MAPPING OF EARTH'SMAGNETIC FIELD WITH THE ØRSTED SATELLITE

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1. Abstract

The Danish Ørsted satellite will carry three science experiments with the objectives of mapping the Earth's magnetic field and measuring the charged particle environment from a 780km altitude sun-synchronous polar orbit. The science data generated during the planned one-year mission will be used to improve geomagnetic models and study the auroral phenomena. Comprehensive and accurate mapping of the geomagnetic field every 5 to 10 years is of particular interest to geophysical studies. As such, the Ørsted science data return will complement the Magsat (1979-80) and Aristoteles (≈ 2000) mission objectives. Two magnetometers will be mounted on an 8 meter long deployable boom together with a star imager for determining the absolute pointing vector for the CSC fluxgate magnetometer. Particle detectors are mounted in the main body of the satellite. Position determination will be provided by a multi-channel GPS receiver. The main body of the 50 kg satellite is shaped as a box with modular electronic boxes and includes sub-systems in areas of Power, Attitude Control, Communication, Command & Data Handling, Structure and Mechanisms. The Ørsted satellite is planned to be launched as an auxiliary payload on either Ariane 4 or a Delta launcher in early 1995.

2. Introduction

This paper describes the key science objectives and system design parameters of the Ørsted Satellite. The project was initiated about a year ago as a cooperative effort between a group of danish universities, institutions and industries active in European space activities.

This group has evaluated the feasibility of developing a small satellite, with the principle scientific objective of performing accurate and sensitive measurements of the magnetic field in a low, polar orbit.

Accordingly, the Ørsted satellite is named after the Danish Physicist Hans Christian Ørsted (1777-1851) who discovered electromagnetism in 1820.

3. Science Overview

The purpose of the Ørsted satellite mission is to conduct a research program in the field of solar-terrestrial physics comprising magnetospheric, ionospheric, and atmospheric physics in combination with research in the magnetic field of the Earth. A wider scope is to create a research environment called the Solar-Terrestrial Physics Laboratory that links together a number of existing research activities, which traditionally have been carried out with only limited coordination at the individual scientist level in Denmark and abroad.

Solar-terrestrial science has attracted much attention during recent years, among scientists as well as in the public. Also, distinction between man-made and natural causes of "Global Change" has become an important issue, and the concept of a steady Sun, expressed for example in the term "the solar constant", has been gradually abandoned and transformed into a broadly accepted concept of a constantly varying outer environment dominated by processes in the Sun. The magnetosphere and the upper atmosphere of the polar regions are the locations where the effects of the variable Sun are most clearly seen. The present poor understanding of the global processes determining the coupled interactions between the electromagnetic and corpuscular emissions from the Sun and the neutral and ionized species in the Earth's environment, does not allow prediction of the response of the system to changes in the solar output. As a consequence, a number of international research programs have recently been initiated to improve our understanding of the solar-terrestrial system.

Danish traditions are prominent in the field of solar-terrestrial and space physics, and Danish science groups have demonstrated a recognized capability in measuring magnetic fields with rockets and ground equipment. Several front-line research results based on this expertise have had international scientific impact. Furthermore, it has been clearly demonstrated during recent years that close collaboration between ground and space based observers is essential. Denmark is there-fore perfectly suited for an optimal exploitation of a new set of high quality magnetic observations from space.

The Ørsted Satellite will contain three science experiments:

- o Compact Spherical Coil (CSC) triaxial fluxgate magnetometer, for measuring magnetic field vectors.
- o Overhauser magnetometer for measuring magnetic field amplitudes
- o Solid-state charged particle detectors for measuring electrons, protons and alpha-particles.

4. Satellite

The satellite will be launched into a polar, circular, sun-synchronous, low earth orbit as an auxiliary payload on an Ariane 4 or MD-Delta II launch vehicle. Fig. 1 illustrates the satellite in the launch configuration and Fig. 2 depicts the satellite in the orbital operation mode with the boom fully deployed. In this configuration the satellite is gravity-gradient stabilized with the boom pointing away from the center of the Earth. Active attitude control is achieved using 3 axis magnetorquer coils. The six charged particle detectors are housed within the satellite body and look out through apertures in the solar panels. An S-band omnidirectional antenna is mounted on the underside of the satellite for communication between the satellite and the ground stations.

Two magnetometers and a star-imager comprise the tip mass which together with the deployed boom provide the satellite with passive gravity gradient stabilisation. Both the CSC magnetometer and the star-imager are mounted and aligned together on an optical bench within the boom structure. In this way the effect of any satellite magnetic field on the CSC magnetometer is reduced to an acceptable level and the requirement for attitude determination of the CSC magnetometer to within 20 arc seconds can be met. Boom flexing effects on the CSC attitude determination are therefore also eliminated.

4.1. Configuration

Ørsted is shaped as a box, 680 mm high x 450 mm wide x 340 mm deep. The four sides and the top of the satellite body accommodate solar cells to provide electrical power. An S-band antenna protrudes from the bottom face of the body.

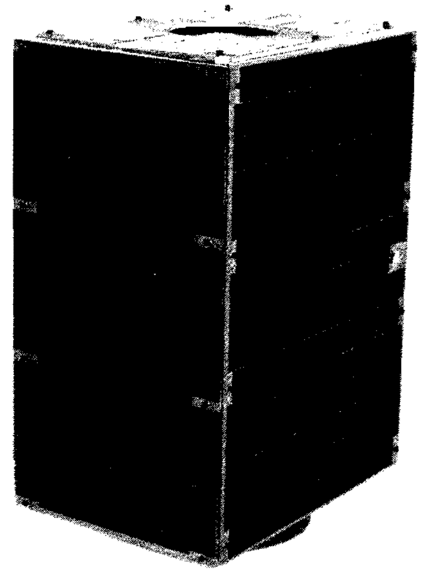

Figure 1: Ørsted Satellite in the Launch Configuration

Within the satellite body are the electronic boxes for the satellite and scientific payload functions, two battery packs, magnetorquer coils, sun sensors, the charged particle detectors, and the deployable boom. Following launch, separation and autonomous attitude acquisition, the boom is deployed into the orbital operation configuration illustrated in Fig. 2.

Interface with the launch vehicle adapter is through the satellite separation mechanism mounted on the underside of the satellite body. Mounted above the separation mechanism are the lower honeycomb platform, the vertical 'H' beam primary structure and an upper platform. Within the main structure envelope are two rows of electronic boxes with space for nearly 10,000cm^2 of printed circuit boards. Prior to deployment, the boom is packaged in a cylinder together with the two magnetometers and the star-imager. Following release of hold-down latches, boom deployment is initiated in response to a ground command.

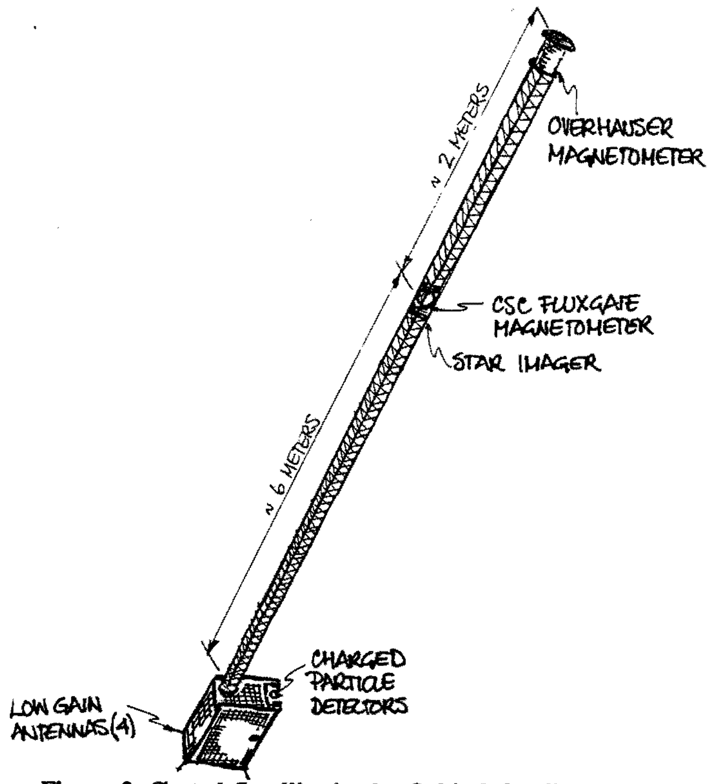

Figure 2: Ørsted Satellite in the Orbital Configuration

The satellite functional block diagram is shown in Fig. 3. Provision is made for ground power supply during testing and the launch campaign. Telemetry data from the scientific experiments plus satellite housekeeping is stored in the on-board memory and then downlinked when a ground station is in view. Attitude determination is based on data from the CSC magnetometer, the sun sensor and the GPS receiver. The attitude control system uses this data for autonomous satellite attitude control via the magnetorquer coils. All command and data are handled by a central computer, capable of providing autonomous control of all satellite functions except for boom deployment which is accomplished under ground control. A hibernation mode is also provided in the event of interruption of ground contact for long periods of time.

As configured in Fig. 1 and Fig.2, the satellite meets the mass constraint of 50 kg maximum imposed by the launch contractor.

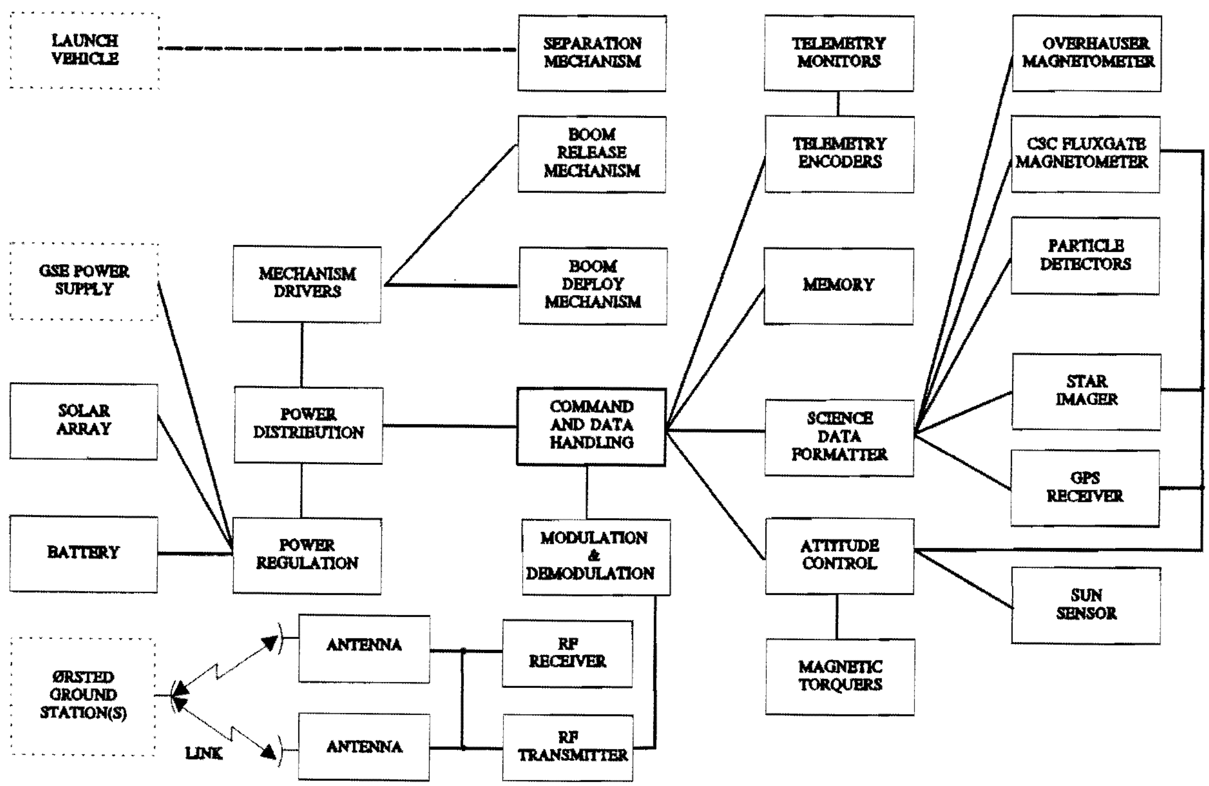

Figure 3: The Satellite Functional Block Diagram

A summary of the satellite mass budget is given in Table 1 which shows a current mass margin of 5.3 kg. Features of the satellite are summarized in Table 2.

ITEM	Mass kg.
Structure (incl. boom)	13.1
Electric Power	11.4
Attitude Control	3.9
Communication	2.7
Command Data Handling	2.2
Thermal	0.3
Integration	2.0
Scientific Payload	9.1
Total	44.7
Allowable Mass	50.0
Mass Margin	5.3

Table 1: Satellite Mass Budget

Body Size	H650 x W450 x D340mm
Mass	50 kg
Stabilisation	Gravity Gradient with yaw controle
Power	GaAs/Ge body mounted solar panels. 44 Watts EOL
Primary Structure	'H' beam with honeycomb platforms
Attitude Control	Passive - Gravity Gradient Active - Magnetorquers
Communication	S-Band with ESA Standard Packet Telemetry
Data Handling	13 hours Data Storage Down-link 150 K bits/sec Autonomous Control
Thermal Control	Passive
Position Det.	GPS
Boom	Deployable 3 longeron
Launch Vehicle	Ariane 4 / MD Delta II

Table 2: Features of the Satellite Design

4.2. Science Payload

The scientific payload fulfills two major objectives. One is to map the Earth's external and internal magnetic fields. The other function is to provide measurements of the high energy particle radiation in the upper polar atmosphere.

Ørsted will carry three science experiments in order to meet these objectives:

- o Compact Spherical Coil (CSC) triaxial fluxgate magnetometer, for measuring magnetic field vectors with an angular resolution of 20 arcsec.
- o Overhauser proton-precession magnetometer for measuring magnetic field amplitudes, with a resolution of 1 nT.
- o Solid-state charged particle detectors (six) for measuring electrons from 30 KeV to 1 MeV, and protons and alpha-particles from 200 KeV to 100 MeV.

Two magnetometers will be mounted on the 8 meter long deployable boom, whilst the particle detectors are mounted in the main body of the satellite.

A Star Imager, mounted on the same boom together with the CSC magnetometer, is also classified as part of the payload. This provides the absolute pointing vector to an accuracy of 20 arcsec for the CSC fluxgate magnetometer.

CSC Magnetometer

A fluxgate magnetometer sensor is a small magnetic field transducer with a ferro-magnetic core inside a secondary pick-up coil. The CSC triaxial fluxgate magnetometer consists of three such fluxgate sensors mounted with their axes at right angles to each other inside a set of small lightweight coils arranged on a spherical surface. The package is 9.0 cm in diameter, 10.2 cm high with a triangular footprint and weighs approximately 300 g.

To meet the less than 3 nT magnetic field disturbance requirement, the satellite must be magnetically clean. Hence, the magnetometer will be mounted on the boom 6 m away from the satellite body.

Overhauser Magnetometer

The Overhauser magnetometer sensor contains coils for proton-resonance excitation and detection, and a resonator for Electronic Spin Resonance (ESR) pumping of a nitro-oxide solution.

The sensor will be mounted on the boom tip 8 m away from the satellite body. It weighs approximately 1.0 kg and is contained in a cylindrical package 15 cm long x 10 cm diameter. This magnetometer has magnetic cleanliness requirements of 1 nT.

Particle Detectors

The solid-state charged particle detector experiment detects electrons in the energy range from 30 KeV to > 1 MeV and protons and alpha particles from 200 KeV to > 100 MeV. Six individual detectors will be mounted in the satellite body approximately 3 cm diameter x 4 cm long with differing fields-of-view between 15 degrees and 45 degrees. The detectors are mounted to look either in the direction of the satellite longitudinal axis such that the detectors are looking upward during passes over the northern hemisphere, or in a direction perpendicular to the longitudinal axis.

Star Imager

The star imager measures the satellite attitude by matching star constellations in the camera field-of-view with an on-board star catalog. Mapping is done by a 604 x 576 pixel charged coupled device (CCD) camera. The CCD is cooled by a passive space radiator. In addition, the CCD chip must not be exposed to the sun for more than a few seconds to prevent chip thermal overload and blooming of the images.

4.3. Electric Power Subsystem

The electric power subsystem comprises a solar array, which converts sunlight to electrical power, a battery which stores electrical energy for use during eclipse, and power control electronics to control battery charging, output voltages and the busloads. Average solar Array Output over an orbit is 44 W. Average power consumption by the satellite is about 23 W (excluding battery charge) leaving a margin of about 10%.

Solar Array

The body-mounted solar array is composed of four solar panels mounted on the X and Y faces of the satellite body and one panel mounted on the top (+ Z) face. These panels are assembled from 2 x 4 cm Gallium Arsenide on Germanium solar cells which provide 44 Watts of electric power at end-of-life (EOL). Average solar cell conversion efficiency is 18.0 % under standard conditions of 1 sun, AMO and of a temperature of +28° C. Figure 4 shows the EOL power output profile of the solar array during the orbit.

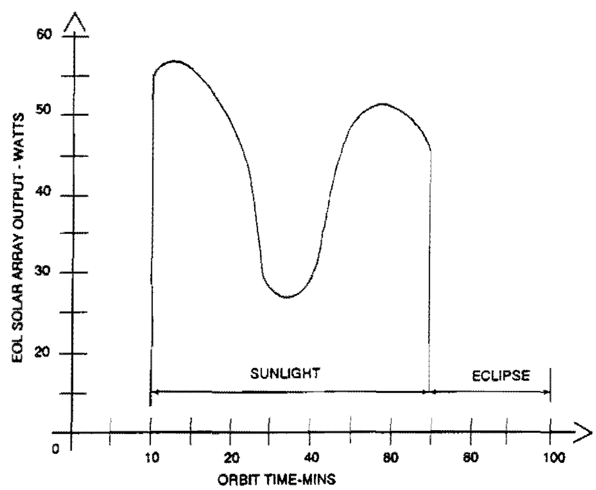

Figure 4: EOL power output profile of the solar array

Each solar cell is covered with a 150 μm thick cover-glass and the adhesives, interconnect materials, etc., are all selected to meet the minimum one year lifetime requirement.

Stringent requirements to reduce magnetic fields from the solar array are met by careful material selection and electrical design. Use of magnetic materials is minimized and all materials will be measured for magnetic moment. Electrical design employs "back-wiring" techniques for each circuit of solar cells as described in References 1 and 2. All power output wiring will be in twisted pairs.

Battery

During eclipse and peak power requirements in excess of solar array capability, power is provided by the battery. The battery consists of two identical packages each comprising five sealed Nickel Cadmium battery cells of approx. 7 ampere hours capacity. Two battery packs are used for mechanical reasons with one pack mounted on each side of the satellite internal structure. All main power wiring is redundant and magnetic moments are minimized. During nominal orbital operation the battery depth-of-discharge will not exceed 15%. Temperature control of the batteries will be accomplished by thermally coupling the batteries to the primary structure.

Power Control Electronics (PCE)

The PCE include a solar array peak-power tracker, separate DC/DC converters for essential and non-essential loads and all necessary load control and protection devices. The peak power tracker maximizes the available power output from the solar array during variations in solar vector angle and the array temperature. It also regulates the power bus voltage and thus provides control of the battery charging regime. DC/DC converters provide the interface between the electrical loads and the power bus. Redundant converters are provided for the essential loads. All DC/DC converters have an efficiency of not less than 90%.

Load control comprises the necessary commandable relays. Protection is provided by a combination of converter voltage fold-back under overload, redundant fuses, current limiters, etc. Further protection of the electric power system is provided by turning off the power to all non-essential loads should the battery voltage fall below the safe limit.

Analogue telemetry (TM) of all voltage, current, temperature and command status for the power sub-system is generated within the PCE. This TM data is forwarded to the Command and Data Handling Sub-system (CDH) for processing.

4.4. Structure

The primary load bearing structure consists of a fabricated "H" beam with upper and lower honeycomb panels as shown in Fig. 5. The primary design drivers leading to the selection of this structural configuration were:

- o Maximize the utilization of the allowed volume
- o Provide a readily fabricated structure and components
- o Maximize the diameter of the boom and thus maximize the boom deployment torque.
- o Accommodate the CSC magnetometer and star imager within the boom envelope.
- o Provide modular electronic boxes for ease of integration and test.

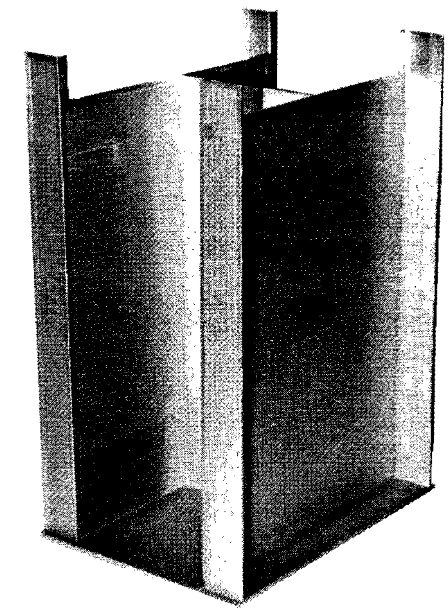

Figure 5: Main Structural "H" beam

The main structural "H" beam is mounted above the separation mechanism. This configuration forms two vertical compartments, one on each side of the web of the "H" beam. The deployable boom assembly is mounted in one of these compartments. Two rows of modular electronic boxes are mounted in the other compartment each row of boxes separated by a center shelf.

A preliminary analysis using data from Reference 3 shows that this configuration has considerable structural margin in the dynamic buckling and compressive modes. The first fundamental frequency in the lateral and longitudinal axes exceeds 100 Hz, and therefore meets the launch vehicle requirements for auxiliary payloads. Detailed structural analyses will be performed later in the project.

The deployable boom consist of three coilable longerons as shown in Fig. 6. The longerons are separated by radial spacers and tensioned by cross-wires when deployed. This boom configuration provides high inherent deployment torque without the need for springs or other mechanical devices except for a restraining wire in the center of the boom to control the deployment. Boom retraction is not required from the fully deployed position.

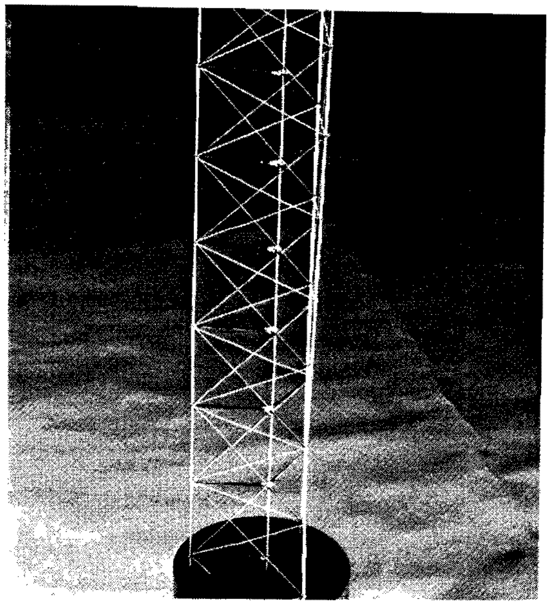

Figure 6: Boom Configuration

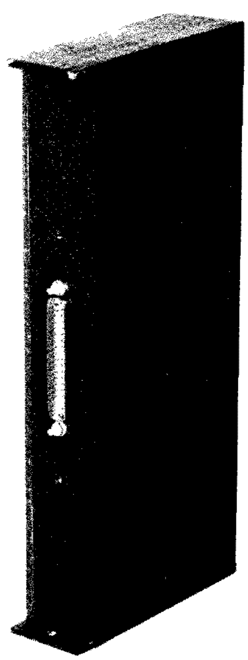

Figure 7: Modular Electronic Box

Fig. 7. shows the modular electronic box which can accommodate two printed circuit boards. Access to the interior of the box is via removable side plates. The box is attached to the web of the "H" beam structure by two long bolts which pass through the box from the front. A thermal interface is provided between the rear surface of each box and the beam structure for removal of heat. Up to three electrical connectors can be accommodated on the front of each box.

Honeycomb solar panel substrates attached to each side and the top complete the structure.

4.5. Command and Data Handling Subsystem (CDH)

The satellite command and data handling subsystem will provide the following processing capabilities.

- o Satellite housekeeping data acquisition and processing.
- o Scientific experiment data acquisition and processing when not included as part of the experiment package.
- o Time management
- o Data memory management
- o Data compression
- o Telemetry (TM) and telecommand (TC) format management
- o Command validation, distribution and execution
- o Attitude control data processing
- o Attitude monitoring and commanding

Data acquired by the various science instrument and housekeeping sensors are routed to the on-board data handling subsystem. Here they are monitored and pre-processed prior to storage. If the monitoring reveals any out-of-limits events, it is reported to the on-board supervisory facility which decides on how to respond to the event and generates commands to be executed. A Master Schedule is included releasing the timetagged telecommands as time comes due. After monitoring and preprocessing the acquired data are stored in the TM memory for later retrieval. When the spacecraft comes into view of a ground station and ground contact is established, data are retrieved, packed, and down-linked. During the periods of ground contact telecommands are also up-linked to the spacecraft, decoded and executed or stored for later execution in accordance to their time-tags.

End-to-end data flow analysis indicates, that a design based on a Central Processing Unit (CPU Intel 80C186XL 20 mHz.) has sufficient processing capacity. The basic approach to timing for the system is that the

CDH issues regular "trigger"signals for each instrument. Depending of the serial interface type, the instrument then responds with its digital signal result, and the CDH will time tag the result according to the time, when the trigger signal was issued. The circuit will contain a 32 bit time counter with one millisecond resolution used for time tagging.

Telecommands will be routed to the CDH via the satellite communication receivers. Telecommand coding and formatting will comply with ESA Packet Telecommand Standard Document ESA-PSS-04-107.

Science instrument data rates, to be handled and stored by the satellite are summarized in Table 3 below.

Autonomous control will include programming to automatically handle the initial satellite operations and following release from the launcher.

Deployment of the boom will be initiated and controlled by ground telecommand. Deployment will take place when the satellite is in view of a ground station and the deployment monitored from the ground by telemetry data. CDH will store telecommands and automatically execute them at the right time and conditions.

The telemetry generation function includes acquisition and formatting of scientific and housekeeping data and preparation of data for transmission via the communication down-link. Telemetry data and format will comply with ESA Packet Telemetry Standard Document ESA-PSS-04-106. Telemetry format will be constructed to avoid ambiguous frame and format identification. The telemetry data bit-stream will be generated by sampling science and housekeeping channels in accordance with sampling rates and sequences which are defined by individual subsystem requirements.

All telemetry data will be handled on a "store and forward" packet basis. The telemetry function will include sufficient channels for satellite housekeeping voltage, current and temperature monitoring and command status.

Storage of scientific and housekeeping data is provided for up to 13 hours continuous satellite operation. Data storage uses semi-conductor memory technology. Due to the radiation conditions in the Ørsted orbit, the memory is protected against bit errors by the use of a hardware Error Detection And Correction circuit.

The RAM disk is a memory used for storage of the scientific data received from the different instruments since last radio contact to the ground. Storage capacity for formatted data will be 16-20 MByte.

Authorized control commands from a ground station will have priority over autonomous operation.It will also monitor electric power levels and battery state-of-charge and if a deficiency is found, it will deactivate selected on-board loads until the power deficiency is corrected.

During normal operation, the CDH will be capable of supervising and operating the satellite and science payload without ground control for periods of at least 12 hours. If ground contact is not established within that period, the satellite will automatically enter an averaging mode to preserve science data until ground contact is established.

All satellite software will be capable of being reconfigured and reprogrammed from the ground. The necessary software and command facilities will be included to accomplish validated software modifications including the autonomous control functions and attitude.

Source	Normal Mode	Burst Mode
CSC Magnetometer	540 bits/second	5.40 kbits/second
Overhauser Magnetometer	20 bits/second	
Star Imager	323 bits/second	1.67 kbits/second
Particle Detectors (Average)	177 bits/second	177 bits/second
Total Science Data Rates	1060 bits/second	7.27 kbits/second

Table 3: Ørsted Science Data Rates.

4.6. Attitude Control Subsystem (ACS)

The attitude of the satellite will be monitored using sun sensors and the magnetometers and controlled during all phases of satellite operation following release from the Launch platform.

Stabilization of the satellite attitude will be accomplished by passive and active techniques. The passive technique employ gravity-gradient stabilization employing a deployable boom with a tipmass consisting of the two payload magnetometers and the star imager. The active technique will use magnetorquer coils located in three axes within the satellite body. The ACS will maintain a yaw angle variation of +/- 10 degrees to optimize the power output of the solar panels and produce exposure of the star imager aperture to the sun. If the sun gets in the star imager field of view, it will not be able to provide attitude data for the CSC. The active attitude control system will be capable of recovering the correct attitude from an "upside-down" attitude where the boom points towards the center of the Earth.

The magnetorquer coils will operate in "High" or "Low" modes. In "High" mode, the coils are activated infrequently to provide large attitude corrections. In the "Low" mode, semi-continuous very low current magnetorquer operation is used to correct attitude drift. In the "Low" mode of operation, the resulting magnetic dipole moment generate a field of less than 1 nT at the CSC magnetometer.

4.7. Thermal Control Subsystem (TCS)

Thermal control of the satellite is accomplished by passive thermal design using multi layer insulation (MLI) and radiative surfaces. All equipment dissipating heat is thermally coupled to the satellite primary structure.

The solar panel temperatures ranges between + 70° C and -34° C, depending upon the location of the panel and the orbital position of the satellite. Satellite internal body temperatures are expected to be within the range of + 4° C to + 12° C under the same conditions. Verification of the thermal design and equipment temperatures will be performed using more detailed modelling and testing.

4.8. Communication Subsystem (COM)

Transmission of TM & TC is accomplished through the communication subsystem when the satellite is in view of a ground-station. Telemetry and payload data are transmitted to the ground via redundant transmitters. The transmitter is powered "on" at a time and with a low duty-cycle to conserve power. Redundant command receivers are provided for the command function. Forward error correction decoding is also provided for the commands. There will be no radio transmission during countdown, the launch and at least 2 minutes following separation from the launch platform.

The telemetry down-link is subjected to both Reed-Solomon and convolutional encoding yielding a negligible data package loss rate. When the spacecraft comes into view of a ground station, telemetry is down-linked to the ground. When the spacecraft is outside the view of a ground station a spacecraft heart beat is sent out comprising spacecraft identification, position, and various status telemetry.

The telecommand up-link is subjected to Reed-Solomon encoding. If telecommand transmission fails the telecommands are re-transmitted. The telecommand up-link provides security facilities to avoid unauthorized access and to secure system integrity.

Key COM parameters are summarized in Table 4 below.

Parameter	Uplink	Downlink
Carrier frequency	2114 mHz.	2296 mHz.
Channel bit rate (BPSK)	256 kbit/s	256 kbit/s
Information bit rate	112 kbit/s	112 kbit/s
Probability of frame loss	1×10^{-7}	1×10^{-8}
Transmitter power (HPA)	13 dBW	0 dBW
Antenna diameter on ground	1.7 m	-

Table 4: Key Communication Parameters

4.9. GPS receiver

The position of the satellite will be continuously determined within 50 meters in any direction by a receiver using the Global Positioning System (GPS).

The GPS equipment comprises a multi-channel, continuous-tracking receiver and antenna. The receiver uses the C/A code on the L1 frequency carrier, and provides three dimensional position and velocity of the satellite together with UTC time all in a digital format to the CDH nominally once per second.

UTC time accuracy is within ± 1.0 microseconds and velocity accuracy $\pm$ 0.5 meters/second.

The GPS Antenna is mounted on the top face of the satellite body to view as many of the GPS satellites as possible.

4.10. Magnetic Cleanliness

Magnetic cleanliness of the 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 and the minimum use of magnetic materials. The guidelines developed for the CLUSTER program (Reference 2) are being implemented for this project.

An Astatic magnetometer is used to screen all parts and materials for magnetic moment. This instrument is fully described in the above reference.

The relationship between the allowable satellite magnetic moment and the boom length for a boom tip magnetic disturbance of 1.0 nT is shown in Fig. 8.

The benefit of increasing boom length can be clearly seen, however, this benefit must be traded against increased boom mass, complexity and cost.

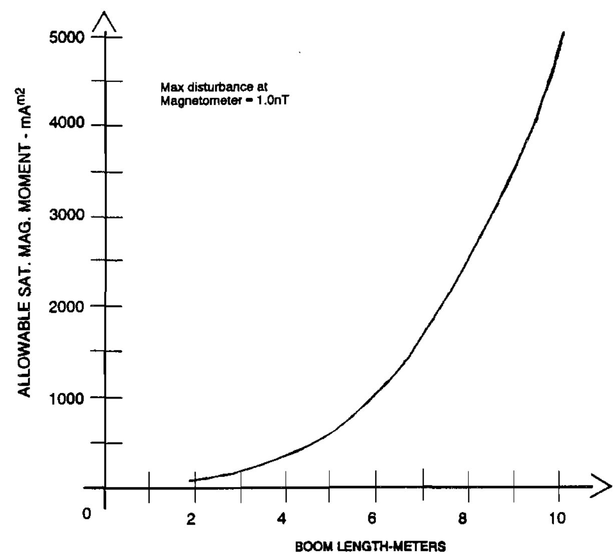

Figure 8: Sensitivity of Satellite Allowable Magnetic Moment to Boom Length

5. Environment

The Satellite will be launched by either the Ariane 4 or the McDonnal Douglas Delta II launch vehicle. The design is compatible with the environments requirements of either launcher vehicle, in terms of shock, vibration, interfaces, temperature, barometric pressure,acceleration, acoustic noise, radiation etc.

6. Integration and Test

The satellite system level integration and test will be performed in clean-room conditions of class 100,000 or better.

Prior to satellite integration all subsystems except primary structure and thermal items will receive environmental and electrical testing. The subsystem environmental testing includes the environments to which the satellite will be exposed. Prior to and following each environmental exposure, electrical parameters are measured. Structural materials and components receive static load tests to verify performance.

Satellite integration and test will follow the plan in Fig. 9. During integration, equipment performance checks will be performed on all electrical subsystems.

Following integration, a complete functional test of all subsystems will be carried out and performance characteristics confirmed. After completion of the post-integration functional test and mass properties verification, a series of environmental tests will be applied to both the qualification and flight model satellites. The flight model satellite acceptance test sequence is shown in Fig. 10. Test levels and duration for the flight model are structured to meet the requirements for both the Ariane 4 and MD-Delta II launch vehicles.

The tests also confirm orbital operation capability and will be conducted in accordance with a satellite environmental test plan.

Magnetic cleanliness of the satellite will be checked by mapping the satellite's static and dynamic magnetic fields in a magnetic test facility. Finally an integrated system test is conducted on the satellite to verify end-to-end performance.

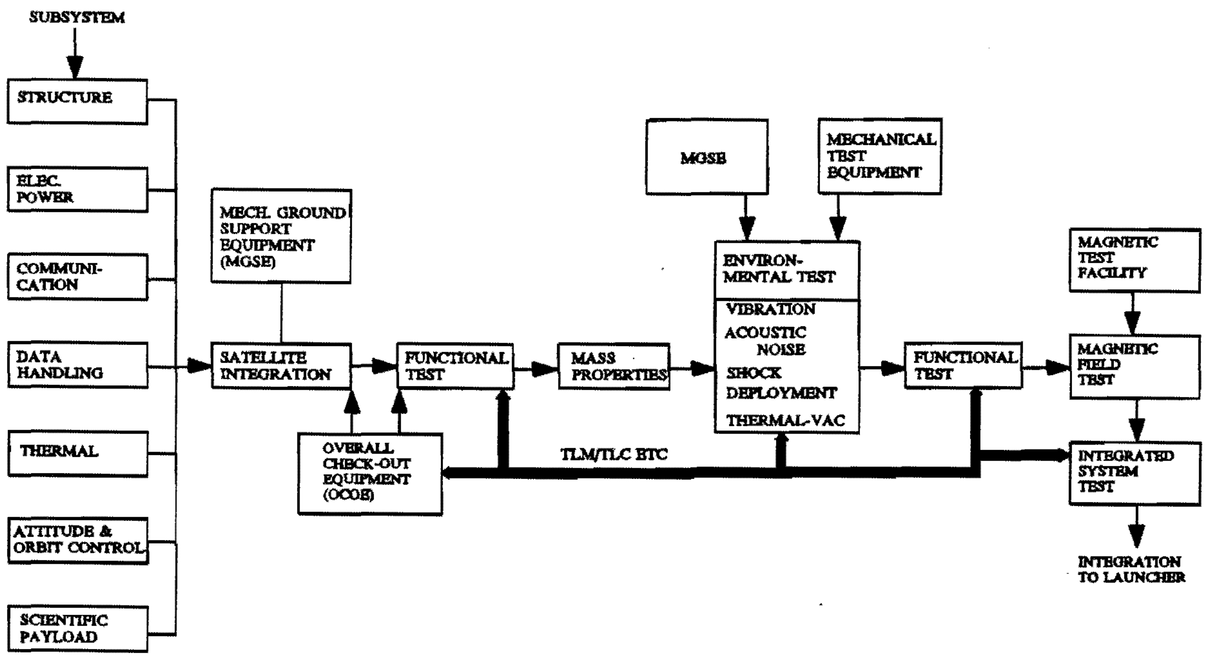

Figure 9: Satellite Integration and Test Plan

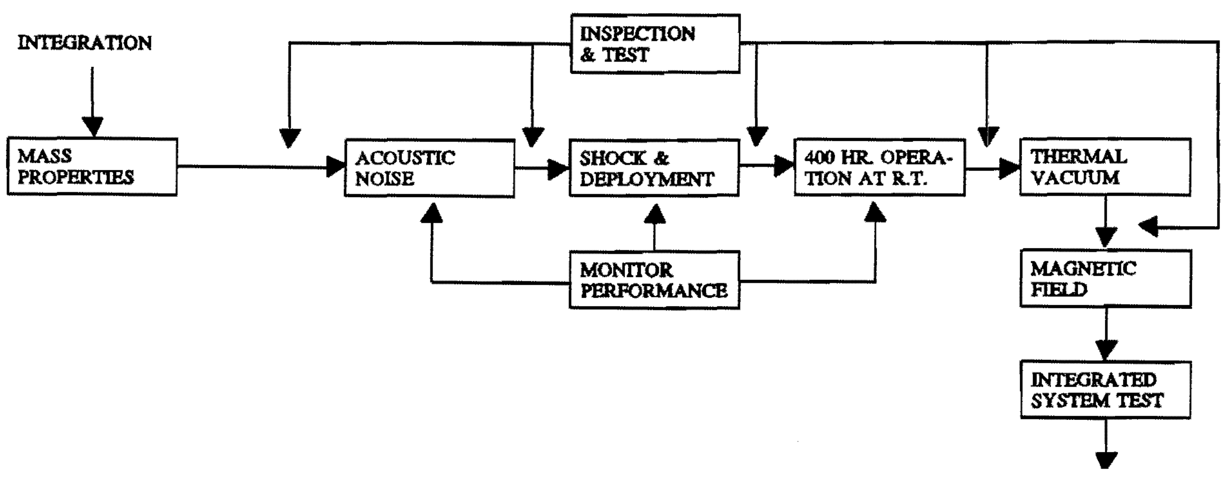

Figure 10: Satellite Acceptance Test Sequence

7. Conclusions

The Ørsted satellite will provide accurate and comprehensive science data for further study of the Earth's magnetic field and charged particle environment by the international scientific community. The science return from the one-year Ørsted mission is expected to be of high quality. It is compatible to the NASA MAGSAT (1979-80) and the planned ESA/NASA Aristoles (around 1998) missions. The Ørsted project organization is comprised of a joint group of Danish universities and private industry involved with European space activities, providing a unique opportunity to demonstrate the visibilityof small satellite missions with well-focused science objectives. The task may be ambitious. However, the science payoff is potentially significant considering the relative low cost and quick turnaround from concept to launch.

8. References

1. W.R. Baron, *"The Solar Array for the Pioneer Deep Space Probe"*, proceedings of the IECEC Conference, Miami Beach, Florida, USA, 13-17 August 1967.

2. *"Guidelines for Magnetic Cleanliness on the CLUSTER Satellite"*, ESA document CL-EST-RS-1002/EID C, Issue 1, Part 02, Section 01; 11 June 1990.

3. James R. Wertz & Wiley J. Larson, *"Space Mission Analysis & Design"*, Kluwer Academic Publishers, The Netherlands, 1991.

4. Hansen, D.B & P.Thomsen, *"Ørsted Satellite System. End-to-End Data and Control Flow Analysis"*, The Ørsted Project, CRI Internal Report, 1992.