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LEO-1: Development Of A University Microsatellite For Flight Testing New Technologies

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The Hawaii Space Flight Laboratory (HSFL) was established at the University of Hawaii at Manoa to educate students and help prepare them to enter the technical workforce, and to help establish a viable space industry that will benefit the State of Hawaii. In 2011 the first mission, STU-1, will be launched. It includes a spacecraft being designed and built by the HSFL called LEO-1. The primary objectives for LEO-1 mission are: (1) to demonstrate the ability of the HSFL to design, build, and operate a small satellite in the 50-kg class as a platform to test new technologies; and (2) support the C-band radar transponder experiment (CRATEX) for the USAF; the Coherent Electromagnetic Radio Tomography (CERTO) experiment for the Naval Research Laboratory; support the LEO-1 Solar Array Experiment; and to demonstrate the suitability of the LEO-1 spacecraft to test technologies suitable for missile detection and tracking. The radar calibration experiment uses C-band transponders and precise orbit determination to test new technologies designed to aid the Department of Defense to calibrate their radars around the world. The CERTO Doppler beacon provides accurate orbit determination using various ground stations around the world, but also provides data that will be used to help characterize the ionosphere for space weather monitoring. A secondary payload consists of two digital imagers to provide color images of the Earth. The LEO-1 spacecraft will be placed into a 600-km circular 9 p.m. ascending Sun Synchronous Orbit to optimize its support of the radar calibration experiment. The 60-kg LEO-1 spacecraft is 3-axis stabilized using three magnetic torque rods and a reaction wheel for attitude control; and two sun sensors, a 3-axis magnetometer, and an inertial measurement unit for attitude determination. Communication is provided by a UHF- transceiver linked to a ground station located in the Leeward Community College in Honolulu and other partner ground stations. Control of the mission will be done in the HSFL Mission Operations Center located on the University of Hawaii campus at Manoa. Integration and testing of the spacecraft will be done in the clean rooms at the HSFL facilities on the UH campus. The HSFL is using a core team of experienced professionals supplemented with graduate and undergraduate students to design, build, and test the LEO-1 spacecraft. Part of the design philosophy includes the development of two nearly identical spacecraft – the Engineering Model which will be used for integrated testing and then transitioned to be part of the tesbed/simulator for operations, and the Flight Model. Another means employed to keep the spacecraft cost to a minimum is the use of commercial-off-the-shelf (COTS) as much as possible.

I. Introduction

The Hawaii Space Flight Laboratory (HSFL) was established at the University of Hawaii at Manoa for two primary purposes: (1) to educate students and help prepare them to enter the technical workforce, and (2) to help establish a viable space industry that will benefit the State of Hawaii.

The Low Earth Orbit Nanosat Integrated Defense Autonomous Systems (LEONIDAS) Program is a Congressionally sponsored program named after the Spartan King who led a small Greek force that held a much larger Persian army at bay at the Battle of Thermopylae in 480 BC. This program is of importance to both the United States and to the State of Hawaii. Its purpose is to develop and demonstrate small-satellite orbital launch capability from the Pacific Missile Range Facility (PMRF). The objectives of the program include establishing a technical work force in the State of Hawaii with a development program that trains students to enter the technical work force. The original vision for the program was to develop a chain of small satellites to pass information to a single ground station and assist with disaster-relief efforts. It later evolved into a constellation of very small (nano-) autonomous

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satellites to aid the Department of Defense (DoD). By the time funding was approved in 2007, the program was designated to cover two launches and two spacecraft. Oversight of this program was given to the U.S. Army's Space and Missile Defense Command (SMDC) with headquarters in Huntsville, AL. In February 2008 the oversight of the program was transferred to the Operationally Responsive Space (ORS) Office with headquarters at the Kirtland Air Force Base in Albuquerque, NM.

The first LEONIDAS mission was to consist of a small (30 – 50 kg) satellite to demonstrate a stable platform for remote sensing (especially missile detection and tracking). The LEONIDAS Missions #1is designated as Science and Technology for the University (STU) #1. This is the official mission launch designation provided to PMRF. The first launch of the LEONIDAS Program and HSFL is STU-1, which includes the LEO-1 spacecraft; other secondary satellites (CubeSats); the Super Strypi launch vehicle, being developed by Sandia National Laboratories for HSFL; and the payload adapter and deployer (PAD), designed and built by the NASA Ames Research Center.

With the transfer of oversight of the LEONIDAS Program from SMDC to ORS, the mission for the first satellite, LEO-1, also changed. Originally it was designed to show the suitability of the microsat bus to be a platform for remote sensing of missile launches and tracking. It then was changed to be a microsat platform to test various technologies of interest to the Department of Defense including the ORS office. During the six months following the switchover, new experimental payloads were found and feasibility studies were performed. Finally the payload manifest was finalized with the inclusion of the C-band radar transponder experiment (CRATEX) for the USAF; the Coherent Electromagnetic Radio Tomography (CERTO) experiment for the Naval Research Laboratory (NRL); and the LEO-1 Solar Array Experiment (LSAE). The current design of LEO-1 has successfully passed its Preliminary Design Review (PDR) in May 2009, and is on course for a launch in 2011.

The LEO-1 is a three-axis stabilized octagonal spacecraft of approximately 60 kg mass and will be launched into a 600-km 9 p.m. ascending node sun synchronous orbit (SSO). Communication is provided by a UHF- transceiver linked to a ground station located in the Leeward Community College in Honolulu and other partner ground stations around the world. Control of the mission will be done in the HSFL Mission Operations Center located on the University of Hawaii campus at Manoa.

II. Mission Description

This section describes the mission architecture, project management, the launch trajectory, and orbits analysis.

A. Mission Architecture

The LEO-1 spacecraft is part of the designated STU-1 launch from PMRF. Figure 1 shows the top level mission architecture for STU-1. HSFL, located at the UH Manoa campus, is at the center of the system and will be responsible for design, fabrication, project management; integration & testing of both the LEO-1 spacecraft and the PAD. All the payloads being flown on STU-1 will be integrated on the PAD at HSFL before shipment to PMRF for launch vehicle integration. HSFL will also be responsible for the ground segment including the ground stations, and mission operations centered in the HSFL Mission Operations Center (HMOC).

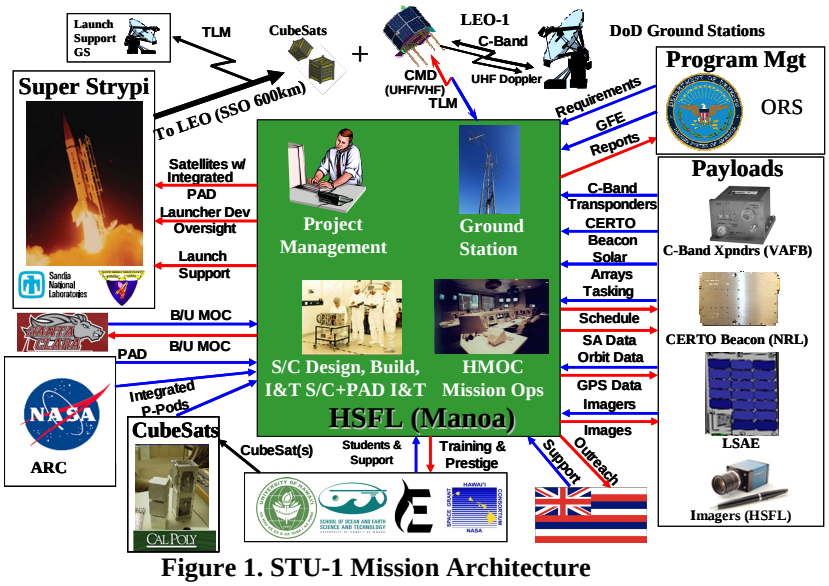


Figure 1. STU-1 Mission Architecture

B. Project Management

1. Organization and Staffing

The HSFL employs a small and dedicated team to lead the effort of developing the LEO-1 spacecraft. To fulfill the university's mission to educate students, HSFL involves several students for these projects. However, students are almost by definition inexperienced and relying solely on them would result in an unacceptable level of risk for a multi-million dollar highly technical project such as LEO-1. The HSFL management has thus hired a core team of experienced experts in the various fields required to complete the development effort. This cadre of experts serves as the mentors for the students and help mitigate the risks inherent to the development of a new launch vehicle and spacecraft.

The HSFL is an element of the University of Hawaii and thus has a primary purpose to educate students. This purpose is reflected in the organizations employed for the LEO-1 project. By employing a number of students, both undergraduate and graduate, the HSFL will accomplish the education of students to prepare them to enter the workforce as already experienced and productive engineers and scientists, and to help keep the development costs of the projects low. The project management and lead engineers are all experienced full-time HSFL staff or part-time faculty. The experience of the key personnel allows them to perform multiple tasks within the organization with the help of undergraduate and graduate students. In some cases the lead engineers will do the work themselves, usually where a high level of technical expertise or experience is required, or they will supervise the performance of tasks by students. Even when the lead engineers do the task themselves, one or more students will be involved if at all possible to observe and learn. The staffing level for most of the project, including staff engineers, faculty, and students, stays in the range of 8 to 9 Full Time Equivalents (FTEs).

The LEO-1 project organization is led by Dr. Trevor Sorensen, project manager, reporting to the HSFL Director, Dr. Luke Flynn. The organization of the LEO-1 project is shown in Figure 2.

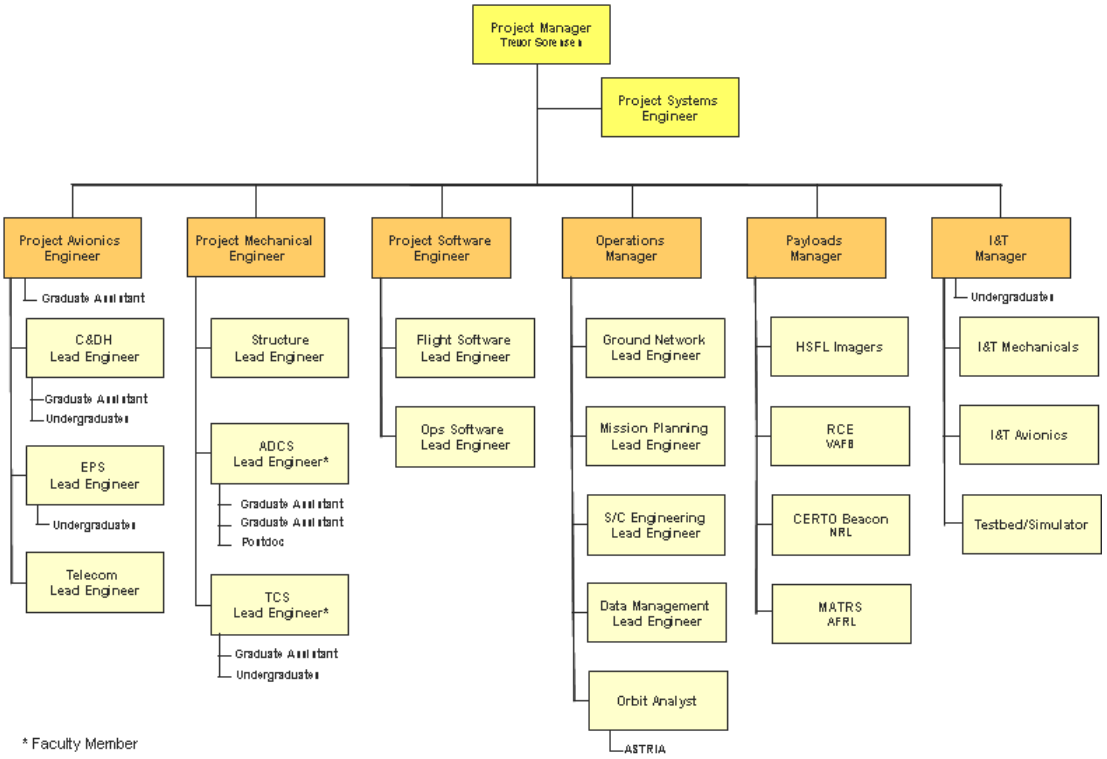


Figure 2. LEO-1 Organization

2. Development Process

The basic design process being followed by the HSFL Team is fairly standard for similar space projects. It is based loosely on the design process outlined in Reference 1. The development process has been divided into five phases as defined in Table 1. The HSFL phase nomenclature is based on the NASA system (A-E) while the

corresponding DoD nomenclature is in parentheses (I-V). Several milestones have been planned into the development process including the following:

1. Conceptual Design Review (CoDR)
2. System Requirements Review (SRR)
3. Preliminary Design Review (PDR)
4. Critical Design Review (CDR)
5. Test Readiness Review (TRR)
6. Mission Readiness Review (MRR)

Table 1. LEO-1 Development Process Phases

Phase	Name	Description
A (0)	Concept Definition	Defines requirements and baseline conceptual design – ends in SRR
B (I)	Definition & Acquisition Planning	Improves on baseline design with further analyses and trades – ends in PDR with preliminary design
C (II)	Detailed Design	Completion of system design – ends in CDR
D (III)	Development	Includes procurement, fabrication, integration & testing – ends in launch
E (III)	Operations	All operations after launch including Launch & Early Orbit (L&EO), Engineering Evaluations & Checkout (EE&C), nominal operations, and terminal operations – ends with reentry or loss of satellite

The developmental process for Phases A & B being followed by HSFL in the development of LEO-1 is shown in Figure 3. The status of the steps in this process at the time of the PDR is indicated by the color: green = completed, yellow = in progress, red = not yet started. The Work Breakdown Structure (WBS) for the project contains over 500 tasks.

The LEO-1 project schedule is shown in Figure 4. A temporary funding shortfall with resultant reduction in force caused the CDR date to be pushed back about six months until May 2010. The spacecraft is currently scheduled to be completed by June 2011.

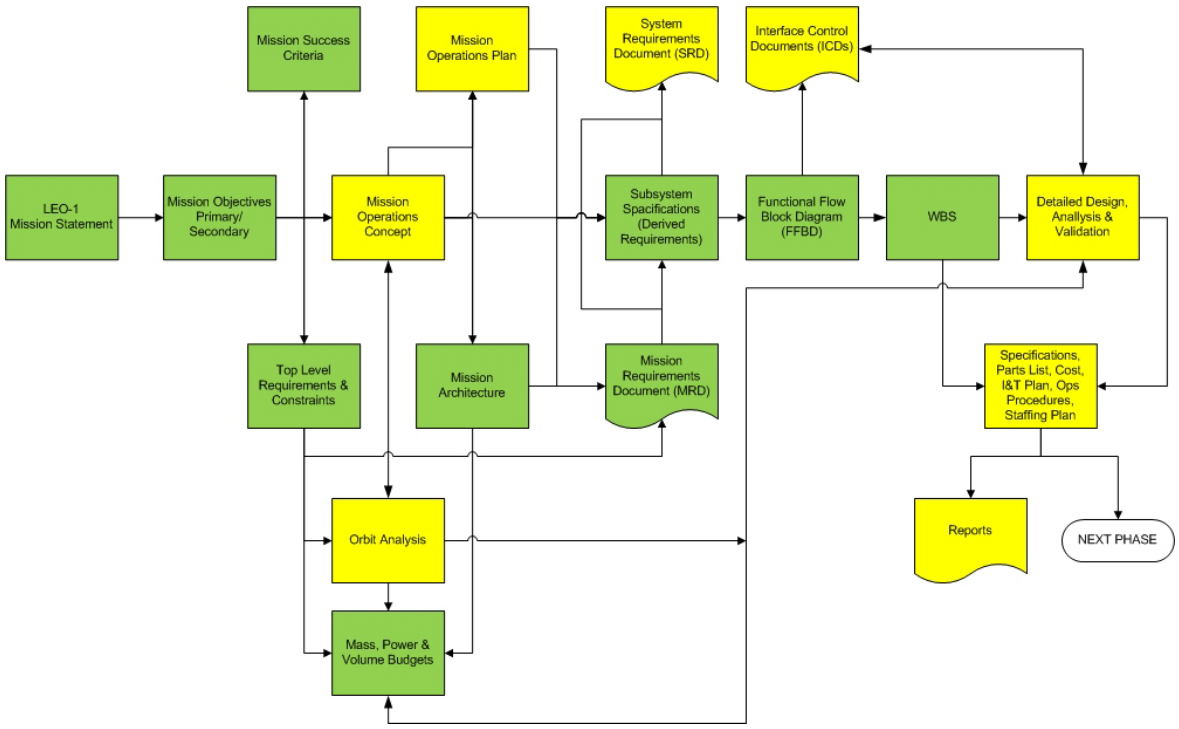


Figure 3. Phases A & B Development Process

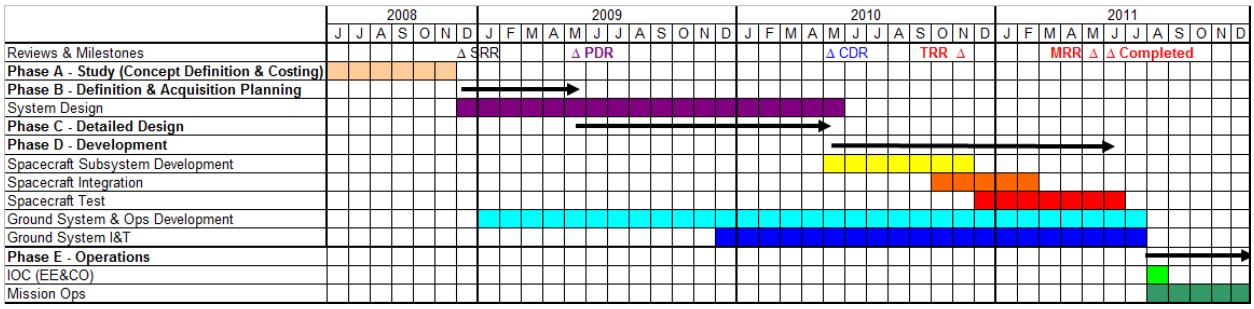


Figure 4. LEO-1 Development Schedule

2. Systems Engineering

The role of systems engineering in the project is to define and develop an operable system capable of meeting mission requirements within the imposed constraints, including performance, cost, risk, and schedule. The HSFL approach follows common systems engineering best practices appropriate to the project and include the following phases: functional analysis; system synthesis; system evaluation and decision; and system definition. The Systems Engineering Management Plan document captures in greater detail the approach to the systems engineering effort.

The project systems engineering effort also includes configuration management; oversight of project documentation; and places a major role in the project risk management strategy.

C. Trajectories and Orbit Analysis

1. Launch Profile

The LEO-1 spacecraft will be ready for launch by June 2011. It will be launched by an enhanced Super Strypi three-stage, spin-stabilized, solid-propellant rocket from the Pacific Missile Range Facility (PMRF) located on Kauai in a southerly retrograde direction. The notional launch profile is shown in Figure 5.

The spacecraft will be deployed from the Payload Adapter and Deployer (PAD), being developed by NASA Ames Research Center. The PAD will be capable of carrying one primary microsatellite and up to 24 or 32 cubesats or equivalents. The nominal insertion orbit is into a 9 p.m. ascending circular sun synchronous orbit (SSO) with an altitude of 600 km. and inclination of ~ 98º. After the third stage combustion has terminated, a yo-yo mechanism will despin the vehicle from 1 rps to no more than 4º per second.

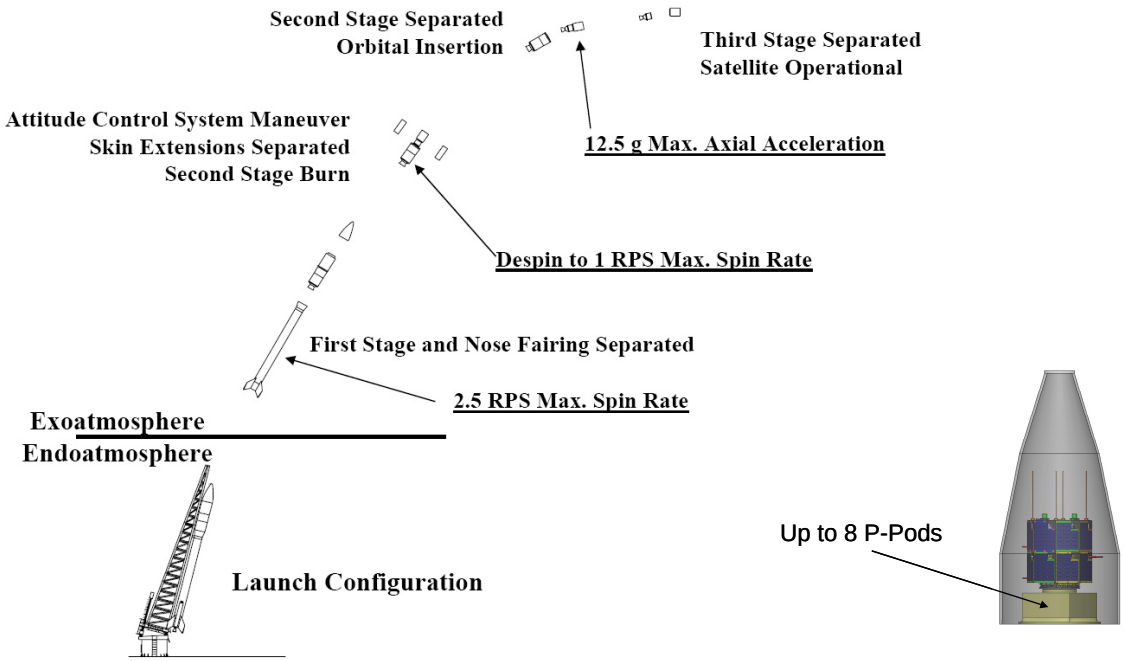


Figure 5. Notional Enhanced Super Strypi Launch Profile[2]

2. Orbit Analysis

The LEO-1 spacecraft has no propulsion subsystem and will be unable to alter its orbit. It is expected to reenter the Earth's atmosphere within 25 years after the end of the design mission (three years). The orbital analysis was performed by the AFRL based upon orbit insertion data provided by Sandia National Laboratories. The assumed baseline orbit is shown in Table 2. The design length of the mission is three years. Dispersion analyses were also done, but are not included in this paper.

Table 2. Baseline Parameters for Orbital Analysis

Launch Date	1 December 2010 20:11:05 UTC
Altitude	600 km
Orbit Type	Circular, Sun Synchronous
Inclination	97.79 degrees
Circular Velocity	7.558 km/s
Orbit Angular Velocity	3.723 deg/min
Period	96.69 minutes
Revolution per Day	14.85
Force Model Parameters	21 x 21 EGM96 Earth gravitation Solar and lunar gravitation $C_D = 2.2$ A/m = 0.0067 m^2/kg Jacchia 1970 Lifetime atmosphere

Three ground stations were assumed for the initial study: Honolulu, Santa Clara CA, and Guildford England. Figure 6 shows the nominal orbit ground traces of the first three complete orbits of the LEO-1. The importance of a ground station in Europe is evident from this plot. The altitude prediction is shown in Figure 7, the inclination prediction in Figure 8, and the ascending node solar time (local time) over the three-year lifetime is shown in Figure 9. This latter plot is important for a desired Sun Synchronous Orbit (SSO) to show how well the satellite keeps to the desired local time. This particular analysis shows the satellite passing a little sooner than the desired 2100 hours local time, but can be tweaked by the launch time.

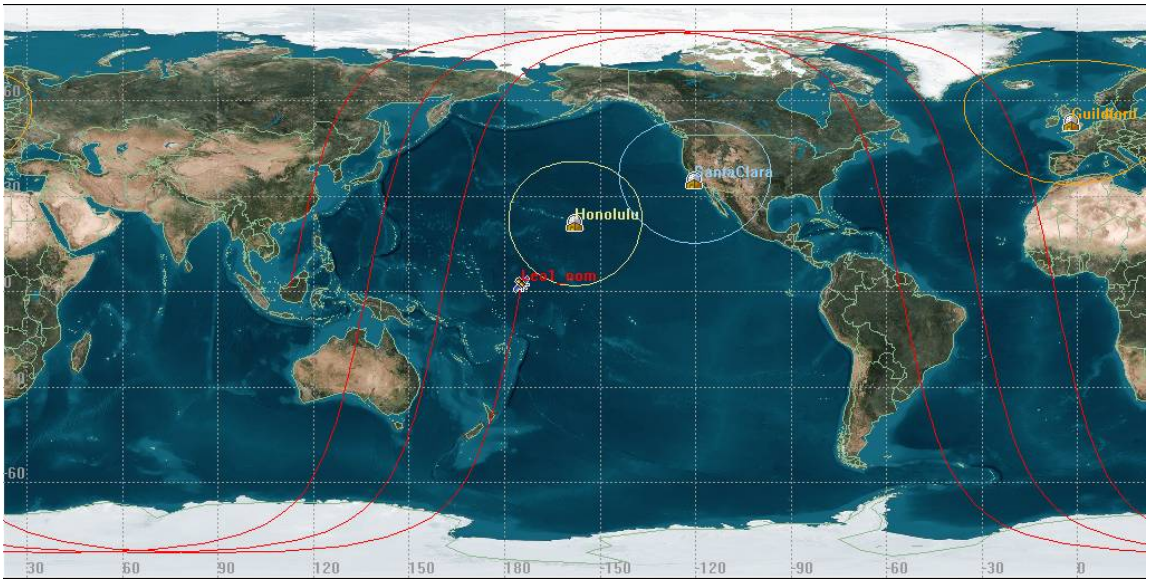


Figure 6. Ground Traces of First Three Nominal Orbits

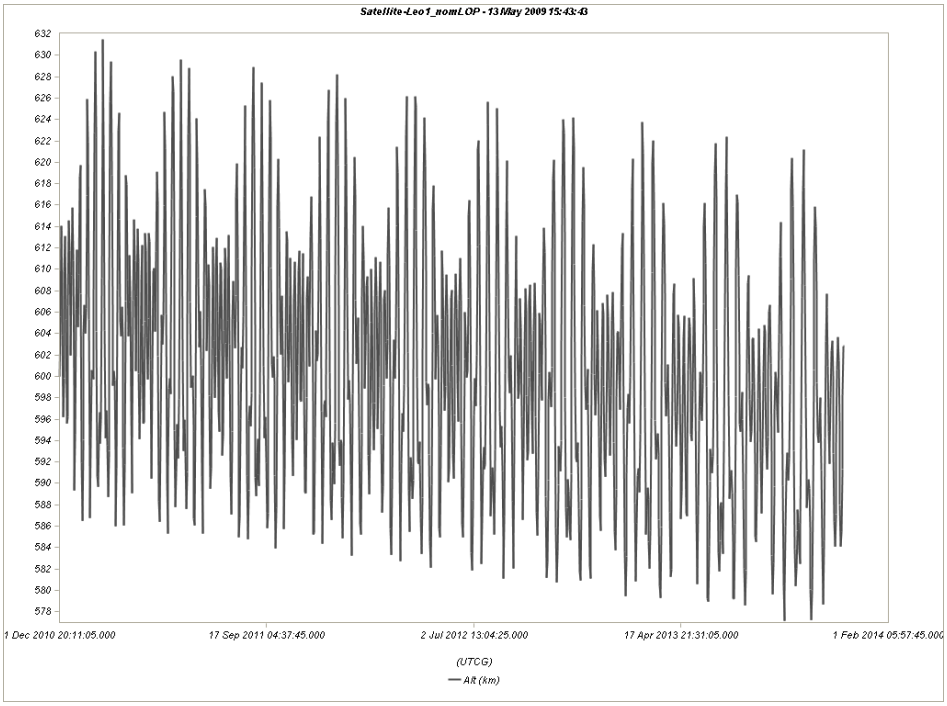


Figure 7. Orbital Altitude Prediction

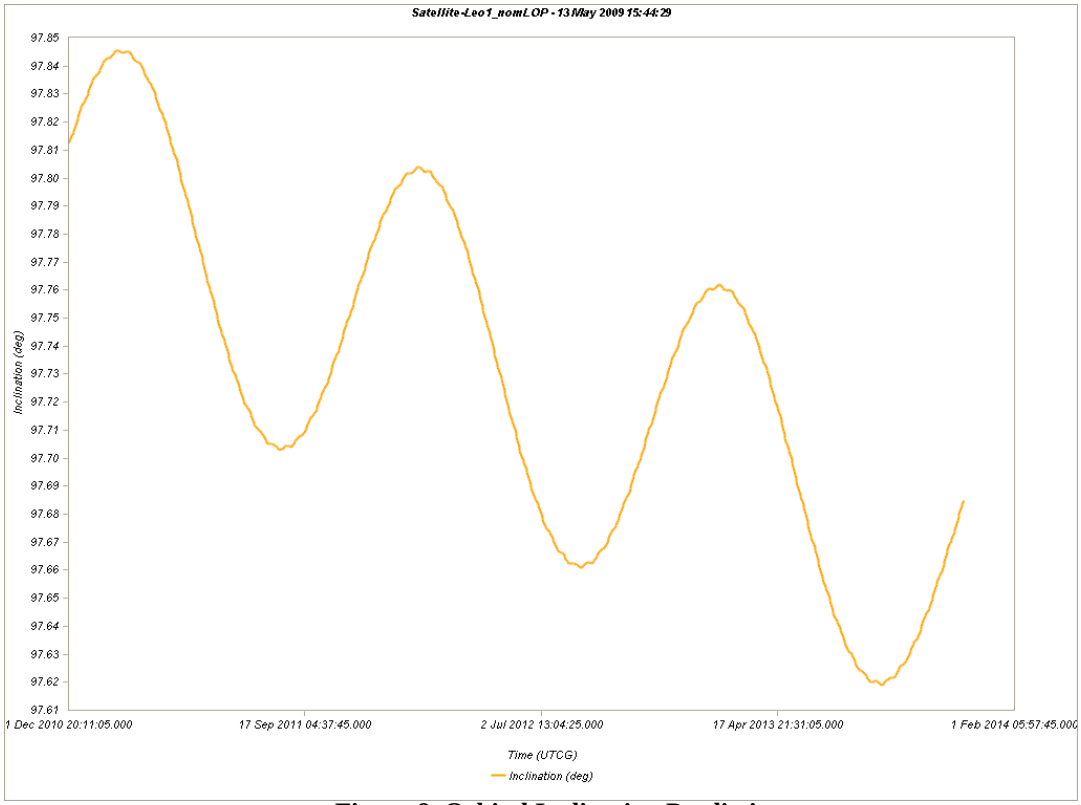


Figure 8. Orbital Inclination Prediction

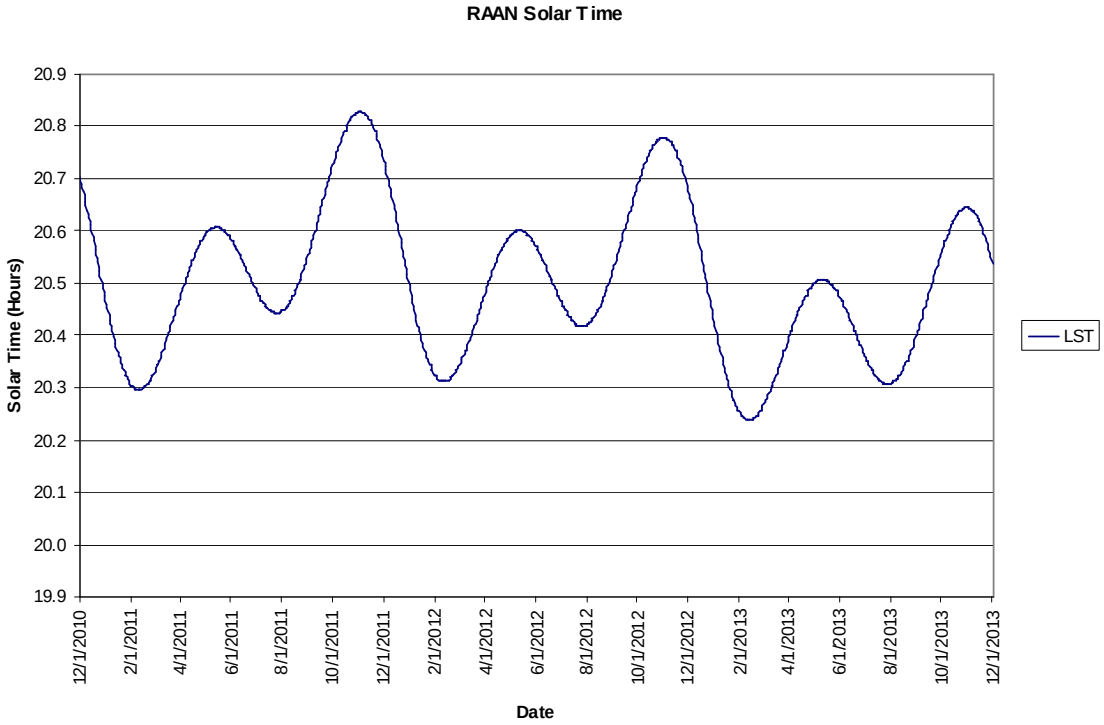


Figure 9. Ascending Node Solar Time

III. Spacecraft Description

This section describes the spacecraft bus, its configuration, and its subsystems.

D. Configuration and Mass Budget

The initial mass budget of the LEO-1 spacecraft is 62 kg. The mass budget is shown in Figure 10. The external and internal configurations are shown in Figures 11 and 12.

	Mass (kg)
Structures	16.3
EPS	8.9
CDH	1.9
Telecom	3.0
ADCS	4.4
Thermal	2.2
Systems	4.4
Payload	11.2
Margin	9.8
TOTAL	62.0

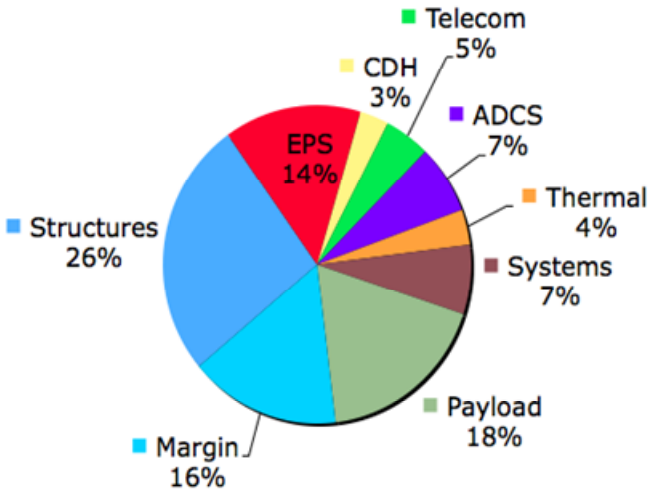


Figure 10. LEO-1 Mass Budget

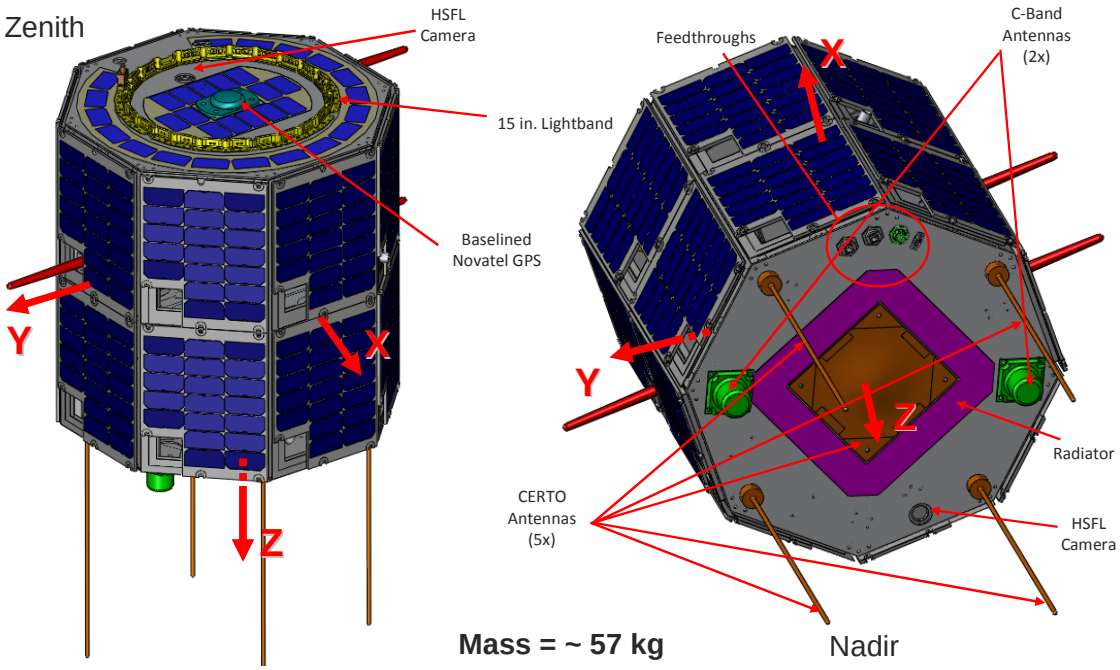


Figure 11. LEO-1 External View

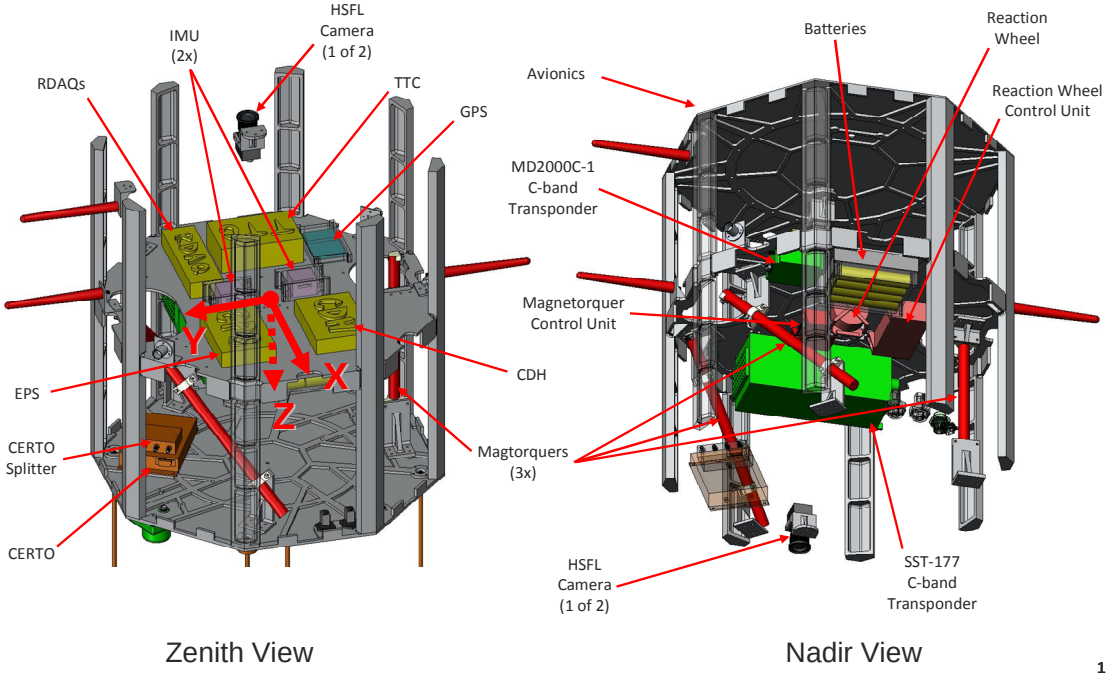


Figure 12. LEO-1 Internal Configuration

B. Bus Subsystems

1. System

The spacecraft functional block diagram (FBD) is shown in Figure 13. The basic philosophy used in the design is a distributed avionics architecture, where each subsystem is controlled by a remote data acquisition (RDAQ) unit that interfaces with the central processing unit (CPU), which provides the top level executive control. Each of the spacecraft bus subsystems will be described in this section. The subsystems are: Command and Data Handling (C&DH), Electrical Power (EPS), Telecommunications (Telecom), Structures, Attitude Determination and Control (ADCS), Thermal Control (TCS), Payloads, and Flight Software (FSW).

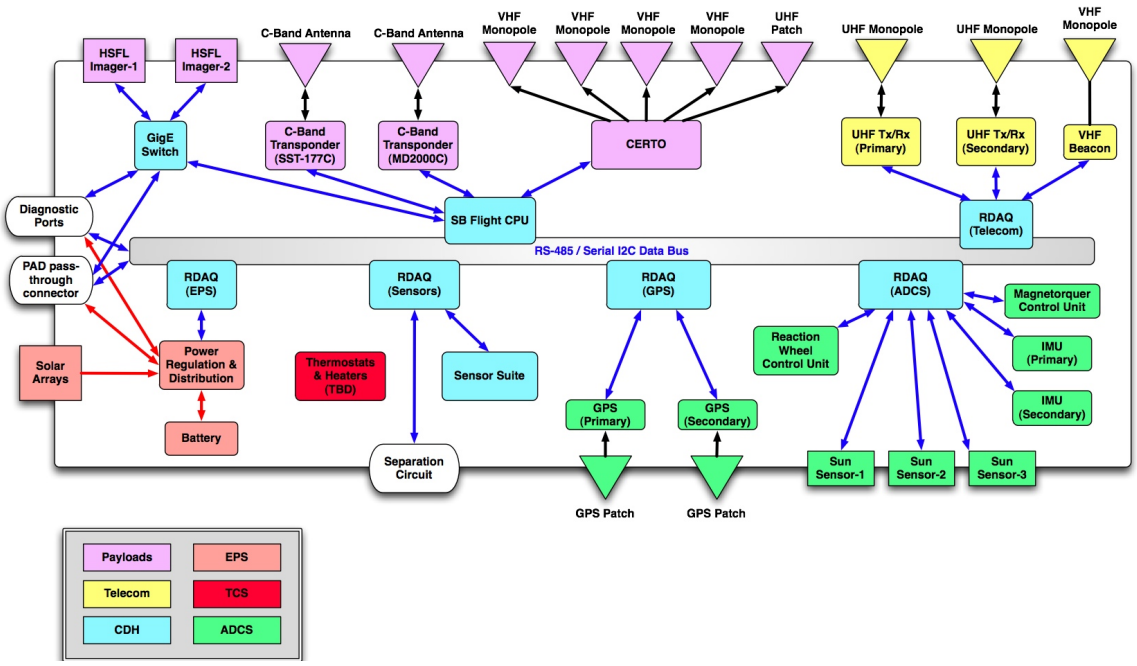


Figure 13. LEO-1 Functional Block

2. Command & Data Handling (C&DH) Subsystem

The spacecraft (S/C) is controllable by real-time uplinked commands, time-delayed command scripts, event-driven command scripts (e.g. establishing lock with a ground station during a pass), and autonomous onboard flight software. It has sufficient onboard storage to store at least 24 hours worth of state-of-health (SOH) and payload data. The S/C is capable of storing delayed commands to support the spacecraft and mission for at least 48 hours.

CDH Subsystem contains one central processor unit (CPU), and multiple microcontroller based Remote Data Acquisition modules (RDAQs). The Main CPU is a PowerPC 405 processor performing attitude control, telemetry, command processing, and CMOS image capture and processing. RDAQ processors are 16-bit Harvard RISC or 32-bit ARM processors performing sensor sampling and Analog-to-Digital, Digital-to-Analog conversion. Communication between the flight computer and RDAQ processors is via a RS-485 bus at $\geq$ 115kbps. Communication between RDAQ processors is via I2C bus at 400 kbps. The MIP 405 CPU has been flight-tested in space.[3]

The Attitude Determination and Control Subsystem (ADCS) can be controlled by the ADCS RDAQ in a limited mode. Telecom transceivers can be handled by COM RDAQ in limited mode. Basic telemetry data sent via separate beacon radio RDAQ inter-module communication on separate I2C data bus.

The functional allocations for the C&DH are as follows:

- MIP405 Flight Computer
  - Command processing and scheduling
  - Wideband telemetry generation
  - ADCS estimation and control
  - HSFL imager interfacing and image data compression

- RDAQ Controllers
  - Housekeeping telemetry generation
  - EPS subsystem control and measurement
  - Launch sensors interfacing (temperature, pressure, acceleration, spin-rate, vibration)
  - ADCS sensors interfacing (IMUs, sun sensors, momentum wheel tachometer)
  - ADCS actuators interfacing (Magtorquer rods, momentum wheel)
  - Payload enable / disable
  - Real Time Camera Control

The C&DH Block Diagram is shown in Figure 14.

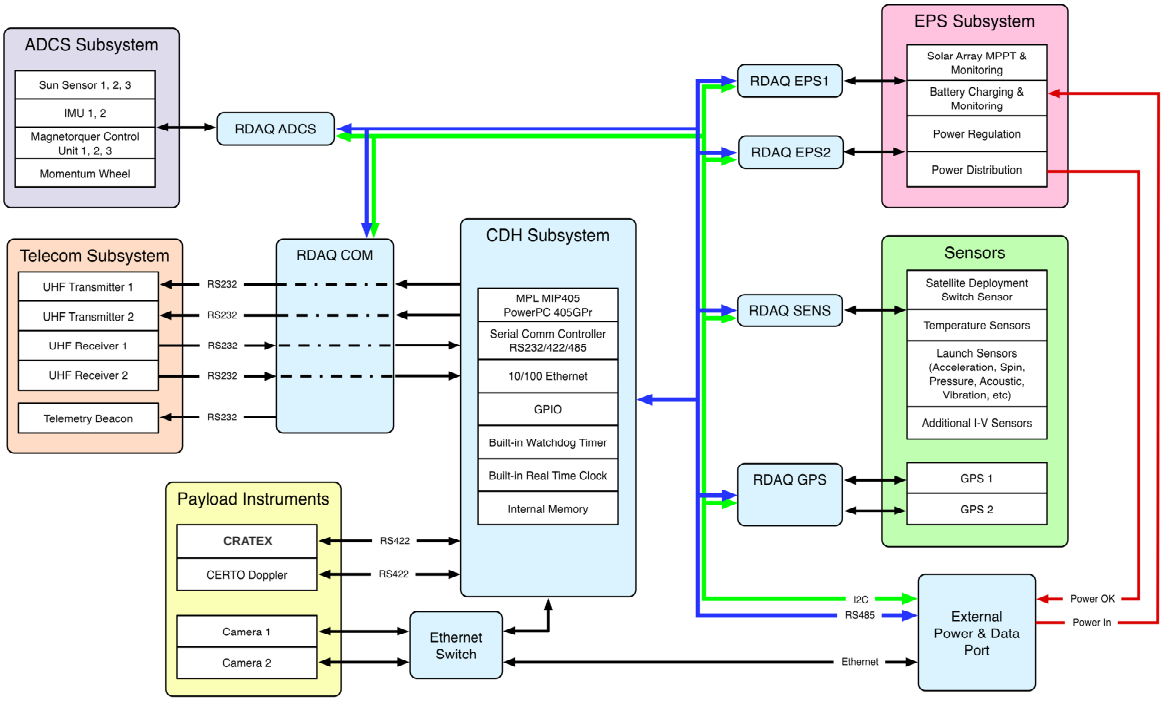


Figure 14. C&DH Block Diagram

3. Electrical Power Subsystem (EPS)

The EPS provides enough power to cover all the spacecraft's needs during all phases of the mission after launch. It uses photovoltaic elements (solar cells) assembled into solar arrays to generate power and secondary batteries to store excess energy until it is required to cover a shortage of power from the solar arrays. The EPS fulfills the power requirements of the fully operating S/C at all times, including during eclipse periods, for the duration of the mission (two years) of the S/C. The spacecraft generates 28 VDC unregulated current which provides 5 VDC and 3.3 VDC on a main power bus to all subsystems as needed. The EPS Functional Block Diagram is shown in Figure 15.

The eight body-mounted solar array panels each consist of two solar array modules (strings) with 19 solar cells per module (Figure 16). To support LEO-1 Solar Array Experiment (LSAE), the arrays are made up of several types of Triple Junction GaAs solar cells – BTJM (28% efficiency), ZTJ (29.5%), XTJ (29.9%), and ITJ (26.8%). Each solar array module has Peak Power Tracking (PPT) controlled by the EPS RDAQ, which also measures the voltage and power of the modules. The solar arrays generate a peak panel power of ~42 W, with average power over an orbit (with losses) of ~25 W. The average power margin over an orbit is ~5.5 W, while in an emergency low power mode the power margin is ~15.9 W.

The LEO-1 batteries are A123 Systems High Power Lithium Ion ANR26650 (LiFePO4) mounted in two arrays providing 242.9 Wh (9.2 Ah @ 26.4 V). These batteries have been flight tested and tested in a HSFL vacuum chamber and are not subject to explosive failures like so other type of Li Ion batteries.

The nominal weekly operations schedule is shown in Figure 17. This determined the loading for the EPS performance shown in Figures 18-21. The nominal S/C attitude is Local Vertical Local Horizontal (LVLH) hold, which keeps the nadir end (+Z) of the S/C always pointed to geocenter.

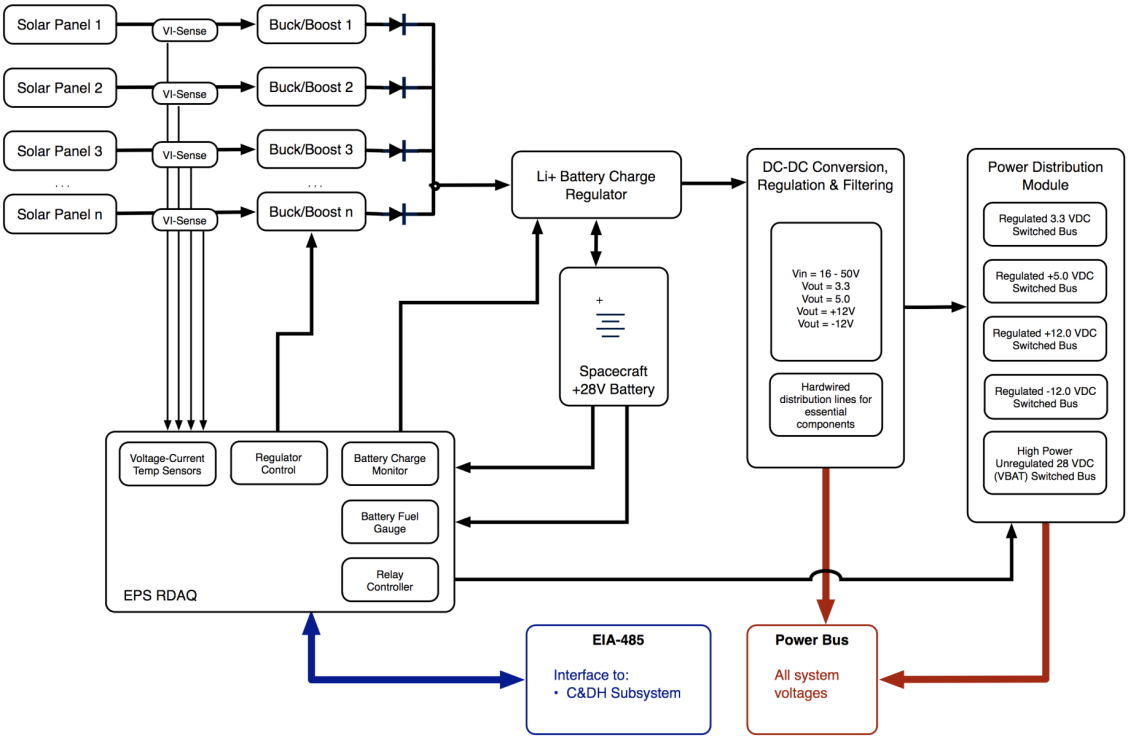


Figure 15. EPS Block Diagram

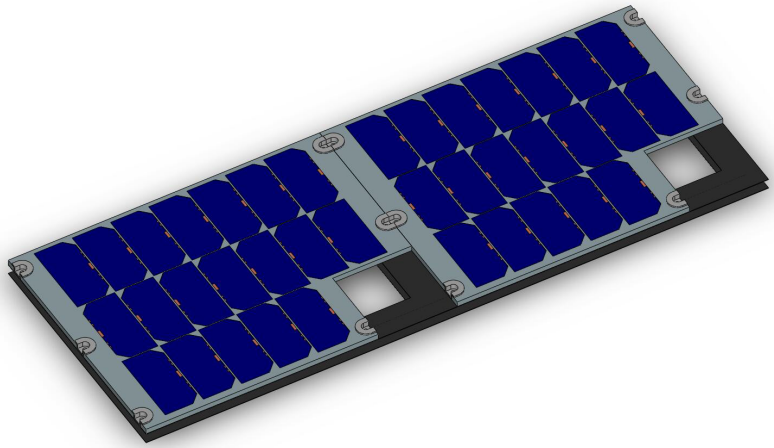


Figure 16. Solar Array Panel Consisting of Two 19-cell Modules

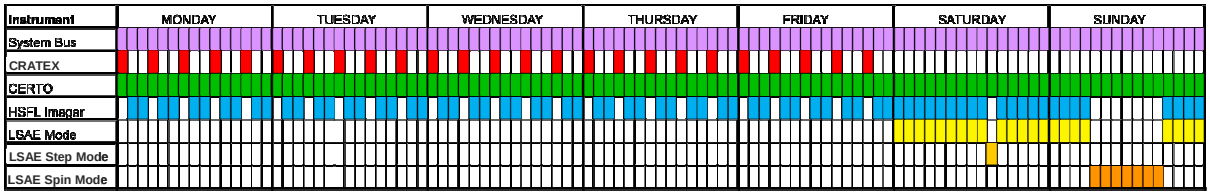


Figure 17. LEO-1 Nominal Weekly Activities

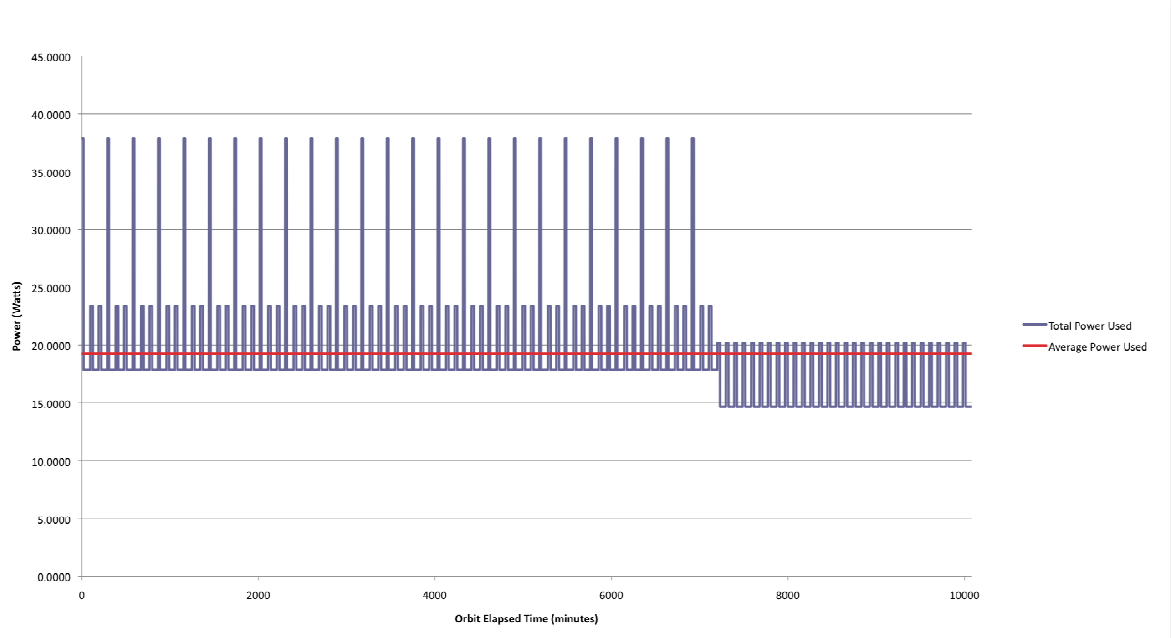


Figure 18. Weekly Payload Power Consumption

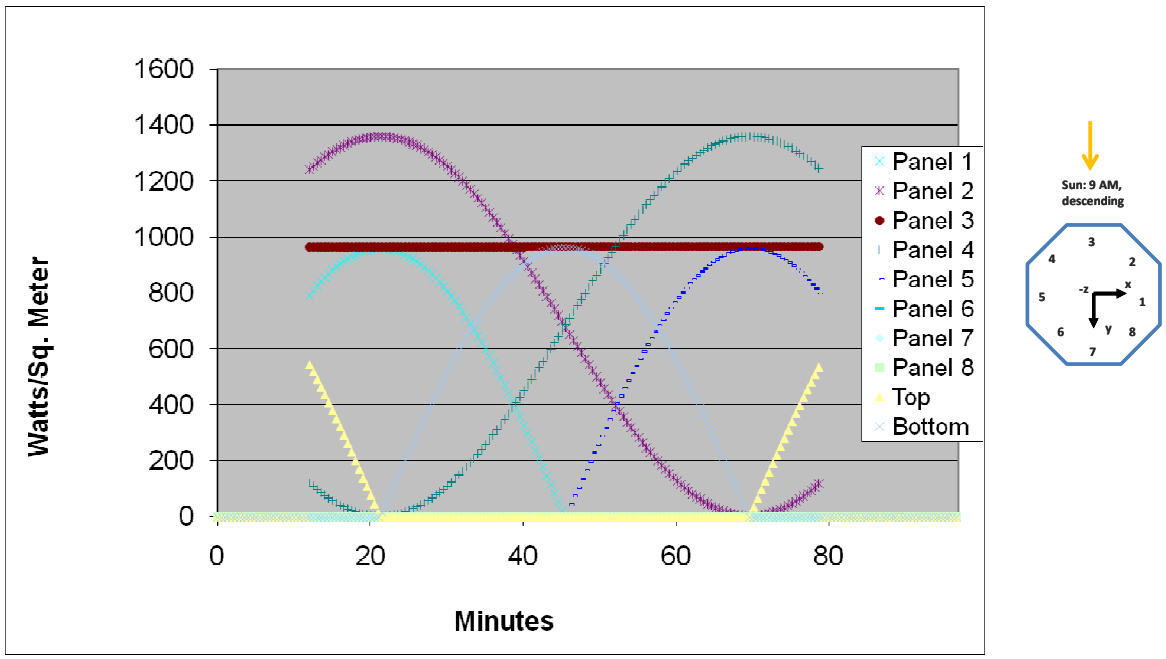


Figure 19. Solar Flux Per Solar Panel (LVLH Mode)

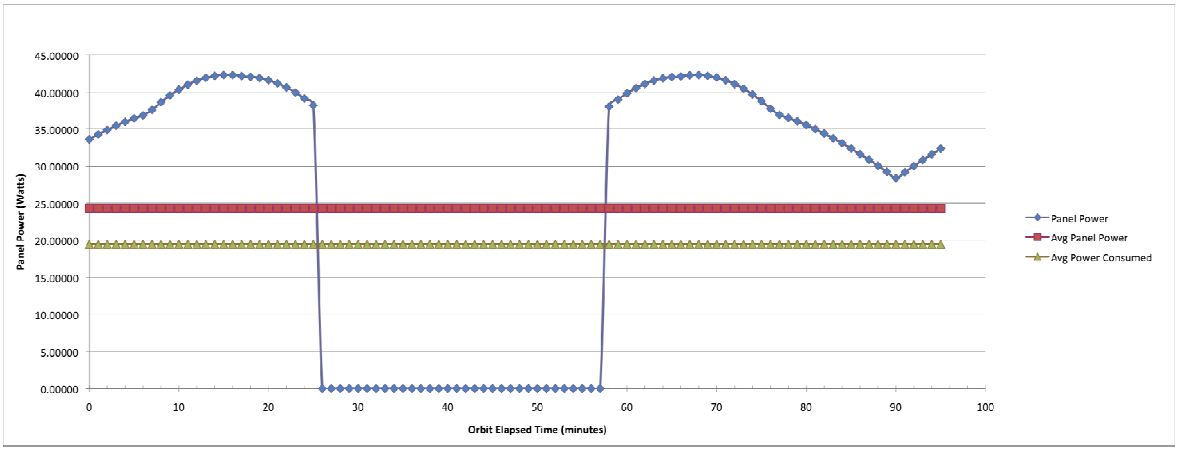


Figure 20. Solar Array Power Over An Orbit

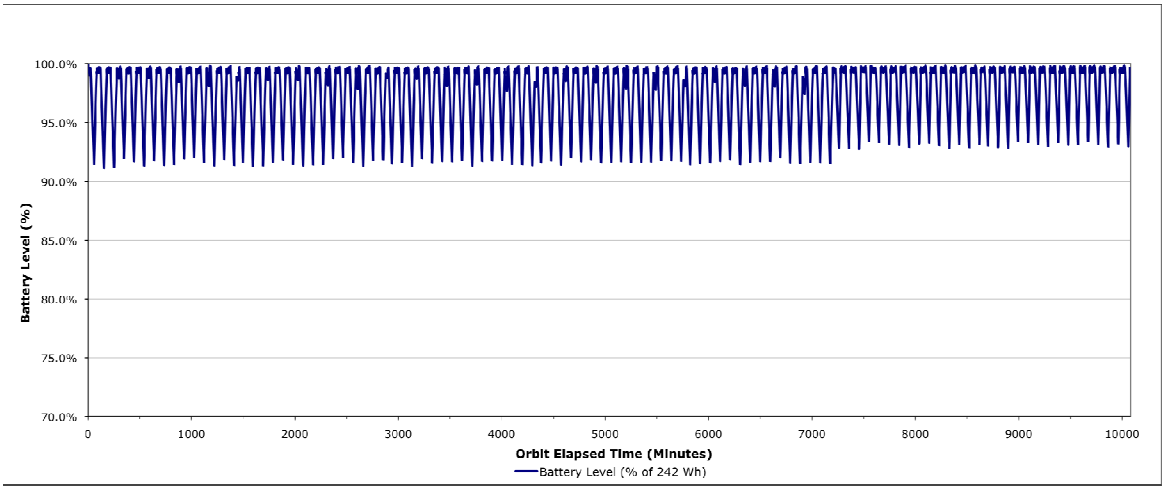


Figure 21. Battery Charge Status Over One Week

4. Telecommunications Subsystem (Telecom)

The spacecraft is capable of two-way communications with the ground regardless of attitude using two redundant UHF full-duplex RF Datatech ART series transceivers to downlink SOH and payload data (excluding the C-band transponders and the CERTO beacon) and to receive uplinked commands and flat files. These transceivers have selectable output power from 50 mW to 5 W, data rate is 150-9600 bps, using 4-level FSK. One receiver is always on (i.e., it is not possible to turn off both S/C receivers at the same time). This is to prevent the spacecraft from accidentally being severed from ground control. The Telecom subsystem is designed to not interfere with payload communications. The Telecom subsystem also includes a simplex VHF beacon that will continuously transmit the spacecraft identification and some essential SOH data. The beacon is a RF Datatech ZRT series radio modem, with selectable power output of 100 mW to 5 W and is configured for 2-level FSK 1200 bps. There are three independent monopole antennas for these transceivers. From the link budget analyses, the UHF downlink margin to the LCC ground station is 18.23 dB, uplink margin of 14.35 dB, and the VHF Beacon downlink margin is 32.05 dB. The Telecom Functional Block Diagram is shown in Figure 22.

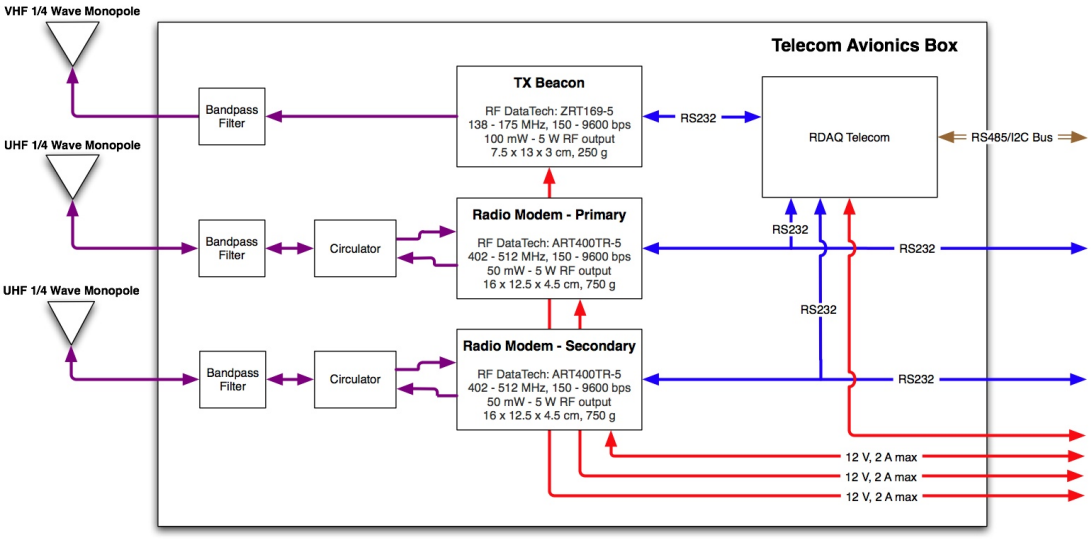


Figure 22. Telecom Subsystem Functional Block Diagram

5. Structures

The structure ensures all subsystem components are housed within the spacecraft's mass and volumetric footprint, and adequately protected from physical forces and environment during all phases of the mission, from integration and testing, through ground transportation, launch, and space flight. The structure also accommodates subsystems mounting and interconnect.

The design of the satellite structure incorporates a number of primary structural members supported by a system of trusses (see Figure 12). The general shape of the structure is an octagon to maximize the power obtained from the solar cells. The solar cell substrate doubles as the side panels for the spacecraft and mount onto the vertical and horizontal members. The upper solar panels mount onto the top surfaces of the frame. The basic material used in the structure is Aluminum 7075 T6. A composite structure was considered, but aluminum was selected because of its reduced cost and risk/complexity. The structure is designed for accessibility, which was part of our criteria for a standard bus baseline. Helping this are removable solar panels (also fulfills an objective of the LSAE), and external electrical/power feedthroughs. The LEO-1 design was also kept mechanism-free for reduced associated risk and complexity.

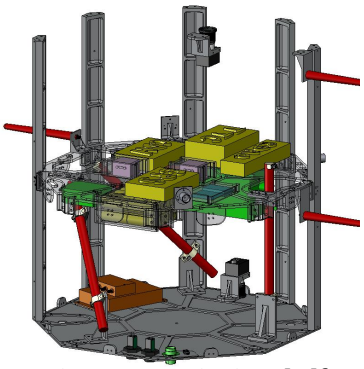


Figure 23. Avionics Shelf

After the System Requirements Review (SRR), a central avionics shelf was added to the S/C as shown in Figure 23. This provided modularity, aided assembly and integration, shortened cable runs, provided additional radiation shielding, and provided stiffening to the S/C structure.

One of the requirements that had to be met in the design of the S/C structure was to not have any object protrude into the field of view (FOV) of the HSFL cameras or the sun sensors. This was accomplished as is shown in Figure 24.

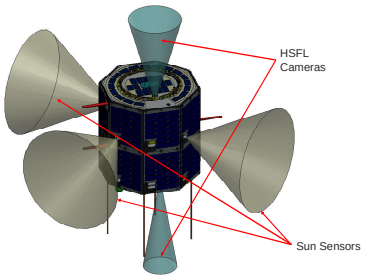


Figure 24. FOV Analysis

The force constraints used during the design of the structure was 16 g axial acceleration, 7 g lateral acceleration, 2.5 rps maximum spin rate, and the launch environment and vibration tests specified in MIL-STD-1540E. This resulted in a limit load of 1.25 times maximum load of 16 g axial and 7 g lateral, 1.25 factor of safety with respect to yield, and 1.40 factor of safety with respect to ultimate. Extensive finite element analyses (FEA) were performed on the LEO-1 structure. The analyses performed included:

static (Von Mises Stress), frequency (up to 15th mode), and thermo-mechanical. An example is shown in Figure 25. The maximum displacement of about 4 mm is considered to be too much and a stiffening of the structure was done post PDR, although the latest information from the launch vehicle designers indicate that the loads will be less than half of what was used in the initial design.

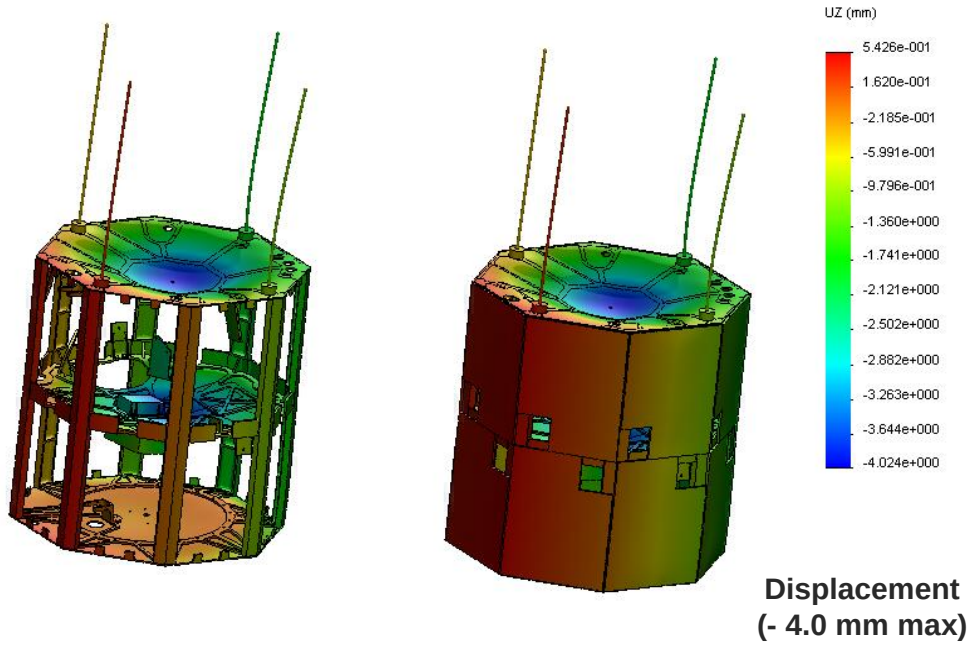


Figure 25. FEA Showing Axial Displacements

6. Attitude Determination and Control Subsystem (ADCS)

The spacecraft is three-axis stabilized and capable of autonomous, closed-loop inertial pointing with an accuracy of 5 deg. or better. Attitude measurement accuracy is adequate to determine where the spacecraft is pointing to 1-2 deg. This accuracy is achievable in real-time, in darkness or sunlight, and during all phases of the flight after deployment.

The ADCS provides the spacecraft with the capability of maintaining the +z face of the spacecraft pointing in a nadir (geocentric) direction with an accuracy of ±5°. This is the attitude required for the C-band transponder, CERTO, and imager payloads. The operational modes of the ADCS are given in Table 3. Most of the mission is spent in the nominal (LVLH Hold) mode with the nadir (+Z-axis) pointing towards geocenter. However, the spacecraft has the flexibility to point to any inertial attitude and even to spin when required. It is also desirable, but not mandatory, for spacecraft to be capable of pointing to and tracking a stationary Earth target. The ADCS is also responsible for maintaining zones of exclusion to prevent the S/C from pointing imagers towards the sun where it could result in damage to

Table 3. ADCS Operational Modes

MODE	DESCRIPTION
Deploy	ADCS inactive after deployment separation
Tumble	Attitude control disabled to allow tumbling (e.g. Lost mode)
Detumble	To some fixed inertial attitude
Nominal	LVLH hold attitude mode with nadir facing Earth and zenith facing Space (along outward radial) and +X-axis pointing along velocity vector – this mode is used for CRATEX contacts
Zenith Imaging	LVLH hold attitude mode with zenith facing Earth and +X-axis pointing along velocity vector
Special	Non-nominal mode for special tasks (e.g., Earth pointing, inertial pointing)
Sun Lock	To attitude that maximizes solar array charging
Safe	To attitude that maximizes power generation and communication coverage
Analysis Spin	1 rpm spin-up and down, for all solar arrays to reach thermal uniformity (± 3° C)

them. The ADCS software, including algorithms and settings, are accessible to the ground and can be updated or replaced from the ground during the flight if needed.

Attitude data obtained from the spacecraft's ADCS sensors are autonomously processed aboard the spacecraft to determine spacecraft orientation. The attitude determination is accomplished using three space-qualified Aero Astro MSS-01 Sun Sensors, each with 60° full-angle circular FOV providing an accuracy of ~1°, and two Microstrain 3DM-GX2 inertial measurement units (IMUs), each of which contains a 3-axis magnetometer, three accelerometers, and a rate gyroscope.

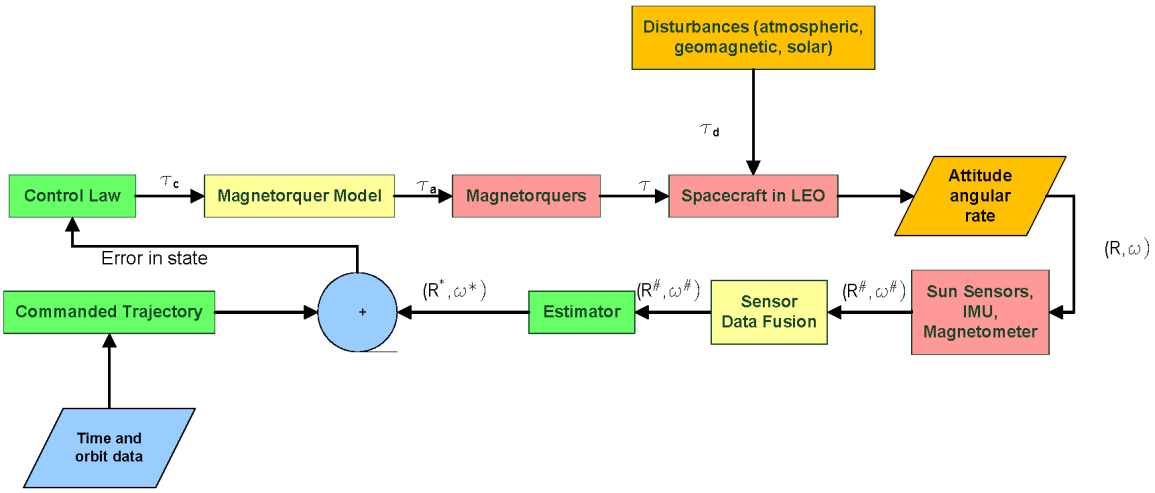


Figure 26. ADCS Functional Block Diagram

The ADCS Functional Block Diagram is shown in Figure 26. The commanded control torque vector and the input vector to the actuators (magnetorquers) are related by: $B(R)\tau_a = \tau_c$. R is the attitude, τ_a is the actuator input vector, τ_c is the commanded torque vector (from the control law). $B(R)=[B_1(R)\times B_g\ B_2(R)\times B_g\ B_3(R)\times B_g]$ with three magnetorquers, where $B_i(R)$, i=1,2,3 are the magnetorquer magnetic fields that depend on the attitude, and B_g is the local geomagnetic field. Whenever one of the magnetorquers is aligned closely with the local geomagnetic field, $B(R)$ cannot be inverted to get the τ_a from the τ_c. The controller has to switch to a slower-acting underactuated control law in this situation. Therefore a small reaction wheel was added for full three-axis attitude actuation capability.

The ADCS provides attitude control by the use of control actuators, which for LEO-1 are three magnetorquers and a small reaction wheel. The magnetorquers are space-qualified Vectronic VMT-35 which each have a magnetic moment of 18 Am2 @ 100 mA and are controlled by a Vectronic Torque Control Unit. The reaction wheel (RW) is a space-qualified Sinclair RW-0.03-4-ASYNC-2-1-0 which has a nominal torque greater than 2 mNm and nominal momentum of 30 mNm-sec @ 5600 rpm. The layout of the ADCS actuators and components is shown in Figure 27.

The ADCS control scheme is nonlinear and continuous, based on feedback of trajectory tracking errors. Control law τ_c designed for global tracking of desired attitude and angular velocity trajectories.[4]

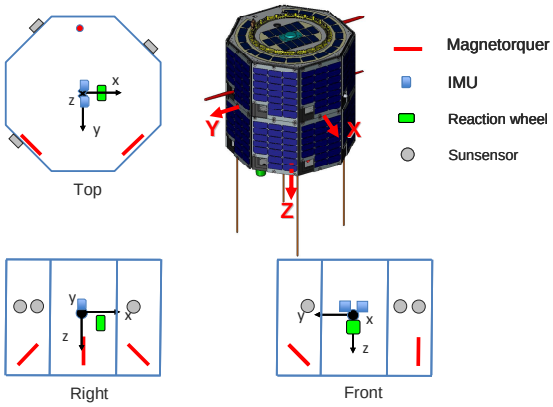


Figure 27. Location of ADCS Components

The actuator model developed generates inputs to actuators (RW and magnetorquers) and the control torque τ_a applied by the actuators. This control torque allocation scheme, for a RW along the pitch axis, is given in Reference 5. This scheme is singular when the local geomagnetic field is perpendicular to the pitch (RW) axis.

The ADCS estimation and filtering scheme is nonlinear and discrete, based on deterministic ellipsoidal bounds on measurement errors and dynamic flow uncertainty. Measurements are: direction vectors (for attitude) and angular velocity vector, all in spacecraft body frame. Estimation scheme for an orbiting satellite without control torques given in Reference 6. Propagation of attitude and angular velocity between measurements carried out using a Lie group variational integrator. Estimation scheme for an orbiting satellite, with the inclusion of a feedback control torque, is being developed and prepared for publication.

The nominal (LVLH Hold) attitude mode requires a pitch rate of 0.063 º/s. The ADCS pitch rate capability is 0.286º/s. ADCS control software simulations suggest it would require about 190 seconds (less than 1/30 of an orbit at 600 km altitude) to rotate the satellite by 30º (to 0.4º accuracy), beginning with an initial attitude error of about 15º, an initial angular velocity error of about 1.15º/s, and a maximum torque <1.8 milli-Nm. De-tumbling maneuver followed by stabilization of nadir-pointing attitude carried out in ADCS simulation shows satisfactory de-tumbling from an initial attitude error of 30º and initial angular velocity of 0.098 rad/s within 1/20 of an orbit at 600 km altitude.

7. Thermal Control Subsystem (TCS)

The TCS is designed to ensure all S/C subsystem components are thermally controlled for operation, both in sunlight and eclipse periods. Due to the fairly benign thermal environment of low Earth orbit, passive thermal control was determined to be sufficient. The internal power dissipation required to be handled by the TCS is from a minimum of 14 W to a maximum of 90 W. The TCS hardware consists of multi-layer insulation (MLI) blankets on the interior S/C panels, appropriate surface finishing (e.g., paints), five thermostats with redundancy, heaters for critical components, temperature sensors, and a radiator. The TCS Functional Flow Block Diagram is shown in Figure 28.

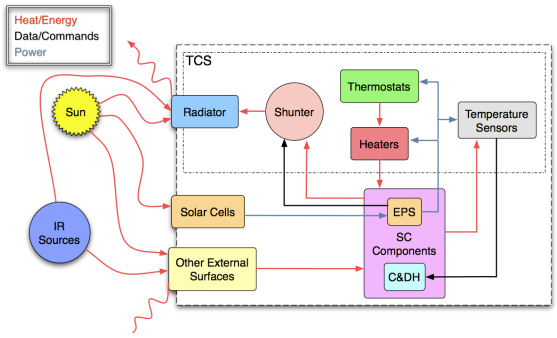


Figure 28. TCS Functional Flow Block Diagram

Detailed simulations were performed of the spacecraft including all sub-systems and payload using Thermal Desktop® using finite differences. Simulations were done for a nominal hot case and a cold case. Trades studies for the thermal management changed the layout of internal components.

One of the LSAE tests require the spacecraft obtain temperature uniformity of the solar array panels by spinning. Simulations were also done of the spinning spacecraft for this test of the solar cells. The results are shown in Figure 29 and show that thermal uniformity of the solar panels can be achieved after only about 30 minutes with a spin rate of 1 rpm. After achieving thermal uniformity, the spacecraft has to continue the experiment for one orbit.

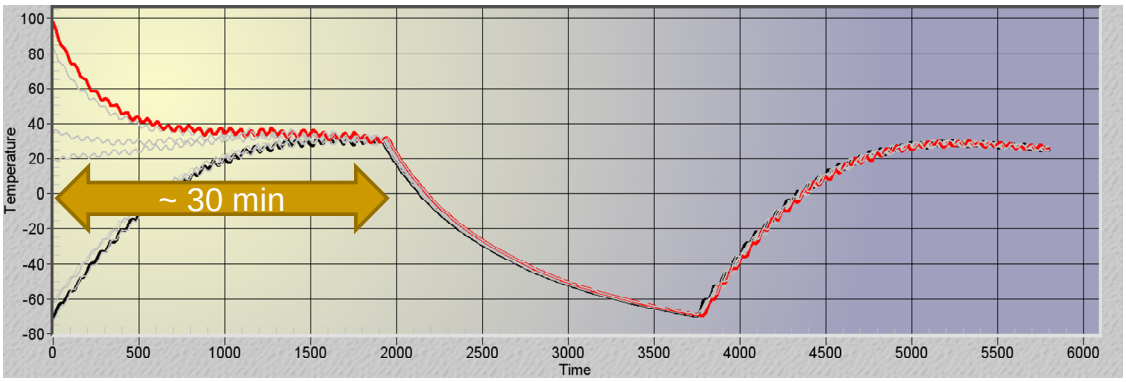


Figure 29. Solar Array Panels Temperature With S/C Spinning at 1 rpm

8. Flight Software(FSW)

The flight software autonomously monitors and maintains the spacecraft's state of health, controls the operation of the spacecraft, monitors the SOH of the payloads, performs the calculations for the ADCS, and can activate or deactivate the payloads. The FSW will always be recoverable from loss of power or function without ground support. It can also be reloaded or modified from the ground without endangering the spacecraft SOH or mission. The FSW architecture is shown in Figure 30.

The FSW operates as a state machine, where the spacecraft operational modes are defined as states. The Image of State is contained in shared memory. There is a subprocess for each important subtask, with the Executive Process controlling system as a whole. Commands are implemented through successive changes in state commands stored as a linear sequence in time on the command queue.

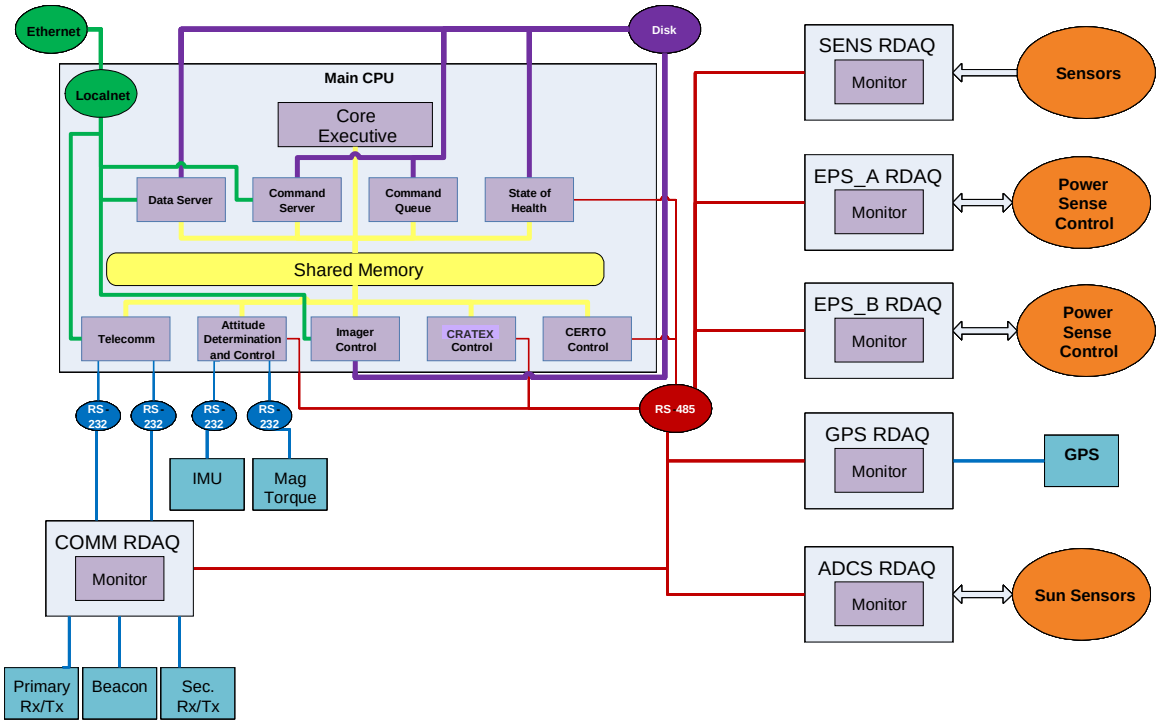


Figure 30. Spacecraft Flight Software Architecture

The FSW performs the navigation calculations for the spacecraft to generate an orbit and position that is used by other subsystems. The current position of the spacecraft can be determined on the ground using the signal from the CERTO Doppler Beacon and then uplinked to the spacecraft as either an ephemeris or state vector. A backup method is to uplink a two-line element (TLE) obtained from the NORAD website. An autonomous method for determining the orbital position is to process onboard the data obtained from the NovaTel GPS unit. Thus the S/C will be able to determine its position even if it is out of touch with the ground for extended periods. The FSW operating system is the QNX Neutrino Real Time OS, which was selected for its stability and support for the MIP 405 C&DH architecture.

IV. Payloads

This section describes the payloads being carried by the LEO-1 spacecraft.

A. LEO-1 Solar Array Experiment (LSAE)

The LEO-1 spacecraft is performing the LEO-1 Solar Array Experiment (LSAE) to test new solar array technologies, including a modular solar array technology developed at DR Technologies, Inc.[7] The technology has Operationally Responsive Space (ORS) applications and would benefit from a flight demonstration. One objective of the LSAE is to flight test plug-and-play solar panel modules and the LSAE incorporates conventional as well as advanced technology photovoltaics that could be used in future space missions.

The LSAE uses high efficiency (~29%) solar cells that are laid out in modules (strings) of 19 cells, with two modules per side panel. There are eight such panels on the spacecraft as shown in Figure 31. Panels 1-5 contain mostly standard technology solar arrays and provide power for the periods when not in a LSAE test mode. The remaining three panels (6-8) contain the experimental modules. Some of the cells have experimental coatings on their glass covers.

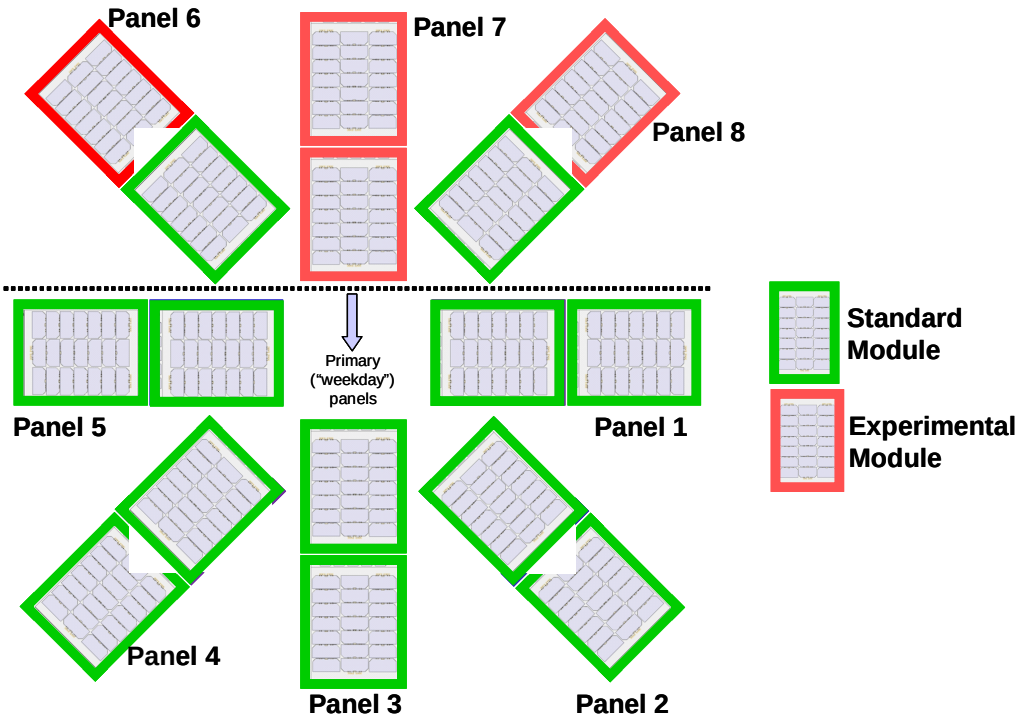


Figure 31. Solar Array Module Arrangement to Support LSAE

During the weekdays, the spacecraft will be oriented such that five of the arrays provide power throughout the sunlit portion of the orbit (indicated as Weekday Arrays in the figure). Each weekend (assuming no other spacecraft operations are planned), the spacecraft will be rotated about the z-axis (major axis of the cylindrical spacecraft) by 180 degrees in order to accumulate exposure time on the three arrays normally shadowed. Approximately once per month, the weekend configuration will include a slow spin about the z-axis (~1 rpm). These LSAE tests are described in more detail in the Operations section.

As part of its normal SOH data collection, the spacecraft collects and stores the following data for each solar array module (string): string current, string voltage, panel temperature, sun sensor readings, and time stamp. SOH data are downlinked to ground stations when possible.

B. C-Band Radar Transponder Experiment (CRATEX)

The CRATEX experiment payload is designed to flight test a replacement to the RADCAL payload currently operating in a degraded fashion on the RADCAL satellite, which was launched in 1993 from Vandenberg Air Force Base (VAFB) with an expected lifetime of one year. LEO-1 with the CRATEX payload will test some new technology that can be used to support ground based Range Instrumentation's performance monitoring functions. Ground based Range Instrumentation users are minimally comprised of Tri-services and NASA Test Ranges. Although the optimal satellite orbit is 850 Km altitude, with an inclination of 65° or higher, the LEO-1 orbital altitude of 600 km in a Sun-Synchronous (inclination ~98°) is considered by VAFB to be satisfactory for testing purposes. A C-Band transponder is used to calibrate the ground radars. An orbit determination system is required that achieves a minimum of 5 meter accuracy. The orbit may be determined on-board, on the ground or through a combination of space and ground calculations, but must be available to users within 48 hours. The primary method used on LEO-1 for orbit determination is the CERTO Doppler system and the backup method is a NovaTel GPS receiver.

VAFB is supplying two C-band transponders – the space-qualified SST-177C (currently being flown on DMSP 15 satellite) and the transponder to be flight-tested in space, the MD2000C (Figures 32 and 33 respectively). Two AntDevCorp Quadrifilar Helix C-Band antennas are being used. Both transponders receive ground C-Band radar signal of 5765 MHz and respond coherently with 5765 MHz on the SST 177C or at 5690 MHz from the MD2000C. This allows monitoring of the tracking performance of C-Band radars, while testing the performance of the MD2000C in comparison to the older technology of the SST 177C. Radar data (TRAE) is compared with a known satellite position (precise ephemeris) obtained from processed Doppler data or GPS data. A coherent transponder is necessary for radars equipped with range rate subsystem. The satellite provides a dynamic target for testing the total radar system. C-Band transponders are normally off and must be commanded on just prior to requested use and off again immediately after each requested use. Track period is software limited to a maximum of 20 minutes. Transponders are routinely activated on an average of 6 times a day.



Figure 32. SST 177C C-Band Transponder



Figure 33. MD2000C C-Band Transponder

Ephemeris must be determined to better than five meters accuracy. The CERTO Doppler Beacon is the primary means of determining precise ephemeris on LEO-1. It achieves less than 5 meters accuracy; the average 2004 Orbit Overlap Total Magnitude RMS was 2.2 meters. The Doppler transmitter remains on at all times. Satellite Doppler data are automatically collected and stored at numerous geodetically distributed sites. Once a day, the data are transferred to Western Range VAFB for initial editing and processing. The edited data are then forwarded to the National Geospatial-intelligence Agency (NGA) for final processing and posting of resultant ephemeris on RPMWEB site for user community access.

C. Coherent Electromagnetic Radio Tomography (CERTO) Experiment

The primary purposes of the CERTO Experiment provided by the Naval Research Laboratory (NRL) are to detect when and where radiowave propagation through the ionosphere is adversely affected by scintillation and refraction and to provide a global map of ionospheric densities and irregularities to improve current models of the ionosphere. The CERTO Experiment makes use of the Doppler beacon required by the CRATEX payload for precise orbit determination. The beacon signal provides a carrier wave transmission to the receiving ground stations, which use the shift in frequencies to help determine the orbit. However, the radio signal gains noise passing through the ionosphere, which is the

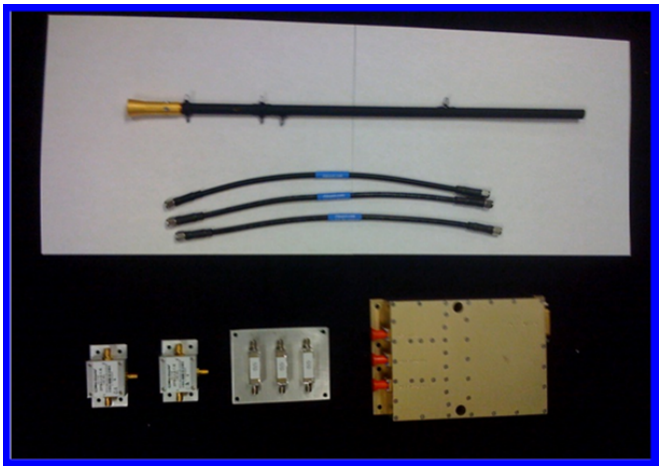


Figure 34. CERTO-R2 Beacon and External Components

phenomenon that is being investigated by the CERTO Experiment. The noise is separated from the carrier signal at the ground station and the resulting carrier signal is used to determine the orbit for the CRATEX experiment and the noise is used by the CERTO Experiment to determine the ionospheric characteristics.

The CERTO-R2 beacon (Figure 34) is dual frequency – 150 MHz (VHF) and 400 MHz (UHF) and each transmission uses its own antennas (four for the 150 MHz and one for the 400 MHz). The phase and frequency fluctuations of the VHF and UHF signals help in the monitoring of scintillations in the ionosphere and resulting regional maps of radio signal disruptions. By tracking the CERTO signals as LEO-1 passes over the ground station allows data from oblique and vertical paths through the ionosphere to be obtained. Additional data are obtained from GPS satellite occultations. The resulting data are used for reconstructions using computerized ionospheric tomography.

NRL wants the CERTO beacon to be operating continuously during the mission if at all possible, but the beacon can be turned to lower power standby mode or off if spacecraft emergency or power state requires it.

D. HSFL Imagers

The HSFL Imagers provide imagery that fulfills a secondary mission objective and will be used by HSFL for public outreach and to inspire interest in science education. Color images of the Hawaiian Islands with little or no cloud cover should provide good public relations value to HSFL and its sponsors. A final product of this experiment is a color image showing the Hawaiian Islands with little or no cloud cover. This can be done with either a single image or a mosaic of images. Two cameras are included in LEO-1. One is a nadir-pointing (in normal LVLH Hold mode) and the other is zenith pointing. This latter imager fulfills a mission requirement to document the separation of LEO-1 from the PAD during orbital deployment, but it would subsequently be a backup camera for the nadir imager.

The Nadir Imager is a Prosilica GE2040C color 2048 x 2048 (4 MP) CCD image sensors with a narrow FOV (NFOV) lens placed on nadir-pointing side of spacecraft (see Figure 35). Its frame rate is up to 15 FPS. The lens selected for this camera is a Zeiss 85mm f/1.4 ZF Planar T manual focus lens with a FOV of 28.5°. Its focus range is 1 m to infinity and it has aperture range of f/1.4 – f/16.

The Zenith-pointing imager is a Prosilica GE1050C 1024 x 1024 (1 MP) color camera with WFOV lens to capture spacecraft deployment from PAD. Its frame rate is 60 FPS. The lens selected for this camera is an Edmund Optics Varifocal 4.0-12 mm manual focus lens with a FOV of 93.6° to 31.2°. Its focus range is 0.3 m to infinity and it has aperture range of f/1.2 – f/16.



Figure 35. Prosilica GE2040C

V. Operations

This section describes the operations concept for the LEO-1 mission.

A. Operations Concept and Phases

The LEO-1 on-orbit mission is divided into three phases.

1. Engineering Evaluation & Checkout to Initial Operational Capability

The Engineering Evaluation & Checkout (EE&C) is the checkout period of the LEO-1 after orbit insertion and is expected to last about a month (or as long as necessary to complete the checkout). This phase concludes with its commissioning when it has achieved initial operational capability (IOC). During the EE&C the following tasks are performed:

- Testing of spacecraft bus subsystems and modes.
- Testing of ground segment and data flow.
- Testing of CRATEX & CERTO payloads:
  - Both C-band transponders and CERTO beacon tested
  - Orbit determination method tested for 5-m accuracy
  - Test interfaces with USAF for tasking and delivery of orbit data
- Testing of LSAE payloads:
  - Experimental solar panels and data acquisition tested
  - LSAE operational modes tested
  - Heat transfer model verified for various LSAE modes
- Testing of HSFL Imagers:
  - Take images using real-time and delayed commands using all cameras with storage and downlink of images.

2. Nominal Operations

Nominal (CRATEX) Mode

The nominal mode for the LEO-1 spacecraft (i.e., the mode in which is spends most of its time) is to support the CRATEX and CERTO payloads. This mode has the following characteristics:

- LEO-1 in LVLH hold (+X or –X forward) attitude (see Figure 36).
- Each ground radar requires calibration at least once per week (using Monday to Friday schedule).
- A weekly schedule request is provided by VAFB (30 Space Wing) >3 days in advance. HSFL integrates requests with the overall LEO-1 schedule. The final CRATEX contact schedule is provided to 30 SW by the HMOC at least two days in advance for nominal commanding.
- A C-band transponder is turned on and off by time-delayed script before and after each calibration contact (script nominally uploaded >1 day in advance). Only one transponder and one antenna are active at a time. Transponders and antennas 1 and 2 are used in alternating months. Transponder 1 (SST 177C) is the reference transponder and Transponder 2 (MD2000C) is the experimental transponder.
- There are a maximum of five calibration passes per day (each of 20 minutes maximum duration limited by watchdog timer).
- The target ground radar activates the C-band transponder and executes a calibration contact. The CERTO Doppler signal is also received by some ground stations.
- HSFL sends GPS position data to VAFB if required (backup mode).
- Ionospheric characterization data extracted from Doppler data are obtained by target ground stations from the CERTO beacon (which is on continuously).

LSAE Modes

During the nominal CRATEX operations period (Monday-Friday), no LSAE activities are scheduled except routine monitoring of solar array performance (solar array SOH data collected and stored at 4 minute intervals). During weekends with no CRATEX contacts, the spacecraft is rotated spacecraft 180° about z-axis (yaw) to put –x axis forward while maintaining LVLH hold attitude This exposes the experimental panels 6, 7, and 8 (refer to Figure 31) to sunlight and puts the standard solar arrays 2, 3, and 4 into shadow during the orbit. Solar array SOH data still collected and stored at 4-minute intervals. This mode, called the *LSAE Mode*, is shown in Figure 37.

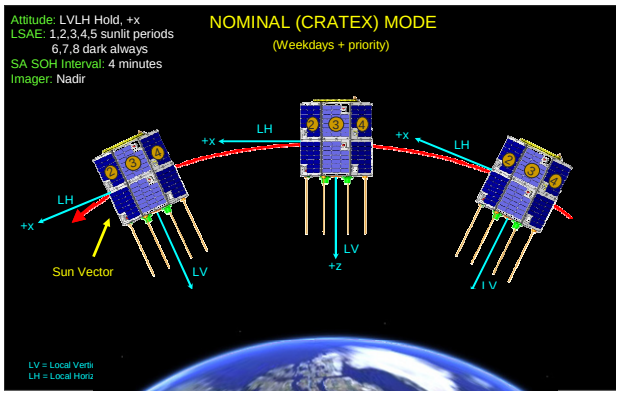


Figure 36. Nominal (CRATEX) Mode

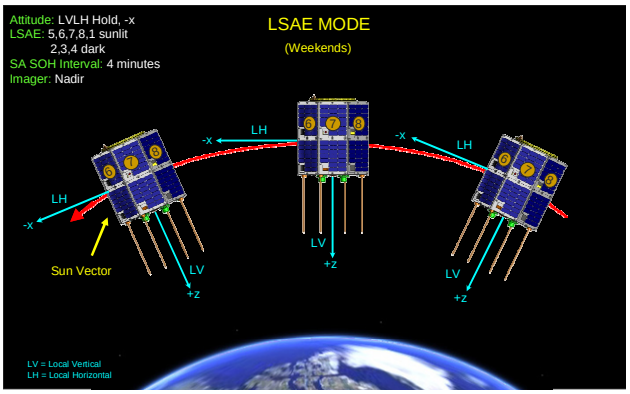
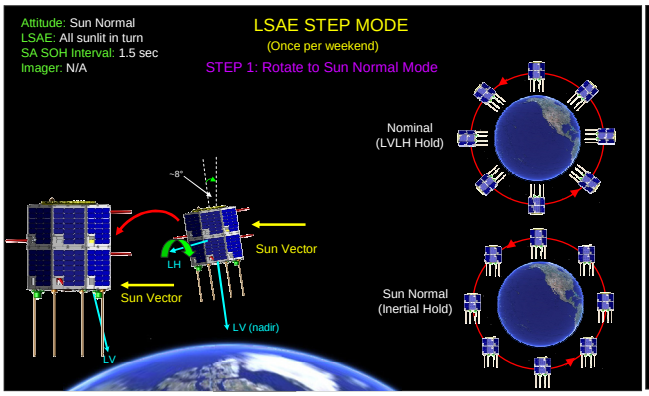
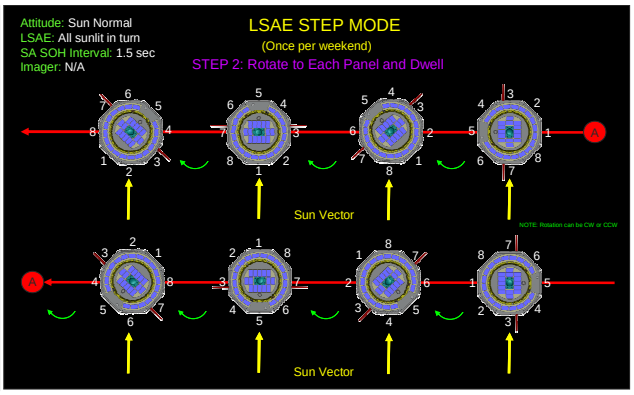


Figure 37. LSAE Mode



a) Rotate to Sun Normal Mode



b) Rotate to Each Panel and Dwell

Figure 38. LSAE Step Mode

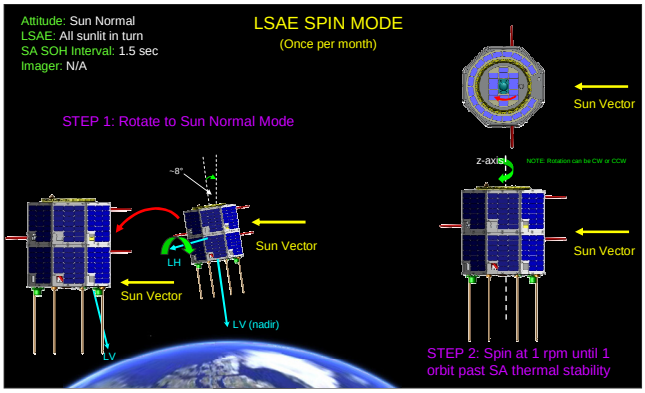


Figure 39. LSAE Spin Mode

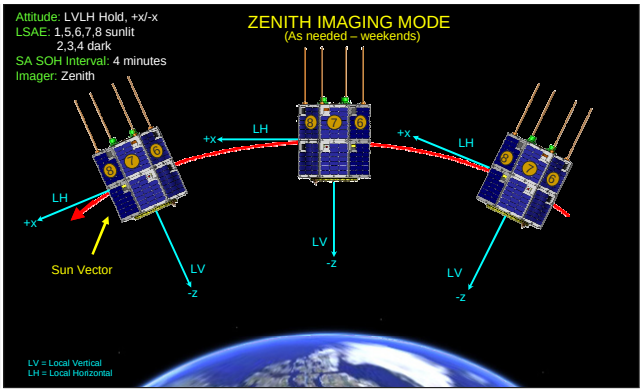


Figure 40. Zenith Imaging Mode

Once a weekend, if possible, the spacecraft collects data for a current-voltage (I-V) curve for each SA module. This is called the *LSAE Step Mode* and is shown in Figure 38. To execute this mode, the following actions are done:

a) Spacecraft goes to inertial attitude with sun normal to side panel
b) Spacecraft points each of the eight sides normal to the sun (beta angle of zero) for long enough to allow for settling of data.

Once a month the spacecraft goes into a slow (1 rpm) spin to achieve thermal stability of the solar arrays (all solar arrays within 3 degrees Celsius) to determine their relative performance with a constant temperature. This mode is called the *LSAE Spin Mode* and is shown in Figure 39. To execute this mode, the following actions are done:

a) Spacecraft goes to inertial attitude with sun normal to side panel
b) Spacecraft goes to an attitude with the sun vector normal to the spacecraft z-axis
c) Spacecraft starts 1 rpm spin about the z-axis
d) Once thermal stability achieved (all solar arrays +/- 3° C) spacecraft completes one orbit with spin. Thermal analysis has shown that this thermal stability can be achieved in less than one orbit. Data are collected at 1.5 second intervals.

Earth Imaging

In Nominal Mode the nadir camera takes images of the Earth. This camera has a comparatively narrow FOV. The similar camera on the Zenith end of the spacecraft that is used to image the separation of the LEO-1 from the launch vehicle, can also be used as a backup for the Nadir imager for obtaining images of the Earth, although with lower resolution and a wider-angle lens. In order to accomplish this, the spacecraft has to be flipped upside-down, so that the zenith end is pointing to the Earth. This mode, called the *Zenith Imaging Mode*, is shown in Figure 40.

Images can be taken by time-delayed script or real-time command on a non-interference basis. They will be stored onboard and then downlinked when convenient.

Nominal Payload Operations Schedule

Having determined the various operational modes to fulfill the objectives of the payloads, a weekly schedule is formed that ensures that there are no conflicts between the spacecraft bus and the various payloads. A nominal weekly schedule showing the operations of the various payloads and modes is shown in Figure 41.

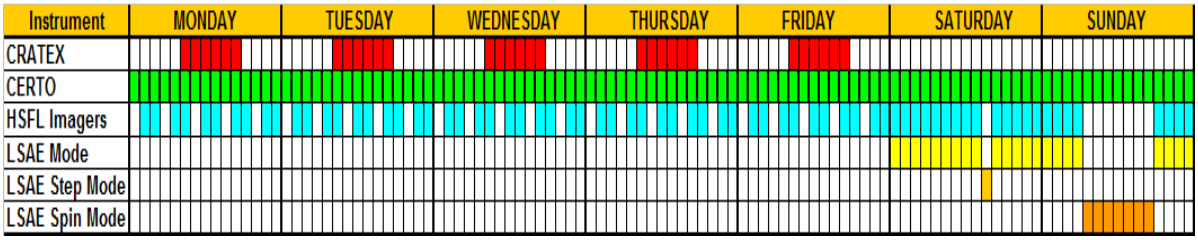


Figure 41. Nominal Payloads Weekly Schedule

3. Mission Termination

New missions launched into LEO have to have a means to dispose of the satellite within 25 years of the mission termination in order to remove it as potential space debris. This can be done either using a system to change the orbital velocity of the satellite (e.g., using propulsive or drag devices), or by natural orbital decay. For LEO-1, which has no propulsion subsystem, the latter method was selected. The current analysis shows that with an initial altitude of 600 km, that the satellite will reenter within 25 years of the nominal mission end.

At the end of mission, the spacecraft payloads and transmitter will be turned off. There is no further action required before reentry into the atmosphere.

B. Operational Modes

A number of operational modes have been identified for the LEO-1 spacecraft to accomplish its mission. These operational modes are identified and described in Table 4. There is a close correlation between these modes and the ADCS modes that were identified in Table 3.

C. Operations Staffing and Process

The basic staffing to support operations of the LEO-1 mission after launch is shown in Table 5. The basic operations philosophy of the LEO-1 mission is to use the spacecraft engineers as operations personnel during the EE&CO period of the first month until commissioning. During *nominal* operations, the engineers will be working other projects and will be used to support operations as needed (e.g., anomaly resolution) or to just a very low level. Most of the operations will be performed by the spacecraft controllers, which will be mainly students. This also provides an educational benefit to the project, to familiarize students with the operation and workings of an operational satellite. The spacecraft is being designed for mostly autonomous operation, including automatic contacts with the ground where command files are uplinked, and data files are downlinked with human intervention.

Table 4. Spacecraft Operational Modes

Spacecraft Mode	Description	Functions/Comments
TEST	S/C is commandable for all subsystems	• Power external or internal
LAUNCH	S/C has active launch sensors to characterize the launch environment. Other subsystems are off.	• EPS on batteries and LV power • RDAQ and environmental sensors on • All other subsystems off
DEPLOY	S/C bus active except ADCS and customer payloads. Records deployment separation event including imaging.	• Mode initiated by microswitch during physical separation • C&DH and EPS nominal • CPU does standard boot up with subsystem checks and then determines whether to go to DEPLOY or SAFE mode • ADCS is on except for actuators - determine position and attitude • Customer payloads OFF • Images separation with zenith imager
START-OPS	Separation imaging discontinued and angular rates zeroed on all three axes and then kept in inertial attitude hold until attitude lock obtained and ADCS mode is changed	• ADCS goes into DETUMBLE mode (zero out rates) • Payloads and TCS turned OFF
TUMBLE	Attitude control disabled to allow tumbling	• ADCS actuators OFF - RDAQ and sensors may be ON or OFF
NOMINAL	Spacecraft in LVLH Hold attitude with nadir face pointed towards Earth and zenith face pointed towards space with +X axis normal to nadir in velocity direction - used for RADCAL II contacts	• Normal operational attitude (nadir pointing LVLH) • Normal payload functions
ZENITH IMAGING	Spacecraft in LVLH Hold attitude with zenith face pointed towards Earth and zenith face pointed towards space with -X axis normal to nadir in velocity direction	• Special attitude to enable zenith camera to image Earth • RADCAL and CERTO not operational • Other subsystems normal
SPECIAL	Spacecraft in non-nominal mode for special experiments or engineering tasks (e.g., LSAE)	
LOW POWER	Turns off non-essential loads as needed to improve power situation. When power is restored and passes threshold it goes to next higher level – from LOW POWER mode it goes to SAFE mode	• Minimal power saving
CRITICAL POWER	Turns off non-essential loads as needed to improve power situation. When power is restored and passes threshold it goes to next higher level - from CRITICAL it goes to LOW POWER mode	• Maximum power saving
SAFE MODE	Maximizes power generation and communication coverage	• CPU does standard boot up with subsystem checks and then determines whether to go to DEPLOY or SAFE mode

Once the mission has achieved a certain level of maturity in which there is a high level of confidence that the automated systems will work, then the mission can enter the *mature* operations, which requires minimal staffing by operations personnel. Only a single 8-hour shift for five days a week is anticipated for nominal and mature operations, while the EE&CO period will require operations personnel for seven days a week. Although the nominal and mature operations are only five days a week, they will usually not correspond to Monday to Friday, since the LSAE modes, which will require supervision, are done on weekends.

Table 5. LEO-1 Operations Staffing

Position	EE&C FTE	Nominal FTE	Mature FTE	Shift h/d	Comment
Operations Manager	1	0.5	0.1	8/5	
Spacecraft Controller	2	0.9	0.1	8/5	8/7 for EE&CO
Ground Network Controller	1	0.1	0.1	8/5	
Scheduler	1	0.5	0.5	8/5	
Mission Planner	1	0.5	0.5	8/5	
Data Manager	1	0.5	0.1	8/5	
S/C Analyst	4	0.5	0.1	8/5	8/7 for EE&CO
Software Analyst	2	0.5	0.1	8/5	8/7 for EE&CO
Orbit Analyst	1	0.5	0.1	8/5	
TOTAL	14	4.5	1.7		

The mission operations process developed for LEO-1shown in Figure 42 includes the following four sub-processes:

a) Planning Process – this takes the Mission Operations Plan, and the Flight Rules & Mission Requirements (all developed before launch), with the current status of the spacecraft, the ground network, and any payload customer requests (such as CRATEX contact requests received from 30 Space Wing at VAFB) and with the aid of the current orbit ephemeris and visibility windows, determines a master schedule, and generates timelines for orbits as necessary. From the timeline a command script is automatically generated, and if necessary a set of pointing quaternions. The command script and quaternions are then verified in the Testbed/simulator, and if successful, passed along to the Execution Process for implementation.

a) Execution Process – this handles the real-time operations of the ground network (ground stations and network) and the interactions with the LEO-1 spacecraft. Contact with the spacecraft are monitored, during which the command script and flat files are uplinked, while the SOH (both real-time and archived) and payload data are downlinked to the ground station and transferred to the HMOC. All contacts and events are logged, both automatically and manually by the controllers.

b) Data Management Process – this takes the downlinked SOH and payload data from the ground station, then archives, processes, and distributes the data.

c) Analysis Process – the processed data are analyzed and trended over time to determine the health of the spacecraft and the success in accomplishing mission objectives. If anomalies are detected, then a spacecraft or payload engineer is alerted to initiate an anomaly resolution process.

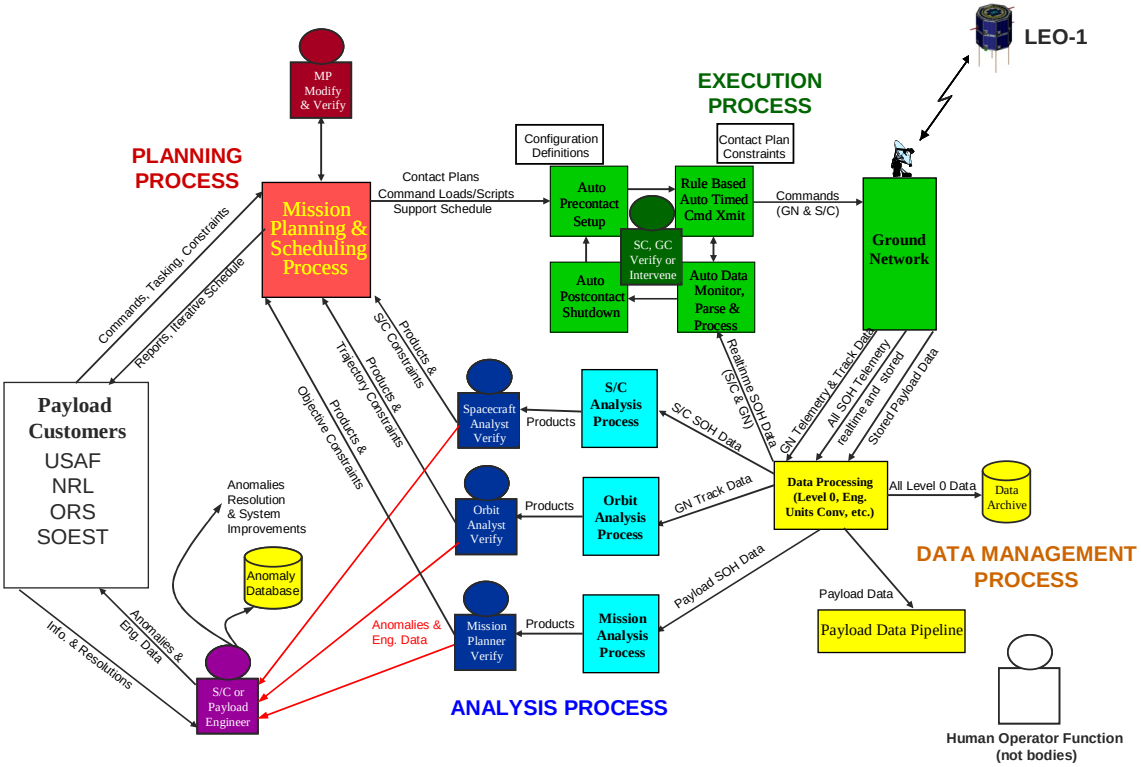


Figure 42. Mission Operations Process

VI. Conclusion

HSFL is well on the path to design, build, and launch a microsatellite. In May 2009 the LEO-1 PDR was successfully completed, with the government customer giving it an "A" grade. The paradigm being used of having a core team of full-time professional engineers and supplementing this with a cadre of professors and students (both graduate and undergraduate) to develop a technologically advanced satellite with a fairly complex mission at moderate cost and risk seems to be working. HSFL plans this satellite to be the first on the path to many and even more ambitious missions.

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