Center For Space Systems Mission Success Assurance: Requirement Verification and Spacecraft Integration and Test Program

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Abstract

This document provides the framework of mission success assurance for space-flight projects developed within the Center for Space Systems (CSS). These guidelines, protocols, and procedures were defined during the development of the first project within the laboratory: the Rapid Reconnaissance and Response (R^3) small satellite mission. The document first details the guidelines and systems put in place for requirement development and verification, the integration and test program, and the hardware development best practices. It then enters into a discussion of how this framework was implemented for R^3, including specific details and examples of the tests planned and executed. The Requirement Verification Matrix (RVM) developed for R^3 is provided as an Appendix in both electronic and hard copy as a model. Templates of all documentation developed to support CSS flight projects are also provided as an Appendix in both electronic and digital versions. All documentation of the R^3 verification and testing program (hardware inspections, test planning forms, test completion records, etc.) is provided in an electronic appendix that is provided alongside this document.

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Nomenclature

ADC = analog-to-digital converter
ADCS = attitude determination and control system
AE = aerospace engineering
AFRL = Air Force Research Laboratory
ASU = Arizona State University
CDH = command and data handling
CDR = Critical Design Review
COTS = commercial off the shelf
CSS = Center for Space Systems
CDH = command and data handling
DAQ = data acquisition product
DIME = Dosimetry Interconnection and Miniaturization Experiment
DMM = digital multimeter
DoD = depth of discharge
DSP = digital signal processing
EDU = engineering design unit
EGSE = electrical ground support equipment
EMC = electromagnetic compatibility
EMI = electromagnetic interference
EP = electric propulsion
EPS = electrical power system
ESD = electrostatic discharge
ESM = Engineering Science and Mechanics
FBSC = full battery state of charge
FCR = Flight Competition Review
FHL = Flight Hardware Lab
FPGA = field programmable gate array
FSW = flight software
GS = ground station
GPS = global positioning system
GTRI = Georgia Tech Research Institute
I&T = integration and testing
LBSC = low battery state of charge
LVDS = low voltage differential signaling
MGSE = mechanical ground support equipment
MK = Montgomery Knight
MLI = multi-layer insulation
MOC = Mission Operations Center
NOAA = National Oceanic and Atmospheric Administration
PCB = printed circuit board
PM = project manager
PMAD = power management and distribution
PSE = project systems engineer
R^3 = Rapid Reconnaissance and Response
RVM = Requirement Verification Matrix
SEL = single event lockup
SEU = single event upset
TCS = thermal control system
UHF = ultra high frequency (radio band)

I. Introduction

THE backbone of a successful engineering project is its ability to thoroughly address two questions: "Are you building the right thing?" and "Are you building it right?". The first question ensures the product fulfills its intended purpose, and the second ensures it meets specifications for functionality. Encompassed in their combination is a fully convincing assurance that the product was completely and adequately engineered for its intended purpose. When applied to a spacecraft development program, the answer to these two questions provides a dependable assurance of mission success. Therefore, they should be the driving forces behind initial concept development, should be revisited regularly throughout design and fabrication, and should guide the entire program for testing and launch preparation.

An expanded description of the above three stages is in order. The first metric for guiding assurance of mission success is to develop sound, specific, and thorough requirements before any design concept has been initiated. These requirements will serve as the framework and decision parameters for the spacecraft's architecture trade space. A design must then been generated for a spacecraft that (a) fits within the framework of requirements developed, and (b) fully incorporates all elements of the framework of requirements developed. Throughout this design process, all analyses (structural/vibrational analysis, thermal analysis, etc.) and component selection must refer back to the requirements document as their metric for necessary performance. Once a design is in its mature stages and components have begun to be fabricated and procured, the project enters a stage of continuous requirement verification within an integration and test program. Every component or software module must be scrutinized under the microscope of all related mission requirements, and a proper system to execute this verification is essential to the mission success assurance process.

This paper explains the system implemented by the Center for Space Systems to address these two questions for the Rapid Reconnaissance and Response flight project. It begins with a discussion of requirement development for the mission, including level definition, traceability, and flow-down. It then describes the methodology employed by the Center for Space Systems for requirement verification, status notation, and tracking of associated documentation. The discussion then moves into an overview of the integration and test program, including purpose, rationale, and a detailed walk-though of the process from initial procurement or fabrication through final integrated software mission simulations. The appendices provide a detailed record of the requirement verification method for the R^3 mission and a complete archive of all referenced and associated documentation.

II. Requirement Development for Assurance of Mission Success

Developing clear, thorough, traceable requirements is the first key element of a space program founded in sound engineering practices. This process begins with a concise, explicit mission statement that describes the objective(s) to be accomplished by the mission, and flows into criteria (needs and conditions) that must be met by the spacecraft system and all encompassed subsystems in order to complete all aspects of this mission statement. Only by defining the problem in this way may one begin to define the possible solutions.

A. Section, Category, and Level Definition

The type of tool used for requirement development (and later, verification) during the R^3 mission is commonly known as a Requirement Verification Matrix. The RVM described throughout this document, however, was crafted as a custom tool specifically for the CSS, and is provided for reference in Appendix A. The R^3 RVM has many sections, denoted by a shaded section breaker and title. These sections fall within both categories and levels, the differentiation made between which is as follows. Categories are defined as requirement groupings or genres, each of which may have many sections and levels. "Mission," "System," and "Subsystem" are the three requirement categories. The color or the shaded section heading denotes which category the section falls under. Levels, on the other hand, define the source and flow-down of the requirement, beginning with the Mission Statement as the overarching purpose and flowing into subsequently more detailed constraints on what the system and its elements must do to accomplish this purpose. Two inherent characteristics of this breakdown system are important to note:

1) A category can have multiple sections. For example, two sections in the R^3 RVM that fall under the System category are "Satellite System" requirements (requirements applicable to the satellite as a whole such as survivability requirements, etc), and "Electromagnetic Compatibility" requirements (which also apply to the satellite as a whole across all subsystems, as opposed to being a subsystem of it).

2) Multiple levels can also exist within a single section, thus a single category. For example, the thermal control system might have Subsystem Requirement B in order to meet the satellite's System Requirement A, and then Requirement B results in Requirements C, D, and E that the thermal control system must also meet in order to accomplish Requirement B. While Requirement B and Requirements C, D, and E are within the same category (subsystem), they are at different levels (B flows down to C, D, and E; or in other words, B is a higher level requirement than C, D, and E).

Table I breaks down of the sections in the RVM for the R^3 mission. Note that categories are denoted by highlight color, levels are denoted by indentation, and that it is common for a section to contain multiple levels, which cannot demarked by the table.

Table I. Breakdown of R^3 RVM Sections

Mission Statement	
MO	Mission Objectives
MSC	Mission Success Criteria
MD	Mission Design
SAT	Satellite System
LVI	Launch Vehicle Interface
EMC	Electromagnetic Compatibility
INS	Science Instruments
ADC	Attitude and Orbit Determination and Control Subsystem
CDH	Command and Data Handling Subsystem
COM	Communications Subsystem
EPS	Electrical Power Subsystem
STR	Structure
TCS	Thermal Control Subsystem
FSW	Flight Software
ALG	Thermal Algorithms
MOS	Mission Operations System
MC	Mission Control
TRAC	Tracking Station
GDS	Ground Data Systems
SMP	Science Mission Planning
DA	Data Analysis
TFAC	Test Facilities
EGSE	Electrical Ground Support Equipment
MGSE	Mechanical Ground Support Equipment

B. Requirement Development, Traceability, and Flowdown

The purpose of the hierarchy of requirements is to break down the mission objectives into smaller, more specific, more assessable tasks to direct the architecture definition and design specifications that will meet them. As such, the two most important aspects of a useful RVM are (1) the traceability of each requirement and (2) a clear and logical flowdown of requirements to and from each level. Any requirement not directly needed in order to accomplish a higher-level requirement is superfluous, and should not be included.

In addition to being traceable to a higher-level requirement, each requirement should be quantifiable to ensure the system has been developed as specified. The various forms of this verification will be discussed in a later section, but knowledge of this aspect guides proper requirement development. Descriptive numbers based in analyses should be used wherever applicable such that requirements are exact, definitive, and testable. Qualitative words such as "sufficient," "capable," and "properly" should be defined by values and quantities or avoided all-together.

Additionally, as a design matures into latter stages and subsystem designs are fleshed out, more detailed requirements will flow not only from mission success criteria and external constraining documents like, in the case of R^3, the binding document for the competition (the Nanosat User's Guide), but also from the needs of various components imposed on the system. These cross-subsystem requirements result in the cyclical nature of requirement development, in which components are selected in one subsystem to meet the requirements of a different subsystem, and those components drive the requirements of other subsystems, potentially including the originally driving one. This cycle is hardware-induced and is thus inevitable, but can be minimized by clear and thorough initial requirement development that considers the complete flow-down and cross-flow of all requirements.

Finally, a good practice after developing all requirements is to scrutinize the flow-down tree of each mission success criteria and ensure it is fully populated. The best way to perform this verification is a rearrangement and visual tree representation of requirements, and can be performed most efficiently in a commercial diagramming program such as Microsoft Visio. Both flow-up and flow-down should be examined: flow-up determines whether each requirement has a justifiable purpose for existence that is traceable directly to a mission success criteria (or external program constraint), and flow-down determines whether the requirements fully encompass all necessary tasks of all involved components to fully achieve each mission success criterion.

III. Requirement Verification Methodology

Requirement verification is the process of performing analyses or tests to fully guarantee the system designed and built will perform all intermediate steps necessary to ultimately achieve mission success. It serves as the bedrock of mission assurance and the guiding force behind a flight project's integration and testing program. The requirement verification tool described in this document was specifically developed for the R^3 mission and its discussion thus relies heavily on the purposes for and objectives of the R^3 mission and satellite. This tool was developed in Microsoft Excel, but the requirement development and verification process can be better automated with more advanced tracking and scheduling software such as the Doors program developed by IBM. The following sections detail the notation and sections of the RVM such that it can be properly interpreted and replicated.

A. Verification Method

There are three methods by which a requirement can be verified: inspection, analysis, and physical testing. Verification by inspection is observation using one or more of the five senses, simple physical manipulation, and mechanical and electrical gauging and measurement to verify that the item conforms to its specified requirements. For instance, a requirement about where port connectors are located on the external housing of the CDH unit is verified by inspection, or visible observation of the port locations. Additionally, any requirement that is stated achievable by a procured part in the provided vendor documentation and is beyond the scope of CSS to test, will be considered verified by inspection as well: inspection of the vendor documentation.

Verification by analysis involves the use of established technical or mathematical models, simulations, and/or algorithms to provide evidence that the requirement is met. This definition can mean two things. The first is that the requirement cannot be tested before launch, and/or does not need to be tested because full proof of compliance can be provided with computational simulation or analysis. The second is actually that multiple tests need to be performed and then further analysis must be complete before the requirement is thoroughly verified. For this case, the result of each test does not give a pass or fail indication, but rather all of the results must be taken in concert and used to perform additional analysis before the requirement is proven to be satisfied.

Finally, verification by testing is the application of scientific principles and procedures to determine the properties or functional capabilities of a component. This method generally requires specialized test equipment, configuration, data, and procedure in order to verify that requirement is satisfied. Testing is the most common method of requirement verification, generally being viewed as the default method unless it is inapplicable or infeasible. The details of the test process are provided in Section IV and V.

B. Status and Document Tracking

The RVM is a living document that is continually updated to monitor the status of the requirement verification program. Each requirement must be demarcated as standing at one of 5 statuses: In Progress, Under Review, Not Met, Designed, or Verified. "In Progress" means that the requirement has not yet been fully addressed by the satellite design. In other words, that aspect of the satellite has not yet been designed or those components have not yet been selected. "Under Review" means the current design might have trouble meeting this requirement. There is cause for concern (but not proof of non-compliance), and investigation (analysis, testing, etc) is currently being done to provide more information. "Not Met" means that the current design cannot meet this requirement, and it is thus a problem area for the program. These red-flag requirements are considered a high priority in development, and must be actively addressed by design changes. An example of this process specific to the R^3 satellite was that a re-analysis of the power budget revealed that the current structural bus volume did not allow enough surface area for solar cells to meet the power requirements of the system. Therefore, design meetings were held to reconfigure the structure and increase the size of the external frame in order to provide the proper surface area for solar cell mounting. "Designed" denotes that the requirement is met by the current design. It has not yet been verified, but preliminary design and analysis predict that the verification process will confirm compliance. Finally, "Verified" means the documentation has been cataloged to evidence the satellite is in full compliance with this requirement. The RVM provides a system for tracking this documentation that is described below.

During preliminary design and requirement development, most requirements will likely be marked as "In Progress." Upon entering the integration and test program, most requirements should have transitioned into the "Designed" status. Table II below summarizes these status categories and their associated color code.

Table II. Status Notation and Color Scheme

In Progress	This requirement has not yet been fully addressed
Under Review	The current design might have trouble meeting this requirement; investigation is being done
Not Met	The current design cannot meet this requirement; this is a problem area
Designed	This requirement is met by the current design (but it has not yet been verified)
Verified	This requirement has been verified; documentation exists

The RVM provides the framework to be the single, central document to guide and trace the mission assurance of a spacecraft development program. As a major element of this framework, the RVM references all relevant documents relating to the test planning and requirement verification. When a requirement is verified by inspection or analysis, a verification form documenting that inspection or analysis is traced by document number beside the requirement definition. This form template is provided in Appendix B. When a requirement is verified by testing, both the test planning documentation and test completion (verification) documentation are traced by document number beside the requirement definition. These forms are also provided in Appendix B.

IV. Integration and Testing Program

The goal of the R^3 I&T phase is three-fold: to perform a series of performance tests on all components to verify conformity to all R^3 requirements and expected equipment performance, to successfully integrate all of these components with each other, and to perform a series of integrated functional tests to ensure collaborative operability of all subsystems in the final integrated satellite configuration. The purpose of implementing a well planned I&T program is to ensure that the R^3 satellite is space-ready and qualified for full mission success.

When visiting the Georgia Tech Space Systems Design Lab and CSS, Bill Nye said that one test is worth 1,000 expert opinions. This section provides procedural and technical descriptions of each phase: initial component inspection, component-level testing, integration, system-level testing, and mission simulation (day-in-the-life) testing. These phases guide a full examination of the satellite with respect to three specific areas: (1) component performance confirmation, (2) requirement verification, and (3) interface testing and integration confirmation to ensure functionality of the overall system after data, power, structural, and software integration.

A. Initial Inspection

When a component is procured from an external vendor, data (mass, dimensions, quantity, supplier, etc.) and associated records (certificate of compliance, order confirmation, test documentation, etc.) about the component must be cataloged using a Receiving Inspection Form, which can be found in Appendix B. A thorough inspection is then performed and any missing elements or damage to the component or packaging integrity are noted. Finally, a series of authorization signatures are required before any further handling of the component is permitted.

When a component is fabricated in house, similar documentation must be completed prior to testing. The completed part must be inspected and approved through a Workmanship Inspection Form, which details basic data about the part as well as any imperfections and special handling or storage requirements. This form can also be found in Appendix B.

B. Component Testing

Each subsystem should be put through a series of performance tests designed to verify both that the subsystems meet all project requirements and, to the extent possible from the ground, that the subsystem performs to the full degree and with the full ranges expected from design. These components will then be integrated with each other in a specific sequence to test the continued functionality of the equipment while in the integrated state. The main architecture of the component testing phase remains constant for all components, while each component will have unique functional tests to be performed within this architecture. Figure 1 and Figure 2 display the flow of this process for both procured and internally fabricated components.

For components fully manufactured external to CSS, team members must perform a thorough receiving inspection and completes the Receiving Inspection Form before beginning any functional testing. If the component is to be fabricated or modified internally, all fabrication and assembly must adhere to guidelines defined in the quality assurance plan discussed in a later section. This fabrication is followed by a thorough Workmanship Inspection as detailed in the same section.

The next step for all components is a test of all electrical interfaces. Once all interfaces have been checked, subsystem leads use external data and power sources to systematically test the performance capabilities of each component to ensure conformity with all expected performance levels and ranges. These performance tests planned for R^3 components are provided electronically in Appendix C. Once each component has successfully passed through all performance tests, it is ready to be delivered for integration. Figure 1 and Figure 2 provide a graphical summary of this process for both internally and externally fabricated components.

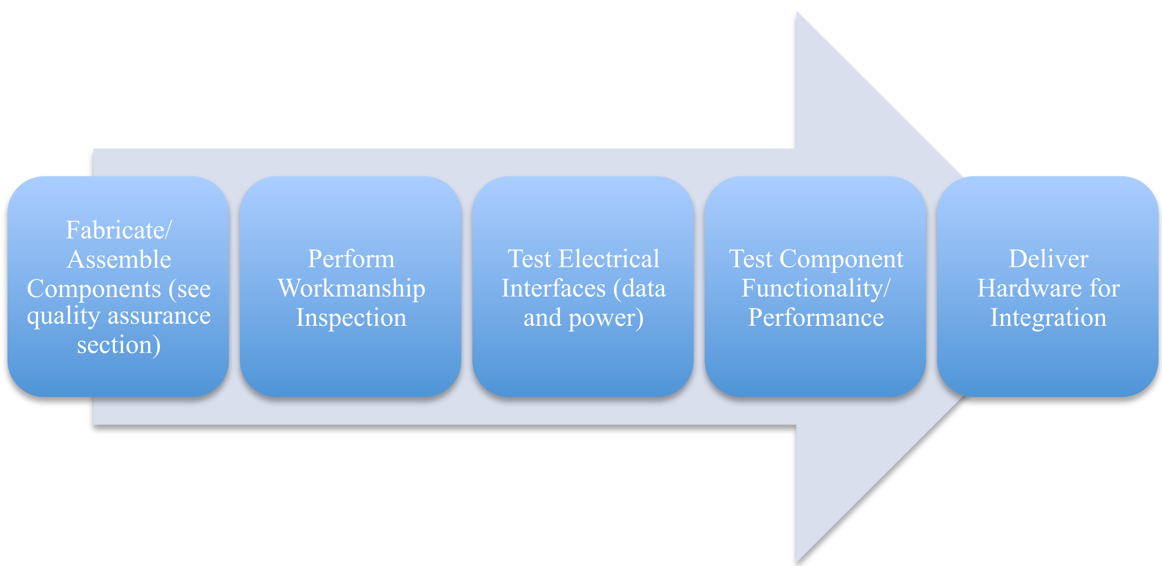

Figure 1: Internally Fabricated Component Test Procedure

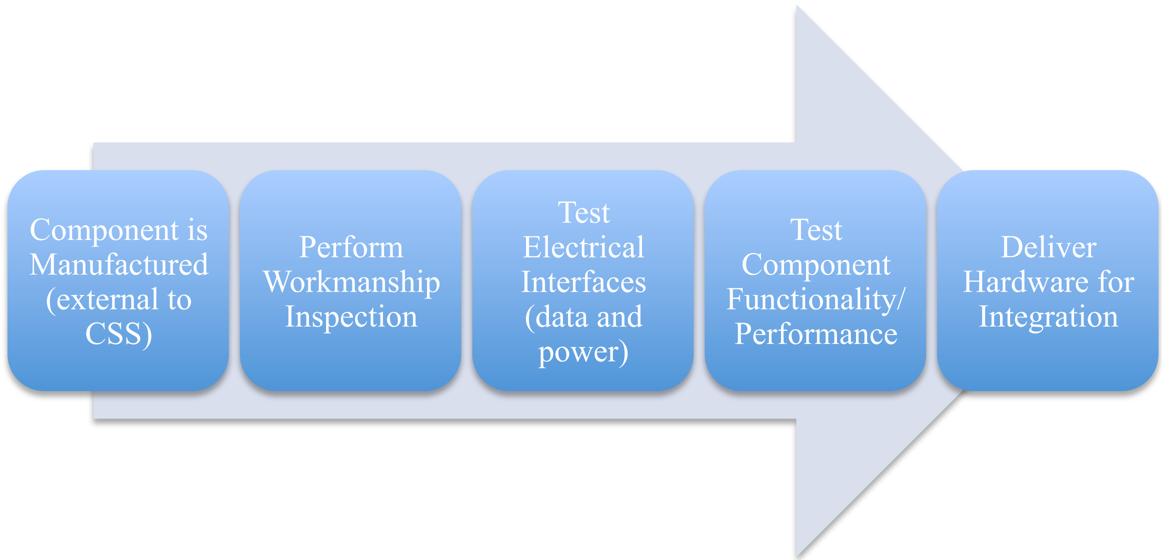

Figure 2: Externally Fabricated (Procured) Component Test Procedure

Environmental tests should be performed largely at the system level. However, the environmental testing must also be performed at the component level before the component is integrated with the spacecraft for certain circumstances. The specific categories of component environmental testing that should occur for the flight components can be found in this section, while examples of components on the R^3 satellite affected by these guidelines is in Section VI.

The first environmental test involves verifying the operational thermal limits of specific components by thermal cycling. This test will only need to occur on the 1-2 components with the tightest hot and cold limits. The intent is not to test the survival of these components at their extreme temperature limits, but instead to explore their performance within their operational thermal limits. If the components cannot perform to their full expected range of functionality at these temperature limits, actions may need to be taken with regard to component selection, mission planning, and/or final thermal design.

In addition to thermal cycling, thermal vacuum tests must be performed on certain components to ensure that outgassing will not affect the performance of the instruments. If the spacecraft contains components that are not traditionally flown in space, or components that have materials not approved by the NASA Outgassing Data for Selecting Spacecraft Materials document, this test must be performed to ensure proper functionality in an in-space environment. Thermal vacuum testing should also be performed on all internally fabricated components, but for a different purpose: this test is the best workmanship examination to analyze the quality of solder joints and other fabricated elements. A sufficient thermal chamber should be capable of bringing the pressure down to around 7 millibars and the temperature down to 0°C. The component should be interfaced with the computer and begin normal operation before the vacuum chamber is initialized. When test pressure and temperature is reached, operation of the component should continue for a representative period of time. Then the chamber should be returned to normal atmospheric conditions, after which the components should continue in normal operation mode for a brief period of time before they are removed from the chamber and evaluated.

The third environmental test is a shock test, which should be performed on any satellite components that contain any form of glass. The entire component (in the same configuration in which it will be mounted to the satellite) must be shock tested, rather than solely the element that contains the glass.

The fourth recommended component-level environmental test is a vibration test (both sinusoidal and random vibration) to verify that the structural integrity of electrical connections is maintained under a vibration environment similar to that of the expected launch environment. This should be performed on solar arrays and all other components with significant wire harnessing or delicate connections. During this test, the component should be monitored for any harmonic motion or physical damage occurring on the assembly. After the vibration test, basic operational tests must be performed to compare baseline open circuit voltages and short circuit current to ensure they do not differ from pre-test values by more than 5%.

C. Integration

In general, this document will refer to integration as three major tasks for each component: mechanical integration (mounting the components securely to the satellite bus or internal framework), EPS integration (interfacing with the satellite PMAD system for the provision and regulation of power to the component), and C&DH integration (interfacing with the flight computer for all necessary data flow to and from the component). More specifically, within these three tasks the system integration phase has four primary objectives. The first is to interface all inputs and outputs for each component to the respective data or power ports so that satellite components are fully integrated with each other through all necessary connections. The second is to verify each of these electrical interfaces, and confirm that power and data can flow to and from each component. The third is to verify the functionality of each component in this integrated orientation by external commands. Note that the performance of the component was thoroughly examined at the component level and will not be re-tested during integration;

simple functionality confirmation via integrated interfaces will be sufficient. The fourth and final objective is to integrate the flight software (FSW) as well, and retest the component functionality when commanded by its respective FSW module. These fully integrated simulation-type tests will be discussed further in Sections IV-D and IV-E. This section, in contrast, will describe the initial procedures and component installations necessary to prepare the satellite for the integration phase, and then discuss the schedule and procedures for subsystem and payload integration.

Before the flight hardware is integrated, it is sometimes advantageous to first develop a flat-sat, in which EDUs arranged on a flat surface and electrically integrated for testing and development before flight units are integrated to the flight structure. In this step, details like wire harnessing are more relaxed for the sake of a functional development and test bed. A flat-sat allows early procedure development for I&T (functional interface tests and other system-level tests) and flight operations. Preliminary hardware and software requirements can also be verified on the EDUs in the flat-sat configuration. Finally, the flat-sat allows for system-level testing of flight software and a more simplistic environment for troubleshooting during flight hardware integration and testing, and even during post-launch flight operations support.

This model was adopted by the R^3 I&T program. A flat-sat board was developed for a preliminary hardware demonstration at the Proto-qualification Review, allowing for work to be done with the electrical interfacing and integration before the satellite structure was complete and assembled. Therefore, as a first step of R^3 integration, the flat-sat C&DH unit, PMAD system, and wire harness were set up with connections for those pieces of satellite hardware to be demonstrated at this review. Note that space-rated components for flight will be acquired only after full integration has been successfully completed using ground testing (EDU) components. Since the development of these two systems is dependent upon the needs of each component during interfacing and integration, they are initially integrated on the flat-sat in a partially complete configuration and their development occurs parallel to the integration schedule of each component. In other words, the initial setup of these three elements should be equipped to interface with the first subsystem components to be integrated according to the schedule. As other subsystems are prepared for integration, the PMAD system and flight computer are developed to accommodate the needs for each component.

For the flight system, the first three elements to be assembled should be the satellite bus structure, the PMAD system, and the flight computer. The satellite bus structure should be laid in an open configuration as the first step of the integration phase, so that as components are electrically integrated to the power and data lines, they can also be mechanically integrated to the satellite structure. In normal operations, the solar cells will collect light from the sun and convert it to electrical power, which will pass through the PMAD system for delivery to each load, including charging of the batteries. During this initial setup, however, an external power source (EGSE) will provide power directly to the PMAD system for distribution.

It is crucial that the PMAD system is tested meticulously, as improper regulation of power could irreparably damage a component. Once the PMAD system is installed on the satellite bus and the power supply is properly connected, the polarity of each connection should be checked with a multimeter. Then, the source and sink of lines in and out will be monitored and evaluated by systematically turning on and off each switch in the power distribution system. As each switch is flipped, it will be verified that the end of the correctly corresponding wire (and only the correctly corresponding wire) receives power regulated to the proper voltage.

When the power distribution system has been thoroughly examined, the command and data handling (C&DH) subsystem should be the first element to be integrated with it. Both the processor board and the interface board should be integrated with the power distribution system interfaces. All data outputs should initially interface with an external source (ie: a computer in the FHL), and be progressively changed over to interface with flight hardware as each element is integrated. An interface test should then be performed to ensure power and data flow is occurring properly between the power distribution system, the C&DH unit, and the external data source/sink. All resistances should be checked, and all data channels should be verified.

Once the power distribution system and C&DH subsystem are integrated in the flat-sat configuration, subsystem integration occurs one component at a time. For the R^3 satellite, this process began with the components demonstrated at the Proto-qualification Review. The R^3 I&T program implemented a phased approach to subsystem integration, such that the component integration schedule is based on the procurement and functionality testing schedule of each component. This method allows for thorough testing to be done on each component individually based on the timeline necessary for full functionality to be confirmed on that component.

This piecewise integration approach also allows parallel development and provides the advantageous opportunity to perform tests at each integration point to determine connection and operability of each component as it is integrated individually. This integration schedule was developed from the component procurement and testing plans, and is reflected by the integration sequence summary in Section VI-B. Note that for flat-sat integration, all elements were complete to the extent of the needs of the flat-sat components, and/or for the tasks of the demonstration. The final R^3 satellite integration involves the added element of mechanical integration with the structure. The structure and C&DH unit in particular should undergo continued development to meet the needs of all the flight components as they are integrated. The full integration and testing schedule can also be found in Section VI-B.

Special measures must be taken when integrating the payload to ensure capable performance to complete the science mission early on in the satellite development schedule. External displays and testing support equipment should be utilized to verify expected science output. For the R^3 mission, pre-integration and partial-integration payload capability were demonstrated at the Proto-qualification Review in the flat-sat configuration as a milestone for the science mission development. However, for final satellite integration, the payload should be integrated last, and only after integrated system functionality of the satellite has been demonstrated to protect the instruments from satellite system anomalies that could cause damage or failure.

D. System and Software Testing

When the satellite has been completely integrated, a series of integrated tests and orbit simulations should be performed to test the cohesiveness of interaction between all hardware components as well as the capability of the flight software to command and control the satellite throughout the full range of scenarios expected during the primary mission. These tests of the fully integrated satellite have been broken down into three major simulation sequences. They begin with flight software-hardware interaction tests for various subsystems. Once flight software seems to be properly commanding a wide range of expected daily on-orbit functions of the satellite subsystems, a series of similar tests will be conducted with the payload instruments. Finally, a series of full orbit simulations will be conducted to ensure the satellite operates competently in a full day-in-the-life scenario.

The first sequence will encompass all satellite subsystems, excluding the payload suite. This simulation is the first transition from action based on external commands to action based on commands written into the flight software and stored in the satellite flight computer. All modules of the flight software will be tested with hardware-in-the-loop and final anomalies will be resolved. It is important that the payload is not included in this initial test sequence in the event that software is improperly implemented and the satellite experiences any irregularity that could potentially harm the payload. This initial simulation explores several complex end-to-end tasks that should occur during the mission; those planned for the R^3 spacecraft are discussed in Section VI-D.

When the flight software is integrated with the payload instruments, the transition from externally commanded hardware actions to flight software commanded hardware actions must be confirmed with hardware-in-the-loop software tests that simulate the full range of performance tasks required of each instrument. This is done last to protect the integrity of the most essential elements of the mission: the science payload. The tasks that will be performed throughout the R^3 mission and will thus be performed during this sequence of integrated testing are also described in Section VI-D.

The full orbit simulations will typically begin with a safe mode orbit simulation, the simplest of the orbital scenarios. At this stage, the test will assume the satellite begins in a safe-mode state. Once cohesive functional capability is established with all necessary tasks within safe mode, the team will move on to an orbit simulation of normal operations (the science performance/acquisition stage of the mission), which is slightly more complex. Finally, the last stage of the orbit simulations is the most complicated of all, in which the satellite will step through all phases of the mission life cycle, from startup and checkout operations to end of mission safing. This full-satellite software verification and validation will be described specific to the R^3 satellite in more detail in Section VI-E.

V. Standards and Protocols

This section provides an overview of policies outlined for the R^3 satellite development, to be followed by future CSS flight projects, as relates to electromagnetic compatibility and quality assurance. The instructions within these two categories comprise a suite of best practices to be implemented by the CSS to ensure sound engineering is applied throughout design and development of all flight projects.

A. Electromagnetic Compatibility

All spacecraft developed within the CSS must conform to standards for Electromagnetic Compatibility (EMC), and will require testing to assure this compliance. Discussed in this section are the analyses and processes required to fully substantiate that no component on the spacecraft will harmfully interfere with any other component to the degree that the mission or spacecraft operation is compromised in any way. These policies were first developed for the R^3 spacecraft's compliance with UNP constraints, so those constraints are summarized before the CSS policies are defined.

The UNP User's Guide, in "Section 6.6.2: Payload Design Requirements," states:

> The nanosat shall be designed for electromagnetic compatibility (EMC) and for mitigation of electromagnetic interference (EMI), specifically susceptibility to launch vehicle and range radiation environments. Simple EMI mitigation techniques (design rules-of-thumb) will be communicated to the universities in an expert area telecon during the course of the Flight Competition. Universities are not required to generate a detailed EMI/EMC analysis, beyond the tabular analysis required as per Section 8.6 of this document. Universities are also not required to perform any EMC testing beyond the self-compatibility testing required as per Section 8.6. Refer to Section 6.7 for further EMI/EMC-related design guidance.

"Section 8.6: Payload Analysis and Test Requirements" governs the basic analysis and testing required by UNP:

> All universities participating in the Flight Competition shall perform an electrical self-compatibility analysis and appropriate testing. Analysis outputs shall include maximum and minimum operating voltages, currents and frequencies for all nanosat components and subsystems, including wire harnesses and exposed field emitters such as communications antennas, propulsion system components, science experiments, etc. If any components or subsystems have known radiated susceptibility levels, those levels shall be included in the analysis.
>
> Self-compatibility analysis shall be validated through electrical performance testing of the nanosat at the subsystem and system levels, to ensure that nanosat components and subsystems are compatible when the nanosat is operated in flight configuration.
>
> Results of the analyses and testing will feed into the tailoring of EMI/EMC test requirements for the flight nanosat. Prior to delivery to AFRL, the flight nanosat shall be tested as follows:
>
> - Verify compliance with the bonding and grounding requirements of Section 6.6.3.
> - Verify that electrical system safety features operate properly.
>
> Integrated system-level EMI/EMC testing will take place at AFRL in conjunction with other environmental testing. Test levels will conform to MIL-STD-461E, at the discretion of AFRL and/or the launch provider. At minimum, testing will consist of the radiated emissions and susceptibility portions of MIL-STD-461E and will be tailored to cover worst-case (noisiest and most susceptible) nanosat operational conditions, including possible failure mode conditions if applicable. Additional testing may be required, depending on the design of the nanosat and the launch vehicle interface.

To meet these expectations, certain US Air Force, Department of Defense, and NASA requirements documents should also be used to ensure compatibility during testing. These requirements documents include MIL-STD-461E (Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment), MIL-STD-464A (Electromagnetic Environmental Effects Requirements for Systems) for reference only, and MIL-STD-1541A (Electromagnetic Compatibility Requirements for Space Systems). From these sources, a list of best practices has been compiled as described below as applies to EMC design, analysis, and testing.

The first action item in the EMI mitigation plan, as per the aforementioned UNP requirements, is that all CSS flight project teams shall keep track in tabular form of operating voltage, current draw, and operating frequency of all components, as well as any known frequency susceptibilities of specific components. Specifically, the EPS and telecommunication subsystem leads are responsible for creating an initial table for all harnessing wires and system components, and updating it throughout the design and testing of the R^3 satellite. The table includes: Operating

Voltages (Average, Maximum, Minimum), Current Draw (Average, Maximum, Peak, Minimum), and Frequency (Average, Maximum, Minimum). Individual component electrical testing shall then be utilized to update the table for more precision. A template for this table can be found in Appendix B.

The second action item in the EMI mitigation plan for CSS projects is to align the hardware design and development with the following strategy of best practices. The most common noise problem encountered in large-scale electronic systems stems from a lack of good grounding practice. Therefore, CSS spacecraft design and development will reflect some basic practices to avoid grounding problems in circuits. First, if several points are used for ground connections, differences in potential between the points can cause ground loops, which will cause errors in voltage readings. A common sign that a ground loop exists, or that a ground is missing, is the presence of induced power line noise in the circuit, so this will be checked throughout board fabrication, wiring, and integration. To ensure that no ground loops are created, the practice of using a "single point ground" will be implemented with all circuitry. While it is not practical to send all connective wiring to a single point ground, the entire bus structure itself can be used as a ground bus. The structure, with low resistance, near proximity to all components and wiring harness locations, and capability to carry the maximum sum total of the load current back to the power supply, will be the best way to practically implement the necessity for a single point ground. Note that as a general practice, analog and digital grounds should be kept separated and connected together only at one single point. A design for the implementation of this practice should be developed prior to integration of any analog components/circuitry.

Basic component-level EMC practices will be implemented when designing any printed circuit boards (PCBs) for in-house fabrication. In board layout, traces should be kept away from high-frequency devices, such as clocks. When selecting component packages, devices with a ground reference in the center of the device are preferable to reduce the ground inductance, and surface mount devices are preferred to through-hole packages for the same reason. Good shielding is recommended for all connectors.

Certain protocols should also be observed and implemented during wire harness design and formation. First, all data lines shall be harnessed separately from power lines to reduce interference between the two. Additionally, within the data wire harnesses and power wire harnesses, pigtail wiring harnesses should be avoided, as a pigtail would undesirably propagate RF signal.

Finally, to ensure full satellite functionality for the entire mission duration, particularly susceptible components such as the flight computer will be encased in EMI shielding housings.

The third action item should be taken within the phase of component level testing. The main test to be performed on each component is a confirmation of successful single point grounding to the structure. When grounding components, each connection between a component and the structure (or between two structural plates) should be tested, and should demonstrate no more than 25 mΩ of electrical resistance. Additionally, there are four standard tests that should be performed at the component level: conducted emissions, conducted susceptibility, radiated emissions, and radiated susceptibility. For the R^3 satellite, as stated in the Users Guide (quoted above), these tests will be conducted at AFRL following delivery of the satellite. For CSS missions not affiliated with UNP, either capabilities to perform these tests in-house should be developed or facility partnerships should be established.

Finally, the fourth action item is to conduct comprehensive testing efforts at the system level. The entire satellite system should be observed and tested to ensure all components are electromagnetically compatible with each other when operating collectively. An EMI profile can be created for this purpose using the anechoic chamber in the testing facilities at GTRI Smyrna.

B. Quality Assurance

To envelope the hardware handling, integrating, and testing process and guarantee quality throughout, an ordered set of documentation has been developed as a guiding force for quality assurance. As discussed in the Integration and Testing Program Section, the first set of documentation and procedures surrounds the initial fabrication or procurement of a component. The subsequent set of procedures involve test planning and execution documentation. While the elements of this process relating to quality assurance are discussed in this section, the documentation policies have already been detailed in the Integration and Testing Section, and will not be reiterated here.

For any component to be fabricated in house, the following protocols should be executed. Prior to fabrication, a fabrication plan must be created and approved (by the PM and PSE) which specifies environment and support equipment required, parts used in component assembly, and a detailed assembly procedure and instructions. Additionally, any personnel involved in fabrication of flight hardware must undergo fabrication training using practice boards in order to rehearse the techniques necessary to properly fabricate the part. Once the component fabrication process has been approved and the team members involved have been sufficiently trained, hardware fabrication can commence. A partnered fabrication system will be instated (at least two people working together at all times to ensure errors are minimized and all steps are followed precisely and thoroughly. Upon completion, the part must be approved through a Workmanship Inspection Form before any regular component testing or integration can commence. Likewise, when a component is procured from an external vendor, a Receiving Inspection Form must be completed in its entirety prior to any testing or handling of the component.

Inventory Management is also a key area of quality assurance, and one important component of Inventory Management is the maintenance of a comprehensive material's list. The CSS shall develop and maintain a Materials List for the protoflight hardware for all flight projects that includes both fabricated and vendor-supplied components. The list shall include metallic materials, non-metallic materials (epoxies, tapes, adhesives, plastic, rubber, composite, glass, lubricants, etc.), and coatings (anodize, plating, iridite, conformal coating, etc.). Payloads are exempt from listing conformal-coated electronics, small mass-produced electronic components, or components on the NASA GSFC Preferred Parts List. As components are procured, this system-wide materials list is maintained and updated. A template of the document developed for this purpose is shown in the Appendix. The intent of this materials list is to ensure compliance with all program material restrictions relating to outgassing, corrosion resistance, and flammability resistance as listed in the UNP User's Guide. Note that while these guidelines were developed for the R3 satellite based on its involvement in UNP, they serve as valuable principles for all CSS flight projects and will be applied as such. Specifically:

- Materials with high resistance to stress corrosion cracking (SCC) shall be used wherever possible. A list of such materials is provided in MSFC-STD-3029.
- Use of non-metallic material shall be restricted to materials that have a maximum collectable volatile condensable material (CVCM) content of 0.1% or less and a total mass loss (TML) of 1.0% or less. Values of CVCM and TML for a wide range of materials may be found on the NASA Outgassing Data for Selecting Spacecraft Materials page: http://outgassing.nasa.gov/."
- Materials with high melting points (ie steels, titanium alloys) shall not be used as structural materials to minimize casualties resulting from reentry debris.
- Toxic and/or volatile fluids or gasses shall not be used.
- Fracture control shall be implemented according to NASA-STD-5003, including multiple load paths and structures built with machined (milled) metals with well-understood properties and having low stresses.
- The primary structure of the satellite shall be machined (milled) and all-metallic.
- All wiring on the satellite shall be copper; aluminum wire shall not be used.

- Use of glass shall be minimized. Where glass must be used, it shall be non-pressurized and subject only to inertial loading, as required by NASA-STD-5003, Section 4.2.3.6.

In addition to maintaining a materials list, all components and parts require proper storage environments suitable to their needs and applications. All storage environments are to be ESD safe and clean, with limited access. In the Flight Hardware Lab (FHL), all working surfaces are ESD safe, and ESD bags are provided for storage of components within the flight hardware cabinets. Components are always stored, tested, and operated within the FHL, which maintains a Class 100,000 clean environment. Finally, FHL access is restricted to project leadership and subsystem leads, protecting flight-critical components from public access.

Finally, flight assurance is the last category of the CSS quality assurance strategy. One important element of this flight assurance is the acquisition of part history tracking records. The CSS acknowledges the importance of part traceability from the forge to the Flight Hardware Lab as a vital component of ensuring quality of flight hardware. Prior to procurement, part traceability for each element of the spacecraft should be confirmed, and an agreement should be reached with the vendor to deliver documentation at the time of purchase. Once supplies and parts are obtained by Georgia Tech, a catalog should be maintained for all part tracking documentation in the FHL.

During assembly and integration of the protoflight and flight hardware, a number of logbooks should be kept. Bolt (torque/untorque) logs and electrical connection (plug/unplug) logs should be made. Connection savers should also be used during integration and testing of the flight hardware to keep electrical connections in faultless flight condition. Every element of the satellite, and any changes made to that element, should be monitored and logged using the appropriate forms to ensure tracking and execution of the quality assurance strategy throughout all handling of the flight system.

The purpose of the Quality Assurance plan is to ensure best practices of all CSS flight projects during the design, fabrication, assembly, and test phases of the program. By being proactive in implementing the methodologies discussed above, the CSS will yield robust designs and spacecraft that are assured for flight readiness once developed.

VI. Mission-Specific Examples from the R^3 Satellite I&T Program

The R^3 team will first execute requirement verification and performance tests on each of the satellite components. Then, the components will be integrated into the satellite in stages to verify sequentially each component's interfaces and integrated functionality before moving to the next component. Once satellite integration is complete, a series of partial- and full-orbit simulations will be run in the integrated configuration to demonstrate the satellite's capabilities under a simulated mission environment.

A. System Description

In order to provide context for the tests detailed in the following sections, a brief description of the R^3 satellite system is provided here. The primary R^3 mission objectives are to evaluate the performance of an uncooled thermal imager (microbolometer) in the space environment and to perform thermal feature identification and geolocation using both thermal and visible imaging. For these purposes, the instrument suite includes a radiation dosimeter to define the radiation environment experienced by the microbolometer for use in assessing its performance, a microbolometer to perform visible imaging and to be monitored for performance anomalies, and a visible imager to provide context images for the thermal feature identification. The attitude determination and control subsystem has both course and fine modes. Course mode is for the applications of initial launch vehicle separation and all non-imaging and non-communication-critical phases, and uses a magnetometer for determination and magnetic torque rods for control. Fine mode is for the application of image-taking or high-gain communication needs, and uses a star tracker for determination and reaction wheels for control. The CDH unit is a DSP/FPGA combination and runs flight software on a Micro-C OS-II Operating System. The communication system utilizes a UHF receiver and monopole receiving antenna for uplink of commands and a stronger / higher-frequency S-band transmitter and circular transmitting antenna for downlink of telemetry and mission data (which is image-heavy and thus requires significant throughput). The EPS is responsible for power production, storage, and distribution, which it accomplishes using body-mounted solar cells for production, Nickel-Cadmium battery cells for storage, and an internally fabricated power management system for regulation and distribution. The TCS is a primarily passive system, using paint and MLI to maintain a high level of control passively. As the satellite is cold-biased, an active system consisting of thermistors and heaters is used throughout the satellite to maintain local temperatures within the tolerances of specific components. Finally, the structure is a simple aluminum bus integrated with L-brackets and aluminum component boxes mounted directly to the face plates.

B. Projected Integration and Testing Schedule

As mentioned in Section IV-C, the R^3 I&T program implemented a phased approach to subsystem integration, such that the component integration schedule is based on the procurement and functionality testing schedule of each component. This method allows for thorough testing to be done on each component individually based on the timeline necessary for full functionality to be confirmed on that component. Table III contains the order of integration and planned timelines for both flat-sat and final satellite integration.

Table III. Order of Integration for the R^3 Satellite

Stage	Component	Start Date	Finish Date
Flat-Sat Integration	Prototype PMAD System	7/1/10	7/26/10
	Flight Computer	7/1/10	8/6/10
	Visible Camera	7/1/10	7/16/10
	Microbolometer	7/1/10	7/26/10
	Dosimeter	7/1/10	8/2/10
	Torque Rod (1)	7/12/10	7/26/10
	Star Tracker	7/15/10	8/4/10
Final Satellite Integration	Structural Assembly	10/7/10	10/13/10
	Visible Camera	9/29/10	10/19/10
	Prototype PMAD with External Power Source	10/11/10	10/22/10
	Battery Box	10/13/10	10/26/10
	Final Computer Development	10/12/10	11/1/10
	Transmitter and Antenna	10/25/10	11/5/10
	Receiver and Antenna	10/25/10	11/5/10
	Prototype PMAD with Batteries as Power Source	10/27/10	11/9/10
	Dosimeter	10/22/10	11/11/10
	Star Tracker	11/2/10	11/15/10
	Torque Rods (3)	11/2/10	11/15/10
	Thermistors	11/1/10	11/26/10
	Heaters	11/1/10	11/26/10
	Microbolometer	11/25/10	12/15/10
	Reaction Wheels	12/6/10	12/24/10
	Magnetometer	1/31/11	2/11/11
	Sun Sensor	1/31/11	2/11/11
	GPS and Antenna	2/7/11	2/18/11
	Solar Array	5/13/11	6/2/11
	Flight PMAD	6/6/11	6/17/11
	MLI	7/1/11	7/21/11

Next, Table IV provides the full detailed I&T schedule for the R3 satellite, including approximate procurement timelines, component testing schedules, and the projected integration calendar.

Table IV. Integration and Testing Master Schedule

Component Testing and Integration	Start Date	Finish Date
Instruments	***5/1/09***	***12/15/10***
Microbolometer	***5/1/09***	***12/15/10***
Camera Procurement	5/1/09	4/14/10
Lens Procurement	4/15/10	9/29/10
Coupler Fabrication	9/30/10	10/27/10
Performance Tests	9/2/10	11/24/10
Integration	11/25/10	12/15/10
Visible Camera	***12/23/09***	***10/19/10***
Procurement	12/23/09	2/16/10
Performance Tests	9/1/10	9/28/10
Integration	9/29/10	10/19/10
Dosimeter	***9/10/10***	***11/11/10***
Fabrication	9/10/10	10/7/10
Performance Tests	10/8/10	10/21/10
Integration	10/22/10	11/11/10
ADCS	***5/1/09***	***2/18/11***
Star Tracker	***5/1/09***	***11/15/10***
Procurement	5/1/09	10/1/09
Performance Tests	7/1/10	7/14/10
Flat-Sat Integration	7/15/10	8/4/10
Integration	11/2/10	11/15/10
Magnetometer	***8/2/10***	***2/11/11***
Procurement	8/2/10	1/14/11
Performance Tests	1/17/11	1/28/11
Integration	1/31/11	2/11/11
Sun Sensor	***11/1/10***	***2/11/11***
Procurement	11/1/10	1/21/11
Performance Tests	1/24/11	1/28/11
Integration	1/31/11	2/11/11
Reaction Wheels	***5/3/10***	***12/24/10***
Procurement	5/3/10	11/12/10
Performance Tests	11/15/10	12/3/10
Integration	12/6/10	12/24/10
Torque Rods	***8/2/10***	***11/15/10***
Fabrication	8/2/10	9/24/10
Performance Tests	9/27/10	10/8/10
Integration	11/2/10	11/15/10
GPS and GPS Antenna	***8/2/10***	***2/18/11***
Procurement	8/2/10	1/14/11
Performance Tests	1/17/11	2/4/11
Integration	2/7/11	2/18/11
CDH	***2/1/10***	***11/1/10***
Processor Board	***2/1/10***	***4/23/10***
Procurement	2/1/10	4/23/10
Interface Board	***6/1/10***	***9/20/10***
Fabrication	6/1/10	9/20/10
Flight Computer Modifications	4/26/10	11/1/10

Flat-Sat Development/Fabrication	4/26/10	7/16/10
Flat-Sat Performance Tests	7/19/10	8/6/10
Final Performance Tests	9/21/10	10/11/10
Integration	10/12/10	11/1/10
Communication	***5/3/10***	***11/5/10***
Transmitter and Transmitting Antenna	***5/3/10***	***11/5/10***
Procurement	5/3/10	9/17/10
Performance Tests	9/20/10	10/15/10
Integration	10/25/10	11/5/10
Receiver and Receiving Antenna	***5/3/10***	***11/5/10***
Procurement	5/3/10	9/17/10
Performance Tests	9/20/10	10/15/10
Integration	10/25/10	11/5/10
EPS	***1/1/10***	***6/17/11***
Solar Array	***1/1/10***	***6/2/11***
Sample Array	1/1/10	11/5/10
Procurement	1/1/10	4/22/10
Array Fabrication	8/2/10	10/22/10
Performance Tests	10/25/10	11/5/10
Procurement	1/1/11	2/24/11
Array Fabrication	2/25/11	4/21/11
Performance Tests	4/22/11	5/12/11
Integration	5/13/11	6/2/11
Battery	***3/1/10***	***10/26/10***
Prototype Battery	3/1/10	8/3/10
Procurement	3/1/10	3/26/10
Performance Tests	7/1/10	7/16/10
Battery Box Fabrication	7/9/10	7/30/10
Flat-Sat Integration	8/2/10	8/3/10
Flight Battery Box	9/1/10	10/26/10
Fabrication	9/1/10	9/28/10
Testing	9/29/10	10/12/10
Integration	10/13/10	10/26/10
PMAD System	***8/9/10***	***6/17/11***
Prototype PMAD	8/9/10	11/9/10
Procurement/Fabrication	8/9/10	10/1/10
Performance Tests	10/4/10	10/8/10
Integration Using External Power Source	10/11/10	10/22/10
Integration Using Prototype Battery	10/27/10	11/9/10
Flight PMAD	1/3/11	6/17/11
Procurement	1/3/11	3/25/11
Fabrication	3/28/11	5/20/11
Performance Tests	5/23/11	6/3/11
Integration	6/6/11	6/17/11
Structure	***7/15/10***	***10/13/10***
Side Plate Fabrication (1)	7/15/10	7/28/10
Side Plate Fabrication (3)	7/29/10	8/25/10
Zenith Plate Fabrication	8/26/10	9/8/10
Nadir Plate Fabrication	9/9/10	9/22/10
Structural Integrity Tests	9/23/10	10/6/10

Assembly	10/7/10	10/13/10
TCS	***8/2/10***	***7/21/11***
Thermistors	***8/2/10***	***11/26/10***
Procurement	8/2/10	8/27/10
Performance Tests	8/30/10	9/1/10
Integration	11/1/10	11/26/10
Heaters	***8/2/10***	***11/26/10***
Procurement	8/2/10	8/27/10
Performance Tests	8/30/10	9/1/10
Integration	11/1/10	11/26/10
MLI	***12/1/11***	***7/21/11***
Procurement	12/1/11	12/28/11
Integration	7/1/11	7/21/11

C. Component Environmental Testing

This section includes details of the specific components to which the environmental test guidelines of Section IV-B apply. Note that descriptions and procedures of all other component tests (performance, requirement verification), as indicated in Section IV-B, are provided in Appendix C.

The first environmental test, thermal cycling, will only be performed on the components with the tightest hot and cold limits, which for R^3 are the visible camera, the battery cells, and the computer's processor board. Thermal cycling testing on the visible camera will be performed at ASU, while that of the other two components will be done in the thermal cycling chambers at GTRI Atlanta.

Thermal vacuum tests must be performed on the microbolometer and the visible camera to ensure that outgassing will not affect the performance of the instruments. This test must be performed since neither instrument is traditionally flown in space, and extra care must be taken to ensure proper functionality in an in-space environment. Since these instruments both have glass optics, any outgassing material that settles on the lenses could partially or fully impair the operation of the instruments and thus prevent the mission success criteria from being achievable. Both imagers will undergo this test in the Spectrometer Lab at ASU. Here, the imager will be placed in a thermal-vacuum chamber. When the imagers are first placed within the chamber, they will be interfaced with the computer and will begin to take images with a frequency of one image per minute. When the test pressure and temperature is reached, imaging will continue for a period of 24 hours. Then the chamber will be returned to normal atmospheric conditions, imaging will continue for one more hour, and then the imagers will be removed from the chamber and the integrity of the optics will be observed. Finally, images taken during the test will also be analyzed for failures due to outgassing. Thermal vacuum tests will also be performed as a workmanship inspection for the dosimeter, the flight computer boards, the PMAD system, and the solar panels.

A shock test, the third environmental test described in Section IV-B, must be performed on the two imaging components: the microbolometer assembled with its lens and the visible camera assembled with its lens, since the lenses for both contain glass. The star tracker also contains glass, but is validated to withstand the shock environment of launch by flight qualification and successful on-orbit operation of similar units. The star tracker utilized for the R^3 satellite was previously used on Mir and on the International Space Station, and has survived multiple launch environments. Finally, the flight computer (both the processor board and interface board) contains crystals, so a shock test must be performed on this component as well.

The fourth environmental test, a vibration test, is recommended for the internally fabricated R^3 solar panels. Both sinusoidal vibration tests and random vibration tests must be performed to verify that the structural integrity of the electrical connections are maintained under a vibration environment similar to that of the expected launch environment. For the sinusoidal vibration test, the solar panel will be secured parallel to (but electrically isolated from) the vibration table. During the test, the solar panel will be monitored for any harmonic motion or physical damage occurring on the assembly. After the test, basic operational tests will be performed to compare baseline open circuit voltages and short circuit current to ensure they do not differ from pre-test values by more than 5%.

D. System-level Testing

The system-level tests planned for the R^3 satellite are overviewed in this section. Detailed Test Plan Forms are provided in Appendix C electronically. The first is a comprehensive exploration of the course attitude determination and control functions. In this test, the magnetometer will take a reading and send the magnetic field vector to the flight computer, which will process the necessary action using ADCS flight code and command the magnetic torquers to control the satellite attitude. This test will need to occur for one axis at a time, so that the satellite can be suspended in such a way that the torquer is able to freely orient the satellite along that axis.

The second test involves a similar procedure to test the fine attitude determination and control functions. In this test, the star tracker will take a simulated input and send the determined quaternion to the flight computer, which will process the necessary action using ADCS flight code and command the reaction wheels to control the satellite attitude. As with the previous test sequence, the test will need to occur for one axis at a time to enable a suspension configuration where that axis can be oriented and controlled freely by the reaction wheels.

The third system-level test planned is an end-to-end downlink communication test using a dummy load. The transmit equipment is easily damaged and must be operated carefully using the proper test equipment. This equipment includes a coaxial cable and a 50 Ohm dummy load that operate at the same frequency as the R^3 transmitter and should have capability to block 5 W of power. This ensures that the 2 W RF power transmitted by the R^3 transmitter will be fully blocked by the load. The coaxial cable will attach the transmitter to the dummy load, which absorbs all RF output power that reaches it. The cable, however, will leak enough radiated power into the surrounding area to be picked up by the ground receiving antenna. Note that nothing is attached directly to the ground receiving antenna or ground receiver, but rather they pick up a low-power leaked and propagated RF signal. There are two ways to accomplish this blockage of RF signal to ensure it is weak enough not to interfere with any other antennas in the area beyond the Georgia Tech ground station. The first is using a termination load (the dummy load described above) attached directly to the transmitter or to the transmitting antenna. If attached to the transmitter, the transmitter will get very hot after approximately one minute and will need to be actively cooled. If attached to the antenna, energy will be radiated so the equipment will not heat up so dramatically, but the antenna will need to be mounted to a ground plane. For this purpose, a 4 x 4 inch square of aluminum should be sufficient. Instead of using a dummy load to dissipate power and using the receiving antenna to pick up this weak signal, a second method involves attaching an attenuation cable directly between the R^3 transmitter and the ground receiver. This receiver must be finely tuned to dissipate the exact amount of RF energy (recommendation: 30-40 dB attenuated). Since cables are not typically sold with a tight enough tolerances to absorb the precise amount of power that are capable of withstanding the head of that load themselves, this method is not recommended.

The fourth system test that immediately follows is an end-to-end uplink communication test. Note for testing that the receiving equipment is not as sensitive as the previously discussed transmit equipment. Using the pinouts given in the R^3 receiver documentation, a connector must first be made for the power supply. The receiver can then be powered using a power supply, monitored to ensure voltage and current limits are not exceeded. The monopole antenna to be used with this receiver must be tuned to the proper frequency by cutting it to the correct length. In this test, a packet of information will be sent from the ground station transmitter, through attenuation cables into a dummy load, and ambient signal will be picked up by the satellite receiver. The receiver will passed on this packet to the flight computer which will provide a readout to an external display to confirm accurate packet transmission.

During all modes except the Low Battery State of Charge (LBSC) Safe Mode, the transmitter will be turned on/off via an overflight table. The GPS will provide orbit data (position and velocity) to mission control, which will determine when the satellite is expected to pass over the Georgia Tech Ground Station. The transmitter will be turned on with enough margin to allow for the equipment start up to be complete by the time the satellite's orbital position reaches 20 degrees in elevation above the ground station, at which point signal acquisition will commence. This overflight table also commands data rates based on the link budget for each expected point on the overflight orbit to maintain a link margin of 3 dB at all times. A final communication test involves assessing the functionality of these tables. To test this function, a sample table with at least two on/off cycles will be uploaded to the computer, which should power on the transmitter at a specific time and begin sending data through it to be transmitted to the ground using the same setup as the transmit test above. This test allows for another step in the direction of full satellite functional integration with realistic mission protocol software.

Next, a series of EPS system-level tests should be performed. The first of these tests is intended to verify that the power distribution and regulation system can distribute power to loads properly and to gain a better understanding of the power characteristics of the system as the battery depletes. This test will need a dummy load, a DMM, inline power meters (or DAQ), and DC power supply. First, the battery should be fully charged and the battery voltage should be recorded. Inline power meters (or DAQ) should be connected behind voltage regulators at each line, and dummy loads should be connected after overload and voltage protection system. A DMM should be connected to the thermistor on the battery, and then the battery should be connected to the power distribution system. These steps complete the experiment setup. Next, the voltage, current, and time started should be recorded, followed by recordings at regular time intervals of the battery thermistor readings and the voltage and current of the inline power meters. The time should be recorded when battery DoD reaches 45%, and the test should then be continued until DoD reaches 90%. The test should then be stopped by disconnecting the battery, and the final battery voltage and temperature should be recorded. Finally, the battery should be returned to the charger and cycled. Record this battery voltage.

The seventh system-level test integrates all elements of the EPS system in an end-to-end power acquisition, storage, and distribution assessment. This test sequence simulates on-orbit power acquisition using a sun simulator to shine light into the solar cells, which will then pass the electric power through the power distribution system to various loads, including battery charging. The test should operate a satellite component as a test load, and should include both sunlight and eclipse simulations (the sun simulator on and off). During sunlight, the solar cells should provide power to the test load, while during eclipse, the stored power in the batteries should power the test load such that it receives continuous power during sunlight, eclipse, and the transition between them. Note that using a classic sun simulator for this test is outside the scope of this project. Therefore, it is recommended that a theater spotlight be used as a viable substitute, as it should produce enough light to properly conduct the test. Georgia Tech's drama production crew, Drama Tech, has a number of stage lighting options, and collaboration with them for potential use of equipment should be investigated.

The eighth system-level test is a flight software control of the temperature sensors and emergency cooling protocols. The purpose of this test is to verify the functionality of the software developed for integrated TCS operation to ensure the system is capable of reading accurate temperatures, identifying critical hot conditions, and entering safe mode as necessary. The integrated satellite will need to be tested within the range of thermal environments expected during the R^3 mission in order to verify system survivability and operability during all thermal conditions. The procedure for this test will require that the thermistor be exposed to a known temperature that is then increased until it reaches a critical value. The test will be proven to be successful if certain components are triggered to shut down when the thermistor is exposed to the maximum operational temperature for that region of the satellite. In order to perform these functionality tests of the thermal control system after it has been fully integrated with the satellite, special equipment will be needed. Thermal vacuum and cycling tests are planned for the entire satellite after delivery to AFRL. However, it is also possible that the heating elements on-board the spacecraft can be verified to produce the correct amount of heat by use of an IR camera. This non-invasive testing method would allow the heaters to be checked in the fully integrated system without the risk of damaging other components in the process.

A few tests will then need to be performed on the tracking station, as the ground segment is a key element of the satellite's mission success. First, a computer software integration test must be completed. The final stage of ground station is Computer Software Configuration, which will be done progressively. First, the R3 team will configure the rotor control assembly box (RC2800PRKXSU) computer program (NOVA for Windows) for remote desktop operation. The team will then configure the Receiver (IC-R9500) and Transmitter (TS-2000X) computer program (Ham Radio Deluxe) for remote desktop operation. The next step will be to operate NOVA for Windows and Ham Radio Deluxe simultaneously, and then to do so again using Windows remote desktop feature.

Then, end-to-end testing will need to be performed with the tracking station, closely paralleling the communications tests discussed previously. Functional testing of the Georgia Tech Ground Station consists of locally and remotely operating the station for receiving and transmitting data. To perform closed loop testing / ground station integration, the team will confirm rotor/receiver/transmitter control with appropriate and properly configured computer software, both locally and remotely. Then, to complete closed loop testing, the team will successfully receive an image from an orbiting NOAA satellite, display it on the computer, and post it to the server Ground Data System.

Finally, the integration of the science payload will occur after subsystem interaction and controllability has been successfully demonstrated. At this point, the following system-level payload tests will be performed. The first of these tests is dosimeter data production. To fully determine the functionality of the dosimeter, the test will have two primary purposes: to confirm its interfaces with internal power and data sources (the satellite batteries and flight computer) enable the dosimeter to power on, collect data, and return radiation information to the flight computer; and to provide an environment with a significant known radiation source to ensure the data provided by the dosimeter is meaningful and accurate. One such test location currently being considered is a local hospital, since x-rays and some cancer treatment equipment are sizeable producers of measurable radiation.

The next system-level test will involve imaging commanded by an overflight table. Since functionality of the imagers will have already been demonstrated during component testing, and the interfaces between the visible camera, the flight computer, and the satellite power distribution will have already been confirmed during integration, the next step is to provide the true on-orbit scenario for image acquisition. During normal mode (the mode in which science data acquisition occurs), imaging will be initiated using a feature overflight table. The GPS will provide orbit data (position and velocity) to mission control, which will determine when the satellite is expected to pass over certain features of interest and develop a timeline of when to acquire image data. For this test, a sample table will be uploaded to the computer, which should command both the microbolometer and visible camera to begin taking images and continue until desired time period has been reached or the desired number of images has been acquired.

Finally, a test must be performed to fully analyze proper image processing, feature identification, and coordinate output. This end-to-end test confirms the third and final element of the R3 science mission: the capability of the microbolometer to acquire image data and pass it to the flight computer, and more importantly, the capability of the flight software to obtain the images from the microbolometer, identify and geolocate the thermal features within them using the on-board processing algorithms, and output all coordinates of the features of interest found within the images. Any feature with a significant temperature difference (ie: a cup of hot liquid, a human body, etc.) could be used as the simulated thermal feature for this processing test.

E. Full Orbit Simulation Testing

The R^3 satellite has three modes of operation: Startup Mode, Safe Mode, and Normal Mode. Each of these is a distinct state of operational capability in which the satellite can function, and is programmed into the satellite's onboard computer to be executed autonomously unless overridden by ground command. Therefore, ground testing of the software associated with these modes must be rigorously performed to explore all possible anomalies and monitor how the satellite autonomously responds. As a foundation for discussing the various modes tests ("day-in-the-life" tests) planned for the R^3 satellite, a brief description of the satellite's three modes is provided below, as defined by Jenny Kelly in the R^3 Mission Plan in April 2010:

> ***Startup Mode:*** Once the satellite enters sunlight, the Command and Data Handling (C&DH) subsystem will activate and begin charging the battery. The satellite will autonomously initiate the startup sequence on the flight computer, which includes monitoring the battery state of charge, initiating thermal control, activating the telecom transceiver (receive only), and initiating the recording of telemetry from subsystems. Upon completion of the startup sequence, the satellite will change the mission phase bit to safe mode.
>
> ***Transitioning to Safe Mode:*** The satellite can enter safe mode in three primary ways: (1) after the battery is initially charged during startup, (2) upon receiving a command from ground, or (3) after anomalous behavior. Potential anomalous behaviors have been identified and include: deviations from predicted performance, loss of attitude/position knowledge, loss of attitude control, unexpected loss of communication with ground, and subsystems reaching a critical point (e.g., computer is reset, battery reaches maximum depth-of-discharge, electrical fault, loss of power production for a period longer than the maximum expected eclipse time, etc.).
>
> ***Safe Mode:*** Upon entering safe mode, the satellite will perform two sequences of autonomous tasks: a low battery state of charge (LBSC) sequence followed by a full battery state of charge (FBSC) sequence. During the LBSC sequence, any other active sequences will be terminated and the statuses of specific spacecraft components will be asserted as shown in Table V. The flight computer will then monitor the battery state of charge.

Table V. Component Statuses Asserted During the Safe Mode LBSC Sequence

Component	Status
Science Instruments	OFF
Reaction Wheels	OFF
Star Tracker	OFF
C&DH Subsystem	ON
Telecom Receiver	ON
Telecom Transmitter	OFF
Magnetometer	OFF
Torque Rods	OFF
GPS	OFF

Thermal Heaters	ON

When the previous sequence is complete and the battery is fully charged, the FBSC sequence will be initiated. Coarse attitude determination and control will be activated by asserting component statuses as shown in the first three rows of Table VI. The satellite will then execute its detumble procedure, damping any spin rates with the magnetic torquers. The telecom transmitter will be powered ON/OFF via an overflight table onboard the satellite (i.e., the transmitter will only be ON during overflights of the Georgia Tech tracking station), also reflected in Table VI. Telemetry-only communication with the ground will be established at a low data rate to maximize signal strength.

Table VI. Component Statuses Asserted During the Safe Mode FBSC Sequence

Component	Status
Magnetometer	ON
Torque Rods (phased with magnetometer)	ON
GPS	ON
Telecom Transmitter	ON/OFF via overflight table

Exiting Safe Mode: When both the LBSC and FBSC safe mode sequences are complete, the satellite can transition to normal mode in one of two ways: autonomously or ground-in-the-loop.

If the safe mode entry was due to exceeding the maximum battery depth-of-discharge, safe mode can be exited through an autonomous exit sequence. By allowing the satellite to respond autonomously, the R^3 team avoids wasting several days of operation on a relatively straightforward problem. The exit sequence will begin with the satellite transitioning to fine attitude determination and control, asserting component statuses as shown in the first four rows of Table VII. The satellite will then prepare for normal operations by orienting for imaging; the imaging attitude is nadir-pointing, with the imager's vertical field of view aligned with the velocity direction. Finally, the satellite will establish two-way communication with the ground at a normal data rate, initiate the imaging sequence, and change the mission phase bit to normal mode.

Table VII. Component Statuses Asserted During the Autonomous Safe Mode Exit Sequence

Component	Status
Science Instruments	ON
Reaction Wheels	ON
Star Tracker	ON
Magnetometer	OFF
Torque Rods	OFF
Telecom Transmitter	OFF
Magnetometer	OFF
Torque Rods	OFF
GPS	OFF
Thermal Heaters	ON

If the safe mode entry occurred for any other reason than exceeding the maximum battery depth- of-discharge, a ground-in-the-loop exit sequence must be used. This exit sequence will be a prepared contingency plan response with different paths for different causes of safe mode entry.

Normal Mode: Upon entering normal mode, the satellite will initiate normal operations. This includes acquiring thermal and visible images of specified regions via sequenced commands from ground control, detecting and geolocating thermal features, and measuring the radiation environment surrounding the microbolometer.

The actions of each of these three modes should be tested in their entirety, according to the modes testing categories described here. The first and simplest of these tests is a safe mode orbit simulation, in which the satellite is assumed to begin the orbit in a state of safe mode, and never leaves this state. Both LBSC and FBSC safe modes should be explored, however, so the transition between these two states will be a part of this simulation. The satellite will begin with faintly charged batteries, and the satellite will command only the components on during LBSC safe mode (flight computer, receiver, heaters if triggered) to start up. A sun simulator (or theater spotlight) will be used to create light for the solar cells to convert to electrical power and use to power the operating components and charge the batteries. Once the batteries are fully charged, the FBSC safe mode sequence will be initiated. The flight computer will command power-up of the magnetometer, torque rods (phased with magnetometer) and GPS to begin course attitude determination and control. The transmitter will be powered on/off during the overflight table created in Test Sequence 1 (above), such that the orbit being simulated contains an overflight of the ground station. Two simulations should be run: one in which the overflight occurs during LBSC safe mode, in which only the receiver is turned on, to confirm the satellite properly receives a test data packet; and a second in which the overflight occurs during FBSC safe mode, in which a test data packet For a full description of the Safe Mode operations to be executed during the mission, refer to the R3 Mission Plan.

In review, this first orbit simulation will confirm that proper flight code has been written and successful interaction of satellite components can be achieved throughout the operations expected to occur in safe mode. These include power acquisition, battery charging, and receiver operation in LBSC safe mode, transition to FBSC safe mode, power-on and operation of course attitude determination and control and transmitter, and successful application of the overflight table as a communication strategy.

The next test to be performed is a Normal Mode (Science Data Acquisition) orbit simulation. This test should be performed only after the previous test has been successfully completed, as it builds upon that test in complexity. The normal mode orbit simulation is assumed to begin in normal mode with batteries fully (or near-fully) charged, and using the fine attitude determination and control system only. Thus, all components are powered on except the magnetometer, torque rods, and transceiver, the last of which is commanded on/off via the communication overflight table. This simulation should include all major operations that occur in normal mode, specifically: science data acquisition (constant radiation data gathering by the dosimeter, and image acquisition by the microbolometer and visible camera at the times designated by the feature overflight table), fine attitude determination and control (gps, star tracker, and reaction wheels only), all functions of the electrical power system (acquisition, storage, management, and distribution) in both sunlight and eclipse, normal-mode communication (uplink of commands and high data rate downlink of science data, at times designated by the ground station overflight table), and thermal monitoring and heater activation when determined necessary.

The final and most complete orbit simulation is a mission life cycle simulation, and will serve to conclusively determine the satellite is ready for delivery to AFRL. This simulation will take the satellite from launch configuration (powered down, batteries discharged), through all inhibits, commands, and actions for initial startup, checkout, and preparation for normal mode. In other words, this simulation carries the satellite through all mission phases that prepare the satellite to begin the science mission. It will then commence simulated normal operations (identical to the previous test) for at least one orbit. Finally, the procedures for shutting down the satellite at End of Mission will be commanded from the ground station to turn off radio transmission, disconnect the batteries from the system, and shorting the batteries and solar arrays to shunt resistors, bringing the satellite to a state that fully prepared for orbit disposal. For complete details of what is required throughout these processes, refer to the R^3 Mission Plan.

VII. Conclusion

In conclusion, the purpose and aspiration of this document is to standardize a set of best practices relating to the flight projects developed within the Center for Space Systems. Specifically, it provides (1) guiding principles for requirement development and verification, (2) procedural and contextual instructions on the stages of an integration and testing program, (3) standards and protocols for quality assurance and hardware handling, and (4) specific examples from implementation of this framework in the R^3 flight project. Intelligent requirement development following the guidelines discussed in this paper is crucial to the setup of a successful flight project. Likewise, a thorough integration and testing program adhering to the methodology outlined in this paper is critical to the final assurance this success. Together, these two elements provide the structure that should guide a space flight project from initial design to launch preparation. This paper aims to provide tools for developing a flight-worthy satellite with full assurance of its capability to achieve mission success.

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Appendix A: R^3 Requirement Verification Matrix

Mission Statement
The Rapid Reconnaissance and Response (R3) mission will characterize the radiation environment in low earth orbit and evaluate radiation effects on an uncooled microbolometer thermal imager. Onboard thermal image processing will be used to geolocate thermal features of interest. The R3 program will provide education and public outreach opportunities by involving K-12 students throughout the development and operation of the satellite.

MO		**Mission Objectives**					
	MO-1	R3 shall monitor the radiation environment in terms of total ionizing dose and single event effects.					
	MO-2	R3 shall characterize the radiation effects on the performance of an uncooled microbolometer thermal imager.					
	MO-3	R3 shall acquire thermal images from low earth orbit, and utilize onboard image processing algorithms to detect and geolocate thermal features having specified signatures.					
	MO-4	R3 shall provide educational outreach opportunities to K-12 students.					
MSC		**Mission Success Criteria**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	MSC-1	The R3 mission shall provide enough radiation data of total dose and single events to fully characterize the radiation environment in low earth orbit.	MO-1	Analysis	Designed	N/A	
	MSC-2	The R3 mission shall compare the microbolometer performance with the radiation environment to determine microbolometer sensitivity to radiation effects.	MO-2	Analysis	Designed	N/A	
	MSC-3	The R3 mission shall acquire thermal images from low earth orbit.	MO-3	Testing	Designed	N/A	
	MSC-4	The R3 mission shall utilize onboard image processing algorithms to detect features having specified signatures.	MO-3	Testing	Designed	N/A	
	MSC-5	The R3 mission shall geolocate the thermal features using onboard processes.	MO-3	Testing	Designed	N/A	
	MSC-6	The R3 mission shall acquire visible images providing context for the thermal images.	MO-3	Testing	Designed	N/A	
	MSC-7	The R3 mission shall provide educational outreach opportunities for a broad range of students during satellite development and mission operations.	MO-4	Inspection	Designed	N/A	
	MSC-8	The R3 mission shall fulfill at least one Technology area of interest to the Air Force, as listed in Appendix B of the Nanosat-6 User's Guide.	NUG	Inspection	Verified	N/A	VV-R3-2010-001
MD		**Mission Design**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	MD-1	The duration of the primary mission shall be 6 months to enable characterization of radiation effects on the microbolometer.	MSC-1,MSC-2	Analysis	Verified	N/A	Mission Plan
	MD-1.1	The R3 satellite must be launched into an orbit with a minimum altitude of 380 km to maintain a usable orbit during the primary mission duration.	MD-1	Analysis	Verified	N/A	Mission Plan
	MD-2	The R3 satellite shall continuously acquire radiation total dose and single event data from orbit while in Normal Mode.	MSC-1	Testing	Designed	N/A	
	MD-3	Initial power-up shall consist of a startup mode in which all potentially hazardous operations (activation of deployables, RF emissions, ACS system activation, etc) are inhibited, as defined according to NSTS 1700.7B.	NUG	Inspection	In Progress	N/A	
	MD-4	Once the R3 satellite has achieved a safe distance from the launch vehicle, [nominally 30 minutes], regular operations can commence.	NUG	Testing	In Progress	N/A	
	MD-5	Deorbiting / moving to a disposal orbit shall occur within 25 years from end-of-life.	NUG	Analysis	Verified	N/A	Mission Plan
	MD-5.1	End-of-life safing shall include turning off radio transmission, disconnecting batteries from the system, shorting batteries and solar arrays to shunt resistors, and deorbiting/moving to disposal orbit.	MD-5	Testing	Designed	N/A	

	MD-5.2	The R3 satellite must be launched into an orbit with a maximum altitude of 600 km to ensure the orbit degrades within 25 years from end of life.	MD-5	Analysis	Verified	N/A	Mission Plan
SAT		**Satellite System**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	SAT-1	The R3 satellite will be completely integrated (including flight software and all interfaces and documentation), ready for environmental testing, and delivered to AFRL by June of 2011.	NUG	Inspection	Under Review	N/A	
	SAT-2	The R3 satellite shall be designed to withstand the launch and the environment of the launch vehicle without failure, leaking hazardous fluids, or releasing anything that could damage the LV or cause injury to the ground handling crew.	NUG	Analysis	Designed	N/A	
	SAT-3	The R3 satellite and all components shall be capable of surviving operation in space for the primary mission duration.	MD-1	Analysis	Designed	N/A	
	SAT-4	All hazards to the satellite equipment shall be identified, controlled, and verified in accordance with NASA Document NSTS 13830, NSTS 1700.7B, and KHB 1700.7C.	NUG	Inspection	Designed	N/A	
	SAT-5	The mass of the R3 satellite shall not exceed 50 kg. The satellite mass includes all mass above the Satellite Interface Plane (SIP) and the bolts used to attach the R3 satellite to the Lightband at the SIP.	NUG	Inspection	Designed	N/A	
	SAT-6	The allowable static envelope for the R3 satellite, illustrated with respect to the SIP in Figure 6-1 of the Nanosat-6 User's Guide, is defined as a volume with linear dimensions of 50 cm in width, 50 cm in length, and 60 cm in height.	NUG	Inspection	Designed	N/A	
	SAT-7	All hardware used in the construction of the R3 satellite shall be traceable and supported by documentation as specified in the Configuration Management (CM) Plan, AFRL Document UN6-0002.	NUG	Inspection	In Progress	N/A	
	SAT-8	Certificates of compliance (C of C's) shall be provided for all protoflight hardware.	NUG	Inspection	In Progress	N/A	
	SAT-9	The R3 satellite shall contain safety features and inhibits for satellite-related hazards. The inhibits shall be designed to allow the verification of inhibits without distubing flight interfaces.	NUG	Inspection	Designed	N/A	
	SAT-10	Use of glass shall be minimized. Where glass must be used, it shall be non-pressurized and subject only to inertial loading, as required by NASA-STD-5003, Section 4.2.3.6.	NUG	Inspection	Verified	N/A	VV-R3-2010-010
	SAT-11	Basic health monitoring shall be performed on all hardware.	NUG	Inspection	Verified	N/A	VV-R3-2010-017
	SAT-12	Temperature measurements shall be tracked for all components.	NUG	Testing	In Progress		
	SAT-13	Current and voltage measurements shall be tracked for solar panels, batteries, power distribution busses, and related components.	NUG	Testing	In Progress		
	SAT-14	All elements of the R3 satellite shall adhere to the International Traffic in Arms Regulations (ITAR), as discussed in Appendix A of the Nanosat-6 User's Guide.	NUG	Inspection	In Progress	N/A	
	SAT-15	All wiring on the R3 satellite shall be copper; aluminum wire shall not be used.	NUG	Inspection	Designed	N/A	
	SAT-16	Wire shall not make contact with dissimilar metals on the R3 satellite.	NUG	Inspection	Designed	N/A	
	SAT-17	The R3 satellite shall be capable of venting at a depressurization rate (LV ascent) of 0.50 psi/sec maximum.	NUG	Testing	Designed		
	SAT-18	Functional testing during environmental tests shall exercise all hardware, including the RF communications subsystem.	NUG	Testing	In Progress		
LVI		**Launch Vehicle Interface**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**

	LVI-1	The R3 satellite shall be designed to be flown in an open configuration, exposed to the interior of the launch vehicle (LV) payload fairing and deployed directly from the LV interface using the AFRL-supplied Planetary Systems Corp (PSC) motorized Lightband low-shock separation system.	NUG	Inspection	Under Review	N/A	
	LVI-2	The R3 satellite shall be subject to the requirements of PSC Document 2000785 Rev. A in order to be compatible with the Lightband separation system.	NUG	Inspection	Verified	N/A	VV-R3-2010-021
	LVI-3	The R3 satellite shall be mounted to the LV interface with a PSC Motorized Lightband system, provided by AFRL.	NUG	Inspection	Designed	N/A	
	LVI-4	The R3 satellite shall contain mechanical interfaces with the Lightband.	NUG	Inspection	Designed	N/A	
	LVI-5	The interface shall consist of a fastener hole pattern that corresponds with the pattern on the Lightband upper adapter ring, with positive backout prevention for the fasteners.	NUG	Inspection	Verified	N/A	VV-R3-2010-022
	LVI-6	Fastener holes shall be sized for 1/4-28 socket head cap screws (NASA Part #1351N4 Series are recommended and are available at supply.gsfc.nasa.gov/fasteners/Default.htm).	NUG	Inspection	Designed	N/A	
	LVI-7	The Lightband upper adapter ring shall be bolted to the base of the R3 satellite at 24 locations.	NUG	Inspection	Designed	N/A	
	LVI-8	The R3 satellite shall be able to interface to the Lightband in any rotational position that matches their 24-hole 1/4-28 bolt pattern.	NUG	Testing	Designed		
	LVI-9	The R3 satellite shall be able to interface to the Lightband in any rotational position that matches their 24-hole 1/4-28 bolt pattern.	NUG	Inspection	Designed	N/A	
	LVI-10	The R3 satellite shall contain electrical interfaces with the Lightband.	NUG	Inspection	In Progress	N/A	
	LVI-11	The R3 satellite shall provide a continuous, electrically conductive path between each major structural component and the launch vehicle interface.	NUG	Testing	In Progress		
	LVI-12	The satellite-side SSP connectors shall provide the following electrical functions to the LV side of the interface: monitoring of the R3 safety inhibits; post-LV-integration ground testing and servicing operations (includes battery charging and functional tests); structural ground connection for the R3 satellite; and Lightband separation detection (generally redundant loopbacks, i.e. four pins).	NUG	Testing	In Progress		
	LVI-13	The Lightband shall be equipped with 2 microswitches at the separation plane which may be used by the R3 satellite as inhibits and/or as a means of detecting separation from the launch vehicle.	NUG	Inspection	Designed	N/A	
	LVI-14	The Lightband upper adapter ring shall remain attached to the R3 satellite after the satellite is deployed from the LV.	NUG	Inspection	Designed	N/A	
	LVI-15	The R3 satellite shall have a wire pigtail on the satellite side of the satellite separation plane (SSP) for connection to the Lightband separation connector. The pigtail must extend approximately 30.5 cm past the SIP for assembly and integration with the Lightband.	NUG	Inspection	In Progress	N/A	
	LVI-16	The R3 satellite shall withstand a tip off of <1 deg/sec relative to the launch vehicle deployable interface.	NUG	Analysis, Testing	Designed		
	LVI-17	The R3 satellite shall withstand a separation speed of 0.1524 - 1.2192 m/sec (inertial frame) imparted by the Lightband.	NUG	Analysis, Testing	Designed		
	LVI-18	The service and test port interface (STPI) on the LV shall be the sole ground operations port for R3 once it is integrated with the LV.	NUG	Inspection	In Progress		
	LVI-19	Functional testing after integration to the launch vehicle shall be performed without free radiation of RF energy.	NUG	Testing	Designed		
EMC		**Electromagnetic Compatibility**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**

	EMC-1	The R3 satellite shall be designed for electromagnetic compatibility (EMC) and for mitigation of electromagnetic interference (EMI), specifically such that launch vehicle and range radiation environments do not adversely affect the R3 satellite.	NUG	Inspection	Verified	N/A	Test Plans IX
	EMC-2	All electronics in the R3 satellite shall be housed in machined (milled) EMI-shielded enclosures, and mesh shall be placed over any openings in an EMI enclosure.	NUG	Inspection	In Progress	N/A	
	EMC-2.1	The CDH Subsystem shall be shielded by an enclosed EMI container to protect components from EMI.	EMC-2	Inspection	Verified	N/A	VV-R3-2010-016
	EMC-3	The EPS design shall utilize EMI mitigation techniques for electromagnetic compatibility (EMC).	NUG	Inspection, Analysis	In Progress	N/A	

INS		**Science Instruments**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	INS-1	The R3 satellite shall use an uncooled microbolometer thermal imager to acquire thermal images from orbit.	MSC-3	Inspection	Verified	N/A	VV-R3-2010-005
	INS-2	The infrared imager (microbolometer and lens) shall provide the spatial and thermal resolution necessary for the image processing algorithms to correctly identify features of interest.	MSC-4	Analysis	Verified	N/A	VV-R3-2010-011, 012, 013
	INS-2.1	The infrared imager shall have an instantaneous field of view (pixel width) of less than 70 arcseconds (170 meters per pixel from 500 km altitude).	INS-2	Analysis	Verified	N/A	VV-R3-2010-011
	INS-2.2	The infrared imager shall have a horizontal field of view (footprint width) of greater than 6.5 degrees but less than 14.5 degrees (56 to 127 kilometers from 500 km altitude).	INS-2	Inspection	Verified	N/A	VV-R3-2010-012
	INS-2.3	The microbolometer shall have a noise equivalent temperature difference (NETD) of 1 K.	INS-2	Inspection	Verified	N/A	VV-R3-2010-013
	INS-3	The R3 satellite shall use a visible camera to provide context images from orbit of the region surrounding the thermal imaging feature.	MSC-6	Inspection	Verified	N/A	VV-R3-2010-006
	INS-3.1	The spatial resolution of the visible camera shall be sufficient to visually identify features of interest while still maintaining a larger footprint size than that of the microbolometer.	INS-3	Analysis	Verified	N/A	VV-R3-2010-007
	INS-3.1.1	The visible camera shall have an instantaneous field of view (pixel width) of less than 40 arcseconds (100 m from 500 km altitude).	INS-3.1	Analysis	Verified	N/A	VV-R3-2010-008
	INS-3.1.2	The visible camera shall have a horizontal field of view greater than 14.5 degrees (127 km from 500 km altitude).	INS-3.1	Analysis	Verified	N/A	VV-R3-2010-009
	INS-4	The R3 satellite shall use a radiation dosimeter to acquire total dose and single event effect data throughout the primary mission.	MSC-1	Inspection	Designed	N/A	
	INS-4.1	The dosimeter shall be placed adjacent to the microbolometer for scientific data correlation.	INS-4	Inspection	Designed	N/A	
	INS-4.2	The dosimeter shall be sensitive to 5 rad of radiation.	INS-4	Testing	Designed	DIME-1 Test User Manual	
	INS-4.3	The dosimeter shall possess a thermistor to measure temperature on the board.	INS-4	Inspection	Verified	N/A	VV-R3-2010-019
	INS-4.4	The dosimeter shall maintain DC isolation of at least 1 MΩ between all supplies and returns and the chassis.	INS-4	Testing	Designed	DIME-1 Test User Manual	
ADC		**Attitude and Orbit Determination and Control Subsystem**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	ADC-1	The R3 satellite shall be three-axis stabilized.	MSC-3,MSC-6	Inspection	Designed	N/A	

	ADC-2	The AODCS shall damp attitude rates within 3 hours of launch vehicle separation.	SAT-3,MD-4	Testing	Designed	Test Plans VI.B.2.a.	
	ADC-3	The AODCS shall be capable of coarse attitude control within 40 deg per axis.	COM-1.4	Testing	In Progress	Test Plans VI.B.2.b.	
	ADC-3.1	The AODCS shall use torque rods for coarse attitude control.	ADC-3	Inspection	Designed	N/A	
	ADC-3.2	The AODCS shall acquire coarse attitude determination within [20] deg per axis.	ADC-3	Testing	In Progress	Test Plans VI.B.2.c.	
	ADC-3.2.1	The AODCS shall use a magnetometer for coarse attitude determination.	ADC-3.2	Inspection	Designed	N/A	
	ADC-3.2.2	The AODCS shall acquire orbit position knowledge at a frequency of 0.5 Hz at an accuracy of 15 m.	MSC-5,ADC-3.2	Testing	Designed	Test Plans VI.B.2.d.	
	ADC-3.2.2.1	The AODCS shall use a GPS unit for orbit position knowledge.	ADC-3.2.2	Inspection	Designed	N/A	
	ADC-3.2.3	The AODCS shall acquire orbit velocity knowledge at a frequency of 0.5 Hz at an accuracy of 0.25 m/s.	MSC-5,ADC-3.2	Testing	Designed	Test Plans VI.B.2.e.	
	ADC-3.2.3.1	The AODCS shall use a GPS unit for orbit velocity knowledge.	ADC-3.2.3	Inspection	Designed	N/A	
	ADC-4	The AODCS shall be capable of fine attitude control within 3.0 deg per axis.	MSC-3,MSC-6	Testing	In Progress	Test Plans VI.B.2.f.	
	ADC-4.1	The AODCS shall use reaction wheels for fine attitude control.	ADC-4	Inspection	Designed	N/A	
	ADC-4.2	The AODCS shall acquire fine attitude determination within [0.5] deg per axis.	ADC-4	Testing	Designed	Test Plans VI.B.2.g.	
	ADC-4.2.1	The AODCS shall use a star tracker for fine attitude determination.	ADC-4.2	Inspection	Verified	N/A	VV-R3-2010-020
	ADC-5	The R3 satellite shall maintain a Sun exclusion zone of [+/- 10 degrees] for the microbolometer and visible camera.	MSC-3,MSC-6	Testing	In Progress	Test Plans VI.B.2.h.	
	ADC-5.1	The AODCS shall use one sun sensor to monitor sun position relative to imaging instruments.	ADC-5	Inspection	Designed	N/A	
CDH		**Command and Data Handling Subsystem**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	CDH-1	The CDH Subsystem shall be capable of interfacing with all subsystem components simultaneously.	MD-3,MD-5.1,SAT-3	Analysis	In Progress	N/A	
	CDH-1.1	The CDH Subsystem shall be operable during all powered modes of the satellite.	CDH-1	Analysis	Designed	N/A	
	CDH-1.2	The CDH Subsystem shall utilize an FPGA for parallel component interfacing.	CDH-1	Inspection	Verified	N/A	VV-R3-2010-003
	CDH-2	The CDH Subsystem shall survive the space environment for the mission duration.	SAT-3	Analysis	Designed	N/A	
	CDH-2.1	The CDH Subsystem shall have a radiation shielding solution capable of enabling component survival.	CDH-2	Analysis, Testing	Verified	Test Plans VI.C.2.a.	VV-R3-2010-014
	CDH-2.2	The CDH Subsystem shall have a processor watchdog to mitigate Single Event Lockups (SELs) and memory scrubbers to mitigate Single Event Upsets (SEUs).	CDH-2	Analysis, Testing	In Progress	Test Plans VI.C.2.b.	
	CDH-3	The CDH Subsystem shall receive thermal images, visible images, radiation data, telemetry, and commands.	MSC-3,MSC-6,MD-2,SAT-3	Analysis	Designed	N/A	
	CDH-4	The CDH Subsystem shall be capable of processing thermal images rapidly.	MSC-4	Analysis	Verified	N/A	VV-R3-2010-015

	CDH-4.1	The CDH Subsystem shall contain a DSP with more than 500 MIPS of processing capability.	CDH-4	Inspection	Verified	N/A	VV-R3-2010-002
	CDH-5	The CDH Subsystem shall temporarily store all data when Ground Station is out of range.	MD-2	Analysis, Testing	Designed	Test Plans VI.C.2.c.	
	CDH-5.1	The CDH Subsystem shall be able to store several image pairs and a couple weeks of telemetry concurrently to bridge communication gaps.	CDH-5	Analysis, Testing	Designed	Test Plans VI.C.2.d.	
COM		**Communications Subsystem**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	COM-1	The COM Subsystem shall provide bi-directional communication between the R3 satellite and the Georgia Tech Ground Station, or other authorized ground stations.	MO-1,MO-3	Testing	Designed	Test Plans VI.D.2.a.	
	COM-1.1	The COM Subsystem shall downlink thermal images, visible images, radiation data, and feature coordinates to the Georgia Tech Ground Station.	COM-1	Testing	Designed	Test Plans VI.D.2.b.	
	COM-1.2	The COM Subsystem shall downlink telemetry to the Georgia Tech Ground Station for satellite monitoring and maintenance.	COM-1	Testing	Designed	Test Plans VI.D.2.c.	
	COM-1.3	The COM Subsystem shall uplink commands only from the Georgia Tech Ground Station, or from authorized ground stations in an emergency situation.	COM-1	Testing	Designed	Test Plans VI.D.2.d.	
	COM-1.4	The COM Subsystem shall maintain a link margin of at least 3 dB.	COM-1	Analysis	Verified	N/A	Mission Plan
	COM-1.4.1	A communication link shall be established with the Georgia Tech Ground Station at the earliest reasonable opportunity (elevation angle = 20 degrees).	COM-1.4	Testing	Designed	Test Plans VI.D.2.e.	
	COM-2	All communication shall abide by ITU and FCC regulations. The R3 team shall obtain the necessary spectrum licenses for operating its space segment radio communication equipment prior to FCR.	NUG	Inspection	Not Met	N/A	
	COM-2.1	The R3 satellite shall possess the ability to cease transmission at any time if it is determined that the satellite is causing harmful interference.	COM-2	Testing	Designed	Test Plans VI.D.2.f.	
EPS		**Electrical Power Subsystem**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	EPS-1	The EPS shall provide for generation, storage, regulation, and control of electrical power to the satellite payloads for the duration of the primary mission.	SAT-3	Inspection	In Progress	N/A	
	EPS-1.1	The EPS shall provide an average power of 87.0 W during sunlight.	EPS-1	Analysis	Designed	N/A	
	EPS-1.1.1	The EPS shall use a shunt to shunt excess power during sunlight power production, up to [TBD] W.	EPS-1.1	Inspection, Testing	In Progress		
	EPS-1.1.2	The EPS shall generate a minimum of 85.0 W-hrs during an average sunlight period using solar cells.	EPS-1.1	Analysis	Designed	N/A	
	EPS-1.2	The EPS shall provide an average power of 49.2 W during eclipse.	EPS-1	Analysis, Testing	Designed		
	EPS-1.3	The EPS shall be capable of providing a peak power of 100 W.	EPS-1	Analysis,Testing	Designed	IT-TP-R3-EPS-013	
	EPS-1.4	The EPS shall provide for charge regulation of the battery cells during sunlight.	EPS-1	Inspection	Designed	N/A	
	EPS-1.5	The EPS shall use a nominal operating bus voltage of 28 ± 1.5 VDC.	EPS-1	Analysis, Testing	Designed		
	EPS-1.6	Harness wires shall conform to [Mil-TBD] standards.	EPS-1	Inspection	In Progress	N/A	
	EPS-1.7	The EPS shall provide interfaces to components with [TBD standards].	EPS-1	Inspection	In Progress	N/A	

	EPS-1.8	The EPS shall be able to recharge the secondary battery within 1 hour under normal sunlight operating conditions.	EPS-1	Inspection, Testing	Designed	IT-TP-R3-EPS-014	
	EPS-1.9	The EPS shall comply with the operational modes of the satellite (Startup, Safe, and Normal Modes), by providing Charging, Checkout, Normal Operations, Detumble, Low Power Safety configurations.	EPS-1	Inspection	In Progress	N/A	
	EPS-1.10	The EPS shall possess an operational storage capacity of 1.86 A-hrs using a secondary battery.	EPS-1	Inspection	Designed	N/A	
	EPS-1.12	The EPS shall provide telemetry, as per Table [TBD] to the C&DH subsystem at a rate of [TBD] Hz.	EPS-1	Testing	In Progress		
	EPS-1.13	EPS -induced ripples shall be less than [TBD] mV peak-to-peak for dc loads and total RMS load shall not exceed [TBD] mV.	EPS-1	Testing	In Progress		
	EPS-1.14	The EPS shall provide loads with regulated power at 5,12,28 V.	EPS-1	Inspection	Verified	N/A	VV-R3-2010-018
	EPS-1.15	The EPS shall be capable of regulating 28,12,5 V at a minimum load of 0,0.1,0.1 A respectively.	EPS-1	Inspection, Testing	Designed	IT-TP-R3-EPS-016	
	EPS-1.16	The EPS shall be capable of regulating 28,12,5 V at a maximum load of 3.75,1.5,.2 A respectively	EPS-1	Inspection, Testing	Designed	IT-TP-R3-EPS-017	
	EPS-2	The EPS shall meet all electrical safety requirements of NSTS 1700.7B, NSTS/ISS 18798 Rev B, and NASA Technical Memorandum 102179.	NUG	Inspection, Testing	In Progress		
	EPS-3	The battery box shall conform to UNP requirements, as listed in Appendix C of the Nanosat-6 User's Guide.	NUG	Inspection	In Progress	N/A	
	EPS-4	The EPS shall interface with the Lightband separation device to initiate charging upon separation from launch vehicle.	NUG	Inspection	In Progress	N/A	
	EPS-5	The EPS shall operate autonomously unless overriden by ground command.	SAT-3	Inspection	In Progress	N/A	
	EPS-5.1	The modes of operation for the EPS shall be controlled by the Mode Controller.	EPS-5	Inspection	In Progress	N/A	
	EPS-5.2	The EPS shall use Power Management Software via the battery charge controller to control battery charging and power management modes.	EPS-5	Inspection	In Progress	N/A	
	EPS-6	The EPS shall utilize three high-side and one ground-side inhibits for the following cases: to prevent power from the solar cells from reaching system loads, to prevent power from the battery from reaching system loads, and to prevent power from the solar cells from reaching the battery.	NUG	Inspection	Designed	N/A	
	EPS-7	The EPS shall utilize fuses for all non-critical loads on the satellite, in accordance with NSTS/ISS 18798 Rev B.	NUG	Inspection	Designed	N/A	
	EPS-8	The EPS shall provide a continuous, electrically conductive path between each major structural component and the launch vehicle interface to serve as a single-point ground.	NUG	Inspection	In Progress	N/A	
	EPS-9	The single point ground shall have a resistance less than 2.5 mΩ.	NUG	Testing	In Progress	IT-TP-R3-EPS-015	
	EPS-10	The EPS shall provide an umbilical cord for [TBD] W of ground power to the bus and for [TBD] A charging current.	NUG	Inspection, Testing	In Progress		
	EPS-11	All metallic surfaces on the interior of battery boxes, including cell retention structures, shall have an electrically non-conductive coating.	NUG	Inspection, Testing	In Progress		
	EPS-12	The EPS shall use Nickel-Cadmium batteries provided by UNP for power storage.	NUG	Inspection	Designed	N/A	
	EPS-12.1	The EPS shall not exceed a depth of discharge of 45% for the battery.	EPS-12	Analysis, Testing	Designed		

STR		Structure	Source	Verification Method	Status	Planned Testing	Verification Source Document
	STR-1	Each material on the Materials List shall comply with requirements for outgassing, corrosion resistance, and flammability resistence as listed in Section 6.3.2 of the Nanosat-6 User's Guide.	NUG	Inspection	Designed	N/A	
	STR-2	Materials selection for the R3 satellite shall comply with MSFC-HDBK-527, Rev. F, the NASA Materials and Processes website, and the NASA Outgassing Data for Selecting Spacecraft Materials website.	NUG	Inspection	Designed	N/A	
	STR-3	Materials with high resistance to stress corrosion cracking (SCC) as listed in MSFC-STD-3029 shall be used where possible. The R3 team will provide a written justification for any materials not found on this list.	NUG	Inspection	Designed	N/A	
	STR-4	Use of non-metallic materials shall be restricted to materials that have a maximum collectable volatile condensable material (CVCM) content of 0.1% or less and a total mass loss (TML) of 1.0% or less. CVCM and TML values should reference the NASA Outgassing Data for Selecting Spacecraft Materials webpage.	NUG	Inspection	Designed	N/A	
	STR-5	Materials with high melting points (ie steels, titanium alloys) shall not be used as structural materials to minimize casualties resulting from reentry debris.	NUG	Inspection	Verified	N/A	VV-R3-2010-023
	STR-6	Any material that can undergo a phase change In the launch or on-orbit environment shall not be used.	NUG	Inspection	Verified	N/A	VV-R3-2010-024
	STR-7	Toxic and/or volatile fluids or gasses shall not be used.	NUG	Inspection	Verified	N/A	
	STR-8	The R3 team shall develop and maintain a Materials List for the protoflight hardware that includes both fabricated and vendor-supplied components. The list shall include metallic materials, non-metallic materials (epoxies, tapes, adhesives, plastic, rubber, composite, glass, lubricants, etc.), and coatings (anodize, plating, iridite, conformal coating, etc.). Payloads are exempt from listing conformal-coated electronics, small mass-produced electronic components, or components on the NASA GSFC Preferred Parts List.	NUG	Inspection	Verified	N/A	VV-R3-2010-030
	STR-9	The integrated R3 system shall be designed to withstand the launch vehicle shock and vibroacoustic environment without failure.	NUG	Testing	Designed	IT-TP-R3-STR-003	
	STR-10	The R3 satellite shall be capable of withstanding an acceleration load factor that correspond to worst-case launch load environments, which is a combination of steady-state, low-frequency, transient loads and high-frequency vibration loads. This design limit load factor is ±20.0 g's on each axis, applied through the center of mass of the analyzed component using the NS-6 coordinate system.	NUG	Analysis, Testing	Designed	N/A	
	STR-11	Fracture control shall be implemented according to NASA-STD-5003, including multiple load paths and structures built with machined (milled) metals with well-understood properties and having low stresses.	NUG	Analysis, Testing	Verified		VV-R3-2010-025
	STR-12	Multiple locking threaded fasteners with backout protection shall be used to join components and assemblies for fastener redundancy and hazard protection.	NUG	Inspection	In Progress	N/A	
	STR-13	As required by JSC 23642, all primary bus structure fasteners shall be #10 or larger and all electronics enclosures shall be fastened to the structure using #8 or larger fasteners.	NUG	Inspection	Under Review	N/A	
	STR-14	R3 shall have a fixed base natural frequency (at the SIP) of >100Hz.	NUG	Analysis, Testing	Under Review	IT-TP-R3-STR-002	VV-R3-2010-026
	STR-15	The center of gravity (CG) for the R3 satellite shall be less than 0.5 cm from the Lightband centerline, including manufacturing tolerances.	NUG	Analysis, Testing	Designed	IT-TP-R3-STR-001	
	STR-16	The center of gravity (CG) for the R3 satellite shall be less than 40 cm above the SIP (+Z axis).	NUG	Analysis, Testing	Designed	IT-TP-R3-STR-001	

	STR-17	All components shall have adequate venting as defined in Section 6.3.3.6 of the Nanosat-6 User's Guide. The R3 satellite shall not have pressure vessels or sealed compartments. Venting analysis shall demonstrate a factor of safety of 2.0.	NUG	Analysis	Verified	N/A	VV-R3-2010-027
	STR-18	The integrated R3 satellite shall be designed to withstand the AFRL Sine Burst Test at 1.2 times the limit loads and a frequency of 0.33 times the lowest natural frequency with no detrimental permanent deformation or ultimate failures.	NUG	Testing	In Progress		
	STR-19	The structure of the R3 satellite shall conform where applicable to the requirements of NSTS 1700-7B.208 and NASA-STD-5003 excepting where specific requirements are defined by the NUG.	NUG	Inspection	In Progress	N/A	
	STR-20	Structural Safety Factors shall meet or exceed 2.0 for yield and 2.6 for ultimate unless otherwise specified by the NUG or other documents listed above.	NUG	Analysis, Testing	Under Review		
	STR-21	Mechanism Load Safety Factors for both operating torque margin and holding torque margin shall meet or exceed 1.0 if based on test and 2.0 if based on analysis.	NUG	Analysis, Testing	Under Review		
	STR-22	The R3 satellite shall have a local flatness of 0.0005 inches per inch at the SIP.	NUG	Testing	Designed		
	STR-23	The primary structure of the R3 satellite shall be machined (milled) and all-metallic.	NUG	Inspection	Designed	N/A	
	STR-24	Pyrotechnic devices/mechanisms shall not be used on the R3 satellite.	NUG	Inspection	Verified	N/A	VV-R3-2010-028
	STR-25	The R3 satellite shall not use welded joints or cast metallic components.	NUG	Inspection	Designed	N/A	
	STR-26	A margin of safety (MS) of zero or greater shall exist for both yield and ultimate stress conditions.	NUG	Analysis, Testing	In Progress		
	STR-27	Bolts used in flight hardware shall be lubricated with space-rated lubricant and torqued to values specified in Table X of MSFC-STD-486B.	NUG	Inspection	In Progress	N/A	
	STR-28	Retaining devices that rely solely on friction as a means of retention (such as, but not limited to: crimps, worm gears, lead screws, and motor detent torques) shall not be used.	NUG	Inspection	Designed	N/A	
	STR-29	The R3 satellite shall accommodate the mounting, placement, and structural support needs of all components of engineering subsystems.	SAT-2	Analysis, Testing	Designed		
	STR-29.1	The R3 satellite shall provide a stable mounting platform for optical instruments on the nadir-pointing face.	INS-1,INS-3,STR-29	Analysis, Testing	Designed		
	STR-29.1.1	Structural deformation shall not permanently compromise the alignment or operation of optical instruments.	STR-29.1	Analysis, Testing	In Progress		
	STR-29.2	The radiation dosimeter shall be mounted close enough to the microbolometer to ensure reasonable scientific correlation.	INS-4,STR-29	Inspection	In Progress	N/A	
	STR-29.3	The star tracker shall be mounted with an unobstructed 180 degree viewing angle to remove the possibility of glints.	ADC-4.2.1,STR-29	Inspection	Designed	N/A	
	STR-29.4	The sun sensor shall have an unobstructed field of view.	ADC-5.1,STR-29	Inspection	In Progress	N/A	
TCS		**Thermal Control Subsystem**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	TCS-1	The thermal monitoring system shall monitor all subsystems with thermistors, using redundancy on critical components, and shall trigger heater activation if minimum temperature is reached.	SAT-12	Testing	In Progress	N/A	
	TCS-2	Thermal Control Subsystem shall maintain all payload elements and subsystems within their operational temperature limits, as listed in Thermal Detailed Design Document.	SAT-3	Analysis	In Progress	N/A	

	TCS-2.1	R3 shall use thermal coatings, surfaces, and insulation where possible to meet component temperature constraints.	TCS-2	Inspection	In Progress	N/A	
	TCS-2.2	R3 shall use resistive heating elements where necessary to meet minimum temperature constraints.	TCS-2	Analysis	In Progress	N/A	
	TCS-3	The TCS shall maintain all components within their survivable temperature range without power for up to [120] minutes during initial startup.	MD-4	Testing	In Progress		
	TCS-4	The TCS shall maintain all components within their operational temperature range during normal mode, and within their survivable temperature range during safe mode if powered off for this mode.	MD-3	Testing	In Progress		
	TCS-5	The TCS shall be capable of maintaining component and satellite thermal requirements within the range of orbits considered for the R3 mission.	MD-1.1,MD-5.2	Analysis	In Progress	N/A	
	TCS-6	The visible camera shall be mounted on a conductive heat sink (ex: aluminum).	SAT-3	Inspection	Designed	N/A	
	TCS-7	Heat from the visible camera case to the bracket shall not be blocked by a non-conductive material (ex: plastic).	SAT-3	Inspection	Designed	N/A	
	TCS-8	Thermal control materials shall only be applied below the forward surface bevel boundary on the Coarse Sun Sensor.	SAT-3	Inspection	In Progress	N/A	
	TCS-9	The Coarse Sun Sensor vent hole shall not be blocked or covered by thermal control materials.	SAT-18	Inspection	In Progress	N/A	
	TCS-10	The processor thermal pad must be soldered to an external ground thermal plane.	SAT-3	Testing	Designed		
FSW		**Flight Software**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	FSW-1	FSW shall interface between all implementations of control logic for all subsystems.	SAT-3	Analysis	In Progress	N/A	
	FSW-1.1	The FSW shall simultaneously handle all control tasks and procedures without the loss of data.	FSW-1	Testing	Designed	Test Plans VI.I.2.a.	
	FSW-1.2	FSW shall be flexible for future design changes of other subsystems.	FSW-1	Testing	In Progress	Test Plans VI.I.2.b.	
	FSW-1.2.1	FSW shall be capable of uploading and replacing updated versions of control logic of other subsystems.	FSW-1.2	Testing	In Progress	Test Plans VI.I.2.c.	
	FSW-2	FSW shall be responsible for gathering telemetry from all subsystems.	SAT-11	Testing	In Progress	Test Plans VI.I.2.d.	
	FSW-2.1	The FSW shall monitor and respond to the status and health of all subsystems.	FSW-2	Testing	Designed	Test Plans VI.I.2.e.	
	FSW-3	FSW shall handle high-level flight modes.	SAT-3	Testing	In Progress	Test Plans VI.I.2.f.	
	FSW-3.1	FSW shall respond to ground commands for changing modes.	FSW-3	Analysis	In Progress	N/A	
	FSW-3.2	FSW shall transition into safe mode operation after flight anomalies designated to trigger safe mode entry.	FSW-3	Testing	Designed	Test Plans VI.I.2.g.	
	FSW-4	FSW shall identify and respond to flight anomalies.	SAT-3	Testing	Designed	Test Plans VI.I.2.h.	
	FSW-4.1	Flight anomalies shall be detected through the use of error codes for individual subsystems.	FSW-4	Analysis	Designed	N/A	
	FSW-5	FSW shall process and respond to uplinked commands from authorized ground stations.	SAT-3	Testing	In Progress	Test Plans VI.I.2.i.	

	FSW-6	FSW shall be designed for modularity.	SAT-3	Analysis	Designed	N/A	
ALG		**Thermal Algorithms**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	ALG-1	On-board image processing algorithms shall be capable of identifying thermal features within images.	MSC-4	Analysis	Verified	N/A	Thermal Algorithms Detailed Design Document
	ALG-1.1	The "blobber" algorithm shall be capable of identifying thermal features having a specified range of intensity, covering a specified range of contiguous pixel area.	ALG-1	Analysis	Verified	N/A	Thermal Algorithms Detailed Design Document
	ALG-1.2	The "edge detection" algorithm shall be capable of detecting areas within thermal images exhibiting thermal gradients within specified ranges.	ALG-1	Analysis	Verified	N/A	Thermal Algorithms Detailed Design Document
	ALG-1.3	The image processing algorithm shall output an error message if the image cannot be processed correctly.	ALG-1	Testing	Verified	N/A	Thermal Algorithms Detailed Design Document
	ALG-1.4	The image processing algorithm to be applied to a particular thermal image shall be identified via ground command.	ALG-1	Analysis	Verified	N/A	Thermal Algorithms Detailed Design Document
	ALG-2	The image processing algorithms shall be capable of calculating all pixel coordinates of the identified thermal features.	MSC-5	Testing	Verified	N/A	Thermal Algorithms Detailed Design Document
	ALG-2.1	The image processing algorithms shall be capable of calculating the boundary pixels of identified thermal features.	ALG-2	Testing	Verified	N/A	Thermal Algorithms Detailed Design Document
	ALG-2.2	The image processing algorithms shall be capable of calculating the centroid pixel of identified thermal features.	ALG-2	Testing	Verified	N/A	Thermal Algorithms Detailed Design Document

MOS		**Mission Operations System**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	MOS-1	MOS shall be capable of full-duplex communications with the R3 satellite.	MO-3	Testing	Designed	IT-TP-R3-MOS-003	
	MOS-1.1	MOS shall provide capability to receive all downlinked science data and R3 satellite telemetry on scheduled passes.	MOS-1	Analysis, Testing	Designed	IT-TP-R3-MOS-001	
	MOS-1.2	MOS shall provide capability to control R3 satellite with uplinked commands during all phases of development and operations.	MOS-1	Analysis, Testing	In Progress	IT-TP-R3-MOS-002	
	MOS-2	MOS shall provide facilities, hardware, and software for development of testbed and support final integrated test plan.	SAT-1	Testing	In Progress		
	MOS-3	MOS shall be capable of archiving all raw data for duration of primary mission [6 months] and provide support for ground processing and dissemination of science data.	MD-1	Testing	Designed		

	MOS-3.1	MOS shall provide commanding and telemetry capturing capability for mission planning and data processing.	MOS-3		In Progress	IT-TP-R3-MOS-004	
	MOS-3.2	MOS shall provide the science team with facilities for archiving and processing science data.	MOS-3	Inspection	Designed	N/A	
	MOS-4	MOS shall provide Public Outreach in the form of workshops and a website.	MSC-7	Inspection	In Progress	N/A	
	MOS-5	MOS shall develop a mission operations system fault tree, and maintain this fault tree throughout operations to minimize single-point failures.	MSC-8	Analysis	In Progress	N/A	
	MOS-6	MOS shall develop and execute a plan to generate command sequence products, validate MOS ability to command during testing and operations phases, and to train flight operations personnel.	SAT-3, MOS-1.2, MOS-2	Inspection	In Progress	N/A	
	MOS-6.1	This plan shall include capability to command protoflight hardware during test phase of project.	MOS-6	Testing	In Progress	IT-TP-R3-MOS-005	
	MOS-6.2	This plan shall include an Operations Test procedure that verifies commanding ability of all sequence products on satellite systems.	MOS-6	Testing	In Progress	IT-TP-R3-MOS-006	
	MOS-6.3	This plan shall include an End to End Test that validates compatibility of tracking station with flight system.	MOS-6	Testing	In Progress	IT-TP-R3-MOS-007	
	MOS-6.4	This plan shall provide the capability to determine, develop, and verify contingency plans to mitigate mission risk.	MOS-6	Testing	In Progress		

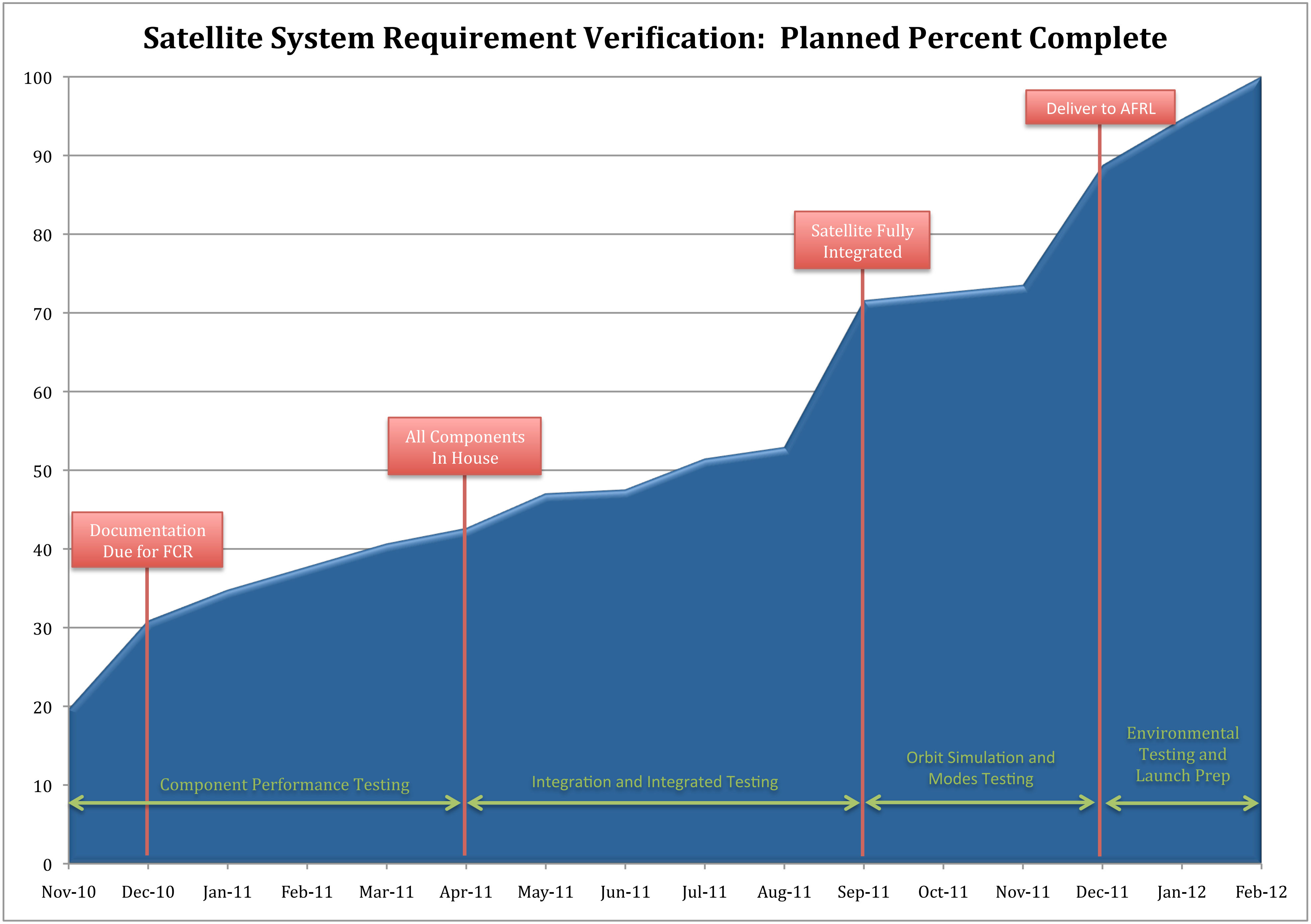
MC		**Mission Control**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	MC-1	MC shall provide capability to receive and monitor engineering telemetry and science data in real time during testing and flight operations.	SAT-11,SAT-12,SAT-13	Testing	In Progress	IT-TP-R3-MC-001	
	MC-1.1	MC shall be able to acquire baseline satellite telemetry within [TBD] minutes after first acquisition of an overpass, and within [TBD] minutes for each subsequent acquisition during that pass.	MC-1	Analysis	In Progress	N/A	
	MC-2	MC shall have the capability to support variable downlink rates during each overpass.	COM-1.4	Analysis	In Progress	N/A	
	MC-3	MC shall provide command and control for all phases of the R3 satellite development and operations.	MOS-1.2	Inspection	In Progress	N/A	
	MC-3.1	MC shall be capable of generating, validating, and uplinking R3 satellite commands and sequences for all mission activities during flight operations.	MC-3	Analysis	In Progress	N/A	
	MC-3.1.1	MC shall verify all command sequence products for resource utilization, constraints, and contingencies before uplink to satellite.	MC-3.1	Analysis	In Progress	N/A	
	MC-3.2	MC shall plan and execute an uplink checkout test after end of inhibited phase.	MC-3	Analysis	In Progress	N/A	
	MC-3.3	MC shall be able to process, store, and uplink [TBD data volume per time period].	MC-3	Testing	In Progress		
	MC-4	MC shall provide uplink encoding, downlink decoding, and delivery of data for necessary operational activities.	MOS-1	Testing	In Progress		
	MC-5	MC shall provide satellite tracking and navigation.	MOS-1	Testing	In Progress		

TRAC		Tracking Station	Source	Verification Method	Status	Planned Testing	Verification Source Document
	TRAC-1	All communication shall abide by ITU and FCC regulations. The R3 team shall obtain the necessary spectrum licenses for operating its space segment radio communication equipment prior to FCR.	NUG	Inspection	In Progress	N/A	
	TRAC-2	The Ground Receiving antenna shall operate in the amateur S-band for downlink.	MOS-1.1	Testing	Designed	Test Plans VI.H.2.a.	
	TRAC-3	The Ground Transmitting antenna shall operate in the amateur UHF band for uplink.	MOS-1.2	Testing	Designed	Test Plans VI.H.2.b.	
	TRAC-4	The Tracking Station shall use computer-controlled rotors to direct the antenna system.	MC-5	Testing	Designed	Test Plans VI.H.2.c.	
	TRAC-4.1	The Tracking Station Main Computer shall control the automated tracking functions for the antenna cluster system.	TRAC-4	Testing	Designed	IT-TP-R3-TRAC-001	IT-TC-R3-TRAC-001
	TRAC-4.2	A designated computer in the Mission Operations Center shall be used to pass command signals to the antenna rotor system.	TRAC-4	Testing	Designed	IT-TP-R3-TRAC-001	IT-TC-R3-TRAC-001
GDS		**Ground Data Systems**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	GDS-1	The Ground Station shall receive mission data and telemetry and post all data to the appropriate workstation computer and server.	MOS-3	Testing	In Progress		
	GDS-2	The Server shall be the central repository for all data collected by the R3 satellite that is successfully transmitted and received by the ground station.	MOS-3	Inspection	In Progress	N/A	
SMP		**Science Mission Planning**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	SMP-1	SMP shall provide specified features for imaging to the mission operations team via the mission planning process.	MOS-1.2	Inspection	Designed	N/A	
	SMP-1.1	Image requests shall be specified via an image request form.	SMP-1	Inspection	In Progress	N/A	
	SMP-2	SMP shall calculate downlink rate based on available data rate for each overpass and intended data collection.	MOS-1.1	Analysis	Designed	N/A	
DA		**Data Analysis**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	DA-1	Data Analysis shall be performed on thermal and visible images and dosimeter readings stored in the Ground Data System by the science team to yield relevant results.	MSC-2, MSC-5	Analysis	In Progress	N/A	
	DA-1.1	Radiation data (TID & SEU) shall be microbolometer performance to assess radiation environment effects on the microbolometer.	DA-1	Analysis	In Progress	N/A	
	DA-1.2	Geolocated thermal features shall be compiled from each set of thermal images and identified.	DA-1	Analysis	In Progress	N/A	
	DA-2	A public website shall be maintained as a database of science mission results for use in the science community, and as a form of public outreach.	MSC-7, MOS-3	Inspection	In Progress	N/A	
	DA-2.1	Raw thermal & visible images, and geolocated features shall be disseminated to the science community via website.	DA-2	Inspection	In Progress	N/A	
	DA-2.2	Ground overflight schedule shall be posted on website to enable science users to request specific thermal or visible images of features of interest.	DA-2, SMP-1	Inspection	In Progress	N/A	
	DA-2.3	Science data and mission information shall be presented on website with intent to educate the public, including K-12 students, on the science mission and satellite operation.	DA-2	Inspection	In Progress	N/A	

TFAC		Test Facilities	Source	Verification Method	Status	Planned Testing	Verification Source Document
	TFAC-1	The protoflight R3 satellite shall be maintained in a Class 100,000 level facility throughout assembly, integration and test.	NUG	Inspection	Verified	N/A	
	TFAC-1.1	All protoflight hardware shall be maintained at the Visibly Clean (VC) level.	TFAC-1	Inspection	Verified	N/A	
	TFAC-2	The clean facility shall be sufficient to assemble and test the R3 satellite.	MOS-2	Testing	Verified		
	TFAC-2.1	Clean room work surfaces shall provide ESD protection to the R3 satellite.	TFAC-2	Inspection	Designed	N/A	
	TFAC-2.2	Clean room work surfaces shall provide secure mounting points compatible with the R3 satellite.	TFAC-2	Inspection	Designed	N/A	
	TFAC-2.3	The R3 satellite shall be lifted securely.	TFAC-2	Inspection	In Progress	N/A	
EGSE		**Electrical Ground Support Equipment**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	EGSE-1	All hazards to the ground equipment shall be identified, controlled, and verified in accordance with NASA Document NSTS 13830, NSTS 1700.7B, and KHB 1700.7C.	NUG	Analysis, Inpsection	In Progress	N/A	
	EGSE-1.1	Documentation shall be presented to the launch organization as listed in NSTS 13830 and KHB 1700.7C.	EGSE-1	Inspection	In Progress	N/A	
	EGSE-2	EGSE shall perform the following functions: battery charging and discharging while satellite is inhibited; inhibit actuation (inhibit/enable satellite); power satellite while satellite is inhibited; support functional testing of satellite, including subsystem level and full "day in the life" testing.	NUG	Testing	In Progress		
	EGSE-2.1	EGSE shall have clearly labeled operations necessary for each function.	EGSE-2	Inspection	In Progress	N/A	
	EGSE-3	EGSE shall have a main power switch with indicator light.	NUG	Inspection	In Progress	N/A	
	EGSE-4	All switches and buttons on EGSE shall be clearly labeled, be sufficiently separated to avoid accidental actuation, and have covers with an automatic-off feature such that when the cover is closed the switch is in the off position.	NUG	Inspection	In Progress	N/A	
	EGSE-5	Circuit protection (fuses or circuit breakers) shall be installed on primary circuits on the load (not ground/return) lines of the EGSE, and shall be readily accessible for inspection, reset, or replacement. Circuit breaker trips and fuse blows shall be readily detectable by visual inspection and circuit protection shall be clearly marked with voltage present and rated amperage.	NUG	Inspection	In Progress	N/A	
	EGSE-6	EGSE shall be self contained and portable.	NUG	Inspection, Testing	In Progress		
	EGSE-7	EGSE shall use standard 120 V, 60 Hz, 3 prong "household" power, preferably through a single plug.	NUG	Inspection, Testing	In Progress		
	EGSE-8	For EGSE batteries, the polarity of battery terminals shall be clearly marked and ventilation shall be provided to ensure concentrations of vapor do not reach 25% of the lower explosion limit.	NUG	Inspection, Analysis, Testing	In Progress		
	EGSE-9	All EGSE shall be designed, fabricated, inspected, and tested in accordance with NFPA 70.	NUG	Inspection, Analysis, Testing	In Progress		
	EGSE-10	All EGSE shall meet the safety requirements of KHB 1700.7C and AFSPC 91-710 Vol 3 Sec 14.2.	NUG	Analysis, Testing	In Progress		

	EGSE-11	EGSE shall be capable of command and control of the satellite without free radiation of RF energy, i.e. through harnessing and/or with RF hats. EGSE shall also be capable of command and control of the satellite through radios and RF. (Note that antenna hats satisfy both of these requirements.) Both communications channels must be available.	NUG	Testing	In Progress		
	EGSE-12	Connectors used in the harnessing between the satellite and the EGSE shall be scoop-proof.	NUG	Testing	In Progress		
	EGSE-13	Battery charging equipment in the EGSE shall be current limited by design and shall provide monitoring and protection to prevent battery damage or failure. The EGSE shall be capable of discharging the battery without enabling the satellite bus/loads.	NUG	Testing	In Progress		
	EGSE-14	All EGSE shall be designed with fuses and diode protection to ensure that failure in ground support equipment will not damage R3 satellite hardware or cause other hazardous conditions.	NUG	Testing	In Progress		
	EGSE-15	EGSE connectors shall not have exposed pins.	NUG	Testing	In Progress		
MGSE		**Mechanical Ground Support Equipment**	**Source**	**Verification Method**	**Status**	**Planned Testing**	**Verification Source Document**
	MGSE-1	The R3 team shall provide MGSE for use in integration operations.	NUG	Inspection	Designed	N/A	
	MGSE-1.1	Tables and workspaces used for integration and testing shall fully secure the R3 satellite.	MGSE-1	Inspection	In Progress	N/A	
	MGSE-1.1.1	The tables and workspaces used for satellite integration shall be moveable and locking.	MGSE-1.1	Inspection	Verified	N/A	
	MGSE-1.1.2	The tabletop stands shall secure the satellite from motion on all axes.	MGSE-1.1	Testing	In Progress		
	MGSE-1.1.2.1	The tabletop stands shall be designed with consideration to the center of gravity of the various satellite configurations.	MGSE-1.1.2	Analysis, Testing	In Progress		
	MGSE-1.2	The tabletop stands shall provide ESD protection to the satellite.	MGSE-1.1	Inspection	Designed	N/A	
	MGSE-2	The tables and workspaces for handling the satellite shall be in a class 100,000 clean room environment.	NUG	Inspection	Verified	N/A	
	MGSE-3	The tabletop stands shall be able to support the R3 satellite with, without, and with only half of the Lightband.	NUG	Analysis, Testing	In Progress		
	MGSE-4	All MGSE shall be designed using a factor of safety of 5.0 for ultimate failure, and be proof loaded to twice the design load.	NUG	Analysis, Testing	In Progress		
	MGSE-4.1	MGSE intended to fit within the shipping container shall be designed using a factor of safety of 5.0 based on expected loads encountered during shipping.	MGSE-4	Analysis, Testing	In Progress		
	MGSE-5	The R3 team shall provide a shipping container that can accommodate the integrated R3 and Lightband for transport from AFRL to the launch site.	NUG	Inspection, Testing	Designed		
	MGSE-5.1	The shipping container shall contain the entire satellite or satellite stack with a clearance of at least [5] cm on all sides.	MGSE-5	Inspection	Designed	N/A	
	MGSE-5.1.1	The shipping container shall enable shipping both with and without the PSC Lightband integrated to the satellite.	MGSE-5.1	Analysis	In Progress	N/A	
	MGSE-5.2	The shipping container shall maintain class 100,000 clean conditions.	MGSE-5	Inspection	Designed	N/A	
	MGSE-5.2.1	The shipping container shall be airtight and posess venting capability.	MGSE-5.2	Inspection, Testing	Designed		
	MGSE-5.3	The shipping container shall be equipped with both fork lift and hanging lift points.	MGSE-5	Inspection	In Progress	N/A	

	MGSE-5.4	The shipping container shall have temperature and humidity sensors in order to monitor the internal environment of the container.	MGSE-5	Inspection	Designed	N/A	
	MGSE-5.5	The shipping container, fully integrated with the satellite, shall have a mass of no more than 90 kg for manual movement from the integration facility to the ground level of the Engineering Science and Mechanics building.	MGSE-5	Inspection	Designed	N/A	
	MGSE-6	The shipping container shall provide ESD protection to the satellite.	NUG	Inspection, Testing	Designed		
	MGSE-6.1	The shipping container shall be fabricated from an electrically non-conductive material.	MGSE-6	Testing	Designed		
	MGSE-6.2	The shipping container shall have a means for grounding the container from an external grounding point before opening container.	MGSE-6	Inspection, Testing	Designed		
	MGSE-7	The shipping container shall measure the shock environment experienced by the R3 satellite during shipping through the use of shock sensors in all 3 axes. Approved shock sensor styles are ball and spring or data-logger type shock sensors – sticker-type shock sensors are not allowed.	NUG	Inspection, Testing	Designed		
	MGSE-7.1	The shock sensors for the shipping container shall be placed on the primary mounting plate/interface to the satellite so as to accurately measure the shock as experienced by the satellite.	MGSE-7	Inspection, Testing	Designed		
	MGSE-8	The shipping container shall provide shock isolation to the satellite.	NUG	Inspection	In Progress	N/A	
	MGSE-8.1	The shipping container mounting interface to the satellite shall be suspended within the shipping container using wire-rope shock isolators.	MGSE-8	Inspection	In Progress	N/A	
	MGSE-9	The R3 team shall provide lifting equipment that can accommodate the integrated R3 satellite and Lightband for movement within the integration facility.	NUG	Inspection, Testing	Designed		
	MGSE-9.1	The lifting equipment shall not contact the satellite at any point other than the designated lifting connection points.	MGSE-9	Testing	Designed		
	MGSE-9.1.1	The lifting equipment shall utilize a spreader frame for load distribution.	MGSE-9.1	Inspection	Designed	N/A	
	MGSE-9.2	The lifting equipment shall provide lifting capability on three axes.	MGSE-9	Inspection	Designed	N/A	
	MGSE-9.2.1	The lifting equipment shall provide 4 lift points for each lifting axis.	MGSE-9.2	Inspection	Designed	N/A	
	MGSE-9.2.2	The lifting points shall be fabricated using low outgassing, space-rated materials.	MGSE-9.2	Inspection	Designed	N/A	
	MGSE-9.3	The lifting equipment shall be capable of a lifting heght sufficient to clear all work surfaces in the integration facility.	MGSE-9	Testing	Designed		
	MGSE-10	The lifting harnesses shall be designed to lift the R3 satellite from a single point above its center of gravity.	NUG	Analysis, Testing	Designed		
	MGSE-11	The lifting equipment shall be constructed out of clean room approved materials and shall not have exposed crevacises or lubricants which could contaminate the clean environment.	NUG	Inspection	In Progress	N/A	
	MGSE-12	The lifting equipment shall be designed such that it will not contact the Lightband during integration and ground handling operations.	NUG	Inspection	Designed	N/A	


Satellite System Requirement Verification: Planned Percent Complete
100
90
80
70
60
50
40
30
20
10
0
Nov-10
Dec-10
Jan-11
Feb-11
Mar-11
Apr-11
May-11
Jun-11
Jul-11
Aug-11
Sep-11
Oct-11
Nov-11
Dec-11
Jan-12
Feb-12
Documentation Due for FCR
All Components In House
Satellite Fully Integrated
Deliver to AFRL
Component Performance Testing
Integration and Integrated Testing
Orbit Simulation and Modes Testing
Environmental Testing and Launch Prep

Appendix B: Templates for I&T Program Documentation

Note that all documents provided in Appendix B are saved as individual full-resolution modifiable files in the electronic appendix CD provided with this document. Also note that in all templates, italicized text denotes fields should be changed accordingly for use (ie: inserting names, dates, revision numbers, etc.).

Reference Template Order (does not correspond to page number)

MATERIALS LIST

SATELLITE NAME

GEORGIA INSTITUTE OF TECHNOLGOY

PROJECT FUNDING ORGANIZATION

Document Number

Revision: #

Date

SIGNATURE PAGE

Prepared by:	*Name* *Title*	Date
Reviewed by:	*Name* *Project Manager*	Date
Reviewed by:	*Name* *Principal Investigator*	Date

REVISIONS

Revision	Description	Date	Approval
-	Initial Release	##/##/####	

MATERIALS LIST

PAYLOAD

Assembly / Part	Material	Specification or Manufacturer's Part #	PPL (Y/N)	TML (%)	CVCM (%)

ADCS

Assembly / Part	Material	Specification or Manufacturer's Part #	PPL (Y/N)	TML (%)	CVCM (%)

CDH

Assembly / Part	Material	Specification or Manufacturer's Part #	PPL (Y/N)	TML (%)	CVCM (%)

COMM

Assembly / Part	Material	Specification or Manufacturer's Part #	PPL (Y/N)	TML (%)	CVCM (%)

EPS

Assembly / Part	**Material**	**Specification or Manufacturer's Part #**	**PPL (Y/N)**	**TML (%)**	**CVCM (%)**

STRUCTURE

Assembly / Part	**Material**	**Specification or Manufacturer's Part #**	**PPL (Y/N)**	**TML (%)**	**CVCM (%)**

THERMAL

Assembly / Part	**Material**	**Specification or Manufacturer's Part #**	**PPL (Y/N)**	**TML (%)**	**CVCM (%)**

Table Acronyms: PPL - Preferred Parts List; TML - Total Mass Loss; CVCM - Collected Volatile Condensable Material

Frequency Tracking Table

In Compliance with Electromagnetic Compatibility Practices

Date Modified: February 10, 2010
By: Sarah McNeese

Operating Frequency Tracking

Subsystem	Component	Operating Voltage (V)			Current Draw (A)			Operating Frequency (Hz)			Validation
		Average	Minimum	Maximum	Average	Minimum	Maximum	Average	Minimum	Maximum	

Frequency Susceptibility

Component A: Frequency (Hz)	Component B: Frequency (Hz)	Component C: Frequency (Hz)	Component D: Frequency (Hz)	Component E: Frequency (Hz)	Component F: Frequency (Hz)

Georgia Tech Center for Space Systems

R^3 Satellite
Flight Hardware Usage Log
LOG-*PROJ-SUB-###*

Component		Home Storage Location	

Check-Out Date & Time	Team Member	Checkout Purpose	Check-In Date & Time	Initials

Georgia Tech Center for Space Systems

Project Name
Receiving Inspection Report
IT-RI-*PROJ-SUB-###*

	Ordered	Shipped	Received	Inspected	
				Accepted	Rejected
Date					
Initials					

Inspection Performed By	
Inspector 1:	
Inspector 2:	

Subsystem	
Component	
Quantity	
Part Number(s)	
Supplier	
Order Number	
Cert. of Comp.	
Materials List	
Additional Documentation	

	Notes/Remarks	Acceptable	Unacceptable	Date	Initials
Appearance					
Dimensions					
Weight					

Is the component damaged?	Yes	No
If Yes, Describe:		
Was the packaging integrity damaged (leakage, denting, etc)?	Yes	No
If Yes, Describe:		
Are any expected/ordered pieces missing?	Yes	No
If Yes, List:		
Does the component require further inspection by a more qualified person?	Yes	No
If Yes, Inspector Recommendation:		

	Approval Signature
Inspector 1	
Inspector 2	
I&T Lead	
Project Systems Engineer	
Project Manager	

*In header, replace italics: "PROJ" with project abbreviation (ie: R3), "SUB" with subsystem abbreviation (ie: COM), "###" with 3-digit component # (001 if this is the 1st component in the subsystem to have a receiving or workmanship inspection, 002 if 2nd, etc). Use same number for all forms related to this component (Test Plan Form, Test Completion Form, etc).

Georgia Tech | Center for Space Systems

Project Name
Workmanship Inspection Report
IT-WI-*PROJ-SUB-###*

	Component Fabricated	Inspected	
		Accepted	Rejected
Date			
Initials			

Inspection Performed By	
Inspector 1:	
Inspector 2:	

Subsystem	
Component	
Quantity	
Materials List	
Additional Documentation	

	Notes/Remarks	Acceptable	Unacceptable	Date	Initials
Appearance					
Dimensions					
Weight					

Does the component have imperfections?	Yes	No
If Yes, Describe:		
Does the component require special handling or storage procedures?	Yes	No
If Yes, Describe:		
Does the component require further inspection by a more qualified person?	Yes	No
If Yes, Inspector Recommendation:		

	Approval Signature
Inspector 1	
Inspector 2	
I&T Lead	
Project Systems Engineer	
Project Manager	

*In header, replace italics: "PROJ" with project abbreviation (ie: R3), "SUB" with subsystem abbreviation (ie: COM), "###" with 3-digit component # (001 if this is the 1st component in the subsystem to have a workmanship or receiving inspection, 002 if 2nd, etc). Use same number for all forms related to this component (Test Plan Form, Test Completion Form, etc).

Test Overview		Approval Initials	
Subsystem		Subsystem Lead	
Component		I&T Lead	
		Project Systems Engineer	
		Project Manager	

Test Description

Test Environment	
1.	
2.	
3.	

Test Support Equipment	
1.	
2.	
3.	
4.	
5.	

Additional Notes:
If you have any specifications about how the test should be performed that is not encompassed by the procedure or the above categories, please list those notes here:

Detailed Test Procedure	
1.	
2.	
3.	
4.	
5.	
6.	
7.	
8.	
9.	

*In header, replace italics: "PROJ" with project abbreviation (ie: R3), "SUB" with subsystem abbreviation (ie: COM), "###" with 3-digit component # previously assigned to this component (either in Receiving Inspection or Workmanship Inspection Form)

Test Overview		Approval Initials	
Subsystem		Subsystem Lead	
Component		I&T Lead	
Test Facilitator 1		Project Systems Engineer	
Test Facilitator 2		Project Manager	
Test Plan Form			

Test Completion Category (Circle One; See Descriptions to Right)	
Noncompliant	The component failed one or more elements of the performance test.
Irregular	An unexpected event occurred, worthy of noting, but it does not render the test a failure.
Successful	The component demonstrated necessary performance capability.

Comments / Test Data Recorded

Noncompliant Results	
Please discuss all failures / noncompliance with test plan:	
Suggested Action Items:	
1.	
2.	

Irregular Results	
Please discuss all irregularities / unexpected events that occurred:	
Suggested Action Items:	
1.	
2.	

Successful Results	
Please list any accomplishments achieved by this test:	
Requirement(s) Verified:	
Operational Test Only:	
Other:	

*In header, replace italics: "PROJ" with project abbreviation (ie: R3), "SUB" with subsystem abbreviation (ie: COM), "###" with 3-digit component # previously assigned to this component (either in Receiving Inspection or Workmanship Inspection Form)

Verification Document
VV-*PROJ-YEAR-###*
Month Day, Year

TO: *Name of PSE*, Project Systems Engineer

FROM: *Your name, Your team/position*

SUBJECT: Requirement *[###]* Verification by *[Inspection or analysis]*

REFERENCES: [1] List any references

Requirement

Reference #	Requirement	Source #
SUB-#.#	*[Enter full requirement text]*	*SUB-#*

Note: Add rows if this inspection or analysis verifies multiple requirements.

Description of Verification Process
Describe the inspection and/or show any analyses done to verify the above requirement, and explain, if necessary, how this work is conclusive to verify it.

Make any comments/notes necessary about the analysis performed, and be sure to discuss any unique or unexpected events occurred.

Conclusion and Final Results
Reiterate the final outcome of the inspection or analysis, and draw conclusions about the finality of the verification process for this requirement.

Appendix C: R^3 Verification, Integration, and Testing Documentation

Appendix C, a compact disc of digital files provided with this document, includes all test documentation from the R3 integration and test program. This file set includes requirement verification documentation, planned testing, performed receiving inspections and workmanship inspections, and completed performance test records. The compact disk also contains Appendix A and B in electronic form for modification and use in future CSS flight projects.