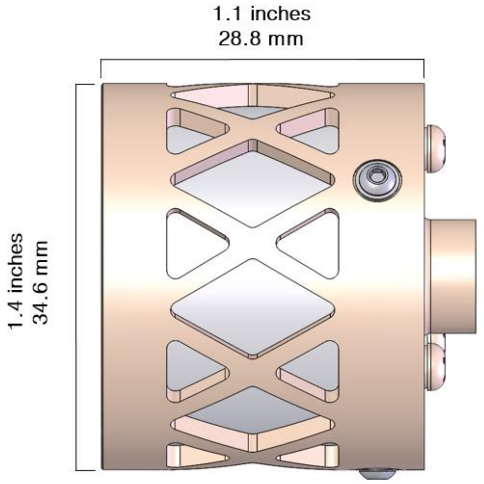


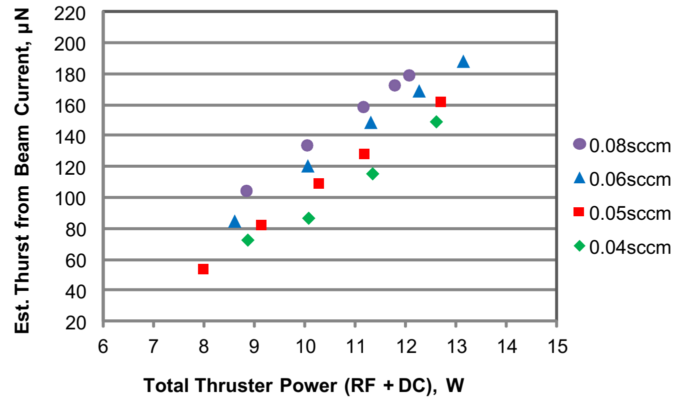
High-Efficiency RF Generator/Amplifier Board in the CubeSat Form Factor

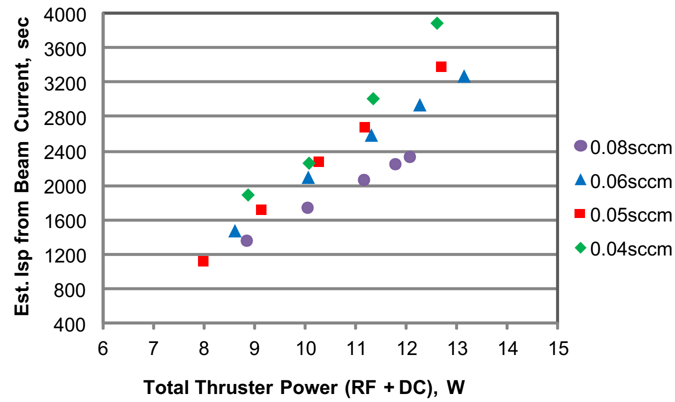
Very precise thrust output is possible with this technology, as well as multiple modes of operation, ranging from higher to lower specific impulse.

BIT-1
Technical Specifications

Envelope Drawing



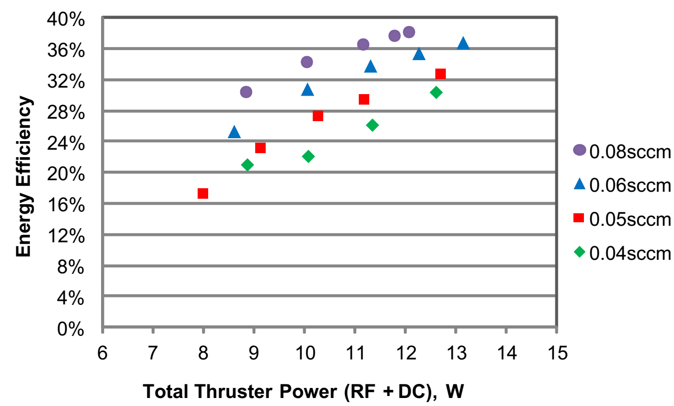




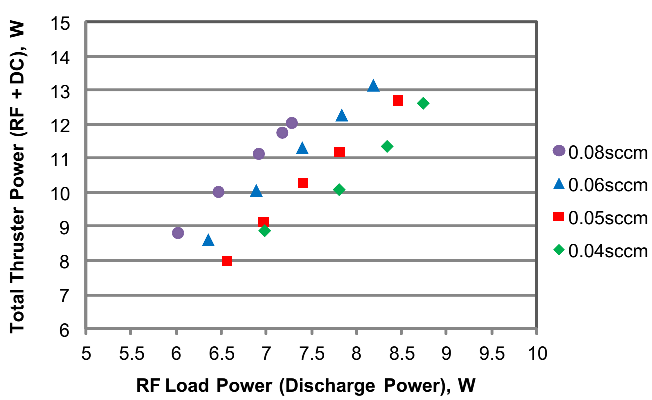
Nominal Specification

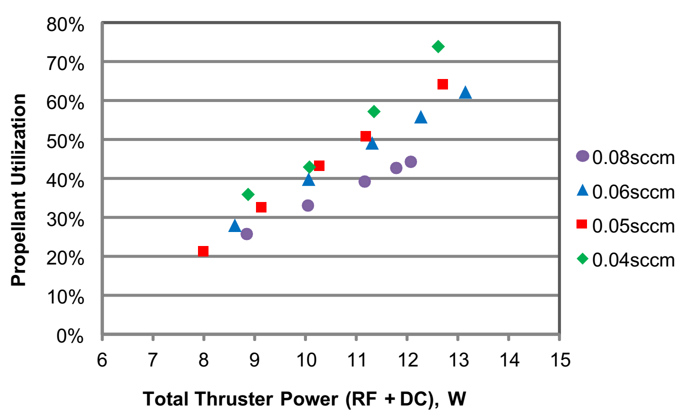
Total Thruster Power*	10 W
Ion Beam Current	1.5 mA
Propellant Mass Flow	4.9 µg/sec Xe
Thrust	100 µN
Specific Impulse	2150 sec
Propellant Utilization	41%
Energy Efficiency**	27%
Grid Input Voltage	2 kV
Thruster Mass***	53 g

** Does not include PPU efficiency or neutralizer consumption*
*** Defined as $P_{grids}/(P_{grids} + P_{RF})$*
**** A complete BIT-1 propulsion system will need to include neutralizer, PPU, feedsystem, and propellant tank*



Performance Characteristics (Xe)





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70011950 RevA

The physics of Nitrous Oxide

Introduction

Most amateur rocketry groups choose Nitrous oxide, often referred to as ‘nitrous’, and sometimes ‘nitrogenous oxide’ or ‘dinitrogen oxide’, (though *not* ‘nitro’ which is nitromethane) as the oxidiser for their hybrid rocket engines.
Another hybrid rocketry nick-name for Nitrous oxide is ‘Nox’, but ‘NOx’ is actually a broad environmental term for any of the various compounds and derivatives in the family of nitrogen oxides, including nitrogen dioxide, nitric acid, nitrous oxide, nitrates, and nitric oxide.

Nitrous oxide’s chemical formula ($N_2 O$) shows a predominance of Nitrogen, which doesn’t help at all with burning; it appears at first sight just to be dead-weight that has to be carried aloft, though it cools the nozzle sufficiently that graphite nozzle inserts can be re-used many times.
But infact the 'inert' nitrogen actually performs a very useful function within hybrid combustion chambers:
Nitrogen mass flow is the majority of the nitrous mass flowing down the central hole or fuel port, and so this does most of the eroding of fresh fuel to burn with the oxygen within the nitrous.
So it significantly aids the fuel erosion rate. (see our’ Introduction to hybrid design’ paper)
That's why it's much harder to get a decent thrust out of pure oxygen hybrids: the **Specific Impulse** is higher, but the fuel erosion rate at **Stociometric** mixture is much lower.
So blasting another inert gas down the port of a liquid oxygen hybrid would help up the fuel liberation (but that's heading back towards nitrous oxide again.)

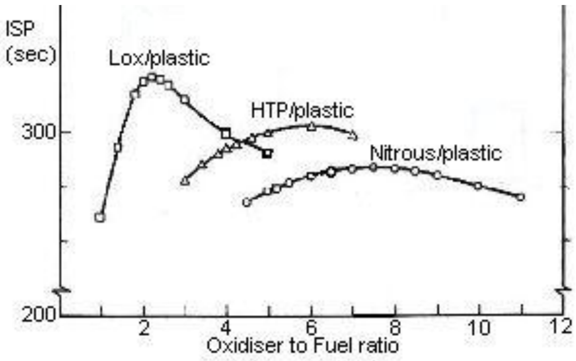
Still, tweaked for performance, nitrous hybrids will outperform most solid motors, so the following are some of the points to consider when designing and/or using a nitrous hybrid.

Note that this paper covers the positive aspects of nitrous oxide; for the negative aspects see our paper ‘Hybrid safety’.

The overtly useful aspects of nitrous are:

1) The simple gas bottles nitrous has to be stored in are a lot cheaper to buy or rent than, say, liquid oxygen or hydrogen peroxide containers, so at the small quantities most amateur groups use, nitrous systems work out cheaper, even though the nitrous itself is quite expensive per litre.
Oxygen has to be chilled below about minus 120 degrees C before it provides a reasonably dense liquid phase: most of the money Aspirespace spends on each H20 hybrid test goes on the containers and cryogens necessary just to keep the liquid oxygen (Lox) this cold for quarter of an hour or so in the test-tank.
Nitrous is readily available from many sources, such as hotrod car shops, whereas a hellova lot of health and safety paperwork has to be done before anyone will sell you Lox or High Test-concentration Peroxide (HTP), or even worse, the utterly toxic Red Fuming Nitric Acid (RFNA).

2) Just like peroxide, a large oxidiser to fuel ratio is required when burning nitrous in the combustion chamber (around 7:1 by mass) which results in a requirement for vast quantities of nitrous, and so a large tank onboard which it’s difficult to keep from being heavy.
This high ratio isn’t all bad news, because as the oxygen within is a low fraction of the total nitrous, you can be quite sloppy with the 7:1 *nitrous* to fuel ratio without altering the actual *oxygen* to fuel ratio within, much.
This means that unlike other oxidisers, a graph of **Specific Impulse** plotted against oxidiser-to-fuel ratio *doesn’t* have a sharp peak at best (**stochiometric**) mix that drops off sharply on either side of the peak.



The graph shown here for Nitrous/plastic combustion (exhausting to a vacuum) is the flattest compared to the others, decreasing by less than 5% of optimum **Specific Impulse** (ISP) over a **stoichiometric** range of 5:1 to 10:1 oxidiser:fuel ratio.

So you'll still get plenty of thrust even if your mixture ratio of nitrous to fuel is way off 7:1, which is good if your test rig can't give you accurate figures to let you tune up the motor: the first few flights will still be adequate, provided the motor doesn't melt.

3) Like bottled $C0_2$, nitrous is **subcritical** at room temperature meaning that both a liquid and a vapour phase can coexist within a *closed* tank.
I'll elaborate on this shortly, but the gist of it is that the moderately dense liquid phase of nitrous can therefore be stored in a compact tank on the pad in the British climate.

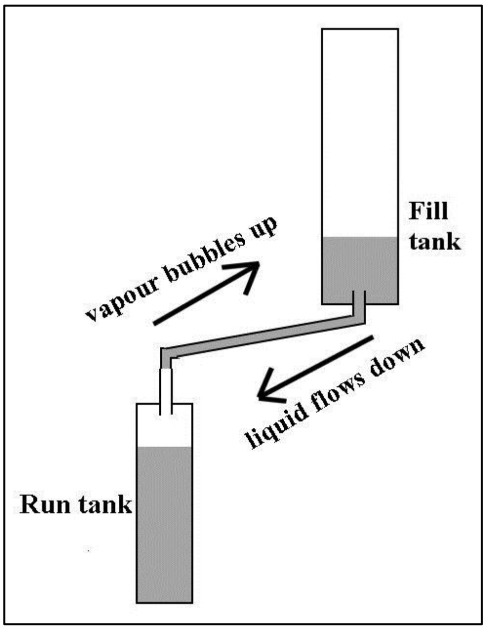
4) At room temperature, Nitrous is *only just* subcritical by a few degrees.
This is nitrous's most unexpectedly useful property, because this close to the Critical temperature, small drops in tank pressure cause large-scale production of extra vapour. This extra vapour strives to maintain the tank pressure at high value as the tank empties.
A traditional blowdown system, e.g. using an ideal gas such as helium, loses tank pressure at a much higher rate during the burn.
This willingness to vapourise with small pressure drops means that the nitrous will vapourise within the orifices of even the crudest injector, typically even a simple single hole.
Injector design is therefore trivial, though noticable improvements still can be gained from more traditionally complex injector designs. (Don't use orifices larger than about 1.5mm diameter for example.)

5) As an added bonus, the pressure of the nitrous gas phase (termed the 'vapour pressure') is seriously high at room temperature, at around 55 Bar (800 PSI).
The gas phase can therefore be used aquajet-style to squirt the liquid phase into the combustion chamber at very high pressure. Some groups call this 'Vapak' pressurization (Vapour pressurization).
This means you can tweak the combustion-chamber to be at almost this high a pressure and the nitrous will still run downstream (in a pressure sense) into the chamber.
The higher the chamber pressure, the higher the Specific Impulse of the motor, particularly at low altitudes. Our AspireSpace hybrids run at about 35 Bar chamber pressure, which is about as high as you can get whilst still having enough of a pressure drop between tank and chamber to prevent a screaming motor.

6) Nitrous has to be raised to a moderately high temperature before it will decompose and release its oxygen.
This is very good from a safety point of view, but it does mean that a lot of heat has to be pumped into the nitrous from some other source at ignition, or the hybrid simply won't light-up.
Once the plastic fuel is burning though, the temperature in the combustion-chamber is high enough to decompose the rest of the nitrous as it feeds-in from the tank during the burn.

Filling the run tank:
I tend to call the beefy container supplied with the nitrous the **fill-tank**, whereas the lightweight tank inside your rocket-vehicle that it fills I call the **run-tank**. (The term 'fuel tank' is just plain wrong; the fuel is the plastic in the combustion chamber.)

The run tank is filled using a difference in pressure between the fill and run tanks:
The pressure difference (or 'head') caused by gravity when the run tank is connected to a fill tank that is physically higher than it will fill the run-tank with the denser liquid phase, while the lighter gas phase will bubble back up into the fill tank. This is how we fill our Rickrock Mk.2 hybrid:



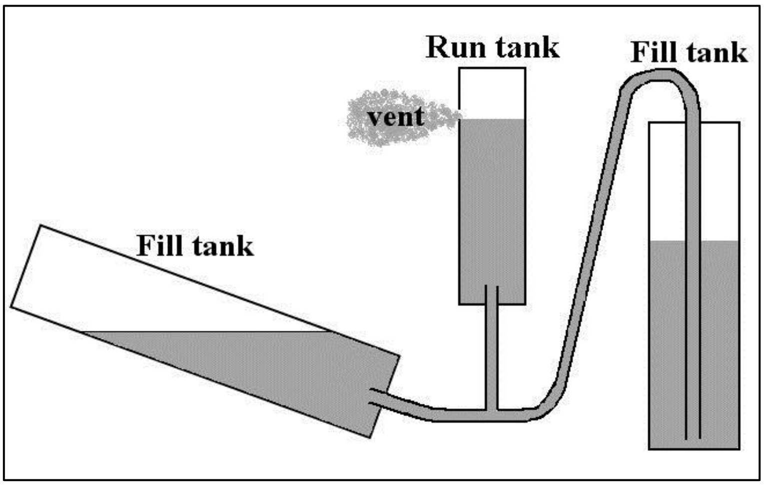
Alternatively, the run-tank has a vent-hole in it which is open to the atmosphere. This lowers the pressure in the run-tank relative to the fill tank (see diagram below).
Then the massive pressure difference between the inside of the fill-tank and the outside air will happily carry the nitrous several metres 'uphill' into the run-tank, so the fill-tank can then be physically lower than the run tank; it is typically lying on the ground whilst the run-tank is up in the rocket up on the launcher.

A question often asked is how full can you fill the run tank?
As we'll see later in this article, the expulsion of the liquid nitrous phase out of the tank during the burn ***is not*** a simple blowdown process, because the nitrous vapour is definitely not an ideal gas.
Nitrous performs much better than this, and infact our test-firings and simulations show that the graph of tank pressure drop with time (during the firing) does not depend upon the amount of nitrous vapour originally in the top of the tank, so you could fill the tank completely full of liquid.
But if the run-tank is to be completely sealed after filling, but then left for some time before firing, then for safety reasons (see our 'hybrid safety' paper, hydraulic overpressure), a small percentage of the tank volume should be deliberately left free of liquid to allow for liquid nitrous expansion with the ensuing increase of temperature.

Vents

On many hybrid systems, this 'ullage' or 'head-space' of gas is achieved by a vent-hole or vent-pipe with an inlet situated slightly below the top of the tank; the outlet is open to the atmosphere outside.
A vent works exactly like the overflow outlet on a bathtub in that the liquid never fills higher than the vent. (Provided that you fill it reasonably slowly.)
The outlet of the vent-pipe can be higher than the vent inlet if required, because the massive pressure difference between inside the tank and outside will happily carry the nitrous several metres 'uphill'.

As soon as the liquid nitrous reaches the level of the vent, you'll see the plume issuing from the vent thicken and whiten appreciably, and that's the time to stop filling.
A dark background behind the vent outlet aids this visual check.
If your hybrid design allows, now's also the time to close the vent hole to stop the loss of nitrous.



Most commercial nitrous hybrid systems keep the vent open permanently, therefore nitrous is continuously being lost.
Although a small enough vent diameter will keep the tank pressure high for some time, this progressively lowers the tank vapour-pressure over time as discussed below, so such a design has to be launched *immediately* after filling.
Faff around on the pad for too long, and significant thrust is lost.

In the above diagram, the fill-tank on the left has to be tilted-up to get liquid phase out, whereas the fill-tank on the right has a 'dip-tube' running down inside it so that it can be sat upright. When you purchase your nitrous, remember to ask whether the fill tank has a dip tube fitted or not.

Subcriticality and Supercriticality:
The apparent simplicity of nitrous hybrids comes at a price.
The nitrous is typically at a temperature where its physics is anything *but* simple, but as in every other branch of rocketry, do thy homework to get thy max performance.

Most substances, below a Critical point (each substance has its own Critical temperature and pressure), can exist as more than one phase simultaneously; they are then termed subcritical. For example any combination of two of the solid phase, liquid phase, or gas phase, can exist together in a tank in 'phase equilibrium', or even all three at the same time at the 'Triple point.'
Nitrous oxide sitting inside a closed container at room temperature is subcritical: partly liquid, and partly gas which being less dense collects at the top of the container.

Strictly, the term subcritical is taken to mean 'just subcritical, but near to the Critical point' but this applies to nitrous as we'll encounter it.

Nitrous properties
Below is a table of nitrous properties reproduced from Ref.1.
ρ is the symbol for density.
Note how the vapour pressure and vapour density increase with increasing temperature, whilst the liquid density decreases with temperature.

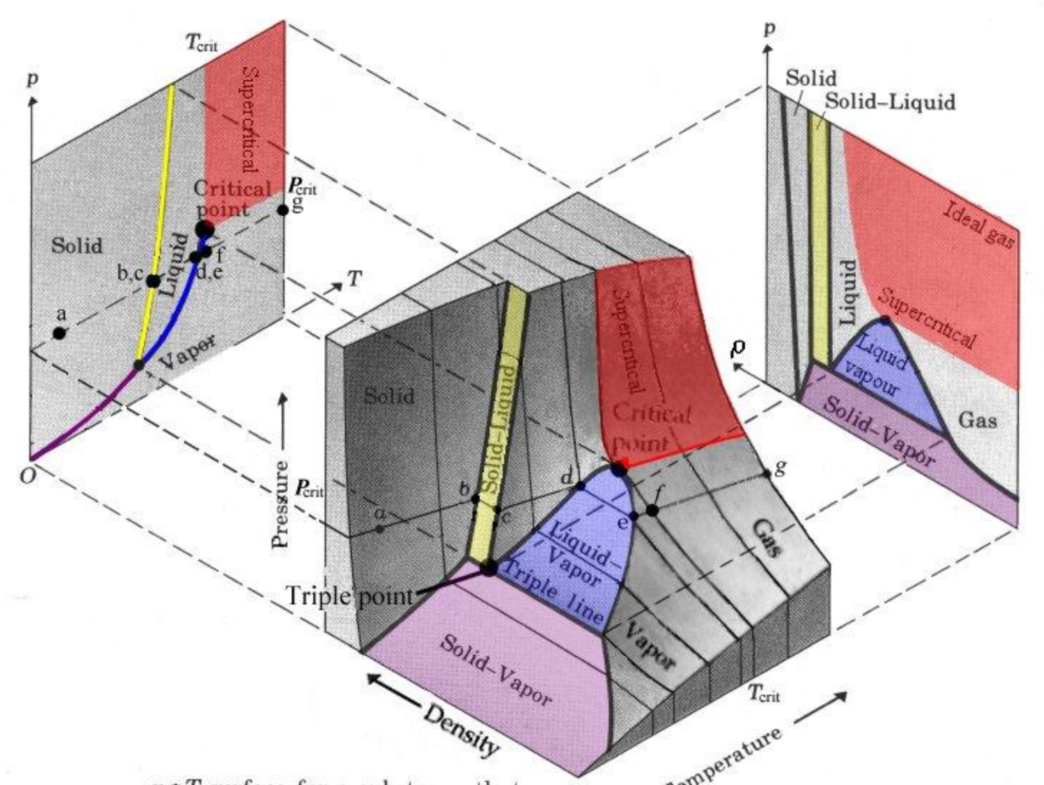
Temperature degrees. C.	Vapour Pressure Bar Abs	ρ_{liquid} kg/m^3	ρ_{vapour} kg/m^3
—20	18.01	995.4	46.82
—15	20.83	975.2	54.47
—10	23.97	953.9	63.21
—5	27.44	931.4	73.26
0	31.27	907.4	84.86
5	35.47	881.6	98.41
10	40.07	853.5	114.5
15	45.10	822.2	133.9
20	50.60	786.6	158.1
25	56.60	743.9	190.0
30	63.15	688.0	236.7
35	70.33	589.4	330.4
T_{crit} 36.42	**72.51**	**452.0**	**452.0**

Firstly, a definition:
The word 'vapour' is usually used to refer to a gas when it's below its Critical temperature and pressure, and so is existing alongside some other phase.
It's purely a matter of context: there's no physical difference between a vapour and a gas, they're exactly the same thing. Technically, nitrous vapour is 'saturated'.

Living at the bottom of Earth's atmosphere as we do, all of our experience of phase changes, usually of water, occur with a constant pressure of 1 atmosphere around us, which usually swamps the results of our experiments.
If the atmosphere wasn't there, water would behave quite differently from our usual experience.
To start with, water's subcritical below 374 degrees C so there are always at least *two* phases present below this Critical temperature.
One phase may well be much less obvious than the other though, infact it's only when the temperature has climbed to 100 degrees C that the pressure of water's vapour phase gets as high as the atmosphere around it.
What we call boiling is when bubbles of water vapour can exist without getting squashed flat by the pressure of the Atmosphere.
So though we're used to thinking that only liquid exists below 100 degrees, and only gas above 100 degrees, this is actually a high school physics simplification. This is Britain after all; we do get the odd rain-cloud.

Nitrous goes supercritical at plus 36 degrees C, so it's very easy to overheat it into supercriticality:
In the heat of the desert launching campaigns in the United States, the nitrous in several hybrids went supercritical.
Supercritical nitrous requires special injector design, so almost all thrust was lost using the standard injectors.

Here's a 3-D graphical representation (not to scale) known as a phase diagram, of the physical properties of any substance that expands on melting, such as nitrous oxide.

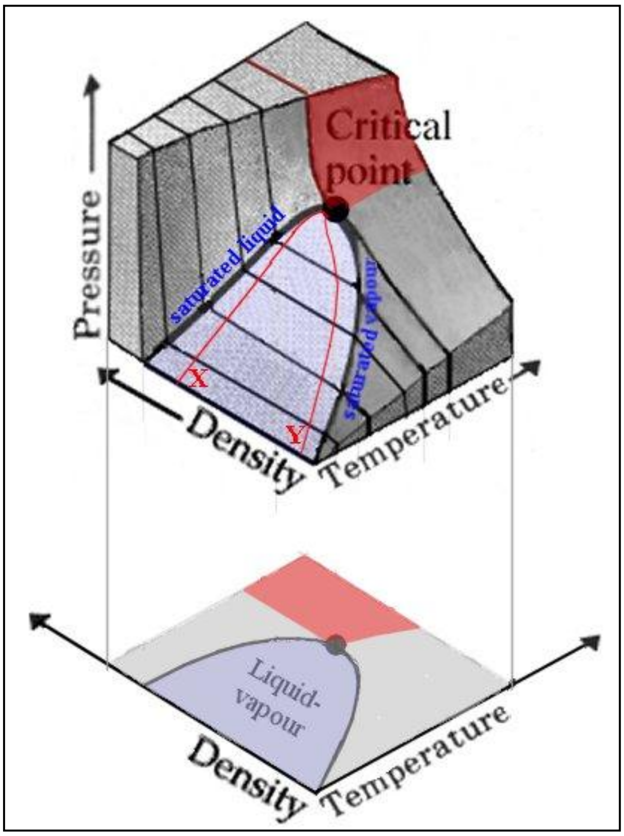


$p\rho T$-surface for a substance that expands on melting. Projections of the surface on the pT- and $p\rho$-planes are also shown.

The slopes of this chunk of 'mount nitrous' represent the values that nitrous physically can exist as; pressure being shown as height.
The path a-g on the upper diagram shows the 'isobaric path' (constant pressure contour), i.e. the experience we're used to with water under the constant pressure of 1 atmosphere around us as described above.
b-c and d-e show the sudden changes of phase at constant temperature that we're used to. (Actually, water is one of the few substances that contracts on melting, so water's phase diagram has 'c' at a higher density than 'b'; its solid-liquid 'cliff' faces away from us instead of towards us as shown for nitrous, but in all other respects the shape of the 'water mountain' is the same.)

On our planet, nitrous' vapour pressure is well above the pressure of the Atmosphere at the temperatures we'll play with it: Boiling point for nitrous is about minus 90 degrees C. So any air trapped in our nitrous tanks that doesn't immediately get squirted out of the vent hole by nitrous' high room-temperature vapour pressure might as well not be there.
The tank behaves as if it contained only pure nitrous.

In this diagram we zoom-in on the range of pressures and temperatures we'll encounter in rocketry.
The density graph shows the view from 'above'.

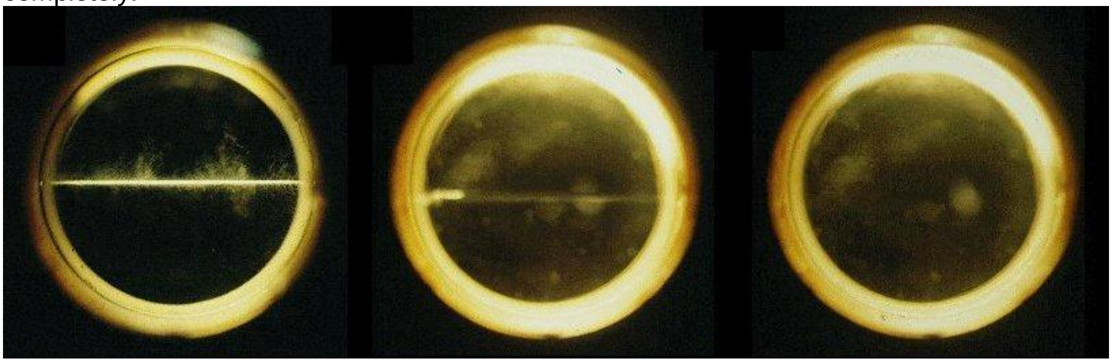


The liquid-vapour area describes what's happening in your tanks: a subcritical region where both the liquid and vapour phases coexist.

When heated, the liquid phase of nitrous follows the saturated liquid line on the graph whereas the vapour phase follows the saturated vapour line.
The series of parallel lines (parallel to the density axis) that cross lines X and Y are known as 'tie-lines', and it's a *convention* to represent how much mass of each phase there is (as a fraction of the total mass in the tank) by the position along the tie-line.
So by this *convention* (each phase *actually* follows its respective saturation line), the exact path up the coloured section depends upon what fraction of the mass of the substance was in the form of each phase when you started heating it:
For example, path X would be a tank of nitrous mostly filled with liquid, whereas path Y would be a tank of nitrous with mostly vapour in it.
By this *convention*, the liquid saturation line is therefore the path of a tank completely full of liquid that is warming up, whereas the vapour saturation line is the path of a tank completely full of vapour.
(In the upper diagram, lines b-c and d-e are tie-lines.)

Notice that as the temperature increases, the density of the liquid saturation line decreases while the density of the vapour saturation line increases.
This phase diagram is based on real data (see the table above from Ref.1): at the Critical point, the densities do become the same; the two phases merge into one single phase, so paths X and Y both pass through the Critical point.

These are photos taken through a tank window as a substance is heated (left to right) to its Critical point. The line of the liquid surface (the meniscus) disappears: the phases merge completely.



Supercritical nitrous can therefore be regarded as either a super-dense gas, or a low density liquid.
At much higher temperatures, the density of supercritical fluid drops much lower: oxygen or nitrogen at room temperature are well-supercritical, hence we refer to them as 'ideal' gasses at these conditions.

Looking at the density versus temperature diagram, you can also see that the change in density of both phases of nitrous per degree change in temperature is largest (steepest) just before the Critical point.
It turns out that the change in vapour pressure per degree C. is also largest just before the Critical point.
For nitrous, even the Scottish climate is still rather close to it's Critical temperature of 36 Degrees C, so sadly, you suffer *big* changes in pressure and density with *small* changes in temperature.
A whopping two Bar decrease in vapour-pressure per degree C decrease in temperature is typical in Britain, so if your nitrous gets too chilly, you'll get a lot less pressure in the tank, so obtain a lot less thrust than you expected.

This close to the Critical temperature, the nitrous *vapour* phase is actually moderately dense and can't be ignored; it has a sizable mass inside the run tank. (and inside the combustion-chamber eventually.)
Conversely, the liquid phase isn't terribly dense, and is progressively less dense at it is warmed: heat it too much and you won't get as much mass of liquid in the run tank's internal volume. (but chill it too much and you lose a lot of vapour-pressure, select your own preferred temperature.)
From the above table of nitrous properties you can see that at 15 degrees C. (standard U.K. day) the liquid phase is only six times denser than the vapour phase.

Historically, it is the liquid phase that is used in the combustion-chamber. The vapour phase then causes extra thrust after the liquid runs out, but its lower density means that the burning is considerably fuel-rich, so the extra thrust it gives is less. More on this later.

<u>Changes in liquid/vapour proportion due to temperature alone:</u>
Going back to the lower diagram, look closely at the tie-lines, recalling what they represent, and you'll notice something odd about the paths X and Y.
The ratio of liquid to vapour within a closed run-tank changes with temperature.
This means that the amount of *liquid* nitrous that you *think* is in your run-tank will change over time if you don't take care to keep its temperature constant between the time that you *start* filling and the time that you launch.

So while it may at first seem a good idea to pre-chill the run-tank to get a good fill of dense liquid phase in there, after several minutes of faffing on the pad the nitrous has warmed up and so the situation has changed.

The nitrous is contained inside the fixed volume of the closed bottle that is the run tank, and so it's mass can't change with time:

$m_{total} = m_{liquid} + m_{vapour} =$ a constant (1)

So it's forced to self-adjust so that it can physically fit inside the tank as the densities of the two phases change with temperature.
The way it physically alters the volumes of the liquid and vapour phases is that a rise in temperature causes some of the liquid to vaporise into vapour, whilst a drop in temperature causes some of the vapour to condense into liquid.

It's forced to follow a volume formula: $V_{vapour} + V_{liquid} = V_{bottle}$ or, $\frac{m_{liquid}}{\rho_{liquid}} + \frac{m_{vapour}}{\rho_{vapour}} = V_{bottle}$

(2) where ρ is density, m is mass.

Actually, this 'self-adjustment' phenomenon is very similar to a reversible chemical reaction:
Temperature is defined as the *average* speed of the molecules of the nitrous: some are moving slower than the average, while some are moving faster, possibly fast enough to break away from the liquid phase and become part of the vapour. This is known as evaporation.
Conversely, some of the slower vapour molecules that 'impact' the liquid phase remain as part of the liquid, a process known as condensation.
At equilibrium (where the nitrous has reached constant temperature and pressure), the rate of condensation is exactly balanced by the rate of evaporation, so no net change occurs with time.
It's only when the nitrous is no longer in equilibrium that one of the rates exceeds the other, and an overall change occurs.

This all occurs within your closed run tank and so you can't see it happening!
Worse still, the total mass of nitrous in the bottle remains the same of course, so weighing scales won't pick up any changes in the proportion of liquid to vapour.

The following resolves this problem:

Fill calculation (after closing the vent valve):
Assuming that you filled the run-tank slowly then you know what mass of nitrous went into the tank, because the volume of tank above the vent-hole should have been vapour alone, and the volume of tank below the vent-hole should have been liquid alone.
So for example if the head space was 15% of the tank volume, then just at the end of filling:

$\frac{m_{vapour}}{\rho_{vapour}} = 0.15(V_{bottle})$ (3) and $\frac{m_{liquid}}{\rho_{liquid}} = (1-0.15)(V_{bottle})$ (4)

If you don't have weighing scales, these two combine to give:

$m_{total} = m_{vapour} + m_{liquid} = \rho_{vapour}(0.15)V_{bottle} + \rho_{liquid}(1-0.15)V_{bottle}$ (5)

The densities of the saturated liquid and saturated vapour phases can be read off of a lookup table such as given above from Ref.1
A run-tank pressure-gauge is invaluable here, perhaps read using binoculars, to discern what temperature caused this run-tank vapour-pressure reading; it may not have reached ambient temperature yet.

The changes in the proportions of the two phases after some time when the temperature has changed (noted by a change in the vapour pressure reading) can then be calculated by rearranging equation (2) and combining with equations (1) and (5) :

$$m_{liquid} = \frac{\left(V_{bottle} - \frac{m_{total}}{\rho_{vapour}}\right)}{\left(\frac{1}{\rho_{liquid}} - \frac{1}{\rho_{vapour}}\right)} \qquad (6) \qquad\qquad m_{vapour} = m_{total} - m_{liquid} \qquad (7)$$

where the densities are those at the new temperature, and m_{total} and V_{bottle} have of course remained constant.

The classic mistake is to forget that pressure gauges measure relative to the atmosphere outside their casing; so one must add one atmosphere (1.013 Bar) to the gauge pressure reading to get the Absolute pressure reading required for the lookup table above.
It's good practice to always label your pressure data 'Bar gauge' or 'Bar abs'.
Some electronic pressure-sensors measure absolute: check their data-sheets.

Proportion changes due to outflow:
Because pressure, temperature, and density are connected, if we cause changes in *pressure* within our run-tank, either during filling, or when we empty its contents into the combustion-chamber, temperature changes will then occur.
And as we've seen, temperature changes cause the ratio of liquid mass to vapour mass in the run tank to change.
Several examples of this occur during hybrid operation:

Firstly, the vent-hole relies on the fact that the vapour-pressure inside the run tank is higher than the atmosphere outside, and so an outflow is established.

The vent should either be of tiny diameter, or be a pipe with a restriction of tiny diameter somewhere along it. (0.3mm diameter is typical.)
A large diameter vent is undesirable because it provides little resistance to the flow pouring out of it, so the drop in pressure between tank and outside occurs more within the tank than within the vent hole. (electronics engineers will recall the principle of a Potential Divider).
The nitrous responds to this low tank pressure by vapourising its liquid away large-scale.
Moreover, the flowrate of nitrous leaving via the vent-pipe is much higher, so it'll all disappear after a short time.
Also, a vent produces gas thrust like any rocket, so you want this flowrate to be small if it's venting sideways.

Similarly, when the **run valve** opens, (the valve between run tank and combustion chamber) the gas phase forces the liquid out of the tank in the manner of an aquajet, because the combustion-chamber connected below the tank is also at lower pressure (unless you've made the nozzle throat too small.)
As the tank empties, the liquid level obviously drops, so the volume available to the vapour phase above the liquid increases, so the vapour expands.
And like any gas, its pressure drops as it expands.

Whatever caused the vapour-phase's pressure to drop, be it venting or emptying, the pressure is now lower than it aught to be (it aught to be at its vapour-pressure) and this drop in pressure is 'sensed' by the liquid phase below it.

Some of the liquid phase will then vaporise in an attempt to create more vapour to raise the tank pressure back up to vapour pressure: the lower the pressure (the bigger the pressure imbalance), the higher the vaporisation rate.

Now the process of vaporising liquid into vapour requires energy (called the latent heat of vaporisation), and this energy has to come from somewhere.
The heat energy required is drained from the nearest available source, which in this case is the remaining liquid nitrous itself, which therefore gets cooled by an amount determined by its Specific Heat Capacity. (Heat capacity per kilo of liquid per degree temperature change.)
Oddly enough, my experiments and simulations show that the metal wall of a nitrous tank doesn't give up heat that quickly into the liquid even though you'd expect it to: the metal may be a conductor, but the liquid isn't.
So the metal of the tank can be ignored as a heat-source for pressure changes, *provided* that they occur in a short time, say the 10 seconds or less that are typical of a hybrid firing.

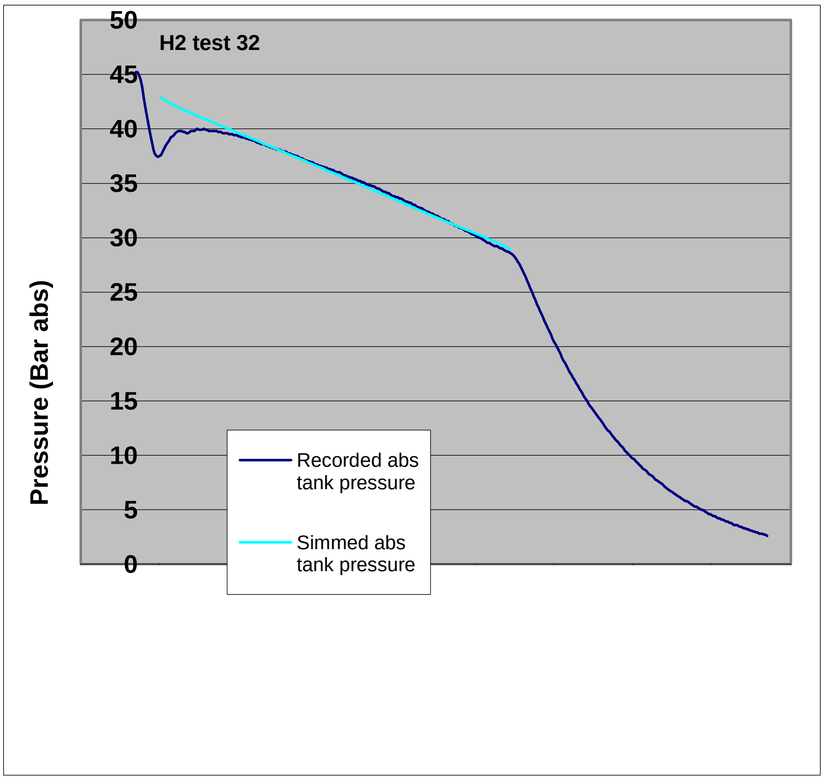
From an engineering point of view, the various thermal layering effects (known as 'stratification') that occur within the nitrous, wherein the liquid and vapour closest to the boundary between liquid and vapour aught to be the coldest because that's where the vaporisation occurs, can be completely ignored in computer modelling.
Perhaps this is because the colder nitrous will be denser, so will try to sink to the bottom of the tank and so the liquid gets evenly mixed. Also, changes in pressure affect the whole of the nitrous at once.

Experiments show (Ref. 4 and confirmed by our own) that other effects cancel stratification out, and so the liquid and vapour can be simply modelled as 'blocks' at uniform temperature.

Read our paper 'Modelling the tank emptying' for a mathematical model and software to model the nitrous leaving the tank.

This cooling of the remaining liquid (and therefore any future gas to be vaporised from its surface as the emptying progresses) means that the vapour-pressure (the tank pressure) will slowly drop over the burn time.

In this graph, burnout was taken as the point when the liquid phase ran out (the graph suddenly steepens):



The lower the pressure drops below vapour-pressure, the more vapour is required to raise the pressure back up, and the more chilled the liquid-phase becomes as it provides this vapour. This is why leaks in any pipe-joints carrying the liquid phase of nitrous oxide show up as regions covered in ice; the nitrous sucks heat out of the atmosphere as it leaks out to atmospheric pressure and vaporises, freezing the water-vapour in the air around the leak.

It'll freeze your eyes, face, or hands too if they're near a leak: **wear goggles and gloves when you work with nitrous.**

So if you crank open the vent (to the atmosphere outside) to huge diameter in an attempt to perform a quick fill, you'll lower the tank pressure *way* below vapour pressure, and so the nitrous will vaporise big-time, chilling itself seriously cold in the process as it drains heat from itself.

If the leak is plugged, for example by shutting a valve on the vent-line, or by shutting the **run valve** mid-burn, liquid will continue to vaporise inside until the vapour-pressure is restored. (albeit the lower vapour-pressure you get at a colder nitrous temperature.)
Then as heat from outside *slowly* trickles back into the liquid through the tank walls (this takes a long time, so the tank *does* count as a heat source), the vapour-pressure will slowly rise again until the liquid is back at ambient temperature, then no more heat can flow in.
This can take a good 15 minutes even for small run-tanks though.

If the nitrous was originally very chilled (from too fast a fill) an awful lot of it will vaporise during this time, so what started out as a run tank nearly full of liquid might well now be mostly vapour.

The rate of decrease of tank pressure with time (the slope of the graph above) depends on how quickly you empty the tank:
Experiments at Surrey Satellites Technology Ltd have shown that if the nitrous is emptied at a tiny flowrate, less than 10 grams per second or so, then the tank pressure remains constant because the small inflow of heat through the tank wall is just enough to compensate.

The vaporisation of the liquid phase into gas is known to resemble conventional 'boiling':
Analogous to the phenomena of supercooling, the boiling of water at atmospheric pressure sometimes doesn't occur at the boiling temperature of 100 deg C.; sometimes the temperature continues to rise higher until some tiny dent or scratch in the container wall (called a 'vapour nucleation site') forms a bubble that breaks loose and sets the wholescale boiling off.
Chemists often drop 'boiling stones' (small porous bubble-producing 'pebbles') into beakers to ensure that boiling occurs at the temperature expected.

Experiments from Ref. 4 show that mechanical agitation will also trigger boiling in fluid that aught to be boiling but as yet is not.
Once any tiny amount of local boiling kicks in, the resulting bubbles agitate the liquid, greatly increasing the boiling rate, and this feedback mechanism then cascades to produce serious boiling.
We see this in our hybrid tests too: the graph shown on the previous page is typical, and shows a downward kink at the start.
It seems that when we open our hybrid **run valve**, the initial drop in the liquid level catches the nitrous 'unawares', and so there is very little vaporisation, and the ensuing graph of pressure drop has the characteristic steepness of an expanding ideal gas.
Once the hybrid fires up though, the ensuing motor vibrations shake the rocket, and hence the tank.
This shaking appears to trigger large-scale boiling, and the tank pressure graph then rises sharply, before descending at the gentler slope that you'd expect from a vaporising subcritical fluid.

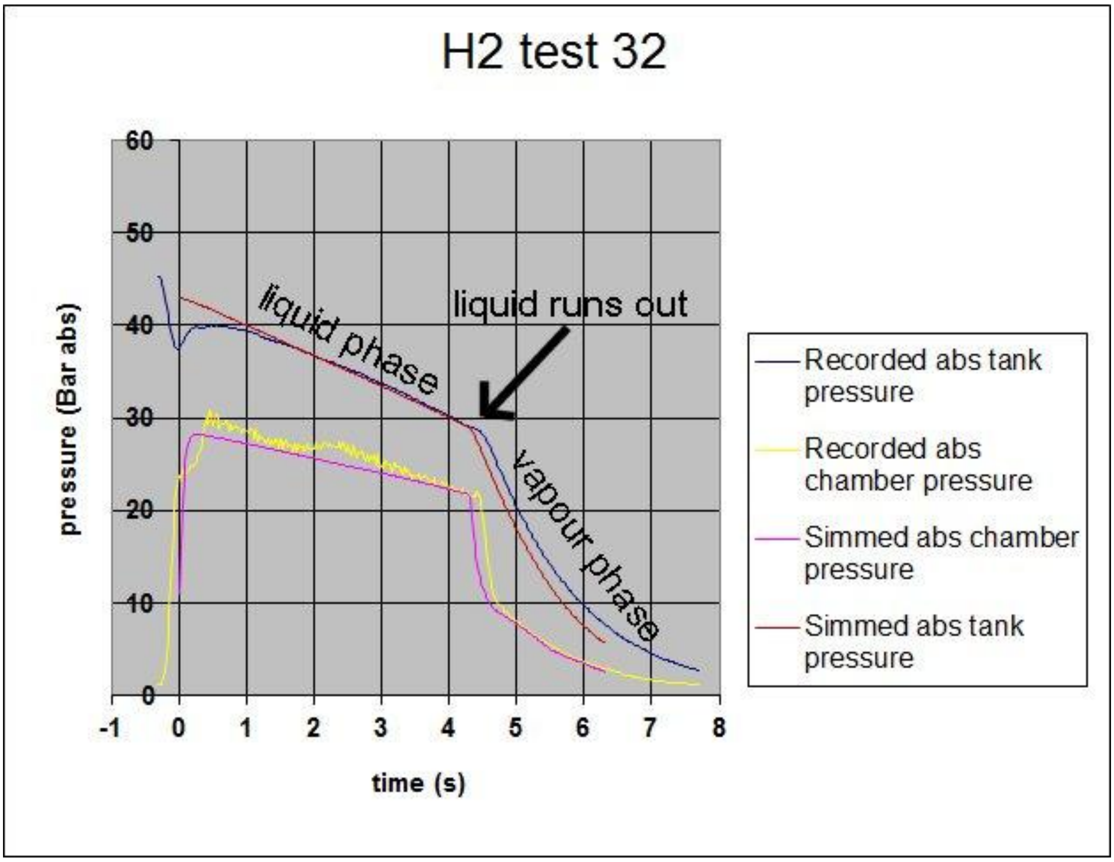
<u>The vapour-only phase</u>
After all the liquid nitrous has run out of the run-tank, there will still be some vapour remaining. Even if you started with a tank completely full of liquid, some vapour will have been created as the tank emptied.

This vapour is dense enough to erode the hybrid fuel grain and so produce thrust, though it burns fuel-rich (too little oxidiser) which lowers the **Specific Impulse**, and this ‘vapour-only’ phase doesn’t last long.

From our hybrid firing data, we’ve learned a few surprising things about this ‘vapour-only’ phase of combustion:

1) It transpires that the pressure loss that occurs as the vapour flows through the injector orifices is identical to when the liquid was flowing through it. This proves that the liquid vaporises completely to vapour inside the orifices as Bernoulii’s principle causes a pressure drop (flow velocity goes up, pressure goes down).
2) The vapour emptying out of the run-tank very nearly follows an ‘isentropic’ process. That means that very little energy is wasted (negligible increase of entropy) during the emptying, and no heat is transferred from the tank walls to the vapour.
3) Therefore the vapour pressure and temperature drop rapidly as the tank empties and the vapour expands.
4) The vapour is not an ‘ideal gas’. Intermolecular forces (the forces between the vapour molecules) are noticeably at work, so nitrous vapour expands differently to that of an ideal gas.

With the above in mind, a simple mathematical model will simulate the tank emptying (see our paper ‘modelling the nitrous run tank emptying’):



On the pad: a quick recap of what all this esoteric physics means to you on the launchpad:

1) If you plug the vent-hole after filling to preserve a higher tank pressure and so better performance, your tank better have a head-space or your innocent-looking run-tank may hydraulically overpressure (go boom, see our 'hybrid safety' paper) a few minutes after filling.
2) It is the liquid phase that we use in the combustion-chamber, so we want to preserve as much of this as possible. Though the vapour phase will cause extra thrust after the liquid runs out, its thrust is much lower.
3) Use a small vent-hole so that the run-tank fills slowly, or a lot of the liquid you put in there will have vaporised by the time you fire it, if it hasn't all leaked away out the vent.
4) If you fill the tank too quickly by cranking open the vent-hole, you'll over-chill the nitrous, so if you fire it straight away, you've got very little tank pressure which will reduce combustion-chamber pressure and so kill most of the thrust.
5) If you quick-fill and then wait several minutes before firing, then (assuming you've plugged the vent) there will be much less liquid in there than there was 5 minutes ago; it'll have vaporised into vapour in the tank.
6) It may seem cool (sic) to pre-chill the tank to densify the liquid phase to get a lot in there, but you'll get all the problems due to over-chilling mentioned in 3) 4) and 5).
7) If it's cold outside, warm the run-tank (remotely!).
8) If it's too hot outside, chill the run-tank to keep the liquid density reasonable, or even to prevent the nitrous going supercritical.

Glossary:

Fill-tank:
The commercial container supplied with the nitrous.

Run-tank:
The lightweight tank inside your rocket-vehicle that is filled from the fill tank.
(In a conventional hybrid, the term 'fuel tank' is just wrong as the fuel is the plastic in the combustion chamber.)

Run-valve:
The valve that lets the nitrous flow from the run-tank into the combustion chamber.

Specific Impulse (ISP):
The miles-to-the-gallon of a rocket propellant combination as it were, equal to the thrust generated (Newtons) per Newton weight of fuel used per second.
Or, thrust per mass flowrate of fuel (kg per second), times the constant of one gravity (9.81).
Units are seconds.

Stociometric:
The fuel to oxidiser ratio that yields best performance (usually that for best **Specific Impulse**).

Subcritical:
A substance at a temperature below its Critical temperature, so that a liquid and vapour phase can both coexist.

Supercritical:
A substance at a temperature above its Critical temperature, so only a dense gas can exist.

References:

Ref. 1: Engineering Sciences Data Unit (ESDU) sheet 91022,
Thermophysical properties of nitrous oxide.
Available in hardcopy from some U.K. University libraries, or accessible over the Web to students with an ATHENS password.

Ref. 2: University Physics 6th edition
Sears, Zemansky, and Young
Addison Wesley world student series ISBN 0-201-07199-1

Ref. 3: Space Propulsion Analysis and Design
by Ronald .W. Humble, Gary .N. Henry and Wiley J. Larson
McGraw Hill Space Technology Series ISBN 0-07-031320-2

Ref. 4: Dr Bruce P. Dunn
University of British Columbia and Dunn Engineering
Several articles on self pressurised peroxide rockets and experiments on propane tanks, as well as email communications with the author on the subject of numerical modelling of the tank liquid emptying process; many thanks.

Ref. 5: Engineering Thermodynamics, Work, and Heat transfer (S.I. units) 4th edition
Rogers and Mayhew
Prentice Hall ISBN 0-582-04566-5

Gary barnhard - Re: ACS parameters for Ed Belbruno - please check

From: Ethan Chew <spacefelix@gmail.com>
To: Nastia Soukhareva <nastia.soukhareva@gmail.com>, Eric Shear <renegade.om...
Date: 2/6/2016 12:41 AM
Subject: Re: ACS parameters for Ed Belbruno - please check
Cc: Eric Dahlstrom <Eric.Dahlstrom@internationalspace.com>, Gary Barnhard <B...
Attachments: Trajectory Design for Alpha CubeSat.pdf; Belbruno Trajectory & Propulsion Capabilities Analysis.xlsx

Alright,

Based off the attached Belbruno trajectory developed, I have determined the propulsion system capabilities in terms of total impulse and total runtime to be able to meet his trajectory's DeltaV requirements.

I've evaluated our baselined NOX-Aluminized Paraffin hybrid HTSD motor as well as our LTLD options from HYDROS, Phase 4 CAT Ambipolar and Busek BIT-1 using propellants that met Belbruno's minimum DeltaV of 180 m/s (per manufacturer's specsheets). Spreadsheet attached.

Conclusions:

- The reduced DeltaV of the Belbruno trajectory allows us to eliminate the combination HTSD-LTLD propulsion system and free up mass and volume for payloads.
- Some propulsion systems have an overly long propulsion runtime (on the order of months to days) to impart the required impulse for the required DeltaV. Belbruno has recommended that propulsive phases be kept to a few minutes per phase.
    - This leaves us with a NOX-Aluminized Paraffin Hybrid HTSD motor or a HYDROS as our best options for propulsion.
- Propellant mass and volume are no longer limiting factors on the mission due to the reduced DeltaV. There is room to add additional propellant and open up further destinations for the ACS mission.

I am going to take the next hour to develop this into a report. Nastia, my apologies for the long time. I will do my best to get this to you shortly.

- Ethan

On Fri, Feb 5, 2016 at 2:06 PM, Ethan Chew <spacefelix@gmail.com> wrote:

> Comments acknowledged and in-work.
>
> Will develop the total impulse calculations for CAT Ambipolar.
>
> Also will add:
>
> - HTSD Propulsion System Engine Chamber and Nozzle design calculations and method.

- Development of combination HTSD & LTLD propulsion system volumes and masses based on updated trajectory DeltaVs and planning (reference to loads for structural system).

Expect delivery by 6pm CST today 2/5/16.

On Fri, Feb 5, 2016 at 7:50 AM, m d <2mdoty@gmail.com> wrote:

> My edits and comments are in green
>
> On 2/5/2016 3:49 AM, Ethan Chew wrote:
>
>> Hello,
>>
>> Attached is the latest draft of the ACS Propulsion Report. It is pending the determination of the mathematical and physics-based relationships between Thrust & I_sp and combustion chamber and nozzle characteristics for HTSD and power and propellant atomic mass for LTLD. Otherwise, we can put them as pending/on-going analyses for engineering. Aaron, if you may assist on this, it would be appreciated.
>>
>> Eric S. and Mike, please continue to provide information in the format within regarding HYDROS and CAT Ambipolar.
>>
>> Trajectories team (Eric D. and Gary), please advise on the orange highlighted sections within.
>>
>> Nastia, please use this report to support your propulsive structural loads design report.
>>
>> - Ethan
>>
>> On Fri, Feb 5, 2016 at 5:15 AM, Ethan Chew <spacefelix@gmail.com <mailto:spacefelix@gmail.com>> wrote:
>>
>> Acknowledged and thank-you.
>>
>> For LTLD propulsion, please use propellants that provide a minimum I_sp of 1,000s. Per the GT-1 Trajectory and DeltaV Analysis, this is the minimum I_sp to achieve the LTLD DeltaV required by the ACS deep space and lunar missions with a limited mass (1kg) and/or volume (1U) of LTLD propellant.
>>
>> Also, please analyze for Iodine even if the I_sp does not meet the 1,000s minimum as the high density of solid iodine may allow the propulsion system to store sufficient mass in the 1U volume limit.
>>
>> - Ethan
>>
>> On Fri, Feb 5, 2016 at 5:04 AM, m d <2mdoty@gmail.com <mailto:2mdoty@gmail.com>> wrote:
>>
>> Attached is data sheet for Cat Ambipolar. The table is for 1 Kg dry mass 3U spacecraft with 50 watts power. We would adjust these numbers for 90 watts power and use 14 Kg total

mass with propellant. our maximum propellant capacity will be 14 Kg - our total dry mass, which I am working on.

On 2/3/2016 9:41 PM, Ethan Chew wrote:

By far tonight, here is the template draft of the GT-2-level Propulsion System Report Outline. Propulsion Team, please respond to request below. Trajectories team, please advise per the below.

Sections of Note:

_

_

Evaluation of candidate HTSD and LTLD propulsion systems against baseline propulsion requirements.

TO PROPULSION TEAM: Eric Shear and Mike Doty, please contribute the parameters for the HYDROS and CAT Ambipolar propulsion systems (as well as others) as follows:

* Classification and HTSD or LTLD
o HTSD: Isp < 500s
o LTLD: Isp > 1,000s
o If system has a dual operating mode between HTSD and LTLD, please state 'Dual' and state specifications for both modes.
* I_sp (s)
* Propellants and Propellant Densities (kg/m^3).
* Thrust (N)
* Total Impulse Imparted (N-s)
* Maximum Runtime (seconds)
* Propellant Safety (Compatible with NASA Cabin Safety Standards, Y/N?)
* If applicable, maximum propellant carriage volume (m^3).
* If applicable, maximum propellant storage lifetime (days).
* If applicable, DeltaV (m/s or else it can be calculated from the I_sp).

Trajectory Basis For Propulsion System Requirements

_

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TO TRAJECTORIES TEAM: Eric D and Gary, may you assist in verifying my statements in the report in regards to trajectory, the design and methods of its optimization are correct? I have highlighted within several areas of concern and need for information. Also, if you may provide an expected amount of trajectories DeltaV savings using the optimization methods of Belbruno and more, that would also be appreciated.

Thank-you.

- Ethan

On Wed, Feb 3, 2016 at 3:45 PM, Eric Dahlstrom <Eric.Dahlstrom@internationalspace.com
<mailto:Eric.Dahlstrom@internationalspace.com>
<mailto:Eric.Dahlstrom@internationalspace.com

<mailto:Eric.Dahlstrom@internationalspace.com>>> wrote:

Gary (& Ethan, Rich, and Mike),
Please check this information for Ed Belbruno. After we confirm the values, we can send it to him so he can begin his calculations. We need to get him information today.
- Eric

Eric.Dahlstrom@InternationalSpace.com
http://www.linkedin.com/in/ericdahlstrom
International Space Consultants +1.202.288.0622 <tel:%2B1.202.288.0622> <tel:%2B1.202.288.0622>
210 Waverley St #6, Menlo Park, CA 94025

Alpha Cubesat (ACS) Information for Ed Belbruno

Scenario:
ISS LEO -> Earth escape (C3≳0, >45000 km) [provided by launch provider]
Earth escape -> 4 million km
4 million km -> EML2 halo [note that the halo orbit is not a requirement, but was identified as a staging point]
EML2 halo -> elliptical Lunar orbit (hp>300 km, ra<10000 km)

Delta-v budget: 1500 m/s [after delivery to Earth escape]

Length of mission: 1 year

Spacecraft wet mass: 14 kg
Form factor: 6U cubesat

Ion propulsion:
4 x Busek BIT-1 thrusters (4 x 100 microN = 0.4 mN)
Isp = 2150 s

'burn for a day' delta-v: 86400 s -> 2.5 m/s

BIT-1 Ion Thruster datasheet
http://www.busek.com/index_htm_files/70011950%20RevA%20Data%20Sheet%20for%20BIT-1%20Ion%20Thruster.pdf

--

Prototype
ISARA Solar / Reflectarray

Hardware Revision: A

Deployable Solar Array with Integrated Reflectarray

Applications
- JPL ISARA

Features
- For 3U PUMPKIN MISC 3 nanosatellite busses
- Three-panel design
- Eight SpectroLab® UTJ cells per panel (8S3P configuration)
- Array folds around CubeSat in stowed position
- Incorporates JPL Reflectarray design on underside of panels
- PMDSAS gen. 5 derivative solar panel design
- Extremely stiff, stable and thermally beneficial design with hybrid laminated construction
- With:
  - Compact 90-degree hinges between panels
  - One CSK body hinge between center panel and CubeSat body
  - Flex interconnect to CubeSat body

ORDERING INFORMATION

Pumpkin P/N 717-01105

Option Code	Configuration
/00	standard

Contact factory for availability of optional configurations.
Option code /00 shown.

CAUTION

Electrostatic Sensitive Devices

Handle with Care

CHANGELOG

Rev.	Date	Author	Comments
A	20140324	AEK	Initial version.

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- MISC™
- CubeSat Kit™ and the CubeSat Kit logo

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email: info@pumpkininc.com

web: http://www.cubesatkit.com/
email: info@cubesatkit.com

PUMPKIN™

SPACE SYSTEMS

PMDSAS™ Solar Panels & Arrays

Profile

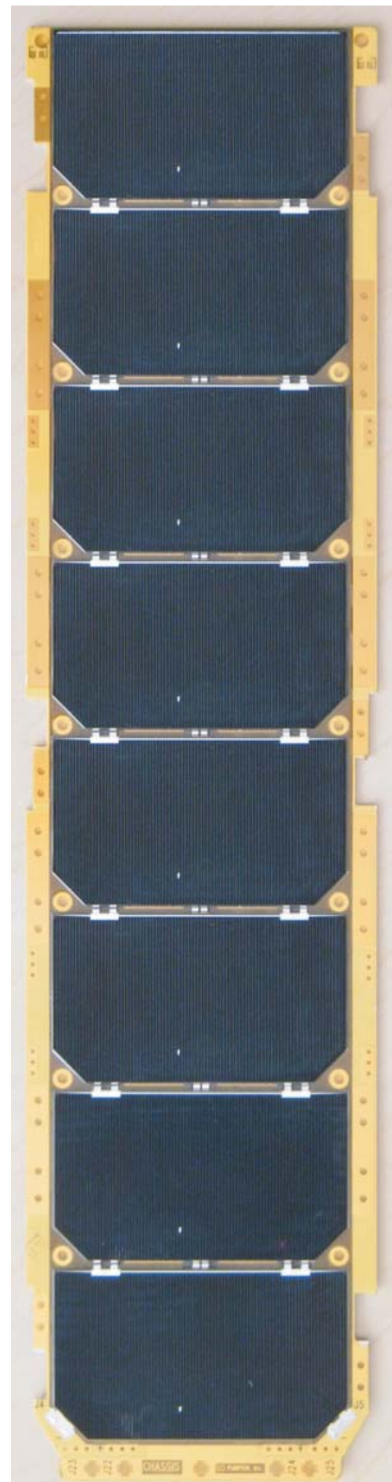
- Pumpkin Modular Deployable Solar Array System
- Lightweight, flexible, volume-efficient and space-proven technology.
- Available in multiple configurations, from 2W to 300W:
  - Fixed panels
  - Deployable panels
  - Deployable arrays
- Available in COTS and custom shapes and sizes.
- Compatible with a wide range of Electrical Power Systems.
- Designed, manufactured, assembled and tested in the USA.

Specifications[1]

Operating Temperature Range (ºC):	-50 to +105
Specific Power (W/kg):	> 90
Stowed Volume Efficiency (kW/m^3):	> 140
Lifetime (yr):	> 2
Fill Factor (8-cell winglet panel, %):	77
Minimum Bend Radius (mm):	< 500
Random Vibe Survival (Grms):	> 11 in all axes
Power per Solar Cell (W, BOL, AM0):	1.05
Maximum Size (cm x cm):	400 x 550
Mass of CubeSat-class panels (g):	22 (1U, 2.1W) 50 (2U, 4.2W) 77 (3U, 7.3W) 82 (3U, 8.4W)

Benefits

- PMDSAS technology delivers extraordinary flatness and light weight, reducing thermal resistance while exhibiting remarkable flexibility and toughness.
- Flexible and modular architecture. Can be customized for a wide range of mission requirements.
- Rapid deliveries (under 1 week from stock).
- Highest-quality panels, made in U.S.A.



PMDSAS 3 8-cell winglet panel. Part of PMDSAS 56W (8S7P) solar panel array that flew on NGC's Caerus/Mayflower in Dec. 2010.

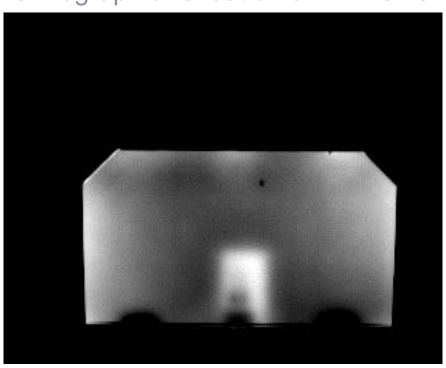
You

- Choose COTS or custom panels.
- Enjoy higher power, greater payload mass and survive extreme vibe levels with PMDSAS solar panels & arrays.

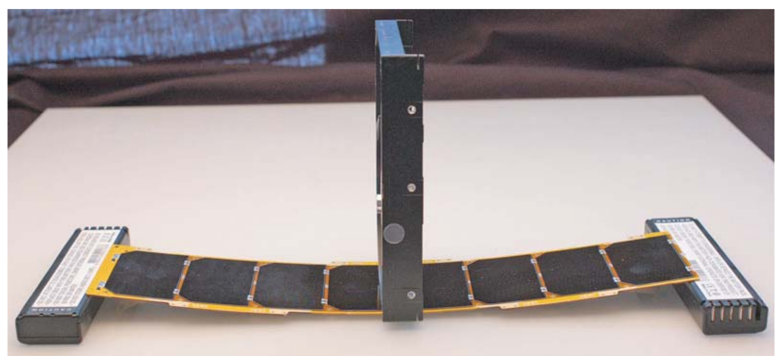
Construction

- Each panel's construction is overseen by a skilled technician.
- All materials used meet NASA outgassing guidelines.
- Multiple redundancies (interconnects, blocking diodes, connections) are employed where appropriate.
- Panels utilize Spectrolab® or comparable triple-junction (min. 28% efficiency) solar cells with integral interconnects, bypass diode and coverglass (CIC).
- Solar cells are affixed to panels via a proprietary and patent-pending PSA-centric procedure. Derived from the Aerospace Corporation's pioneering approach in 2009 that utilized NuSil® CV4-1161-5, with advancements for enhanced flatness, reduced mass, speed of assembly and thermal performance.
- Thermal encapsulation, where required, is achieved via a thermally conductive epoxy.
- Substrate-to-PSA-to-cell design is inherently devoid of trapped bubbles, validated via thermographic testing.
- Panels are built using a variety of substrate materials, optimized for specific applications.
- Integral Kapton® coverlay is used on top and/or bottom surfaces.
- No discrete or hand-wired point-to-point interconnects. All interconnects are integral to the solar cells or the panel substrates themselves.
- Copper layers on substrate material are carefully mapped to ensure maximum possible symmetry and coverage for enhanced heat flux and to minimize local hotspots.
- Panels use a "sea of vias" and other PCB layout and construction techniques specifically tailored for best heat flux and minimal magnetic signature.
- Interconnects and components are soldered with leaded solder to preclude tin whiskers.
- LM335 precision temperature sensors and coarse sun sensors available on most models.
- Custom harnesses available for all models.
- CubeSat-standard Hirose® DF13-series connector fitted as standard. Optional connectors available.
- Isolation resistor connects solar array substrate ground to chassis ground as per NASA-HDBK-4002A.
- Compatible with CubeSat Kit Solar Panels Clips, screw fasteners and RTV/epoxy bonding methods.
- Flatness is maintained throughout the manufacturing process.
- Every unit is laser serialized for tracability.

Thermographic validation of PMDSAS 2



Simple bend test of PMDSAS 3 winglet panel



Fourteenth C1B (AFRL). Has PMDSAS 4 panels

Hinge detail from 56W PMDSAS 3 array

Contact

PUMPKIN™
SPACE SYSTEMS

750 Naples Street
San Francisco, CA 94112 USA
tel: 415-584-6360
fax: 415-585-7948
web: www.cubesatkit.com
email: info@cubesatkit.com

1. All power figures assume 28.5% efficient solar cells of 26.62cm^2 area under AM0 illumination. Specifications are representative of PMDSAS 5. Masses shown are for 0.031" (0.8mm) thick panels. Larger sizes possible.

Specifications subject to change without notice. Made in USA.

707-00805-B 08/2012

PUMPKIN™
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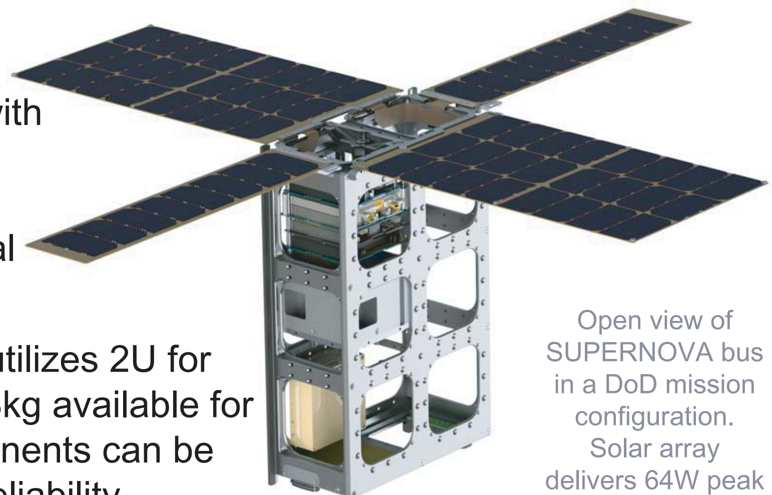
Update Q2 2015

SUPERNOVA™

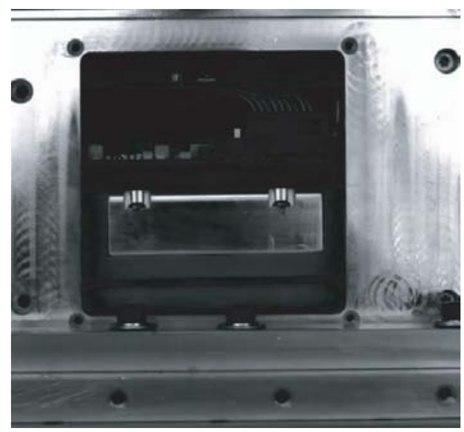
- Pumpkin has delivered the first SUPERNOVA™ block III structures. Developed and tested in partnership with the Air Force Institute of Technology (AFIT), the block III design has a supersymmetric design with six internal unit cells of 100 x 100 x 100mm each.

  A typical SUPERNOVA configuration utilizes 2U for the bus components, leaving 4U and 8kg available for the payload. Any CubeSat-size components can be accommodated inside. A single high-reliability resettable pin puller is used to release all deployables.

  SUPERNOVA™ is compatible with Planetary Systems Corporation's flight-proven Canisterized Satellite Dispenser (CSD). Testing at AFIT has confirmed the exceptional stiffness that derives from the SUPERNOVA design and how internal components are mounted. First flight is scheduled for Q4 2015.



Open view of SUPERNOVA bus in a DoD mission configuration. Solar array delivers 64W peak.



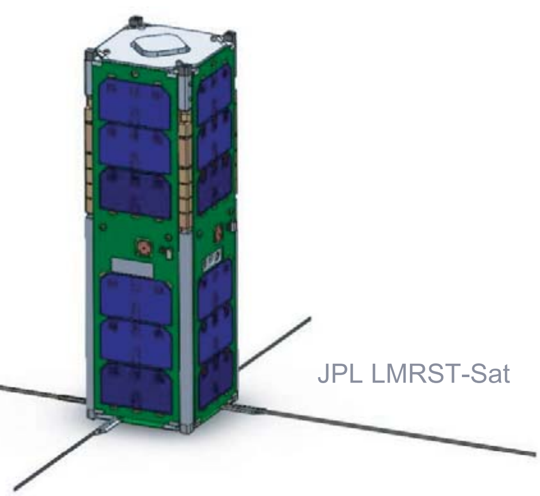
High-speed image of SUPERNOVA undergoing -12dB, -6dB & 0dB random sine vibe testing

NASA

- PUMPKIN delivered its first CubeSat-compatible High-performance Processing Engine (HiPPiE™) to NASA Ames Research Center in February. This 1.5U-size unit is suitable for use in static installations, ROVs, UAVs, aircraft and nanosatellites.

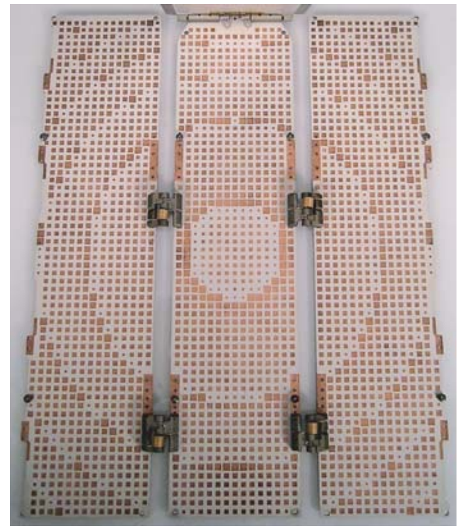
CubeSats

- JPL recently completed environmental testing of its 3U-size LMRST-Sat CubeSat. Stanford's Space & Systems Design Lab (SSDL) designed the bus, and created the flight software, ground station software, bus-to-payload interface, an SGP4 orbit propagator and other subsystems. Pumpkin supplied the CubeSat Kit Pro chassis, five PMDSAS solar panels, the C&DH module and a GPSRM 1 GPS receiver module with dual orbit propagators (Vinti7 & SGP4). LMRST-Sat is on the August 27, 2015 NRO Atlas V launch.



JPL LMRST-Sat

Reflectarray



Reflectarray side of three-panel deployable PMDSAS array

- Pumpkin recently completed the first set of custom PMDSAS solar panels for JPL's ISARA project. This three-panel depoloyable reflectarray array for a 3U CubeSat has dual purposes -- a 24W solar array on top, and a beam-forming Ka-band reflectarray underneath.

  Working with JPL's Spacecraft Antennas Group, Pumpkin has co-developed a manufacturing process that can combine RF radiators or antennas of arbitrary shape and complexity on the underside of a solar panel with solar cells on the top side. This process remains compatible with Pumpkin's extensive deployable array hinge offerings, while maintaining adequate flatness.

Custom

- Standardization and mass production are hallmarks of Pumpkin's approach to product development. Yet each nanosatellite mission has unique requirements. Our specialty is integrating CubeSat systems to increase functionality within a constrained form factor. Whether your requirements are for a particular processor, or a choice of radios, antennas, or other systems, no one can integrate nanosatellite systems like Pumpkin Space Systems. Our proven track record in space, modular architecture, rapid engineering services, supplier relationships and broad assortment of standard components allow us to rapidly reconfigure each spacecraft to suit your particular mission, at attractive prices.

Partners

- Pumpkin Space Systems is seeking complimentary technologies to incorporate into its product lineup. Our goal is to qualify a second source for each major system, and to offer customers multiple configurations based on mission requirements. Advanced solar cells, radios, antenna systems, micropropulsion, and deorbit devices are among the systems we seek to incorporate or upgrade. Certification as a *Pumpkin Space Partner* gives your company access to the highest volume nanosatellite spacecraft market. Contact us if your company currently builds or plans to build high-quality components for small satellites and would like to be included as optional equipment in Pumpkin's expanding MISC family of nanosatellites.

Contact

Pumpkin Space Systems serves demanding government, commercial and educational customers with P-POD compatible nanosatellite spacecraft and buses. Our integrated designs are based on our own flight-proven CubeSat Kit™ components and have completed flight qualification.

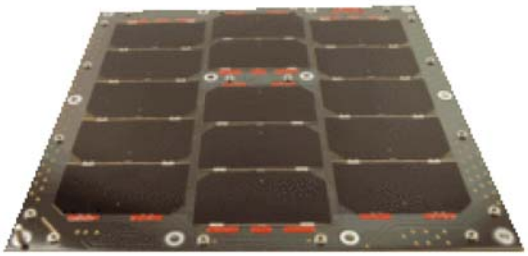
PUMPKIN™ SPACE SYSTEMS

CUBE SAT KIT™

750 Naples Street
San Francisco, CA 94112 USA
tel: 415-584-6360
fax: 415-585-7948
web: www.cubesatkit.com
email: info@cubesatkit.com

Specifications subject to change without notice. Made in USA.

707-00419-K 04/2015



Zoom (http://www.website.com/documents/580/580-large.gif)

6U CubeSat SIDE Solar Panel

Cost:$14,300.00

6U CubeSat SIDE Solar Panel

- High quality PCB Substrate with space-grade Kapton coverlay (picture opposite is not a 6U panel, but is indicative of the size and configuration of a 6U Panel).
- 18 large area triple junction cells with 28.3% efficiency minimum
- Temperature sensor, reverse bias protection diodes and harness connector
- Compatible with CubeSatKit structure (ISIS [and other]structure version available on request)
- Manufactured to Clyde Space qualified, flight heritage processes using space grade materials
- Other sensors available for inclusion on solar panel (i.e. coarse and fine sunsensors). Magnetometers and/or rate sensors can be included, but we advise that these are placed in a more thermally benign location due to the device temperature coefficients.
- Also available with integrated magnetorquers
- All manufacturing at Clyde Space is to our ISO9001:2008 accredited processes by our ESA trained assembly staff (see our Quality Manual (http://www.website.com/about_us/quality_management_clyde_space))
- Solar_Panel_Datasheet (http://www.website.com/documents/2625)
- 3D Model available on request.

Performance Specifications of this Solar Panel:

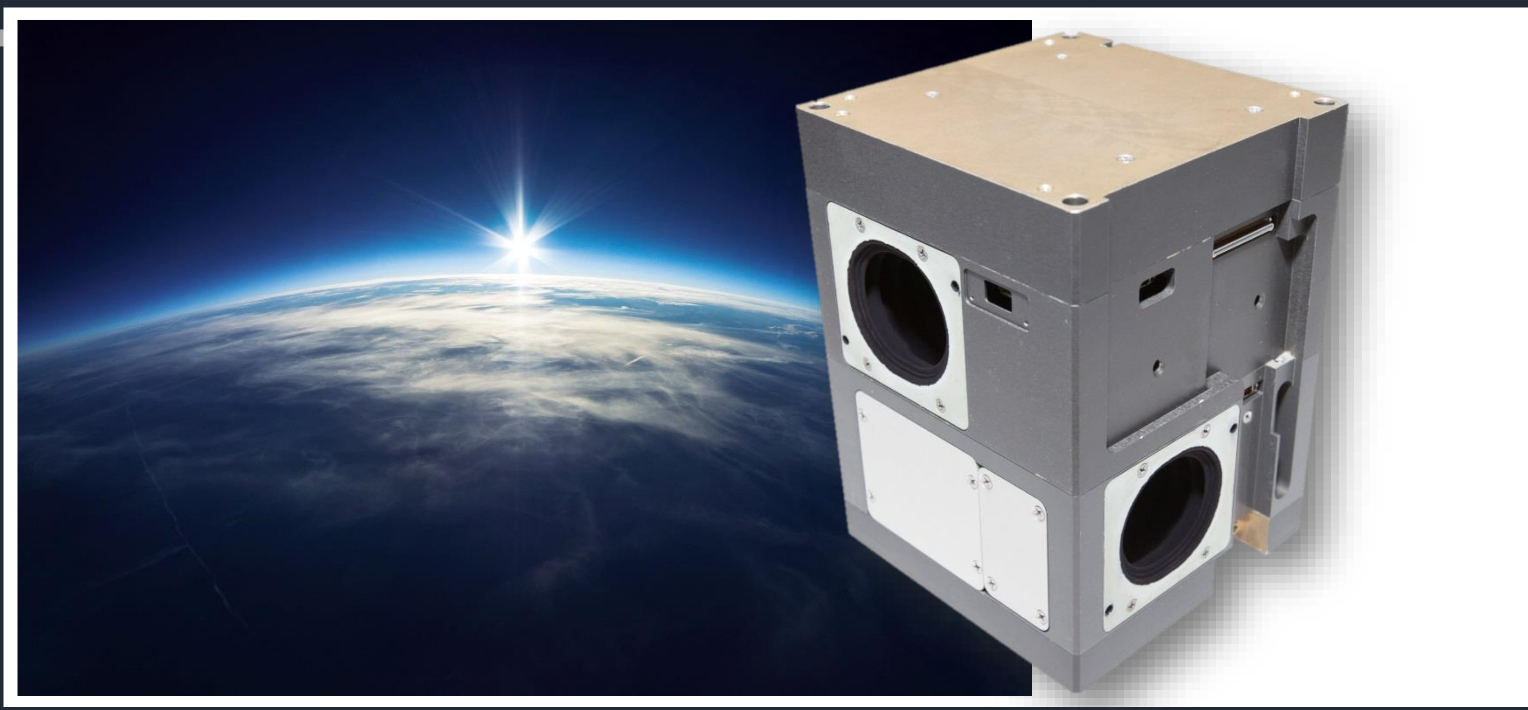
Parameter	Units	6U Side Panel
BOL Voc at -40°C	(V)	27.60
BOL Vmpp at -40°C	(V)	25.13
BOL Vmpp at 80°C	(V)	16.94
BOL Vmpp at 28°C	(V)	21.15
BOL Power at -40°C	(W)	22.18
BOL Power at 80°C	(W)	16.10
BOL Power at 28°C	(W)	18.75
Mass 1.6mm PCB no MTQ	(g)	290
Mass 1.6mm PCB w/ MTQ	(g)	340
Magnetic Moment 1.6mm PCB w/ MTQ	(Am2)	TBD

Add to basket (http://www.website.com/basket/add/335?dept=180)

XB1™

High-Performance 1U CubeSat Bus

Ready for your payload – right out of the box

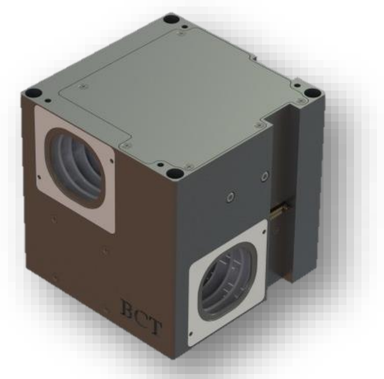


Key Features

- High-precision pointing performance from Dual Micro Star Trackers
- Bus functionality for GN&C, EPS, Thermal, C&DH, RF Communication, SSR, and Flight Software
- Interfaces and control provided for propulsion, solar arrays, and multiple payloads
- Maximizes payload volume
- Supports configurations up to 27U

Total Integrated Mission Solutions

XB1 provides a complete CubeSat bus solution in a highly integrated, precision spacecraft platform including: Ultra high-performance pointing accuracy, robust power system, command and data handling, RF communications, propulsion interfaces, and multiple flexible payload interfaces. Precision stellar-based attitude determination & control provided by dual star trackers. Supports precision orbit propagation of multiple target objects with flexible pointing commands to enable a wide range of missions. The **XB1 Flight Software** and simulation environment supports user-developed flight applications unique to your mission.



XB1 Stacked Configuration

XB1 Side by Side Configuration

	XB1 Parameter	Value/Notes
GN&C	**Pointing Accuracy**	±0.002° (1-sigma), 3 axes, 2 Trackers
	Pointing Stability	1 arc-sec/sec
	Maneuver rate	10 deg/sec (typical 3U CubeSat)
	Orbit knowledge	4m, 0.05m/s
CDH	**Data Interfaces**	Serial: LVDS, RS-422, or SPI available
	Onboard Data Processing	Configurable via user loadable software
	Telemetry Acquisition	6 12bit Analog, 6 discrete inputs
	Commands	Real-time, stored, macro
	Onboard Data Storage	32 Gbytes
EPS	**System Bus Voltage**	10 – 20 V (battery and array dependent)
	Energy Storage	Standard: 25Whr, expandable
	Solar Panels	Customer or BCT Provided Solar Panels (Details available per request)
	High Current Capability	Unregulated up to 60W
	Payload Power Feeds	QTY 6 (12, 5, 3.3V or Bus voltage)
Comm	**Frequency**	UHF or SBand
	Uplink	CCSDS, SGLS
	Downlink	250 kbps / 5 Mbps
	Encryption	AES 256
	Solid State Recorder Capacity	32 Gbytes
Prop	**Heater Controllers**	Up to 6 independently controlled zones
	Propulsion System Drive	Or up to 6 Thruster drivers or Latch Valves Drivers
	Telem. Interfaces	2 Temperatures, 4 voltages, 6 discrete IO
	Mass / Volume	1.5 kg / 10 cm x 10 cm x 10 cm
	XACT-Bus Nominal Power	< 6.3W
	Orbit Altitude / Orbit Lifetime	LEO / > 3 years

XB1 Modules (5 x 10 x 10 cm) can be stacked or placed side-by-side

- Provides the highest-available pointing performance from Dual Micro Star Trackers
- Bus functionality for GN&C, EPS, Thermal, C&DH, SSR, RF Communication
- Interfaces and control provided for Payload, Propulsion, and Solar Arrays
- Supports configurations up to 27U
- Compatible with multiple CubeSat deployment systems

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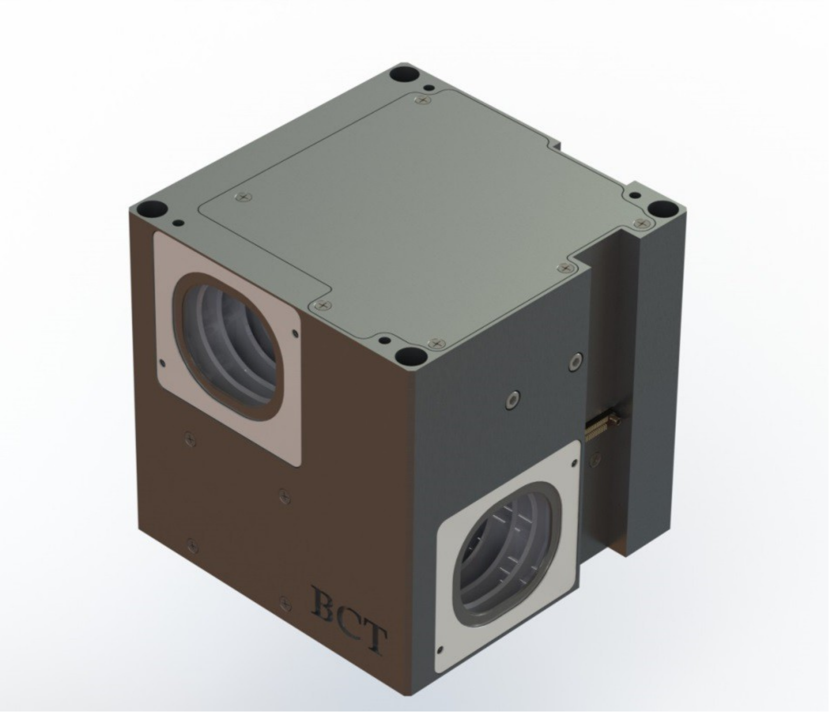


The XB1 Precision CubeSat Bus:

A New Paradigm for Space Exploration Platforms

Dan Hegel

Blue Canyon Technologies

720.458.0703

BlueCanyonTech.com

About BCT

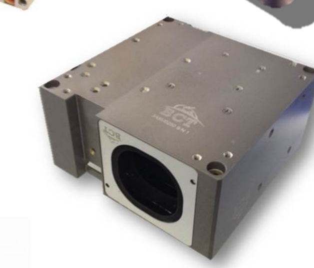
Blue Canyon Technologies is a small business founded in 2008 by industry veterans who have developed, tested and flown components and systems on more than 27 diverse space missions

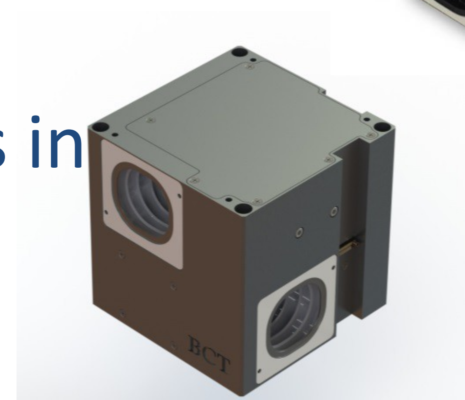
Advancing the state of the art in affordable space access

Current customers include: US Air Force,
NASA (JPL, Marshall, Johnson), Southwest Research,
University of Boulder, other commercial.

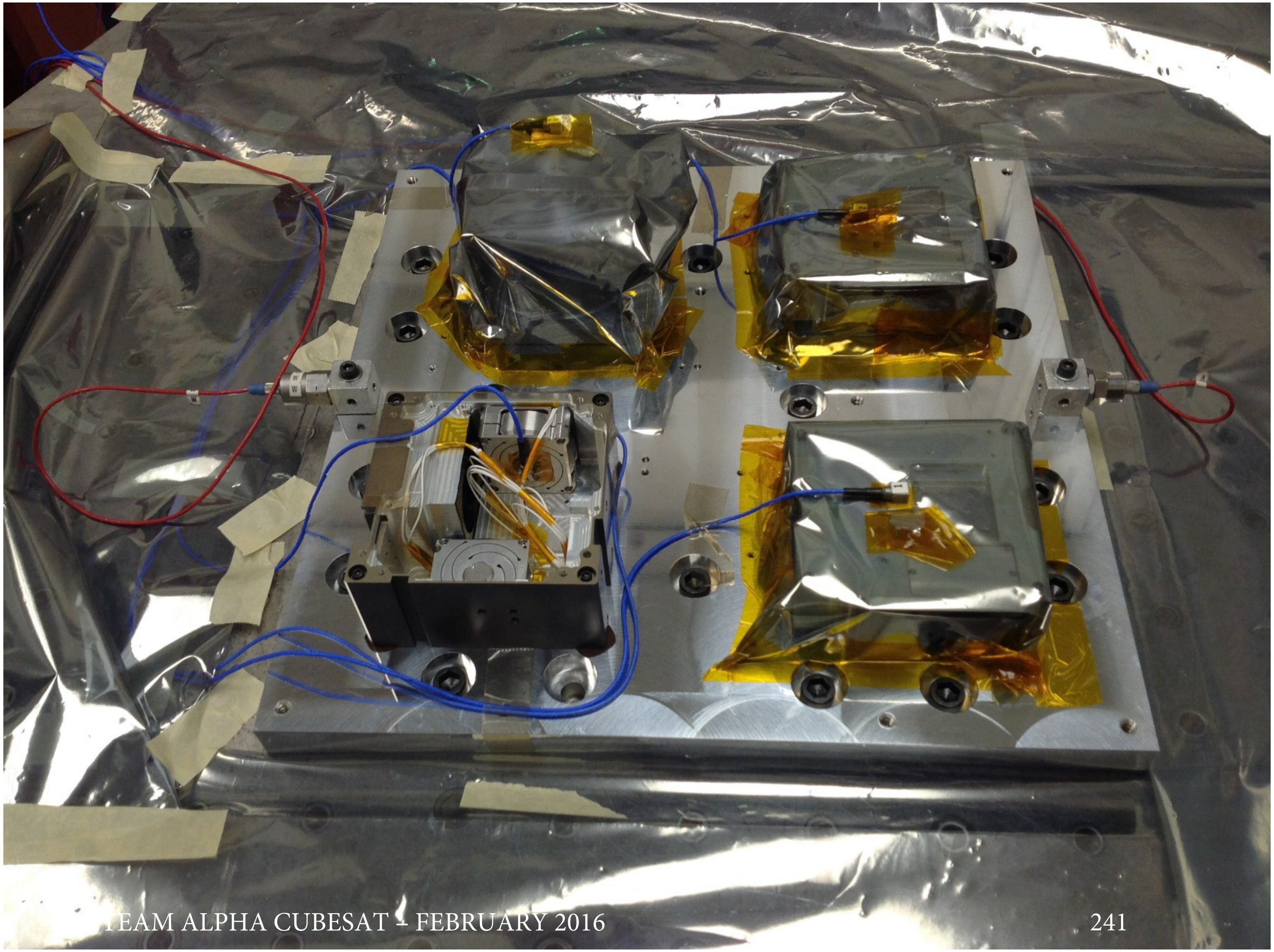
High Performance Products

- **Nano ST** – High performance, ultra-small Star Tracker
- **Reaction Wheels** – Nano, CubeSat, and Micro-Sat sized Wheels
- **XACT** - Complete CubeSat GN&C System (1/2U with Precision 3-Axis Pointing)
- **XB1** - Complete CubeSat Bus in 1U, based on XACT





Recent Vibration Test of Various Hardware

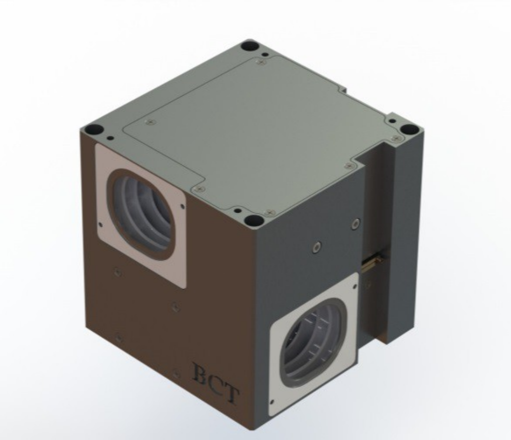


Integrated Spacecraft Design

- XB1 represents a paradigm shift
  - Complete spacecraft bus (GN&C, Power, Thermal, C&DH, RF-Comm, propulsion control, and flight software)
  - Ready straight out of the box, much like laptop computers and smart phones today
  - No programming or assembly required (except for your payload)
- And in the paradigm of smart phones, the XACT-Bus Development and Operations Environment (using model-based design) will provide users the ability to develop their own flight “apps” to operate their payload, process payload data, and control XB1
  - For example, new RPOD algorithms
  - Mission specific onboard processing of payload data
  - The user needs only to provide the mission-dependent payload
- Increases mission capabilities by maximizing payload volume, power availability, and autonomy

XB1

XACT-Based High Performance CubeSat Bus



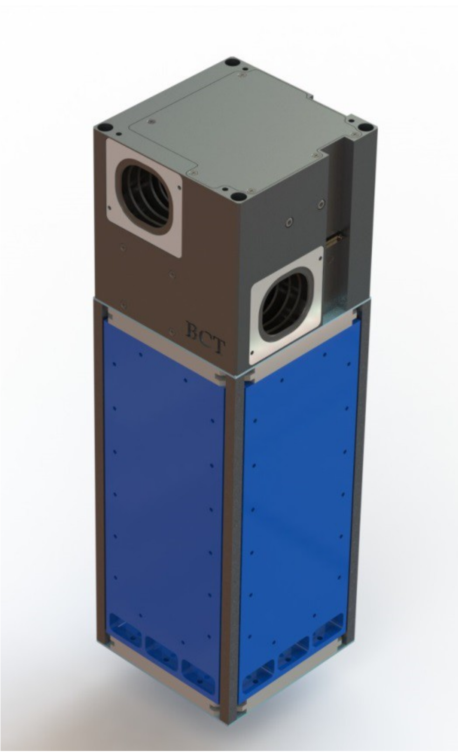
- Highest-available pointing performance from *Dual* Micro-Star Trackers
- Bus functionality for GN&C, EPS, Thermal, C&DH, SSR, RF Comm*
- Interfaces and control provided for Payload, Propulsion, and Solar Arrays
- Supports configurations up to 27U

* optional 1-cm slice

	XB1 Parameter	Value/Notes
GN&C	Pointing Accuracy	±0.003° (1-sigma), 3 axes, 2 Trackers
	Pointing Stability	1 arc-sec/sec
	Maneuver rate	10 deg/sec (typical 3U CubeSat)
	Orbit knowledge	10m, 0.15m/s (GPS)
CDH	Data Interfaces	Serial: RS-422, I2C, SPI, LVDS
	Onboard Data Processing	Configurable via user loadable "apps"
	Telemetry Acquisition	16 12bit Analog, 32 discrete inputs
	Commands	Real-time, 10,000 stored, macros
	Onboard Data Storage	4 Gbytes (option)
EPS	System Bus Voltage	12 ± 2 V
	Energy Storage	>20Whrs
	Payload Power Feeds	QTY 3, 12V or Regulated 1.2V to 5.0V
Comm*	Frequency	UHF or SBand
	Uplink	CCSDS, USB, SGLS
	Downlink	250 kbps / 5 Mbps
	Encryption	AES 256
	Solid State Recorder Capacity	4 Gbytes
Prop	Heater Controllers	4 independently controlled zones
	Propulsion System Drive	8 Thruster drivers, 2 Latch Valve Drivers
	Telem. Interfaces	1 Temperature, 1 Pressure, 2 Status
	Mass / Volume	1.5 kg / 10 cm x 10 cm x 10 cm
	XACT-Bus Nominal Power	<2.5W

Supports Multiple Configurations of BCT Structures

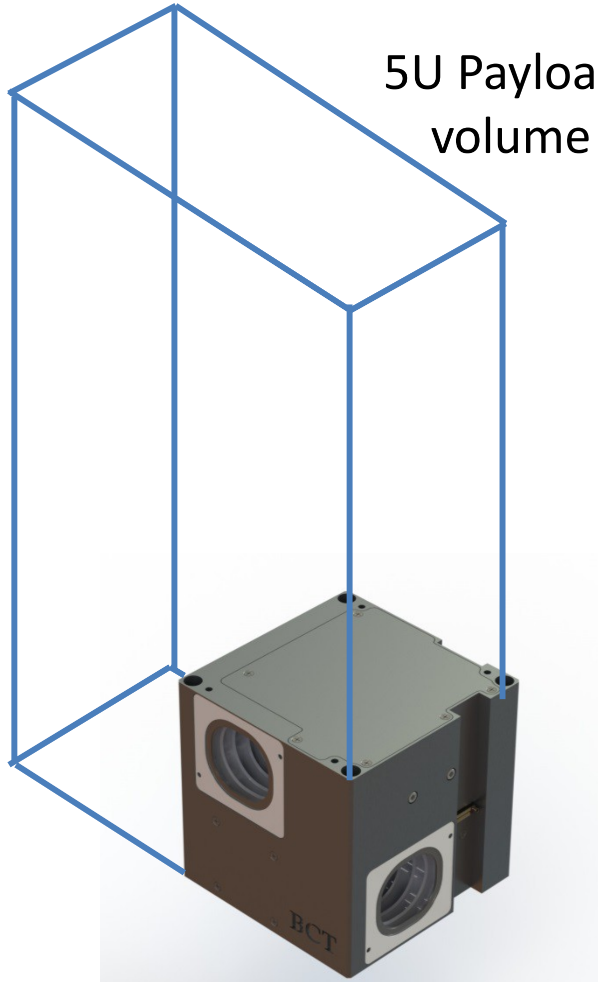
3U



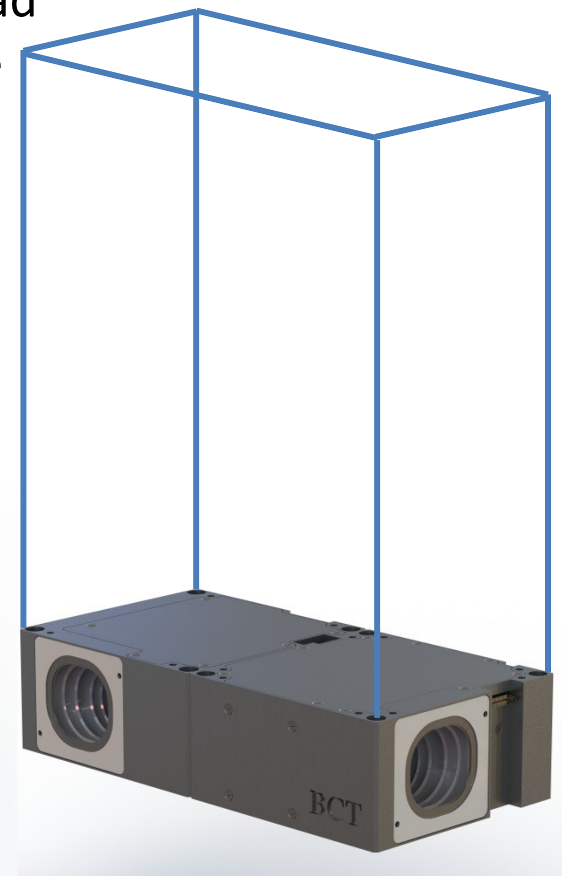
2U Payload volume

6U 'Stack'

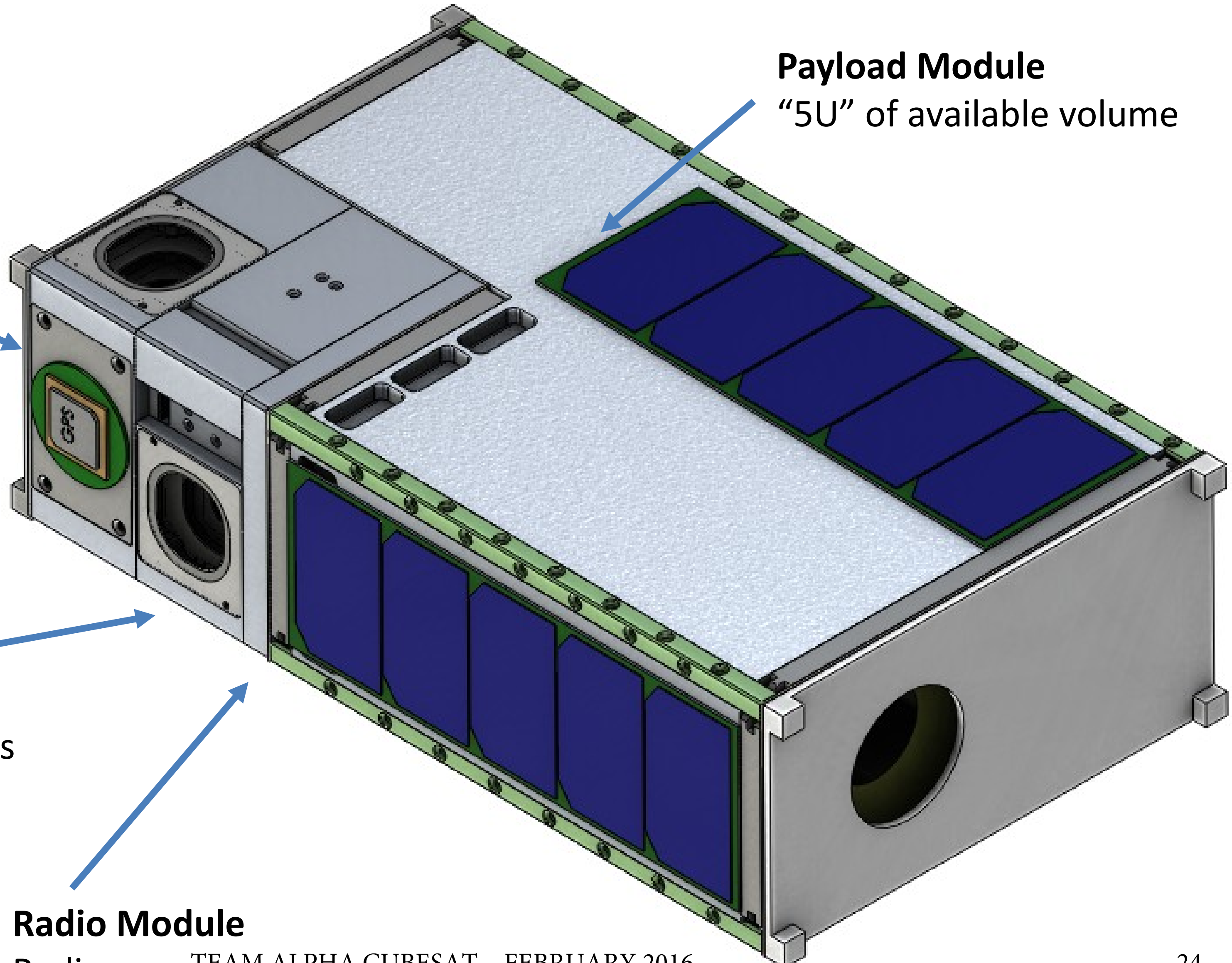
5U Payload volume



6U 'Side-by-side'



6U Stack Configuration



Payload Module
"5U" of available volume

XB1 Module
Star Tracker
Power Control
Battery
GPS

XACT Module
Star Tracker
3X Reaction Wheels
3X Torque Rods
Coarse Sun Sensor
Magnetometer
MEMS Gyro

Radio Module
Radio

XB1 Flight Software Highlights

- Highly autonomous operation
- Precision stellar-based attitude determination & control
  – Operates with stars down to 7.5 magnitude (over 21,000 stars in catalog)
  – Lost-in-space star identification in less than 2 seconds
- Supports precision orbit propagation of multiple target objects
- Flexible pointing commands support a wide range of missions
  – *e.g. Inertial, LVLH, Earth-Fixed, Solar, object tracking*
- Supports user-developed payload apps
  – *Built-in 'hooks' for high rate, low rate, and asynchronous task processing, with easy access to all XB1 data, including raw star camera images*
  – *XB1 interface functions allow user apps to receive commands and send telemetry*
  – *XB1 interface functions allow user apps to command the XB1 (e.g. a wide-field payload detects lunar feature of interest, then commands XB1 to point narrow-field payload for more accurate data.)*
- Supports 10,000 stored commands, as well as real-time, macro sequences, and commands from user apps
- Multiple telemetry formats

XB1 Development & Operations Environment (XDOE)

- XDOE supports user through all stages of satellite life cycle.
- Model-based design (using Matlab/Simulink) supports flight software and simulation software in one unified environment.
- All-software simulation of spacecraft (provided out-of-the-box) supports mission analyses and training.
- Customizable with user payload models and flight apps.
- Auto-code generation of custom models and apps.
- Test console supports real-time closed-loop testing of XB1.
- Command, telemetry, and 3d animation displays.
- Generation of uploadable flight parameter tables.

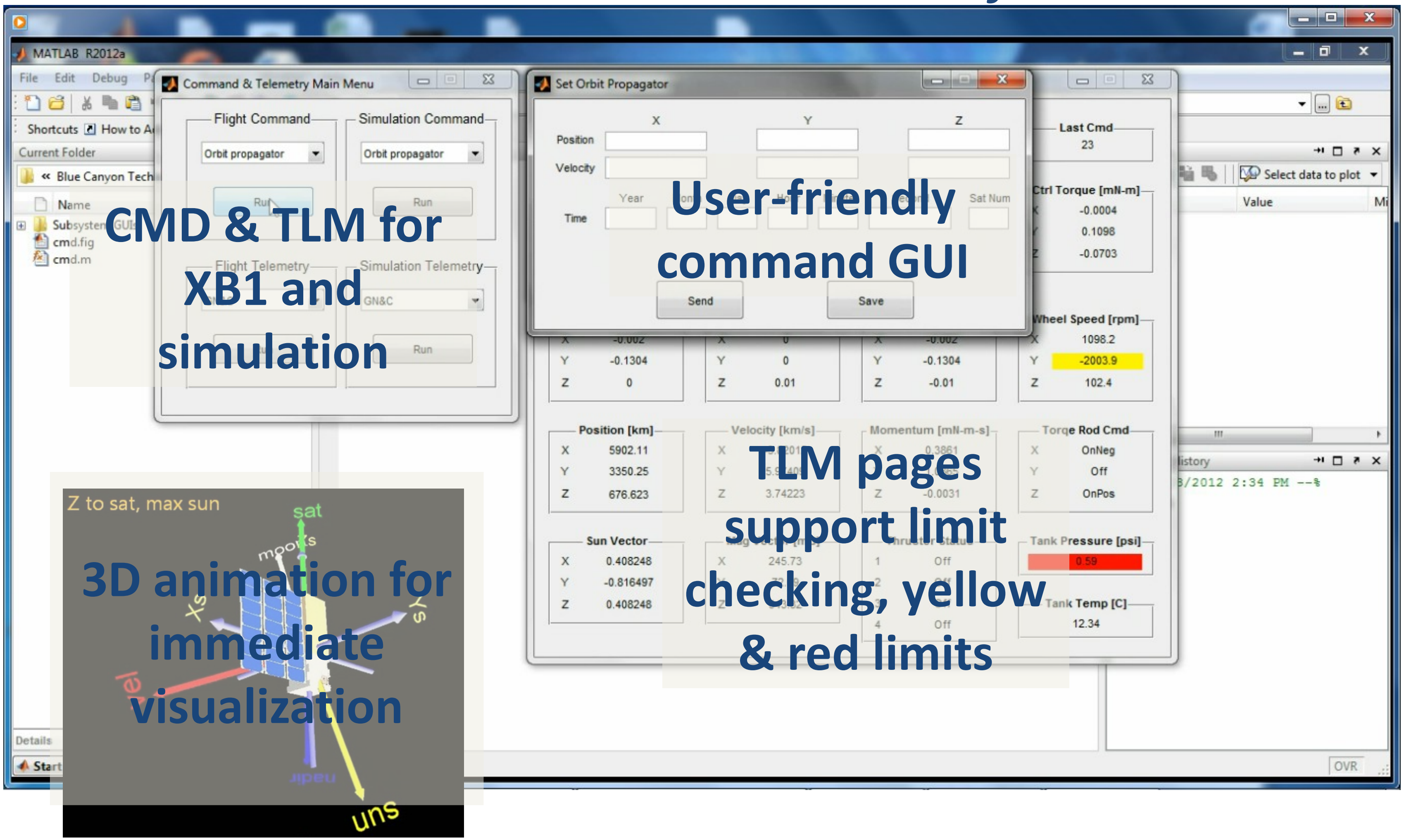
All the tools you need to quickly get to the science

XDOE Simulation Highlights

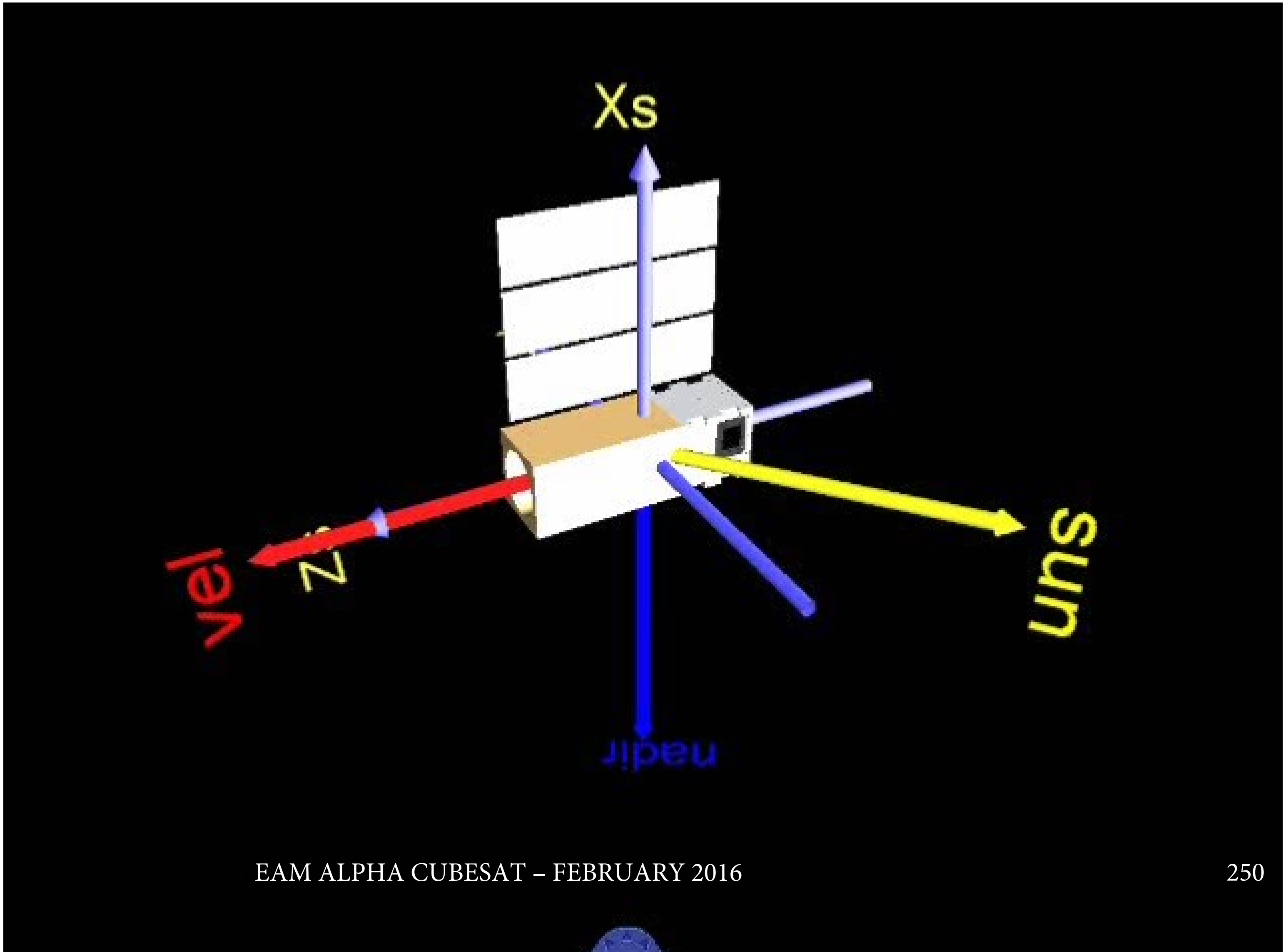
- Supports constellation of 99 satellites (each independently configured and controlled)
- Variable run speed (<<real-time to >>real-time)
- Command script or GUI control
- Selectable gravity field model with user-friendly initialization command features to support formation flying (can use earth or lunar harmonic model)
- Sun, moon, star field vector models
- High-fidelity GN&C component models
- Built-in 3d animation driver for user-provided VRML model
- Real-time STK “connect” interface, or play-back files supported

XB1 Integrated Command, Telemetry & Animation

Control and visualize the XB1 out of the box



XB1 Flexible Pointing Demo



XB1
Enabling A New Realm of CubeSat Science

- Remote Sensing
- Formation Flying
- Rendezvous, Proximity Operations & Docking
- Autonomous Operations
- Inter-satellite Communications Networks
- Thruster control for lunar orbit insertion
- Lunar impactor steering

Contact BCT for more information

Dan Hegel

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XACT Lite™

Attitude Control for CubeSats

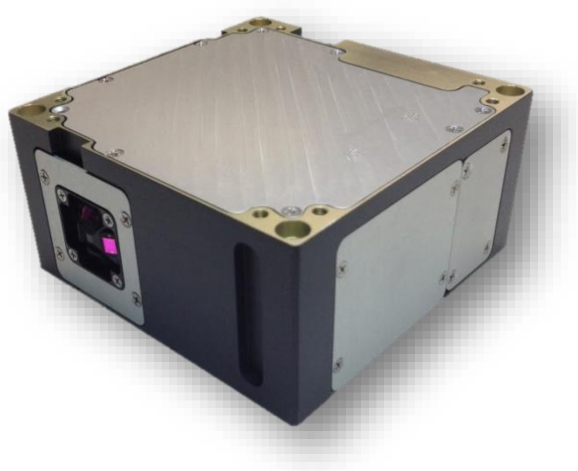
3-axis attitude determination in a micro-package

Key Features

- Low cost 3-axis attitude determination
- 0.5U Micro-package
- Multiple pointing reference frames: Inertial, LVLH, Earth-fixed, Solar
- Low jitter 3-axis reaction wheel control (also sold as single wheel)
- User-friendly software for simulation, integration and customization
- Self-calibrating reaction wheels, with advanced digital controls, provide unparalleled torque precision

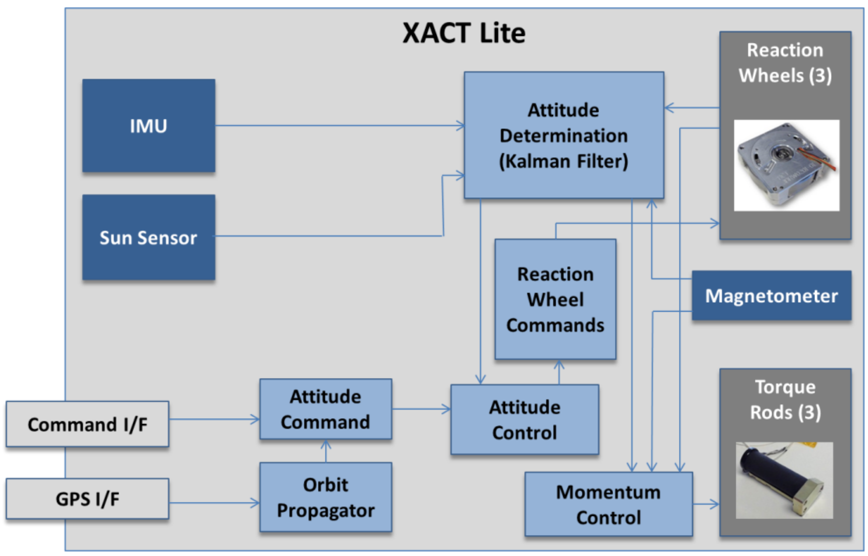
Total Integrated Mission Solutions

XACT Lite is a lower cost and lighter version of the BCT XACT. The XACT Lite is for those missions that don't need the exquisite pointing of the stellar-based XACT, but want to utilize all of its other flexible, capable features. It is a reliable CubeSat attitude control system compatible with a variety of configurations and missions. The highly integrated XACT architecture leverages a powerful processing core with BCT's Micro Reaction Wheel assemblies to enable a new generation of highly capable, miniaturized spacecraft. XACT features 3-axis Attitude Determination in a micro-package. Built-in flexible commanding allows for multiple pointing reference frames: Inertial, LVLH, Earth-Fixed, and Solar. Precise 3-axis control is provided by low jitter reaction wheels, torque rods and integrated control algorithms. Software is available to support simulation, system integration, and customization of the ADCS functionality.



XACT Lite Capability	
Specification	**Performance**
Spacecraft Pointing Accuracy	± 1.0 deg (1-sigma)
Spacecraft Lifetime	3 Years (LEO)
XACT Lite Mass	0.7 kg
XACT Lite Volume	10 x 10 x 5 cm (0.5U)
XACT Lite Electronics Voltage	5V
XACT Lite Reaction Wheel Voltage	12V
Data Interface	RS-422 (can support SPI)
Slew Rate (8kg, 3U CubeSat)	≥10 deg/sec

Operational Case	Power (W)
XACT (5 Hz operation)	0.75
XACT + 3 TR (ON STATE)	1.50
XACT + 3 RW (@ 600 rpm)	0.89
XACT + 3 RW (@ 600 rpm) + 3 TR (ON STATE)	1.64
XACT + 3 RW (@ 1500 rpm)	1.90
XACT + 2RW (@600 rpm) + 1 RW (max speed @6000 rpm)	2.53



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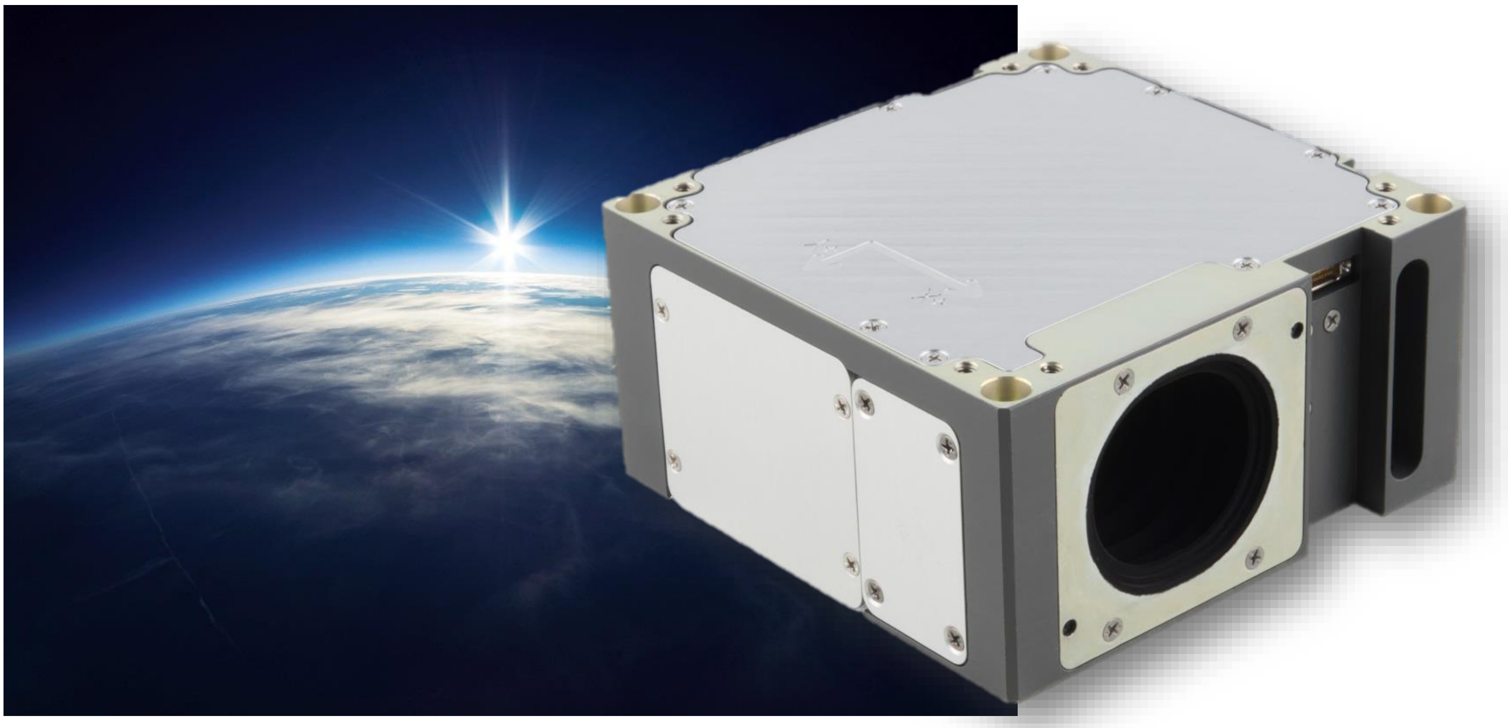
XACT™

High-Performance Attitude Determination for CubeSats

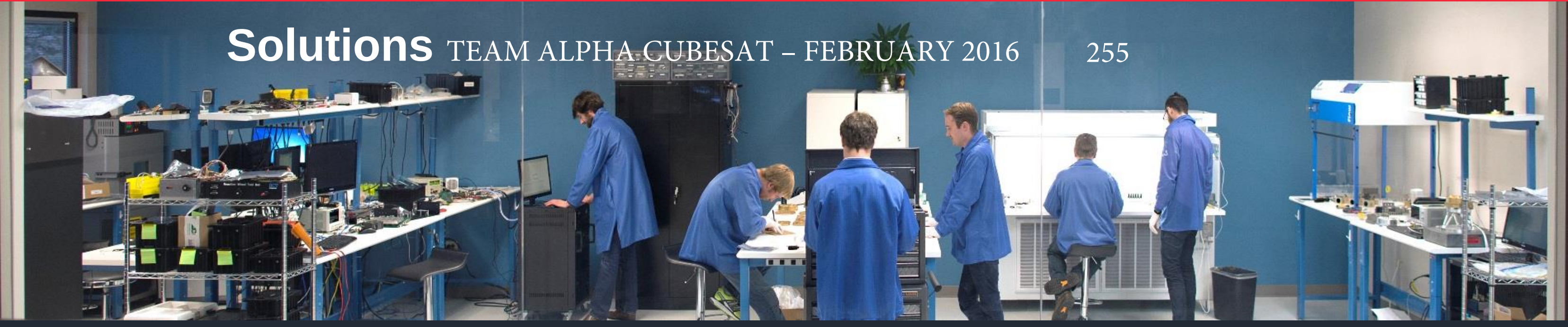
Precise 3-axis stellar attitude determination in a micro-package



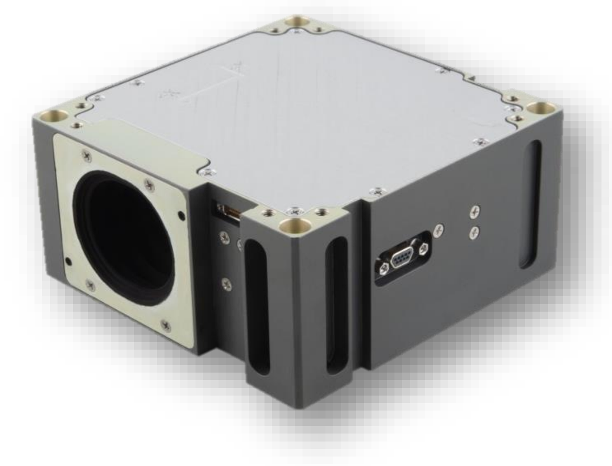
Key Features

- 3-axis Stellar Attitude Determination with integrated stray light baffle
- 0.5U Micro-package
- Multiple pointing reference frames: Inertial, LVLH, Earth-fixed, Solar
- Low jitter 3-axis reaction wheel control (also sold as single wheel)
- User-friendly software for simulation, integration and customization
- Self-calibrating reaction wheels, with advanced digital controls, provide unparalleled torque precision

Total Integrated Mission Solutions



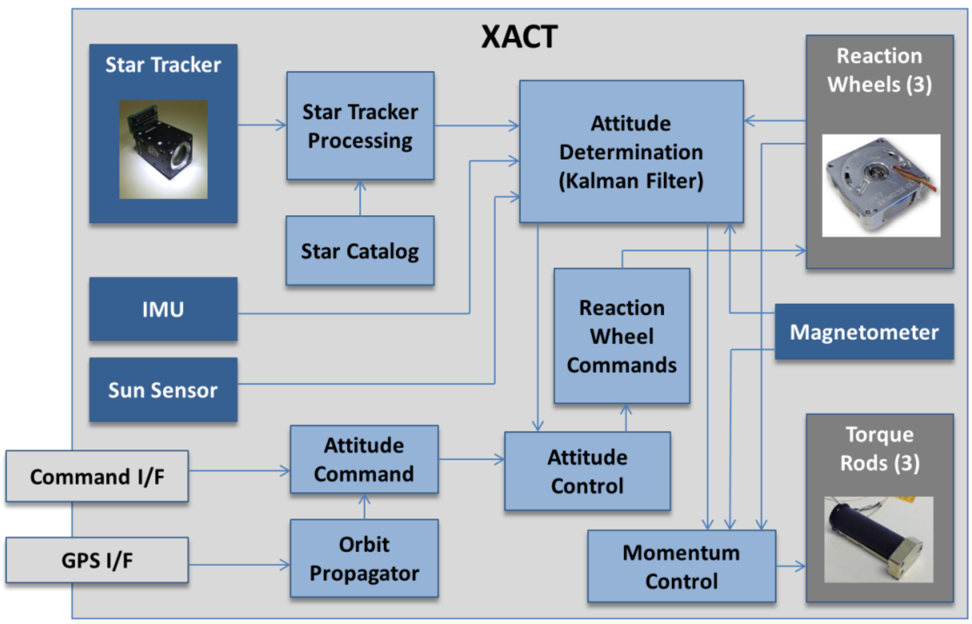
XACT is a reliable CubeSat attitude control system compatible with a variety of configurations and missions. The highly integrated XACT architecture leverages a powerful processing core with BCT's Micro Star Tracker and Micro Reaction Wheel assemblies to enable a new generation of highly capable, miniaturized spacecraft. XACT features 3-axis Stellar Attitude Determination in a micro-package. Built-in flexible commanding allows for multiple pointing reference frames: Inertial, LVLH, Earth-Fixed, and Solar. Precise 3-axis control is provided by low jitter reaction wheels, torque rods and integrated control algorithms. Software is available to support simulation, system integration, and customization of the ADCS functionality.



XACT Capability

Specification		Performance
Spacecraft Pointing Accuracy		± 0.003 deg (1-sigma) for 2 axes
		± 0.007 deg (1-sigma) for 3rd axis
Spacecraft Lifetime		3 Years (LEO)
XACT Mass		0.85 kg
XACT Volume		10 x 10 x 5 cm (0.5U)
XACT Electronics Voltage		5V
XACT Reaction Wheel Voltage		12V
Data Interface		RS-422 (can support SPI)
Slew Rate (4kg, 3U CubeSat)		≥10 deg/sec

Operational Case	Power (W)
XACT (low power standby mode)	0.85
XACT (5 Hz operation)	1.05
XACT + 3 TR (ON STATE)	1.80
XACT + 3 RW (@ 600 rpm)	1.19
XACT + 3 RW (@ 600 rpm) + 3 TR (ON STATE)	1.94
XACT + 3 RW (@ 1500 rpm)	2.20
XACT + 2RW (@600 rpm) + 1 RW (max speed @6000 rpm)	2.83



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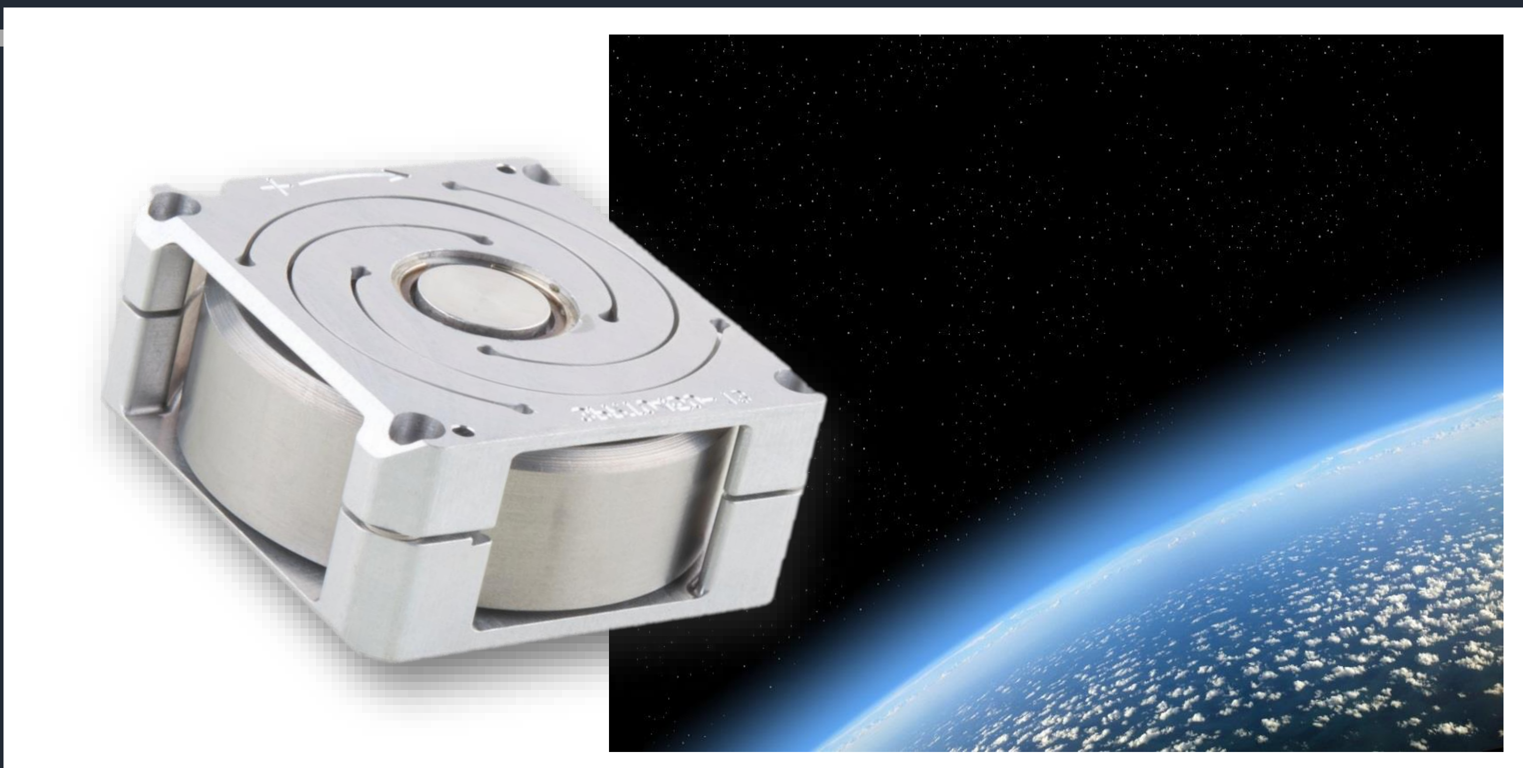
Micro Reaction Wheel ™

High-Performance Attitude Determination for CubeSats

Reaction Wheel for precise attitude control of Nanosats



Key Features

- Micro volume packaging
- Highly efficient design
- Low jitter
- Long life bearings
- Patent Pending

Total Integrated Mission Solutions

The **BCT Micro Reaction Wheel** is a reliable CubeSat attitude sensor compatible with a variety of configurations and missions. It is designed with a revolutionary micro size, power, and mass. The BCT Micro Reaction Wheel is creating a new level of performance for Nano size spacecraft.

Micro Reaction Wheel Capability

Specification	Performance
Momentum	15 mN·m·s
Max Speed	6,500 RPM
Max Torque	6 mN·m
Torque @ 3000 RPM	3 mN·m
Lifetime	>5 Years
Mass	115 g
Volume	43 x 43 x 18 mm
Power @ 600 RPM	0.1 W
Power @ 3000 RPM	0.9 W
Power @ 6000 RPM	1.7 W
Power @ Max Torque	8.0 W
Operating voltage	+12V (variable down to +8V)
Data interface (optional drive electronics board available)	RS-422 (can support SPI)
Fine Dynamic Imbalance – Static	≤0.35 g·mm
Fine Dynamic Imbalance – Couple	≤4.55 g·mm^2

Waterfall plots available upon request

The BCT Micro Reaction Wheel uses a BCT-built motor and a long life hybrid bearing and lubrication system. The wheels undergo extensive testing to characterize their electrical and mechanical performance, including jitter and life test.

BCT

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VII. CONCLUSION

Through this article, we have tried to bring a personal contribution which consists of an adds in the design of future satellite communication system for Ka band that propose a multimedia application, first we have tried to choose an architecture to our system, we have chosen a simple architecture for our study consisting of a transmit earth station a receiver and a geostationary satellite, In the practice case this architecture is spread on several users (multi-user) especially with the use of satellite equipped with multibeam antenna. Secondly we developed software that enables the link budget calculation of any satellite link. Then, we have set up the different parts that make up the system.

Finally, we use the developed software for checking the feasibility of implementing such system with the proposed parameters and the results are very satisfactory because at the end of calculation, we conclude to a margin of error at 8.17 dB for the uplink and at 8.2 dB in the downlink, for both cases is a fairly comfortable margin since it is higher than the limit that was set previously and which is 8 dB, so even if we will have to strong atmospheric disturbances (especially the case of strong rain) or other unexpected sources of losses, our system will function normally and the service will be provided.

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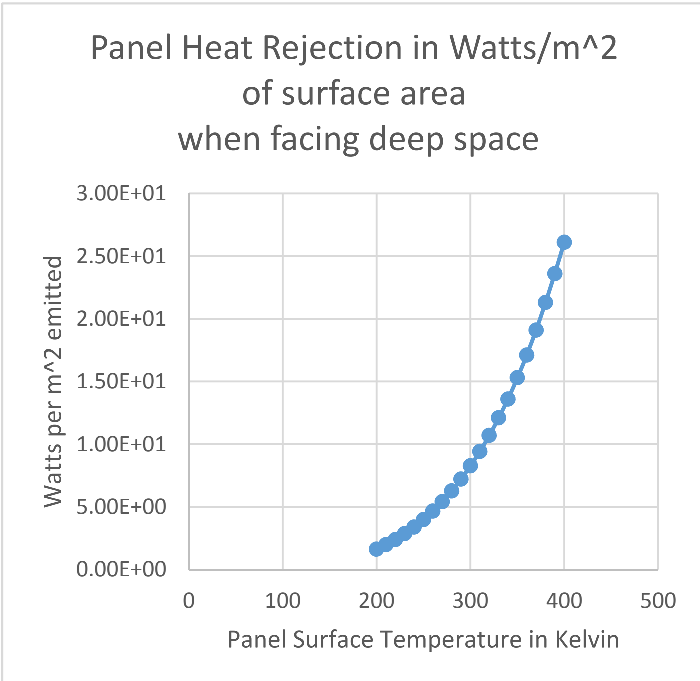
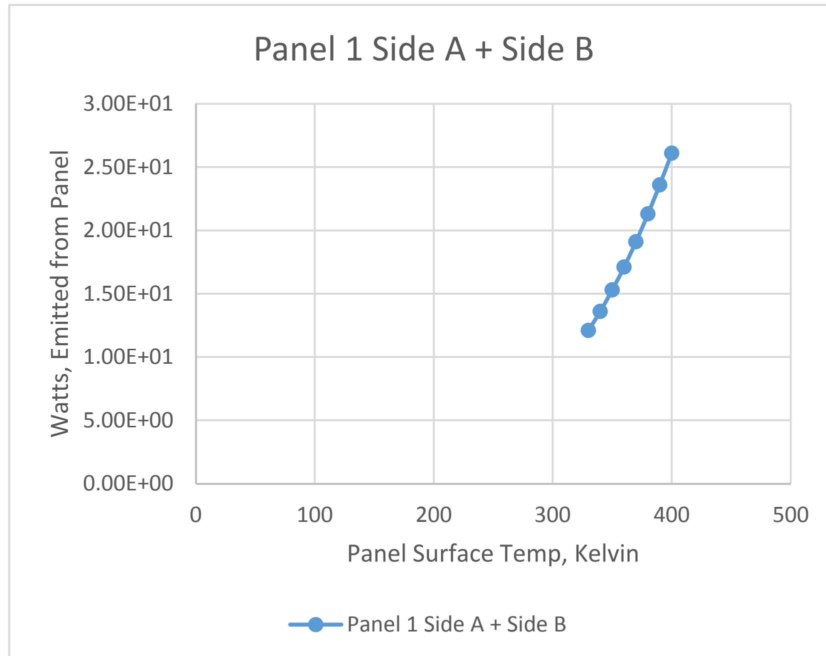
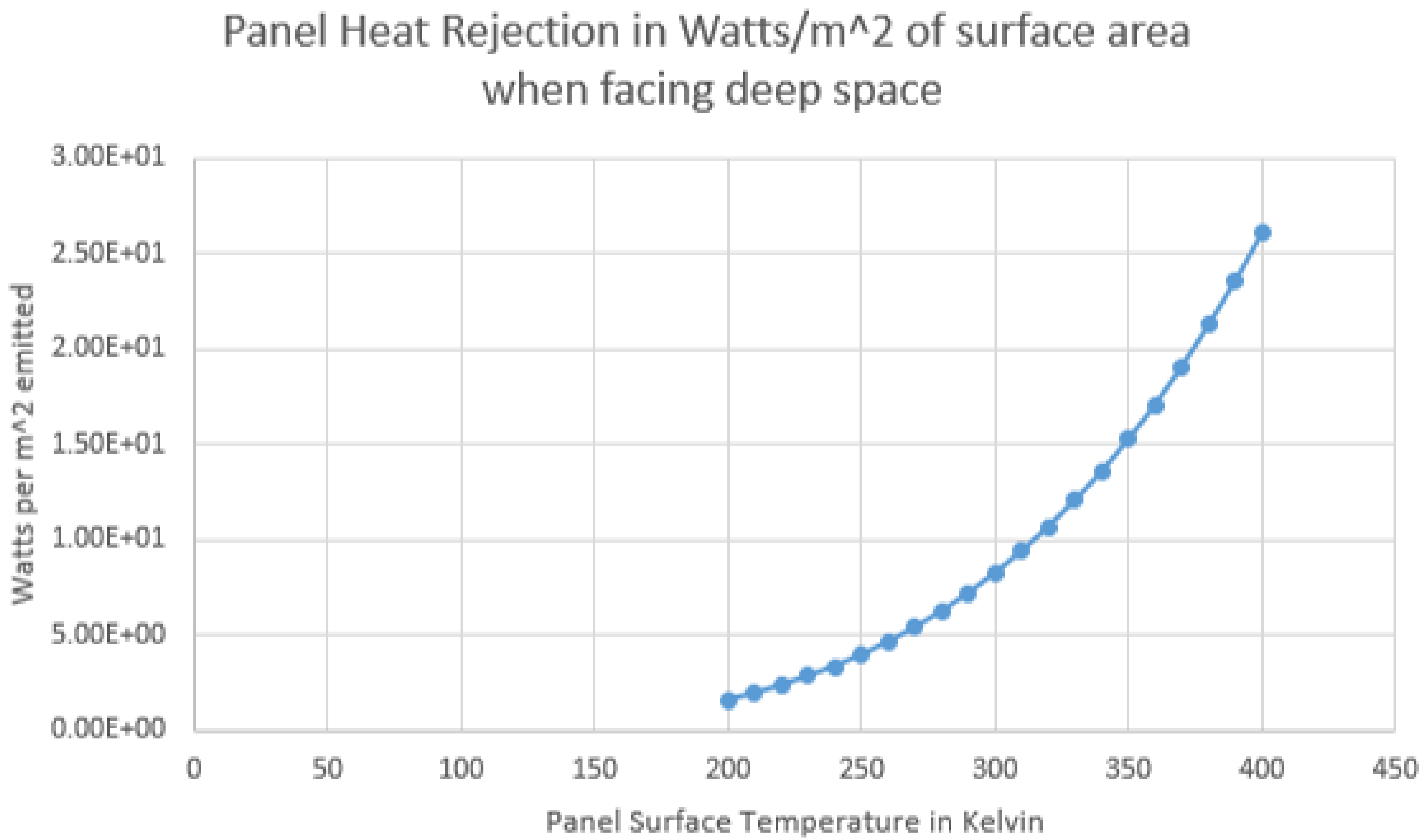
Watts emitted per Sq. meter Panel 1 Side A + Side B	Emissivity	m^2 Area	m Length	m Width	W/(m^2*K^4) Stephan-Boltz	K Temp-spac	View Facto	K Panel Temp	Watts emitted per Sq. meter Panel 1 Side A + Side B
2.61E+01	0.9	1	1	1	5.67E-10	4	1	400	2.61E+01
2.36E+01	0.9	1	1	1	5.67E-10	4	1	390	2.36E+01
2.13E+01	0.9	1	1	1	5.67E-10	4	1	380	2.13E+01
1.91E+01	0.9	1	1	1	5.67E-10	4	1	370	1.91E+01
1.71E+01	0.9	1	1	1	5.67E-10	4	1	360	1.71E+01
1.53E+01	0.9	1	1	1	5.67E-10	4	1	350	1.53E+01
1.36E+01	0.9	1	1	1	5.67E-10	4	1	340	1.36E+01
1.21E+01	0.9	1	1	1	5.67E-10	4	1	330	1.21E+01
1.07E+01	0.9	1	1	1	5.67E-10	4	1	320	1.07E+01
9.42E+00	0.9	1	1	1	5.67E-10	4	1	310	9.42E+00
8.27E+00	0.9	1	1	1	5.67E-10	4	1	300	8.27E+00
7.22E+00	0.9	1	1	1	5.67E-10	4	1	290	7.22E+00
6.27E+00	0.9	1	1	1	5.67E-10	4	1	280	6.27E+00
5.42E+00	0.9	1	1	1	5.67E-10	4	1	270	5.42E+00
4.66E+00	0.9	1	1	1	5.67E-10	4	1	260	4.66E+00
3.99E+00	0.9	1	1	1	5.67E-10	4	1	250	3.99E+00
3.39E+00	0.9	1	1	1	5.67E-10	4	1	240	3.39E+00
2.86E+00	0.9	1	1	1	5.67E-10	4	1	230	2.86E+00
2.39E+00	0.9	1	1	1	5.67E-10	4	1	220	2.39E+00
1.98E+00	0.9	1	1	1	5.67E-10	4	1	210	1.98E+00
1.63E+00	0.9	1	1	1	5.67E-10	4	1	200	1.63E+00


Panel Heat Rejection in Watts/m^2 of surface area when facing deep space
Watts per m^2 emitted
3.00E+01
2.50E+01
2.00E+01
1.50E+01
1.00E+01
5.00E+00
0.00E+00
0
100
200
300
400
500
Panel Surface Temperature in Kelvin


Panel 1 Side A + Side B
Watts, Emitted from Panel
3.00E+01
2.50E+01
2.00E+01
1.50E+01
1.00E+01
5.00E+00
0.00E+00
0
100
200
300
400
500
Panel Surface Temp, Kelvin
Panel 1 Side A + Side B



Panel Heat Rejection in Watts/m^2 of surface area when facing deep space
Watts per m^2 emitted
3.00E+01
2.50E+01
2.00E+01
1.50E+01
1.00E+01
5.00E+00
0.00E+00
0
50
100
150
200
250
300
350
400
450
Panel Surface Temperature in Kelvin

Energy Balance in LEO Case: Aft Facing Sun Ram Facing Earth	7/3/2015 Alpha Cubesat Revision 1	**Assumptions: Power off, .9 surface emissivity; high reflective surface on aft**
Power ON	Eric Gustafson	**Energy Balance 6U AlphaCube Sat w/ Aft facing Sun and Ram facing earth**

Ref: Heat Transfer, 8th Ed., Holman, JP, McGraw Hill, 1997
http://wiki.naturalfrequency.com/wiki/Absorptance_and_Emittance

Item Description	Watts emitted Energy (Watts)	Emissivity/ absorption	Area (m^2)	Length (m)	height (m)	W/(m^2*K^4) Stephan-Boltz
Panel 1 (AFT) Solar Heat Flux	5.58	0.2	0.02	0.1	0.2	5.67E-08
Panel 2 (Ram) Emit	-2.45E+00	0.9	0.02	0.1	0.2	5.67E-08
(Emitted to Earth)	-4.10E+00	0.9	0.02	0.1	0.2	5.67E-08
Panel 3 (Zenith) Emit	-3.67E+01	0.9	0.06	0.3	0.2	5.67E-08
Panel 4 (Port) Emit	-3.67E+01	0.9	0.06	0.3	0.2	5.67E-08
Panel 5 (Nadir) Emit	-3.67E+01	0.9	0.06	0.3	0.2	5.67E-08
Panel 6 (Starport) Emit	-3.67E+01	0.9	0.06	0.3	0.2	5.67E-08
System Electronics Power	26.5					
Ion Thruster Power	40					
Ion Thruster Radiative loss	-8.62E+01	0.96	0.13194678	0.942477	0.14	5.67E-08
Extended Radiative surfaces	-7.39E+00	0.96	0.011309724	0.03769908	0.3	5.67E-08
Solar Panel Radiate	-1.18E+02	0.96	0.18	0.6	0.3	5.67E-08
Solar Panel Absorb	203.391	0.9	0.18	0.6	0.3	5.67E-08
Solar Panel Radiate	-1.18E+02	0.96	0.18	0.6	0.3	5.67E-08
Solar Panel Absorb	203.391	0.9	0.18	0.6	0.3	5.67E-08
Energy Balance, Watts	-3.436940185					

K		K	W/m^2
Temp -Radiate	View Factor	Panel Temp	Heat Flux
6000	1		1395
4	-0.2	331	
289	-0.8	331	
		331	
4	-1	331	
		331	
4	-1	331	
		331	
4	-1	331	
		331	
4	-1	331	
		331	
		331	
		331	
4	-1	331	
4	-1	331	
		331	
4	-1	331	
4	0.9	331	1395
4	-1	331	
4	0.9	331	1395

Table ________ ACS Spacecraft Requirements Matrix

Rule Number	Rule Text	Classification	Spacecraft	System
I	Abide by the prevailing Cube Quest Challenge rules as defined in Document No.: CCP-CQ-OPSRUL-001 Cube Quest Challenge Ground Tournaments, Deep Space Derby, and Lunar Derby Operations and Rules December 4, 2014 Revision C, December 30, 2015 and subsequent revisions as made applicable.	Admin & Technical	Yes	All
II	ACS Spacecraft Requirements Abstract from Document No.: CCP-CQ-OPSRUL-001			
	Eligibility and Registration			
	Rule 1: Eligibility to Compete and win prize(s)			
	Rule 1.A: In order to be eligible to win a Prize, the Team Leader must be (i) a citizen or permanent resident of the United States, or (ii) an Entity that is incorporated in and maintains a primary place of business in the United States. Competitor Teams must furnish proof of eligibility (including proof of citizenship or permanent resident status, for Team Leader, and proof of incorporation and primary place of business, for an U.S. Entity) that is satisfactory to NASA in its sole discretion. A Competitor Team's failure to comply with any aspect of the foregoing requirements shall result in the Competitor Team being disqualified from winning a Prize from NASA.	Admin		
	Rule 1.B: A Competitor Team is comprised of one or more Team Members. A Team Member can be an individual or an Entity. If a Team Member is an individual, the individual has to be a citizen or permanent resident of the United States. If the Team Member is an Entity, the Entity must be a U.S. Entity (incorporated in and maintains a primary place of business in the United States). Foreign nationals may own up to 49% of an otherwise eligible U.S. Entity. Foreign nationals may only participate as either owners, employees, or students of an otherwise eligible U.S. entity.	Admin		
	Rule 1.C: No Team Member shall be citizens of a country on the NASA Export Control Program list of designated countries. (The current list of designated countries can be found at http://oiir.hq.nasa.gov/nasaecp/).	Admin		
	Rule 1.D: A Federal Entity or Federal Employee may not participate in the Cube Quest Challenge if acting within the scope of their employment.	Admin		
	Rule 1.E: An Entity Employee, or Entity, contracted by the US. Government and physically located at a Federally Owned Facility may not participate if acting within the scope of the contract.	Admin		
	Rule 1.F: Each Team Member shall acknowledge by their signature in the Registration Data Package that NASA shall make Prize payments to the Team Leader, also indicated in the Registration Data Package. Any failure of the indicated Team Leader to make payments of any kind to Team Members is the responsibility of the Team Leader and not the responsibility of NASA.	Admin		
	Rule 1.G: A Competitor Team may only submit a single CubeSat into competition to win a Cube Quest Challenge Prize; however, a Team Member may support more than one Competitor Team.	Admin		
	Rule 2: Competitor Team Responsibilities and Agreements			
	Rule 2.A: Competitor Teams are responsible for compliance with all applicable regulations and laws including obtaining any necessary approvals for foreign student or employee participation.	Admin		
	Rule 2.B: Prospective Competitor Teams shall submit their notice of intention to compete, and a Registration Data Package (defined in Section 5.0), to the Email address given in Section 5.2. In addition, Competitor Teams must submit a Mission Concept Registration Data Package, as defined in Rule 3, within 60 calendar days after their registration. The prospective Competitor Team will receive a formal acknowledgement receipt of their package within 5 business days of submittal and a formal acceptance as Challenge Competitor Teams within 15 business days.	Admin		

	Rule 2.C: Liability insurance - All Team members agree to assume any and all risks and waive claims against the Federal Government and its related Entities, except in the case of willful misconduct, for any injury, death, damage, or loss of property, revenue, or profits, whether direct, indirect, or consequential, arising from their participation in the competition, whether such injury, death, damage, or loss arises through negligence or otherwise. For the purposes of this paragraph, the term `related Entity' means a contractor or subcontractor at any tier, and a supplier, user, customer, cooperating party, grantee, investigator, or detailee. Team Members must obtain liability insurance or demonstrate financial responsibility, in the amount of $1,000,000 for claims by- A. A third party for death, bodily injury, or property damage, or loss resulting from an activity carried out in connection with participation in a competition, with the Federal Government named as an additional insured under the registered participant's insurance policy and registered participants agreeing to indemnify the Federal Government against third party claims for damages arising from or related to competition activities; and B. The Federal Government for damage or loss to Government property resulting from such an activity.	Admin		
	Rule 2.D: Use of NASA Name and Insignia Competitor Teams may not use the name or insignia of NASA on its hardware and printed materials related to the participation of Competitor Teams in the Challenge without NASA's prior written consent. Competitor Teams agree that unauthorized use of such names, trademarks, and insignias shall result in elimination from Challenge participation if Competitor Teams continue unauthorized use after being notified to cease and desist by NASA.	Admin		
	Rule 2.E: Compliance with Existing Laws - Competitors will comply with all U.S. laws, regulations and policies, including those relating to export control and nonproliferation, and the laws of relevant state and local jurisdictions that NASA Centennial Challenges pertain to or govern any activities conducted by Competitors in connection with the Challenge.	Admin		
	Rule 2.F: Reporting - On a monthly basis, Competitor Teams agree to provide NASA with a written total (a single amount) of the following: Competitor Team's incremental and cumulative financial, property (capital), personnel, and any other investments, and/or expenditures (direct or in-kind) made to conduct any and all activities related to or required by participation of the Competitor Team in the Challenge. NASA will not make this information public except in aggregate form for all Competitor Teams competing in the Challenge.	Admin		
	Rule 2.G: Media Rights The Competitor Team retains all Media Rights related to the story of its participation in the Challenge. The Competitor Team agrees that NASA will retain all Media Rights related to the story of the Challenge. Each Team Member agrees to let NASA use the name and likeness of such Team Member (without charge) as may be reasonably required in connection with the media material prepared and distributed by NASA relating in any way to the Challenge. The Competitor Team agrees to provide NASA reasonable amounts of video footage or access for recording activities related to participation of Competitor Team in the Challenge and the right to use said footage for public affairs and/or educational purposes. The Competitor Team agrees that its failure to furnish video footage or access for recording purposes based on NASA's reasonable requests may result in the Competitor Team's removal from participation in the Challenge.	Admin		
	Rule 2.H: Purchase and Sales Rights The Competitor Team agrees that NASA retains the non-exclusive right to purchase from Competitor Team the resultant or derived product, service, or technology used to win the Challenge. This section does not guarantee a purchase of the resultant or derived product, service, or technology and is subject at all times to the parties reaching mutual agreement after the Challenge. The Competitor Team retains all rights to sell the resultant or derived product, service, or technology used to win the Challenge to whomever they wish, provided they abide by all local, state, and federal laws and regulations regarding the sale and export of technology.	Admin		

	Rule 2.I: Intellectual Property Rights Notwithstanding anything to the contrary in these rules, NASA claims no intellectual property (IP) rights from the Competitor Team. All trade secrets, copyrights, patent rights, and software rights will remain with each respective Competitor Team. To the extent the Competitor Team owns IP resulting from its participation in Challenge, the Competitor Team agrees to negotiate in good faith with NASA for a grant of a nonexclusive, nontransferable, irrevocable, license to practice or have practiced for or on behalf of the United States, the intellectual property throughout the world, at reasonable compensation, if NASA chooses to pursue such a license.	Admin		
	Rule 2.J: Delay, Cancellation or Termination The Competitor Team acknowledges that circumstances may arise that require the Challenge to be delayed indefinitely or cancelled. Such delay or cancellation, and/or the termination of the challenge, shall be within the full discretion of NASA, and the Competitor Team accepts any risk of damage or loss due to such delay, cancellation, and/or termination.	Admin		
	Rule 3: Competitor Teams shall submit to NASA a Mission Concept Registration Data Package within 60 calendar days after their registration (and 30 calendar days before they may participate in any of the GTs). The Mission Concept Registration Data Package is defined in a separate document (available on the Cube Quest Challenge website). It includes at least the following content: • Statement of Intent to Compete • Concept of Operations • Conceptual Mission Design • Conceptual method for CubeSat disposal • Satellite Communications Concept	Admin		
	4.2 EM-1 Launch and Schedule The Ground Tournaments schedule and the EM-1 payload delivery, payload integration, and launch schedules shall be according to a separately published Cube Quest Challenge schedule (CCP-CQ-SCHED-001). Schedule will be published on the Cube Quest Challenge website. If any reason arises such that payload integration, launch, and deployment on the EM-1 mission cannot take place as planned for the Cube Quest Challenge, NASA will investigate launch alternatives. If no reasonable alternatives are found to be available, NASA reserves the right to postpone, modify, or cancel the in-space portion of the Challenge.	Admin		
	4.2.1 Notification to Competitors of EM-1 Deployment Trajectory NASA provides updates on the Centennial Challenge Program Cube Quest Challenge website of the planned orbital elements of the Space Launch System (SLS) upper stage after its disposal maneuver. The final orbital elements of the EM-1 Secondary Payloads will be posted within 24 hours after the actual EM-1 Secondary Payload deployment maneuver. Competitor Teams deployed from EM-1 SLS upper stage will be notified of confirmation of successful deployment of their CubeSat as soon as possible after the event. This time constitutes the "Start of Competition" as defined in Rule 15.	N/A		
	4.3 Design Requirements			
	Rule 4: CubeSat Mass, Volume, and Interface Requirements			
	Rule 4.A: To be eligible for NASA EM-1 Launch, the Competitor's CubeSat shall meet all the requirements of the SLS Secondary Payload Deployment System Interface Definition Requirements Document (IDRD). In the event of a conflict between the SLS IDRD and these Competition Rules, the SLS IDRD shall take precedence. The IDRD will be available to Competitor Teams no later than GT-2.	Admin		
	Rule 4.B: For both EM-1 and non-EM-1 launches, payloads shall meet 6U size and mass requirements as defined in the latest version of the SLS Secondary Payload IDRD.	Technical	Yes	All
	Rule 4.C: A Competitor Team may submit and operate only one single payload, compliant with the 6U volume and mass constraint as specified in the SLS Secondary Payload IDRD, eligible for Prizes.	Technical	Yes	All

Table ________ ACS Spacecraft Requirements Matrix

	Rule 4.D: Competitor Teams with non EM-1 launches shall be responsible for determining, and complying with, their own respective responsibilities and requirements with the third-party launch vehicle provider. NASA will not assist with compliance with third party launch provider requirements.	Technical & Admin	Yes	All
	Rule 4.E: In case of any discrepancy between the volume and mass allowances of NonEM-1 launch providers and those of the NASA EM-1 launch, the allowances of the SLS Secondary Payload User's Guide and IDRD shall take precedence for Challenges eligibility.	Technical	Yes	All
	Rule 4.F: Competitor Teams with non-EM-1 launches shall submit a Required Data for Competitor Teams with Non-NASA Launch package (defined in a separate document) at least 2 weeks prior to payload integration, and shall allow a Challenge-designated government inspector to verify by inspection, test, or other method of verification, the data it contains.	Technical	Yes	All
	Rule 5: Radio Frequency Authorization			
	Rule 5.A: Competitors agree that use of Radio Frequencies (RF) for any purpose, such as spacecraft tracking and control, information (data) transmission to and from the spacecraft, or active sensors, will be in accordance with all U.S. laws and regulations, and with the International Radio Regulations promulgated by the International Telecommunication Union (ITU). The controlling organization for each CubeSat shall obtain Federal Communications Commission (FCC) radio frequency authorization in accordance with the Rules and Regulations, Title 47, of the Code of Federal Regulations. FCC Public Notice DA: 13-445 (http://www.fcc.gov/document/guidance-obtaining-licenses-small-satellites) is useful in deciding authorization options to consider.	Technical		COMM
	Rule 5.B: For all communications, including communications eligible for these Challenges, any electromagnetic spectrum frequency (e.g., RF, infrared, visible light, etc.) is allowed, subject to all applicable RF licensing and spectrum allocation Rules.	Technical		COMM
	Rule 5.C: Competitors are responsible for obtaining necessary RF operating licenses for both their CubeSat space stations and for all ground stations under their control, and are responsible for abiding by National and International Rules governing radio operators in their operating spectrum.	Technical		COMM
	4.4 Monitoring and Inspections			
	Rule 6: Competitors shall permit NASA to non-invasively monitor any space-based communication relevant to the Challenges, using NASA's resources without prior notification to the Competitors. This monitoring may be used to verify compliance with the Challenge Rules and may be used to validate Competitor Team's submissions. This monitoring will not be used as a Competitor Team's official entry into competition. Competitor Teams may not use data encryption (other than encryption authorized by NASA) for transmission of commands or data relevant to the Challenges.	Admin		
	Rule 7: Competitors shall permit NASA visits to Competitor's operations sites, and permit inspection of cubesats, dispensers, ground equipment and operating procedures. Visits may be used to verify compliance with the Challenge Rules.	Admin		
	4.5 Rules for Ground Tournament As specified in the Rules below, GT scores are based on judges assessment of each Competitor Team's compliance to specific Challenge Rules and SLS Interface Requirements, and assessment of mission success probability for meeting the minimum requirements for either (or both) the In-space Prizes (depending on which In-space Prize(s) the Competitor Team indicates they intend to enter).	Admin		

Table ________ ACS Spacecraft Requirements Matrix

	4.5.1 Ground Tournament Constraints Any Competitor Team may participate in any or all of the Ground Tournaments (GTs). To participate in any GT, it is not necessary to have competed in the previous GTs. For example, a Competitor Team is not required to have participated in GT-1, 2, or 3 in order to participate in GT-4. However, the Competitor Team must submit their Mission Concept Registration Data Package (defined in Rule 3) at least 30 calendar days prior to their first GT in which they participate or by the published date. Judging criteria and expected degrees of design maturity advance progressively for each successive GT, and all Competitor Teams (whether they are pursuing an EM-1 spot or a third party launch) are judged by the same technical criteria at each GT.	Admin		
	Rule 8: Constraints on Ground Tournament Participation			
	Rule 8.A: Registered Competitor Teams may participate in any, or all, of the Ground Tournaments (GT). Competitor Teams that arrange for independent, thirdparty launches may, but are not required to, participate in any GT.	Admin		
	Rule 8.B: Competitor Teams shall submit a Mission Concept Registration Data Package (defined in Rule 3) at least 30 calendar days prior to participating in their first GT.	Admin		
	Rule 8.C: Before each GT, Competitor Teams shall declare whether they intend to compete in either the Deep Space Derby or the Lunar Derby or both. Competitors may change their declaration prior to each GT. These declarations may be made publicly available on the Challenge website.	Admin		
	Rule 8.D: Prior to each GT, Competitor Teams shall declare their intention to compete for integration and launch on EM-1, or their intention to arrange for their own independent, third-party launch. Competitors may change their declaration until GT-2 at which point they must make a final declaration. These declarations may be made publicly available on the Challenge website.	Admin		
	Rule 8.E: Competitors shall participate in at least GT-2 to be considered for selection as a secondary payload on the EM-1 launch.	Admin		
	Rule 8.F: The SLS Program requires a series of four Payload Safety Reviews (Phase 0 - Phase 3 Safety Reviews) before any CubeSat is accepted for integration to launch on EM-1. Only the top 5 winners of GT-1 and GT-2 will be submitted to Phase 0 and Phase 1 Safety Reviews, respectively. Only those Competitor Teams that pass the Phase 0 or the Phase 1 Safety Review may proceed to the Phase 2 Safety Review. Only those Competitor Teams that pass the Phase 2 Safety Review and are a top 5 winner in GT-4 may proceed to the Phase 3 Safety Review. The effect of these constraints is that only the top 5 winners of GT-1 or GT-2, who proceed to be top 5 winners of GT-4, will be eligible to launch on EM-1.	Admin		
	4.5.2 Procedures and Judging for Ground Tournament Ground Tournaments (GTs) require Competitor Teams to deliver submittal materials specified in the Ground Tournament Workbook, and to deliver interactive presentations to judges, either by video conference or in person at locations to be specified for each GT. Judges will consult with a NASA design center and/or third-party experts, and run mission simulations and analysis using product specifications and performance projections submitted by each Competitor Team 30 calendar days prior to GT. Judges will provide scores to Competitor Teams using standardized criteria, based on a scale of 1 (low, poor) to 5 (high, superb). A score of zero will be given for elements in which insufficient or no data was submitted. Judges will provide scores to Competitor Teams within two weeks of their GT.	Admin		
	Rule 9: Ground Tournament Judging			
	Rule 9.A: For each GT, Competitors shall submit required documents and data as listed on the Judges Score Card on dates specified in the published GT schedules. GT judging templates will be provided in advance to the Competitor Teams.	Admin		

Table ________ ACS Spacecraft Requirements Matrix

	Rule 9.B: Competitors shall permit Judges, or designee, (upon request) to conduct site inspections, inspections of competition hardware and/or software, and allow component or subsystem tests witnessing in order to verify submitted documentation.	Admin		
	Rule 9.C: Competitor Teams shall allow their composite scores to be posted on the Challenge website after each GT. (Competitor Team technical Intellectual Property will not be publicly released.)	Admin		
	Rule 9.D: All Competitor Teams shall be judged by the same criteria at each GT for probability of mission success, and for compliance with specific Challenge Rules.	Admin		
	Rule 9.E: For each GT, 40% of each Competitor Team's assigned total score will be determined by the probability of mission success using the Judge's Scorecard.	Admin		
	Rule 9.F: For each GT, 60% of each Competitor Team's assigned total score will be determined by compliance to specific Challenge Rules and (a) for Teams that state their intention to launch on EM-1 SLS: compliance with SLS Interface Requirements as defined in the SLS Secondary Payload Deployment System IDRD; or (b) for Teams that state their intention to launch on a vehicle other than SLS: compliance with the written interface and safety requirements of the team-procured launch service provider.	Technical & Admin	Yes	All
	Rule 9.G: Competitor Teams that arrange for their own third party launches must submit information required in Required Data for Competitor Teams with Non-NASA Launch, and will be judged for compliance with interface and safety requirements of their own launch operators, instead of for compliance with SLS Interface Requirements.	Technical	Yes	All
	4.5.2.1 Rules and Requirements for GT-1 Competition Judges will provide Competitor Team scores based on standardized assessments. Every Competitor Team (up to maximum of 5 Competitor Teams) whose composite score is greater than 3.0 will be awarded $20,000 each; however if more than 5 Competitor Teams score greater than 3.0 (composite score), only the 5 highest scoring Competitor Teams will be awarded $20,000 each. Only the GT-1 winners will submit their CubeSat designs for the SLS Phase 0 Payload Safety Review. Only Competitor Teams that pass SLS Phase 0/1 Payload Safety Reviews are eligible for future Safety Reviews and eligible to integrate and launch on EM-1.	Admin		
	Rule 10: To participate in the GT-1 and be eligible for GT-1 Prize Awards, Competitor Teams shall provide to NASA the input listed on the Judges Score Card.	Technical	Yes	All
	4.5.2.2 Rules and Requirements for GT-2 Competition Judges will provide Competitor Team scores based on standardized assessments. Every Competitor Team (up to maximum of 5 Competitor Teams) whose composite score is greater than 3.0 will be awarded $30,000 each; however if more than 5 Competitor Teams score greater than 3.0 (composite score), only the 5 highest scoring Competitor Teams will be awarded $30,000 each. Only the GT-2 winners will submit their CubeSat designs to SLS Phase 0/1 Payload Safety Review. GT-1 winners that successfully completed the Phase 0 payload safety review but not selected in GT-2 will submit for Phase 1 payload safety review. Only Competitor Teams that NASA Centennial Challenges pass SLS Phase 0 and 1 Payload Safety Reviews are eligible for future Safety Reviews and eligible to integrate and launch on EM-1.	Admin		
	Rule 11: To participate in the GT-2 and be eligible for GT-2 Prize Awards, Competitor Teams shall provide to NASA the input listed on the Judges Score Card.	Admin		
	Rule 11.A: Prior to GT-2, Competitor Teams must declare their final intention to compete for selection to launch on EM-1 4.5.2.3 Rules and Requirements for GT-3 Competition Judges will provide Competitor Team scores based on standardized assessments. Every Competitor Team (up to maximum of 5 Competitor Teams) whose composite score is greater than 3.0 will be awarded $30,000 each; however if more than 5 Competitor Teams score greater than 3.0 (composite score), only the 5 highest scoring Competitor Teams will be awarded $30,000 each	Admin		

	Rule 12: To participate in the GT-3 and be eligible for GT-3 Prize Awards, Competitor Teams shall provide to NASA the input listed on the Judges Score Card. 4.5.2.4 Rules and Requirements for GT-4 Competition The GT-4 is the final ground competition. Participation in GT-4 is required for all Competitor Teams who qualified in GT-1 or GT-2, passed all required safety reviews, and are requesting integration and launch on EM-1. Judges will provide Competitor Team scores based on standardized assessments. Every Competitor Team (up to maximum of 5 Competitor Teams) whose composite score is greater than 3.0 will be awarded $20,000 each; however if more than 5 Competitor Teams score greater than 3.0 (composite score), only the 5 highest scoring Competitor Teams will be awarded $20,000 each.	Admin		
	Rule 13: GT-4 Rules and Requirements			
	Rule 13.A: Prior to GT-4, Competitor Teams must also declare their final intention to compete in the Deep Space Derby, or the Lunar Derby, or both.	Admin		
	Rule 13.B: To participate in the GT-4 and be eligible for GT-4 Prize Awards, Competitor Teams shall provide to NASA the input listed on the Judges Score Card	Technical	Yes	All
	Rule 13.C: Only the top 5 highest scoring Competitor Teams that achieve all the following: • receive a GT-4 score of at least 3 and are in compliance with all Challenge requirements and Space Launch System Secondary Payload Deployment System Interface Definition and Requirements Document (IDRD) requirements, and • declared before GT-2, their intention to launch on EM-1, and • passed SLS Phase 2 Safety Review will be submitted to SLS Phase 3 Safety Review to become qualified for integration, launch, and deployment on EM-1.	N/A		
	4.5.3 Down Select Launch Candidates (conditional) Rule 14: In the event that the total number of qualified CubeSats exceeds the number of SLS dispenser slots assigned to Cube Quest Challenge, then the following down-select Rules shall apply:	N/A		
	Rule 14.A: Judges shall rank all Competitor Teams in order based on the GT-4 total score. In case of a tie, the tie breaker will be the highest cumulative score across all GTs.	Admin		
	Rule 14.B: At the present time, there are three slots on EM-1 allocated to CubeQuest Challenge. The top 3 teams that successfully pass Phase 3 SRB will be integrated on EM-1.	N/A		
	Rule 14.C: Teams 4 and 5, if they successfully pass the Phase 3 SRB, will be used to backfill in the event that any EM-1 selected competitor team cannot deliver their Cubesat for vehicle integration. "Runner's up" should be prepared (at a moment's notice) to replace any selected Competitor Team up until actual vehicle integration date.	N/A		
	Rule 14.D: Deleted			
	4.6 General Rules Applicable to Both In-Space			
	Rule 15: In-Space Competition Start ("Start of Competition")			
	Rule 15.A: Competitors that have arranged their own third party launches shall notify Judges within one day of their deployment confirmation receipt. The positive deployment confirmation time shall be considered the start time of the first competition day of their respective "Start of Competition".	Technical & Admin		

Table ________ ACS Spacecraft Requirements Matrix

	Rule 15.B: For Competitors with CubeSats deployed from EM-1, the positive deployment confirmation time shall be considered the start of the first competition day for all Competitor Teams with CubeSats deployed from EM- 1. (Note that the SLS Payload User's Guide and/or the SLS Secondary Payload Deployment System Interface Definition and Requirements Document may specify a timed delay before CubeSats may begin powered operation after the deployment from the SLS. Nevertheless, the deployment confirmation time shall be considered the "Start of Competition" for CubeSats deployed from EM-1.) In support, NASA will notify Competitors within one day of their successful deployment from EM-1 SLS.	N/A		
	Rule 16: Competitor Ground Stations			
	Rule 16.A: Competitor Teams may communicate with, and update, their CubeSat as often as desired within the competition period. This includes commands, revised operating instructions, software updates, etc.	Technical		Ground Systems COMM
	Rule 16.B: Earth-based transmissions and receptions may be performed from the same ground station or differing ground stations.	Technical		Ground Systems COMM
	Rule 16.C: Competitor Teams may not use Government controlled stations as their primary data communications stations for the purposes of communications NASA Centennial Challenges achievements eligible for in-space Prizes, unless appropriate compensation is provided and the station is also made available under the same terms to all Competitors.	Technical & Admin		Ground Systems COMM
	Rule 16.D: Competitor Teams will not be charged for communications monitoring by Government-controlled stations strictly for the purpose of authenticating claimed communications distances, or for verifying the achievement and maintenance of lunar orbit. See Required Navigation Artifacts for Authenticating Claimed Comm Distances and Verifying Achievement and Maintenance of Lunar Orbit.	Technical & Admin		Ground Systems COMM
	Rule 16.E: Ground station operators may be Team Members (Rules 1 and 2 apply), or ground station services or facilities may be procured by the Competitor Team (Rules 1 and 2 do not apply, except for Rule 1.C).	Technical & Admin		Ground Systems COMM
	Rule 17: Planetary Protection			
	Rule 17.A: Competitor Teams shall submit Orbital Debris Assessment Reports (ODARs) and End of Mission Plans (EOMPs) that are compliant with NASA-STD- 8719.14 Process for Eliminating Orbital Debris, in order to be compliant with U.S. National Space Policy of the United States of America (June 2010), the U.S. Government Orbital Debris Mitigation Standard Practices (February 2001), and other National and International policies and guidelines for limiting Earth-orbiting debris.	Technical & Admin	Yes	All
	Rule 17.B: Competitor Teams shall submit their ODARs and EOMPs to Judges no later than Ground Tournament 4.	Technical & Admin	Yes	All
	Rule 17.C: Competitor Teams with CubeSats that enter lunar orbit shall submit an End of Mission Plan that, to the satisfaction of Judges, complies with "NASA's Recommendations to Space-Faring Entities: How to Protect and Preserve the Historic and Scientific Value of U.S. Government Lunar Artifacts" found at http://www.nasa.gov/sites/default/files/617743main_NASAUSG_LUNAR_HISTORIC_SITES_RevA-508.pdf	Technical & Admin	Yes	All
	Rule 17.D: Competitor Team mission designs must be compliant with requirements of NPR 8020.12 Planetary Protection Provisions for Robotic Extraterrestrial Missions. For Competitor Teams that demonstrate to the satisfaction of Judges (by trajectory simulation/analysis or other documentation) that their CubeSats will not encounter any protected planet (beyond Earth and Earth's moon), then written documentation compliant with NPR 8020.12 is the only requirement for planetary protection. (Tests and demonstrations would not be required.)	Technical & Admin	Yes	All

Table ________ ACS Spacecraft Requirements Matrix

	Rule 17.E: Competitor Teams shall submit a letter to Judges explaining their planetary protection plan at GT-1. Competitor Teams shall submit their final planetary protection plan at GT-4. Competitor Teams shall submit a Pre-launch report for purposes of compliance with NPR 8020.12 at L-60 calendar days. Competitor Teams shall submit a post-launch report at L+60 calendar days. Competitor Teams shall submit an EOMP at mission end.	Admin		
	Rule 18: Communications Competition Procedure for Both In-Space Challenges. The exact details of the implementation of the following Rules are contained in the supplemental document Communications Procedure for Both In-Space Challenges (CommsProc).	Technical & Admin		DMS COMM
	Rule 18.A: Each Competitor Team shall inform Judges a minimum of 24 hours prior to the start of each operating period (as specified in CommsProc). If the Competitor Team does not announce operating periods, then Judges will not consider any operations that day for competition purposes.	Technical & Admin		Ground Systems & Payload Systems
	Rule 18.B: Competitor Teams shall generate their random data using the algorithms and protocols specified in CommsProc. Judges will not accept data generated by any other methodology.	Technical		DMS
	Rule 18.C: The Competitor Team shall supply a CubeSat communications log to the Judges to verify competition timing.	Technical		DMS COMM
	Rule 18.D: Competitor Teams may choose to wrap data blocks in a convenient protocol for transmission to assist with block accounting and sequencing as long as the Judges can verify that data were generated by the prescribed algorithm.	Technical		DMS COMM
	Rule 18.E: The Competitor Team shall receive the data blocks over the communications link, perform any required error correction deemed necessary, and arrange the blocks in correct sequence. Any blocks that are not completely received within the operating period will not count towards the operating period total.	Technical		
	Rule 18.F: The Competitor Team shall deliver to NASA properly sequenced, unique (nonduplicative) error-free data blocks received at the ground station(s) within 10 minutes of the operating period closure. If the Competitor Team requires a data retransmission to achieve an error-free block, the Competitor Team must complete that transaction by the end of the operating period.	Technical		
	Rule 18.G: As specified in CommsProc, the Competitor Team shall provide the evidence that authenticates actual transmission achievement from their spacecraft in space and ground station receipt to the Judges. The Competitor Team shall make raw data available to the Judges at the same time as the Competitor Team presents the sequenced data. Judges shall also receive contact logs from the ground station operators. Logs are to include (at minimum) pointing data, AZ/EL coordinates, and receiver start/stop times. Competitor Teams shall provide documented	Technical		
	Rule 19.G compliance procedures before GT-3.			
	Rule 19: Competition End for Both In-Space Challenges ("End of Competition")			
	Rule 19.A: For Competitor Teams that have arranged their own third party launch, all activities for the purposes of these Challenges shall end exactly 365 competition days after their respective CubeSat deployment confirmation time, or exactly 365 competition days after the EM-1 deployment confirmation time, whichever occurs first.	Technical	Yes	All
	Rule 19.B: For Competitor Teams deployed on EM-1, all activities for the purposes of these Challenges end exactly 365 competition days after the EM-1 deployment confirmation time.	N/A		
	Rule 19.C: No activity taking place later than exactly 365 competition days after the EM-1 CubeSat deployment shall be counted for Challenge purposes, regardless of the respective launch dates.	Admin		

Table ________ ACS Spacecraft Requirements Matrix

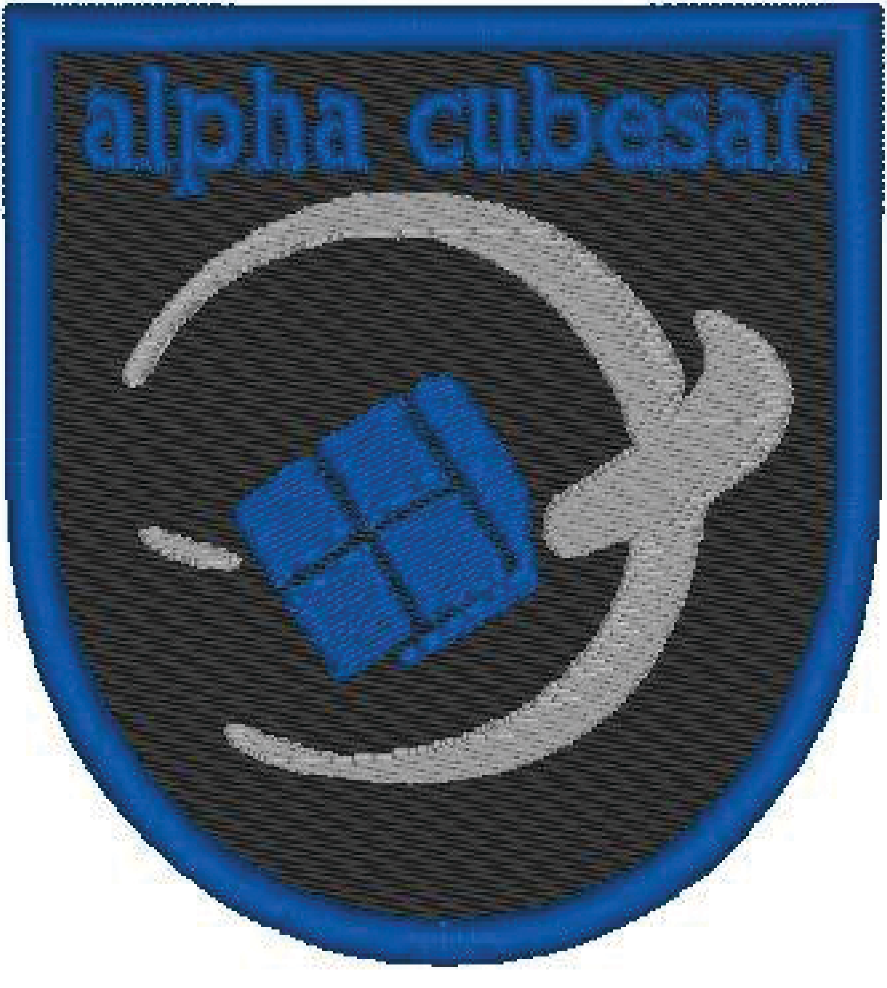
	Rule 19.D: For Competitor Teams that have arranged their own third party launch, data transmissions after 365 calendar days will only be used for longevity category entrants regardless of data burst rate or data aggregate volume. Data transmissions must meet minimum requirements based on the prize.	Technical		COMM
	Rule 20: If, for any reason, a CubeSat does not successfully deploy from EM-1 (a dispenser malfunction, for example), then that Competitor Team shall be ineligible for any In-space Prizes.	N/A		
	Rule 21: Competitor Teams shall acknowledge that NASA reserves the right to share information about Competitor Team accomplishments and progress, after verification by Judges, throughout the Challenge period. Accomplishment or progress information may include, for example, the data volumes communicated, time of lunar orbit, and cubesat distances from Earth. NASA also reserves the right to publicly announce when Competitor Teams are planning to attempt a communications task or propulsion maneuver before results have been confirmed by Judges.	Admin		GN&C COMM Ground Systems
	4.7 Additional Rules for Deep Space Derby			
	Rule 22: Achievement and Maintenance of Verifiable Minimum Required Distance from Earth			
	Rule 22.A: Competitor CubeSats shall achieve and maintain a verifiable minimum required distance from Earth's surface of at least 4,000,000 kilometers (+/- 4,000 km allowable tolerance) during any operations that would count toward the Deep Space Derby Prizes achievements.	Technical	es	GN&C COMM Ground Systems
	Rule 22.B: Competitors shall provide evidence that demonstrates, to the Judges' satisfaction, the spacecraft distance from Earth. (Acceptable evidence to be submitted to NASA for purposes of authenticating the claimed distance from Earth is specified in Required Navigation Artifacts for Authenticating Claimed Comm Distances and Verifying Achievement and Maintenance of Lunar Orbit, a separate document.)	Technical		GN&C COMM Ground Systems
	Rule 22.C: In the event that no CubeSat successfully reaches the minimum distance from Earth (Rule 22.A) within 365 competition days of the EM-1 launch, NASA will declare the Deep Space Derby over with no winner and no prizes awarded.	Technical & Admin	Yes	All
	Rule 23: Deep Space Derby Prizes			
	Rule 23.A: Best Burst Data Rate: $225,000 will be awarded to the Competitor Team that receives the largest, and $25,000 will be awarded to the Competitor Team that receives the second largest, cumulative volume of error-free data (above the minimum volume of one 1024 bit data block) from their CubeSat over a 30-minute period while satisfying Challenge Rules and definitions. If only one Competitor Team achieves more than the minimum volume, they are awarded $250,000. If no Competitor Team achieves more than the minimum volume, no Best Burst Data Rate prize will be awarded. In case of a tie, all qualifying tied Competitor Teams will receive an equal portion of this prize amount.	Technical	Yes	All
	Rule 23.B: Largest Aggregate Data Volume Sustained Over Time: $675,000 will be awarded to the Competitor Team that receives the largest, $75,000 will be awarded to the Competitor Team that receives the second largest, cumulative volume of error free data (above the minimum volume of one thousand 1024 bit data blocks) from their CubeSat over their best contiguous 28-day (calendar days) period while satisfying Challenge Rules and definitions. If only one Competitor Team achieves more than the minimum volume, they are awarded $750,000. If no Competitor Team achieves more than the minimum volume, no Largest Aggregate Data Volume prize will be awarded. In case of a tie, all qualifying tied Competitor Teams will receive an equal portion of this prize amount.	Technical	Yes	All

Table ________ ACS Spacecraft Requirements Matrix

	Rule 23.C: Spacecraft Longevity: $225,000 will be awarded to the Competitor Team with the longest elapsed number of competition days, and $25,000 will be awarded to the Competitor Team with the second longest elapsed number of competition days, between the date of their first and last, confirmed reception of error-free, 1024-bit data blocks from their CubeSat while maintaining at least the minimum required distance from Earth, and before the "End of Competition" (above the minimum number of 28 elapsed competition days) while satisfying Challenge Rules and definitions. If only one Competitor Team achieves more than the minimum number of 28 elapsed competition days, they are awarded $250,000. If no Competitor Team achieves more than the minimum number of 28 competition days, no Longevity Contest prize will be awarded. In case of a tie, all qualifying tied Competitor Teams will receive an equal portion of this prize amount.	Technical	Yes	All
	Rule 23.D: Farthest Communication Distance From Earth: $225,000 will be awarded to the Competitor Team that receives from the CubeSat at least one, error-free, 1024-bit data block, from the greatest, and $25,000 will be awarded to the Competitor Team with the second greatest distance from Earth (above the minimum distance of 4,000,000 km), and before the "End of Competition", while satisfying Challenge Rules and definitions. If only one Competitor Team receives at least one, error-free 1024-bit data block (above the minimum distance of 4,000,000 km from Earth), they are awarded $250,000. If no Competitor Team receives data, no Farthest Communication Distance prize will be awarded. In case of a tie, all qualifying tied Competitor Teams will receive an equal portion of this prize amount.	Technical	Yes	All
	4.8 Additional Rules for Lunar Derby			
	Rule 24: Achievement and Maintenance of Verifiable Lunar Orbit			
	Rule 24.A: Competitor CubeSats shall achieve and maintain a verifiable lunar orbit, during any operation that would count towards the Lunar Derby Prizes achievements.	Technical		GN&C
	Rule 24.B: For the purpose of the Lunar Derby, a lunar orbit is defined as at least one complete orbit of minimum distance always above the lunar surface of 300 km, and with an aposelene that never exceeds 10,000 km.	Technical		GN&C
	Rule 24.C: Competitors shall provide evidence, to the Judge's satisfaction, that demonstrates that they have successfully achieved a lunar orbit, as defined in Rule 24.B. (Acceptable evidence to be submitted to NASA for purposes of authenticating claimed lunar orbit is specified in Required Navigation Artifacts for Authenticating Claimed Comm Distances and Verifying Achievement and Maintenance of Lunar Orbit, a separate document.)	Technical		GN&C
	Rule 24.D: Competitor Teams shall provide evidence demonstrating their CubeSat has maintained a minimum altitude of at least 300 km above the lunar surface at all times, before intentional end-of-mission disposal maneuvers.	Technical		GN&C
	Rule 24.E: Competitor Teams shall provide evidence, to the Judge's satisfaction, demonstrating that their CubeSats has maintained a lunar orbit (as defined in Rule 24.B) during any operations counting towards competition achievements or prize awards.	Technical		GN&C
	Rule 24.F: In the event that no CubeSat successfully achieves verifiable lunar orbit (as defined in Rule 24.B) within their respective 365-day (calendar days) competition, NASA will declare the Lunar Derby competition over with no winner and no prizes awarded.	Technical & Admin		
	Rule 25: Lunar Derby Prizes Rule			
	25.A: Lunar Propulsion: All contestant Competitor Teams that successfully demonstrate their CubeSat has achieved at least one verifiable lunar orbit and satisfy Challenge Rules and definitions shall be awarded an equal share of the $1,500,000 Lunar Propulsion Competition Prize, with a maximum of $1,000,000 to any one Competitor Team.	Technical	Yes	All

Table ________ ACS Spacecraft Requirements Matrix

	Rule 25.B: Best Burst Data Rate: $225,000 will be awarded to the Competitor Team that receives the largest, and $25,000 will be awarded to the Competitor Team that receives the second largest, cumulative volume of error-free data (above a minimum volume of one 1024 bit data block) from their CubeSat over their best 30-minute operating period while satisfying Challenge Rules and definitions. If only one Competitor Team achieves more than the minimum volume, they will be awarded $250,000. If no Competitor Team achieves more than the minimum volume, no Burst Data Rate prize will be awarded. In case of a tie, all qualifying tied Competitor Teams will receive an equal portion of this prize amount.	Technical	Yes	All
	Rule 25.C: Largest Aggregate Data Volume Sustained Over Time: $675,000 will be awarded to the Competitor Team that receives the largest, $75,000 will be awarded to the Competitor Team that receives the second largest, cumulative volume of error free data (above a minimum volume of one thousand 1024 bit data blocks) from their CubeSat over their best contiguous 28-day (calendar day) period while satisfying Challenge Rules and definitions. If only one Competitor Team achieves more than the minimum volume, they will be awarded $250,000. If no Competitor Team achieves more than the minimum volume, no Aggregate Data Volume prize will be awarded. In case of a tie, all qualifying tied Competitor Teams will receive an equal portion of this prize amount.	Technical	Yes	All
	Rule 25.D: Spacecraft Longevity Contest: $450,000 will be awarded to the Competitor Team that achieve the longest elapsed number of competition days between the first and last confirmed reception (greater than a minimum number of 28 elapsed competition days), and $50,000 will be awarded to the Competitor Team with the second longest elapsed number of competition days, of an error-free, 1024-bit data block from their CubeSat while satisfying Challenge Rules and definitions. If only one Competitor Team achieves more than the minimum number of 28 elapsed competition days, they will be awarded $500,000. If no Competitor Team achieves more than the minimum number of competition days, no Longevity Contest prize will be awarded. In case of a tie, all qualifying tied Competitor Teams will receive an equal portion of this prize amount.	Technical	Yes	All
	4.9 Additional Cube Quest Challenge Rules			
	Rule 26: The Centennial Challenge Program (CCP) has made significant effort to develop fair and just competition rules. In the event that the CCP deems it necessary, additional rules or requirements may be administered with the concurrence of all currently registered Competition Team(s). Failure to adopt or follow such additional rules or requirements shall be grounds to terminate a Competition Team and all Team Members from the Challenge.	Admin		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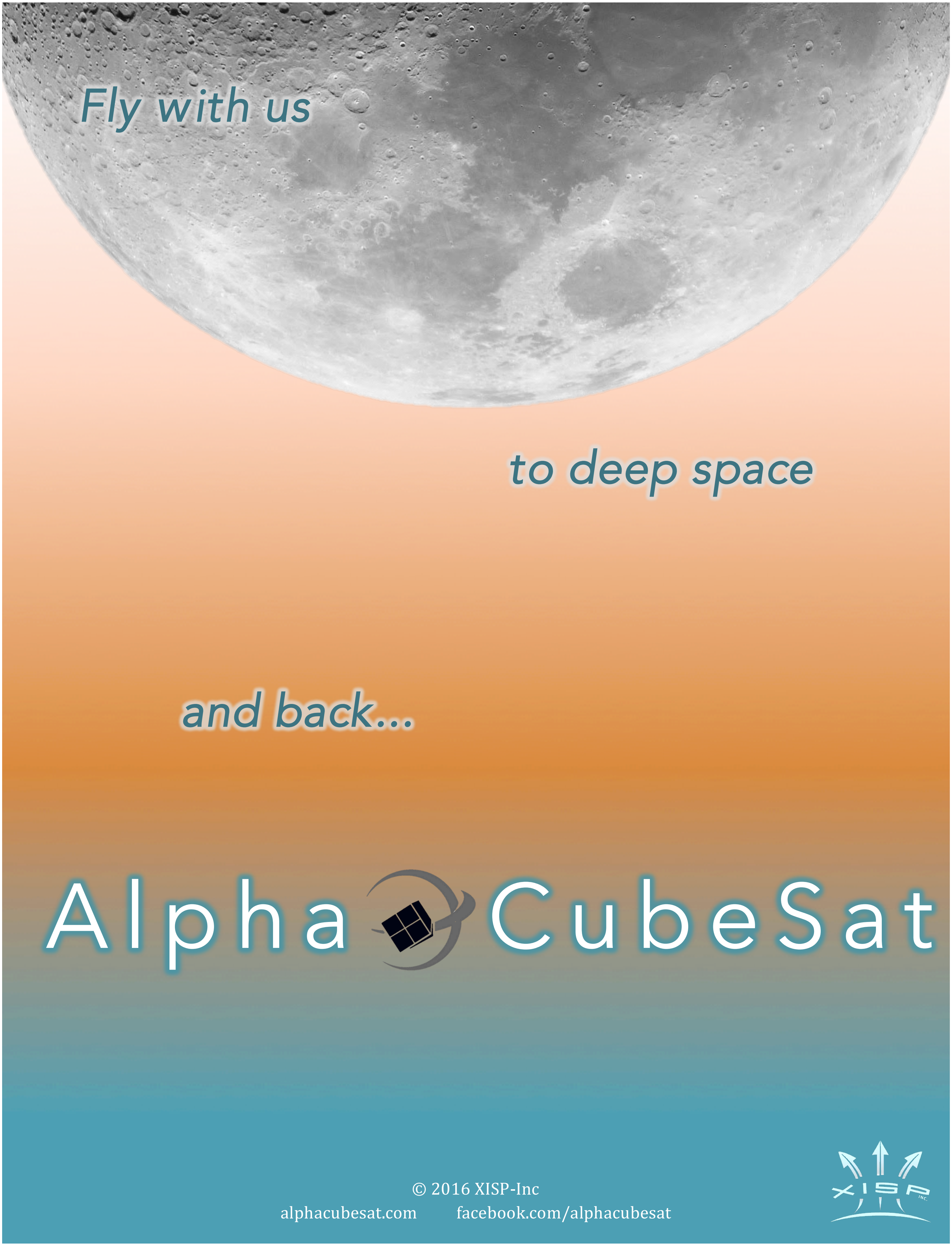
Fly with us
to deep space
and back...
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The Mission

The first mission of the Alpha CubeSat heritage will set an operational precedent for nanosatellites through: technology demonstration, deep space communication, launch & deployment, maneuvering, and lunar orbit. Success will occur through a combination of competition and cooperation.

The Alpha CubeSat Team is out to win the NASA Cube Quest Challenge. The Cube Quest Challenge, sponsored by NASA's Space Technology Mission Directorate Centennial Challenge Program, offers a total of $5 million to teams that meet the challenge objectives of designing, building, and delivering flight-qualified, small satellites capable of advanced operations near and beyond the moon.

> Alpha CubeSat will secure cheap and on demand access to space. With the use of new launch and deployment methods, the door for other nanosatellites' access to orbit will be blown open!

Alpha CubeSat will demonstrate innovative satellite instrumentation while following progressive, low-energy trajectories to reach a deep space altitude of 4 million km (about 10x farther than the moon!) before returning to the moon and establishing a strategic resonance orbit.

Design freedom and launch options afford an intrepidity lacking in new satellite missions: the courage to prove never flown before instruments, demonstrate efficient experimental orbits, and develop new launch opportunities for future cubesats.

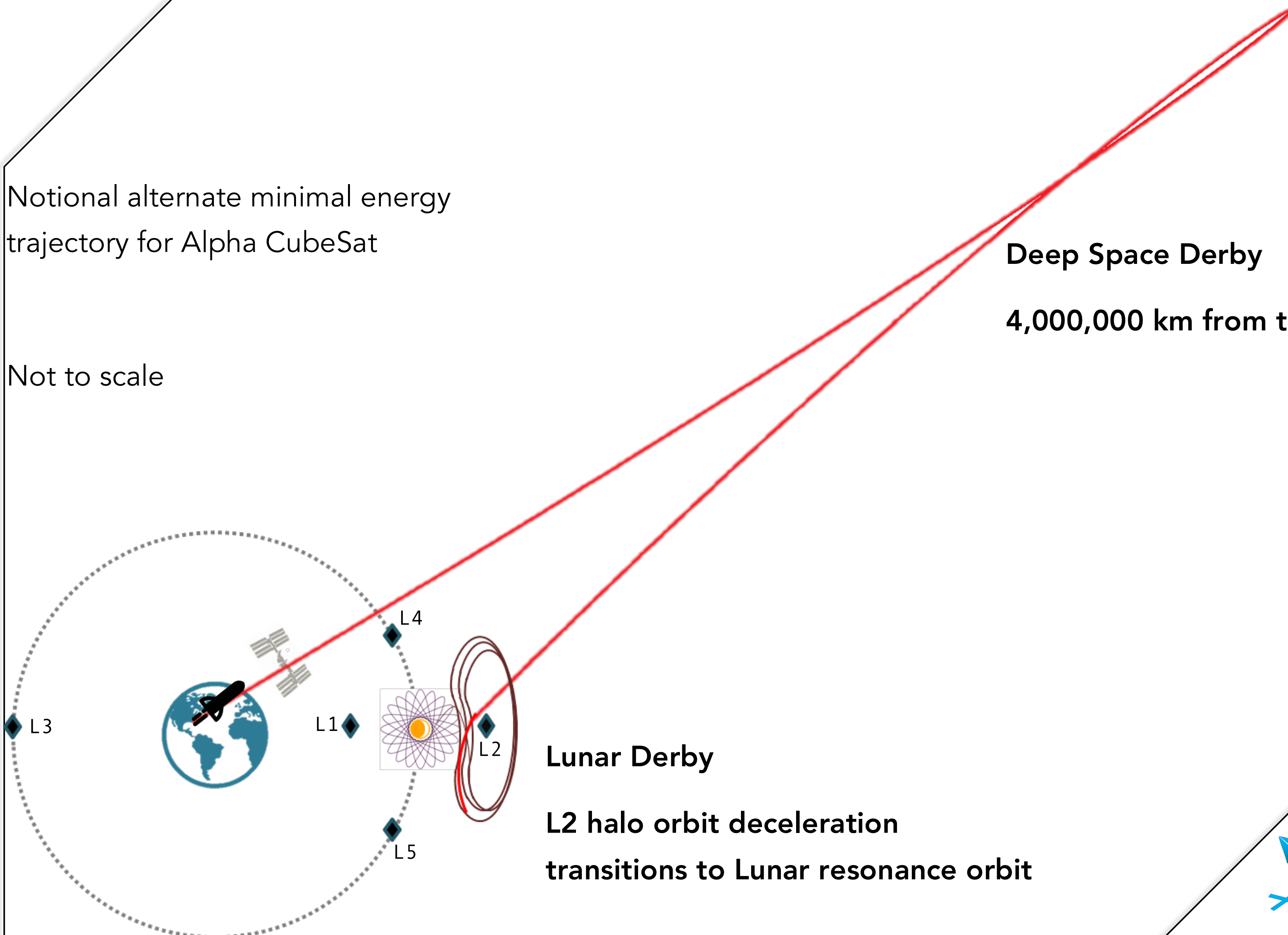
Innovative trajectories and orbits will provide high definition access of the moon's surface as well as backup communication provisions for independent space missions.

Xtraordinary Innovative Space Partnerships, Inc. (XISP-Inc) is the founding sponsor of Alpha CubeSat.

The Trajectory

Notional alternate minimal energy
trajectory for Alpha CubeSat

Not to scale



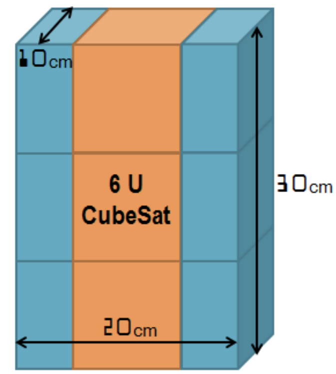
Deep Space Derby

4,000,000 km from the Earth

Lunar Derby

L2 halo orbit deceleration
transitions to Lunar resonance orbit

Alpha CubeSat Design



STRUCTURE

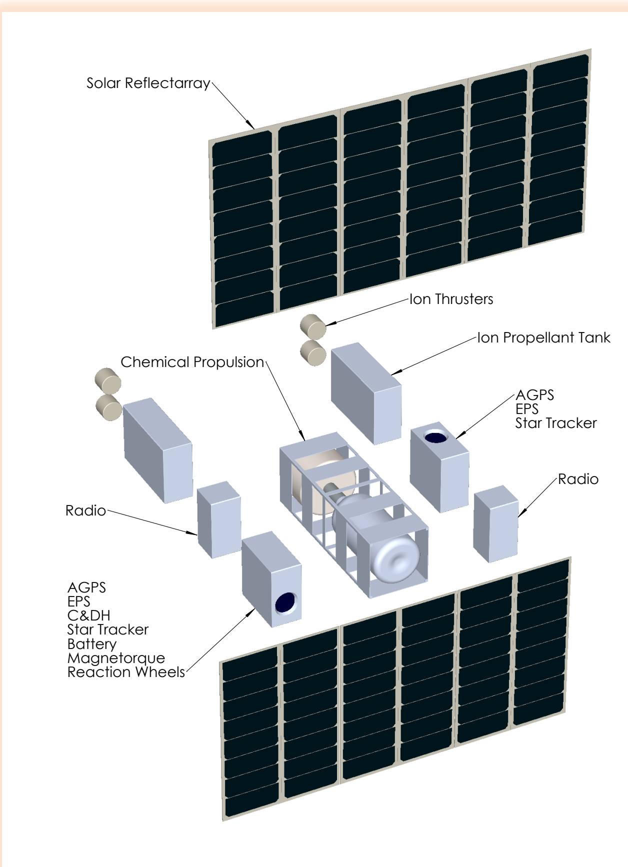
6U (10cm x 20cm x 30cm) CubeSat with deployable solar arrays. Nominal Mass 14 kg as constrained by NASA CubeQuest Challenge requirements.

COMMUNICATIONS

Ka Band is the frequency baseline for communications and should provide certainty with data acquisition during flight. The use of a new Ka Band nanosatellite transceiver will be one example of new technology to be demonstrated onboard Alpha CubeSat.

PROPULSION

A combination of low-thrust-long-duration and high-thrust-short-duration propulsion systems will be used by Alpha CubeSat after deep space trajectory insertion. A combination of Ion, electric, chemical, and thermal thrusters will be used to provide low-thrust-long-duration propulsion capabilities. In addition, the use of a high-thrust-short-duration propulsion system is baselined for high thrust trajectory maneuvers if required. An in-line hybrid Nitrous Oxide and Acrylic/Paraffin propulsion system and use of the International Space Station (ISS) in-situ resources are the leading alternatives at this time.

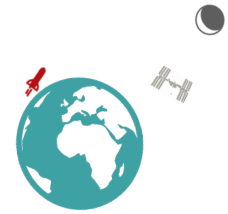


THERMAL

Alpha CubeSat will spend most of its life after leaving LEO in full sun. To manage thermal changes- likely scenarios include the need to turn the transmitter on often enough to help keep the satellite warm and to turn it off/throttle when it is in danger of overheating. Passive systems such as shading, coloring, selective placement of system/subsystem components as well as some active deployment of shades and louvers are being designed into the system.

To learn more details about our design contact us at info@alphacubesat.com or call (301)-509-0848.

Alpha CubeSat Flight



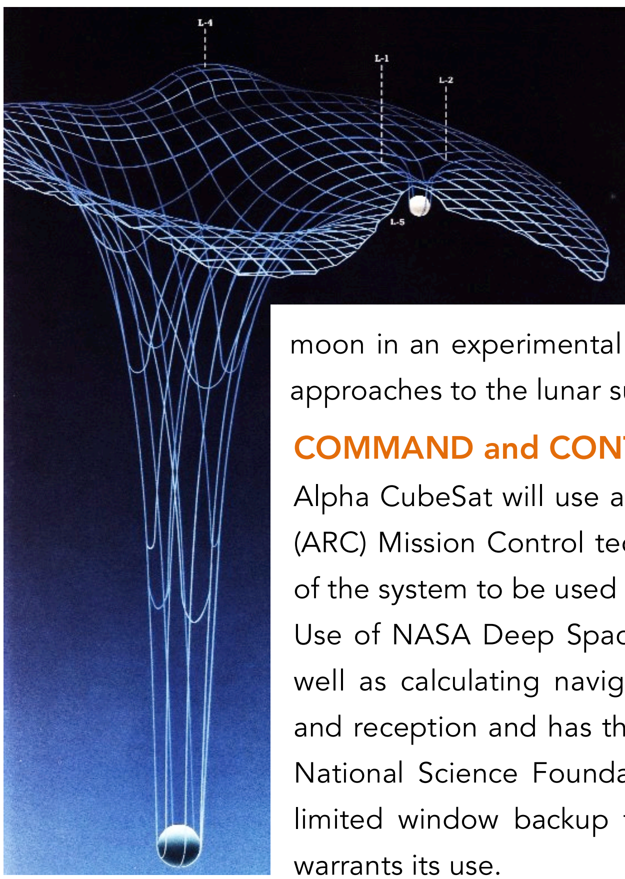
LAUNCH and DEPLOYMENT

The largest number of launch opportunities for CubeSats would be afforded by being manifested as ISS commercial cargo.

Baseline: Soft Pack Pressurized International Space Station (ISS) Cargo & ISS IntraVehicular Activity (IVA) Japanese Experiments Module (JEM) airlock transition to ExtraVehicular Robotic (EVR) Low Earth Orbit to Deep Space and CIs-Lunar Trajectory Insertion.

Alternate 1: EVR Deployed Unpressurized ISS Cargo & ISS logistics storage (JEM back porch) to EVR Low Earth Orbit to Deep Space and Cis-Lunar Trajectory Insertion.

Alternate 2: Leverage the expanding fleet of expendable launch vehicles such as secondary payload on SpaceX's Falcon 9, OrbitalATK's Antares, ULA's Atlas/Delta/Vulcan, or NASA's SLS Secondary Cargo & the Payload Planetary Services Systems release mechanism.



TRAJECTORIES

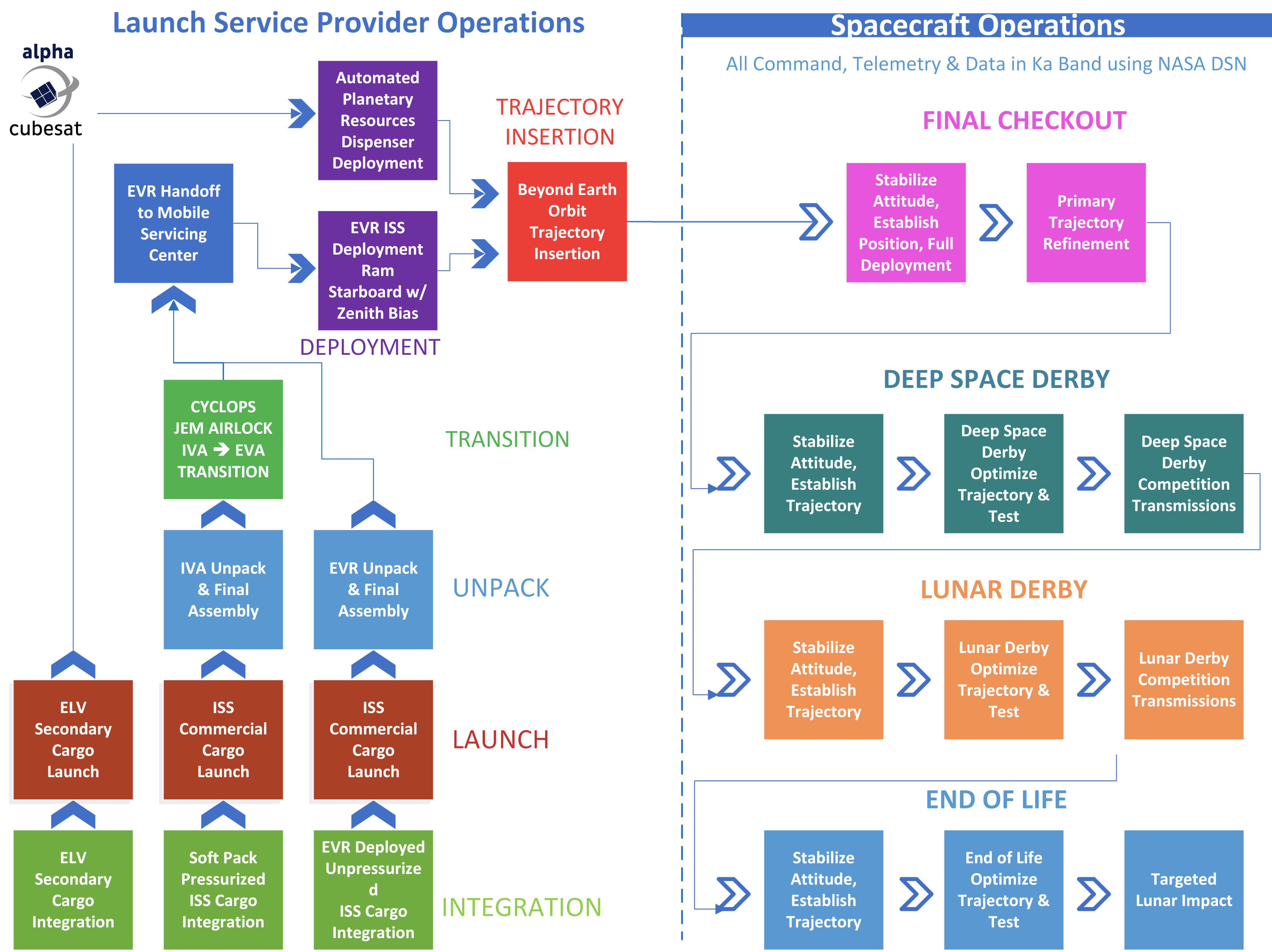
Inspired by Dr. Edward Belbruno and the late Dr. Robert Farquhar's trajectories for the ISEE-3 spacecraft, Alpha CubeSat will fly to an altitude of 4 million km using minimal fuel and taking advantage of the Earth-Moon gravity wells and Lagrange points. The ultimate goal is to orbit the moon in an experimental resonance orbit that will provide 50+ years of close approaches to the lunar surface with minimal orbital maintenance!

COMMAND and CONTROL

Alpha CubeSat will use an augmented set of the NASA Ames Research Center (ARC) Mission Control technologies suite enabling a near realtime state model of the system to be used to manage all command, telemetry, and data streams. Use of NASA Deep Space Network (DSN) is baselined for all transmissions as well as calculating navigation elements. DSN supports Ka band transmission and reception and has the largest number of readily available ground stations. National Science Foundation's Arecibo Observatory has been identified as a limited window backup facility in the event of an emergency condition that warrants its use.

To learn more details about our concept of operations contact us at info@alphacubesat.com or call (301)-509-0848.

Alpha CubeSat Concept of Operations



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The Future

1. Alpha Cubesat launches.
2. The cost of access to space for nanosatellites will dramatically decrease.
3. New nanosatellites will begin to lay the groundwork for space and terrestrial information beaming.

Space based information beaming will mean:

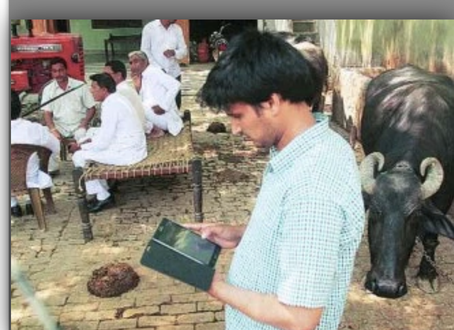
Immediate data transfers will be directed by satellites in low earth orbit. Consistent Earth-wide WiFi beamed from satellites will connect many struggling communities to opportunities for improving their life.

Nanosatellites can provide deep space mission communication support for independent missions by lending bandwidth for sending data packets or beaming power between spacecraft.

Creating jobs in struggling communities with Ground Station Development: space based solar power beaming technology will require Antennas on the Earth to receive energy transmissions.

Clean Energy! Space Based Solar Power will produce no toxic byproducts during operation. Growing economies that are dependent on fossil fuels will drastically decrease their carbon footprint.

Fostering international cooperation for our collective further advancement by demonstrating new technologies and advancing science.



Safety, health, and educational opportunities dramatically increase in struggling communities with affordable space access.

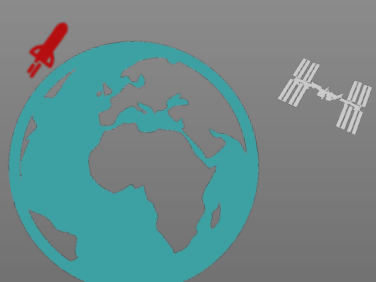
Call for Participants

Join Team Alpha CubeSat!

Alpha CubeSat will create a market for affordable space access. Our high visibility, cost effective, resource-rich platform will enable access to ISS and NASA ground center laboratories as well as close flyby's of the moon and deep space communication demonstrations. The result of flying Alpha CubeSat will set a new precedent for ease of integration and ease of launch for new space technologies.

Is your company looking for first flight opportunities for new technologies? Join our technology demonstration platform that will enjoy wide international exposure between 2017-2019.

Participate! Fly your instruments on Alpha CubeSat and leverage the value of the spacecraft by utilizing our existing design to support your technology demonstration.

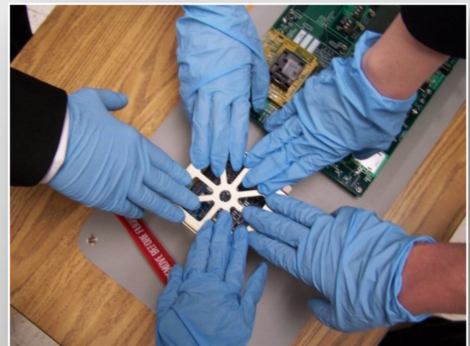


Contact us today!

Email info@alphacubesat.com or call Gary Barnhard at (301)-509-0848

Partners and Payloads

Join us as a Partner!



We are looking for subsystems or components to fly on Alpha CubeSat for the cost of the equipment.

Join Xtraordinary Innovative Space Partnerships, Inc as a partner and provide an essential component to the mission success of Alpha CubeSat.

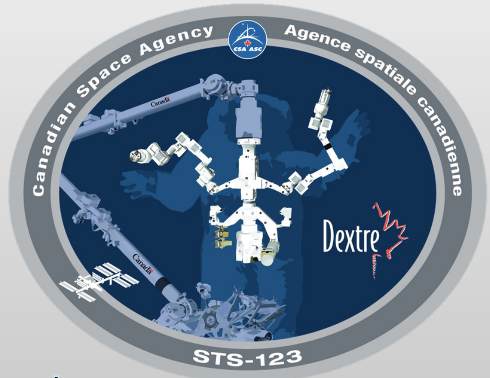
Join us as a Customer!

Provide a technology demonstrating payload to fly on Alpha CubeSat.

We are offering access to the deep space environment extending out past 4 million km as well as an opportunity for long duration, repeated high definition data acquisition of the moon.

Team Alpha CubeSat

Our team includes engineers with over 150 years of collective experience on the development of successfully flown spacecrafts.



Team Alpha CubeSat brings together an extraordinary combination of proven systems engineering talent, specialized discipline skills, and a shared commitment to build a mission of enduring value. Name a leading NASA contractor and it is guaranteed we have experience there!

The engineers on Team Alpha Cubesat have worked at leading space companies and on numerous rocket and satellite programs highlighted by the mission patches on this page. The following missions have flown with direct involvement from an Alpha CubeSat team member: NASA Galileo spacecraft; JPL Microwave limb Sounder on the Upper Atmosphere Research Satellite; Boeing 376 spin-stabilized spacecraft; body-stabilized Boeing 702 spacecraft; GOES N 601 Geostationary Operational Environmental Satellite; the International Space Station.

Our specialist advisors range from orbital mechanics to virtual reality experts - telecom and satcom innovators to presidents/founders/CEOs of prestigious space consultancies and leading asteroid mining companies.

Team Alpha CubeSat

FOUNDING SPONSOR:

Xtraordinary Innovative Space Partnerships, Inc. (XISP-Inc)


Team Lead
CEO/Systems Engineering
Gary Barnhard


Engineer - Propulsion Systems
Ethan Chew


Engineer – Structures & Mechanisms
John Tascione


Engineer – CAD/ Systems Integration
Mike Doty


Engineer - Guidance, Navigation, & Control
Brian Martin


Engineer – Thermal Systems
Eric Gustafson


Engineer - Radiation & Shielding
TJ Mckinney


Engineer – Propulsion Systems
Eric Shear


Multimedia Production
Jamie Pulliam


Engineer – Attitude Control
Justin Siples


Engineer - Structures & Mechanisms
Anastasia Ford


Contract Specialist Documentation
Joseph Rauscher

COMMERCIAL TEAMMATES:

Barnhard Associates, LLC
Deep Space Industries, Inc.

NON-PROFIT TEAMMATES/SPONSORS:

Space Development Foundation
National Space Society

ADVISORS:


Advisor Communication Systems
Pat Barthelow


Advisor Propulsion Systems
Craig Foulds

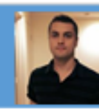

Advisor Trajectory Consultant
Ed BelBruno


Advisor Communications Systems
Aaron Harper


Advisor Astrophysics
Eric Dahlstrom

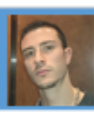

Advisor STK & Orbital Dynamics
Chris Cassell


Advisor Lunar Science Liaison
David Dunlop

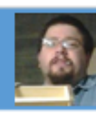

Advisor Mechanical Systems
James DiCorcia

INTERNATIONAL LIAISONS:


Int. Liaison Systems Engineering
Daniel Faber


Int. Liaison Commercial Collaboration
Joe Hatoum


Int. Liaison Electrtical Engineering
Isaac Desouza


Int. Liaison Attitude Control Systems / CAD
Matteo Borri

ALLIED BUSINESS RESOURCES:

General Counsel: Copilevitz & Canter, LLP
Intellectual Property Counsel: Tucker & Ellis, LLP
Accounting: May & Barnhard, PC
Banking: Capital Bank, Maryland
Insurance: Kurek Insurance Associates, Inc.
Internet Service Provider: Xisp.net


Launching in 2018
XISP
INC.
alpha
cubesat
Don’t wait for the future, help us build it!

Applicable Safety Requirements

1. The Alpha CubeSat (ACS) spacecraft must meet or exceed the International Space Station (ISS) safety requirements for pressurized cargo from the delivery to the applicable ground cargo processing facility (Kennedy Space Center or the Mid-Atlantic Regional Spaceport) until the Launch Service Provider (LSP) Trajectory Insertion Bus (TIB)passes outside the ISS Keep Out Sphere (KOS) with the ACS attached.

2. The ACS must meet or exceed the LSP TIB safety and interface requirements from the point of integration (TBD: ground or ISS depending on cargo vehicle accommodations) until the TIB executes the equivalent of a Planetary Services Deployment Mechanism release of the ACS spacecraft.

3. The following ISS Safety Requirements Documents are applicable:

- **SSP 50021 – ISS Safety Requirements Document**
- **SSP 50021 DCN 001**
- **SSP 50021 DCN 002**
- **SSP 30599 Revision E – Safety Review Process**
- **SSP 30559 Revision D – ISS Structural Design and Verification Requirements**
- **SSP 52005 Revision D – Payload Flight Equipment Requirements and Guidelines for Safety-Critical Structures**
- **SSP 41172 Revision U – Qualification and Acceptance Environmental Test Requirements**
- **SSP 30558 Revision C – Fracture Control Requirements for Space Station**

4. The ISS SSP 30599 Revision E – Safety Review Process Document begins with a Phase I Safety Review which typically occurs after the preliminary design is complete. In anticipation of the need to meet the requirements of the Phase I Safety Review after CubeQuest Challenge GT-2 the ACS Team has developed the following annotated abstract of the Phase I Safety Review process to document our readiness to comply with the applicable requirements.

5.1 PHASE I SAFETY REVIEW

The phase I safety review is the first safety meeting among the appropriate safety and engineering personnel representing NASA, IPs, contractors, and the ISS safety review panels in which safety of the ISS equipment and associated operations are addressed. The objective of the meeting is to identify all hazards and hazard causes inherent in the preliminary design, evaluate the means of eliminating, reducing, or controlling the risk, and establish a preliminary method for safety verification.

5.1.1 PHASE I DATA REQUIREMENTS

The following data is required for the phase I safety reviews:

A. GSE and Flight Hardware Ground Operations at KSC

> *1. Flight Element description based on subject mission.*
>
> *2. Descriptions of GSE and flight hardware subsystems that present a potential hazard during ground processing, and the ground operations involving these items. Schematics and block diagrams with safety features and inhibits identified shall be included. Design data for hazardous*

systems (pressure, lifting, etc.) shall be summarized in a matrix. Contact the GSRP Chair for sample formats.

3. Ground operations scenarios including post-flight ground operations at the primary, alternate, and contingency landing sites. The scenarios shall highlight unique requirements, such as continuous power through a T-0 umbilical.

4. Ground HRs and appropriate support data.

5. Ordnance data required by KHB 1700.7

6. Demonstration that the preliminary design is in compliance with design requirements of KHB 1700.7. The following are basic hazard groups applicable to ground operations: structural failure of support structures and handling equipment; collision during handling; inadvertent release of corrosive, toxic, flammable, or cryogenic fluids; loss of habitable/breathable atmosphere; inadvertent activation of ordnance devices; ignition of flammable atmosphere/material; electrical shock/burns; personnel exposure to excessive levels of ionizing or nonionizing radiation; use of hazardous/incompatible GSE materials; inadvertent deployment of appendages; working under suspended loads; and rupture of composite epoxy overwrap pressure vessels. SSP 30599 Revision E 5-2

7. Planned on-dock arrival date at KSC.

B. Flight System Design and Operations

1. An overview description of the design and flight operations of the hardware being addressed in the review. This includes descriptions of: hardware elements; flight and ground systems related to ISS on-orbit manned and unmanned operations; airborne support equipment; operational scenarios related to assembly, start-up sequences, and orbital operations; and LP, assembly, and stage configurations of the hardware. Briefly describe the hardware and operations in terms of significant characteristics and functions. Include figures or illustrations to show all major configurations and identify all hazardous systems and subsystems.

2. Detailed descriptions and schematics/block diagrams (at a PDR level of detail) for safety-critical systems and subsystems and their operations. In lieu of uniquely generated safety descriptive data, and with prior coordination with the SRP, references can be made to other ISS descriptive documentation made available to the SRP.

a. The schematics and block diagrams should be prepared with safety features, inhibits, etc., identified. Describe the major elements of the end item or segment with the information organized by technical disciplines (See below).

b. Describe the design, function, planned operation, and safety features of each system/subsystem.

c. The following list of technical disciplines may be used to organize the data: structures, materials, mechanical systems, pyrotechnics and ordnance systems, pressure systems, propulsion and propellant systems, avionics systems (including electrical power distribution, computer-controlled systems), command and control systems, optical and *laser systems, human factors, hazardous materials, thermal control systems, and interfaces and provided services.*

3. Flight HRs and appropriate support data (see paragraph 5.1.2).

4. A summary listing in the description section, of safety-critical services provided by other ISS segments or the Orbiter.

5.1.2 PHASE I HAZARD REPORTS

A phase I HR shall be prepared for each hazard identified as a result of the safety analysis on the preliminary design and operations. The focus shall be on cause description and controls. Instructions for completion of phase I HR forms are contained in Appendix D.

5.1.3 SUPPORT DATA - PHASE I HAZARD REPORTS (FLIGHT ONLY)

Critical procedures/processes, which require special monitored verification, shall be identified in preliminary fashion. Also, for those hazards controlled by "design for minimum risk," rather than failure tolerance requirements, a minimum set of support SSP 30599 Revision E 5-3 data, defined herein for phase I are required. (Appendix D contains the complete list of data elements for design for minimum risk hazards.) For COTS and non-complex hardware, ISS subsystem manager and SRP with appropriate discipline expert (EEE, material, battery, etc) will provide guidance to the appropriate level of detail required for HR generation. (Note 1: Reference to submitted and approved document by number and title is sufficient unless given specific request.)

A. Unpressurized Structures:

1. Preliminary plan for structural verification in accordance with SSP 30559, Structural Design and Verification Requirements, (including beryllium, glass [in accordance with SSP 30560, Glass, Window, and Ceramic Structural Design and Verification Requirements], and composite/bonded structure) (Note 1)

2. Fracture Control Plan in accordance with SSP 30558, Fracture Control Requirements for Space Station (Note 1)

B. Pressurized Systems:

1. Fracture Control Plan (Note 1)

2. Summary of design conditions for each pressurized system and certification approach

C. Pyrotechnic Devices:

1. Identification of pyrotechnic devices and functions performed

D. Ionizing Radiation:

1. Ionizing radiation data sheet for each source (JSC Form 44 Ionizing Radiation Source Data Sheet - Space Flight Hardware and Applications, See Appendix G)

E. Electrical Systems:

1. Top level wiring diagrams illustrating the approach to wire sizing and circuit protection

F. Components and Elements of Mechanisms in Critical Applications:

1. Mechanical Systems Verification Plan (MSVP) – Preliminary Version (Note 1). Include in the MSVP a summary of critical procedures and processes to meet safety requirements using either a) failure tolerant approach or b) Design For Minimum Risk (DFMR) approach that required compliance with JSC letter MA2- 00-057, Mechanical Systems Safety, September 28, 2000. A fault tolerant approach that combines a) and b) above will be accepted. A link to the MSWG website and the MA2-00-057 letter is available on the ISS SRP web page at http://srp-sma.jsc.nasa.gov/default.cfm.

5. The following Space Launch System (SLS) safety requirements while not specified as applicable are included as a reference:

Hazard Analysis Verification

Reference SLS-SPIE-RQMT–018 IDRD Sect 4.0 and App B VCRM

Submit analysis method of verification of safety hazard mitigations as defined in SLS-SPIE-RQMT–018 IDRD Sect 4.0 and App B VCRM

1-lists analysis w/plans of when performed;

3-all above & provides some initial analysis

5-all of the above plus some detailed analyses

Hazard Analysis Test/Demonstration

Reference SLS-SPIE-RQMT–018 IDRD Sect 4.0 and App B VCRM

Submit test or demonstration method of verification of safety hazard mitigations as defined in SLS-SPIE–RQMT–018 IDRD Sect 4.0 and App B VCRM

1 - lists tests w/plans for development;

3 - all above & plans for verification testing

5 - all above & draft test procedures available

Inspection

Reference SLS-SPIE-RQMT–018 IDRD Sect 4.0 and App B VCRM

N/A

Safety Data Package (SDP)

Reference: SLS-RQMT-216 SLSP EM-1 Safety Requirements for Secondary Payload Hardware & SLS-PLAN-217 EM-1 Secondary Payload Safety Review Process or equivalent for selected launch vehicle

Initial Safety Data Package with hazards identified

1 - completed Phase 0 submission material, but no material for Phase I review

3 - completed Phase 0 submission material, & draft SDP for Phase I with hazards identified

5 - all of the above, plus methods to close hazards

Schedule

Submit your development schedule, showing milestones relative to phased safety review milestones, demonstrating compliance with SLSPLAN-217 SLS Secondary Payload Safety Review Process, Sect. 4. Detail plan to GT3 w/milestone events to other GTs

1 - low confidence that SDP and payload development will be sufficiently mature for phased payload safety review;

3-adequate confidence that SDP and payload development will be mature as required for phased payload safety review milestones

5-excellent progress in SDP; excellent payload development progress relative to required phased safety review milestones