Aspects of SUNSAT main structure design and manufacturing

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Abstract

This paper describes some overall mechanical design aspects of the Stellenbosch University satellite, 'SUNSAT'. An introduction to the SUNSAT project is given. The mechanical design requirements are described. The design of the module trays, which constitutes a major part of the main structure, is discussed. The design and manufacturing of the horizon sensor housing and the imager bearing housings are described to illustrate the influence of the design requirements, the use of solid modelling CAD software and the application of design for assembly principles in the satellite's mechanical design. The resulting design satisfies the functional and financial constraints.

Introduction

This paper describes the mechanical design and manufacture of the trays that form the main structure of SUNSAT, as well as the horizon sensor housings and the imager bearing mountings. The particular requirements that the mechanical design has to satisfy, and the ways in which these requirements were met, are outlined. A companion paper describes the design of the base plate and launcher mountings in detail.[1] The detail design for other mechanical components, e.g. the imager optics and the reaction wheels, is still in progress.

SUNSAT is an acronym for Stellenbosch University Satellite. The primary motivation behind SUNSAT is to increase engineering design opportunities for graduate students, promote interest in technology through school interaction programs, and increase industrial and international interaction. The current SUNSAT development team consists of about 30 post-graduate students (mostly in electronic engineering) and academic and technical staff members of the University of Stellenbosch. SUNSAT is therefore a typical educational micro-satellite, with a weight of 60 kg and a cubic size of 450 mm. It is designed to carry an amateur radio transponder, a store-and-forward communication system and a three-colour imaging system with a resolution of 15 meter. Its planned functional lifetime in orbit is 4 years.

The Department of Electrical and Electronic Engineering at the University of Stellenbosch started the SUNSAT micro satellite project in 1989. Studies of the Department's capabilities and other programs at the universities of Surrey and Berlin led to the January 1992 baseline design compatible with ARIANE launch requirements. In 1994, NASA expressed interest in launching SUNSAT in the same mission as the Danish Oersted magnetic research satellite,[2] both as secondary payloads on the Argos/P91-1 Delta II mission in October 1996. In exchange, SUNSAT will provide data gathered by a precision GPS receiver (that NASA will supply) and the mounting of a set of laser retro-reflectors on SUNSAT. The proposed orbit is near-polar with an inclination of 96° and an altitude varying from 450 to 850 km.[3]

Review of previous work

Thorough surveys of other international activities involving small spacecraft have been given in previous papers.[4; 5] Only a brief summary of relevant aspects will be given here.

The University of Surrey has pioneered micro-satellite technologies, beginning with its UoSAT program in 1979. Surrey's first experimental micro-satellites (UoSAT-1 & 2) were launched free-of-charge as 'piggy-back' payloads through a collaborative arrangement with NASA on Delta rockets in 1981 and 1984, respectively. Since then, a further eight low-cost, yet fairly sophisticated, micro-satellites have been placed in low Earth orbit for a variety of customers using the ARIANE Auxiliary Payload Adapter.

UoSAT-1 and 2 both used a rather conventional structure – a framework 'skeleton' onto which modules containing the various electronic subsystems and payloads were mounted. However, the need to accommodate a variety of payload customers within a standard launcher envelope, coupled with increased demands on packing density, economy of manufacture and ease of fabrication, led to the development of a modular design of a multi-mission micro satellite platform. This structure is based around a series of module trays that house the electronic circuits and themselves form the mechanical structure, onto which solar arrays are mounted as seen in Figure 1.[6]

Design requirements

The design requirements can be classified as either financial or functional requirements. The functional requirements can be further grouped according to the environment that imposes the particular requirement, i.e. before

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launch, during launch, and operation in space. Griffin & French [7] described the general requirements in some detail. Those that have an appreciable effect on SUNSAT's mechanical design are outlined below.

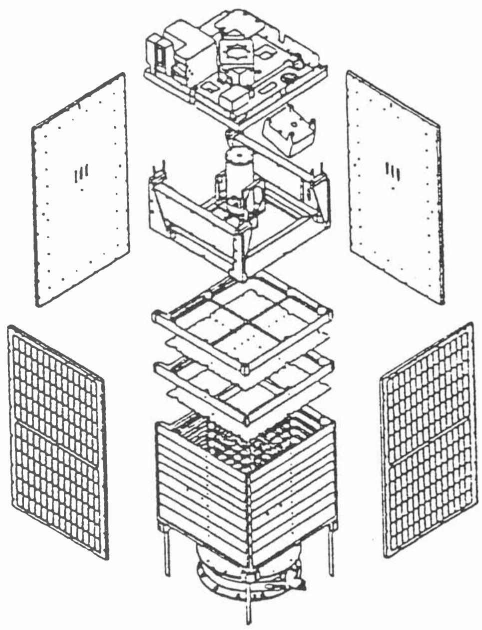

Figure 1 Exploded view of the Surrey micro satellite structure [6]

Financial requirements

As an educational project, SUNSAT is subject to severe financial restraints. The main impact in the mechanical design is that as many as possible of the structural components had to be designed to be manufactured in the university's workshop. Conventional lathes, milling machines and a 3-axis NC machine are available in the workshop. Basic sheet metal working and welding facilities are also available, but aluminium welding and manufacturing in composite materials could not be accommodated in this workshop.

A further impact of the financial restraints is that readily available, moderately priced materials and components had to be used, as far as possible.

Although it was not a primary requirement, the satellite had to be modular to allow future changes to be incorporated as easily, and therefore as cheaply, as possible, whilst not substantially increasing the cost of the first version.

Requirements related to the before-launch environment

Materials used, such as lightweight structural alloys, must not be susceptible to undue atmospheric corrosion, or must be protected against it.

In SUNSAT's mechanical design, it was assumed that the satellite will be protected during transport sufficiently that mechanical vibration and shock loads imposed will not exceed (in magnitude and duration) those during launch.

The design of the satellite must be amenable to being assembled and disassembled a number of times during testing.

Requirements related to the launch environment

NASA imposed tight constraints on the total mass (60 kg maximum) and the location of the centre of mass (not further than 280 mm from the separation plane and within a cylinder of 20 mm diameter centred on the longitudinal axis).[8] This limits the moment that the payload imposes on the satellite-launcher interface and ensures a safe separation after launch. If the centre of gravity is too far from the longitudinal axis, the separation springs would force the satellite into an undesirable tumbling movement.

Dimensional limitations are imposed on SUNSAT by the space available in the launch vehicle. The maximum dimensions for the body are 450 mm × 450 mm × 500 mm.

One of the most stringent design requirements imposed by launch is the stiffness requirement. NASA requires that the first natural frequency of SUNSAT mounted on the launcher must be at least 70 Hz. This requirement dominated the design of the base plate.[1]

Launch is characterised by a highly stressful environment for the spacecraft for a relatively brief period (typically a few minutes). The spacecraft is subjected to significant axial loads by the accelerating launch vehicle, as well as lateral loads induced by steering and wind gusts. A 10g steady acceleration, acting simultaneously in the axial and both lateral directions, was used as steady load criterion for large components.[1] Finite element analysis methods are typically used for the strength design of these components. Manual stress calculations were used for designing brackets, sensor housings, support beams, etc. These small components were designed to sustain 20g, because of greater uncertainties in their load conditions and the simple analysis methods used in their design.[9] In lieu of structural static load testing to provide flight qualification, a 'no flight' factor of 2.0 times the maximum flight stress levels (limit load factors) was used in the structural analysis.[8] This safety factor of 2.0 was required for both main structural components and smaller components, because neither will be statically tested.[9]

Shock loads during satellite-launcher separation, substantial mechanical vibration and severe acoustic energy input, particularly while the rocket engine noise reflects from the ground, have to be sustained without structural damage. Dynamic loads of typically 13g rms acceleration at frequencies in the 10 to 2 000 Hz band must be sustained. These requirements are discussed in more detail in [1].

Enclosed volumes have to be adequately vented to avoid the build-up of pressure differentials due to the rapid reduction in ambient pressure during the launch.[7]

Requirements related to the Space environment

The space environment is characterised by a very hard vacuum, very low gravitational acceleration, ionising radiation, extremes of thermal radiation source and sink temperatures, and micro-meteoroids.

Vacuum

Material selection is crucially affected by the extent to which materials outgas (emit vapour) in a vacuum environment. Outgassing should generally be minimised in SUNSAT as the vapour can condense on optical parts of the imager or sensors. Lubricants commonly used on earth cannot be used in space because they will evaporate, resulting in cold welding of the lubricated parts.

The vacuum in low Earth orbit is however not total. Orbital operations during periods of greater solar activity, and consequently higher upper atmosphere density, produce more rapid orbit decay due to greater aerodynamic drag.[7] Minimising drag is therefore a design requirement, but with a low priority.

Weightlessness

Even though body forces are negligible, high stiffness remains a design requirement. Re-alignment of structures and instruments due to the reduction in gravitational effect has to be minimised where it is important (e.g. in the imager and the relative positions of sensors).

Radiation and micro-meteoroids

For low Earth orbit, radiation is not a major design consideration. The main design requirement derived from radiation is that all electronic components must be protected from radiation by at least 2 mm aluminium to restrict degradation due to total dose effect and malfunctions induced by so-called single-event upsets.[7] A secondary requirement is that materials susceptible to degradation due to radiation must not be used where they would be exposed to such radiation.

The 2 mm wall thickness requirement stated above also provides sufficient protection against damage caused by micro-meteoroids.[7]

Thermal environment

The maintenance of the temperature of all the spacecraft's electronic components within appropriate limits over the mission lifetime is essential to their effective functioning. The analysis of the overall thermal balance of the satellite is not considered in this paper, but the temperature variations in mechanical components are. The sun has a characteristic blackbody temperature of 5 780 K, which acts as an energy source, while dark space, at 3 K, acts as an energy sink. Entry into eclipse and re-emergence into the sunlight results therefore in rapid cooling and heating, respectively, of external surfaces and low mass extremities. Thermally insulated portions of a satellite can typically experience temperature variations from 200 K during darkness to 350 K in direct sunlight.

The satellite structure must therefore be able to sustain substantial thermal stresses and fatigue, and the design must be such that thermal deformation must not affect the accurate pointing of sensors.

Design of general mechanical configuration

Main components

Figure 2 shows the general geometric configuration of SUNSAT. The main components from a mechanical perspective are:

- Solar panels on the 4 sides, for power supply, and rechargeable NiCd-batteries for energy storage in the lower tray.
- Onboard computers mounted on rectangular PC-Boards.
- A high resolution imager for remote sensing in the lower tray.
- Sensors to determine the satellite's attitude mounted on the top plate, i.e. the star sensor, horizon sensors, sun sensor and magnetometer.
- Devices for controlling the attitude, i.e. the gravity gradient boom (on the top plate), magnetotorquers (integrated with the solar panel assemblies) and reaction wheels (in the lower tray).
- A payload adapter mechanism (attached to the base plate) to mount the satellite on the launch vehicle and to ensure safe separation from it in orbit.

The main structure and payload adapter were optimised using a finite element analysis. The satellite was experimentally qualified to the expected stress levels, with a margin of safety, to confirm that it will be able to function after the launch.[1]

Mechanical structure

SUNSAT is built like a sandwich with a base plate and a top plate, and trays in between. The stacking of the trays forms the central structure. The advantages of a tray structure are the optimal use of the available volume and ease of fabrication, especially for the electronic hardware, and modularity of the design. Assembly costs did not have a strong influence on the design, because SUNSAT will be produced in very small numbers. A larger number of individual structural components were therefore used to simplify manufacturing and provide interfaces that promote damping of vibrations during launch.

Each tray, as well as the top and base plates, have holes on their corners through which a tie rod passes. Nuts

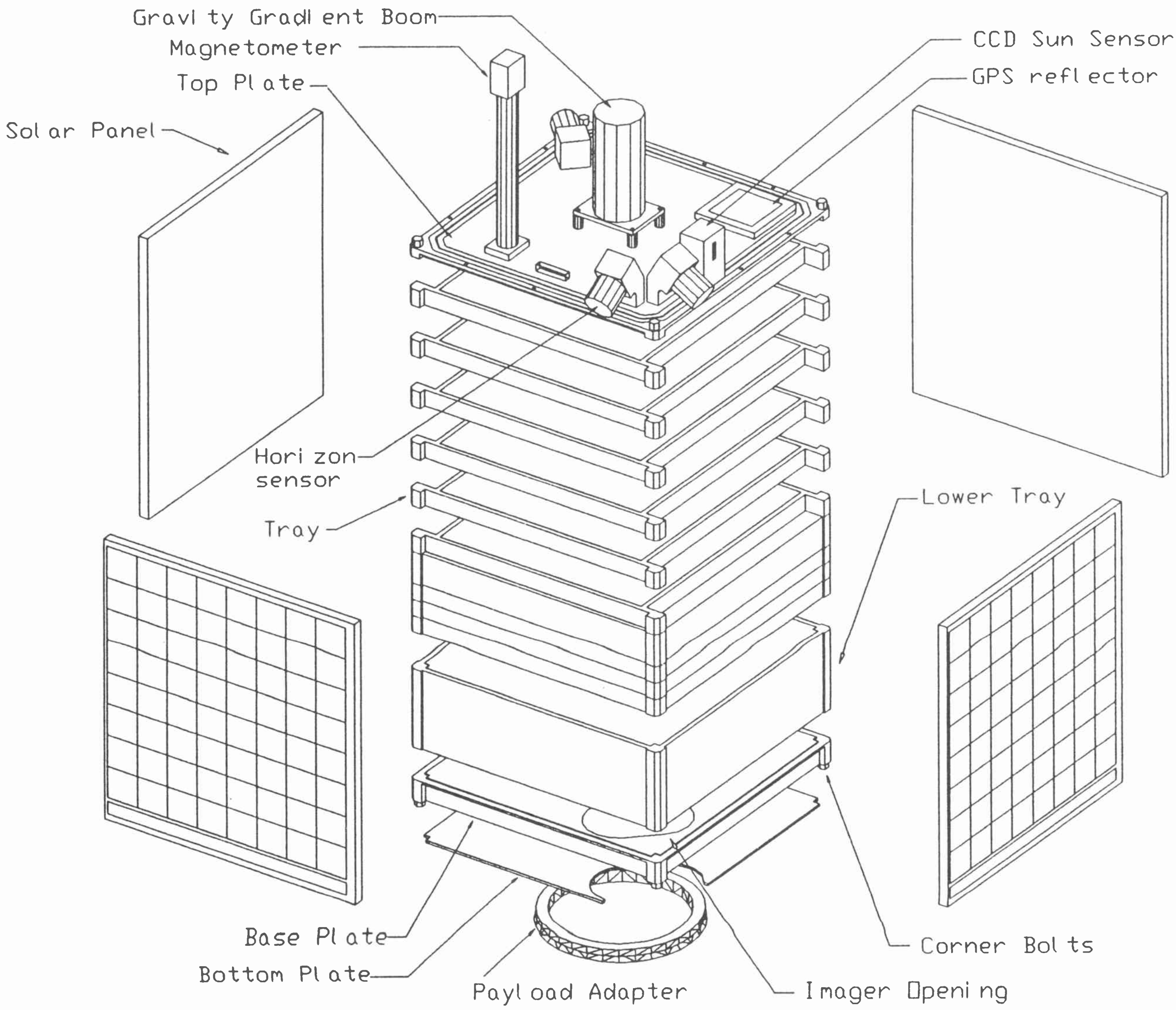

Figure 2
SUNSAT general structure configuration (exploded view)

on the threaded ends of the tie rods are torqued to press the satellite body together.

Each of the 11 trays houses particular electronic and mechanical components. A typical tray consists of 4 sides, 2 support ribs, and a printed circuit board, and forms one subassembly. Figure 3 shows an exploded view of a tray without board and support ribs. The boards are attached to the trays by screws on all sides. The two ribs reach from one side of the tray to the opposite. They support the board in the middle to increase its stiffness. The tray sides interlock with tenon joints. Pins fix the joints after assembly. Solid models of the tray sides were constructed in the CAD system during design to clarify the joint concept, ensure that sufficient material was provided where needed and to confirm assembly sequences. The Aluminium structure consisting of top plate, trays, base plate, and Payload Adapter Assembly weighs approximately 15 kg. This is 25% of the satellite's total mass.

Although a complete tray could be machined from a solid block to avoid problems in joining processes and to follow the minimum component count philosophy, the decision was made in SUNSAT to machine the four sides separately. This reduced manufacturing costs and improved functional flexibility. This flexibility proved to be a significant advantage when the additional GPS (Global Positioning System) electronics had to be accommodated at a late stage in design. The joints further provide frictional surfaces to improve damping of vibrations.

Structural materials and methods of fabrication

Aluminium alloy 7075 T6 was selected for most of the structural parts. It has a good strength- and stiffness-to-weight ratio, can sustain the expected temperatures, has a high thermal conductivity to reduce thermal shock stresses, is readily available at reasonable costs and is easy to machine. Although they were eventually not needed for SUNSAT, a number of surface-coating processes are available to allow tailoring of surface characteristics for hardness, emissivity, absorptivity, etc.

Composite materials were initially considered due to the high specific stiffness that can be achieved with these materials, but they were not used for structural components. The selection of aluminium instead of composites was based on the following reasons:

1. Space-class composite materials (that do not outgas) are not readily available in small quantities in South Africa and require more expensive manufacturing processes than aluminium. The cost constraints on a University project of this kind is such that the additional costs involved in using space class, high stiffness composite materials could not be justified.
2. Certain composite materials (e.g. carbon fibres) on the outside of the satellite are susceptible to oxidation in space due to the presence of high energy atomic oxygen. These composites have to be adequately covered to prevent this oxidation, which negates some of the mass advantages.
3. Aluminium provides better radiation shielding than low density composites. The minimum wall thickness is often determined by the radiation requirement and not only the stiffness requirement.

Stainless steel was selected where a relatively high mass or non-magnetic behaviour is needed. The reaction wheels, for example, are from stainless steel in order to obtain the required inertia.

Joining methods

Various tests and changing of electronic or mechanical components required the possibility to dismantle the satellite. SUNSAT is therefore mainly screwed together. For the final flight model, the screws will be either wirelocked or locked with Locktite, or self-locking nylon insert nuts will be used. Welding has been avoided for following reasons:

- The post welding heat treatment of aluminium 7075 T6 is costly because it has to be very accurate controlled to achieve the same material properties as before.
- Welding does not allow for disassembly.

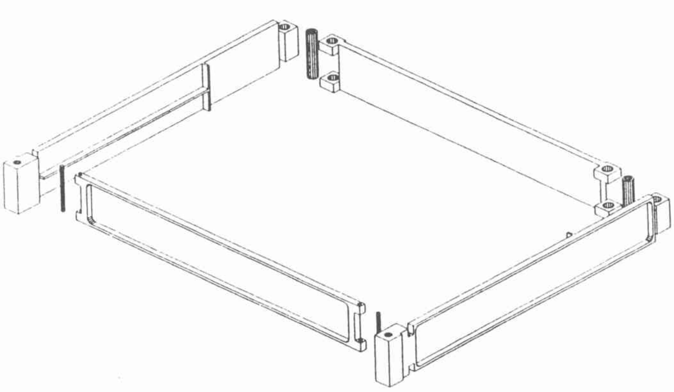

Figure 3 Example tray assembly

- Welding may lead to dimensional distortions, resulting in additional machining after assembly.
- Welding introduces the risk of trapped welding process particles coming loose in space.
- Welded interfaces will not contribute to damping of vibrations to the same extent as screwed interfaces.

Mass properties control

A detailed list of all components was maintained to ensure that the mass property constraints from NASA were adhered to. A solid modeler, AutoCAD's Advanced Modelling Extension Release 2.1,[10] was used to build a mass property model at the part level, as well as on the assembly level. In this way the mass properties could be controlled in the design stage.

Heavy parts, e.g. batteries, have been mounted on the base plate to keep the centre of mass low. All components are placed so that the resulting centre of mass lies near the centre line. The freedom in placing the GPS receiver, which does not occupy the whole space within its tray, will be used to correct mass imbalances in the lateral plane at the end of the design stage.

Environmental protection

The SUNSAT structure and, especially, the sensors were designed to provide at least a 2 mm thick aluminium shell between the electronics and the outer space. This gives protection against the total dose effect of radiation and meteoroids described above.

Four means to minimise temperature extremes and its detrimental effects are used:

- The sensors on top of the satellite are shielded with thermal blankets.
- All internal heat sources are connected to the main satellite structure via internal conduction paths.
- The satellite slowly rolls about an axis normal to the sun.
- A special carbon fibre based material, with a very low coefficient of thermal expansion, is used for the solar panels. These components need to be exposed to incoming radiation and will therefore experience the largest temperature variations.

Horizon sensor housing design

Introduction

The horizon sensors are situated on top of the satellite (Figure 2) and determine the attitude of the satellite body relative to the Earth. They consist, from an optical viewpoint, of two identical units mounted perpendicular to each other. Each contains a linear CCD image sensor connected to a printed circuit board, and a lens that focuses on the Earth's horizon.

Particular functional design requirements

- The housing must fit in the given space on the top of the satellite. That means, that the height must be less than 80 mm, and both lateral dimensions less than 100 mm.
- The housing must position the two lenses with their focal paths at 90° to each other, and both must look down by 25° without any obstructions in the optical path.
- The lenses must focus within 0.3 mm of the plane of the CCD imager sensor.
- The housing must weigh less than 100 gram.
- No light must enter the housing other than through the lens.

Previous design

In an initial design [11] the charged coupled device (CCD) sensor, the associated printed circuit board (PCB), and the lenses were mounted on two different parts. Figure 4 shows the exploded view of the sensor mounting and the complete assembly. The CCD sensor was clamped into a close fitting recess of one mounting, that held it firmly in position and perpendicular to the lenses. The printed circuit board was fixed to the back of the same mounting by 4 bolts. The lenses screwed into a collar that could be screwed in or out of the other mounting to adjust the focus. The two mountings slotted together and left a shoulder, that gave the assembly the required angle when clamped onto the walls of a box like housing.

Redesign

The assembly discussed above turned out to be too difficult to manufacture, weighed too much, was too large, and the single parts were difficult to assemble. The sensor housing was therefore redesigned, to optimise the functionality, size, manufacturability, and assembly requirements. Figure 5 shows the redesigned horizon sensor housings without lenses and electronics, mounted on the top of the satellite.

The mountings of the lenses and the CCD sensor were machined out of one piece. The distance from the lenses to the focus plane could therefore be more accurately controlled. The machining tolerance was thus within the tolerance of the focus plane distance, and the previously used collar could be omitted.

Reducing the part count automatically helped to avoid assembly problems, simply because there are fewer assembly operations.

Two housings had to be made in the new design, versus one in the original. This leads, at first, to an increase in the part numbers, but this is well compensated for by the reduction of space used and easier manufacturing. Each horizon sensor's complete housing was milled out of one solid block, using a conventional vertical spindle milling machine. The PCB is totally enclosed by the housing in order to shelter it against radiation. The hollow foot enables the cables to run through it and provides a good heat conduction path. A lid closes the opening through which the PCB and the CCD sensor are mounted. It will be press fitted into the housing and glued on for the final assembly.

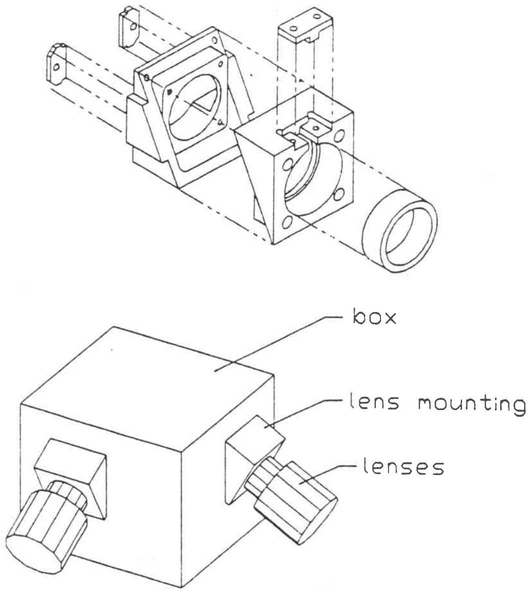

Figure 4 Previous horizon sensor housing design [11]

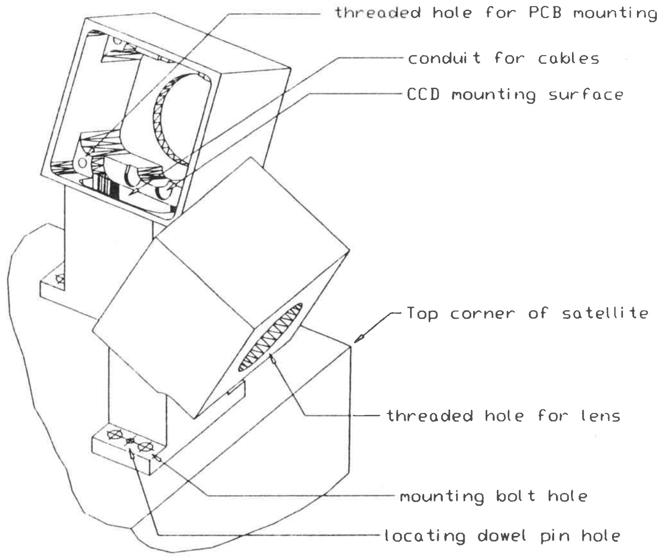

Figure 5 Improved horizon sensor housing

Dowel pins through the top plate and each housing foot align the two housings relative to each other, as well as to the satellite body. The holes for the pins in the housing and in the top plate were machined by an NC machine in order to obtain close tolerances.

The solid modeller from AutoCAD AME [10] was used as a design tool. The total mass could therefore easily be calculated and controlled. The 3-dimensional drawing and the hidden line algorithm were already used to check the appearance of the housing at the design stage. One complete sensor including electronics and optics weighs 215 grams, of which the aluminium housing constitutes 40%.

Imager bearing housings design and manufacturing

Description of the imager

SUNSAT's push broom imager is a single tubular assembly. A detailed functional description of the SUNSAT imager itself is given by Du Plessis & Milne.[12] The imager will be mounted diagonally across the lower tray of the satellite and stub shafts on both ends allow a rotation around its axis. This movement will permit stereo imaging. It is necessary to change the viewing angles as quickly as possible to minimise loss of earth coverage during the transition from one angle to the other. This is the reason for rotating the tube rather than the satellite.

The main tube serves as a basic structure to which the other components are attached. To accommodate the differential thermal expansion of the tube and the base plate, one side of the tube is axially located in its bearing mounting, while the other side is free to move axially.

Selection of the bearings

In this application, with the low number of shaft revolutions, a commercial dry bush bearing could be considered. Dry bearings provide a large contact surface, which is favourable for a shock and vibration environment. The chosen DU bearing from Glacier has the following advantages.[13] DU is a composite material in which a porous bronze sinter is bonded to a steel backing and impregnated and lined with PTFE. It is designed to operate without lubrication at temperatures between −200°C and 280°C, and can withstand loads up to 250 MPa.

The steel backing provides the underlying strength, while the bronze inter-layer provides both a strong mechanical bond for the PTFE lining and dimensional stability. It also improves thermal conductivity thereby promoting heat dissipation from the bearing surfaces.

Design of the bearing housings

The bearing housings hold the tube in position on the base plate while allowing the tube to rotate about its axis and expand in the axial direction. The design is aimed at minimum weight, high rigidity and ease of manufacturing. A rib structure with through-holes was therefore chosen. Figure 6 shows the two bearing housings.

The ribs are only $3-4$ mm thick. To provide sufficient stiffness in the lateral directions, the ribs were placed as far apart as possible, and made as wide as could be accommodated. The vertical middle rib transmits loads from the bearing to the middle of a web in the base plate.

The manufacturability of the rib structure is improved by the use of through-holes. The milling can thus be done from one direction and in one fixture. If a numerically controlled machine is not used, a turntable is needed for milling the ribs that lie at an angle.

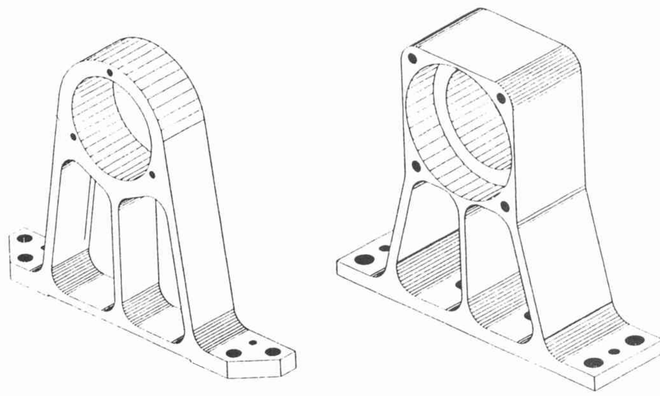

Figure 6 Imager bearing housings

Conclusions

The requirements have been described for the design and manufacturing of a low cost micro satellite, including the financial and technical constraints. The overall mechanical configuration, the tray structure, the horizon sensor housing and the imager mountings have been discussed. The parts are designed and manufactured within the allowable cost and time-scales, to satisfy all functional requirements.

Although SUNSAT is physically small, it is nevertheless complex and exhibits many of the characteristics of a large satellite. This makes it particularly suitable as a focus for educating and training students by providing a means for direct, hands-on experience of all stages and aspects (both technical and managerial) of a real satellite mission – from design, construction, manufacturing, test and launch through to orbital operation.[6]

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