SPARTAN LITE: GENESIS OF A SMALL SATELLITE

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Abstract

Goddard Space Flight Center's (GSFC) Space Science Directorate (SSD) came to the Special Payloads Division (SPD) to discuss the idea of extending the mission life of sounding rocket class instruments with a small satellite. The SSD provided a preliminary study that recommended using a five cubic foot Hitchhiker (HH) or Get Away Special (GAS) canister to eject an ASTRO HITCHHIKER satellite into a Space Transportation System (STS) orbit. The SPD interdisciplinary engineering team reviewed this preliminary study and identified restrictions on the proposed satellite's design. The SPD then asked the SSD to provide requirements for various types of instruments. Combining the new requirements and the initial study, the team optimized the services provided to the instruments. The resulting satellite design called SPARTAN LITE accommodates a wider range of instruments than initially proposed and successfully extends the mission life of sounding rocket class instruments. This paper will introduce the reader to ASTRO HITCHHIKER, trace its evolution to SPARTAN LITE, and describe SPARTAN LITE in detail.

Background

The GSFC SSD preliminary ASTRO HITCHHIKER study proposed avionics-grade components for a satellite to support a short mission (1 - 3 months). The candidate instruments required arc-second pointing accuracy. The three-axes stabilized satellite used the Global Positioning System (GPS) with the instrument providing the fine pointing error signals. The study suggested that the satellite's principal moments of inertia be equal and the satellite's center of gravity co-located with the center of pressure. The configuration included body mounted solar cells producing a total of 40 watts of orbital average power for the spacecraft and science instrument. The instrument weight was 21 pounds and the telemetry down link rate, at least 1 Mbps.

The SPD interdisciplinary engineering team included experienced Spartan and Small Explorer (SMEX) Project personnel who provided many recommendations to modify the ASTRO HITCHHIKER design. A decision to relax the moments of inertia limits provided the greatest impact. This relaxation eliminated the need for large ballast weights and relocation of spacecraft components, thus allowing greater instrument weight and volume.

The team also recommended increased electrical power by deploying a portion of the solar array. The deployed solar array allowed one side of the satellite to face away from the sun, easing the thermal design.

The recommendations for these new capabilities included a penalty: the aerodynamic torques on the spacecraft ACS increased considerably. The team compensated for this problem by using larger momentum storage reaction wheels. A SMEX design magnetometer allowed accurate unloading of the wheels with stock torque rods. These design changes increased the instrument weight to 75 pounds with 40 watts orbital average power solely for the instrument. Additionally, the down link rate increased to 2 Mbps. The resulting satellite design called SPARTAN LITE accommodates a wider range of instruments than initially proposed with a low cost access to space.

In designing the vehicle to meet the science needs the team followed a synergistic approach by combining interdisciplinary engineering and scientific needs coupled with an iterative design process. This approach optimized the instrument volume, weight, and power, while minimizing operational and design costs through simplified approaches and existing hardware.

ASTRO HITCHHIKER

Interstel Inc. was tasked by GSFC SSD to develop the ASTRO HITCHHIKER spacecraft for low energy, low cost, three-axis-fine pointing. The satellite concept was low cost and simple. The spacecraft ejects from the shuttle into a low earth orbit with the HH ejection system and becomes a non recoverable free flyer. Depending on the magnitude of the solar wind, the concept of ejecting from a HH canister at shuttle altitudes gave a minimum lifetime of three months before reentry. This lifetime expectancy was quite acceptable for a low cost system. Existing Hitchhiker hardware, canister, Motorized Door Assembly (MDA), and the ejection system were baselined for the study. The HH ejection system is capable of launching a 250 pound satellite that is the full diameter and length of the HH canister. Figure 1 shows SPARTAN LITE in the easily manifested sidewall configuration, which was also specified for the ASTRO HITCHHIKER.

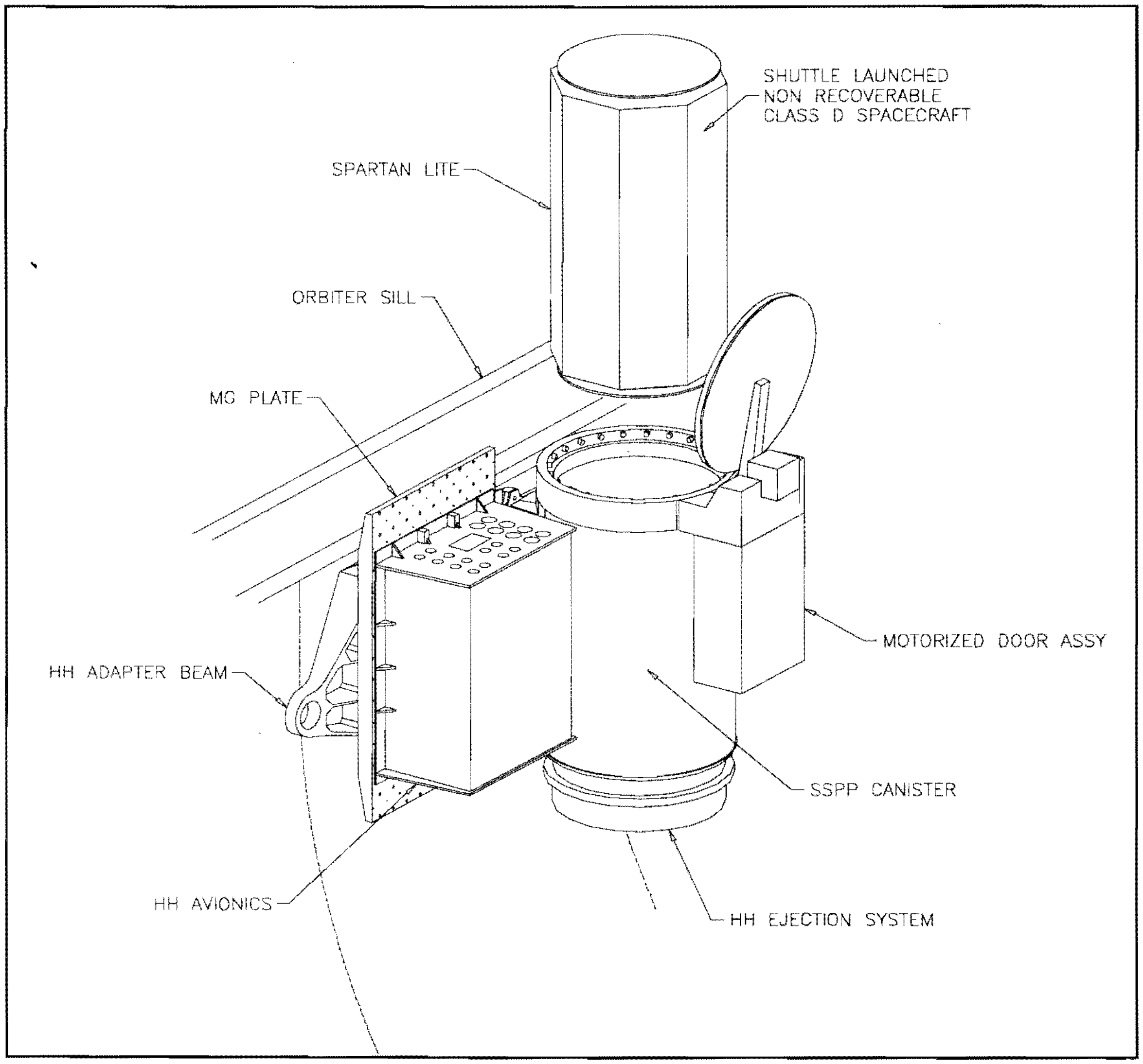

Figure 1 - SPARTAN LITE Sidewall Configuration

Basic Configuration and Attitude Control

For a satellite with a lifetime of three months reducing the cost is imperative. GPS is used for the ACS acquisition, eliminating the need for all sensors except for instrument fine pointing. Figure 2 shows the exterior of the ASTRO HITCHHIKER spacecraft. The simple low energy approach dictated body-mounted solar arrays and the elimination of the two environmental disturbances that are most detrimental to fine pointing: gravity gradient torques and

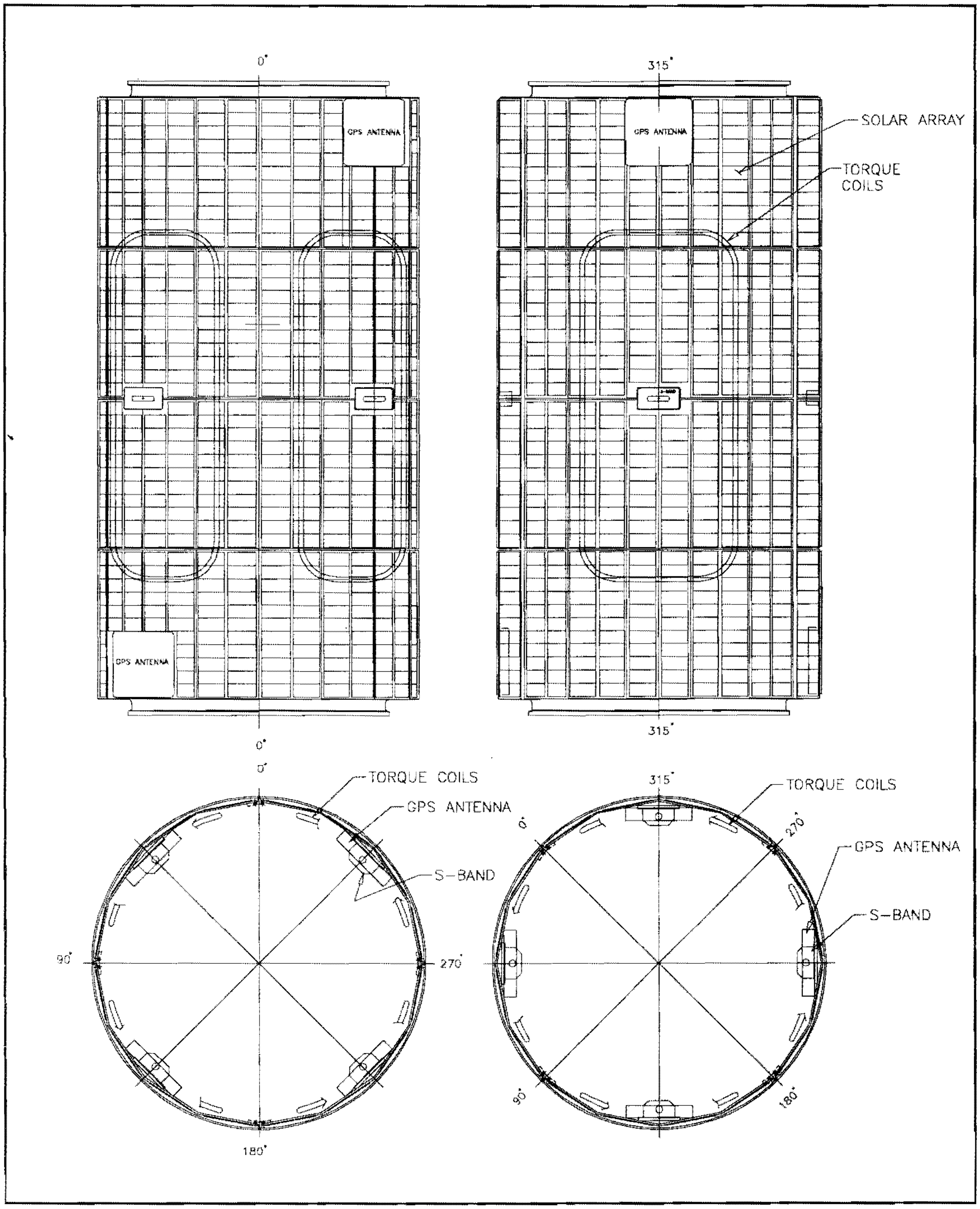

Figure 2 - ASTRO HITCHHIKER Exterior

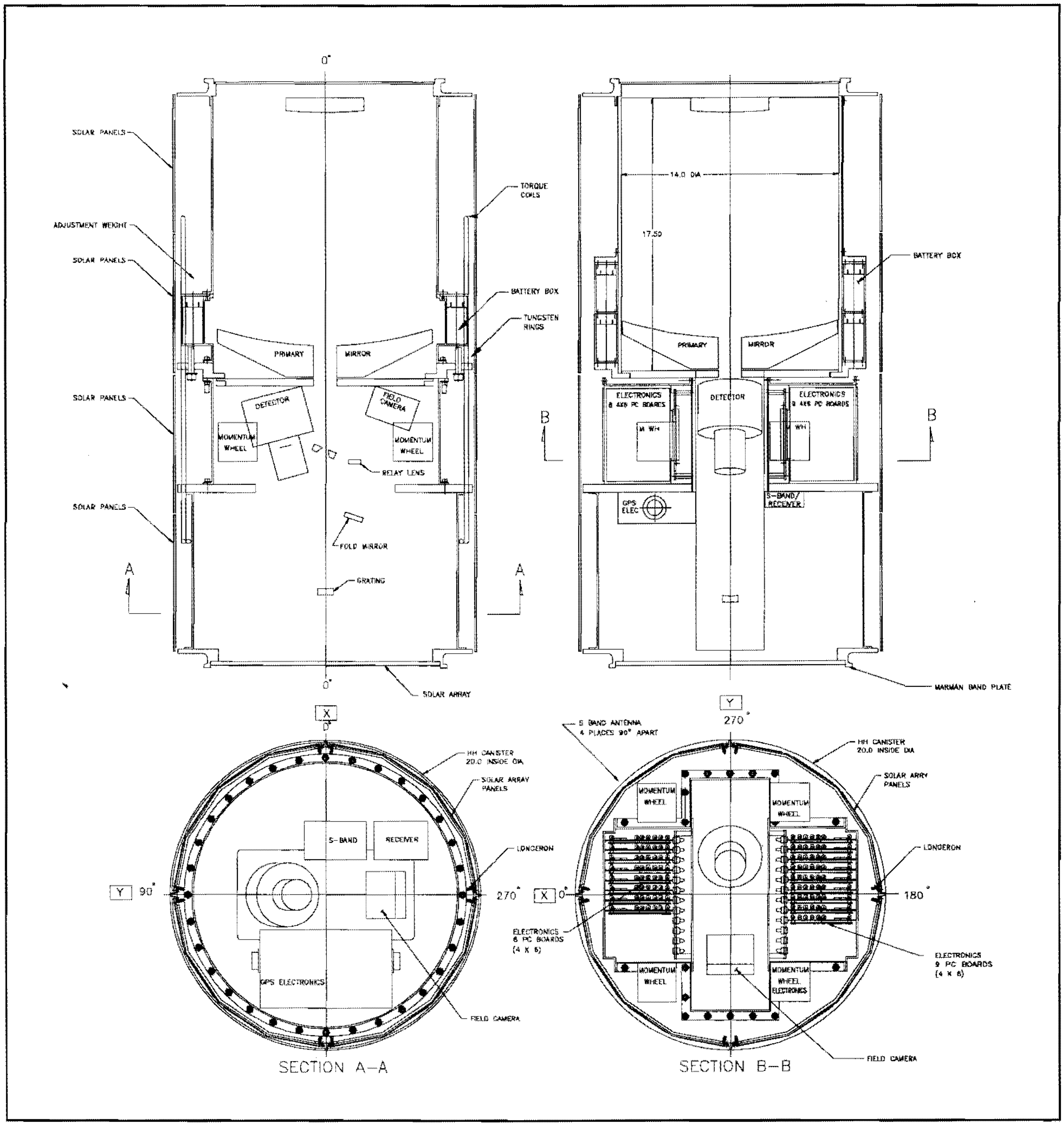

Figure 3 - ASTRO HITCHHIKER Component Placement

aerodynamic torques. The requirement for the ratios between the principal moments of inertia was less than 1%. The center of pressure was required to match the center of mass within a fraction of a millimeter. No flexible appendages were allowed. Since the shape of the ASTRO HITCHHIKER is defined by the inside of the HH canister it is ejected from, the equal moments of inertia required a ring of batteries and a 36 pound tungsten ring around the midpoint of its cylindrical shape. With no deployables and the center of gravity at the center of the cylindrical shape, aero torques are limited to drag produced by orientation of the cylinder to the velocity vector. Small stepper motors that used the rotational inertia of the motor core to function as a reaction wheel would produce enough torque for control. These small, low-power reaction wheels are also capable of achieving fine pointing to the arc-second level. Magnetic torque coils are carried to unload the reaction wheels, and their operation is controlled from a magnetic field model in the spacecraft computer. This approach requires accurate placement of the spacecraft and science instrument components and infers an integrated science instrument. The resulting 250 pound spacecraft has 21 pounds available for the science instrument.

Science instrument

A strawman science mission that needed fine pointing but had no requirements for slewing speeds, target acquisition times, or high data rates was selected for prototype. The mission selected for prototype monitors a class of variable stars known as Cataclysmic Variables (CVs). This mission needed long uninterrupted observing runs (several days) at fixed pointing near the orbit pole. Selected targets avoided earth occultation. The specific instrumentation is a telescope with a UV spectra-photometer and an imaging optical photometer. The data product was continuous low resolution UV spectra combined with continuous high time resolution photometry. No UV light reaches the ground because of atmospheric absorption. None of these targets can be monitored continuously from the ground without weather or coverage interruptions. Figure 3 shows the preliminary concept of the integrated science instrument and satellite bus and the locations of the various components of the ASTRO HITCHHIKER satellite.

Spacecraft Electronics

The Command and Data Handling is performed by a Central Processing Unit (CPU) using a radiation hardened Intel 80386 microprocessor. This processor is fast enough to control all electrical subsystems and interfaces and perform the ACS functions as well.

The power system is composed of a body mounted Gallium Arsenide (GaAs) array feeding a direct energy conversion system. The satellite's maximum external surface area is the interior surface area of the canister. This surface area limits the power generation of the solar arrays to a total available orbital average power of 40 watts. Twenty standard Avionics quality Nickel Cadmium (NiCd) D-type cells per battery are used in this design. Duty-cycling is used to best match the science with the power availability. A power distribution box contains solid state relays to distribute raw bus power to subsystems and experiment.

RF System

An S-Band communications system uses a small antenna size in the 2200 to 2300 MHz frequency range. The small antenna size does not cause variances in the moments of inertia and aerodynamic torques. Four small patch antennas are placed around the circumference of the satellite for transmission and reception. Since position and attitude are known by the GPS, RF switching to the appropriate antenna is performed. A commercially available S-Band transmitter and receiver are used. The internal modules (modulator, series regulator, power amplifier, multipliers and filters) are housed in separate enclosures to maximize RF isolation.

ASTRO HITCHHIKER becomes

SPARTAN LITE

The ASTRO HITCHHIKER has limited science accommodations of 21 lbs and 20 watts available power with the instrument integrated into the spacecraft. Since this capacity greatly limited the spacecraft's appeal, the science community, approached the Special Payloads Division to refine the satellite concept.

SPD put together an interdisciplinary team of personnel from Interstel, Lockheed-Martin, Global Science & Technology, Space Science Directorate scientists and Special Payloads Division engineers to work on the concept. At the weekly meetings, the team identified several areas that needed attention. Increasing the power and weight capabilities of the spacecraft would interest a significantly larger number of science experimenters in this spacecraft. Thermal considerations were also a concern since the full body mounted solar arrays impede the rejection of excessive heat. The GPS antenna placement also required work. The resulting redesign is called SPARTAN LITE.

Deploying the solar arrays increased the power generation of the spacecraft and eased the thermal design. However, this deploying caused non radial symmetry and increased the aerodynamic torques. Additionally, symmetrical moments of inertia were abandoned and the tungsten ballast was removed, providing the science instrument and spacecraft the additional weight previously reserved for the tungsten ballast. Larger reaction wheels were required, since the balanced-moments-of-inertia requirement was dropped and aero torques increased. Since more power was available, larger standard SMEX heritage momentum wheels replaced the stepper motors. Amorphous solar arrays on flexible sheets wrapped around the cylinder that flattened upon deployment appealed to the low cost simple approach but the efficiencies were around 5.5%. Therefore more efficient and costly flat gallium arsenide arrays (18.5%) were chosen. The Nickel Cadmium batteries baselined in the ASTRO HITCHHIKER study were replaced with lightweight Nickel Metal Hydride batteries since their mass was no longer an asset to balance the inertias.

The current mechanical configuration evolved from a design with a circular opening and electronics boxes placed in the middle, to a design with an octagonal opening and electronics boxes placed along the sides. This design defines the internal octagonal cylinder as the experimenter's envelope and places flat solar arrays along the cylinder's exterior. The ACS components are grouped in one end of the cylinder with the spacecraft computer, and the electronics, RF components, and batteries are spread out on the outside of the octagon under the arrays. This configuration leaves the entire center section clear and available for the science instrument. The resulting accommodations are an octagonal cylinder 14 inches across and 28 inches long. Forty watts of orbital average power and 75 lbs of mass are available to the science payload. Over 30 unique science payloads have been identified to date that can use this accommodation.

The ASTRO HITCHHIKER computer was based on a custom software configuration. A substantial saving is realized for the SPARTAN LITE spacecraft which a derivative of the computer that was used on SMEX \ SWAS with elements of the SPARTAN ACS. The primary attitude information comes from GPS with four antennas and a magnetometer. Error signals are required from the science instrument for pointing closer than half a degree. GPS will continue to provide the location of the spacecraft at all times and the spacecraft can download data to the ground station when it is in range. The RF down link was originally intended to transmit at 1 Mbps, but available data compression software and a slightly higher output transmitter have achieved 2 Mbps. The ground station is linked to the Principal Investigator (PI) by the internet to deliver data. Uplink of commands and down link of spacecraft status and housekeeping is accomplished through the GLOBESTAR LEO communications system. Normal uplinks provide new target commands. The spacecraft could contact the PI if it malfunctions through the cellular communication network from nearly any where on earth. The spacecraft is designed for 72 hours of autonomous operation.

SPARTAN LITE in Detail

Mission operations

SPARTAN LITE operations are based on an autonomous spacecraft controlled by the principal investigator. The electrical GSE for the spacecraft is PC based. A version of that simulates the spacecraft interface to instrument electronics is given to the experimenter during instrument development. During integration this equipment acts as a testing platform. The GSE interfaces directly into the spacecraft with a GSE debug port for ground testing. The PI crew learns to control the spacecraft. Upon launch, the GSE goes back to the PI as do the nominal operations of the spacecraft. Data is dumped to a ground station and relayed to the PI by way of the internet. Commands are uplinked to the spacecraft by way of the GLOBESTAR communications system. If the spacecraft goes into safehold and contacts the PI, a SPD crew will assist in getting the spacecraft back on line. The spacecraft is capable of 72 hours of autonomous operation to preclude the necessity of a twenty-four hour seven day a week operation. The PI can work normal hours directing the SC and instrument to record data.

Mechanical

SPARTAN LITE uses different structural configurations depending on the experiment requirements. The primary difference between the solar pointing and celestial pointing configuration is the configuration of the solar array panels after deployment.

The spacecraft structure is a unitized octagonal frame of cast aluminum or aluminum beryllium with cast integral mounts. The structure is a octagonal cylinder with a fourteen inch span between opposing walls (see figure 4). The overall length is 37.35″ with a plate set 28 inches into the structure which separates the instrument volume (4,547 in^3) from the spacecraft bus volume (1,437 in^3). The frame has attachment locations for the 75 lbs of available science instrument weight. Each side of the structure is a 37.35″ x 5.1″ x 0.125″. The solar array panels are 35.75″ x 7.2″ x 0.4″ aluminum honeycomb panels that wrap around the spacecraft structure in the stowed configuration. There is a 1.6 inch gap between the structural panels and the solar array panels. Spacecraft components are mounted inside the spacecraft bus volume and

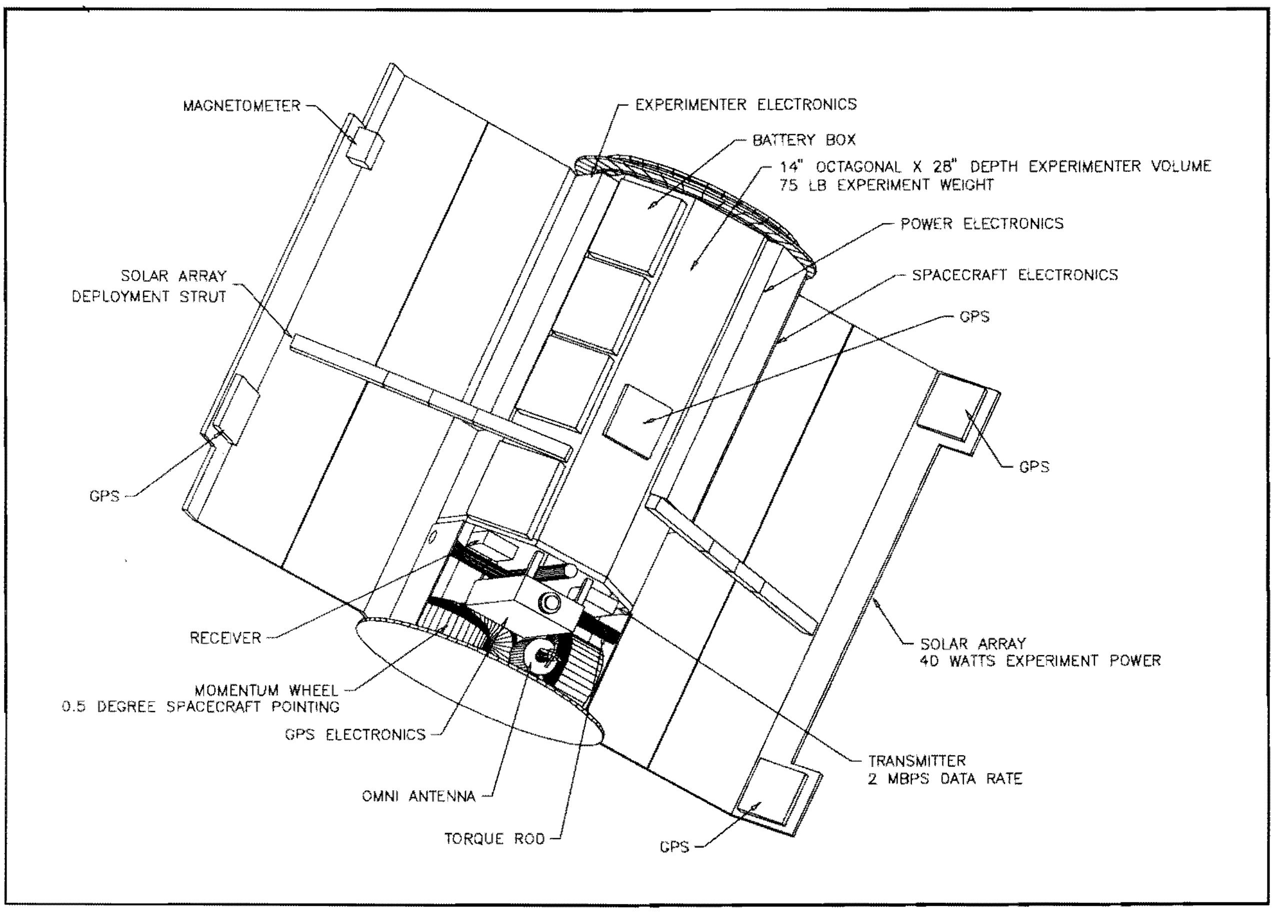

Figure 4 - Celestial Pointing SPARTAN LITE

on the outside of the structural panels in the gap.

The typical instrument configuration would mount the instrument inside its designated volume with its view angle pointed out of the instrument volume. Additionally, one slot on the outside of the structural panels is reserved for the instrument command and data handling unit. This panel could accommodate side viewing ports if required.

Celestial Pointing Configuration

The celestial pointing configuration consists of three body mounted solar array panels and two wings deploy laterally (see figure 4). Each wing consists of two solar array panels with the primary panel longitudinally hinged to the structure and the secondary panel longitudinally hinged to the primary panel. A flexible strut located between the spacecraft structure and the secondary panel provides the deployment force and the structural stability after deployment. One of the body panels is mounted parallel to the plane of the solar wings, and the other two panels are mounted at opposing 45° angles to the plane of the solar wings.

Solar Pointing Configuration

The solar version has all 8 panels hinged at the top of each side with simple torsion springs and spring struts on 4 panels which carry GPS antennas and the magnetometer. The solar array panels deploy longitudinally to form a flower petal formation around the opening of the instrument volume (see figures 5 and 6). A flexible strut located between the spacecraft structure and the primary panel provides the deployment force and the structural stability after deployment. These are retained before deployment by a single restraint wire that is cut by a cable cutter for release.

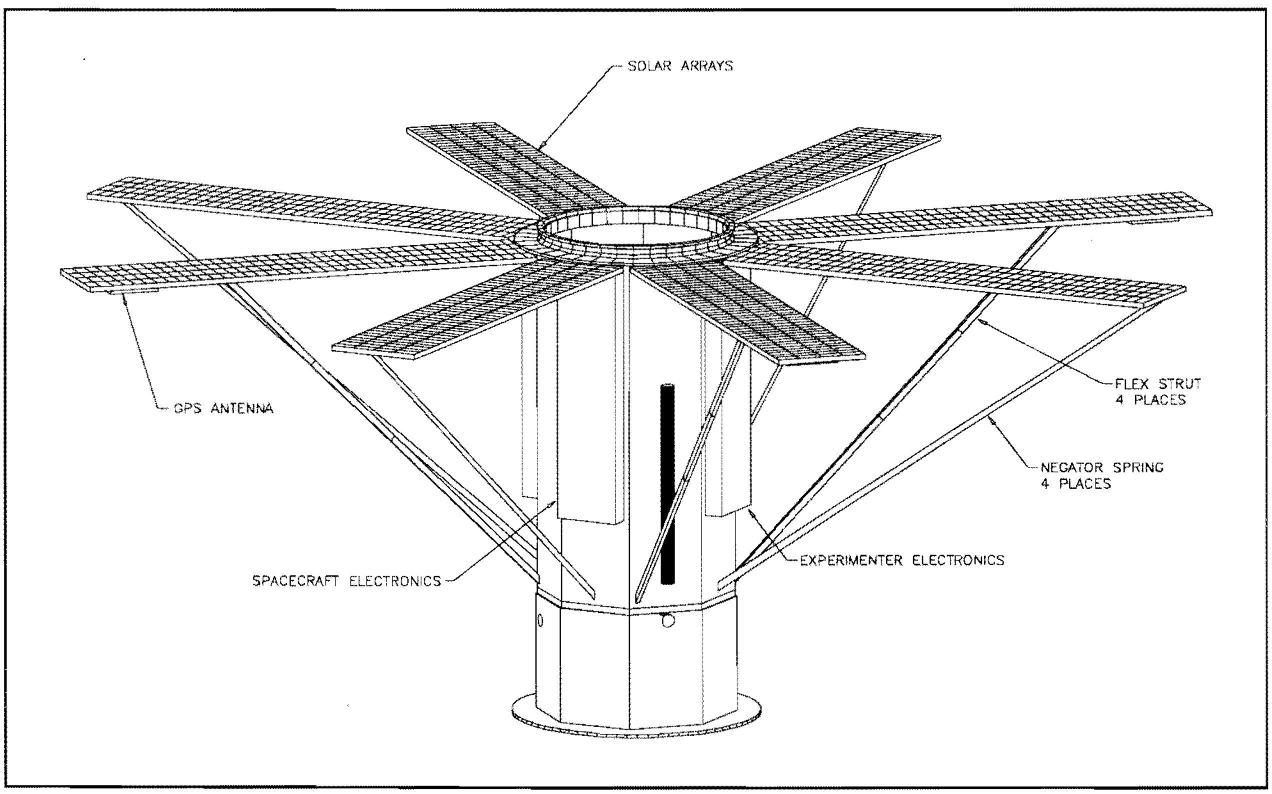

Figure 5 - Solar Pointing SPARTAN LITE (Side 1)

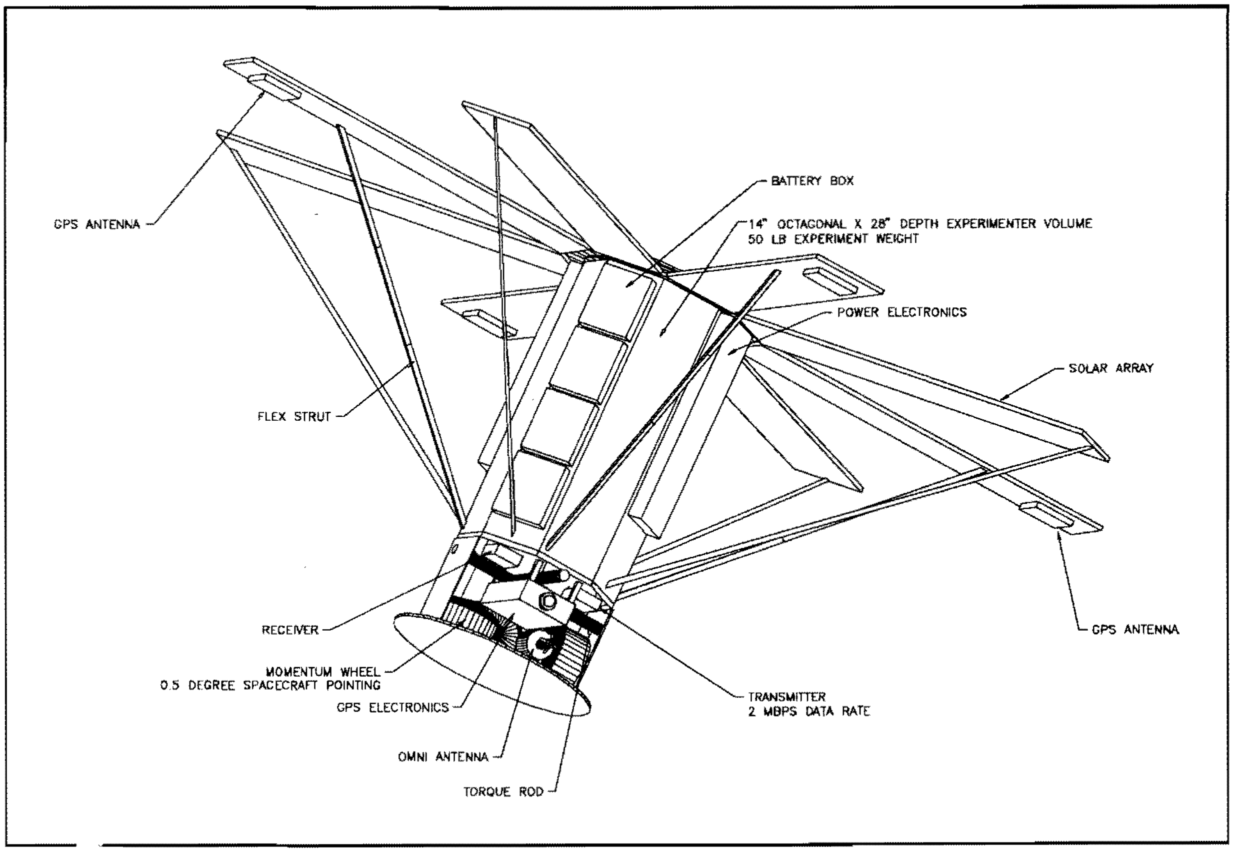

Figure 6 - Solar Pointing SPARTAN LITE (Side 2)

Thermal

SPARTAN LITE primarily uses a passive design in its thermal control system. The survival temperatures are expected to be - 40° to 60° C for the spacecraft and science instrument. Individual subsystems will either radiate heat directly to space or will conduct heat to the spacecraft structure that will radiate the heat to space. The system will be optimized by tailoring internal conductive paths, using multi-layer insulation (MLI) to protect those surfaces requiring heat retention, and using high emittance paint to enhance heat dispersion from surfaces requiring heat radiation.

The spacecraft will use heaters to maintain critical components above their minimum survival temperatures if there is a powered down condition. Any instrument heater power requirements will come from the instrument power budget. All heaters will be controlled by thermostats.

Electronics

The current SPARTAN LITE block diagram is shown in Figure 7. The integrated nature of the spacecraft bus electronics does not permit simple distinct modules. The spacecraft bus electronics is integrated into two electronics boxes called the Control Unit Electronics (CUE) and the Power Control Electronics (PCE). CUE and PCE functions are distributed between the two electronics boxes as space permits.

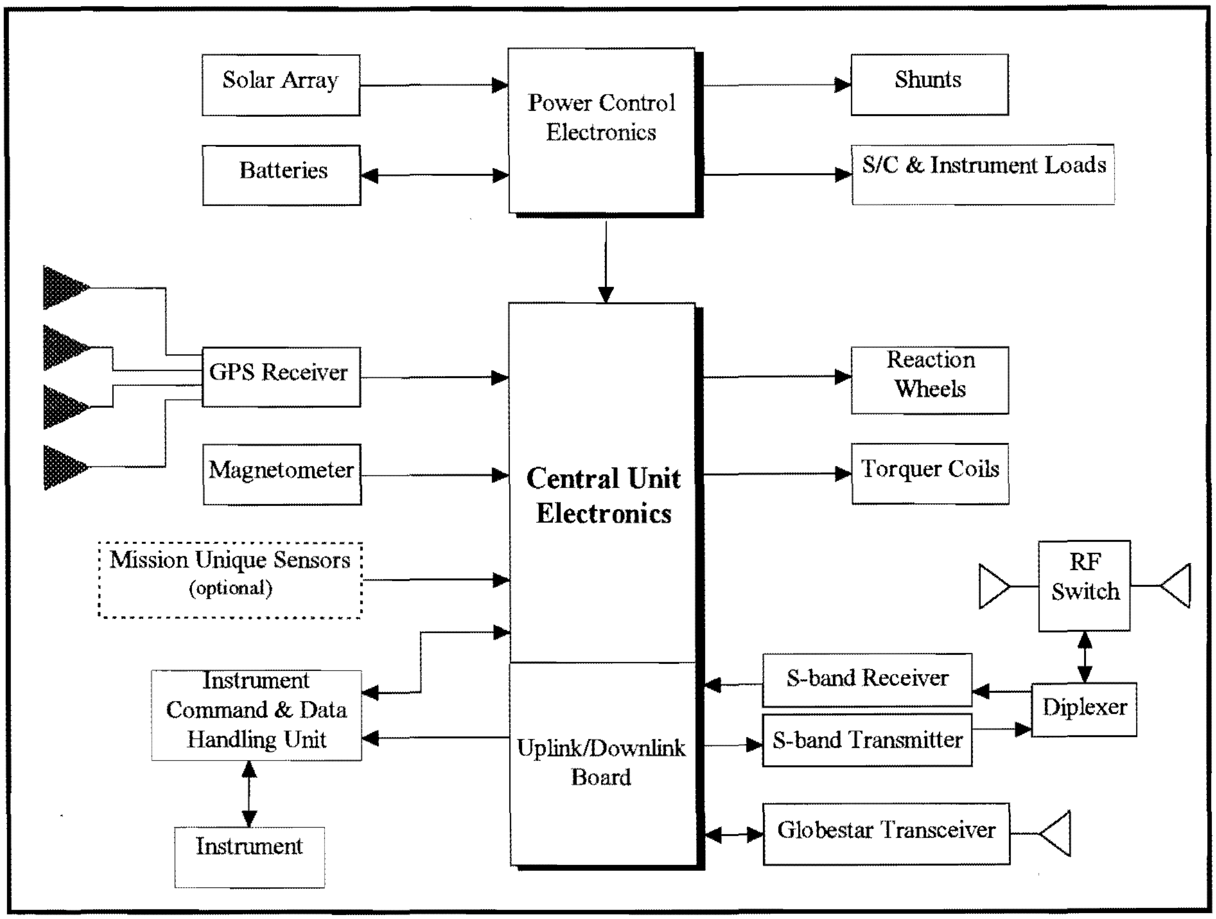

Figure 7 SPARTAN LITE Block Diagram

Command & Data Handling

The CUE is an on-board computer that provides command and control of the spacecraft bus, housekeeping data storage for the spacecraft bus, attitude control, and the interface with the RF communications subsystem (see figure 8). It is a programmable unit designed to provide autonomous operations of the spacecraft. The CUE does not control the instrument, process instrument data, or store instrument data. The PCE controls the gallium arsenide solar arrays, charges the NiMH batteries, and switches power to instruments and spacecraft components. Both of these electronics boxes are based on SMEX and Spartan designs but have been repackaged in a flat configuration to mount on the outside of spacecraft under the solar arrays in their undeployed state. Each unit will fit within a 20.1″ x 5.25″ x 1.6″ volume and will contain six printed circuit boards a processor debug port is available on the CUE for ground testing. This placement puts them on the outside for thermal control and access. The central cavity is therefore free of obstructions for the experiment. The CUE has eight components that perform all the necessary functions to command and control the spacecraft. Management of spacecraft data collection and command distribution is controlled by a 80386 Intel CPU and a 80387 coprocessor, both of which operate at 16 MHz. The CUE uses a 24 bit address bus and an eight bit data bus. The memory module contains 256 kbytes of EPROM for the operating software and 2 Mbytes of RAM for temporary storage of spacecraft data. The digital input/output (I/O) component provides three serial communications ports, 32 TTL I/O channels, 16 +28 VDC I/O channels, and three types of timers. The timers include watchdog, spacecraft clock and time distribution and synchronization. The actuator interface provides

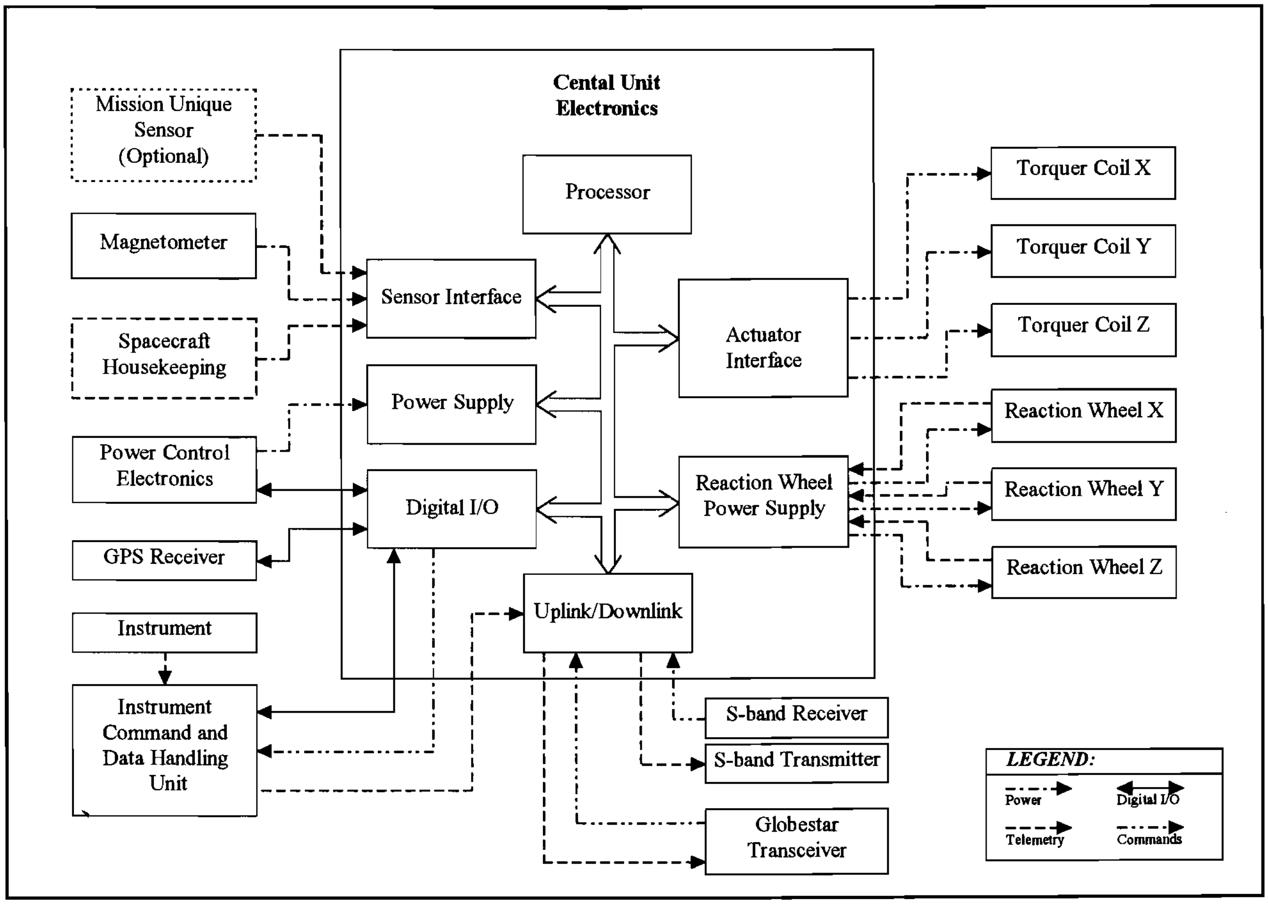

Figure 8 CUE Block Diagram

the torque coil drive voltages and optional gyro servo loops. The sensor interface provides the magnetometer interface, 32 analog input channels, a high resolution analog-to-digital converter, and a programmable gain amplifier. The power conditioning module converts the +28 VDC from the PCE to +5 and ±15 VDC for the spacecraft bus and provides eight switched voltage outputs. The reaction wheel power supply monitors the reaction wheels' electrical outputs and provides the motor current for up to three reaction wheels. The uplink/downlink board accepts downlink data from two sources, the spacecraft or a high speed serial port for downlinking science instrument data. The uplink\downlink board then converts the downlink data to CCSDS format, encodes the downlink data, forwards the downlink data to the transmitter, decodes uplinked commands, and forwards them to the CUE or the instrument command & data handling unit.

Power

The power control electronics (PCE) uses a direct energy transfer architecture. The PCE controls the bus voltage, the battery recharging, the shunts, and the power distribution. The power subsystem supplies unregulated +28 volts to the spacecraft and consists of five components (see figure 9). The solar array panels use GaAs solar cells to generate approximately 150 watts of electrical power. The operational mode provides 40 watts of continuous power to the science instrument. The standby mode provides 10 watts survival power to the science instrument. The Nickel Metal Hydride (NiMH) batteries provide power during eclipse conditions and store up to 4.8 amp-hrs of energy. The digital shunts short the output of a solar array panel during overvoltage conditions. The linear shunts drain power for overvoltage

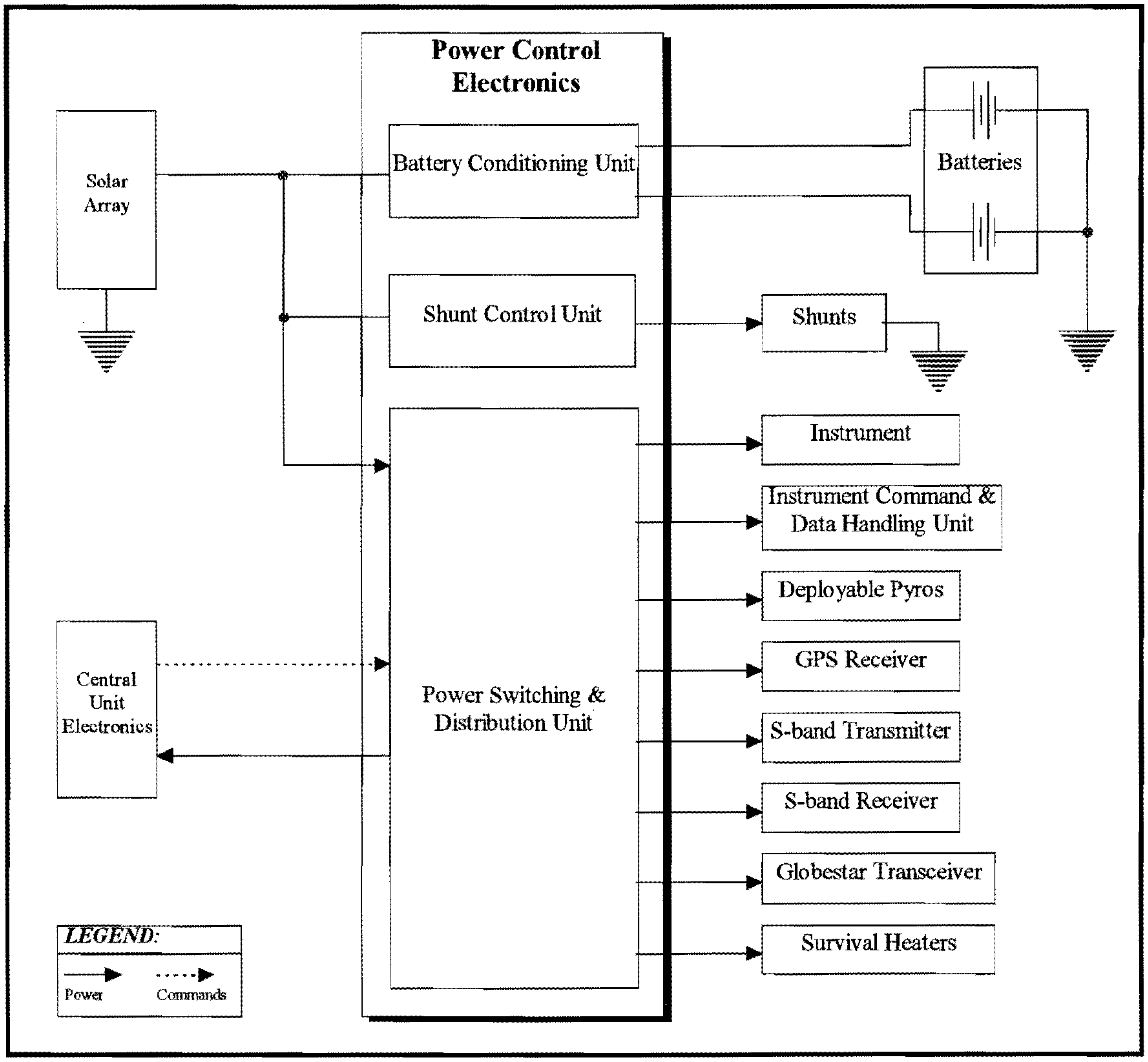

Figure 9 Power Subsystem Block Diagram

conditions which require less correction than shorting a solar panel.

Attitude Control

The attitude control system (ACS) determines the spacecraft position, orbital elements, and attitude (see figure 10). The commercially available GPS receiver will determine the instantaneous position and attitude of the spacecraft. The GPS is used for coarse orbit position and attitude control. Fine pointing is accomplished by the aid of fine error signals from the science instrument or from an instrument star sensor. Three antennas placed around the spacecraft will allow a coarse attitude determination to about 0.5 degrees. This determination is sufficient to locate the target in the optical imaging sensor. The GPS has an attitude accuracy proportional to the relative distance between reception antennas; the greater the distance between antennas, the better the accuracy. Consequently, the greatest allowable separation between the externally mounted GPS antennas is desired.

A magnetometer and array derived solar error signals are baselined in addition since it can provide attitude information when spacecraft obscures the GPS constellation. SPARTAN LITE puts the GPS antennas on strut braced solar arrays to provide the largest possible

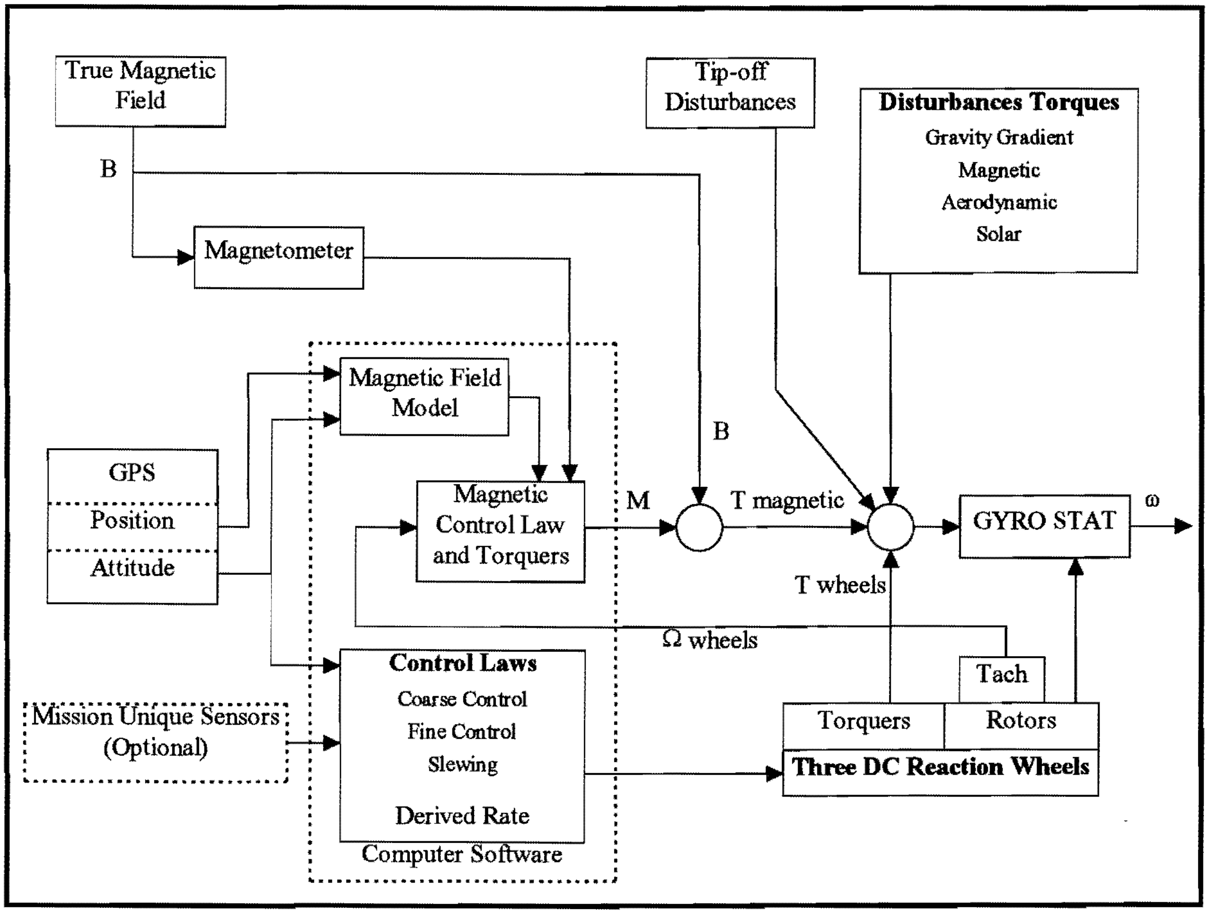

Figure 10 ACS Block Diagram

baseline in a plane. This arrangement also provides a ground plane to reduce multipath signals. The magnetometer, also mounted on an array, is an inexpensive, low power SMEX developed model that provides redundant attitude information for mission success. Three small momentum wheels of SMEX heritage drive the spacecraft with 3 off the shelf 10 AM^2 torque rods to unload the system. These attitude control elements are grouped together at the base of the spacecraft. Fine error signals are the responsibility of the science instrument that provide a fine pointing signals of 5 to 15 Arc seconds for spacecraft pointing. Mission unique sensors for the different science mission are part of the instrument weight and power budget. For solar pointers a SISS is baselined, if the instrument does not itself provide the error signals. For stellar pointers a small, low cost remote head star tracker is baselined. For Earth pointers a low cost horizon scanner is baselined.

RF

SPARTAN LITE uses an S-band communications system (see figure 11) to downlink. The system consists of a transmitter, a receiver, diplexer, RF switch, and two antennas. The transmitter and receiver use BPSK modulation. The diplexer permits simultaneous signal reception and transmission from the same antenna. The RF switch permits the CUE to choose the best antenna for signal transmission and reception. The science instrument will be switched to the standby mode during downlink. A minimum single ground station configuration placed far south will provide data reception. The downlink transmission rate is 2 Mbps and the uplink transmission rate is 2 Kbps. The

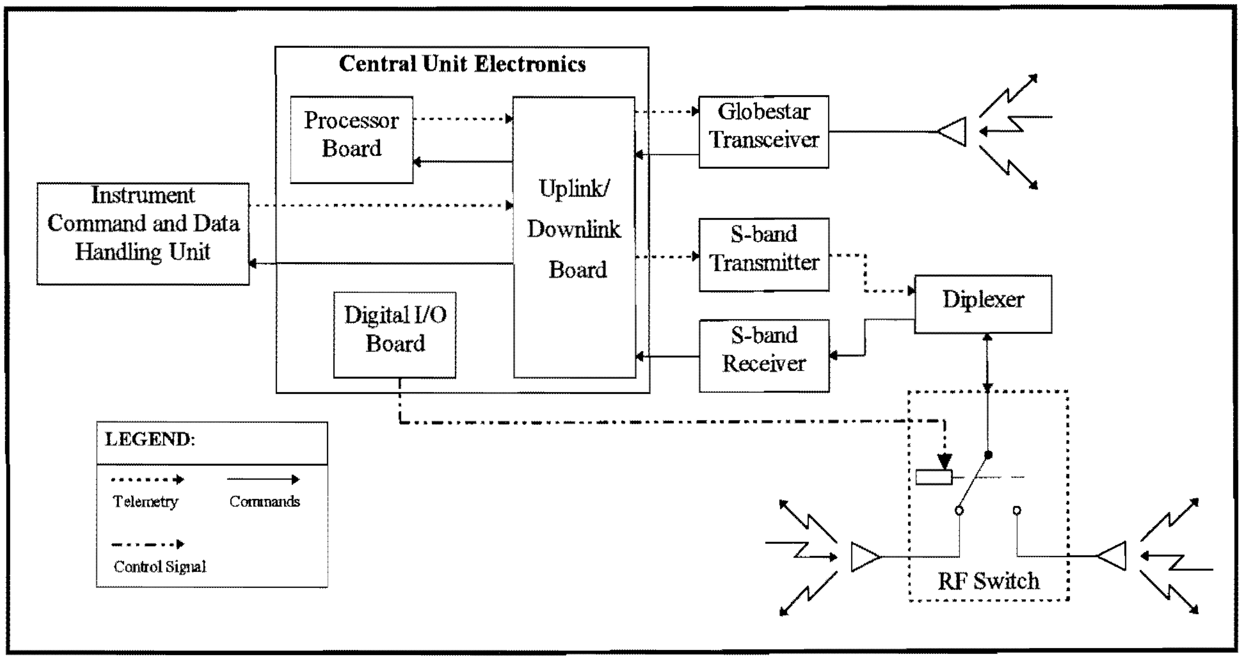

Figure 11 RF Communication Subsystem

maximum potential dump time for one orbital pass is greater than three hundred seconds. Data is sent through the internet to the PI. The PI manages the spacecraft after launch as explained in the operations section.

SPARTAN LITE plans to use the GLOBESTAR L.E.O. communications system for the uplink and downlink of spacecraft housekeeping. The world-wide access provided by GLOBESTAR has distinct advantages. If the S/C goes into safe hold it can contact the PI from anywhere instantaneously and the commands do not have to wait for a pass over the data ground station. If GLOBESTAR works out, the receiver will be removed.

Ground Support Equipment

The SPARTAN LITE Spacecraft will use existing Hitchhiker Ground Handling and support equipment. The existing lifting slings and handling dollies will be used for the transportation and integration of the spacecraft.

An autonomous PC based ground station is baselined for the SPARTAN LITE spacecraft. This ground station will be purchased by the SPD and leased to the PI during the life of the mission.

Weight and Power Budget Summary

Preliminary weight and power budgets usually change when components are delivered and tested. The weight and power budget summaries are provided as a reference only and to substantiate the level of effort that has gone into the design of the SPARTAN LITE spacecraft. The Weight Budget Summary is shown in Table 1. A conservative approach was used when determining the weights of the various components, consequently the values presented here are control weights. Actual weights should be lower for the subsystems and will be substituted for the control weights as the hardware is delivered.

ITEM	TOTAL WEIGHT
SPACECRAFT	139.69
Structure	52.00
Attitude Control System	26.11
Command & Data Handling	20
Power	37.00
RF Communications	7.28
Thermal	2.15
Science Instrument	75.00
TOTAL	219.54
HH CANISTER CAPABILITY	250.00
CONTINGENCY 12.2%	30.46

Table 1 Weight Budget Summary

The Power Budget Summary is shown in Table 2. The assumptions for the power generation include using the median sun (1353 watts/sq m), an orbital altitude of 190 nautical miles, the end of solar cell life efficiencies, and a normal to sun array angle. This realistic approach to the power generation allows accurate determination of duty cycling and shunting requirements.

ITEM	ORBITAL AVERAGE POWER (Watts)
SPACECRAFT BUS	33.04
Attitude Control System	10.30
Central Unit Electronics	11.91
Power Control Electronics/Wiring	5.76
RF Communications	3.21
Thermal Subsystem	1.85
SCIENCE INSTRUMENT	39.27
TOTAL POWER REQUIRED	79.61
TOTAL POWER AVAILABLE	86.04
CONTINGENCY 7.48%	6.44

Table 2 Power Budget Summary

Current configurations

Figure 1 Shows the Shuttle launched configuration. The design of the SPARTAN LITE spacecraft is compact and versatile enough to be launched in a group of three as UELV Primary (see figure 12) or in a more common single configuration as a UELV secondary (see figure 13). These options allow non STS (ie polar) orbits to be flown. The ejection system for these options is under development.

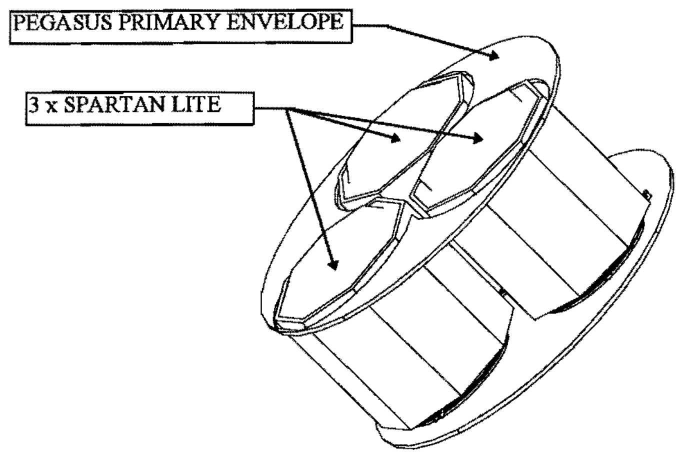

Figure 12 - UELV Primary

Conclusion

This paper introduced the reader to the key concepts behind the ASTRO HITCHHIKER. The most important of which was the simplification of the attitude control system by diminishing the effects of gravity gradient and aerodynamic torques. Diminishing these torques reduced the size of the reaction wheels and magnetic torque coils. The use of off the shelf avionics grade components for the RF, GPS, and reaction wheels was well suited for the class D nature of the spacecraft.

These key concepts in ASTRO HITCHHIKER provided an excellent foundation for SPARTAN LITE. The use of avionics grade components for the class D missions was continued. The power generation increased by a factor of 2 for the science instrument, while the allowable weight for the science instrument increased by a factor

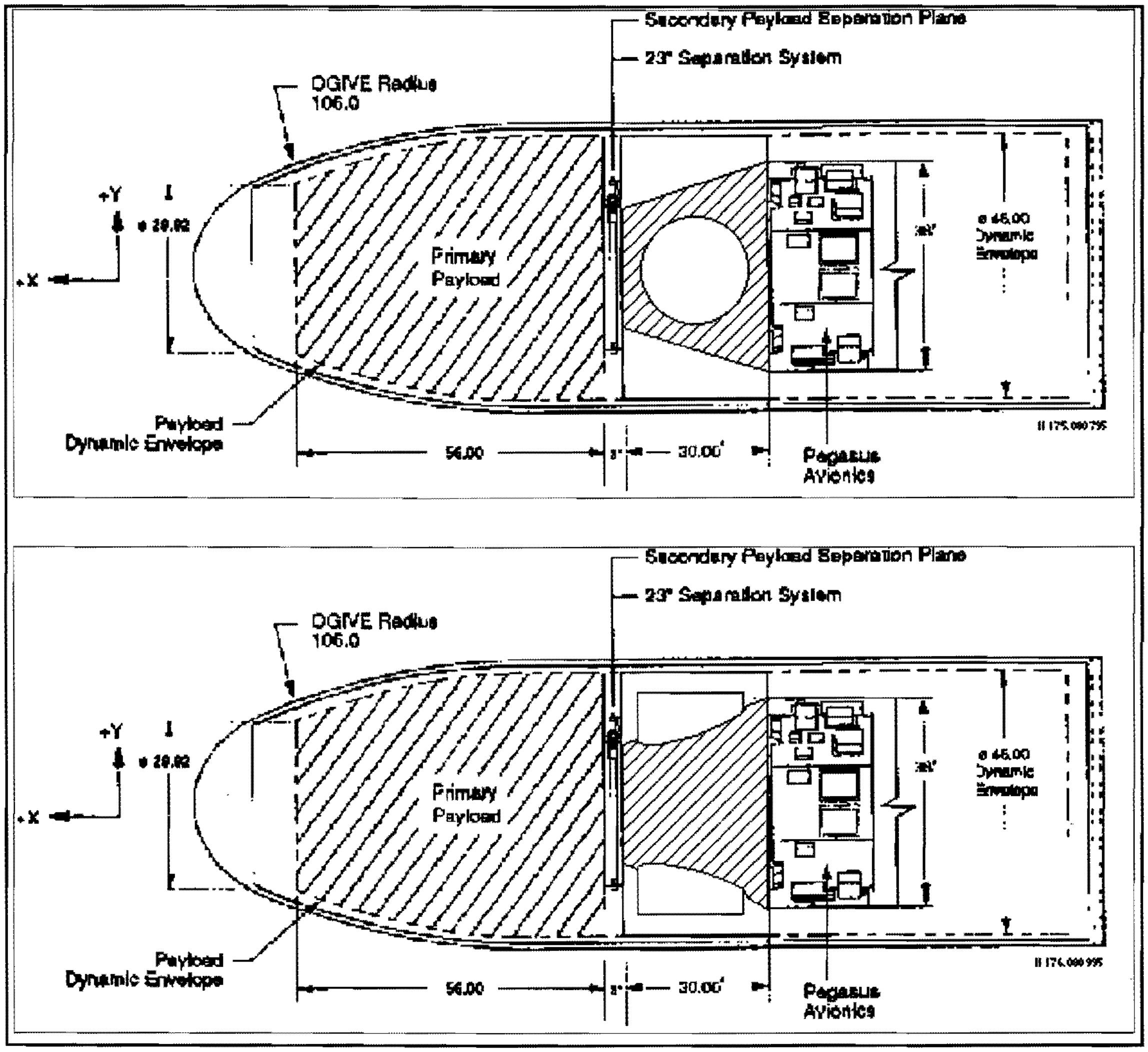

Figure 13 - UELV Secondary

of over 3. The data downlink rate increased by a factor of 2, while the cost for all of these improvements was kept down.

The objectives of the SPARTAN LITE project are to compliment the current SPARTAN and HITCHHIKER program by providing a new small class of satellites between sounding rockets and SMEX/MIDEX. Cost effective use of existing designs, equipment, and resources of the Special Payloads Division (SPD) allow a useful low cost technology demonstrator and experiment proof-of-concept tool. This class D shuttle launched 3-axis stabilized spacecraft has an expected mission lifetime of three months. Numerous different candidate missions have been explored as possible experiments and their requirements appear to match SPARTAN LITEs capabilities. The major classes of experiments include solar, celestial, and Nadir pointing experiments.

The SPARTAN LITE program is currently moving into Phase B which is the definition phase. The Preliminary Design Review (PDR) for SPARTAN LITE is currently scheduled for March 1996. The PDR will be for one of the three candidate configurations. The first flight of the SPARTAN LITE spacecraft will act as a proof of concept flight for the new components on the spacecraft.

Authors Biographies

Evan K. Goldstein received his Bachelors of Science in mechanical engineering from the University of Maryland. His Masters of Science in Mechanical Engineering from The George Washington University, and is currently pursuing a Ph.D. Evan is employed with Interstel Inc. as a mechanical systems engineer. Evan was involved in the launch of BREMSAT. Evan currently is involved in other small satellite projects that are using the SPD Hitchhiker system including: SEDSAT, SLOSHSAT, and MIGHTYSAT.

Stewart C. Meyers received a BSAE from Northrop Institute of Technology. He has worked for the GSFC in the old Sounding Rocket Division- now the Special Payloads Division- for over 30 years. He has personally designed built and flown many payloads and spacecraft form sounding rockets and balloons to Spartans and Hitchhikers. He is now the head of the Structures and Mechanical Systems Branch. This team has designed and flown PEGSAT and SAMPEX ; has FAST, SWAS, and PEGSTEP waiting for a ride, and has TRACE and WIRE under design.

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Mr. Thomas Guilmette	Interstel
Mr. Gregory Waldo	LMM
Mr. William Dorsey	LMM
Mr. Linkh Parikh	GST