PRELIMINARY DESIGN OF ASUSat 1

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Abstract

ASUSat 1 represents a revolution in space science. The project will be smaller, faster, and cheaper than most comparative missions. The students at Arizona State University are challenged to design, fabricate, and launch a 10-pound-class (4.5 kg) satellite to perform meaningful science in space. Orbital Sciences Corporation is developing the deployment system for this prototype payload from the Pegasus launch vehicle. The project will allow for low-cost measurement of micro particles from interplanetary dust and asteroids in low Earth orbit. The satellite will be placed in a near-polar, sun-synchronous orbit to detect the flux, mass, velocity, trajectory, and temperature of these particles. The timeline for the project is approximately two years from project definition to launch. References [1-7] provide additional background.

Introduction

The thirty-five students working on the satellite design range from high school through Ph.D. level and come from several different science and engineering disciplines. They are divided into the following sub-systems: science, structures, dynamics, communications, power, thermal, commands, ground support equipment (GSE), software & data analysis, and systems integration.

The science team is in charge of designing the space debris detector, and analyzing the returned data. The structure team is designing and building the satellite bus. The dynamics teams' responsibility is acquisition and control of satellite pointing. The communications subsystem is designing the transmitter and receiver, along with the antenna. The power group is in charge of supplying the power to the experiment throughout the mission. The thermal group is providing a model to predict and control the temperatures the satellite will be experiencing. The commands team is designing the system that will function as the control center of the experiment. The GSE team is providing a portable interface between the satellite and the testing environment. The software & data analysis team is in charge of all on-board and ground station programming required to transfer the data into a usable format. The systems integration team is supervising the necessary communication between sub-systems during the experiment via the connectors and cabling harness.

Mission Requirements

The primary objective of ASUSat 1 is to measure the flux, mass, velocity, trajectory, and temperature of asteroidal and interplanetary dust particles in low earth orbit. Particles must be detected from all directions on the leading edge of the satellite within one degree of accuracy. A minimum frontal area of 650 square centimeters must be provided for the detectors. Coverage of the entire earth is necessary, with a position accuracy within two-tenths of a kilometer. A two-year mission lifetime is desired.

In order to accomplish these objectives, ASUSat 1 will be placed in approximately a 450 km, sun synchronous orbit. The exact altitude is not critical, as long as sun synchronous is achieved for power requirement concerns.

For further information on the scientific goals of ASUSat 1, see reference 3.

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Mission Design

Science

The Microparticle Recognition Experiment (MRE) will consist of two sensor arrays, an outer array, on the leading edge of the satellite, and a parabolic sensor array, designed to be incorporated within the body of the satellite. The open end of the parabola will be oriented toward the leading edge of the satellite, thus optimizing particle encounter. This system is passive and operates independently of ground control. Its continuous monitoring and low storage requirements maximize data return for its two year mapping and reconnaissance mission. The MRE's passive design and extremely light weight provide significant advantages over active and ground controlled systems.

The sensors will be constructed of Polyvinylidene fluoride (PVDF). This is a semi-crystalline polymer (50% crystalline, 50% amorphous) that through treatment becomes piezoelectric. When an external mechanical force acts on the film, it is deformed. This results in a change in the surface thickness of the material. The induced charge will be recorded by the supporting systems, and relayed to the ground station.

PVDF film is flexible, extremely light, and has a plastic toughness. It has a good response in temperature ranges from -50° C to 100° C.

A simple schematic of the MRE design is shown in Figure 1. Details can be found in reference 3.

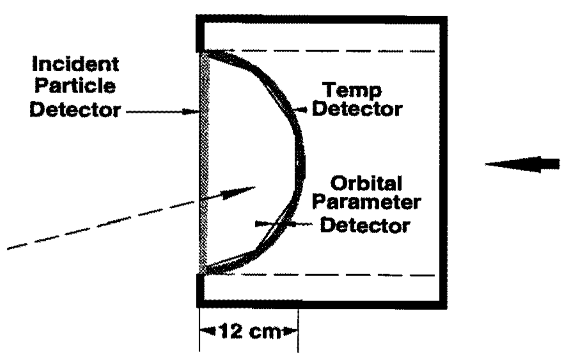

Figure 1

Structure

The structure is required to support all subsystems during all phases of flight. It must also provide adequate ground access for subsystem assembly, and pre-flight tests. A specific goal of this project is to design approximately a 10 lb (4.5 kg) satellite. The structure will be limited to only 25% of this weight. The payload envelope will measure approximately 30 cm by 30 cm. The structure will be required to withstand the accelerations shown in Table 1.

Table 1 : Payload Acceleration Environment [5]

Event	Axial	Horiz.	Vertical
Taxi, Captive Flight	+/- 1.0	+/- 0.5	+/- 2.2
Drop Transient	0.0	+/- 1.0	+/- 4.0
Aerodynamic Pull up	-4.2	+/- 1.5	+/- 3.6
Stage Burn out		+/- 1.2	+/- 1.2
Abort Landing	+/- 0.6	+/- 0.6	+/- 3.5

All loads must meet a yield factor of safety of 1.1 and an ultimate factor of safety of 1.25.

The satellite bus will be a cylinder, 30 cm in diameter by 30 cm in length, with 60 mils thick side walls. The structure itself will weigh approximately 2 pounds. Solar arrays will be rigidly mounted to the structure.

Two types of composite fibers are being considered: PAN and PITCH fibers. In addition, two types of resins, 934 epoxy and a cyanate ester are also being reviewed. The layup will be (0,45,90,135). Figure 2 shows a three dimensional view of the satellite bus. An instrument tray insert will be designed to slide into the cylinder.

Estimated weights of all components are given in Table 2. These estimates show our current weight to be about 2 pounds over the 10 pound goal. Weight reducing techniques will be investigated in the detailed design.

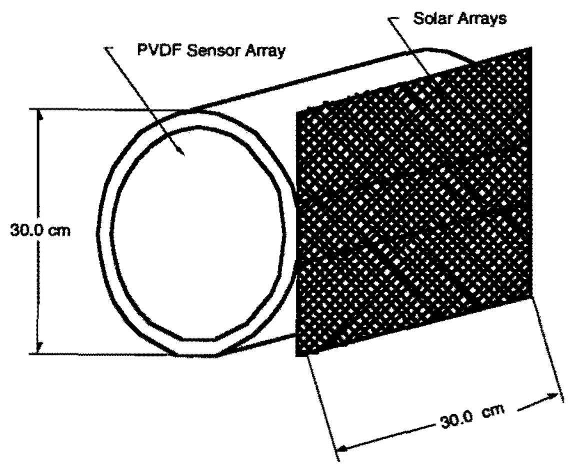

Figure 2

Table 2 : Weight Budget

Weight			4.5 kg = 4500g	(10 lb)
Budget	**Component**	**Dimensions**	**Weight**	
			(grams)	
Structures	Main Structure		681	
	Mounts		50	
	Fasteners		50	
Dynamics	Torque rods		200	
	Magnetometers		100	
	Sun sensor		100	
	Horizon sensor		100	
Power	Batteries		1000	
	Solar arrays		800	
	Bus		100	
Thermal	Thermistors		20	
	Blankets		100	
GSE	Connectors		450	
Science	PAS		450	
	Control Electronics		225	
Data Analysis	Communicator		350	
	Decoder		100	
Commands	Circuit board		450	
		total=	5326	11.73

Dynamics

Dynamics and control is the section of the design concerned with the acquisition process after release from the launch vehicle. This includes location of the Sun and Earth, and proper orientation of the axes of the satellite. The MRE must be maintained in the orbital flight direction, and the solar arrays must be maintained in the direction of the Sun. After acquisition, pointing for the science detector and the solar array must be maintained.

Current pointing requirements dictated by the MRE are ±5° error from the orbital path. Power requirements, in terms of array pointing accuracy, have not been determined. Also, the determination of attitude at the moment when particles impact upon the detector must be made as accurately as possible. The power needed for acquisition is to be determined. The current battery life will help to determine the speed of the acquisition maneuver.

The satellite will be placed in a 450 ± 25 km orbit. This will be a 6 a.m. - 6 p.m. (see Figure 3) sun-synchronous orbit with an inclination of 97°.

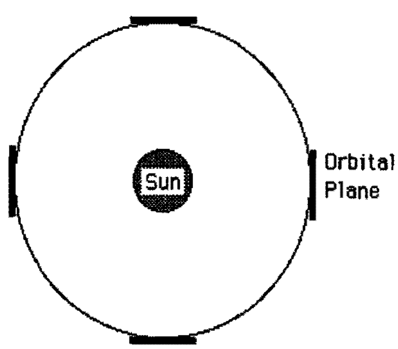

Figure 3

The control system will use passive stabilization (of pitch and roll) through use of a gravity-gradient boom and active control of the yaw axis through the use of an electromagnet. The controls apparatus expected to be used in this system includes: a two-axis magnetometer for yaw and roll angles, a horizon sensor for pitch angle determination, and an electromagnet.

The basic acquisition sequence for ASUSat 1 is shown in Table 3.

Table 3 : Gravity Gradient (GG) Basic Acquisition Sequence

Event	Associated Hardware	Remarks
Release from Pegasus	None	GG stabilized mode
GG Boom Extension	Boom release mechanism, telescoping boom	Pre-stressed metallic tape possible
Yaw axis control	Magnetometers, torquerods	Time to stabilize an issue
Position acquisition	Magnetometer	Proceed to on-orbit control

Components required for control are shown in Table 4.

Table 4 : Stabilization/Electromagnet System

Component	Weight	Size [cm^3]	Power Required (max/ave) mW
Magnetometer	60g	8.9x5.3x2.5	30/20
Horizon Sensor	92g	5.6x4.2x4.7	90/60
Torquerod [1]	78g	1.3x12.7 dia	250/30
Electronic Assemblies	120g	19.9x11.4x.1	50/50
Sun sensors [2]	65g	30x60x60mm	90/60

Total Weight: 480 g

At a 450 km altitude, the orbital period will be 1.57 hours and this is the rate at which our satellite must spin in order to maintain the correct pointing.

In the event that the orbit into which ASUSat 1 enters differs from our selected orbit, precession of the orbit may take us into the shadow of the Earth. The maximum time in this shadow per orbit will be of use to the power team in the determination of back-up power requirements.

For the 450 km orbit, the maximum time in the Earth's shadow per orbit was found to be 4.7 minutes for an inclination of 97°.

Communications

Information will be encoded from all instruments and sensors to enable the data to be transmitted as a telemetry signal. Telemetry will be relayed through an on-board transmitter to the ground station. The possibility of cooperation with AMSAT exists, and is being pursued at this time.

The information to be sent includes sensor measurements of temperatures, currents, voltages, and time stamps. These analog signals will be sampled periodically and converted to digital format. Time stamp information is also important for analyzing the data. The transmitter and antenna options are being considered at this time. A single, omni-directional antenna will most likely be used.

Power

The power system is responsible for providing electrical power to the various subsystems including command and data handling, the MRE, the communications, dynamics, and thermal systems.

The requirements are as follow:

- Provide 10V and 5V to the various subsystems.
- Provide ground access to recharge the batteries.
- Provide enough energy for a two year active lifetime.

Power will be supplied by a solar/battery combination. Initial power will be provided by the batteries, which will be re-charged by the solar cells during the mission. The solar cells will be attached directly to the satellite on the sun facing side. The determining factors involved in the design of the solar array are: the surface area of the solar array, the efficiency of the solar cells, the temperature of the solar cells, the zenith (incident) angle to the sun, the material of the solar cells, the size of the solar cells, the solar array configuration and time of eclipse. The instrumentation and communication needs will determine the power requirements. The power units and control electronics will supply electricity to other subsystems on a common supply line. It will interface with telemetry, tracking, and commands for monitoring, and with ground support equipment for charging.

The power usage of the satellite has been estimated based upon values given by each subsystem. The power requirements (Table 5) are estimated considering the most taxing situation.

The battery will be an Eagle-Picher Industries Nickel-Hydrogen RNHC-6-1, which provides 7-8 volts with a 2-3 ampere current. This will give a power output of 14 - 24 watts. This will further be narrowed by the day to day time of eclipse.

The electrical circuit will include a peak power tracker. This will send a high charge to the battery as soon as the satellite leaves an eclipse. The voltage from the battery and solar array will be bucked down to the lower voltages used by the onboard equipment, by way of a transformer.

Energy density is approximately 1350 W/m^2. With a GaAr/Ge (gallium arsenide on germanium) solar array, we expect to generate about 18 W with an array efficiency of 18% and total array area of 824 cm^2.

Table 5 : Power Requirements

Device/System	Current (A)	Voltage (DC)	Power (W)	Occurrence	Duration (s)	Energy (J)	Amp-Hr
Particle Sensing Film	0.378	10	3.78	1	5880	22226.4	0.62
Modulator	0.4	5	2	4	420	3360	0.19
Demodulator	0.2	5	1	4	420	1680	0.09
Telecommand/Telemetry	0.2	5	1	1	5880	5880	0.33
On-board Computer	0.4	5	2	1	5880	11760	0.65
Sun Sensors	0.02	5	0.1	1	5880	588	0.03
Horizon Sensor	0.018	5	0.09	1	5880	529.2	0.03
Magnetometer	0.006	5	0.03	1	5880	176.4	0.01
Torque Rods	0.05	5	0.25	1	5880	1470	0.08
Electronic Assembly Dyn	0.01	5	0.05	1	5880	294	0.02
Data Storage	0.2	5	1	6	100	600	0.03
Total			11.16	20	36220	47740.8	2.03

Thermal

The purpose of the thermal subsystem is to determine as accurately as possible the thermal loads that will be placed upon the satellite, its structure and components. After having determined the thermal loads, the constraints for each component and temperature control must be determined.

Table 6 shows the estimated thermal loads during specified stages of launch.

Table 6 : Pegasus Thermal Loads [5]

ENVIRONMENT	TEMP RANGE (Degrees C)
VAB/Ground operations	18 - 29
Carrier Mate	18 - 29
L-1011 Taxi	18 - 29
L-1011 Captive Carry	0 - 29
L-1011 Abort/Contingency Site	0 - 29

Thermisters will be placed at strategic locations on and around the instruments for monitoring their operating temperatures. These locations will be determined after computational modeling of the ambient temperatures. The need for passive cooling devices such as thermal blankets and system isolation will also be determined. Possible placement of solar cells and their effects on the instruments will also be a major thermal issue.

Energy absorption, internal heat generation and external heat radiation are currently being modeled.

Commands

The command subsystem is responsible for sequencing all satellite events and recording and storing data until it can be sent as telemetry, as well as for monitoring the experiment environment. The command hardware controls and/or monitors all subsystems.

The command subsystem consists of a micro controller, its associated PROM program memory and a bank of data memory. The data memory is battery backed static RAM.

Control provided includes:

- Data storage which will comply with science requirements of speed and data volume, and be nonvolatile to avoid data loss.
- A centralized control system to facilitate subsystem control (i.e. thermal monitoring/control), monitor and control power to subsystems, control timing of experiment

(active/passive mode) and provide a ground support interface.

- Relays for the following: power to Microparticle Recognition Experiment (MRE), power to deploy gravity gradient stabilization boom, power to control torque rod, and power for communications. All relays provide a specified voltage to be determined by each subsystem.

The micro controller will receive, decode, and execute commands sent through the communication link from the ground station. The micro controller will also control the modes of flight operation, including passive, active, and fail-safe/shutdown. It will monitor operating temperatures and voltages, and commence instrument shutdown if any specified limits are exceeded.

Figure 4 is the preliminary design of the on-board command system.

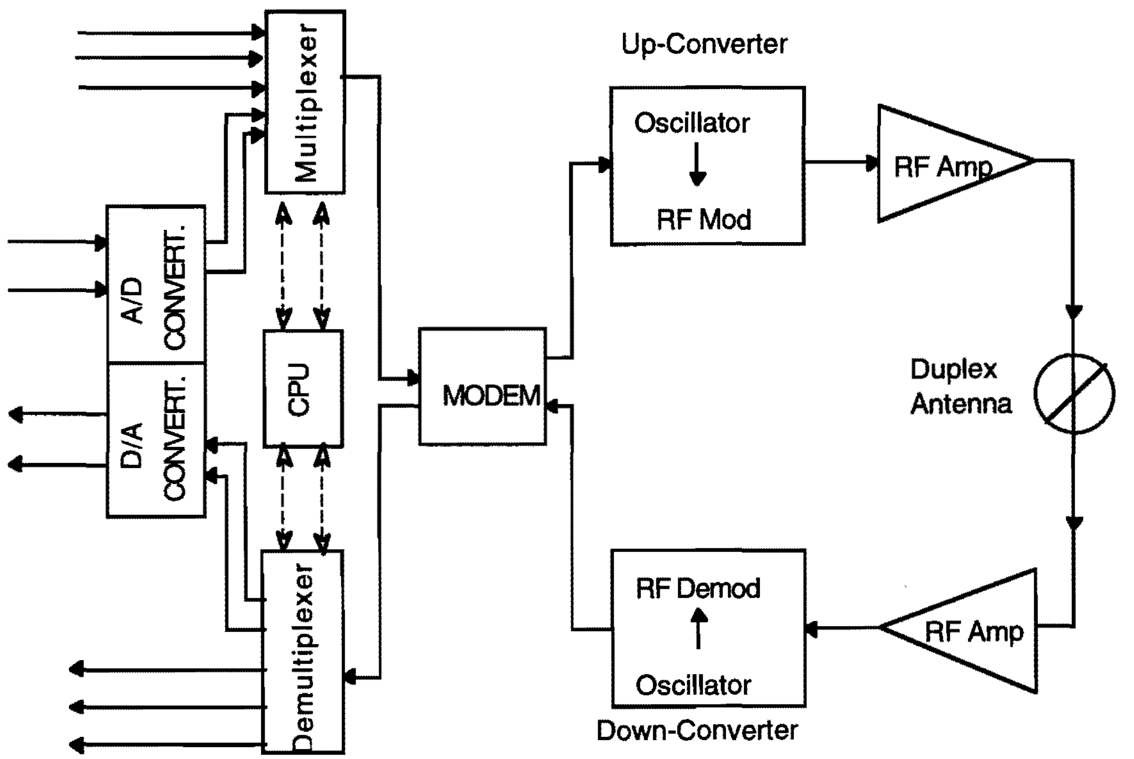

Figure 4

Ground Support Equipment

The ground support equipment is necessary for pre-flight tests to monitor payload activities. These include voltage and current outputs from the instruments and data lines. It will also be used for battery charging and to provide an interface for mission simulations and testing.

GSE requirements are as follows:

- Provide a means of verifying the micro controller is operating within set parameters.
- Provide ground monitoring of the instruments and subsystems.
- Provide battery charging facilities.
- Provide external power to the satellite.
- Provide a simulation of payload release signal.

The GSE subsystem will be a portable, self contained interface between the satellite and the ground based technician/operator. Satellite/GSE connections will be made via the Signal Interface Cable (SIC) and the Power Interface Cable (PIC) connectors on the GSE. Signals will be monitored by using a GSE mounted voltmeter and ammeter or, through the use of external meters via banana jacks.

The GSE will contain a power supply for charging the satellite batteries if necessary. Both the current and voltage of the batteries will be monitored throughout the charging process by the GSE. Whether charging the batteries or not, the GSE will be able to power the satellite externally for mission simulations.

The GSE also includes facilities for telemetry, tracking, and commands during satellite operations, and equipment for data processing during the mission. The ground station will consist of two PC's. One will be used for tracking, while the other receives housekeeping information and experiment data.

Software & Data Analysis

The incoming telemetry at the ground station will be decompressed and separated into the housekeeping data and the actual scientific measurements. The housekeeping data includes the thermal measurements, the various voltages, positioning sensors, and the cpu clock. All information will then be analyzed for critical limits and displayed in an easy to read format individually on a computer monitor.

Science data will be stored in a standard form. Software will be designed to analyze data according to scientific goals of university partners or customers.

The data analysis/software subsystem encompasses all aspects of project computer use. The main areas of design include a ground receiving program, scientific analysis software, and ensuring all aspects of project software systems are compatible.

The ground station module must be able to store incoming telemetry and link to the communications module. The data analysis software must be able to display impact wave forms graphically and calculate object masses, velocities, trajectories, and temperatures. The software must also be flexible enough to accommodate customer dependent science goals.

Tracking information will be obtained from data received from an on-board sun sensor and a horizon sensor. NORAD Level 1 orbital elements will be used to predict position for communications. Positioning will be accurate up to 5 km with elements being updated daily as needed. The tracking software, SPG-4, will be used.

The orbital inclination and ground track may be verified by a global positioning system (GPS). This data would be used to update NORAD elements in order to gain position accuracy to within 0.5 km. The horizon sensor will provide information on pointing accuracy. This information is important for data calibration.

The ground communications program will check incoming telemetry for accuracy, split it into component parts, perform an initial critical value check on housekeeping data, and save all data to disk for later analysis.

Figure 5 is a block diagram giving a general outline of the program's functions.

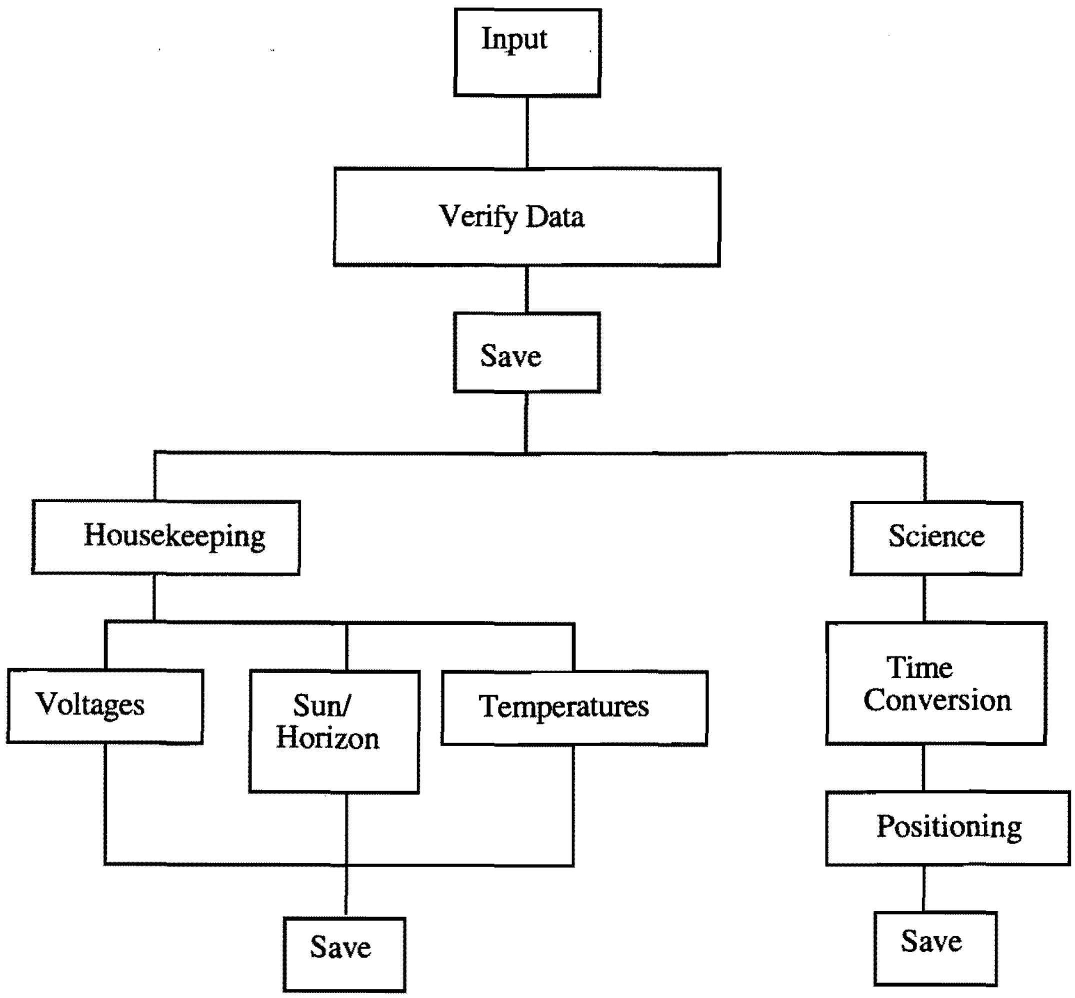
Input
Verify Data
Save
Housekeeping
Science
Voltages
Sun/
Horizon
Temperatures
Time
Conversion
Positioning
Save
Save

Figure 5

Systems Integration

The integration team will develop assembly and test plans and procedures to ensure proper functionality of the satellite and to satisfy the range safety test requirements. They will also define and supply all cabling and connectors necessary to provide the required interfaces between the associated subsystems.

A schematic will be drawn showing the general physical layout of the satellite and the interfaces between the internal components as well as the connections with the GSE. Connectors and cabling will be defined at that point. The connectors to be used in this project will be defined by the individual component requirements. The only requirement on all connectors is that they are capable of performing in a vacuum without off-gassing or otherwise failing. Cabling used in the satellite will be investigated according to industry standard. The only requirement is to ensure functionality in the on-orbit vacuum.

Assembly and testing of the hardware components of the project will also be addressed. Tests will ensure the required functionality. Environmental tests will be performed to satisfy range safety requirements and to validate internal analyses.

A comprehensive assembly plan will be developed that will encompass all phases of construction of the satellite. This plan will include calls for functionality tests and checks at various stages of assembly. Currently, all assembly of flight hardware will take place in the student lab at Arizona State University.

Standardized diagnostics and functionality tests will be developed for each component, each sub-system, and the system as a whole. These tests will be performed during assembly and after any event which might effect the hardware or software. These events include each stage of any environmental test as well as any time the satellite is moved. The functionality tests will be standardized to ensure completeness.

The structure and the complete satellite system will be subjected to a series of tests to determine the ability of the equipment to survive launch conditions and operate in space. These test results will be used for structural and thermal model verification.

The random vibration and shake test will verify that the entire satellite will survive the Pegasus launch environment. The assembled satellite will be instrumented with accelerometers and placed on a shake table at Orbital Sciences Corporation in Chandler, Arizona. The satellite will be subjected to random vibrations at different frequency ranges. Detailed test procedures and equipment requirements are to be determined.

The thermal vacuum test will verify the thermal analyses that have been performed and ensure that the satellite can operate in space within the temperature limits of all components. This test will be performed on the complete, assembled set of equipment at Orbital Sciences Corporation's facilities in Chandler. Detailed test procedures are to be determined.

Conclusions

The successful completion of this project will provide students with hands on experience in a real space program. They will be able to participate from the initial concept, through the design and instrumentation, and on through flight, ground operations, and data recovery. This is an important research endeavor for science and industry that will result in many technical papers and theses. The small satellite is an important training tool for the next generation of space scientists and engineers.

Acknowledgments

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This paper is a conglomeration of results of all students involved with ASUSat 1 from its conception in October, 1993 through May, 1994.

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