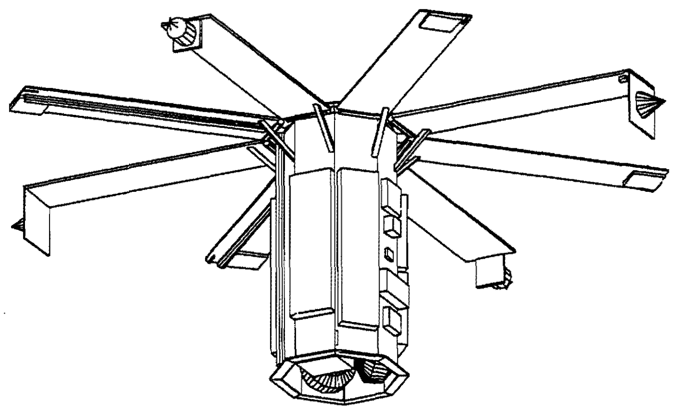
SPARTAN LITE STATUS REPORT

S. C. Meyers

Special Payloads Division
Goddard Space Flight Center
Greenbelt, MD



Abstract

Spartan Lite is an inexpensive, three axis, fine pointer with mission unique pointing capabilities to arc seconds. It is a low cost, class D, single string, spacecraft designed to be launched from a Hitchhiker eject system on the Shuttle or as multiple payloads on a Pegasus class vehicle. The science payload is a cylinder 14 inches in diameter, 28 inches long, and weighing 75 pounds with 40 watts of experiment power. The spacecraft electronics wrap around the payload. Solar arrays are rigidly deployed at one end in a "Flower Petal" configuration. Attitude control actuators and sensors and antennas are mounted on the arrays. Existing hardware designs and software have been used where possible to keep the costs down. The highly autonomous spacecraft is operated in the 'Principal Investigator' mode with simplified mission operations. Fixed science accommodations can support different instruments capable of doing solar, celestial, or earth pointing missions. The Shuttle launched solar version had its Preliminary Design Review in May 1996. As a result, the payload capabilities have been enhanced to 40 inches in length and 100 pounds with more power available for the experiment. Other mission types are under study for the Critical Design Review in March 1997.

Background

Last year, at the Small Satellite Conference, we introduced Spartan Lite, a small satellite designed to be launched from a Hitchhiker ejection system in a Get Away Special (GAS) can on the side wall of the Shuttle. The design has gone through many improvements and has matured over the year. Reducing costs has been a prime consideration of the study. The cost has decreased as the design has matured and uncertainties are reduced. Scientists have been an important part of the development team driving the design to accommodate the needs of the scientific community.

The enabling concept for the design is to wrap the spacecraft electronics around a central science cavity, with Attitude Control System (ACS) sensors and actuators as well as antennas on the deployed solar arrays. A single "Flower Petal" design has emerged that can do celestial and Earth pointing missions as well as solar science. In order to do meaningful design and analysis, it was necessary to pick straw man missions. Over 50 possible missions were examined. For the Preliminary Design Review (PDR) in May 1996, slim resources were used to concentrate on a single mission.

A Shuttle launched solar pointing mission was selected to envelop several possible instruments as a more or less universal solar case. Celestial and Earth pointing missions are diverse in their requirements, and are harder to define with representative straw men. Care was taken in the design not to preclude it being flown in other configurations.

Design Philosophy and Requirements

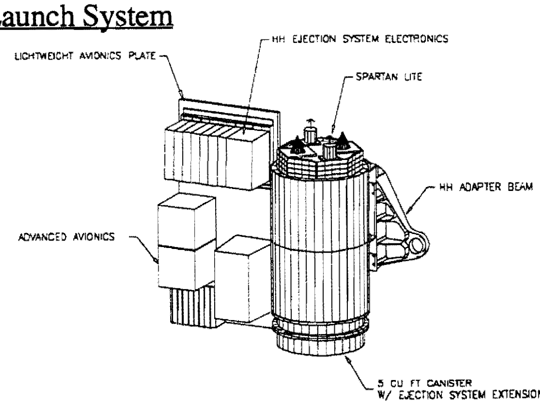
Most scientific satellites, often employing multiple instruments, have been custom built to maximize the science. At the extreme upper end are the 'Great Observatories'. The challenge posed by our scientific colleagues was to build an inexpensive fine pointer at the other end of the scale. It became obvious early on that advances in computers and electronics allow smaller and possibly cheaper systems. The challenge was in the mechanical, attitude control, mission operations, and traditional requirements areas. The Spartan Lite is considered to be class 'D'mission. A single string systems approach is used. Commercial off-the-shelf components (COTS) have been selected where possible and existing designs and software reused where possible.

The PDR configuration

The solar mission calls for half degree spacecraft pointing with improved pointing to arc seconds dependent on mission unique sensors or instrument error signals. A single basic design acts as a "to-build-to" target for science instruments. The instrument accommodations are 75 pounds, 14 inches diameter and 28 inches long mounted from one end. Forty watts are available to the instrument during science observations. The experiment stores its own data for telemetry and supplies the fine pointing error signals. To reduce costs, a single instrument with a single Principal Investigator (PI) was base lined. Spartan Lite is operated in the 'PI' mode where the PI controls the highly autonomous spacecraft from his institution after the first few orbits to reduce mission support costs.

The command and telemetry conforms to the Consultive Committee for Space Data Systems (CCSDS) format and has a downlink rate of 2 Mb/s with a 2kb/s uplink rate. The power system is a 28 volt regulated energy transfer system. Thermal design range is 0° to 40°C with a passive system with heaters. Radiation limits are 1k rad with no latch-up in critical components. The spacecraft is designed to 11g's in all axes with primary structural frequency above 50 Hz when mounted to the eject system. Antennas are provided for future use of the GPS for attitude information and a Space Mobile Satellite Services (SMSS) phone link for complete global coverage. Only existing systems are required for the demonstration flight while the scars are there for future enhancements. Shuttle safety was taken into consideration during the design but was not specifically addressed in the PDR package.

Launch System



SHELS ON A SIDEWALL CAN

The PDR base line case ejects from a Hitchhiker can mounted on a Shuttle sidewall beam. The Shuttle Hitchhiker Ejection & Launch System (SHELS) mounts to the bottom of a standard Hitchhiker/GAS can. Hitchhiker avionics are used to control the launch.

A three month minimum life time with up to a year possible was the design goal. A review of future Shuttle manifests shows 80% of the orbits are above 412 km. At 28° inclination or above, an altitude of 400 km results in an orbital life time for Spartan Lite of between 12 to 26 weeks depending on the solar flux.

Mechanical Systems

The structure is an octagonal cast aluminum frame with bonded sheet aluminum close out panels which are thinner than can easily be cast. The structure is designed to meet the Shuttle low risk fracture classification and to minimize thermal gradients. Finite element analysis shows maximum stress to be 8 ksi in the casting.

The momentum wheels occupy a compartment at one end. The instrument fills the rest of the cavity and is cantilevered from the open end with a simple support at the other end. Spacecraft electronics wrap around the outside of the casting.

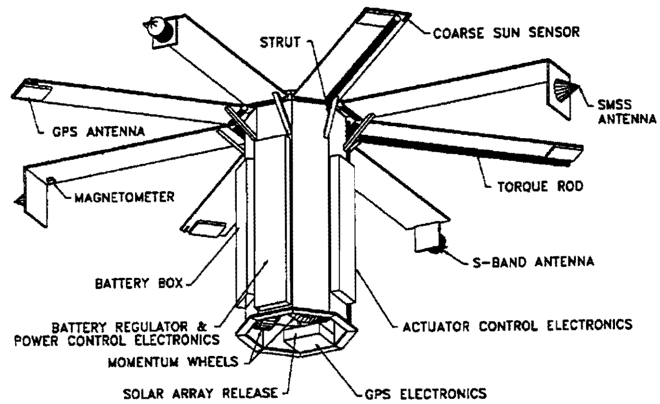
Eight solar arrays built from composite face sheet and stiffening ribs are hinged from the science end of the structure. The release mechanism, located at the momentum wheel end, is an eight link bellcrank with individual turnbuckle adjustment. The bellcrank is released by an pyro pin puller and rotated by a compression spring .

The arrays are deployed by torsional springs over the first 45° of motion, short carpenter rule snap struts drive them to the final locked position. The arrays are then rigidly locked in a repeatable position which allow them to be used to mount GPS antennas, ACS sensors, and torque rods. These snap struts have been built and the rigidity of the array tested on a mechanical mockup of the solar array and hinge. The release mechanism has not been mocked up and tested yet.

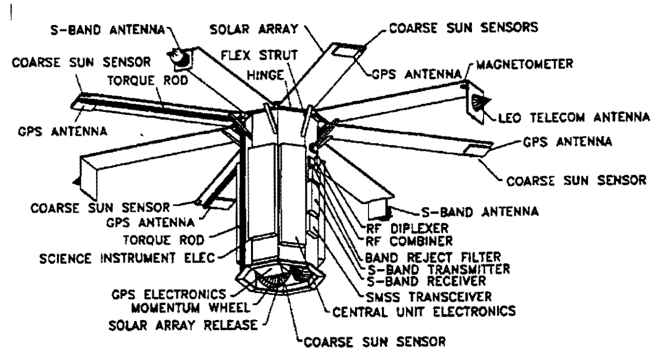
A casting drawing for the spacecraft has been made and sent to the caster for critique and estimates. Recommended changes have been incorporated and it is ready to be released.

Attitude Control Approach

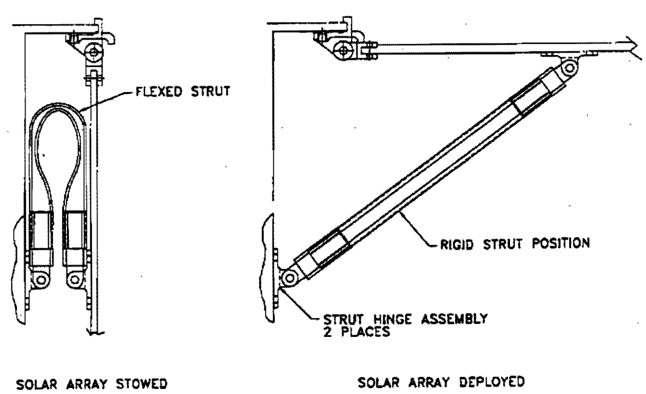
In order to minimize cost, maximum reuse is made of Small Explorer (SMEX) ACS: control laws, algorithms, simulation package, sensors, actuators, software, and testing. In addition, Global Positioning Satellite (GPS) signals can be used for position velocity and time (PVT). The design of the science pointing system allows for GPS, solar, and/or celestial pointing fine sensors a –"plug and play" type architecture. The controller uses a momentum based eigen axis slewing implementation.



FRONT VIEW



REAR VIEW



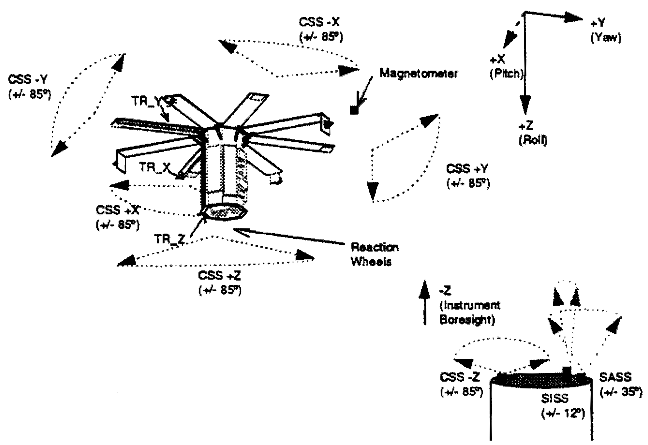
SNAP STRUTS

No ephemeris is required for initial acquisition and safehold. These are a fully autonomous control modes. The ACS system is designed to point the spacecraft to within 10° of the sun line with initial body rates of up to 6°/sec from any attitude without discharging the battery below 60%. The maximum maneuver rate is 50°/min. The spacecraft is allowed to drift during eclipse and can reacquire the sun within 3 minutes after coming out of an eclipse.

The system then steps through normal mode to sun point and then to the science mode. In the science mode the system will hold the experiment on target to the limit of the fine sensor errors which are typically in the arc-second range. SMEX based simulations have provided independent verification of the system.

Attitude Control Hardware

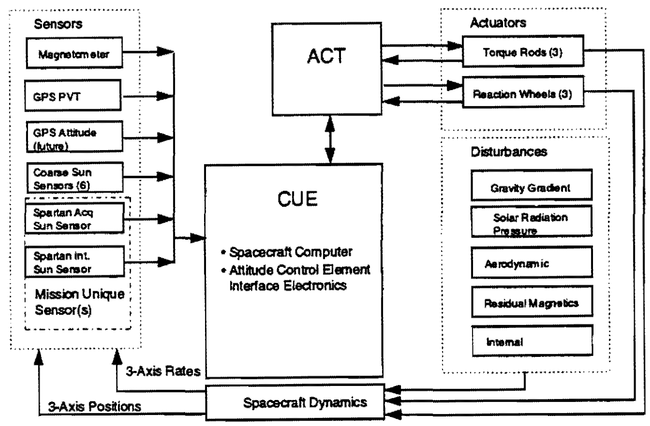
Spacecraft sensors and actuators consist of 6 coarse sun sensors (CSS) covering 4 pi steradians, a Spartan acquisition sun sensor (SASS) with a 35° field of view, a 3-axis magnetometer, 3 orthogonal reaction wheels, 3 orthogonal torque rods and a GPS receiver for PVT data. A Spartan intermediate sun sensor (SISS) is base lined in the experiment for fine pointing.



ACS sensor configuration

The Momentum wheels have 2.8 Newton-Meter-Second momentum storage and are a derivative based on the SWAS design. The rest of the hardware items are existing designs used on other Spartan and SMEX spacecraft.

Attitude Control Electronics

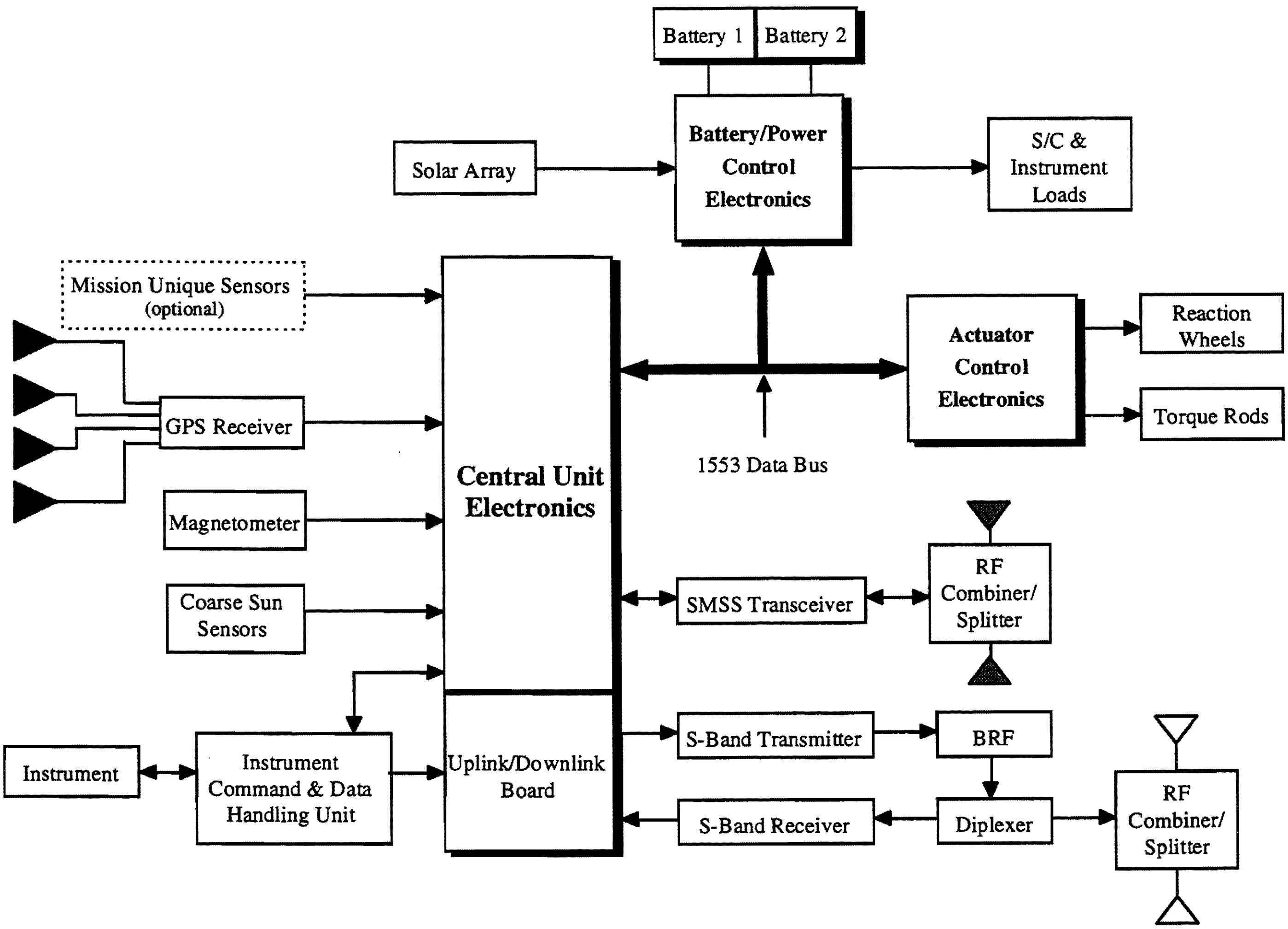


ACS Block Diagram

All computation is handled by the Central Unit Electronics (CUE), while the torque rods and momentum wheels are driven by the ACTuator control electronics. The ACT repackages exiting electronics from SMEX in the flat form necessary to fit on the outside of the spacecraft under the solar arrays before their deployment. The ACT shares a nearly identical housing to the CUE and Power Control Electronics (PCE).

Central Unit Electronics

The Central Unit Electronics receives all commands and determines the command type, instrument or spacecraft, and forwards all instrument commands in CCSDS format to the instrument processor. It commands and controls all spacecraft functions and provides failure detection and correction. The CUE supports 72 hours of autonomous operation using stored commands. It performs all attitude control and determination calculations and power system processing. The CUE converts analog monitors using a 12 bit analog to digital converter (ADC) and stores up to 4M bytes of housekeeping telemetry. It limit checks housekeeping telemetry and processes all telemetry in CCSDS format for downlink. The downlink Rates are 900Kb/s and 2.0Mb/s with four encoding options: Convolutional, Bi-Phase L, Convolutional-Biphase, and NRZ-L. The SMSS uplink/downlink at 9600 Baud is also handled by the CUE. It has a 1553 bus and RS-422 interfaces. The CUE uses an industrial 80486 single board computer with a 80387 math co-processor. It also has an interface card, an I/O card, an uplink/downlink card, a housekeeping card with the ADC and magnetometer electronics, and a spare card for future expansion. The CUE software is over 90% reused from SMEX.

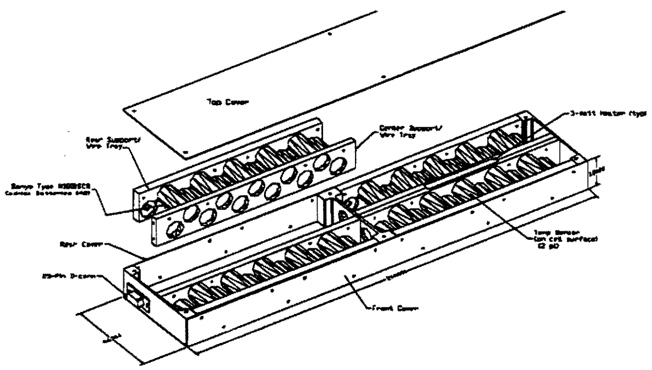


SYSTEM BLOCK DIAGRAM

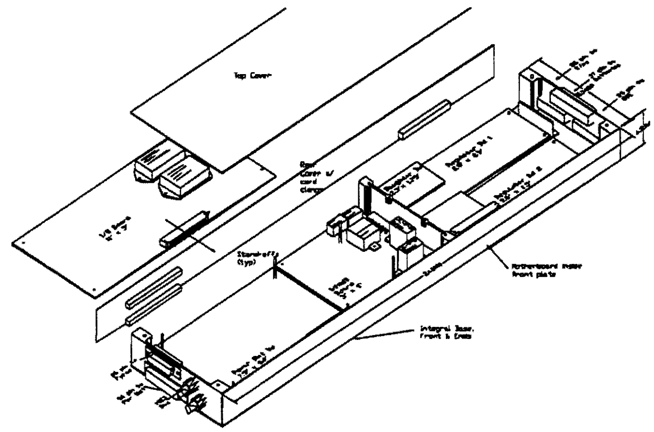
Power System

The power system is a regulated energy transfer system used on a number of current spacecraft. The eight arrays provide 1.18 square meters of area covered with Gallium Arsenide cells. The projected end of life solar array power is 219 watts. Spacecraft peak required solar array power is 164 watts.

Two battery packs of twenty-one 1.8 ampere hour Sanyo Ni-Cd sub-C cells provide 3.6 ampere hours with a bus voltage of 28+/- 4. The commercial cells are subjected to selection and conditioning. A separate charger is used for each pack. The PCE box contains both the charge regulators as well as the power distribution circuits and pyro inhibits.



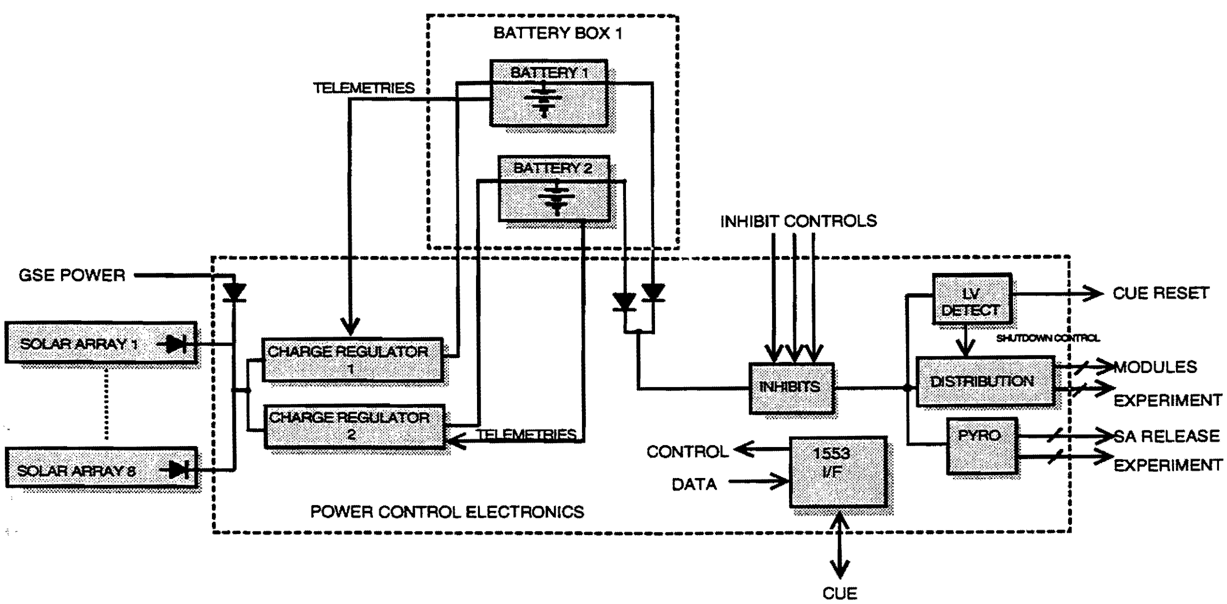
BATTERY BOX



POWER CONTROL ELECTRONICS

POWER BUDGET

	Duty cycle	Watts
ACS	100%	14.5
CUE	100%	7.5
ACT	100%	4.5
S-Band (RX&TX)	2.43%	1.24
SMSS Active	21.85%	5.46
SMSS Rcv Only	78.15%	1.46
Power Electronics	100%	11.68
Heaters	39.69%	5.0
Science (normal)	60%	18
Science (standby)	100%	10
Total Orbit Average Power		**79.09 Watts**



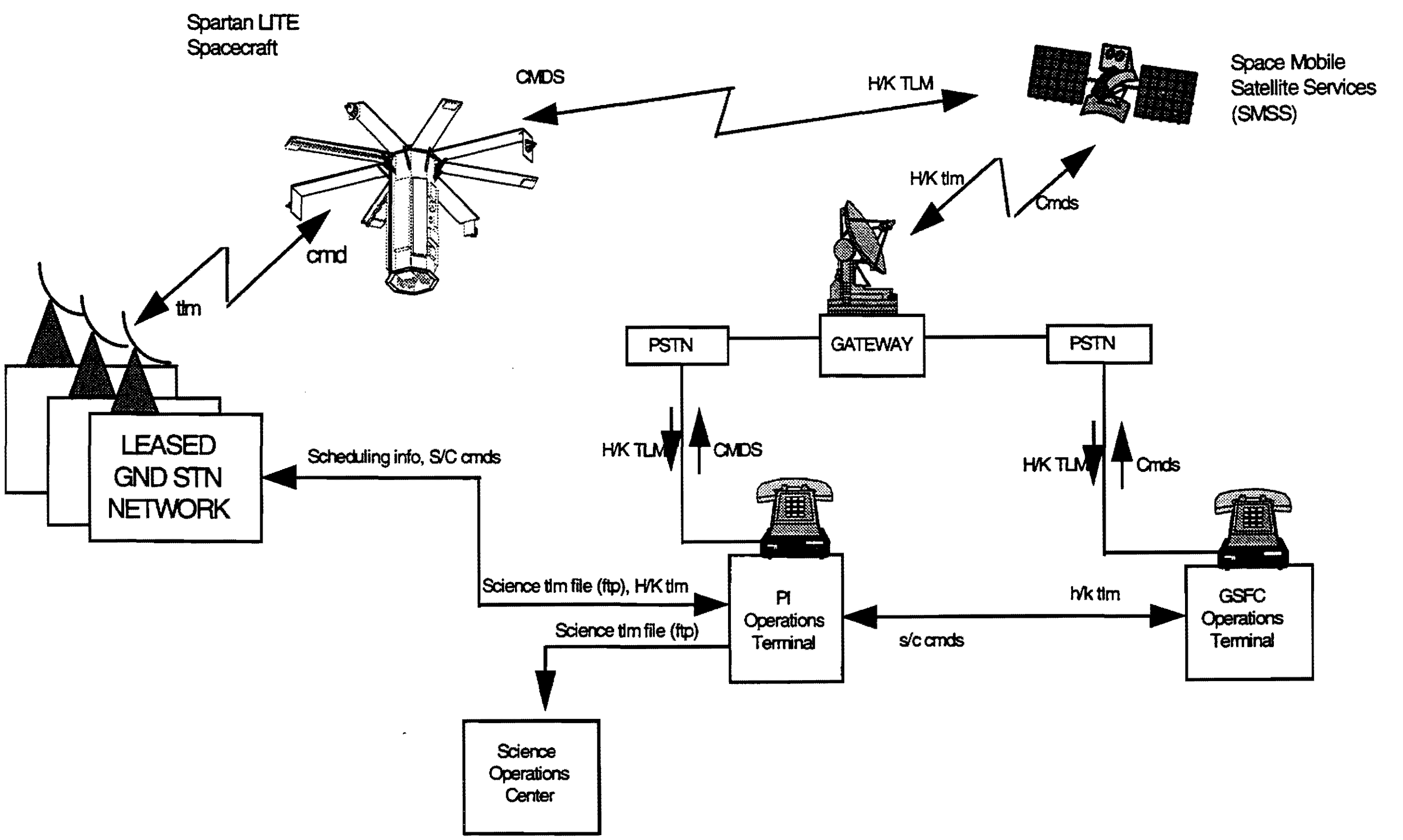
Mission Operations

In order to greatly reduce mission operations cost, normal spacecraft operations are turned over to the Principal Investigator (PI) at his institution. GSFC performs the launch and early orbit operations, backup spacecraft monitoring, and emergency trouble-shooting. The PI provides at least two people to operate spacecraft during I&T activities.

The same Ground Service Equipment (GSE) is used for I&T and operations (GSFC provides the PI with equipment to be returned upon mission completion). The GSE is a PC based work station with commercial add-on boards. This spacecraft and GSE are capable of 72 hour autonomous operation and can page on-call personnel upon detecting an anomaly. Relaxed requirements for data integrity allow it to be transmitted over the World Wide Web.

Time is leased on an automated ground station network for high-rate TLM downlink and backup CMD uplink allowing 3-4 passes per day. The ground station provides the PI with quick-look data and delivers science data as time permits via file transfer over the web.

This same scheme is being proposed by other SMEX and Spartan missions.



Post PDR (Getting rid of the Can)

The proposed launch system for Spartan Lite is the SHELS which can be attached to the bottom of a standard GAS/Hitchhiker can. For the PDR configuration we had eliminated the lid of the GAS can to allow the satellite to extend beyond the can. The extra length was needed for antennas and the weight saving of the lid was used to increase the satellite weight. Thermal analysis showed heaters on the can were sufficient to keep the spacecraft within limits. Discussions during the PDR made it clear that the can provides no particular benefit to the spacecraft such as structural containment, pressurized atmosphere or contamination control.

The can does impose a serious shape and volume penalty in addition to a weight burden. A scheme was being explored to mount the SHELS to the sidewall beam with a light frame work to form a thermal controlled shell around the spacecraft. The design of this system is now well underway, along with an optimum set of Hitchhiker avionics to control it.

Freed of the constraint of the can, the spacecraft has been lengthened to 50 inches and the payload to 40 inches. The external diameter of the spacecraft has been expanded from 19 to 23 inches. The diameter of the science cavity stays the same resulting in more room under the solar arrays in the stowed configuration. This eases the most stringent constraint on electronics box thickness from 1½ to 2 inches. This may not seem like much, but it allows us to use some standard components and shave costs. A COTS GPS receiver can now also be mounted on the outside rather than in the end cavity with the wheels. We can also use existing SMEX momentum wheels rather than developing a smaller derivative that had been base lined at PDR. The solar arrays now can be made of honeycomb, reducing the cost and increasing the stiffness.

More area on the spacecraft is now available because of the greater length and larger diameter. The solar arrays have increased to 1.9 Sq. M. The batteries have been changed from C cells to D cells, reducing the depth of discharge and extending power system life to 18 months. By adding an additional D cell pack (13 lbs charged to the experiment) the spacecraft orbit average power can be increased to 150 watts making 100 watts available to the experiment. The spacecraft target weight has been raised to 375 pounds which is easily accommodated by the 400 pound plus capacity of the SHELS. The instrument weight has been raised from 75 to 100 pounds. Even with this enhanced design two Spartan Lites can fit side-by-side under a Pegasus payload shroud.

Spartan Lite at PDR Mass Budget Pounds

Structure	44
Attitude Control System	44
Command and Data handling	33
Power	45
RF Communications	11
Thermal	8
Spacecraft Total	184
Science instrument	75
Total	259
Spacecraft Target Weight	300

Spartan Lite After PDR Mass Budget Pounds

Structure	50
Attitude Control System	41
Command and Data handling	36
Power	69
RF Communications	11
Thermal	16
Spacecraft Total	223
Science instrument	100
Total	323
Spacecraft Target Weight	375

The CDR

A CDR is scheduled for March 1997. ACS analysis is being run to characterize the larger spacecraft in both the Solar and Celestial configurations. A minimum controllable altitude is being established. It is doubtful that orbital life times will change much from the PDR study since the ballistic coefficients of the satellite have not changed much. Design of the electronic boxes will be carried from schematics to layouts. A full scale working model of the solar array release mechanism and a flight like array will be built. A casting pattern will also be built and if the budget allows, the structure will be cast. The spacecraft computer, an COTS industrial 80486 based single card unit, is being programmed to gain experience running the modified SMEX routines on it. The CUE should be built by CDR and going into test. The SHELS should also be at CDR level by next March. Adaptation of a SMSS receiver to work with small satellites is ongoing.

Spartan Lite components especially the CUE and its software are being adapted to other Spartans.

Beyond the CDR

We have submitted a Form 1628 to manifest Spartan Lite on the Shuttle. Selection of a real science mission is the next step. We are also looking at propulsion systems to extend orbital life for future flights.