Advanced features and specifications of IRS-1C spacecraft

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Here we report the advanced features and specifications of the IRS-1C spacecraft. The spacecraft consists of three payloads, viz. a panchromatic camera, a four-band multispectral camera and a two-band wide field camera. The mainframe consists of a structure, thermal, mechanisms, power, communication and data handling, attitude and orbit control systems. Many new developments, including the payloads, have taken place for realizing the IRS-1C spacecraft.

THE IRS-1C mission envisaged the realization of a satellite-based remote sensing system to provide continuity of services to the users and also to enhance the potential applications compared to IRS-1A/1B. The principal component of the IRS-1C system is the spacecraft carrying the three optical cameras onboard. The specifications of the cameras have been arrived at on the basis of the requirements from the users.

The spacecraft is three-axis stabilized and is derived out of the heritage of IRS-1A/1B but includes many new features to meet the enhanced mission requirements. The mission requirements for the space segment are:

A panchromatic camera (PAN), providing 5.8 m resolution and swath of 70.5 km with off-nadir steering capability up to ± 26° to provide stereo imageries.

A multispectral camera, called Linear Imaging Self Scanner-III (LISS-III), providing 23.6 m resolution in three bands, viz. green, red, near IR and a resolution of 70.8 m in shortwave infrared band (SWIR) and a swath of 142 km in the first 3 bands and 148 km in the SWIR band.

A wide field camera (WiFS) working in two bands, viz. red and near infrared, providing a resolution of 188 m and a wide swath of 804 km.

To support the above payloads, the following mainframe elements are required:

- A structure to house the payloads and mainframe elements and to give mechanical and thermal stability during launch and orbital phases respectively.
- A thermal control system to provide the thermal control for all elements of the spacecraft as per the requirement of the subsystems.
- Mechanisms to deploy the solar panel and PAN camera on-orbit to enable rotation/steering of these systems on-orbit.
- A power system to support the functioning of all the systems during the sunlit and eclipse phases of the orbit.
- A control system to provide the three-axis stabilization to the required pointing accuracies using the reaction wheel for each axis within the acceptable drift rates and jitter. Earth, sun and inertial sensors (gyros) are required for sensing the attitude errors about all the three axes. The control system should also be able to provide orbit manoeuvre capability using thrusters for adjusting the orbit to remove the initial orbit dispersions arising due to the launch and also the orbit disturbances caused by the orbit perturbing forces from time to time.
- A star sensor for accurate post facto attitude determination.
- A communication and data handling system to give the telemetry, telecommand, tracking (TTC) functions of the spacecraft and also for the payload data handling and transmission to the data-receiving stations.
- An onboard tape recorder for recording the payload data over non visibility regions for subsequent playback over the ground station.
- To provide a spacecraft reliability of 0.75 over three years of mission life.

To support all the above functions, a sun synchronous orbit of 817 km altitude has been choosen with 10.30 AM as local time of equatorial crossing at descending node. Sun synchronous orbit ensures constant illumination of the scenes at a given latitude. The sun synchronous orbit at 817 km altitude has been chosen to provide the required ground station area coverage for the data, the required sun angle for imaging from applications point of view and to take care of the other constraints arising out of the satellite operations pertaining to the frequency of the orbit manoeuvre for maintaining the specified orbit.

System features and specifications

Payloads

All the three payloads work in the push broom scanning principle, wherein the required swath is provided by the number of pixel elements in the charge coupled devices (CCD) being used as detectors and the along track coverage is provided automatically as the CCD scans along track using the satellite motion. The specifications of all the cameras are given in Table 1.

Panchromatic camera (PAN): PAN camera is a high resolution camera working in the panchromatic band (0.5–0.75 μm) and uses a reflective off axis optics using three mirrors to realize the required focal length to achieve 5.8 m resolution.

The reflective optics consists of three-mirrors with a total field of view of ±2.47°. A three-mirror reflective optics has been chosen since a refractive optics will be too voluminous to get a focal length of 980 mm. The mirrors are polished to a surface accuracy of better than lambda/10 to provide the required modulation transfer function (MTF) at the Nyquist frequency of spatial sampling. The camera uses three 4000 element charge coupled devices as detectors which are mounted in a staggered fashion to provide a total swath of 70.5 km and an instantaneous geometric field of view (of each pixel) of 5.8 m on ground. The mounting structure of the optics is made of invar material to ensure the specified optical performance even with thermal gradients across the optics.

The data from each CCD are separately processed in a processing electronics and all the three CCD data are merged subsequently in the data handling system before transmission to the ground. Also, a provision has been

Table 1. Spacecraft specifications

Mass of the spacecraft : 1246 kg

Payloads

	PAN	LISS-3		WiFS
		VNIR	SWIR	
Spatial resolution (m)	5.8 (Nadir)	23.6	70.8	188
Swath (km)	70.5	142	148	810
Spectral bands (μm)	0.5–0.75	B2 0.52–0.59 B3 0.62–0.68 B4 0.77–0.86	B5 1.55–1.70	B3 0.62–0.68 B4 0.77–0.86
No. of grey levels	64	128	128	128
Signal-to-noise ratio (at saturation)	> 64	> 128	> 128	> 128
Square wave response (at Nyquist)	> 0.20	B2> 0.40 B3> 0.40 B4> 0.35	> 0.30	B3> 0.34 B4> 0.20

Spacecraft platform

Power	Solar panel	:	720 W at end of life
	Battery	:	21 AH × 2 (Ni–Cd)
	Distribution	:	DC/DC convertors
TTC	Band	:	S-band
	Telemetry downlink	:	1 W power output
	Telecommand uplink	:	–110 dbm command sensitivity
	Tracking accuracy	:	10 m in range 10 cm/sec in range rate
Data handling	Chain 1	:	84 MBS for PAN
	Chain 2	:	42.5 MBS for (LISS-III + WiFS)
	Modulation	:	QPSK modulation
	RF power output	:	40 W (X-band)
AOCS attitude pointing accuracy		:	±0.15° about pitch and roll ±0.2° about yaw
	Attitude drift	:	Better than 3×10^{-4} deg/sec
	Jitter	:	Less than 1/10 of PAN pixel on ground
Attitude determination accuracy		:	±0.01° using star sensor
	Thrusters	:	1 Newton for attitude control 11 Newton for orbit control
	Fuel	:	Monopropellant (hydrazine) 84 kg
Reaction wheels angular momentum		:	5 NMS for each axis
	Torque	:	0.05 NM
Gyroscope stability		:	0.05°/h
Onboard tape recorder storage		:	62 GB/21 minutes of storage

kept to verify frequently the responsivity calibration of CCDs using LEDs placed appropriately in the optics for illuminating the CCDs. The block schematic of PAN camera is given in Figure 1.

Multispectral camera (LISS-III): The multispectral camera uses an eight element lens refractive optics as was used in LISS-I/LISS-II of IRS-1A/1B but with a focal length of 347 mm. The refractive optics is required for providing the specified field angle. The optics is individually implemented for each of the four bands and for band selection an interference filter is used in the front.

The mounting structure of the optics is made of invar for the reasons of thermal stability. Each band uses 6000 element linear silicon CCD array providing an instantaneous geometric field of view of 23.6 m for green, red, and near IR bands and a 2000 element Indium Gallium Arsenide CCD to provide 70.8 m for SWIR band. The SWIR band CCD is maintained at –10°C for giving the required signal to noise ratio. A swath of 142/148 km is provided for first three bands/SWIR band respectively. Each band CCD output is processed in a separate processing electronics before final merging during the data formatting at data handling system end. The block schematic of LISS-III is given in Figure 2.

Wide field sensor (WiFS): The WiFS camera uses an 8 element refractive optics like in LISS-III but with a focal length of 57 mm and a field of view of ± 13°. In order to realize the required wide field of ±26°, two such cameras are mounted with overlapping pixels of imaging. The cameras work in two bands, viz. red and near IR. The detectors used are 2000 element silicon CCDs providing instantaneous geometric field of view of 188 m on ground and a total swath of 804 km with two overlapping cameras. The WiFS data are processed separately but formatted along with the LISS-III chain data. The block schematic of WiFS camera is given in Figure 3.

Structure system

The structure system not only houses various elements of the spacecraft but is also able to withstand the launch loads and provide the required mechanical integrity along with alignment and thermal stability. The structure is derived from IRS-1A/1B with a central cylinder as the main load-bearing element but modified to meet the compatibility requirements with Russian Molniya launcher or Indian PSLV launcher. The main platform of the structure consists of a central cylinder attached to the bottom and the top decks which are connected to the four vertical decks on the four sides of the spacecraft. In addition, a thermally isolated (from the main platform) payload deck is introduced in IRS-1C to make for better thermal and alignment stabilities for

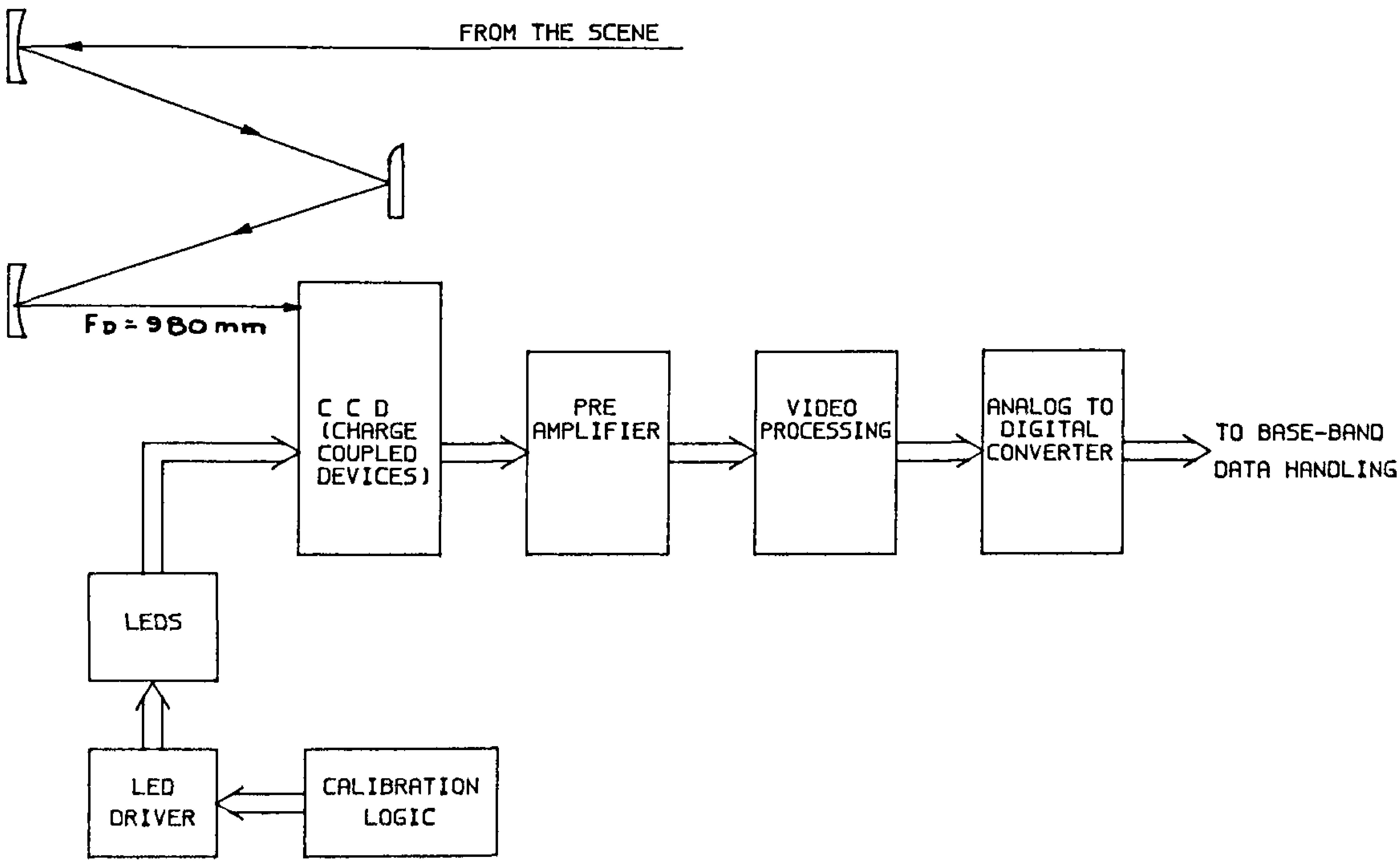

Figure 1. Block schematic of panchromatic camera.

the payload sensors. On the same payload deck where payload optics are mounted, attitude sensors are also mounted to provide acceptable attitude reference accuracies with respect to the payload pointing (optical) axis. Further stiffening of the IRS-1A structure using sheer-webs connecting the bottom and the top decks to increase the global longitudinal/lateral resonance frequency and the inclusion of a CFRP cylinder which thermally isolates the mainframe structure from the payload deck to enable the stringent thermal control requirements of the payload were the significant design modifications from IRS-1A. The total mass of the spacecraft is 1246 kg out of which the structure mass is 138 kg. Figures 4 and 5 show the assembled and disassembled view of the spacecraft.

Thermal system

The thermal system provides the thermal control for all the elements of the spacecraft, where all the electronics packages are controlled between 0 and 40°C. All payload optics are controlled between 17 and 23°C with a

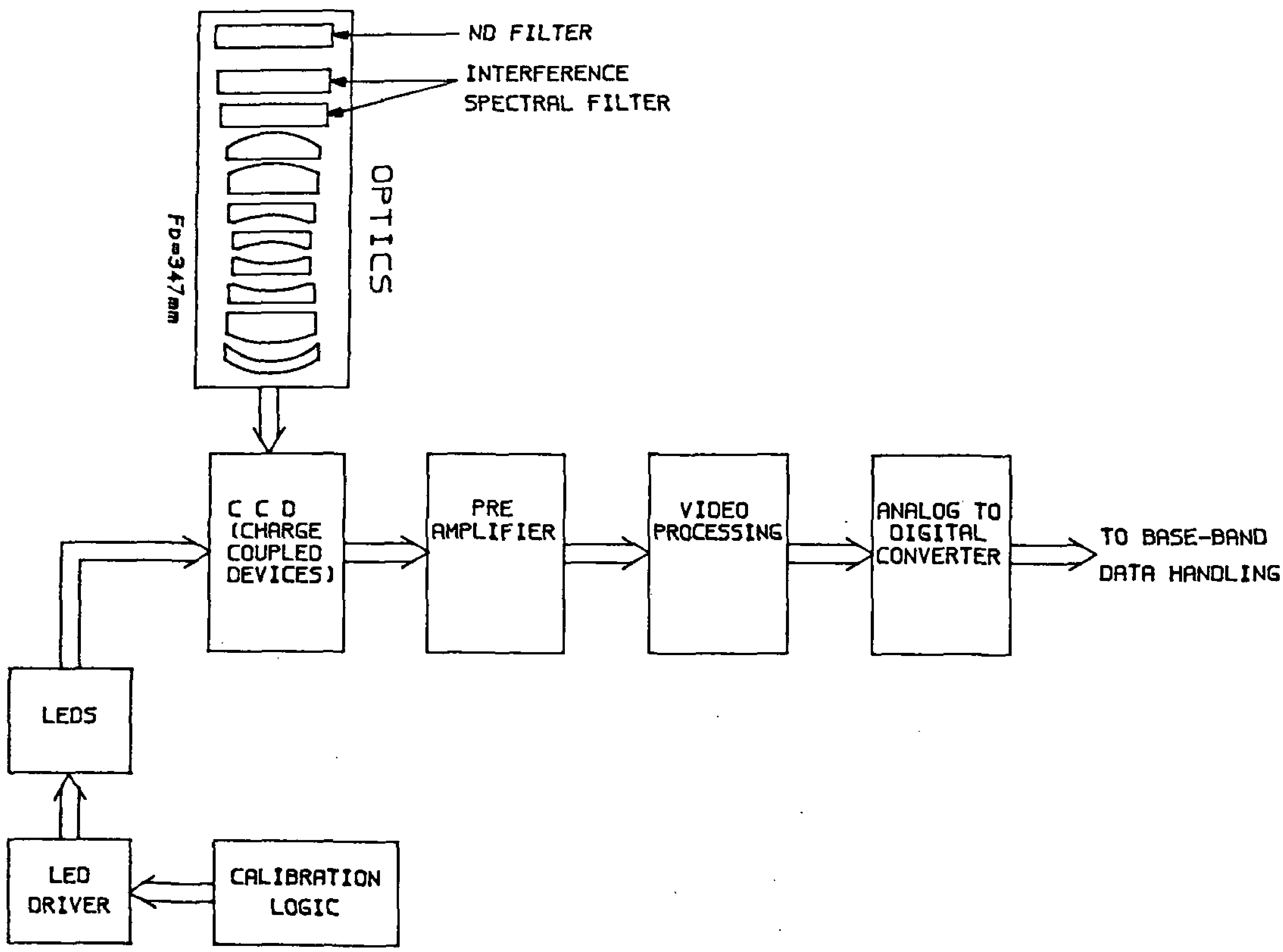

Figure 2. Block schematic of linear imaging self-scanner-III.

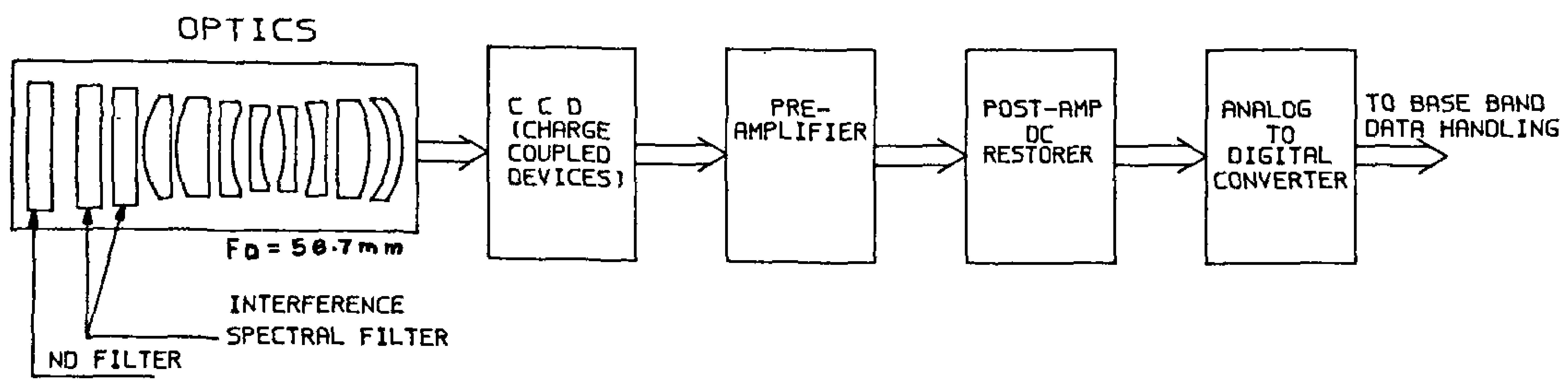

Figure 3. Block schematic of wide field camera.

gradient of less than 2°C across the optics structure. The nickel–cadmium batteries are controlled between 0 and 10°C to provide long life for the batteries. The thermal control is based on conductive/radiative coupling design using heaters and passive thermal control elements like thermal insulation blanket, optical solar reflectors and paints to provide appropriate thermal conductivity/emissivity ratio for the radiative surfaces. A special thermal design is made for controlling the temperature of the SWIR band detector at –10°C using conductive coupling with the help of a copper braid connecting the device and a radiator plate facing the outer space. Separate thermal covers with a view port cutout are incorporated for each camera optics and their temperatures are controlled using automatic ON/OFF temperature controllers. The star sensor requires the detectors to be cooled to –20°C and this is achieved using a peltier cooler.

Mechanisms

The IRS-1C spacecraft contains two deployment and steering mechanisms. The solar panels of span length 4.4 m on each side is to be deployed on orbit and to be driven subsequently using a sun sensor on the solar panels to track the sun. This system is the same as that used in IRS-1A/1B. The new mechanism for IRS-1C is the panchromatic camera deployment on orbit to make it free for steering to provide the stereo pair of imageries of a scene at two different viewing angles. The PAN

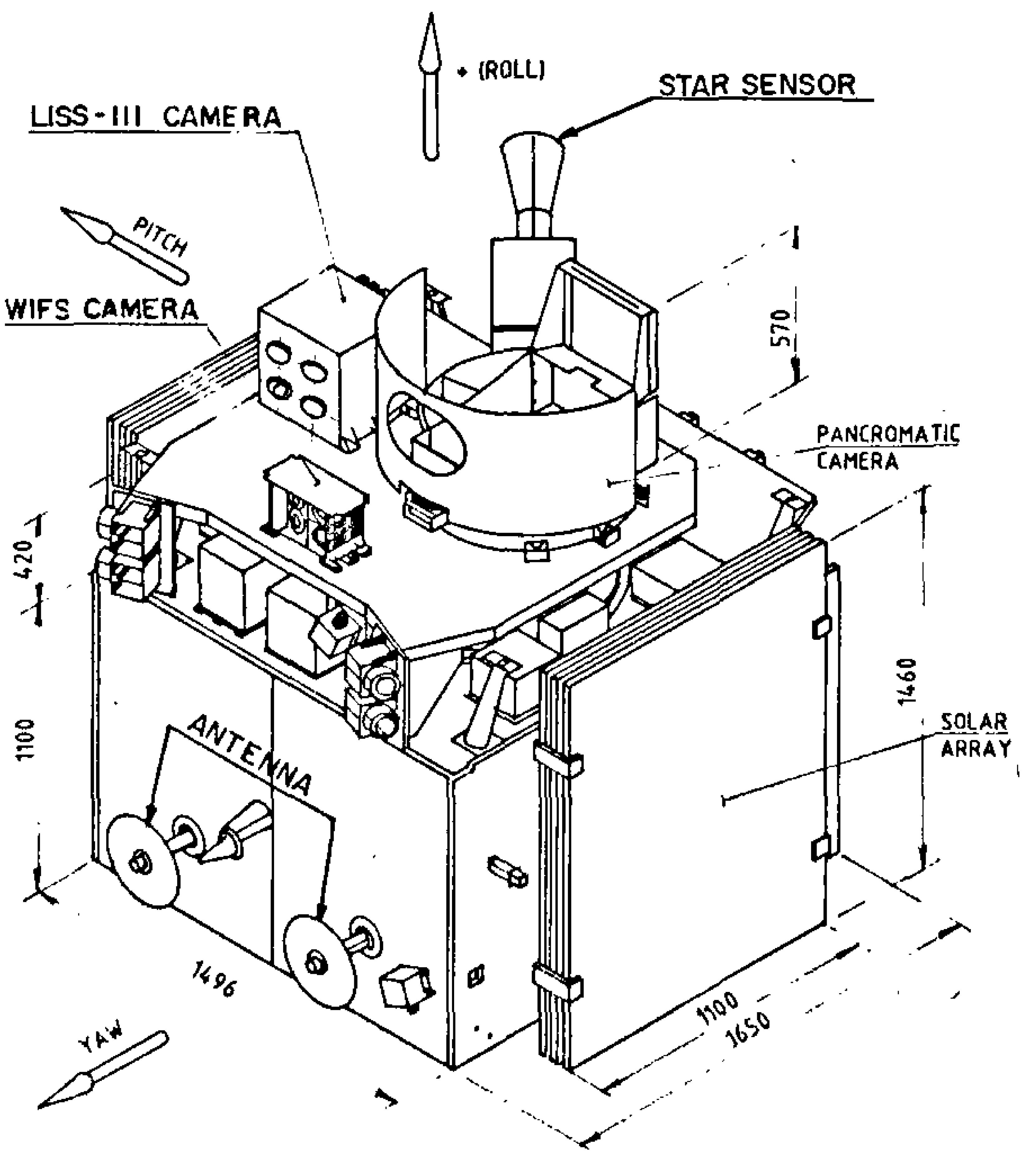

Figure 4 *a*. Launch configuration of IRS-1C.

camera and the connected steering mechanism should be held during launch phase to ensure that the launch loads are not transmitted to the steering mechanism. A pyro cutter cuts a metallic rope and releases the spring loaded clamps to release the payload from hold down condition. The payload-steering mechanism uses a diaphragm suspended shaft to offload the launch loads and after release uses a stepper motor using a 1 : 20 gear ratio for steering the camera in steps of 0.09° up to ± 26° in the off-nadir direction. Resolvers read out the accurate rotational position of the payload when payload is rotated to take the off-nadir position. The important requirement in this is that the camera, though free to rotate, should have a mechanical stability of less than 0.1 arcsec to avoid any jitter during imaging. This has been ensured by providing appropriate dampers about all the three axes.

Power system

The power system consists of two solar panels, one on each side making available a total area of 8.58 m^2 for tracking the sun and providing end of life raw power of 720 W to support the spacecraft. The solar panel uses silicon solar cells of 2 × 4 cm size. Two nickel–cadmium batteries of 21 AH each support eclipse and peak load operations. These batteries were developed inhouse using procured cells. The power conditioning and distribution is done according to the subsystem requirements with an efficiency of better than 75% and low noise. A taper charge regulator is used for controlling the charging of the battery.

Communication and data handling

The communication system caters to both the downlink and uplink communication requirements from/to the spacecraft. The uplink consists of the telecommand link providing 704 ON/OFF and 46 data commands whereas the downlink caters to the house keeping telemetry providing about 1000 channels. The above two communication links are also used for sending range tracking tones to track the spacecraft by determining the range and range rate of the spacecraft with respect to the ground station providing an accuracy of 10 m in range and 10 cm/sec in range rate. The TTC link works in S-band through an onboard near omni antenna. The telemetry, telecommand and tracking system are derived from IRS-1A/1B but with a mission specific design.

The data handling system provides for payload data telemetry downlink for all the three payload cameras. The data handling system formats the payload data into 84 MBS data format for PAN camera and 42 MBS format for LISS-III + WiFS data. The data QPSK modulates the individual x-band carrier for each chain and the modulated carrier is amplified using a travelling wave tube amplifier to provide 40 W of transmission power and is radiated through a shaped beam earth coverage antenna. The data handling system along with QPSK modulators is newly developed inhouse for IRS-1C.

Onboard tape recorder

An onboard tape recorder has been incorporated in the spacecraft to provide for recording of the payload data over areas of interest as required by the users, without any visibility constraint from the ground station. The data thus recorded using a time tag command can be played back when the spacecraft is visible over the ground station. The capacity of the tape recorder is such that about 21 minutes of payload data can be recorded in a single or multiple segments, providing 64 G bits of storage.

Attitude and orbit control system (AOCS)

The attitude control system provides for control of the spacecraft in different phases. During the initial phase, the spacecraft (and solar panels) should be controlled in sun acquisition mode till the spacecraft is 3-axis stabilized in the subsequent orbit. In 3-axis stabilized mode, the control system uses many sensors in the control loop. While the controlling sensors are gyroscopes in all the three axes, they are frequently corrected for their inherent drifts using pitch, roll and yaw sensors. The gyroscopes developed for IRS program are dry Dynamically Tuned Gyroscopes (DTG) providing low drift and long life. The pitch and roll errors are measured by earth sensor, sensing the earth radiation in 14–16 μm band whereas yaw error is sensed by a sun sensor. The systematic errors seen in the earth sensor arising due to the earth oblateness, eccentricity of the orbit and earth radiance variations can also be corrected in IRS-1C using a systematic error correction package. The schematic of AOCS is given in Figure 6. The actuators for attitude control are 1 Newton thrusters for coarse

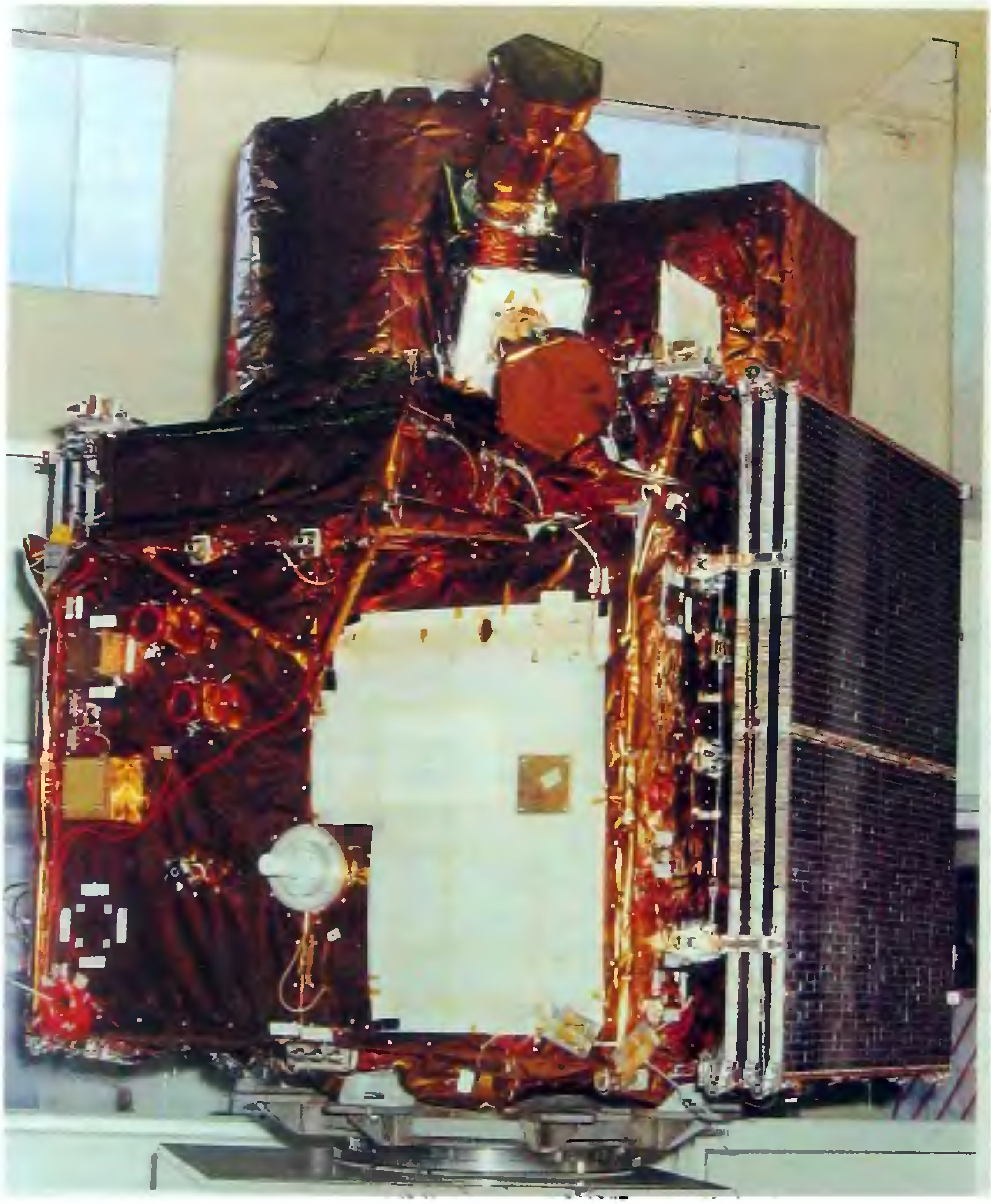

Figure 4 *b*. Assembled view of IRS-1C.

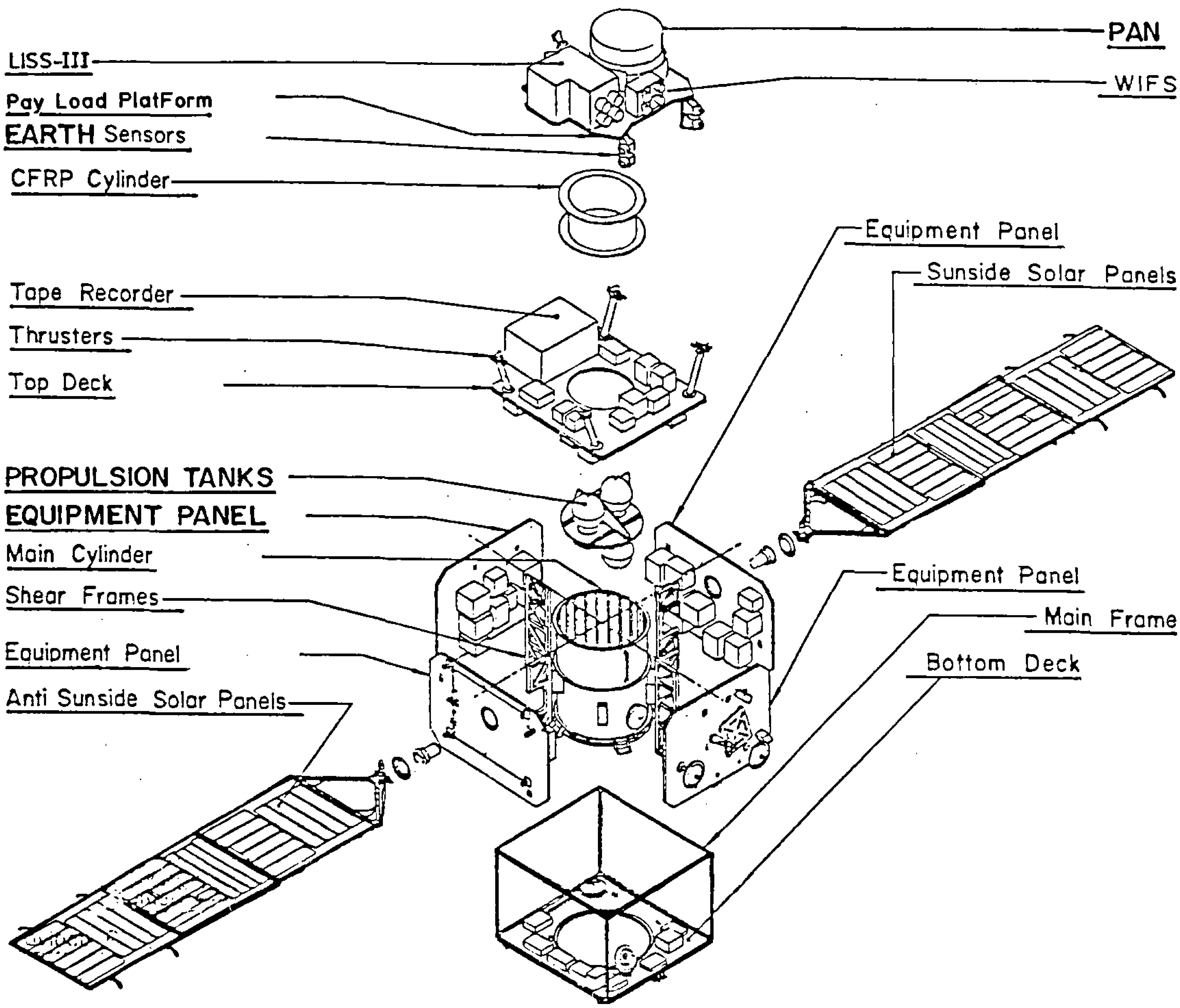

Figure 5. Dis-assembled view of IRS-1C.

attitude control (in the initial phase) and reaction wheels for fine attitude control about each axis in the normal phase. The control system is designed around both software-based/hardware-based systems. While hardware-based system is identical to IRS-1A/1B, the software-based system is developed for low earth orbit application in IRS-1C timeframe. The software system is based on linear controllers for all the axes, while the hardware system is based on pulse width pulse frequency modulation (PWPFM) controllers. The software system using the linear controllers and an update Kalman filter can give better pointing accuracy and drift stability compared to hardware system. Many control modes, viz. update Kalman filter, relative quarternion and dynamic observer are built into the control schemes, which can be selected as the situation demands. The new elements developed for IRS-1C are 3 gyro cluster configuration, processor-based control system for low earth orbit and certain elements of reaction control system like latch valves, filters, thermocouples, etc. The specifications of attitude control system are given in Table 1.

The orbit control system is a part of the same control

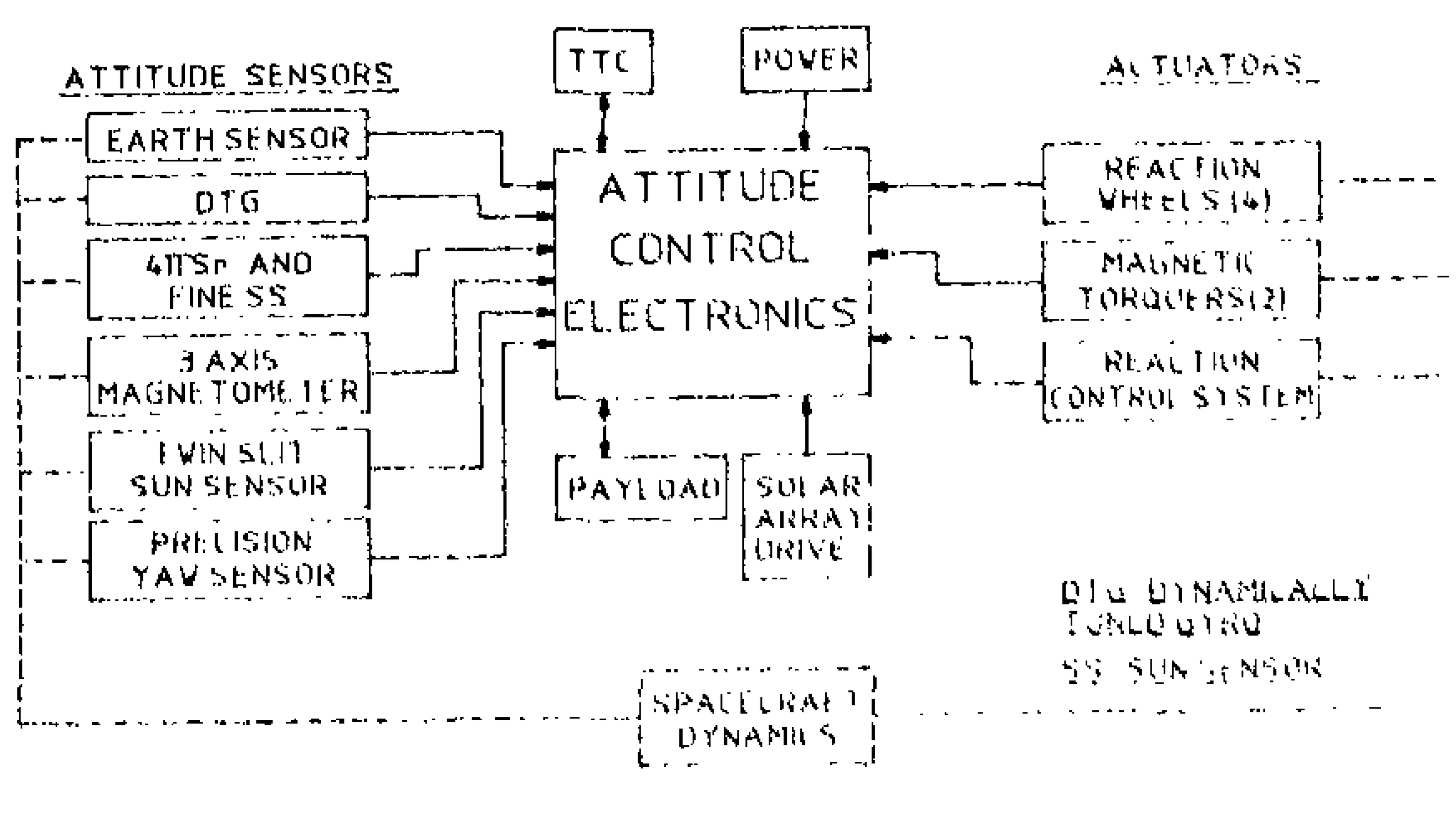

Figure 6. Schematic of IRS attitude orbit control system

system which is being used for removing the initial launch dispersions and frequent orbit adjust manoeuvres resulting from orbit disturbances due to many perturbing forces. This system uses large angle manoeuvre of the yaw axis of the spacecraft and firing of 11 Newton thruster mounted along the negative roll axis. Depending on the out of plane or inplane corrections required, the spacecraft is tilted (by 90°) or not, before the firing of 11 Newton thruster. This scheme of large angle yaw manoeuvre and 11 Newton thrusters are developed for IRS-1C spacecraft. All thrusters use monopropellant (hydrazine) for generation of the thrust when the hydrazine gets decomposed when it is passed through the thruster catalyst bed.

A star sensor which can provide pitch, roll and yaw attitudes to an accuracy of ±0.01° has been developed for IRS-1C based on area array CCD. Two such star sensor heads are deployed in the spacecraft to increase the number of stars sighted at any given time. The star sensor can sense stars of 5th magnitude. The semi-processed data are downlinked for attitude computation on ground using the star catalogue.

Realization stages

The realization of IRS-1C took about six years including the development of the new elements and their qualification. The steps followed in the realization of IRS-1C are:

a) Define the mission objective/mission requirements.
b) Derive specifications for the payloads and the spacecraft.
c) Conduct tradeoff studies to select appropriate configurations for various systems.
d) Baseline/preliminary design.
e) Review of baseline/preliminary design for further configuration control.
f) Configuration control mechanisms to analyse any further changes in the configuration/design proposed during implementation phase to take a view for a decision to accept or reject such proposed changes.
g) Development of the new systems and their qualification through the environmental conditions.
h) Critical design review to verify the adequacy of the design for its duplication in the flight model.
i) Flight model realization and integrated testing of the systems at the spacecraft level and environmental testing.
j) A preshipment/flight readiness review to verify the adequacy of the flight model test results before clearing the spacecraft for shipping to the launch site and subsequent flight.

Model philosophy

The model philosophy consisting of several models was decided for the qualification of the new design. The structure/thermal/engineering model was meant for the structural qualification in terms of static and dynamic mechanical load tests. The engineering cum thermal model is realized with live qualified systems with the same power dissipation as in the flight model. This model goes through all the electrical integrated tests at spacecraft level and also the environmental tests like thermovacuum test, vibration/acoustic test, to verify the performance of the system (Figure 7). Also, the same spacecraft is put through a solar simulation test, where the spacecraft is tested in a vacuum chamber with sun illumination using a sun simulator (Figure 8). The thermal performance of the spacecraft and various systems are verified using this test. Following this, the flight model was realized and tested through the complete acceptance tests.

Quality and reliability aspects

Ensuring the quality of the subsystems which finally results in their reliable performance in orbit is the essence of the implementation. Steps to achieve this include review by expert committees for design adequacy, selection and procurement of parts and materials of the required quality levels, upgradation of non standard parts by in-house screening, adopting only the qualified fabrication processes with the QC inspection at every

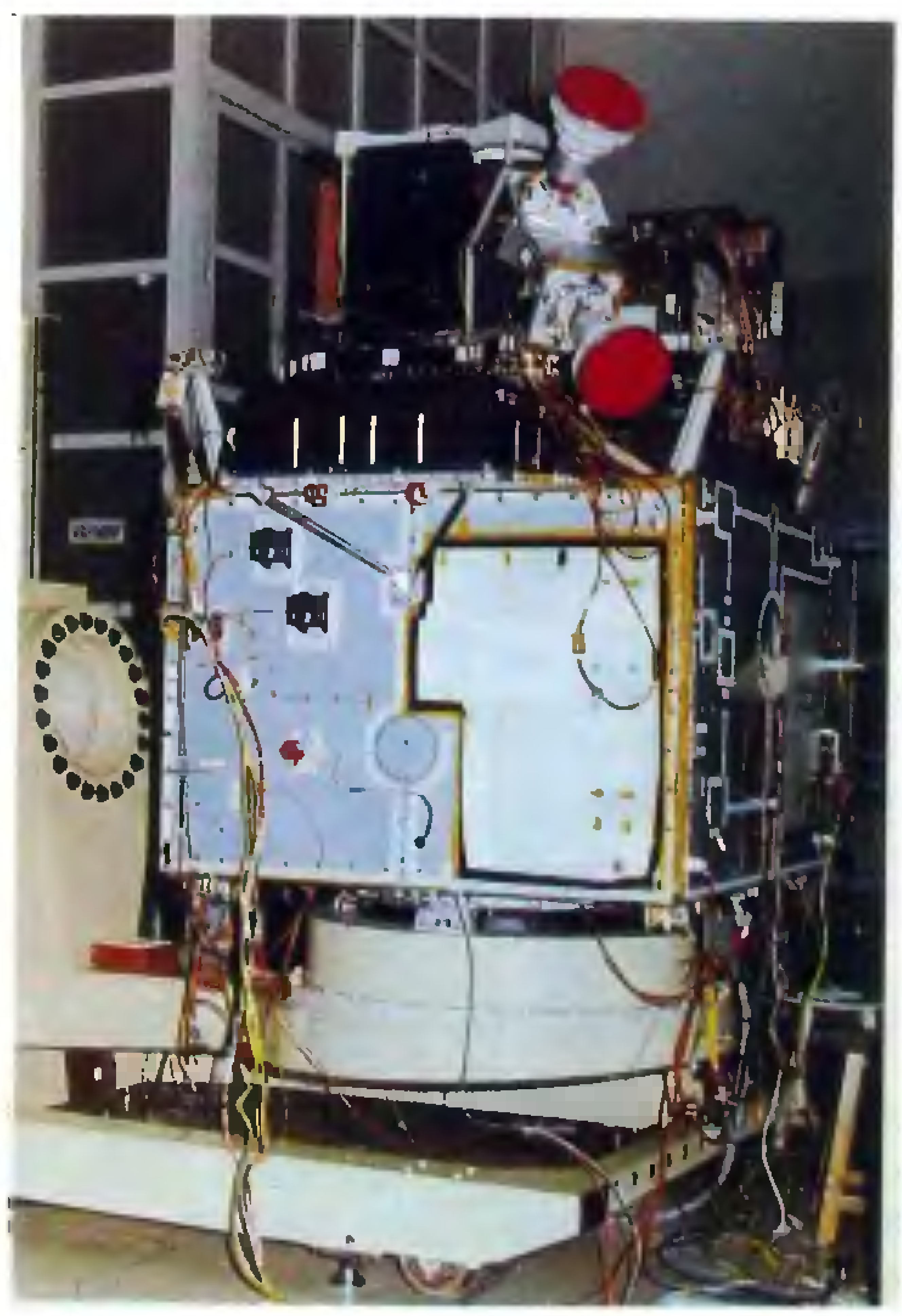

Figure 7. IRS-1C in vibration test.

Figure 8. IRS-1C in solar simulation test.

stage, and testing the elements at the sub-assembly and the assembly levels.

The process of getting the final product is not smooth and many a times, quality problems are faced which could be because of the non-availability of proper parts and materials, design inadequacy showing up and any variation in the approved fabrication process. In all such cases, the problems are analysed, causes are identified and solutions are applied to satisfy the laid-down project requirements.

Management aspects

The matrix management system was followed for managing the IRS project, wherein focus on project activities by the designated project personnel and the availability of the specialized technical expertise from the functional groups for all the projects running concurrently in the centre were used. Better utilization of resources is possible in the above system of management.

Planning and schedule

The project schedule is worked out at the beginning of the project, considering the new developments required for the various subsystems. The schedule is mainly dictated by the development phase followed by the fabrication and test duration of the flight systems. The PERT network analysis is used for arriving at optimistic, realistic and pessimistic time schedules. The optimistic schedule forms the baseline schedule and the critical path is closely monitored and corrective actions are taken when necessary by assigning priorities and allocating the resources in order to meet the end dates. Due to technical problems in the new developments, delays do occur which are compensated to some extent, by work around plans. The schedule control mechanism is mainly based on periodical status reviews by senior functionaries.

Conclusions

The IRS-1C spacecraft has been realized, launched and operationalized on orbit successfully. The on-orbit performance indicates that the systems, supporting the mission have worked meeting their specifications and good quality imageries have been transmitted by the spacecraft. The IRS-1D spacecraft is planned to be launched during 1997–98 to provide the continuity of data to the users.

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