# A GEOMAGNETIC SMALLSAT OBSERVATORY FOR OPERATION IN A 200 KM ALTITUDE LOW EARTH ORBIT

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Abstract

*An altitude of 200 km can improve the accuracy of measurements of the earth's crustal fields and low altitude ionospheric currents. However, atmospheric drag at this altitude requires altitude maintenance to provide useful orbital lifetime. In this study, a small-satellite conceptual design is developed. to meet those requirements. The box shaped satellite basic configuration consists of science and electronics module mounted above a propulsion module. Overall satellite body size is 0.71 m x 0.71 m x 1.58 m long. The science and electronics module accommodates an 8 meter long deployable, non-magnetic boom carrying a scalar magnetometer, a star imager and a compact spherical coil magnetometer. Also mounted in this module are the modular electronic boxes, the $NiH_2$ battery, a charged particle detector, and a position receiver utilizing both the GPS and the GLONASS systems.*

*Major design drivers are the need to minimize the 'ram' area to reduce the amount of propellant for altitude maintenance, coupled with the requirement to accommodate the satellite within the volume and mass constraints of the Pegasus XL launch vehicle. With the chosen configuration the 'ram' area is reduced to 0.5 $m^2$, using one flight proven propellant tank. The total wet mass of the satellite is 264 kg including margin.*

## Introduction

Forming a part of the ESA Explorer program and at the request of the magnetometry mission working group (MMWG), ESA is examining the role of small satellites to study the earths magnetic field and to provide a coordinated follow-on to the current science/geomagnetic satellite projects (Ørsted, SAC-C, CHAMP, etc.). As part of this activity, CRI A/S has performed studies covering a high-altitude Geomagnetic Space Observatory (GSO-H) in a 600 km polar orbit and a separate low-altitude GSO-L operating in a 200 km polar orbit. This paper focuses on the GSO-L satellite design.

All subsystems of the satellite design are described. Special attention is paid to the altitude and orbit control subsystem (AOCS) which uses clusters of combined thrusters to achieve dual functionality and to maintain 3-axis stability with the boom trailing behind the satellite in the negative velocity direction.

## Science Overview

Three district sources are considered to generate the Earth's magnetic field. They are: - convection in the Earth's fluid core generating the main field; remanent and induced magnetization in the Earth's crust and mantle generating the anomaly or lithospheric field and currents in the ionosphere and magnetosphere generating the external field.

As part of the ESA Earth Explorer program, a Magnetometry Working Group determined the scientific objectives and observation requirements of a magnetometry mission [8] to further define the Earth's magnetic field. Conflicting requirements between lithospheric field studies needing short duration, low altitude measurements and main field studies needing long duration measurements, led to the consideration of two satellites called Geomagnetic Space Observatories (GSO). These observatories are: -

- GSO-H For measuring main and external magnetic fields in a drifting high inclination, polar orbit at 600 km altitude, 5 years mission duration and proposed launch 2001.
- GSO-L For measuring lithospheric and external fields in a high inclination, polar orbit, altitude 200-250 km, 6 months observation time and proposed launch 2003.

The dual satellite concept proposed above will accomplish the major mission objectives summarized below: -

- The accurate separation of the Earth's main, lithospheric and external magnetic fields on a global scale.
- The determination of the Earth's main magnetic field and derive an up-to-date main

field model together with its secular variation, mantle conductivity and core-mantle coupling.

- Map the long wavelength (>250 km) lithospheric field thus bridging the gap between satellite and aeromagnetic mapping and identifying magnetic sources in the oceanic lithosphere.

- Determination of the Earth's ionospheric magnetic fields to improve understanding of the global coupled sun-Earth system, ionosphere structure and processes and provide an empirical model of the global ionospheric current distribution.

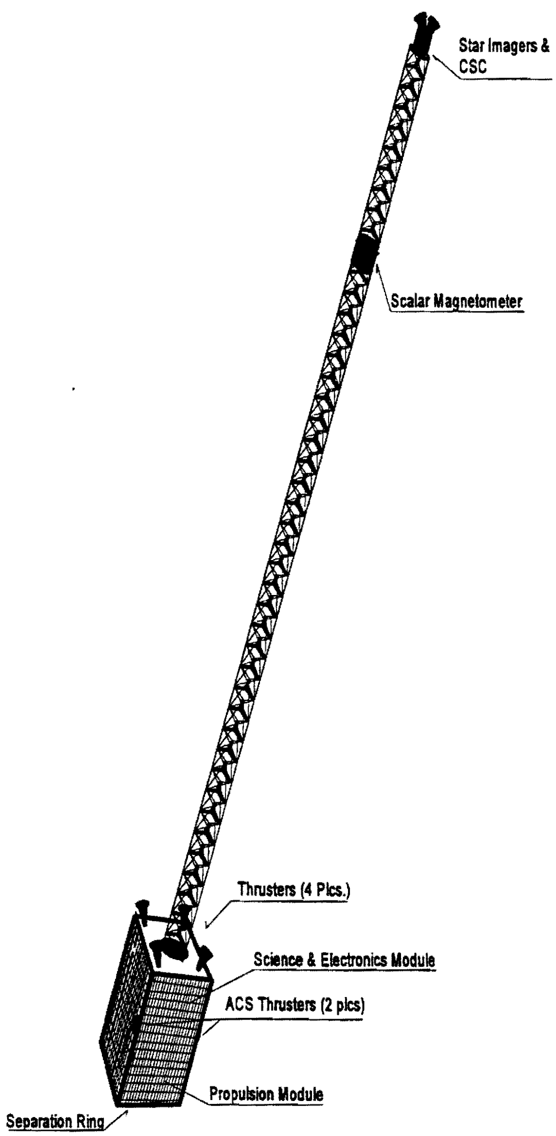


**Fig. 1: GSO-L Satellite in the Orbital Configuration**

## Satellite

### Overview

The satellite is to be placed in a polar, high inclination, low earth (LEO) with resultant nodal drift to provide local time coverage and thereby achieve the highest possible value of the science data. To reduce the risks following separation from the launch vehicle, the satellite is to be initially placed in a 280 km altitude LEO. Following separation from the launch vehicle, autonomous 3-axis attitude stabilization is achieved, and an 8 m long boom is deployed in the negative velocity vector direction to bring the satellite into the orbital operating configuration shown in Fig. 1. During the commissioning phase, the satellite altitude is allowed to decay further to the operational altitude of 200 +/- 3 km.

Two magnetometers and two star imagers are mounted on the boom. The vector (CSC) magnetometer and the two star imagers are mounted and aligned together on an optical bench within the boom envelope to eliminate the effect of boom flexing. A separate scalar magnetometer provides absolute calibration of the CSC magnetometer. The long boom reduces the effect of the satellite magnetic field on the magnetometers to an acceptable level.

The approach used in the conceptual design of the GSO-L satellite is to follow the strategy of a simple, modular satellite body with a negative velocity vector pointing deployable boom. Such an approach provides a compact, lightweight design which is essential for meeting the major payload requirements of minimum magnetic disturbance from the satellite and the ability to maintain an altitude of 200 km for 6 months. At the same time it utilizes and improves upon the considerable experience gained during the Ørsted satellite project [1], and maximizes cost efficient utilization of commonality between GSO-H and GSO-L.

The overall satellite is configured as a box-shaped body comprising a science/electronics module and a propulsion module. Fig. 2- shows the preliminary internal arrangement of the satellite. Such an arrangement provides a compact, minimum weight design to reduce magnetic disturbances to an absolute minimum, while providing ease of manufacture, assembly and integration.

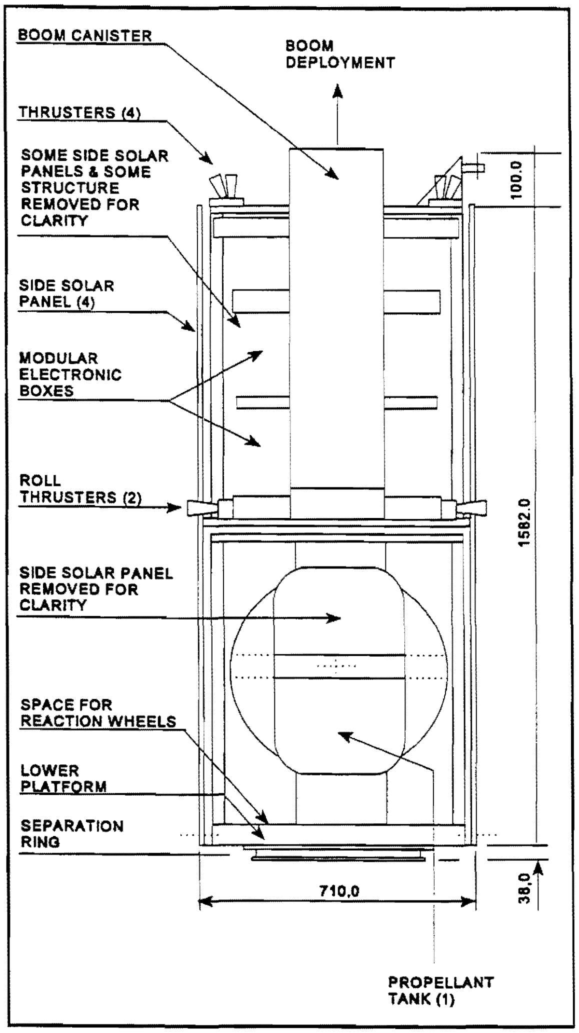


***Fig. 2: GSO-L Satellite Internal Layout***

Atmospheric drag considerations at 200 km altitude dictate a minimal satellite body area in the "ram" or velocity vector direction, to minimize the amount of propellant required for altitude maintenance. Further, the maximum distance between the center of mass and the center of pressure shall ensure a stable attitude. The body area must, however, be consistent with the required solar panel size, necessary to provide adequate electrical power to the satellite and the space available within the launch vehicle fairing. In addition, the satellite launch mass must be within the capability of the launch vehicle to reach the required altitude. The GSO-L satellite as currently conceived meets these major constraints.

The center of mass lies close to boom end of the satellite body while the center of pressure lies a little further up the boom (when the boom is deployed). During the mission when the propulsion tanks are emptied the two points move closer to each other thereby increasing the possibility of instability in the attitude control. Initial calculations based on the current mass budget and the current mass allocation on units show margin in the necessary distance between the two points.

Electrically the GSO-L satellite science/electronics module is configured as four major functional areas, an electrical power subsystem, a command and data handling subsystem, an attitude and orbit control subsystem and a communication subsystem. Dedicated harnesses for power distribution, signals and RF interconnect the satellite subsystems and payload units. The major characteristics of the satellite are presented in Table 1.

The satellite electrical power demand is met by a body mounted solar array and a battery. This power is conditioned and distributed by the power subsystem via a voltage regulated bus and current limiting switches.

The command and data handling subsystem performs the primary control functions of the satellite and consists of two central processing units. This subsystem distributes commands, received via the combined uplink unit and communication subsystem, and acquires and encodes telemetry from the payload and satellite subsystems. This telemetry is delivered either to the downlink unit and from there to the communication subsystem for real time transmission to the ground, or to the on-board memory for later transmission.

Communication with the satellite is established through the communication subsystem. Internally the communication subsystem interfaces with the command and data handling subsystem through the uplink and downlink units. The communication link comprises an uplink capability to receive telecommands, and a downlink capability to transmit telemetry.

The propulsion module houses the single mono-propellant hydrazine tank, the necessary valves and pipework and the AOCS reaction wheels. Location of the reaction wheels was chosen to be on the propulsion module lower platform, to assist in lowering the satellite center of mass. AOCS thrusters are located on the boom end of the satellite body, with two roll thrusters located approximately at the junction of the science/electronics and propulsion modules. The separation ring, which forms part of the Pegasus 23 inch payload separation system is attached to the underside of the propulsion module lower platform.

An overview of the satellite subsystems (and the redundancy) is shown in the block diagram of Fig. 3.

| CHARACTERISTIC | | Description |
|---|---|---|
| Configuration | | Box shaped body |
| Body Size | - H<br>- W<br>- D | 1582 mm<br>710 mm<br>710 mm |
| Solar panels | - Quantity<br>- Solar cell type<br>- Output EOM | Five<br>GaInP/GaAs<br>240.5 W * |
| Battery | - Quantity<br>- Type | One<br>10 Ah $NiH_2$ |
| Power Control Units | - Quantity<br>- Battery Charge Regulators<br>- Power Distribution | Two<br>Buck topology<br>FET Switches |
| Primary Structure | - Science and electronics Module<br>- Propulsion module | 'H' beam & platforms<br>Load and shear panels,<br>lower platform |
| AOCS | - Type<br>- Attitude control<br>- Orbit and altitude control<br>- Propellant | Active<br>Reaction wheels and thrusters.<br>Thrusters in negative velocity vector<br>Monoprop. Hydrazine |
| ACS Sensors and electronics | - 1-axis sun sensors<br>- 3-axis sun sensors<br>- turnrate sensors | Two<br>Two<br>Four |
| Communication | - Quantity<br>- Frequency | Two transceivers<br>'S' Band |
| Antennas | - Quantity<br>- Type | Two<br>turnstile |
| Computers | - Quantity<br>- CPU | Two<br>RISC 6000 (or Intel 80386 or Intel 80486) |
| Up-link Unit/Down-link Unit | - Quantity<br>- Storage included in DLU | Two<br>56 Mbyte |
| Position Determination | - Quantity<br>- Type | One<br>GPS/GLONASS |
| Boom | - Length<br>- Type | 1 x 8 m<br>Deployable, 3 longerons |
| Thermal Control | | Passive and Active |
| Mass (excluding suppuration mechanism) | | Approx. 264 kg at launch |

* Excluding boom shadows on top panel.

**Table 1: GSO-L Satellite Major Characteristics**

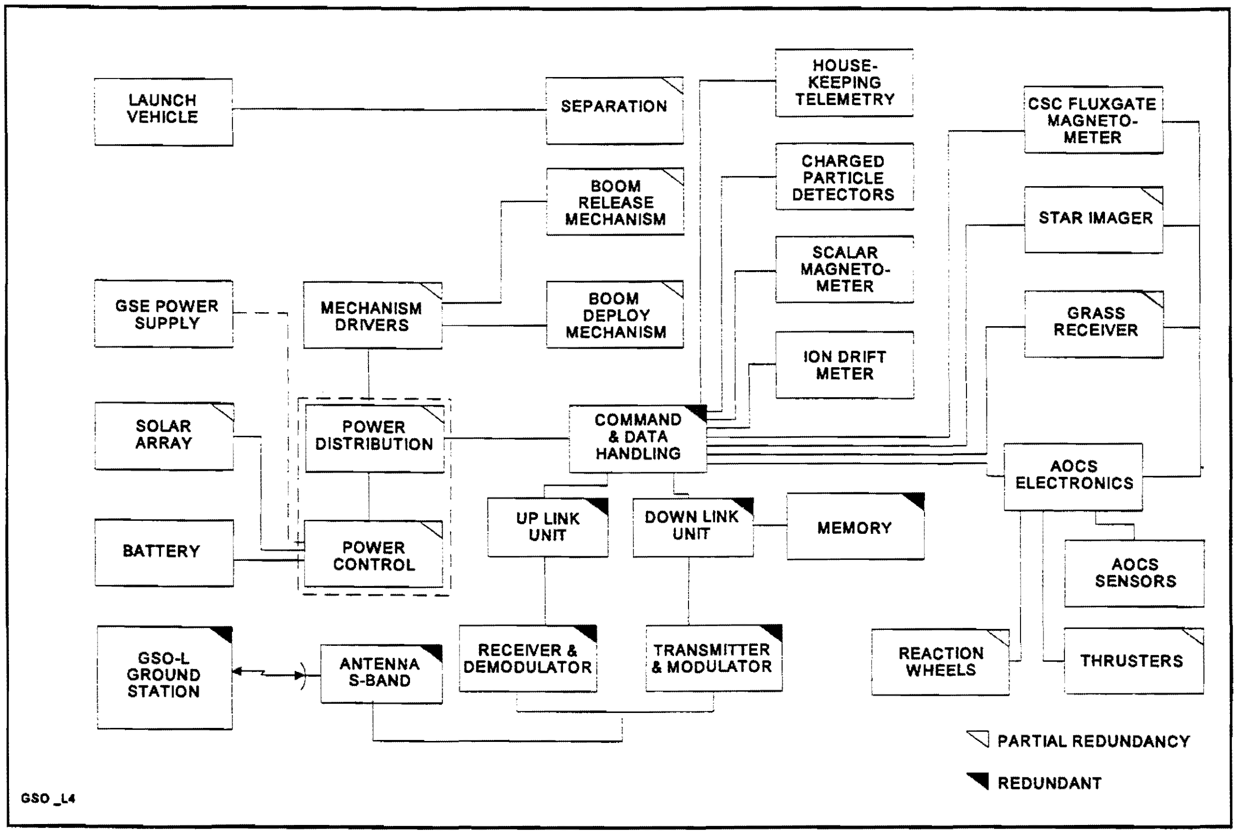


**Fig. 3: GSO-L Satellite Functional Block Diagram**

Science Payload

The complement of science instruments consists of two magnetometers (scalar and vector) to map the geomagnetic field, a dual camera head star imager and six high energy charged particle detectors to measure the charged particle environment. Further an Ion Drift Meter has been added to the primary payload based on recommendations made in the Earth Explorer Magnetometry Report. The instruments use state-of-the-art technology. The following discussions provide an overview of the individual instruments.

**Magnetometers** . . .Two magnetometers are used to measure the Earth's magnetic field:

- Scalar Proton Free Precession Magnetometer
- Triaxial CSC Fluxgate Magnetometer

The CSC magnetometer measures the relative magnetic field vector components in three mutually perpendicular axes. The Proton magnetometer measures the absolute magnetic field amplitude.

**Scalar Proton Free Precession Magnetometer. . .**The objective of this instrument is to measure the absolute scalar magnitude of the Earth's magnetic field along the satellite orbit. The measurement range is 15000 nT to 60000 nT. The instrument provides one scalar field sample per 1-2 minutes. To achieve the design goal of an absolute accuracy better than 0.5 nT the sensor is mounted approximately 6 m away from the satellite body on a boom which is about 8 m in total length. The magnitude measurement is not influenced by boom deflection.

**CSC Magnetometer** . . . The CSC triaxial vector feedback fluxgate magnetometer measures the Earth's magnetic field vectors. The measuring range is +/-65000 nT with 20 bits resolution corresponding to 0.1 nT.

**Star Imager** . . . The Star Imager (SIM) supplies the absolute attitude referred to the celestial coordinates. This is achieved by viewing and recognizing stellar constellations, and by fine-tune matching the data to an on-board high quality star catalogue in real time. The objective of the SIM is to determine the attitude of the CSC magnetometer within 2 arc-seconds. The SIM is mechanically coupled to the CSC fluxgate magnetometer by a non-magnetic optical bench. The angular alignment between the SIM and the CSC axes are determined during the calibration.

**Optical Bench** . . . The instrument placing the most severe constraint on attitude determination is the CSC vector magnetometer. In order to achieve a calibrated error angle between the CSC and the star imager (SIM) camera heads of less than 1 arc second, the distance between the star imager and the CSC is minimized, and the two units are mounted on a non-metallic, non-magnetic, thermally stable optical bench.

**Charged Particle Detectors** . . . The solid state charged particle detector (CPD) experiment detects electrons in the energy range from 20 keV to 2 MeV, protons from 150 keV to 30 MeV and alpha particles from 600 keV to 124 MeV. The CPD recommended for GSO-L consists of several solid state detectors for energetic electrons, protons, and alpha particles. The detector array scans solid angles around 0° and 90° (relative to the boom direction) and bin up to 8 energy levels for each detector.

**Ion Drift Meter** . . . The Ion Drift Meter (IDM) measures the vector drift velocity of ambient ions. The detector is mounted on the ram surface and measures the angle of incoming ion flux. This angle is given by a combination of the spacecraft velocity in the detector coordinate system and the ion drift velocity. The IDM includes a retarding potential analyzer (RPA), which measures the velocity component parallel to the IDM viewing direction and a segmented anode, which determines the angle of the incoming flux.

Structure

As described earlier, the overall satellite is configured as a box-shaped body comprising a science and electronics module mounted over a propulsion module as shown in Fig. 2.

Primary design drivers leading to the selection of this structural configuration are:

- Size limitations of the Pegasus XL fairing.
- Provide a readily fabricated structure and components.
- Maximize the diameter of the chosen boom configuration and thus maximize the deployment torque
- Accommodate the Scalar and CSC Magnetometers and the Star Imager within the boom envelope.
- Provide modular electronic boxes for ease of integration and test.
- Maximize the stiffness of the structure.
- Keep the Center of Mass as low as possible and as close as possible to the center of the separation system.
- Reduce the satellite body "ram" area to minimize propellant required to maintain altitude for the required lifetime.
- Accommodate required propellant tank containing fuel for altitude and attitude control (unloading of reaction wheels).
- Provide sufficient solar array area and heat radiation area.

In this brief description of the satellite structure design concept it is therefore convenient to consider the structure as divided into two items - one for each of the above modules.

**Science and Electronics Module** . . . The structure for the science and electronics module draws upon the basic approach used for the Ørsted satellite design. The primary structure comprises a fabricated vertical "H" beam, complete with four 45 deg angled stiffening panels, all mounted on the module lower platform. In this way five basic compartments are created within this structure. The center compartment houses the deployable boom assembly. Two rows of modular electronic boxes are mounted on each side of the boom center compartment, to accommodate the majority of the satellite bus and science payload electronics. Each row of electronic boxes is separated by a central cable tray, with additional cable trays provided above and below each row of boxes. The remaining two compartments accommodate the battery and the reaction wheel electronics. The reaction wheels are placed in the lower propulsion module. Shear panels enclose and also stiffen the primary structure.

The structure also supports four side- and one top solar panel substrates, to which are mounted the solar cells, providing electric power to the satellite. A sketch of the general arrangement of the science and electronics module showing the primary structure is given in Fig. 4. The charged particle detector is mounted on a small upper platform attached to the top of the primary structure.

In addition to stiffening the "H" beam, the four 45 deg panels allow load transfer from the corresponding lower propulsion module stiffener panels. They also improve radiative heat transfer from the "H" beam to the solar panels and improve conductive heat transfer down to the lower propulsion module.

**Propulsion Module** . . . The primary structure considered for the propulsion module is mounted on the standard 23 inch Pegasus separation mechanism and consists of a square lower platform with vertical load bearing columns located at each corner, to which are attached vertical stiffening panels. This configuration is shown in Fig. 5. In addition to stiffening the columns, the panels also allow mounting of the propellant tank in the existing flight-proven manner, together with the necessary valves, hydrazine lines etc. It should be noted that the stiffener panel locations correspond to similar panels located in the science and electronics module, allowing load- and heat transfer between the modules.

Sufficient space is allowed between the propellant tank and the lower platform, to accommodate the four AOCS reaction wheels. These reaction wheels are mounted on local aluminum honeycomb panels oriented to meet the required mutual wheel angles. The wheels are mounted in this module to maximize the distance to the magnetometers and for lowering the COG.

Shear panels enclose and stiffen the primary structure. The structure also supports the four side solar panels, which are the full height of the satellite body.

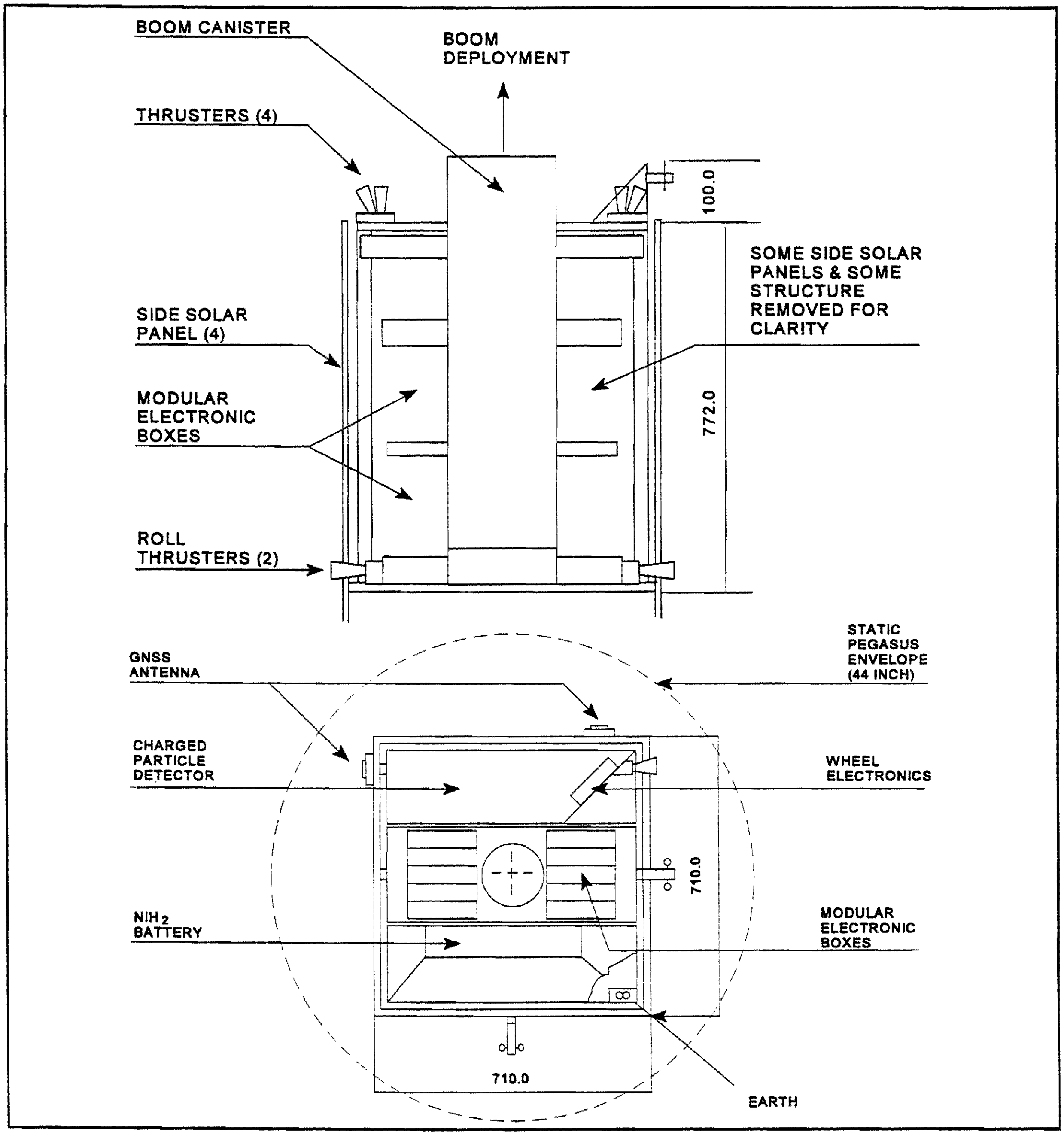


*Fig. 4: Science and Electronics Module*

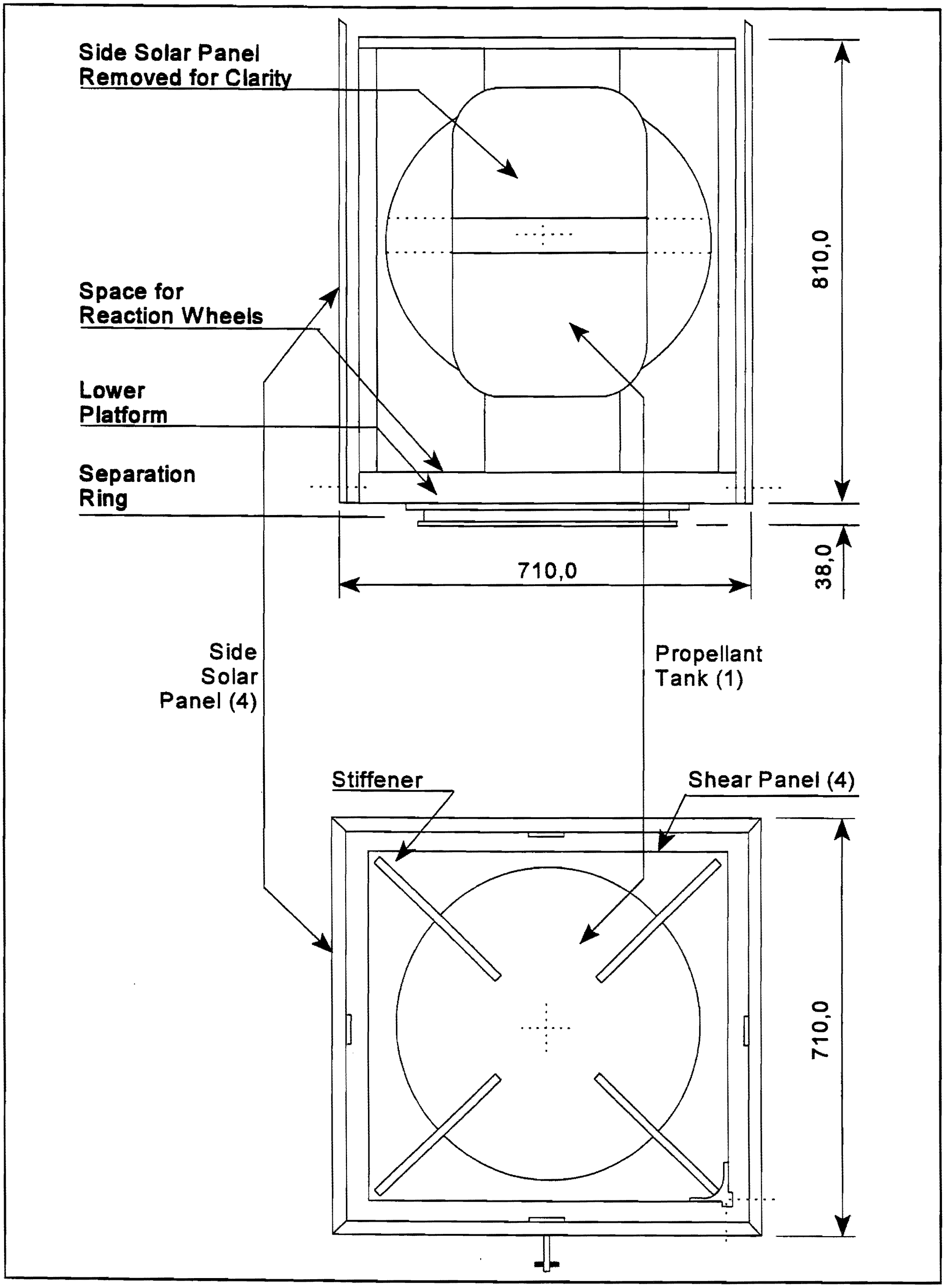


*Fig. 5: Propulsion Module*

Mechanisms

**Deployable Boom** . . . One of the compartments formed by the science and electronics module primary structure houses the deployable boom assembly consisting of the boom, the two magnetometers, the two star imagers and the boom release and deployment mechanisms.

The selected boom design for GSO-L is the tested Ørsted design due to the limited weight, the non-metallic parts and materials, the reasonably easy way of separating and mounting the two instrument packages, and the limited sizes and mass of the deployment mechanisms. A diagram of part of this boom configuration is shown in Fig. 6. The deployable boom consists of three coilable longerons. The longerons are separated by radial spacers and tensioned by cross-wires when deployed. This boom configuration provides sufficient inherent deployment torque without the need for external springs or other mechanical devices except for a restraining wire in the center of the boom to control the rate of the deployment.

**Boom Release Mechanisms** . . . Prior to boom deployment, the boom complete with magnetometers and star imager is released from the stowed launch condition by redundant non-explosive initiators and latches.

**Boom Deployment Mechanism. . .**The deployment of the boom is controlled by redundant motors, a differential gear drive and optical sensors.

**Launch Vehicle Separation Mechanism. . .**The underside of the satellite structure is fitted with a separation ring which forms part of the separation mechanism provided and controlled by the Pegasus XL launch vehicle.

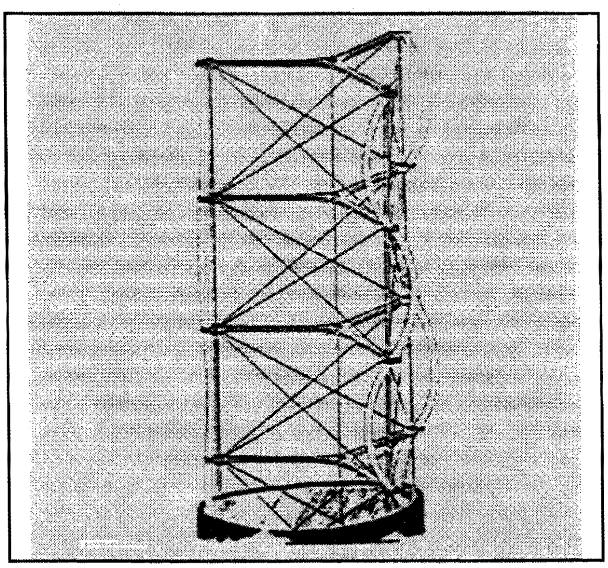

*Fig. 6: Section of Deployed Boom*

Command and Data Handling

Two identical redundant Command and Data Handling (CDH) units are provided. Only one CDH unit is activated at a time and provision is also made to deactivate the CDH units by ground command. All data handling electronics are connected to the MIL-STD-1553B on-board data-bus as shown in Fig. 7. If one CDH units fails the other CDH unit can take over the bus-controller function and still have access to all subsystems and instruments

Telecommands are uplinked from the ground station, decoded by an UpLink Unit (ULU), routed to the destinations, and executed. Further, time tagged telecommands are stored in a Master Scheduler for time controlled retrieval and execution.

Telemetry is routed to the On-Board Computer (OBC), forwarded to a DownLink Unit (DLU), encoded, stored, retrieved during ground passes, and downlinked. Further, during ground passes real time telemetry is downlinked directly from the DLU without being stored first.

Time information is provided by a GPS/GLONASS GRASS receiver and is routed to all instruments and subsystems that perform telemetry time stamping via a dedicated time-bus. The data rates from the instruments and satellite subsystems is given in Table 2. The data is Reed-Solomon encoded by the DLU before it is stored. In this way radiation caused bit-flips can be 'washed-out' when data are received by the ground station and decoded.

**Mass Storage** . . . To simplify the operational interface of the instruments, mass storage is accessible from the DLU only. Mass storage provides the storage capacity needed for a store-and-dump strategy facilitating data taking and storage at any point in orbit. The total mass memory capacity needed to store the data during non-contact periods and assuming a downlink rate of 1 Mbit/second is 4.6 x $10^7$ byte. This capacity is provided by a 56 Mbyte cold redundant memory with a margin of 28%.

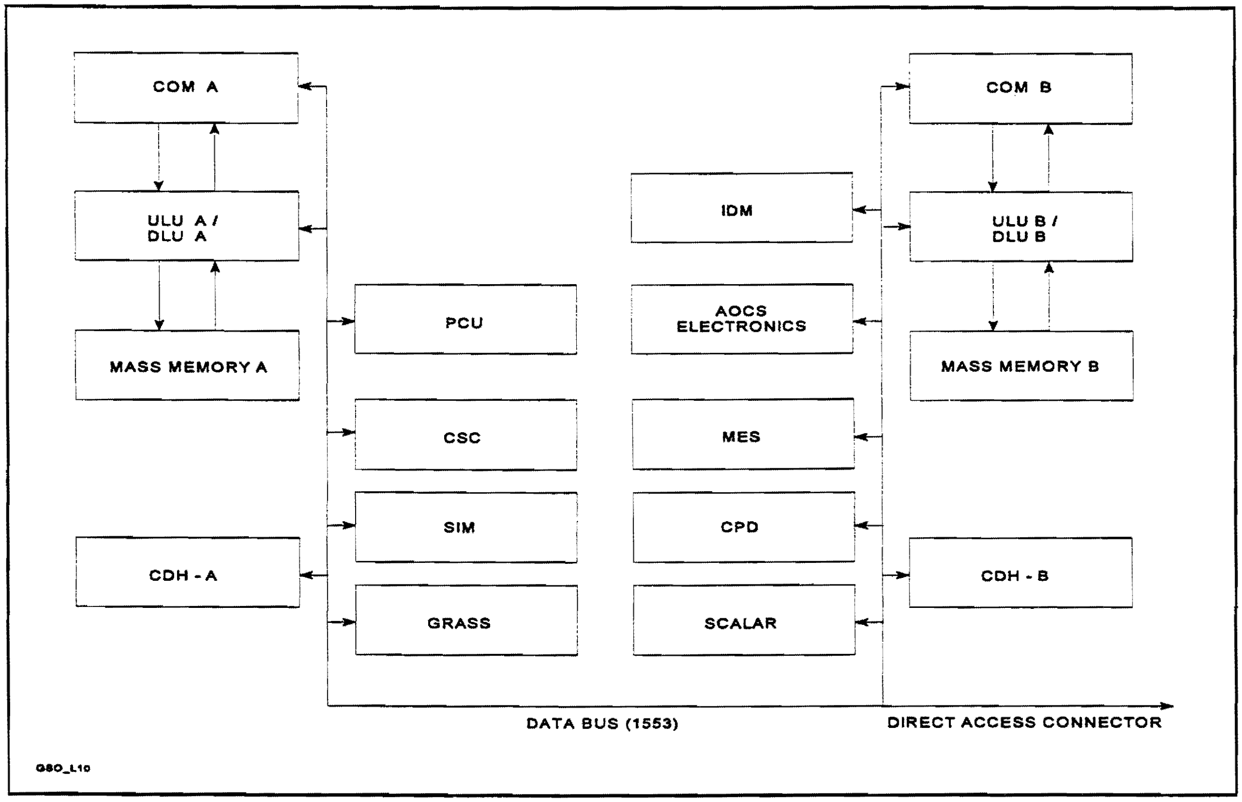


**Fig. 7: GSO-L On-Board Data Bus Structure**

| Source | Normal Mode (bits/second) | Burst Mode (bits/second) |
|---|---|---|
| CSC Magnetometer | (200) | **2000** |
| Scalar Magnetometer | **30** | n/a |
| Star Imager | **400** | n/a |
| CPD | **200** (average) | 2600 |
| IDM | **700** | n/a |
| GRAS Receiver | (1000) | **5000** |
| Satellite Housekeeping | **120** | n/a |

**Table 2: Satellite Data Rates**

Electric Power Subsystem

The Electric Power Subsystem (EPS) consists of a solar array, a battery, two cold redundant Battery Charge Regulators (BCR), two central DC/DC convertors and two Power Distribution Units (PDU) which control the loads on the subsystem. Minimum solar array output at end-of-mission (EOM) allowing for solar flare protons in [2] is 240 watts averaged over the sunlit portion of the noon-midnight orbit at summer solstice. Average power consumption by the satellite in the nominal science data gathering mode is 213 watts thus providing a margin of 13% worst case at EOM.

All voltage regulation electronics for the EPS are housed in two identical electronic boxes called Power Control Units (PCU). Each PCU contains one battery charge regulator (BCR) and one central DC/DC converter. A separate power distribution unit (PDU) controls the switchable loads and also the circuitry to control the activation of the power switches. Telemetry of selected voltage current, temperature and status are provided to the Command and Data Handling Subsystem for storage and later downlink.

**Solar Array** . . . The GSO-L Solar array provides power to the satellite and charges the battery during the

sunlit part of each orbit. It comprises five solar panels - four mounted on the sides of the satellite body and one boom end solar panel as shown in Fig. 1.
There are 5024 solar cells forming the solar array arranged in single circuits of 18 series cells on the side panels and 17 series cells on the top panel. The solar cells are nominally 20 x 40 mm, $GaInP_2$/GaAs and each solar cell is protected by a ceria-doped microsheet cover glass 150 μm thick. Each solar cell circuit is isolated from the main power output bus by a blocking diode mounted on the rear of each solar panel. Average solar array output during the sunlit portion of the orbit as a function of mission lifetime is shown in Fig. 8.

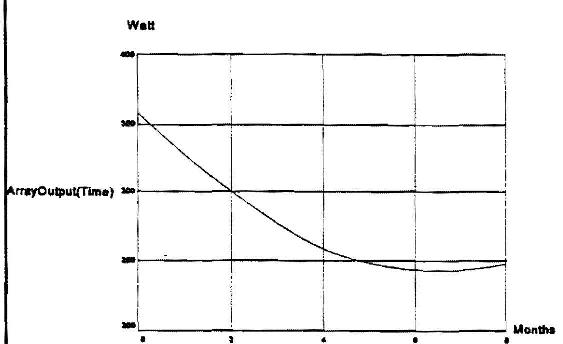


**Fig. 8: Solar Array Output vs Mission Time**

**Battery** . . . A single $NiH_2$ battery provides power to the satellite during eclipse periods and also if pulse loads exceed the output capacity of the solar array. During the sunlit portions of the orbit the battery is charged by the solar array.

During many years of operation $NiH_2$ batteries have established a reputation for reliable performance in satellite orbital operation. The current trend towards small, lower mass satellites in LEO with consequent reduced power requirements, provides the opportunity to apply $NiH_2$ battery technology to this emerging small-satellite market with consequent savings in mass coupled with high energy density, long cycle life, high reliability and simplified charge control. This technology is now available for small satellite applications by the use of flight qualified, small-diameter, combined pressure vessel (CPV) $NiH_2$ battery cell design. The battery assembly considered for the GSO-L mission consists of ten 10Ah $NiH_2$ CPV aerospace grade battery cells connected in series for a nominal output voltage of 25 V DC.

**Battery Charge Regulator. . .**The BCR is based upon a Buck topology with the duty cycle controlled by both the solar array voltage control and the state-of-charge current control. The state-of-charge current control has priority.

The BCR is capable of charging a fully discharged battery with the input from the solar panels controlled by a latching relay to provide a cold-redundancy function with the second BCR. Interchange of BCR's can also be made by ground command. Output from the BCR is fed to the main unregulated power bus without protection fuses as the BCR circuit itself is short circuit point failure free.

**DC/DC Converter. . .**The DC/DC converter is implemented as a current-mode-push-pull converter. It converts the unregulated bus into $\pm$5V and $\pm$8V regulated output lines. Input to each converter is through redundant fuses to protect the main bus and the battery against a short-circuit in the converter or the PDU connected to the output. Each converter is started by a continuous turn-on signal from the separation switch.

**Power Distribution Unit. . . .** Each PDU controls and distributes various voltages to the switchable loads, which are electrically divided into two groups..

All switches used in the distribution of loads are solid state Field Effect Transistor (FET) type, except the boom release and deployment circuits which have an additional mechanical relay in series with the FET switches to provide complete isolation. In this case, the FET switches and the relay require two separate telecommands (TC) for operation. All FET switches are controlled in a "soft" turn-on mode to prevent noise-spikes.

Attitude and Orbit Control Subsystem

Design of the Attitude and Orbit Control Subsystem (AOCS) is based on the requirements and the findings in the study [7] on aerodynamic simulations for the 250 and 200 km altitudes.

Normal operation - the science observation phase - was found to be most feasibly done in a "comet" mode of the satellite, i.e., with the boom in the wake. The motion in comet mode was found to exhibit a limit cycle between two marginally stable equilibria. If large motions occur, a random and undamped motion can be initiated. With the magnitude of forces involved, it was decided to use reaction wheels as the primary actuator for attitude control. A set of four wheels was chosen - rather than the minimum set of three. The four wheels are mounted in a canted arrangement such that a failure in any wheel can be compensated by coordinated control of the others. Wheel de-spin is done by use of the orbit control thrusters for the pitch and yaw motions, a smaller dedicated thruster is employed for roll axis de-spin. De-tumbling control, attitude control before and during boom deployment, and possible maneuvering during altitude decay from the initial orbit down to the science observation altitude are well supported by using reaction wheels as actuators.

Instrumentation of the AOCS is partly the science observation instruments: GRASS, SIM and CSC. Additional instruments are needed, however, as SIM is only available after boom deployment. Before this, CSC and a set of wide angle sun sensors are used to deduct attitude information. The CSC is not available during boom deployment due to undefined orientation, and a set of rate sensors was incorporated to obtain sufficient information for attitude and rate estimation during boom deployment. Rate sensor measurements are also used to obtain a faster converging Kalman filter than obtainable from sun sensor and CSC measurements alone. Orbit control is limited to compensation of average drag force to maintain the desired altitude. Firing of hydrazine thrusters for propulsion requires tight attitude control where the attitude set point is calculated from orbit data before each burst. The orbit control will also take due account of necessary wheel de-spin in pitch and yaw. The AOCS configuration is shown in Fig. 9.

Faults in the AOCS will be detected by dedicated detectors, and the state of the satellite and AOCS will be observed by an AOCS manager module (S/W). Fault accommodation actions will be taken by the AOCS manager module through dedicated efforts. An effector may, as an example, shift to an alternative wheel control strategy if one wheel is detected to have anomalous operation.

**Orbit Versus Attitude Control** . . .There are two main causes for interaction between orbit and attitude control. One is the use of the set of orbit control thrusters for pitch and yaw axes wheel de-spin. Another is inaccuracy in the balance of thrust delivered by the main thrusters, together with some uncertainty in the open/close timing accuracy of the thrusters.

The use of the main thrusters for de-spin was motivated by the fact that the capacity of the main thrusters (1 N, and minimum opening time 10-15 m sec - equivalent to a minimum impulse of 0.01 Ns) is not much more than the available minimum size thrusters for attitude control. With the selected wheels (maximum torque 0.03 Nm and momentum 4 Nms), we obtain a reasonable opening time of the main thruster for wheel de-spin.

The motion of the satellite during de-spin is simply determined by available thruster force, arm of this force, which depends on CG location, maximum wheel torque and the inertias. The ratio between the maximum momentum of the wheel and the wheel torque determines the time the attitude control can stay active, i.e., the time before a de-spin is required. The hard limit on de-spin control is found to be the roll axis due to the low inertia and the requirement of a maximum rate of 0.5 deg/s from the SIM.

With the suggested configuration, roll axis de-spin must be carried out by pulsing the roll thruster in a 6% duty cycle. The roll thruster is activated until the roll rate is 0.5 deg/s ( approx 1 second) the wheel is then simultaneously commanded to de-spin using maximum torque. The roll thruster is re-activated when the roll turn rate reaches the other 0.5 deg/s limit. 20 cycles are required for full de-spin of the roll axis wheels.

Pitch and yaw de-spin must be conducted as part of the orbit control activity. A 20 - 60 second period is required for full de-spin depending on available thrust and torque.

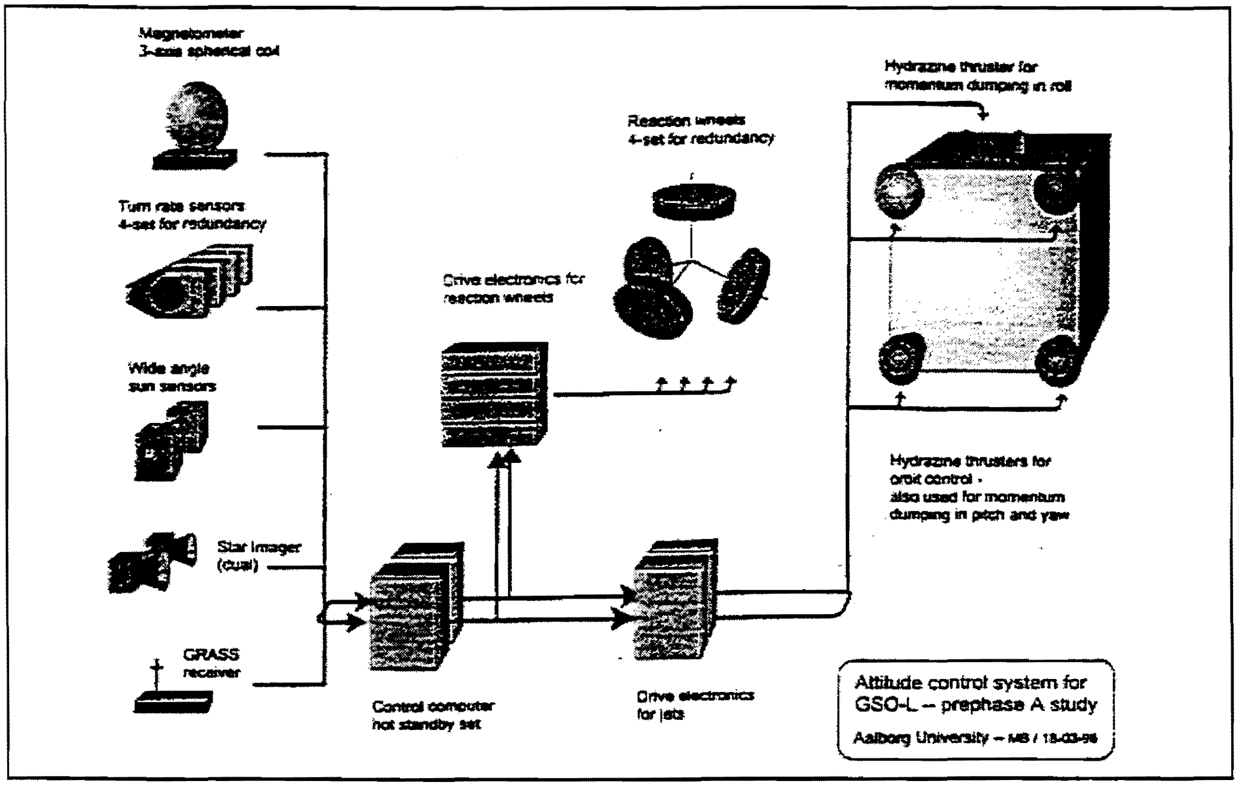


**Fig. 9: Attitude and Orbit Control System (AOCS) Configuration Proposed for GSO-L (By Courtesy of AUC)**

**Reliability and Fail-Safe Operation** . . . With the satellite in a very low orbit, an estimated lifetime without attitude maintenance down to one week in a 250 km orbit, and one-two days at 200 km, component failures in the altitude control system become extremely critical. Attitude component failures may cause large attitude fluctuations which degrades the scientific quality of results and reduces life-time due to increased air drag. The worst-case consequence of attitude components failure is uncontrolled tumbling.

An AOCS failure mode and effects analysis was performed to identify major possible failure modes together with failure consequences, criticality, and the consequence if the fault is not accommodated. The results of this analysis were used to identify requirements of the control system architecture.

A detailed evaluation was performed to establish whether electrical propulsion or chemical propulsion was feasible for the GSO-L satellite. Among the electrical propulsion systems examined were Pulsed Plasma Thruster and Ion Thrusters. The analysis showed that the power necessary to maintain the satellite in a 200 km orbit with an electric propulsion system was so large (>>100 W) that the deployable solar panels necessary to generate this amount of electric power, would not only reduce the reliability of the total system, but would also increase the cross-sectional area of the satellite and thereby increase the air drag. To counteract this increased air drag, more electric power would be required and thus larger solar panels and so on. Consideration of the mass favorable electric propulsion systems was therefore terminated.

Based upon the above information a decision was made to proceed with the well-known monopropellant hydrazine $N_2H_4$ propulsion concept.

**Operational Concept of Hydrazine System** . . . One single diaphragm tank contains both propellant and pressurant at the Beginning of Life (BOL). The propellant feed concept is fully blow-down. The pressure (and the thrust level) decrease during the propellant depletion taking place within the operative lifetime. BOL pressure is approx. 22 bar and End of Mission (EOM) pressure is approx. 5.5 bar.

Two fully redundant Reaction Control Thruster (RCT) branches are used, each branch with isolation capability in the event of leakage failure on one RCT. Four sets of thrusters are used pointing in the "anti-velocity" direction to maintain the altitude, two sets to maintain the attitude in the roll direction

The subsystem schematic is shown in Fig 10. Total mass of the propulsion system is 22.7 kg.

The existing PSI tank utilized for the EURECA OTA subsystem is used. The tank is 653 mm long and 587

| | |
|---|---|
| Average blow-down specific impulse for the system (between 22 and 5.5 bar) | 2100 m/s |
| Available propellant mass (excluding non usable residuals) | 83.5 kg |
| Equivalent $\Delta V$ value | 798 m/s |
| Propellant necessary for orbit maintenance | 55.5 kg |
| Propellant necessary for attitude maintenance | 10 kg |
| Margin when propellant tank is full | 27% |

**Table 3: Major Performance Characteristics of the GSO-L Propulsion System**

mm diameter and can accommodate up to 85 kg of propellant in order to provide approximately 24 liters of ullage for the pressurant accommodation in the BOL condition. The thruster for altitude maintenance and pitch and yaw attitude control is the CHTI.O catalytic hydrazine thruster manufactured by DASA. Attitude control in the roll direction is provided by the CHT0.5 thruster.

**Contamination** . . . The potential impingement of thrust exhaust particles on the boom mounted sensors has been briefly investigated. Initial discussions with ESTEC and NASA indicate that the important factor to eliminate any potential problem, will be to keep the temperatures of any sensitive surfaces or optics higher than 100° K. Where temperatures are lower there will be a risk for particles to contaminate the surfaces.

However, thermal analysis indicates that the temperatures of the sensors will be significantly higher at all times. Further the preliminary analysis performed by ESTEC shows that the resulting mass flux will be very limited at the distances of the boom mounted sensor platforms.

**Performance** . . . Table 3 presents a summary of the propulsion subsystem major performance characteristics. used for maintaining altitude can be calculated from established formulae in [3]. For the GSO-L satellite configuration the necessary amount of propellant is 55.5 kg which combined with attitude control requirements and residuals results in a total needed propellant mass of 67 kg. Thus the 85 kg propellant tank provides a 27% margin assuming a passive drop from the initial 280 km orbit to the 200 km science data orbit. The majority of the propellant is used to maintain the satellite in a 200 km altitude orbit for 6 months.

It is important to note that these numbers are based on the 200 km orbit. As the initial operational orbit is somewhat higher (<250 km) the margin will be higher than the 27 % indicated in Table 5. The margin will depend on the time when it is decided to lower the satellite orbit to 200 km.

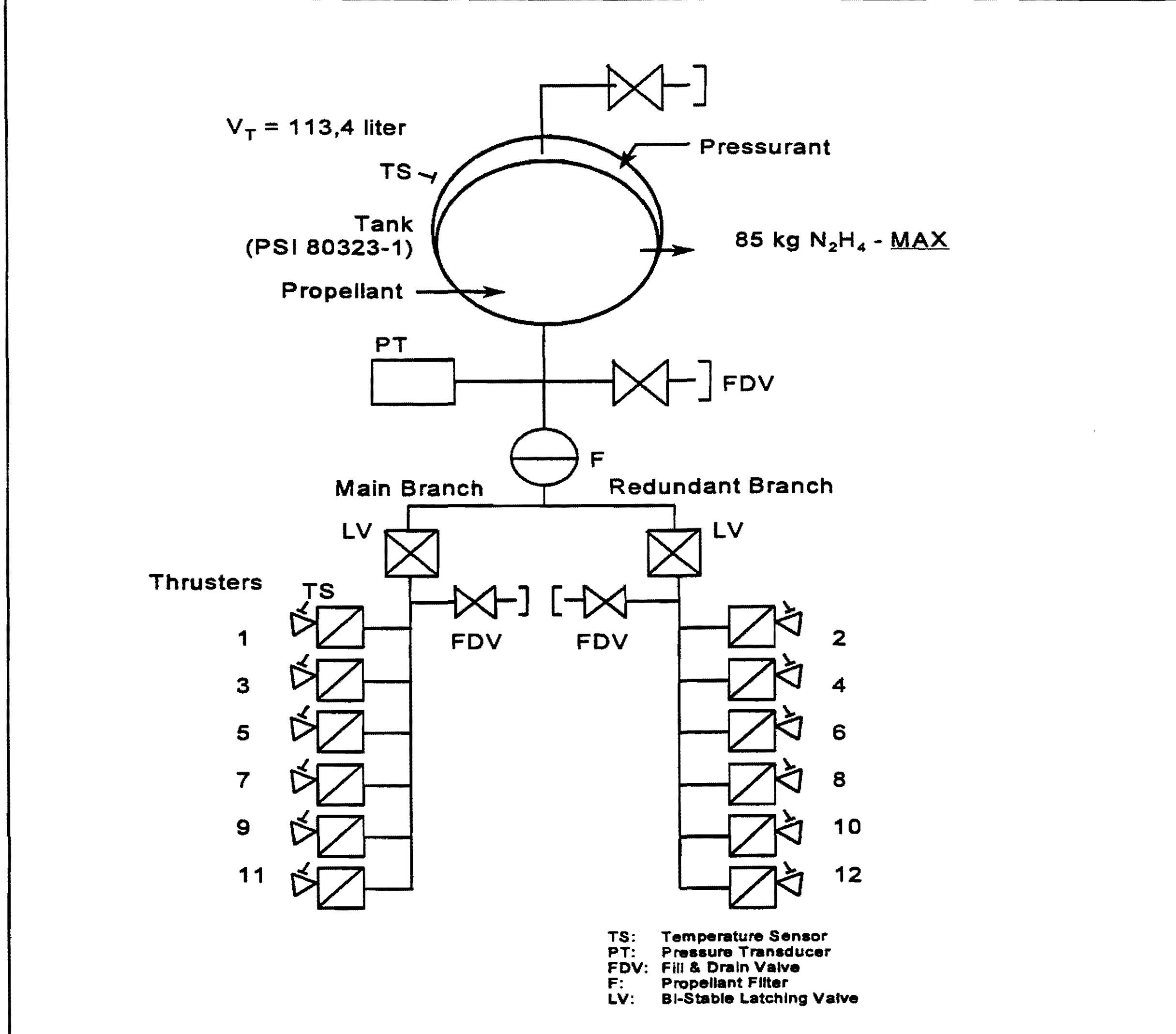


**Fig. 10: Propulsion Subsystem Schematic**

Communication Subsystem

Transfer of telemetry data from the satellite and telecommands to the satellite is accomplished through a highly reliable redundant communication subsystem, when the satellite is in view of the Kiruna ground station. A summary of the communication parameters is given in Table 4.

Only one of the redundant transmitters is powered "on" at a time. The high data rate and high output power mode is only activated during ground station passes to conserve overall power. When the satellite is outside the view of a ground station and prior to the ground station visibility window, the downlink data is transmitted at a low data rate, enabling the output power to be reduced substantially while maintaining positive downlink signal margin. The low data rate combined with low output power functions as the satellite beacon, containing the satellite identification and essential housekeeping data for evaluation of the health of the satellite. While the primary transmitter is active the redundant transmitter is in a cold stand-by mode, ready to take over if any anomaly is detected in the primary transmitter. The active receiver enables uplink of telecommands from the ground station, which is forwarded as a serial synchronous data stream for decoding in the Up Link Unit (ULU), before being forwarded to the on board computer, one of the bus subsystems, or to the discrete command distribution unit which is part of the Up Link Unit. There is no radio transmission during count-down and launch.

A highly reliable and compact antenna [5] is used for each receiver and transmitter pair. The antennas are located on the two Earth facing surfaces of the satellite body and have nearly spherical coverage. The spherical coverage also enables communication with the satellite when an incorrect attitude occurs.

| CHARACTERISTIC | | DATA |
|---|---|---|
| Uplink Channel Data Rate | (Low for satellite tumbling)<br>(High) | 1 kbit/s<br>4 kbit/s |
| Ground Station EIRP | | ≥ +49 dBW |
| Satellite Uplink System Figure of Merit G/T | | > -30 dB/K |
| Target Uplink Bit Error Rate | | ≤ $10^{-6}$ |
| Downlink Channel Data Rate | (Low for signal acquisition)<br>(High for data transmission) | 3.906 kbit/s<br>1 Mbit/s |
| Satellite Transmitter EIRP | (Low)<br>(High) | -15 dBW<br>-2 dBW |
| Target Downlink System Margin at 5° elevation | | 3 dB |
| Target Downlink Probability of Frame loss | | ≤ $10^{-8}$ |
| Ground Station Figure of Merit G/T | | ≥ 17 dB/K |

**Table 4: Communication Parameters**

Due to the high bit rates used for the telecommand and telemetry links 'S' band carrier frequencies are used. Although not a requirement, the up and down frequencies are paired at a fixed 240/221 ratio to enable communication with other ESA ground stations than the selected primary ground station at Kiruna (Sweden). Binary Phase Shift Keying (BPSK) with square baseband pulse shaping directly modulated onto the carrier is selected as the baseline modulation format for both the telecommand uplink and the telemetry downlink. BPSK is very robust to interference and distortions and one of the formats recommended by [6].

Position Determination

GSO-L satellite position and timing in orbit is determined using the GPS/GLONASS systems.

The GRASS receiver now under development in Europe can operate on both the GPS and GLONASS satellite navigation systems. It provides satellite position and timing data via a patch antenna located on the satellite body to obtain maximum viewing of the satellites in the GPS/GLONASS constellations. The GRASS receiver will also operated via a second patch antenna located on the satellite body which provides a view if the Earth limb and the opportunity to acquire some ionospheric and atmospheric sounding data as the position satellites recede from view of the GSO-L satellite and their signals pass through the Earth's atmosphere/ionospheric profile.

The GRASS system provides a position accuracy of <30 m real time. In order to meet the required position accuracy of ±4 m, post processing on ground is likely to be required. Such post processing techniques used for increasing the accuracy are well known.

The GRASS system will also provide the onboard timing reference in UTC. In addition the time pulse can be used as a frequency reference for the Scalar Magnetometer. The Receiver is turned on by ground command.

Thermal Subsystem

A passive thermal control subsystem supplemented by local active heating is used to control the temperature of the GSO-L satellite. This is achieved by thermally coupling all heat dissipating equipment (except boom mounted instruments) to the primary structure and providing local heating of the propellant lines and valves as necessary.

The thermal control subsystem can be considered as divided between the science and electronics module and the propulsion module. Construction of the science and electronics module allows the use of a passive thermal control strategy. In this strategy, all payload and satellite equipment dissipating heat (except boom mounted instruments) are thermally coupled to the satellite primary structure, to form integrated thermal mass with a long time constant.

The propulsion module contains heavy propulsion equipment such as the hydraizine tank, valves etc., a large volume of hydrazine, four ACS reaction wheels and a substantial primary structure. However, the ability of this module to maintain acceptable equipment operating temperatures in orbit cannot rely on a passive thermal control strategy. For example, hydrazine fuel line temperatures are maintained within the range +7 to +35°C by careful routing of the lines plus local heating by thermostatically controlled electric heaters.

A brief examination of the overall thermal situation confirms that heat must be conducted from the science and electronics module to the propulsion module, to improve operating temperature equalization. This can be accomplished by tight thermal coupling of the interface between the two modules, plus locating the high-power dissipation electronic boxes closest to the science and electronics module lower platform. Operating temperatures are adjusted by applying thermal coatings to selected areas of the satellite body, the use of MLI in selected locations, also the use of active local heating where required.

Thermal analysis of the boom mounted instruments performed for the Ørsted satellite, shows that a passive thermal design can be used to maintain the required instrument operating temperature ranges [4]. This design included the use of MLI around the instrument assemblies and a thermal control coating on the external surfaces of the instrument assembly enclosure.

## Magnetic Cleanliness

Magnetic cleanliness of the GSO-L satellite is required to ensure quality and validity of the scientific data gathered during the mission. Control of magnetic moment is made by using good design practices at the subsystem and satellite levels, careful part selection with magnetic screening of each item, material controls, the minimum use of magnetic materials and use of non magnetic integration tools. An Astatic magnetometer is used to screen all parts and materials for magnetic moment. The CSC magnetic error budget includes an allowance of 0.4 nT for the satellite magnetic moment.

This relationship between the allowable satellite magnetic moment and the distance from the satellite center of gravity to the CSC magnetometer, for a boom tip (CSC) magnetic disturbance of 0.40 nT is shown in Fig. 11. Also shown is the selected design point, which determines the maximum allowable total satellite magnetic moment which can then be allocated among the satellite components and subsystems on a basis of weight.

The magnetic cleanliness program aims at reducing associated costs by very early design phase control of magnetic materials content, by education of personnel in the use of simple magnetic test instruments, and in understanding the philosophy of a low and controlled magnetic budget. Magnetic tests are carried out by the Project at unit acceptance, and the satellite magnetic survey is considered part of the system calibration.

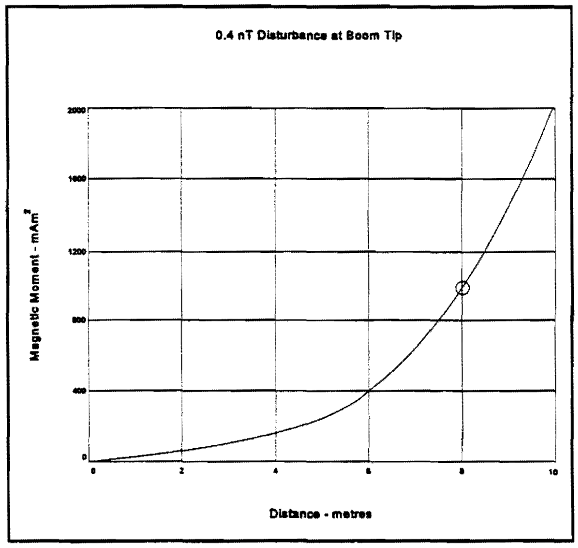


**Fig. 11: GSO-L Satellite Allowable Magnetic Moment Versus Distance from CSC Magnetometer to Satellite Center of Gravity**

## Conclusions

This study shows that a small satellite approach can be implemented for a 200 km high inclination LEO. It provides a low-cost option to the traditional larger platforms thus expanding the role of satellites for Earth observation and particularly the study of the Earth's magnetic field.

Magnetic cleanliness is a critical requirement for magnetometry missions. With the low mass of small satellites, it is easier to control the total magnetic moment of the satellite. The suggested GSO-L design is therefore an example of how a small satellite, even with a classical approach to the propulsion system (monopropellant hydrazine), can be used to provide world-class scientific data in a very low earth orbit.


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W.R. Baron is the Senior Consultant for the Small Satellite Department of CRI A/S Denmark. Prior to joining CRI, Mr. Baron was a senior engineering specialist with Ford Aerospace Corporation (now Space Systems Loral) Palo Alto and a member of the technical staff of TRW systems, Redondo Beach, California. During this period of approximately twenty-five years, he contributed to many successful satellite programs with these organisations. Mr. Baron is a member of three technical societies, is registered as a European Engineer and is the author of six patent disclosures. He has published and contributed to numerous technical papers.

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P.L. Thomsen is the lead System Engineer for the Small Satellite Department of CRI A/S, Denmark. He joined CRI in 1991 after his graduation at the Danish Engineering Academy. Mr. Thomsen was assigned as system engineer during the initial design phase of the Ørsted project (1992) and primarily contributed to the design of electrical interfaces, and operations. Mr. Thomsen has authored and co-authored several technical papers in the field of mini-satellites.

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K.S. Schultz is a system engineer for the Small Satellite Department of CRI A/S, Denmark. Prior to joining CRI, Mr. Schultz was co-investigating engineer at Technical University of Denmark, developing digital hardware and software for the star imager which is an autonomous attitude determination instrument for satellite applications. As part of the activities at CRI, Mr. Schultz has designed the on-board data handling and communication subsystem for the GSO satellites.