INSAT-2A and 2B DEPLOYMENT MECHANISMS

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ABSTRACT

The Indian National Satellite (INSAT) 2A and 2B have deployment mechanisms for deploying the solar array, two C/S band antenna reflectors and a coilable lattice boom with sail. The mechanisms have worked flawlessly on both satellites. The configuration details, precautions taken during the design phase, the test philosophy, and some of the critical analysis activities are discussed.

1.0 INTRODUCTION

The INSAT-2A and 2B are the first two indigenously built operational communication satellites. Both satellites are identical in their configuration and include mechanisms for deployment of a solar array, two C/S band antenna reflectors, and a coilable lattice boom with solar sail. Figure 1 shows the satellite with deployed appendages. All the mechanisms have functioned flawlessly on both INSAT-2A and INSAT-2B Spacecraft. All the deployment indications were seen unambiguously.

This article describes some of the special features of these mechanisms, precautions taken during design phase, the test philosophy, and the analyses that are behind the consecutive total successes. Some of the details which are common to all the mechanisms are highlighted below.

- Use of pyrocutters with simple designs, adequate margins, and mechanical and electrical redundancies.
- Minimizing the number of deployment phases in each mechanism and using simple configurations.
- Use of simple designs for the hold-down and release mechanisms.
- Provision of spring-actuated pushers at all separation planes to ensure a positive release and first motion.
- Provision of compensation features at hold-down interfaces/close control loops (CCL) and incorporation of flexibilities in hold-down bolts to account for differential thermal expansions.
- Meticulous and elaborate planning and implementation of the test and evaluation plan for each of the mechanisms at component level,

sub-assembly level, and system level, and establishment of dedicated test facilities.

- One-hundred percent participation by independent quality assurance teams.

2.0 SOLAR ARRAY DEPLOYMENT MECHANISM

2.1 CONFIGURATION

The solar array consists of a yoke, three large panels of 1.8m x 2.15m, and two small panels of 1.073m x 1.8m. The two small panels are stowed at the back side of the first large panel and are held down by a secondary hold-down loop. The yoke, first large panel with two stacked small panels, and the other two large panels are stowed on the spacecraft deck using six hold -own assemblies interconnected as shown in Figure 2. Figure 3 shows the array deployment in two stages, namely primary deployment and secondary deployment. Primary deployment consists of deployment of the yoke and three large panels, and secondary deployment consists of deployment of the two small panels. Three distinct advantages of this solar array configuration are:

- No need for partial deployment during transfer orbit by proper sizing of array. Transfer orbit power is obtained by orienting the south side of the Spacecraft to sun.
- 75% of power is available on deployment of large panel and the array is steerable after the first stage of deployment.
- Primary deployment is of accordion type which reduces the shock load considerably.

In any deployment mechanism configuration selection, the number of deployments should be kept at a minimum as this results in the reduction of pyrocutters. The availability of 75% of power at the end of the first stage of deployment itself is a positive aspect from a mission point of view. The choice of accordion type of deployment is preferred. The shock at each joint is minimized because the energy gets countered due to the change in direction of rotation between successive panels .

2.2 HOLD-DOWN BLOCK

Figure 4 shows a typical hold-down assembly. A flexible wire rope is used instead of a rigid rod used in most hold-downs [1]. The flexibility in hold down allows for minor misalignment due to assembly as well as thermal distortions and ensures positive release. Adding to the wire rope flexibility in the hold-down bolt, hinging has been included for smooth withdrawal and release of the long hold-down bolt.

In addition, in each of the hold-down base assemblies, a spring is provided to ensure the release of the hold-down lever/plunger elements immediately after cutting the hold down loop, even though the reaction forces in the hold down are enough for this release.

Prior to the deployment in the geostationary orbit, the array stack has to withstand the thermal loads expected during the transfer orbits. These loads can distort the panels and can cause hindrance to the deployment. To prevent the building up of thermal loads, a thermal slip provision is made in the hold-down block at the interface of panels, the details of which are shown in Figure 5.

The in-plane loads on the panels expected during launch do not exceed the friction loads acting at various interfaces. For generating the required frictional resistance at the outer-most panel hold-down block, at the next panel serrations at the interface, and at the first panel level, a grooved configuration has been used. Thus a graded friction has been adopted in the design.

At each of the interfaces between panels, spring-actuated pushers have been used to give first motion to the panels even though the springs at the hinges have enough margin over the frictional torque. These pusher springs are located away from the hinges, thus producing a large torque at the start of the motion for a small angular movement. However, this does not increase the deployment energy considerably and the value is about 3% of the deployment energy.

2.3 CLOSE CONTROL LOOPS (CCLs)

The CCLs are used to coordinate the deployment direction. Figure 6 shows a typical CCL. Each CCL consists of a preloaded wire rope loop passing over two pulleys mounted at the hinges. This CCL has the feature that the turn buckle and spring are combined. A compression spring is used instead of a tension spring to make the assembly compact. The loop has two springs, one on each side with a provision to adjust the preload. The temperature differentials expected in the orbit change the length of the wire rope. This change is absorbed by springs. The springs are also designed to maintain the preload in the loop well within the specified value. Thus it is ensured that the coordinated control is not affected.

2.4 SNUBBERS

The yoke is triangular in shape and supports two shunt regulators. This yoke is supported at three hinge points. The two-meter span beam of yoke has a low frequency, if unsupported. This frequency is increased by using two preloaded snubbers. The snubbers are made of space-qualified silicone rubber. This design eliminates the need for a separate yoke hold down. Similar snubbers have been used to support the two small panels. This design has been successfully implemented to limit and damp the vibration amplitudes. The design has been validated through qualification tests at spacecraft level and its successful on-orbit performance.

2.5 SMALL PANEL HOLD-DOWN SYSTEM

A hold-down system shown in Figure 7 is a simple restraint mechanism without any rigid clamping. This is a unique, compact and simple design adopted in the system for small panel hold down and release system.

3.0 C/S BAND ANTENNA REFLECTORS DEPLOYMENT MECHANISM

INSAT-2 has two C/S-reflectors of size 1.772 m x 1.772 m each. These reflectors are stowed parallel to the East and West faces of the satellite deck and when deployed through 73.61°, the characteristic value of the paraboloid, they will have a northward tilt of 3.77° corresponding to the beam center of 22° N latitude. Figure 8 shows the stowed and deployed configurations.

3.1 HINGE LINE DEFINITION

The accuracies required on deployment of the reflectors were of the order of +0.02 deg over the above-mentioned angles of 73.61 deg and 3.77 deg. To accommodate the reflectors within the specified envelope in the stowed condition, the edges of the reflector must be kept parallel to the satellite faces. At the same time, in the deployed configuration, a 3.77 degree northward tilt was required at the end of deployment. This complex requirement was met by an accurate definition of the hinge line.

The stowed and deployed coordinates were considered. Intersection of spheres with appropriate solid geometry relations has been used for finding the hinge line. This line was further checked by using rotation transformation matrices to ensure that the stowed coordinates when rotated about the defined hinge line would give the required deployed coordinates of the reflector.

The defined hinge line had a tilt about two axes. Designing the hardware to meet this requirement and subsequent fabrication and inspection operations have been very challenging. A typical hinge is shown in Figure 9 with associated locking linkage and flexure. The double-tilt bracket seen in the figure was fabricated using CNC milling with appropriate programs. The inspection of this complex component has been carried out using a 3D measuring machine. The hinge line defined by analysis has been implemented in the hardware and the pointing accuracies realized on assembly have been checked by using optical theodolites, autocollimation prisms, and associated accessories.

3.2 HOLD-DOWN MECHANISM

The C/S-reflector and solar array hold down and release mechanism concepts are similar. They include a provision for thermal slip at hold down and spring-actuated pushers at separation planes. The two hold downs used in this system are interconnected with a straight wire rope and a single cutter, unlike multiple explosive bolts used in other satellites, thus increasing the reliability of the system.

3.3 FLEXURE

Flexures have been used in the hinge outboard bracket to the CFRP antenna interface to take care of the effects of thermal differentials. These elements have been designed to have a low stiffness along the CFRP rib direction and high stiffness in the deployment direction to withstand the latch-up moment. A typical flexure can be seen in Figure 9.

3.4 LOCKING LINKAGE

Figure 9 also shows the locking linkage position in the hinge assembly. These linkages ensure a precise and positive locking for the reflector when it deploys through a predetermined angle of 73.61 deg. Based on range tests, if any change in this angle is required, a provision exists in this mechanism for fine tuning the opening angle by +0.5 deg from the nominal orientation. These linkages have been designed to take tensile load at latch-up, unlike the compression mode in designs used in other spacecraft.

4.0 SOLAR SAIL/BOOM

The coilable lattice boom with a conical-shaped sail balances the solar radiation torque acting on the solar array. The deployable boom is 14.95 m long and 0.26 m diameter. The solar sail at the end of the boom is 1.5 m diameter at the bottom, 0.79 m diameter at top, and 4.4 m long. Figure 10 shows the stowed and deployed configurations of the coilable lattice boom with sail. Stowed sail and boom are held down to the north panel using a launch restraint assembly and a preloaded tie rod. The boom in its stowed condition is housed inside a very compact canister, with the stowed height of the boom being 2% of its deployed length. The boom has self deploying capability but to control the rate of deployment, a lanyard type of deployment mechanism is used along with a drive motor with worm gear speed reducer. A pyro bolt cutter is used for cutting the tie rod and releasing the hold down for on-orbit deployment. Six microswitches are used for monitoring the performance of the boom during deployment.

During the fabrication of boom, elaborate tooling and fixtures have been developed to ensure the boom geometry is well within the desired limits and the axis of the boom is maintained within +0.3 deg consistently for all models.

4.1 HINGE

The boom uses hinges with two degrees of freedom to connect the longerons with battens. The diagonals are connected to these hinges through spherical terminals as shown in Figure 11. These hinges are dry lubricated with MoS_2 on all the bearing surfaces to minimize friction and ensure a smooth deployment. The hinge parts have been configured for ease of assembly and disassembly for replacements, if required.

4.2 FIRST-MOTION SPRING ASSEMBLY

The characteristic of this type of boom is that the self deployment force at the start of deployment is low if both ends of the boom/longerons are stowed flat. Also, the friction at the end hinge assemblies is high. To overcome these problems and to aid the deployment of the boom in the initial phase, a wedge support with an 8° taper and a spring-actuated first-motion spring assembly are incorporated below each of the longeron end fittings at the base end. These features give a force of 7 kg over an initial plunger movement of 10 mm. A typical kick-off plunger assembly is shown in Figure 12. This design ensures base-end deployment, which is an essential feature for a trouble-free deployment.

4.3 LANYARD SPOOL ASSEMBLY

The boom with sail is released at a controlled rate using a lanyard. One end of the lanyard is attached to the tip plate of the boom with the other end wound on a spool that is driven by a DC motor through a worm gear speed reducer to preclude the possibility of the boom driving the motor. The lanyard is attached to the spool by an end hook that automatically gets released from the spool in the event of failure of the motor auto off feature at the end of deployment. This feature avoids the backwinding of the lanyard on spool.

4.4 AUTO-MOTOR-OFF SWITCH ASSEMBLY

Figure 13 shows this actuator. A spring-loaded lever dips into a recess provided in the lanyard spool soon after full deployment of the boom and in turn actuates two microswitches that cut off power to the DC motor. The design is such that the lever will not interfere in the rotation of the spool even when there is no lanyard on the spool.

4.5 GROUNDING TECHNIQUE OF SAIL

To minimize the build up of static charges on the large area solar sail surface, use of aluminized Kapton film with a conductive coating on the Kapton side and grounding it would have been a simple option. However, considering the prohibitive cost of this material, a special grounding technique has been developed and qualified. This technique involves the use of standard aluminized kapton film with conductive tabs at both top-mid and mid-bottom cone interfaces. This has resulted in considerable saving in cost. All the joints have undergone extensive static charge testing and qualified for the expected on-orbit conditions.

5.0 PYROCUTTERS

Pyro wire rope cutters are one of the critical elements for the successful functioning of the mechanisms. The pyrocutters used in the solar array and C/S band antenna were qualified earlier during the development of mechanisms for Indian Remote Sensing Satellites. The bolt cutter used for boom mechanism was developed during the INSAT-2 project. All pyrocutters have both electrical and mechanical redundancies with adequate margin of safety.

6.0 ANALYSIS

The analysis activities carried out for each of the above systems are discussed in brief. The deployment dynamics of the INSAT-2A and 2B Solar Array and C/S band antennae have been carried out in detail for both ground and on-orbit conditions. However, in case of 2A, the predicted deployment time did not match with the on-orbit deployment time. Hence a post launch analysis has been carried out using the high-speed camera data analysis obtained during ground tests of INSAT-2B. The updated initial velocity values were used for predicting the deployment times of the INSAT-2B primary array deployment, secondary array deployment, and the C/S band antennae. The predicted values are in close agreement with the on-orbit deployment time. The post-launch analysis is discussed in reference [2]. The mismatch between initial prediction and 2A on-orbit values has been assessed to be due to initial velocities imparted to the system by the snubbers and spring-actuated pushers ,which give a small amount of energy into the system for a few milliseconds.

The hinge line definition which was discussed in C/S band antennae is an important analysis that has been carried out. Here an application of solid geometry, intersection of spheres, angle between lines, planes, and rotation transformations have been used in accurately defining a hinge line. The intersection of spheres results in a set of nonlinear algebraic equations. After obtaining the solution, it was checked thoroughly for required angular accuracies. The process was repeated iteratively until the accuracies required were met.

A best-fit paraboloid analysis has been carried out in defining the vertex shift, focal length changes, focus shifts, and corresponding tilts. A least-square fit was used. This is discussed in detail in reference [3].

One of the most fascinating analysis was the elasto-plastic analysis of the lanyard. The lanyard experiences a shock load from the release of the stowed energy of the boom, preloaded tie rod, tip plate, and launch restraint rods when the tie rod is cut. This energy was found to be greater than the elastic energy carrying capability of the lanyard. Consequently, the lanyard was found to yield. So an elasto-plastic analysis with a cumulative damage study was conducted. The number of cycles the lanyard could withstand before failure was found. Based on this analysis, the maximum number of allowable tests on the flight model lanyard was defined and implemented. This is discussed in detail in reference [4].

The shock analysis for the primary deployment, secondary deployment and C/S reflector has been carried out. This provides the basic input for the design of hinges.

The boom free vibration and thermal distortion study has been carried out. The deflection of the boom with sail from its nominal direction due to thermal differentials, superimposed with acceleration loads acting during controlling of satellite, has been found. The study has been carried out to ensure the sail middle cone does not come within the field of view of the VHRR cooler which is very sensitive to heat radiation.

7.0 TESTING

To ensure successful working of these mechanisms, a detailed test matrix and associated test plan was generated for all the critical components, subassemblies and assemblies. These were meticulously planned and implemented. A few of them are listed below.

- Strength and stiffness measurements
- Calibration of springs
- Characterization of harness loops
- Destructive and non-destructive testing of various boom elements
- Coupon testing of adhesive joints.
- Friction measurements.
- Alignment using autocollimation
- Fine-motion study using high-speed camera.
- Non-contact distance measurements using ECDS (Electronic Coordinate Determination System)
- Angular error measurement and correction.

Further, for the testing of the mechanisms, a few sophisticated or dedicated facilities have been established. These include:

- Zero-"g" fixture for solar array deployment.
- High-bay test facility for vertical deployment of boom with sail
- Water-trough facility for horizontal deployment of the boom.
- Electronic Coordinate Determination System for alignment and non-contact distance measurements.
- High-speed camera for measuring fast motions such as hold-down release.
- Air-bearing facility for Zero-"g" tests on C/S antenna reflectors.

One of the important tests used in the qualification of the Coilable Lattice Boom (CLB) was a stress rupture study of longeron. Stress rupture (static fatigue or delayed failure) is the failure under sustained loads over a long period of time. Stress rupture of glass fiber composites is controlled by surface defects of fiber, matrix failure due to visco-elastic deformation, etc. In an application like CLB of INSAT-2A/2B, it may become necessary to store the boom in a stowed condition for a long period due to various reasons during fabrication, testing and prelaunch phases. Typically, a storage life of about five years is specified under a flexural strain of 1.1% or a stress of 60 - 65 Kgf/mm^2. On the continuous longerons of the CLB, stress rupture data based on a 15-year study as a function of sustained stress versus life under tensile loading on composites is available in the literature. However, the type of loading in our application is flexural. Stress rupture behavior on longeron elements at 2% strain level has been verified by coiling on a mandrel of suitable size on the INSAT-2A boom structural model and storing for more than 4 years without any failure of longerons.

The two small panels are stowed at the back of the first panel. The pyrocutter used for cutting the hold-down cable is mounted in the back side of the solar cells on the first panel. To ensure that the solar cells on the first panel are able to withstand the shock due to pyro, a few tests were conducted. Acceleration levels were measured and the solar cells mounted on first panel were found to be intact. With these tests, the use of a pyrocutter mounted on the first panel was cleared.

8.0 MISSION

Both the solar array and antenna are made of CFRP. As these panels will be facing sun before deployment, the temperature of the array can go beyond 70 deg C, the qualification temperature of the hinge interfaces. Hence, reorientation of the satellite is necessary to bring down the temperature of the array below 70 deg C. This is done to ensure that the hinge interface loads at latch-up are well within the limits to which the hardware was qualified. To minimize the thermal differential within the CCL wire rope, which in turn can produce change in tension of CCLs wire rope and consequently an increase in friction torque, a small tilt was given in the satellite. The tilt angles are 60 deg in the Roll-Pitch plane away from the sun to bring down the temperature, and 6 deg from the Roll-Pitch plane towards the earth-viewing face to avoid thermal differential within CCLs.

To facilitate monitoring deployments, an adequate number of microswitches have been used. In the solar array, and C/S band antennae, microswitches have been used for monitoring the cutting of wire rope, system first motion, and locking of hinges. The coilable lattice boom cutting of bolt, initial motion, motor-release function, and sail deployment have been monitored through microswitches. In case of any anomaly, sufficient data can be obtained through these indications for further analysis.

As can be expected, a mechanism would work better if temperatures close to laboratory conditions are created in space. This philosophy was adopted in the C/S band antennae deployment. The East reflector was deployed with the east face of the satellite facing sun, and the West reflector was deployed with the west face facing sun. With this, the temperatures of the hinges were close to 20 deg C. The solar sail boom was deployed when the temperature of the motor was around 20 deg C. This was adopted in both INSAT-2A/2B. All mechanism deployments were smooth and all indications were obtained unambiguously.

9.0 CONCLUSIONS

The configuration of the deployment mechanisms used in INSAT-2A and 2B has been discussed. Some of the design features are discussed. The thermal compensation features and flexibility in hold down have been discussed showing how thermal differentials have been taken care in the design. The spring-actuated pushers give a large torque acting for a few milliseconds in the initial phase of deployment. Detailed analyses carried out to support the design and testing phases of the mechanisms have been brought out. The meticulously planned testing at various levels and development of dedicated test facilities has been highlighted. Wherever possible the mission sequence has been finalized so as to ensure that the temperature of the hinges is around room temperature for smoother performance of mechanisms.

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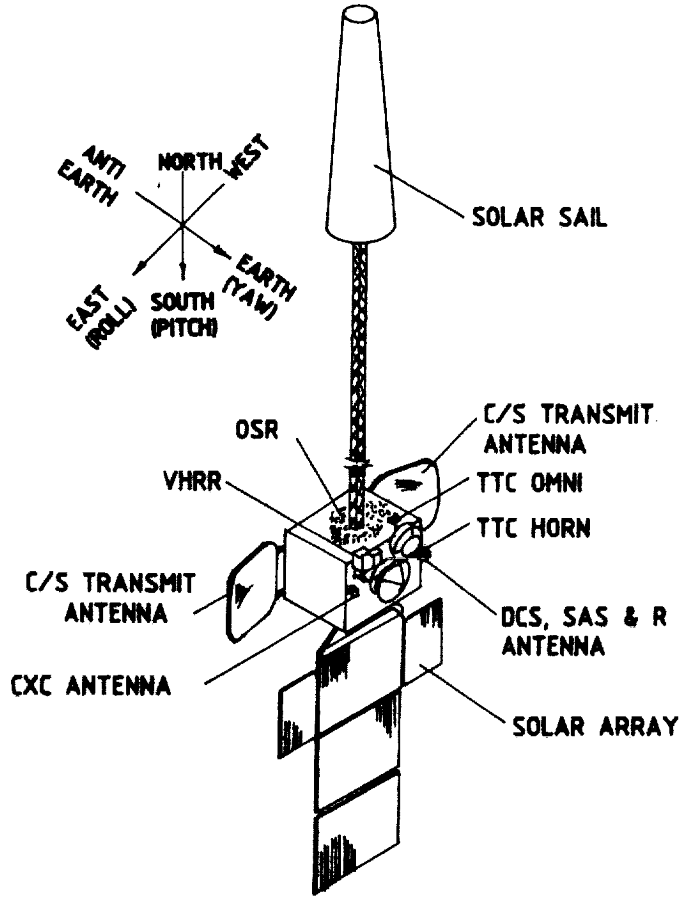


FIG.1 ONORBIT CONFIGURATION-INSAT-2 SPACECRAFT

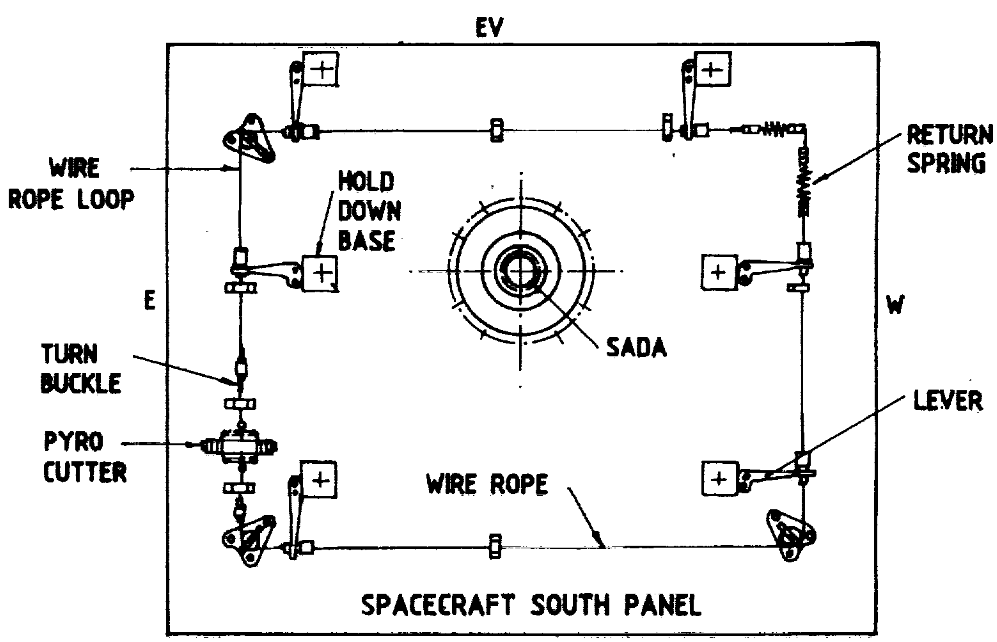


FIG.2 PRIMARY HOLD DOWN LOOP

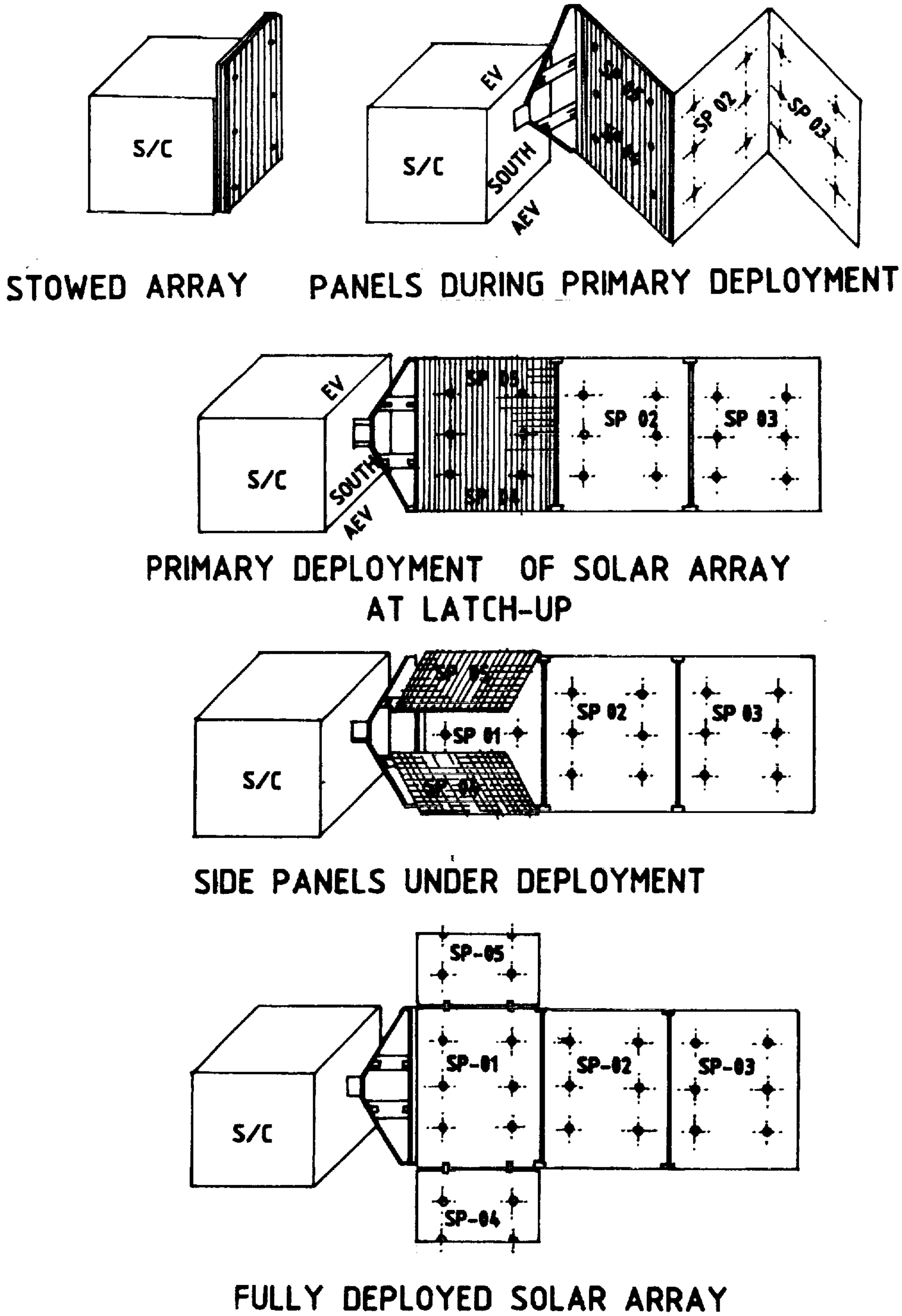


FIG.3 SOLAR ARRAY DEPLOYMENT SEQUENCE

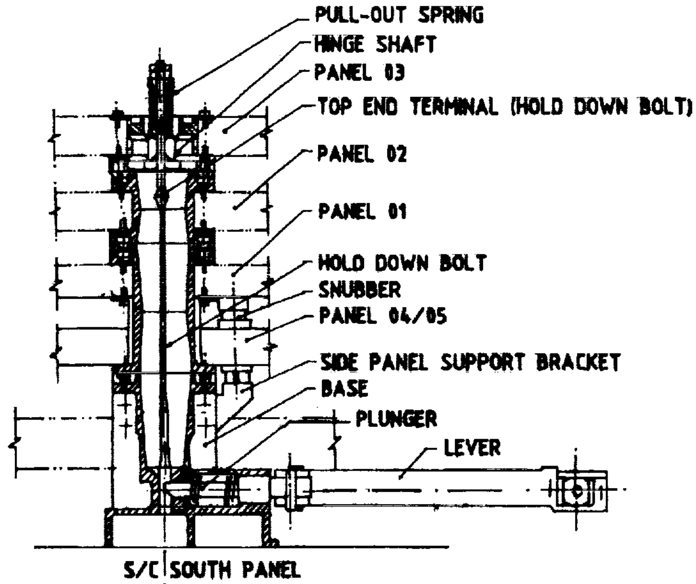
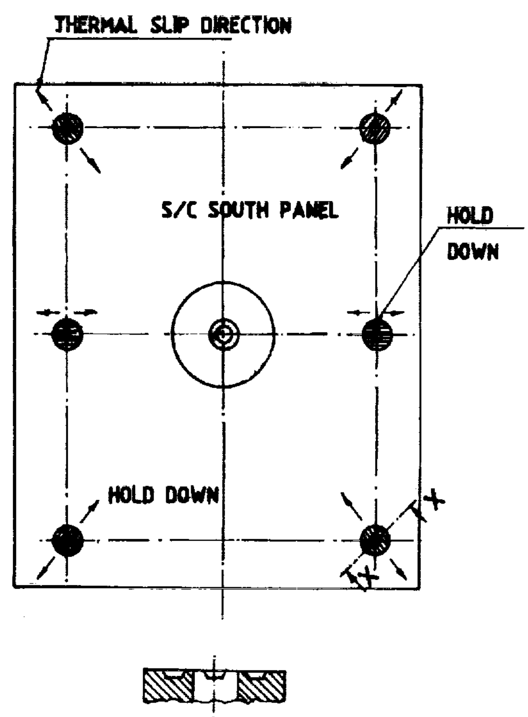


FIG.4 HOLD DOWN ASSEMBLY DETAILS



SECTION-XX
SPIGOT DETAIL

FIG.5 SOLAR PANEL STACK SPIGOT
LOCATIONS FOR THERMAL SLIP

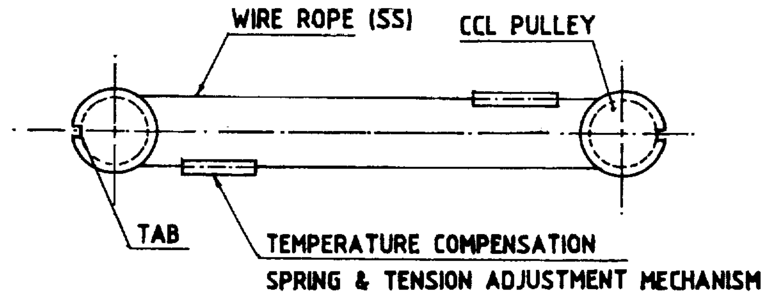


FIG.6 CLOSE CONTROL LOOP

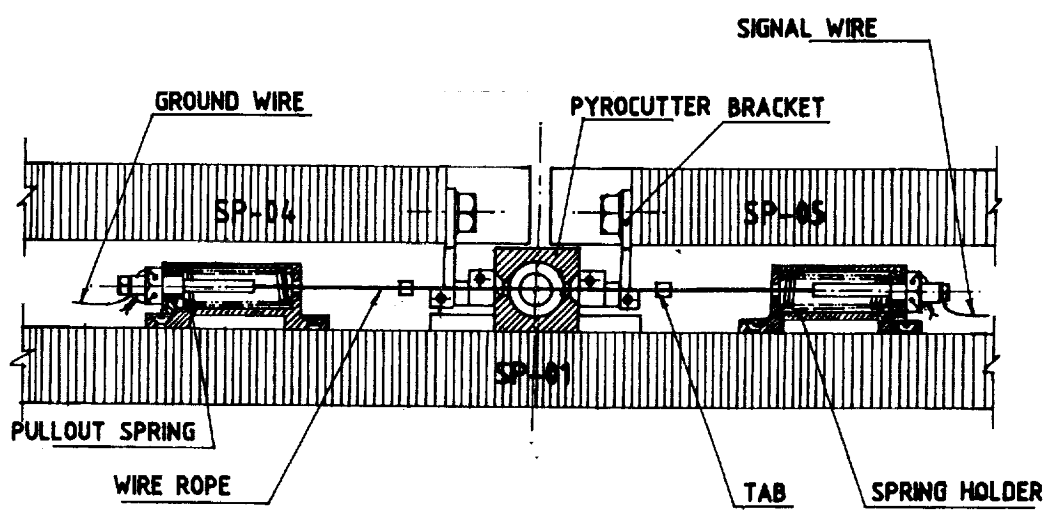


FIG.7 HOLD DOWN & RELEASE SYSTEM FOR HALF PANELS

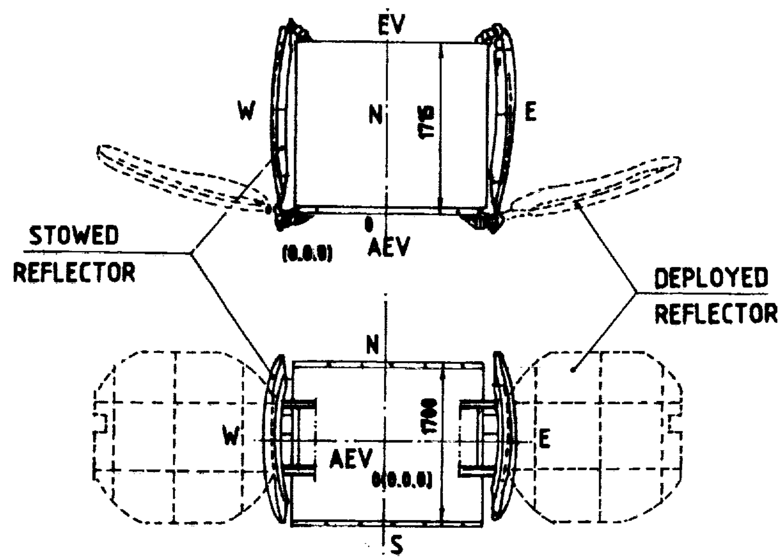


FIG.8 STOWED AND DEPLOYED CONFIGURATION OF C/S-REFLECTOR

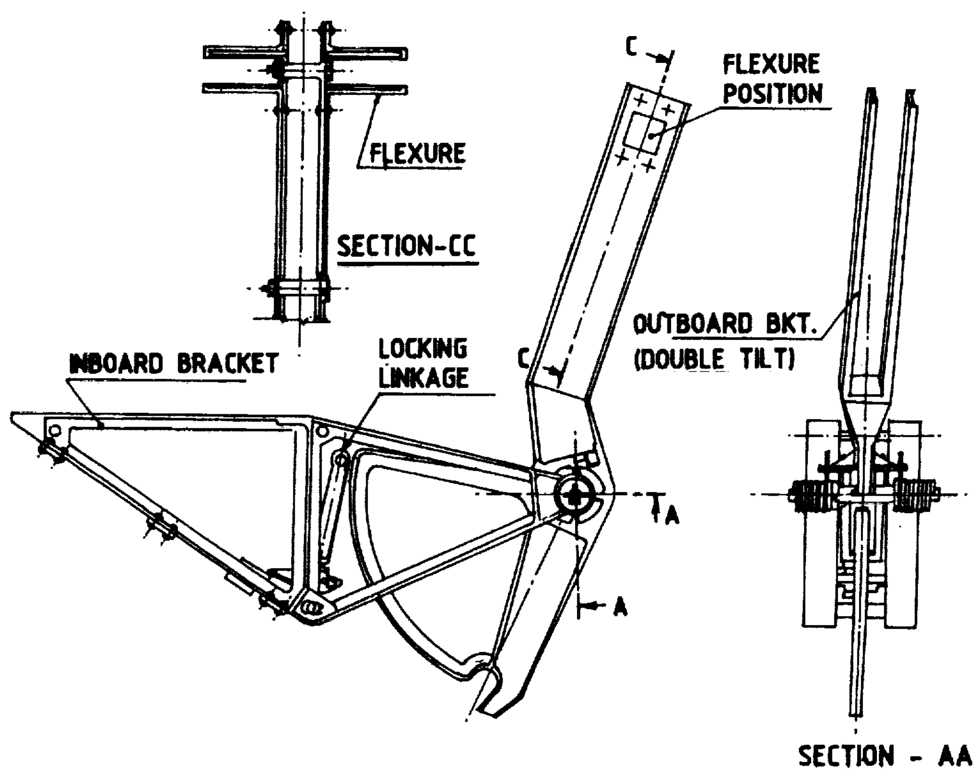


FIG.9 TYPICAL HINGE ASSEMBLY DETAILS

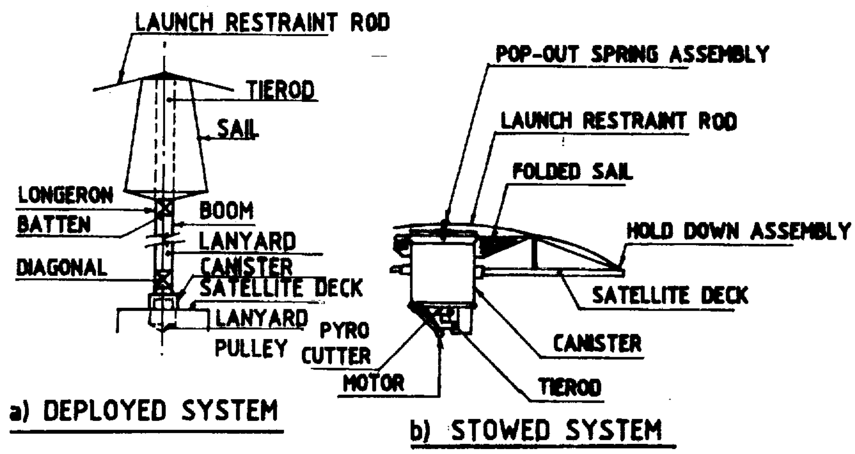


FIG.10 STOWED AND DEPLOYED CONFIGURATION OF SAIL BOOM

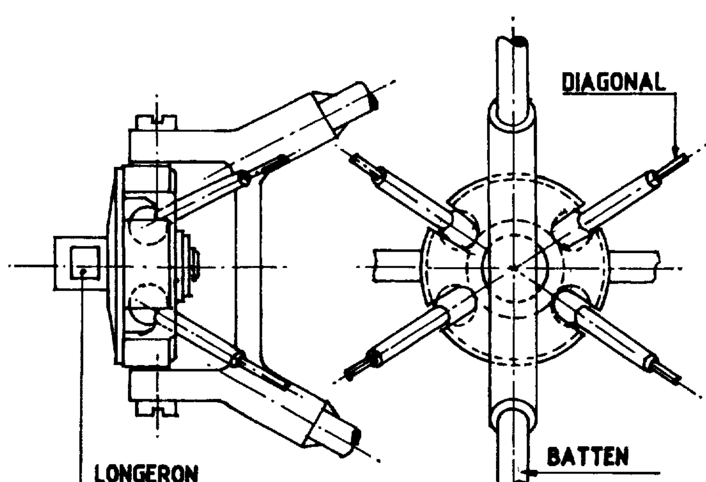


FIG.11 TWO DEGREE OF FREEDOM HINGE ASSEMBLY

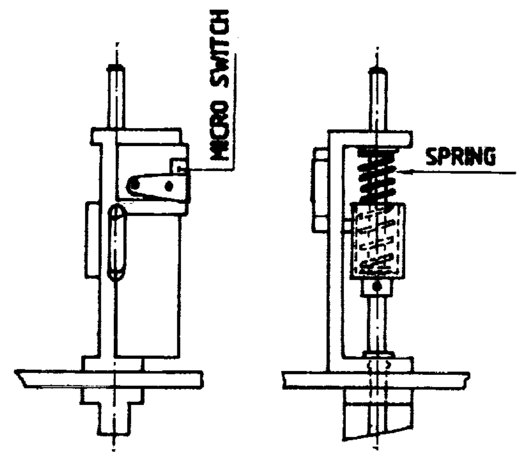


FIG.12 FIRST MOTION SPRING ASSEMBLY

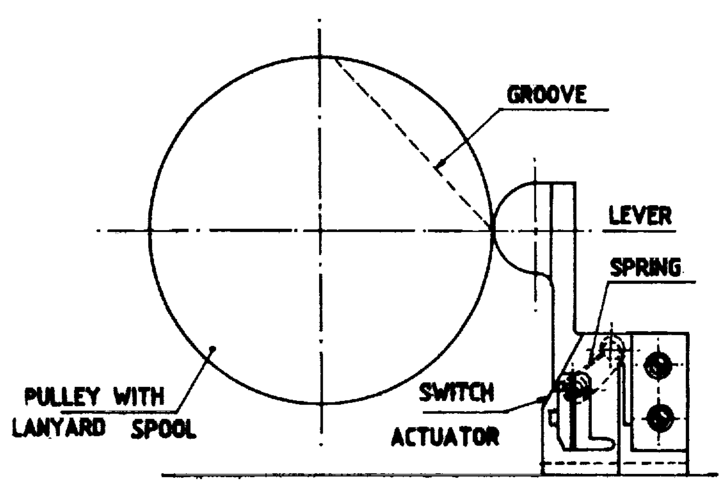


FIG.13 AUTO MOTOR OFF SWITCH ASSEMBLY