Doc no: ISRO-ISAC-TR-1445

Indian Remote Sensing Missions & Payloads A Glance

January 2018

ISRO Satellite Centre
Indian Space Research Organisation
Bengaluru

Doc No: ISRO-ISAC-TR-1445

Indian Remote Sensing Missions & Payloads- A Glance

Space Science and Ground Segment Section

IRS & SSS PROGRAM MANAGEMENT OFFICE

January- 2018

ISRO SATELLITE CENTRE
INDIAN SPACE RESEARCH ORGANISATION
DEPARTMENT OF SPACE
BANGALORE

Doc No: ISRO-ISAC-TR-1445

Indian Remote Sensing Missions & Payloads- A Glance

Prepared by	Nitu Pathak	Nitu Pathak
	Yogyata Agrawal	Yogyata Agrawal
	Buchi Ramachandra Reddy	B. Ramachandra Reddy
	Y V Sai Bhageerath	Y V Sai Bhageerath
	Vivek Kumar Gautam	[illegible]
	Nashiket Parate	[illegible]
Reviewed by	Ravi Chandra Babu G	[illegible]
	Dr. P Murugan	P. [illegible]
Approved by	G Nagesh	[illegible]

January- 2018

IRS & SSS PROGRAM MANAGEMENT OFFICE
ISRO SATELLITE CENTRE
INDIAN SPACE RESEARCH ORGANISATION
DEPARTMENT OF SPACE
BANGALORE

Preface

The goal of the Indian Space Programme is to harness space technology for applications in the areas of communications, broadcasting, meteorology, disaster warning, search and rescue operations, navigation and remote sensing. Operational systems have been realized in all the above areas during the past two decades.

Remote sensing is an important part of the Indian Space Programme and the Department of Space (DOS), Government of India, is the nodal agency for the realization of the National Natural Resources Management System (NNRMS), the National Resources Information System (NRIS) and the Integrated Mission for Sustainable Development (IMSD), besides several other national level application projects like Crop Acreage and Production Estimation (CAPE), National Drinking Water Mission and Wasteland Mapping etc in close collaboration with the user agencies. Operational satellites have been indigenously built and launched, which cater to land and ocean applications.

The Indian remote sensing programme has come a long way since the first experiment conducted by aerial surveys to monitor coconut wilt disease in early seventies. Noting the immense potential benefits of remote sensing in the areas of resources survey, inventory and management, the Indian Space Research Organisation (ISRO) has embarked upon the development of this technology in a major way through the use of orbiting satellites. This goal was initiated through the experimental missions, namely, Bhaskara-1 and Bhaskara-2 in the late seventies and early eighties and then graduating to more sophisticated state-of-the-art operational missions of Indian Remote Sensing Satellite (IRS) series.

The aim of this document is to collect information about the launched Indian Remote Sensing payload and spacecraft available till date at various sources and consolidate in a single document. This document provides brief, condensed information of various IRS Missions and payloads. This document will serve the new entrant engineers to understand the evolution of Remote Sensing Programme in India. Information provided in this document is collected from various sources like Project reports, internet sources, Journals and books.

The Space Science and Ground Segment section team in PMSG is thankful to Dr. M Annadurai, Director ISAC for his continuous motivation and encouragement. The team thankful to Shri G Nagesh, Programme Director, IRS & SSS Programme for valuable guidance and encouragement. The team acknowledges with thanks the encouragement and support provided by Dr. P Murugan, Group Head, PMSG. The team also acknowledges Shri Ravi Chandra Babu G, Head, ISSD for his valuable guidance.

Change History

Version No	Date	Change Information	Remarks
1.0	May 2012	Initial Document	N/A
1.1	Jan 2018	Updated document with launched spacecraft till Jan-2018	N/A

CONTENTS

Acronyms

4 Pi SS	4 Pi Sun Sensor
ADC	Analog to Digital Converter
ADCS	Attitude Determination and Control Subsystem
AH	Ampere Hour
AHB	Advanced High speed Bus
AIT	Assembly Integration & Testing
Al	Aluminium alloy
AOCE	Attitude and Orbit Control Electronics
AOCS	Attitude and Orbit Control System
APB	Advanced Peripheral Bus
APD	Avalanche Photo Detector
APS	Active Pixel Sensor
ASIC	Application Specific Integrated Circuit
ASLV	Augmented Satellite Launch Vehicle
ATC	Auto Temperature Control
ATE	Automated Test Equipment
ATJ	Advanced Triple Junction
AWiFS	Advanced Wide Field Sensor
BAPTA	Bearing and Power Transfer Assembly
BBR	Band-To-Band Registration
BCD	Binary Coded Decimal
BDH	Baseband Data Handling
BDR	Base-line Design Review
Be	Berillium
BER	Bit Error Rate
BFL	Back Focal Length
BMU	Bus Management Unit
BOL	Beginning of Life
BPSK	Binary Phase Shift Keying
BRC	Bit Rate Clock
CAN	Controller Area Network
CARTOSAT	Cartographic Satellite
CBT	C-Band Transponder
CCD	Charge Coupled Device
CCGA	Ceramic Column Grid Array
CCL	Closed Control Loop
CCSDS	Consultative Committee for Space Data Systems
CDR	Critical Design Review
CFRP	Carbon Fibre Reinforced Plastic
CGP	Central Grounding Point
CIP	Command Interface Port
CMG	Control Moment Gyro
CMOS	Complementary Metal Oxide Semiconductor
CMRB	Configuration Management Review Board
CNES	Centre National d'Etudes des Spatiales
COTS	Commercial - Off –The- Shelf
CP	Circular Polarized (RCP, LCP)
CPM	Charge Particle Monitor
CPSK	Coherent Phase Shift Keying
CRC	Cyclic Redundancy Code
CTE	Coefficient of Thermal Expansion
CTF	Contrast Transfer Function
CVD	Chemical Vapor Deposition
CZT	Cadmium Zink Telluride
dB	Decibel
DCU	Data Compression Unit
DDR	Detailed Design Review
DE	Detector Electronics
DEC	Decoder
DFT	Discrete Fourier Transform
DFU	Data Formatting Unit
DGA	Dual Gimbal Antenna (DGA)
DH	Data handling
DIP	Dual Inline Package/Data Interface Package

DMA	Direct Memory Access
DMSS	Dynamic Multi Star Simulator
DN	Digital Number
DP	Data Products
DPCM	Differential Pulse Code Modulation
DPSK	Differential Phase Shifting Keying
DQE	Data Quality Evaluation
DRAM	Dynamic Random Access Memory
DSER	Deserializer
DSN	Deep Space Network
DSP	Digital Signal Processing
DSS	Digital Sun Sensor
DTG	Dynamically Tuned Gyroscope
EDAC	Error Detection and Correction
EED	Electro Explosive Device
EFL	Effective Focal Length
EID	Electrical Interface Document
EIRP	Effective Isotropic Radiated Power
EM	Engineering Model
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOL	End of Life
EOM	Electro-Optical Module
EOS	Earth Observation System
ESD	Electro Static Discharge
FCC	False Color Composite
FD	Flange Distance
FM	Frequency Modulation
FM	Flight Model
FOG	Fiber Optic Gyro
FOV	Field of View
FPA	Focal Plane Arrays
FPGA	Field Programmable Gate Array
FRR	Flight Readiness Review
FSK	Frequency Shift Keying

G/T	Gain/Temperature
Gbps	Giga Bits Per Second
GCP	Ground Control Point
GMT	Greenwich Mean Time
GPS	Global Positioning System
GSLV	Geo-Synchronous Satellite Launch Vehicle
HEX	High Energy X-ray Payload
HILS	Hardware In-Loop Simulation
HK	House Keeping
HMC	Hybrid Microwave Circuit
HR	High Resolution
HRMX	High Resolution Multispectral
HySI	Hyper spectral imaging instrument
IC	Integrated Circuit
IFOV	Instantaneous Field Of View
IGFOV	Instantaneous Geometric Field Of View
IIMS	Integrated Information Management System
IISU	ISRO Inertial Systems Unit
IMS	Information Management System
IMS	Indian Micro Satellite
INCOIS	Indian National Committee for Ocean Information Services
InGaAs	Indium Gallium Arsenic
IOC	Integrated Optic Chip
IP	Internet Protocol
IP core	Intellectual Property core
IR	Infra-Red
IRS	Indian Remote Sensing Satellites
IRU	Inertial Reference Unit
ISPRS	International Society for Photogrammetry and Remote Sensing
ISRO	Indian Space Research Organisation
ISS	International Space Station
ISSDC	Indian Space Science Data Centre

IST	Integrated Spacecraft Testing
ISTRAC	ISRO Telemetry Tracking and Command Network
ITT	International Telephone & Telegraph
JPEG	Joint Photographic experts group
K	Kelvin
LAXPC	Large area Xenon Filled Proportional counter
LCD	Liquid Crystal Display
LED	Light Emitting Diodes
LENA	Low Energy Neutral Atom
LEO	Low Earth Orbit
LEOS	Laboratory for Electro Optics Systems
LHCP	Left Hand Circular Polarisation
Li-ion	Lithium ion
LISS	Linear Imaging Self-Scanning Sensor
LLRI	Lunar Laser Ranging Instrument
LNA	Low Noise Amplifier
LO	Local Oscillator
LOCO	Low Complexity Lossless Compression
LPSC	Liquid Propulsion Systems Centre
LTC	Light Transfer Characteristics
LVDS	Low Voltage Differential Signaling
LWIR	Long Wave Infrared
M3	Moon Mineralogy Mapper
MADRAS	Microwave Analysis and Detection of Rain and Atmospheric Structures
Mbps	Mega Bits Per Second
MCT	Mercury Cadmium Telluride
MEO	Medium Earth Orbit
MEOSS	Monocular Electro-Optical Stereo Scanner
MHDMSS	Multiple Head Dynamic multi star simulator
MI	Moment of Inertia

MID	Mechanical Interface Document
MIL-STD	Military Standard
MIP	Moon Impact Probe
MIR	Medium wave Infra-Red
MLI	Multi-Layer Insulation
MMU	Mission Management Unit
MOS	Multispectral Opto-electronic Scanner
MRB	Material Review Board
MRR	Mission Readiness Review
MSMR	Multi-frequency Scanning Microwave Radiometer
MTC	Magnetic Torquer Coil
MTF	Modulation Transfer Function
MX	Multispectral
NDC	NRSC Data Centre
NI-Cd	Nickel Cadmium Batteries
NIR	Near Infra-Red
Nm	Newton Metre
NMS	Newton Metre Second
NRSC	National Remote Sensing Centre
OBC	On-Board Controller/Computer
OBT	On-Board Time
OCM	Ocean Colour Monitor
OCP	Over Current Protection
OILS	Onboard Software In Loop Simulation
OSR	Optical Solar Reflector
P	Pitch axis
PAA	Phased Array Antenna
PAN	Panchromatic
PCB	Printed Circuit Board
PCM	Pulse Code Modulation
PDR	Preliminary Design Review
PEB	Project Executive Board
PFZ	Potential Fishing Zone
PID	Parameter Identification

PINFET	Positive Intrinsic Field effect Transistor
PIU	Payload Interface Unit
PLE	Payload Electronics
PLL	Phased lock loop
PM	Phase Modulation
PMB	Project Management Board
PMO	Programme Management Office
PMT	Photo Multiplier Tube
PPC	Payload Power Converter
PPC	Pointed Proportional Counter
PPR	Payload Power Regulator
PrEB	Programme Executive Board
PrMB	Programme Management Board
PRNU	Photo Response Non-Uniformity
PROM	Programmable Read Only Memory
PSK	Phase shift keying
PSLV	Polar Satellite Launch Vehicle
PSR	Pre-Shipment Review
QPSK	Quadrature Phase shift keying
R	Roll axis
RADOM	Radiation Dose Monitor
RAM	Random Access Memory
RCS	Reaction Control System
RF	Radio Frequency
RHCP	Right Hand Circular Polarization
RISAT	Radar Imaging Satellite
RISC	Reduced Instruction Set Computing
ROM	Read Only Memory
ROSA	Radio Occultation for Sounding of Atmosphere
RS	Reed-Solomon, Receive/send
RTO	Regenerative Thermal Oxidizer
RTX	Receive/Transmit
RW	Reaction Wheel

Rx	Receiver
S/N	Signal-to-Noise Ratio
SAA	Sun Aspect Angle
SAC	Space Application Centre
SADA	Solar Array Drive Assembly
SAMIR	Satellite Microwave Radiometer
SAR	Synthetic Aperture Radar
SARA	Sub KeV Atom Reflective Analyzer
SARAL	Satellite for Argos and Altika
SCC	Spacecraft Control Centre
SCD	Swept Charge Device
SEO	Satellite for Earth Observation
SER	Serializer
SGCMG	Single Gimbal Control Moment Gyro
SiC	Silicon Carbide
SLV	Satellite Launch Vehicle
SNR	Signal to Noise Ratio
SOC	System On Chip
SPS	Satellite Positioning System
SPSS	Solar Panel Sun Sensor
SRC	Standing Review Committee
SROSS	Stretched Rohini Satellite Series
SS	Star Sensor
SSM	Scanning Sky Monitor
SSPA	Solid State Power Amplifier
SSPO	Sun Synchronous Polar Orbit
SSR	Solid State Recorder
SSRB	Subsystem Review Board
SST	Sea Surface Temperature
SWIR	Short Wave Infrared
SWR	Square Wave Response
SXT	Soft X-ray imaging Telescope
TC	Telecommand
TCXO	Temperature Controlled Crystal Oscillator
TDI	Time Delay Integration

TES	Technology Experimental Satellite
TIFR	Tata Institute of Fundamental Research
TM	Telemetry
TMC	Terrain Mapping Camera
TSG	Thermal Systems Group
TTC	Telemetry Tracking and Command
TWTA	Traveling Wave Tube Amplifier
Tx	Transmitter
UHF	Ultra high Frequency
USB	Universal Serial Bus
UTMC	United Technologies Microelectronics Center
UV	Ultra Violet
UVIT	Ultra Violet Imaging Telescope
VHDL	Very high speed Hardware Description Language
VHF	Very High Frequency
VNIR	Visible and Near InfraRed
VSSC	Vikram Sarabhai Space Centre
VSSGCMG	Variable Speed Single Gimbal Control Moment Gyro
WDE	Wheel Drive Electronics
WiFS	Wide Field Sensor
XSM	X-ray Sky Monitor
Y	Yaw Axis
ISSP	Indian Scientific Satellite Project

1. Introduction

The Indian Space Research Organization (ISRO) planned a long-term Satellite Remote Sensing programme in seventies, and started related activities like conducting field & aerial surveys, design of various types of sensors for aircraft surveys and development of number of application/utilization approaches. These were followed by planning, designing, fabrication of experimental satellite remote sensing missions, Bhaskara 1 & II etc. These missions gave experience in developing remote sensing satellites, setting up of ground-based data reception and processing systems, experience in over-all mission management etc.

The launch of IRS-1A satellite on 17th March, 1988 into the orbit is the start of operational remote sensing era of IRS programme. The IRS Satellites are providing imagery data for many national important projects/applications. Some of remote sensing applications being catered are provided below

Agriculture and soils
Phase level information of soils
Improved multiple crop discrimination
Crop monitoring & condition assessment
Crop canopy water stress
Crop yield estimates
Crop management
Cropping system analysis
Damage assessment
Surveillance of pests and diseases
Forestry
Inventory and updating
Forest landscape analysis
Forest infra-structure mapping
Forest encroachment
State of forests
Wildlife habitat analysis
Bio-diversity
Fire damage
Implementation of forest policies
Environment
Hydrologic units
Land unit maps
Soil contamination maps
Quarries and waste identification

Desertification analysis
Oil spills
Point and non-point sources of pollution
Environmental impact assessment
Geology and Exploration
Rock type mapping
Tectonic geo- structure mapping
Mining pollution analysis
Off/on shore seep analysis
Coal fire analysis
Mining subsidence analysis
Landslide vulnerability / risk
Geo-energy
Water cycle study
Ocean application
Phytoplankton observation
Chlorophyll content,
Yellow substance
Suspended sediments
Sea surface winds,
Sea roughness monitoring
Sea surface temperature
Identifying the potential fishing zones,
Coastal zone management,
Ship routing,
Operations of offshore oil rigs

Meteorology
Water vapour in an atmospheric column,
Cloud formation
Low pressure zone identification
Cyclone movement speed & direction
Weather predictions,
Infrastructure and Utilities
Road networks
3D-city models
Infrastructure maps
Siting of hydro-power locations
Site suitability
Rural and urban infrastructure
Structural and hydrological inventory
Municipal GIS
Utility corridor mapping
Transportation network
Rural road connectivity
Tracking changes in road
Telecom facilities
Recreation facilities
Tourism
Violations
Damage assessment

Cartography
Updating topo-maps
Augmenting Databases
Image maps as base maps
Watershed management
Terrain evaluation
City models
Road and infrastructure maps
Site suitability assessment
Cadastral map generation
Defense
Strategic target monitoring
Mission planning
Training
Treaty verification
Demining
Lunar & Stellar Observation
Understand the way planets created
Stellar movement,
Celestial body feature study
Studying radiations coming from stars
Different elements available on stellar objects

Above applications can be categorized in following way as part of National Natural Resources planning as well as in other areas.

- Multiple crop production estimates,
  - Area estimate, crop health estimate
- Land and Water Resources optimization,
- Urban planning and management,
  - Infrastructural plan
- Coastal zone studies and regulation
  - Fishing, chlorophyll, phytoplankton etc.
- Mapping and inventory of forests, wastelands, land use,
- Lunar & stellar observation missions

The aim of this document is to collect information about Indian Remote Sensing payload and spacecraft available at various sources and consolidate in a single document. This document provides brief, condensed information of various IRS Missions and payloads. This document will serve the new entrant engineers to understand the evolution of Remote Sensing Programme in India. Information provided in this document is collected from various sources like Project reports, internet sources, Journals and books.

IRS missions are classified based on their applications as follows.

- Land and Water Resource Observation Series
- Ocean and Atmospheric Observation Series
- Cartographic Satellite Series
- Microwave Remote Sensing Series
- Space Science Series
- Micro and Nano Satellite Series
- University Satellites

1.1 Land and Water Resource Observation Series

This series satellites cater the requirements of Agriculture, forestry, land use, land cover, soil, geology, terrain, water resources, disaster management like flood, forest fire, drought, land slide, etc. applications.

Satellites launched for these applications are IRS-1A, IRS-1B, IRS-1C, IRS-1D, IRS-P6 (Resourcesat-1), Resourcesat-2/2A and IMS-1A. Following IRS Satellites are operational and providing data for land and water resource applications.

- Resourcesat-2 satellite with improved performance having three payloads namely, LISS-IV with 5.8m Resolution and 70 km swath in 3 bands, LISS-III with 23.5m Resolution and 140 km swath and AWIFS with 56m Resolution and 740 km swath was launched on 20thApril 2011 and successfully operationalized and is providing data for above said applications.
- Resourcesat-2A (RS-2A) mission was launched on 07th December 2016 to provide continuity of operational services of AWiFS, LISS-3 and LISS-4 payloads. Primarily the spacecraft configuration of RS-2A is similar to RS-2 with few incremental changes in some of the systems to take care of the in-orbit observations of RS-2 and improve the overall mission operations management. A new payload Solid state C Band Transponder (SCBT) is added to aid the calibration of C-Band Radars at SDSC, SHAR.

Following Satellites are planned in future to provide continuity services with improved features.

- Resourcesat-3 Series
- Resourcesat-3S Series

1.2 Ocean and Atmospheric Observation Series

Under Ocean and Atmospheric Observation Series, satellites were launched to meet the needs of potential fishing zone (PFZ) forecasting, sea state forecasting, and coastal zone studies. It also provide inputs for weather forecasting and climatic studies:

Satellites launched for this application are Oceansat-1, Oceansat-2, Megha Tropiques, SARAL, Scatsat-1. Following Satellites are operational and providing data for Ocean and atmospheric observation applications.

Oceansat-2: Satellite with Ocean Colour Monitor (OCM) payload with 360m spatial resolution and 1420 km swath and Ku-Band pencil beam scatterometer and Radio Occultation for Sounding of Atmospheric (ROSA) was launched on 23rd September 2009 was successfully operationalized for providing data for the above applications.

MEGHA-TROPIQUES: Under Ocean and Atmospheric Observation series to study the convective systems that influence the tropical weather and climate. The Megha-Tropiques mission was launched on 12th October 2011. The Megha-Tropiques spacecraft carries 4 payloads namely:

- MADRAS (A Microwave Imager to study the precipitation and cloud properties).
- SAPHIR (A Microwave Sounder for the retrieval of water vapour vertical profiles).
- SCARAB (A Radiometer for the measurement of outgoing radiative fluxes at the top of the atmosphere) and
- ROSA (Radio Occultation for Sounding of Atmosphere) to provide vertical profiling of temperature.

The Megha-Tropiques mission provides sampling of water and energy budget of the Tropical Convective Systems in the inter-tropical band.

SARAL: Another international co-operative mission in the area of Ocean and Atmospheric series is SARAL (Satellite for ARGOS and ALTlKA) mission to provide data for marine meteorology and sea-state forecasting, operational oceanography, seasonal forecasting and climate monitoring and to provide data for ocean, earth system and climate research.

The SARAL spacecraft carries two payloads namely, ALTlKA (a Ka-Band Altimeter) and ARGOS (a data collection system from ARGOS platform). The payloads provided by CNES, France and the mainframe and the Assembly, Integration & testing and launching is by ISRO.

Scatsat-1: ScatSat-1 is a continuity mission for Scatterometer payload (Scat-1) on-board Oceansat-2 spacecraft. This mission is providing continuity of weather forecasting services to the user communities, as Scat-1 payload was declared non-operational. The data from this payload was being used by many national and international users. Hence, demand for a new satellite with only Scatterometer has come up.

ScatSat-1 carries Ku-Band Scatterometer named as Scat-2 payload (Oceansat-2 Scatterometer was Scat-1) similar to the one flown on-board Oceansat-2 but with enhanced features. Scatterometer is an instrument working on the principle of back-scattered energy and is used to measure the wind velocity (speed and direction) over the ocean.

Following Satellites are planned in future to provide continuity services with improved features.

- Oceansat-3 Series
- Scatsat-1A

1.3 Cartographic Satellite Series

In order to meet the large scale and thematic maps for urban & rural infrastructure development & management and to provide National Digital Elevation Model (DEM) for cadastral overlay these satellites were launched. They cater cartographic applications, coastal features mapping, coral reef mapping and for mineral studies.

The features of these satellites are : TES with <1 m resolution PAN, Cartosat-1 with 2.5m Resolution PAN and 30 km swath with along track stereo imaging capability, Cartosat-2/2A/2B with 0.8m Resolution PAN with 9.6 km swath

In order to meet the increased demand for large scale mapping and other cartographic applications for cadastral level and for urban and rural infrastructure development & management, Cartosat-2A and Cartosat-2B with 0.8m Resolution PAN with 9.6 km swath were launched for above applications.

Cartosat-2 Series Satellites are high resolution remote sensing satellite configured with panchromatic camera and a 4-band multispectral camera operated in '*Time Delay Integration (TDI)*' mode. It provides scene specific imageries of 0.64m spatial resolution in panchromatic camera and better than 2m in multispectral camera with a swath of 10 km.

Satellites launched for this application are TES, Cartosat-1, Cartosat-2/2A/2B, Cartosat-2S. Following Satellites are operational and providing data for cartographic applications.

Following Satellites are planned in future to provide continuity services with improved features.

- Cartosat-2S
- Resourcesat-3S Series
- Cartosat-3 Series

1.4 Microwave Remote Sensing Satellite Series

In order to provide data during the cloud cover seasons (Kharif) over the tropical regions for many applications like agriculture, and damage assessment during flood and for mitigation effects Microwave Remote Sensing Satellites are planned.

The **RISAT-1** with C-Band Synthetic Aperture Radar (SAR) is in the advanced stage of Assembly, Integration, and Testing. The SAR payload with C-Band TR modules have undergone extensive Testing and evaluation at SAC, Ahmedabad and delivered for Integration with the spacecraft. The mainframe systems are also in the advanced stage of integration

The spacecraft can operate in different modes of operation and provides various resolution imageries like:

- High Resolution Spotlight Mode (HRS) provides 1 - 2m resolution with 10 km x 10 km spot scenes.
- Fine Resolution Strip Mode (FRS-I) provides 3 - 9m resolution with 30 km x 30 km scenes.
- Fine Resolution ship mode (FRS-2) provides 6 - 9m resolution with 30 km swath.

- Medium Resolution Scan SAR Mode (MRS) with 25m resolution with a swath of 120 km.

The RISAT-1 is providing all-weather day and night imageries for applications in the areas of Agriculture for identifications, detection and classification for acreage estimation, forest type plantations and accurate bio-mass estimation, flood mapping to provide accurate flood inundation zones for early relief measures, soil moisture and Hydrology including snow cover and snow wetness, etc.
Following Satellites are planned in future to provide continuity services with improved features.

- RISAT-1A
- RISAT-2A
- NISAR

1.5 Space Science and Planetary Series

The study of Lunar and stellar sources provide better understanding about the universe and planet creation etc. Following satellites are in this series.

IRS-P3: Indian X-ray Astronomy Experiment (IXAE) payload on IRS-P3 satellite launched in 1996. Observational studies of many bright X-ray binary star systems including X-ray pulsars and stellar mass black hole candidates have been carried out. Studies of mass accretion around neutron stars and black hole binaries were some of the major outcomes from this experiment.

Chandrayaan-1: The launch of Chandrayaan-1 has demonstrated the technological capabilities of reaching the outer planets and has confirmed the scientific findings of the previous International Missions, like the presence of water molecules and other precious elements on the surface of the moon.

The significant scientific finding have provided impetus to further space research activities in the country and has created special awareness and enthusiasm among the younger generation.

Mars Orbiter Mission: Mars Orbiter Mission is ISRO's first interplanetary mission with an orbiter craft designed to orbit Mars in an elliptical orbit of 366 km x 80000 km. Mars Orbiter Mission can be termed as a challenging technological mission considering the critical mission operations and stringent requirements on propulsion, communications and other bus systems of the spacecraft. The primary driving technological objective of the mission is to design and realize a spacecraft with a capability to perform in Earth Bound Manoeuvre (EBN), Martian Transfer Trajectory (MTT) and Mars Orbit Insertion (MOI) phases and the related Deep Space mission planning and communication management with a distance varying from nearly 60 million km to 380 million km. Given that the Round-trip Light Time (RLT) from Earth to Mars can vary anywhere between 6 to 43 minutes, it would be impractical to micromanage the mission from Earth. Therefore, autonomous Fault Detection, Isolation and Reconfiguration (FDIR) becomes vital for the mission.

ASTROSAT: The multi-wave length Astronomical Observatory the ASTROSAT is an IRS-class satellite with a mass of about 1515 kg, and power generation capacity of about 1600 W. It is launched from Satish Dhawan Space Centre (SDSC), Sriharikota on 28th September 2015 by PSLV C30 (XL) to a 650 km near-equatorial orbit with an inclination of about 6 degrees.

The ASTROSAT carries onboard a total of 6 experiments i.e. Ultraviolet Imaging Telescope (UVIT), Soft X-ray Telescope (SXT), Large Area X-ray Proportional Counter (LAXPC), Cadmium Zinc Telluride Imager (CZTI), Scanning Sky Monitor (SSM) and Charged Particle Monitor (CPM).

Following Satellites are planned in future for space exploration.

- Chandrayaan-2
- Aditya-L1
- Mars Orbiter Mission-2
- Venus Exploration

1.6 Micro And Nano Satellite Series

IMS-1: With the advances in miniaturization and the advances in high performance devices and techniques, it has become feasible to realize the functions of bigger satellites into Micro & Nano satellites. The ISRO's first Mini Satellite (IMS-1) with 36m Resolution Mx with 141 km swath and 505 m resolution Hyper-spectral Imager (HySI) with 64 bands has been realized within 85 kg and was launched successfully during April 2008.

Youthsat: This satellite is second in this series, carrying three payloads namely SOLRAD, LiVHiSI and RaBIT. In was launched into space on 20thApril 2011 by PSLV-16 along with Resourcesat-2. The SOLRAD payload monitors the Solar activities through hard X-rays gamma rays and particle mostly electrons and protons. The effect of the solar activities on atmosphere is studied by the RaBIT. The effect on the thermosphere, which co-exists with the ionosphere is monitored by LiVHySI.

Microsat: Microsatellites (Microsat) are small satellites with small volume, low power requirements and weighing 100 kg. MICROSAT is an advanced optical remote sensing satellite for providing spot imageries with a high spatial resolution in the panchromatic band and night imaging using IR payloads. Payload requirements are met by choosing Sun-Synchronous near Earth orbit of 300 Km x350 Km with 10:30 AM descending node local time. The satellite is designed with heritage from IMS-1 and Youthsat. This is a technology demonstrator and the fore runner for future satellites of this series. The satellite bus is modular in design and can be fabricated and tested independently of payload.

The interest and the enthusiasm created by the launch of many Nano satellites in a single launch by the PSLV, induced many students community from Colleges, Universities and Indian Institute of Technologies to involve in the development of many Micro and Nano satellites for various applications.

Indian Nano Satellite (INS) is a versatile and modular Nano-satellite bus system envisioned for future science and experimental payloads. With a capability to carry up to 3 kg of payload and a total satellite mass of 11 kg, it offers immense opportunities for future use. The INS system is developed as a co-passenger satellite to accompany bigger satellites on PSLV launch vehicle. Its primary objectives include providing a standard satellite bus for launch on demand services and providing opportunity to carry innovative payloads.

INS-1A and INS-1B: Nano Satellites offer a compelling alternative to large space projects, with a capability to support commercial, governmental, academic applications in a responsive and cost effective manner.Also, Nano Satellites have lot of potential in future and will give rise to lot of new developments in various technologies and are needed from strategic point of view in near future. In view of this, ISRO has developed and launched the first two ISRO Nano Satellites (INS) namely INS-1A and INS-1B on 15th February 2017.
INS-1A carried a twin payload package from SAC. Science objectives of INS-1A:

1. SBR: BRDF (Bidirectional Reflectance Distribution Function) of the Earth surface. To take readings of the reflectance of different surface features due to sun albedo
2. SEUM to monitor single event upsets (SEU) occurring due to high energy radiation in the space environment in COTS components.

INS-1B carried EELA payload from LEOS and Miniature Multispectral Payload Technology Demonstration (MMX-TD) payload from SAC.

1. EELA: Registration of terrestrial exospheric line-of-sight neutral atomic hydrogen Lyman-alpha flux. Estimation of interplanetary hydrogen Lyman-alpha background flux by means of deep space observations.
2. MMX-TD: Remote Sensing Colour camera with a novel lens assembly for optical realization in a small package. Scope for future scalability and utilization on regular satellites.

INS-1C: Indian Nano Satellite-1C is the third satellite in the Indian Nanosatellite series. INS-1C carries Miniature Multispectral Technology Demonstration (MMX-TD) Payload from Space Applications Centre (SAC). Data sent by this payload can be utilized for topographical mapping, vegetation monitoring, aerosol scattering studies and cloud studies.

1.7 University Satellites:

The STUDSAT, the first Nano satellite conceived and designed by 7 Engineering Colleges of Karnataka and Andhra Pradesh was successfully launched during July 2010.

JUGNU from IIT-Kanpur and SRMSAT were successfully launched on 12th October 2011. SATHYABAMASAT and SWAYAM were successfully launched on 22nd June 2016. PRATHAM and PISAT were successfully launched on 26th September 2016 and NIUSAT was successfully launched on 23rd June 2017.

Some more premier educational institutions of India are ready to get an opportunity for making satellites.

1.8 List of Satellites and Payloads

Table 1-1 List of Satellites and Payloads

Sl.No	Satellite Name	Payload	Abbreviation
1.	Bhaskara 1 , 2	SAMIR, TV Camera	**Sa**tellite **Mi**crowave **R**adiometer
2.	Rohini Satellites RTP, RS-1	LVMI	**L**aunch **V**ehicle **M**onitoring **I**nstruments
3.	IRS 1A & 1B	LISS-I, LISS-II	**L**inear **I**maging **S**elf **S**canner **-I** **L**inear **I**maging **S**elf **S**canner **-II**
4.	IRS-1E	LISS-I MEOSS	**L**inear **I**maging **S**elf **S**canner **-I** **M**onocular **E**lectro-**O**ptical **S**tereo **S**canner
5.	IRS-P2	LISS-II	**L**inear **I**maging **S**elf **S**canner **-II**
6.	IRS-P3	WiFS MOS IXAP	**Wi**de **F**ield **S**ensor **M**ultispectral **O**pto-Electronic **S**canner **I**ndian **X**-Ray **A**stronomy **P**ayload (IXAP)
7.	IRS 1C & 1D	PAN, LISS-III, WIFS	**Pan**chromatic **L**inear **I**maging **S**elf **S**canner **-III** **Wi**de **F**ield **S**ensor
8.	IRS-P4	OCM, MSMR	**O**cean **C**olour **M**onitor **M**ulti-frequency **S**canning **M**icrowave **R**adiometer
9.	TES	PAN	Panchromatic
10.	IRS-P6	LISS-IV, LISS-III, AWiFS	**L**inear **I**maging **S**elf **S**canner **-IV** **L**inear **I**maging **S**elf **S**canner **-III** **A**dvanced **Wi**de **F**ield **S**ensor
11.	Cartosat-1	PAN-AFT, Pan- Fore	Pan camera - Looking Forward Pan camera - Looking Forward
12.	Cartosat 2,2A,2B	PAN	Panchromatic
13.	IMS-1	Mx, HYSI	Multispectral Camera Hyper Spectral Camera
14.	Chandrayaan-1	TMC	Terrain Mapping Camera
		HySI	Hyper Spectral Imager (0.2u to 0.9u)
		LLRI	Lunar Laser Ranging Instrument (LLRI)
		HEX	High Energy X-ray payload (HEX)
		MIP	Moon Impact Probe(MIP)
		LEX	Low Energy X-ray (LEX) Payload (CIXS).
		MINISAR	Mini SAR from Applied Physics Laboratory (APL, USA
		SIR-2	SIR-2 from Max Plank Institute / ESA
		RaDoM	Radiation Dose monitor from Bulgarian Academy of sciences.

Sl.No	Satellite Name	Payload	Abbreviation
		SARA	Sub-KeV Atom Reflecting Analyser (SARA) Experimental developed jointly by IRE Sweden, SPL-VSCC India, ISAS/JAXA Japan and VBE Switzerland
		MMM	Moon Mineralogy Mapper (M3) from HJPL, USA
16	Oceansat-2	OCM, Scatterometer, ROSA	Ocean Colour Monitor Scatterometer Radio Occultation Sounder for Atmosphere
17	Resourcesat-2	LISS-IV, LISS-III, AWiFS HIP-1	**L**inear **I**maging **S**elf **S**canner **-IV** **L**inear **I**maging **S**elf **S**canner **-III** **A**dvanced **Wi**de **F**ield **S**ensor **H**osted **I**ndian **P**ayload-1
18	Youthsat	SOLRAD, LiHySI, RaBIT	**Sol**ar **Rad**iation Monitor Limb-View Hyper spectral Imager **Ra**dio **B**eacon for **I**onosphere **T**omography
19	Megha-Tropiques	MADRAS SAPHIR ScaRaB, ROSA	Microwave Analysis and Detection of Rain and Atmospheric Structures Soundeur Atmospherique du Profile d'Humidite Interopicale par Radiometrie Scanner for Radiation Budget Radio Occultation Sounder for Atmosphere
20	RISAT-1	C-Band SAR	Synthetic Aperture Radar
21	SARAL	ARGOS, ALTIKA SCBT	ARGOS Data Collection System Ka band Altimeter Solid State C-band Transponder
22	Mars Orbiter Mission	MENCA LAP TIS MSM MCC	Mars Exospheric Neutral Composition Explorer Lyman Alpha Photometer TIR Imaging spectrometer Methane Sensor for Mars Mars Colour Camera
23	Astrosat	LAXPC CZT SXT SSM UVIT CPM	Large area xenon-filled proportional counter Cadmium Zink Telluride Soft X-Ray imaging Telescope Scanning Sky Monitor Ultra Violet imaging telescope Charge Particle Monitor
24	Cartosat-2 Series 2C/2D/2E/2F	PAN Mx	Panchromatic Multispectral
25	Resourcesat-2A	LISS-IV	**L**inear **I**maging **S**elf **S**canner **-IV**

Sl.No	Satellite Name	Payload	Abbreviation
		LISS-III AWiFS A & B SCBT	**L**inear **I**maging **S**elf **S**canner **-III** **A**dvanced **Wi**de **F**ield **S**ensor **S**olid State **C-B**and **T**ransponder
26	Microsat	PAN MIR LWIR	Panchromatic Mid Infrared Long Wave Infrared

Table 1-2 Experimental Satellites

Satellites	Payloads	Altitude km	Inclination deg	Mass kg	Power W	Launch Vehicle	Launch Date
Bhaskara-1	TV Camera Micrometer	519 x 541	50.6	442	47	C1-Intercosmos	07-06-79
RTP	LV Monitor instruments	**Not achieved**		35	3	SLV-3	10-08-79
RS-1	LV Monitor instruments	305 x 919	44.7	35	16	SLV-3	18-07-80
RS-D1	Landmark tracker	186 x 418	46	38	16	SLV-3	31-05-81
Bhaskara-2	TV Camera micrometer	541 x 557	50.7	444	47	C1-Intercosmos	20-11-81
RS-D2	Smart sensor, L-Band Beacon	371 x 861	46	41.5	16	SLV-3	17-04-83
SROSS-1	GRB, MEOSS	**Not Achieved**		150	90	ASLV	24-03-87
SROSS-2	MEOSS	**Not Achieved**		150	90	ASLV	13-07-88
SROSS C	GRB RPA	267 x 391		106.1	45	ASLV	20-05-92
SROSS-C2	GRB RPA	430 x 600	45	115	45	ASLV	04-05-94
Youthsat	RaBIT LiVHySI SOLRAD	822±20	98.731	92	230	PSLV-C16	20-04-11
INS-1A	SBR SEUM	505	97.4	8.4	24	PSLV-C37	15-02-17
INS-1B	EELA Origami Camera	505	97.4	9.7	24	PSLV-C37	15-02-17
INS-1C	MMX-TD	583	97.7	11	27	PSLV-C40	12-01-18

Table 1-3 Operational Earth Observation Satellites

Satellites	Payloads	Orbit	Inclination	Local Time	Mass	Power	Launch Vehicle	Launch Date
IRS-1A	LISS-I, LISS-II	904	99.08	10.30	975	600	Vostak	17-03-88
IRS-1B	LISS-I, LISS-II	904	99.08	10.30	975	600	Vostak	29-08-91
IRS-1E	LISS-I MEOSS	**Not achieved**			846	415	PSLV-D1	04-05-94
IRS-P2	LISS-II	904	98.68	10.30	804	510	PSLV-D2	15-10-94
IRS-1C	LISS-3 PAN WiFS	817	98.09	10.30	1250	809	Molniya	28-12-95
IRS-P3	WiFS, MOS IXAE	817	98.68	10.30	920	817	PSLV-D3	21-03-96
IRS-1D	Liss-3 PAN WiFS	740 x 817	98.6	10.30	1250	809	PSLV-C1	27-09-97
IRS-P4 (Oceansat-1)	MSMR, OCM	720	98.28	12.00	1050	750	PSLV-C2	26-05-99
TES	PAN	572	--	9.30	1108	800	PSLV-C3	22-10-01
IRS-P6 Resourcesat-1	Liss-3 Liss-4 AWiFS	817	98.7	10.30	1360	1250	PSLV-C5	17-10-03
IRS-P5 (Cartosat-1)	PAN (Fore) PAN(Aft)	618	97.4	10.30	1560	1100	PSLV-C6	05-05-05
CartoSat-2	PAN	635	98.7	9.30	650	900	PSLV-C7	10-01-07
Cartosat-2A	PAN	635	98.7	9.30	690	900	PSLV-C9	28-04-08
IMS-1 (TWSat)	Imaging	635			83	220	PSLV-C9	28-04-08
Oceansat-2	OCM, Scatterometer ROSA	720	98.28	12.00	960	1360	PSLV-C14	23-09-09
Cartosat-2B	PAN	630	97.71	9.30	694	930	PSLV-C15	12-07-10
Resourcesat-2	Liss-3 Liss-4 AWiFS, COMDEV	822	98.731	10.30	1206	1250	PSLV-C16	20-04-11

Satellites	Payloads	Orbit	Inclination	Local Time	Mass	Power	Launch Vehicle	Launch Date
Megha Tropiques	MADRAS SHAPHIR SCARAB ROSA	867	20.00	---	1000	1325	PSLV-C18	12-10-11
RISAT-1	C-Band SAR	536	97.552	6AM/ 6PM	1858	2200	PSLV-C19	26-04-12
SARAL	ARGOS, ALTIKA SCBT	781	98.538	6PM	407	906	PSLV-C20	25-02-13
Cartosat-2 Series-2C	PAN Mx	505	97.48	9:30	737.5	986	PSLV-C34	22-06-16
Scatsat-1	Scatterometer	720	98.1	9:30	371	280	PSLV-C35	29-09-16
Resourcesat-2A	LISS-IV LISS-III AWiFS A & B SCBT	817	98.69	10:30	1235	1550	PSLV-C36	07-12-16
Cartosat-2 Series-2D	PAN Mx	505	97.46	9:30	714	986	PSLV-C37	15-02-17
Cartosat-2 Series-2E	PAN Mx	505	97.44	9:30	711	986	PSLV-C38	22-06-16
Cartosat-2 Series-2F	PAN Mx	505	97.47	9:30	710	986	PSLV-C40	12-01-18
Microsat	PAN MIR LWIR	353 x 376	96.877	10:30	120	286	PSLV-C40	12-01-18

Table 1-4 Lunar and Stellar Observation Satellites

Satellites	Payloads	Orbit	Mass	Power	Launch Vehicle	Launch Date
Chandrayaan-1	TMC, HySI, LLRI, HEX, MIP, CIXS, SIR-2, SARA, MINISAR, M3, RADOM	100 km x 100 km : Lunar Orbit	1380	700	PSLV-11	22-10-08
Mars Orbiter Mission	MENCA, LAP, TIS, MSM, MCC	Martian Orbit	1337	840	PSLV-C25	05-11-13

Astrosat	LAXPC, CZT, SXT, SSM, UVIT, CPM	650 km, 6 deg Inclin	1515	1900	PSLV-C30	28-09-15

Table 1-5 List of University / Academic Institute Satellites

Satellites	Orbit (km)	Mass (kg)	Launch Vehicle	Launch Date
ANUSAT	550	40	PSLV-C12	20-04-09
STUDSAT	630	>1	PSLV-C15	12-07-10
SRMSAT	867	10.9	PSLV-C18	12-10-11
JUGNU	867	3	PSLV-C18	12-10-11
SATHYABAMASAT	505	1.5	PSLV-C34	22-06-16
SWAYAM	505	1	PSLV-C34	22-06-16
PRATHAM	670	10	PSLV-C35	26-09-16
PISAT	670	5.25	PSLV-C35	26-09-16
NIUSAT	505	15	PSLV-C38	23-06-17

Feature of electro optical cameras in IRS Missions

	IRS-A/1B		IRS-1C/1D			IRS-P2	IRS-P3	IRS-P4	IRS- P6			TES	P5	C2
	LISS-1	LISS-II	LISS-III	PAN	WiFS	LISS-II	WiFS	OCM	LISS-III	LISS-IV	AWIFS	PAN		PAN
EFL (mm)	162.2	324.4	347, 301	980	56.47	324.4	56.47	20.0	347.5, 301	980	139.5, 181.35	3920	1945	5600
FOV(deg)	±4.7	±4.7	±4.5	±2.5	±13 X 2	±4.7	±13 X2	±43	±5	±2.5	±12.5 X2	±0.85	±1.3	±0.6
F/Number	4.5	4.5	4.3,4.2	4.5	6	4.5	4.5	4.3	4.5	4	5	7	4	7
Opt. Type	Refrac. 8 lens	Refrac 8.lens	Refrac. 8.Lens	Off-Axis	Refrac	Refrac	Refrac	Refrac	Refrac	Off-Axis	Refrac	RC	Off-Axis	RC
Spectral Band	B1,2,3,4	B1,2,3,4	B2,3,4,5	.5-.75	B3,4	B1,2,3,4	B3,4	#	B2,3,4,5	B2,3,4	B2,3,4,5	.5-.85	.5.85	.5-.85
IGFOV(mm)	72.5	36.25	23.5	5.8	188	32.74	188	360	23.5	5.8	56	1	2.54,2.19	0.8
Al.Tr.Res	72.5	36.25	23.5	5.8	188	32.74	188	352	23.5	5.8	56	1	2.54	
Ac.tr.Res	72.5	36.25	23.5	5.8	188	32.74	188	360	23.5	5.8	56	1	2.54,2.19	0.8
Swath km	148	145	141,148	70.5	810	2x67	770	1360	141,141	70.8	740	13.8	30,27	9.6
Repetivity	22	22	24	48	5	24	5	2	24	48	5		126	
# pixels	2048	2048	6000, 2100	3 x 4K	2048	2048	2048, 2100	3730	6000	12K	6000	4 x 4K	12K	2 X 12K
Pixel size(microns)	13 x 13	13 x 13	7 x 10, 26 x 26	7 x 7	13 x 13	13 x 13	13 x 13	7 x 10	7 x 10 13 x 13	7 x 7	7 x 10 13 x 13	7 x 7	7 x 7	7 x7
Quantisation	7	7	7	6	7	7	7	12	8,10	10>7	10	7	10	10
SWR(>)	40,40, 30,20	40,40, 30,20	40,40, 35,30	23	34,20	40,40, 30,20	34,20,20	34 TO 9	40,40 35,20	23	30,30,20,20	10	20	20
Int. time (ms)	1.2	5.6	3.6, 10.8	0.873	28.8	5.6	28.8	34.75	3.6.3.3	.877	9.96	.883	0.336	0.336
Power (W)	33 , 35	34 x 2	58	46	19.2	32, 34	50	134	75.3	48 x 3	111	92	<110 x2	<110
Mass_EO Electronics	27.5 38.42	70x2 165	76 117	105.5 124.5	18.8 26	73 98	25	64 142	75.3 106	94 169.5	29.5 x 2 103.6	146	224 267	

Spectral bands of OCM: 0.402-0.422, 0.433-0.453, 0.480-0.500, 0.660-0.680, 0.745-785, 0.660-0.680, 0.745-785, 0.845-0.88

Spectral Bands of Mx: B1: 0.45-0.52 B2: 0.52-0.59 B3: 0.62-0.68 B4: 0.77-0.86 B5: 1.55-1.7

Specification of Current High Resolution Satellites

Mission or Satellite	Ikonos-2	EROS A1	Quickbird-2	SPOT 5	Orbview-3	FormoSat-2 (Formely ROCSat-2)	IRS-P5 (Cartosat-1)	Corona (KH-4), many missions	KH-7, many missions	Cosmos[1], many missions
Sensor	OSA	PIC	BHRC60	HRG, HRS	OHRIS	RSI	2 PAN cameras	Stereo panoramic cameras	High resolution surveillance camera	KVR 1000 Panoramic camera (2 working alternatively)
Country	USA	Israel	USA	France, Belgium, Sweden	USA	Taiwan	India	USA	USA	Russia
System Type	Commercial	Commercial	Commercial	Commercial	Commercial	Commercial	Commercial	Military declassified	Military declassified	Commercial
Launch date or duration	9/1999	12/2000	10/2001	5/2002	6/2003	5/2004	5/2005	1960-1972	1963-1967	1981-2000
Sensor type	Digital	Digital	Digital	Digital	Digital	Digital	Digital	Film	Film	Film
PAN GSD (m) (across x along track)	1 (actually 0.82)	1.9, 1 or 1.4 (oversampled)	0.61	5 or 2.5-3 (oversampled) HRG 10 x 5 HRS	1	2	2.3	2-140	At nadir down to 0.45-0.5	2
PAN pixels of line CCD/ pixel spacing (µm)	13,816/ 12	7,043 (2 lines)/ ca. 13	27,568/ 12	12,000 (2 lines for HRG/ 6.5)	8,000/ 6 x 5.4, numbers shown here for 2 staggered line)	12,000/ 6.5	12,288 (x 2 staggered lines)/ 7	NA	NA	NA
Flying height (km), Focal length (m)	681, 10	Ca. 500, 3.4	450, 8.832	181-833, 1.082 HRG 0.58 HRS	470, 2.77	888, 2.896	618, 1.945	Variable, 0.6069	Variable, 0.96	Variable (190-270), 1
No. of MS Channels/ GSD (m)	4/4	0	4/2.44	(excl. Vegetation instrument) 4/ 10 and 20	4/4	4/8	0	0	Very few color & CIR images	0
Stereo[2]	Along-track, across-track	Along-track, across-track	Along-track, across-track	Along-track, across-track	Along-track, across-track	Along-track, across-track	Along-track	Along-track	Few images in stereo	No stereo
Swath width (km) or image film dimensions (cm)	11	14, 10 for oversampled images	16.5	60 HRG, 120 HRS	8	24	27	5.54 x 75.69 (across)	22.8 x variable (across)	18 x 72 (across)
Field of regard [3] (deg)	45, up to 60 deg images shot	45	45	27 (HRG, only across track)	50	45	23 (across)	NA	NA	NA

Mission or Satellite	Ikonos-2	EROS A1	Quickbird-2	SPOT 5	Orbview-3	FormoSat-2 (Formely ROCSat-2)	IRS-P5 (Cartosat-1)	Corona (KH-4), many missions	KH-7, many missions	Cosmos[1], many missions
TDI	Y	N, Asynchronous scanning	Y	N	N, asynchronous scanning equivalent to 10 TDI lines, and 4 variable integration times	N	N	NA	NA	NA
Along track triplette ability	Y	Y	?	N	?	?	N	N	N	NA
Body rotation angular rate [4] (deg/sec)	Up to > 1	1.8	0.5-1.1	NA	?	0.4-.075	?	NA	NA	NA
FOV (deg) or film area coverage	0.93	1.5	2.12	4.13 HRG 7.7 HRG	0.97	1.54	2.49	14 x 189 km (typical)		40 x 160 km (typical)
Quantization bits	11	11	11	8	11	12	10	NA	NA	NA
Scale Factor	68,100	145,000	51,100	762,500 HRG, 1,422, 500 HRG	170,000	307,000	312,000	Variable, ca. 250,000 typical	Variable	190,000-270,000
Stereo overlap (%)	Up to 100	Up to 100	Up to 100	Up to 100	Up to 100	Up to 100	Up to 100	Up to 100	?	6-12
B/H ratio	Variable	Variable	Variable	Up to 1.1 HRG, 0.8 (40 deg) HRS	Variable	Variable	0.62 (31 deg)	0.60 (30 deg)	?	NA

Specification of Radar Satellites

Satellite	Country	Year	Band	Frequency (GHz)	Wavelength (cm)	Incident Angle (deg)	Polarization	Pulse Bandwidth (MHz)/ (Range Resolution (m))	Azimuth Resolution (m)/ (looks)
SEASAT	USA	1978	L-band	1.275	23.5	23	HH	19/ (7.9)	6/ (1)
SIR-A	USA	1981	L-band	1.275	23.5	50	HH	6/ (24.9)	6.5/ (1)
SIR-B	USA	1984	L-band	1.275	23.5	15-65	HH	12/ (12.5)	6/ (1)
ERS-1/2	Europe	1991/95	C-band	5.25	5.7	23	VV	15.5/ (9.7)	25/ (3)
ALMAZ	USSR	1991	S-band	3.0	10	30-60	HH	-/ 15	15/ (2)
JERS-1	Japan	1992	L-band	1.275	23.5	39	HH	15/ (10)	30/ (4)
SIR-C/ X-SAR	USA Germany	 1994	L-band C-band X-band	1.25 5.3 9.6	23.5 5.7 3	15-55 54	HH, HV, VH, VV VV	10/ (15) 20/ (7.5)	7.5/ (1) 6/ (1)
RADARSAT-1	Canada	1995	C-band	5.3	5.7	20-50	HH	11.6/ (12.9) 17.3/ (8.6) 30/ (5)	28/ (4) 50/ (2-4) 100/ (4-8)
SRTM	USA Germany	2000	C-band X-band	5.25 9.6	5.7 3	54 54	HH, VV VV	20/ (7.5) 8/ (18.7)	15/ (1) 8-12/ (1)
ENVISAT	Europe	2002	C-band	5.25	5.7	15-45	IIII, IIV, VH, VV	9/ (16.6)	6/ (1) 150/ (12) 1000/ (18-21)

2. Experimental Satellites

2.1 Bhaskara-1 & 2

2.1.1 Introduction

The dream of making remote sensing satellite by India came to true through the BHASKARA-I mission. It is the first experimental remote sensing satellite built by India and launched in 7th June 1979. This was followed by a follow-on mission BHASKARA-II with some modification and was launched on 20th November 1981. The technologies developed for ARYABHATA were used in BHASKARA with some improvements such as spin rate and spin axis control, using Infrared Horizon sensor for earth reference and high bit rate telemetry for payload data transmission. These missions provided a system experience on End-to-End basis, configure, design, develop, and assemble a satellite for remote sensing to reception and processing of remotely sensed data and generation of data products as per user requirement.

2.1.2 Mission Objectives

The primary objectives of Bhaskara-I mission were:

- To conduct earth observation experiments that would yield useful data in the areas of metrology, hydrology and forestry using a two band TV camera system operating in the 0.54 to 0.66 microns visible band and 0.75 to 0.85 micron near Infrared band (The earth imagery obtained from an altitude of 525 km provides a spatial resolution of 1 km X 1 km in a picture frame of 341 x 341 km)
- To conduct ocean surface studies using a three chain radiometer operating at microwave frequencies.
- To evolve the methodology of reception, processing and dissemination of data and thus establish visibility of management of earth resources through remote sensing satellites.

The secondary objectives are

- To develop the technology for relaying data collected from the unattended platforms to a central receiving station to obtain useful meteorological data on an experimental scale, from presently inaccessible regions at short turn-around times and thus develop the expertise and infrastructure for large scale applications of automatic data collection platforms.
- To study the performance of indigenously developed solar cells, thermal paints and heat pipe under prolonged exposure to space environments.
- To study the time variations of celestial X-ray sources and detect transient sources.

The Basic objective of the Bhaskara – II was to provide continuity to the Bhaskara-I experiment.

2.1.3 Orbit Details

Bhaskara I & II comparison

Table 2-1: Orbit details of Bhaskara Satellites

Parameter	Bhaskara-I	Bhaskara-II
Mass	442 kg	444 kg
Power	47W	47W
Altitude	519 x 541 km	541 x 557 km
Eccentricity deg.	0.0023	0.002459
Orbit	Near circular	Near circular
Inclination deg.	50.6	50.7
Stabilization	Spin Stabilization	Spin Stabilization
Launch Date	June 7, 1979	Nov. 20 1981
Orbital Time	95.2 min.	95.2 min.

2.1.4 Salient features of Bhaskara-I & II Systems

The Bhaskara-I project, originally known as the Satellite for Earth Observation (SEO) was conceived as one of the key intermediate steps towards going for a full-fledged operational remote sensing satellite system for India.

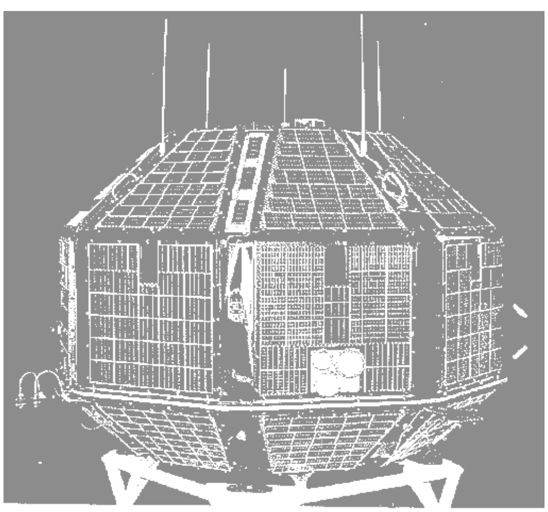


Figure 2-1 View of Bhaskara

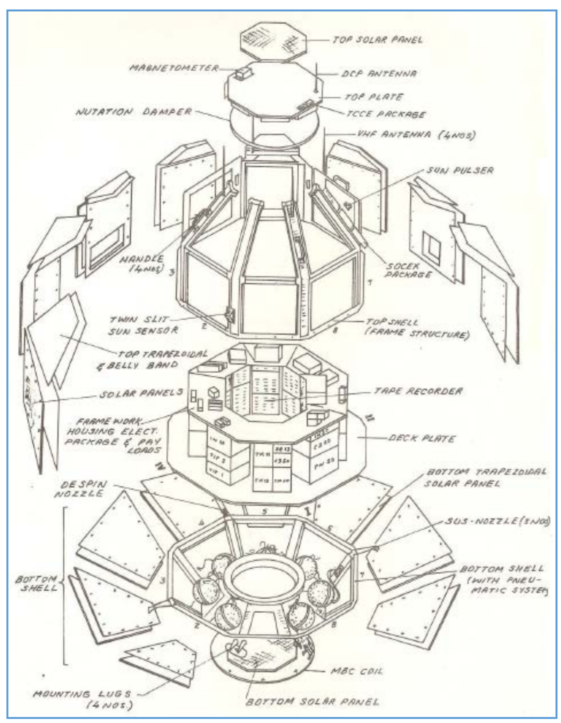


Figure 2-2 Exploded view of Bhaskara

Table 2-2 Features of Bhaskara Systems

Parameter	Values
Overall height including antenna	1559 mm
Weight	442 kg (Bhaskara-1) / 444 kg (Bhaskara-2)
Primary Payloads	TV slow scan Videocon cameras operating in the visible band 0.54 – 0.66 microns and IR Band 0.75 to 0.85 microns Microwave Radiometers operating at 19 and 22 GHz (in Bhaskara II, 32 GHz chain included)
Satellite Main Frame Power system TTC Uplink TTC Downlink LBT (224 Bits/Sec) TTC downlink HBT (91 Kbs) Tracking System	 Body mounted solar panels backed by 10 AH NiCd Battery PCM/FM/AM at 148.25 MHz. PCM/FM/AM at 137.20 MHz. PCM/BSK at 137.20 MHz 148.61 MHz uplink; 137.1MHz downlink with tones 32 Hz to 20 KHz.
AOCS Type of stabilization Spin Rate range Spin axis orientation Attitude determination	 Cold gas jet –spin stabilization 6 to 11 RPM Within 3 deg of orbit normal Within 1 deg.

2.1.5 Payloads

In an ideal case an earth observation satellite system should be three axes stabilized so that the sensors can point towards the earth continuously. However to gain time, the satellite configuration of the first Indian satellite ARYABHATTA was adopted and images were acquired while spacecraft was spinning. 'BHASKARA-I' was launched on 7th June 1979 from a Soviet Cosmodrome in a near circular orbit of mean altitude 534 km and inclination 50 deg. The 444 kg satellite carried two major remote sensing payloads namely a two band TV camera system and a two frequency microwave radiometer. The satellite was spin 21ealized21d with its spin axis maintained at right angles to the orbital plane. Such configuration enables both the payloads to 'look' along the local vertical once during every spin. 'Bhaskara-II' was launched on November 20, 1981. It is an improved version of 'Bhaskara-I' having **a three frequency radiometer** to enable differentiation between water vapour and liquid water in the atmosphere.

2.1.5.1 Multispectral TV Camera

Bhaskara TV payload system consists of two TV cameras, one operating in the 0.54 to 0.66 micron band and the other in the 0.75 to 0.85 micron band. Each picture frame covers an area of 340 km x 340 km with a ground resolution of about one km and a typical over lap of 10% between successive picture frames. The built in marks and in flight radiometric calibration help in producing geometrically and radio-metrically corrected picture on the ground. The camera is mounted on the spacecraft with its optical axis at the right angles to the spin axis. The cameras are exposed at the instant when the optical axis of the camera points to the local vertical. Read out takes place at a slower rate commensurate with the telemetry capability of the spacecraft. The basic sensor of the TV camera payload is a Super Videocon Camera Tube consisting of an image intensifier with a gating facility coupled to a storage type 22ealized tube. A specially designed multi-element lens is used for each camera. The focal length of the lens and the active face plate area together decide the field of view of the camera. A summary of the camera specifications is given in Table 2-2.

Table 2-2:TV Camera paylaod specifications

Parameter	Value
Sensor Type	Slow scan vidicon coupled to an image intensifier
Imaging lens	F/no.1.9, Focal length 18.46 mm, FOV 49.37°
Spectral channels	Camera-1 0.54 – 0.66 microns Camera-2 0.75 – 0.85 microns
Picture Frame	340 x 340 km^2 for a 525 km altitude
Ground Resolution	About 1 km
Exposure control	1, 1.5, 2 ms selectable by gorund command
Power	22.5 W average

The system can be put in 'calibration mode' by ground command. The mechanical shutters do not operate during the calibration mode and the tube face plate is illuminated by flashing an LED source. In one calibration cycle, the cameras are exposed to four different intensity levels one of which is zero illumination and other three are spread out, over the dynamic range. This calibration cycle then repeats itself during the calibration mode. The exposure duration can be changed by ground command to get additional calibration levels.

The initial 'switch on' of BHASKARA-I TV camera payload was not successful. Extensive ground simulation studies indicated that the anomalous behavior during the switch on of the TV camera was due to a corona discharge in the high voltage section of the payload. Poor adhesion of the potting compound with the high voltage standoff, coupled with trapped air, caused the corona. With time, the trapped air leaked out and camera-l was switched on successfully on May 16, 1980.

BHASKARA-II payload was suitably modified to take care of the problems observed in BHASKARA-I and the camera performance was satisfactory in BHASKARA-II. The imagery received from both bands was comparable in quality to any other imagery of similar resolution. Multiband imagery from the TV payload has been received over the complete Indian subcontinent. The multiband imagery received from BHASKAR A-II has been used to demonstrate various applications in the field of geology, hydrology, and forestry.

2.1.5.2 Satellite Microwave Radiometer (SAMIR)

The SAMIR system of BHASKARA-I consisted of three independent channels operating at 19.1, 19.6 and 22.235 GHz frequency bands. Each channel contains a scalar horn antenna, dicke switch, mixer/preamplifier, square law detector, suitable D.C. amplifiers, and telemetry interface circuits. In the case of BHASKARA-II one of 19 GHz channels has been replaced by a channel at 31.4 GHz.

In BHASKARA-I the spatial resolution of the 19 GHz radiometer was 150 km and the spatial resolution of the 22 GHz radiometer was 230 km respectively. In BHASKARA-II all the three radiometers had same spatial resolution of 125 km. Broad specifications of the radiometers are given in Table 2-3.

Table 2-3:Specificatios of SAMIR

	BHASKARA-I	BHASKARA-II
Frequency (GHz)	19.1 (R1) 19.6 (R2) 22.235 (R3)	31.4 (R1) 19.35 (R2) 22.235 (R3)
System Noise Figure (dB)	6.5 (R1) 6.5 (R2) 7.5 (R3)	8.5 (R1) 6.5 (R2) 7.5 (R3)
Predetection bandwidth(MHz)	100	100
Integration Time Constant (ms)	350 (R & R2) 470 (R3)	300 (All)
Spatial resolution (km)	150 (R1 & R2)	125 (All)
Brightness Temperature range (°K)	4 – 320	4- 320
Temperature sensitivity (°K)	1	1

The SAMIR system can be operated in two possible modes, depending upon the spin-axis orientation. In the 'Normal Mode' the spin axis of the satellite *is* normal to the orbital plane and hence the antenna would scan along the satellite track. In the 'Alternate Mode' the spin axis of the satellite would lie in the orbital plane, tangential to the orbit at a certain latitude, thus converting the radiometers effectively into a scanning system. In the 'Alternate Mode' data will be sampled at fourteen different angular positions and the effective coverage during each orbit will be around 1000 km with a 125 km ground resolution at nadir.

Analogue data from all the channels *is* sampled at various angular positions around nadir, depending upon the mode of operation. As the data acquisition and telemetry transfer rate are not synchronous, data is held in various sample and hold circuits, till it is transferred to the satellite data stream.

Various tests conducted during the initial phase operations and operational phase have confirmed the consistent performance of SAMIR Radiometers onboard BHASKARA-I & **II.** SAMIR data was used for a number of meteorological applications. These include estimation of water vapor and liquid water content, rain fall estimation over ocean area, estimation of wind speed over ocean, study of floods etc.

After realizing the mission objectives the Bhaskara-II mission was decommissioned in March 1981.

2.2 Rohini Satellite Series and Stretched Rohini Series (SROSS) Satellites

2.2.1 Introduction

The Rohini satellites were launched with various remote sensing payloads for X-ray observations and as payloads for the SLV launch vehicles which were under development.

2.2.2 Rohini Satellite Series

Rohini Satellite Series had four satellites of 35 kg class namely

- RTP (Rohini Test Project)
- RS
- RS-D1
- RS- D2

2.2.2.1 RTP – Rohini Test Project

RTP carried Launch vehicle monitor equipments. The mass of the satellite was 35 kg. It was launched on 10-08-79. **Launch Failed.**

RTP (Rohini Test Project)	Mission	Experimental
	Weight	35 kg
	onboard power	3 Watts
	Communication	VHF band
	Stabilization	Spin stabilized (spin axis controlled)
	Payload	Launch vehicle monitoring instruments
	Launch date	August 10,1979
	Launch site	SHAR, Sriharikota, India
	Launch vehicle	SLV-3
	Orbit	Not achieved

2.2.2.2 RS-1

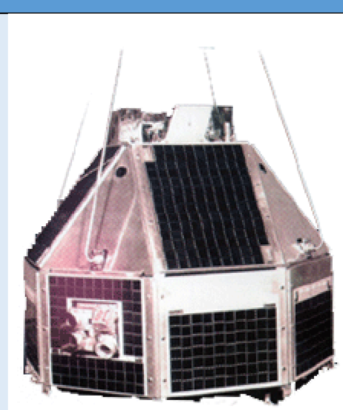
Mission Objectives: To monitor the launch vehicle performance.

RS-1	Mission	Experimental
	Weight	35 kg

onboard power	16 Watts
Communication	VHF band
Stabilization	Spin stabilized
Payload	Launch vehicle monitoring instruments
Launch date	July 18,1980
Launch site	SHAR, Sriharikota, India
Launch vehicle	SLV-3
Orbit	305 x 919 km
Inclination	44.7 deg.
Mission life	1.2 years
Orbital life	20 months

2.2.2.3 RS-D1

Mission Objectives: Carried a Land Mark sensor payload whose solid state camera performed to specifications. The satellite re-entered the earth's atmosphere nine days after launch on account of the launch vehicle's injecting the satellite into a lower than expected altitude.

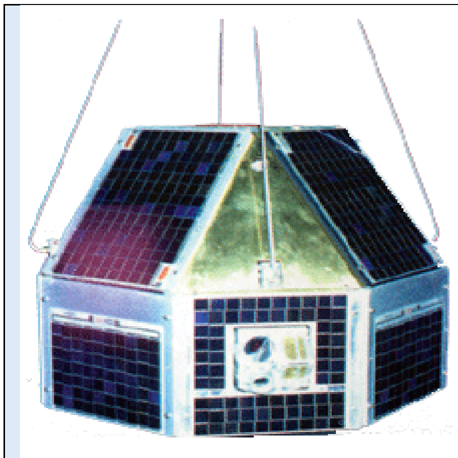


RS-D1	Mission	Experimental
	Weight	38 kg
	onboard power	16 Watts
	Communication	VHF band
	Stabilization	Spin stabilized
	Payload	Landmark Tracker (Remote sensing payload)
	Launch date	May 31,1981
	Launch site	SHAR Centre, Sriharikota
	Launch vehicle	SLV-3
	Orbit	186 x 418 km (achieved)
	Inclination	46 deg.
	Orbital life	9 Days

2.2.2.4 RS-D2

Mission Objective: The Smart Sensor Camera was the primary payload on board the satellite. It was operated for over five months and sent more than 2500 pictures frames in both visible and infrared bands for identification of landmarks and altitude and orbit refinement. The camera had on-board processing capability to use the data for classifying ground features like water, vegetation, bare land, clouds and snow. After completing all its mission goals, RS-D2 was closed down on Sept. 24, 1984.

RS-D2	Mission	Experimental



Weight	41.5 kg
Onboard power	16 Watts
Communication	VHF band
Stabilization	Spin stabilized
Payload	Smart sensor (remote sensing payload), L-band beacon
Launch date	April 17, 1983
Launch site	SHAR Centre, Sriharikota, India
Launch vehicle	SLV-3
Orbit	371 x 861 km
Inclination	46^{o}
Mission life	17 months
Orbital life	Seven years (Re-entered on April 19, 1990)

2.2.2.5 Payload Smart Sensor Onboard ROHINI Satellite

Rohini series of satellites are launched by the Indian launch vehicle SLV-3. A two band solid state camera was designed for Rohini Satellite. A 256 element photo diode array is used as the basic detector. The satellite is spin 26ealized26d with spin axis normal to the orbital plane. During each spin the camera scans the earth approximately ±4.5° to the local vertical producing 80 scan lines thereby generating a picture frame of 250 km x 80 km. The image resolution is about 1 km from 500 km orbit.

One of the unique features of the camera is that it is capable of carrying out limited feature identification onboard. This is realized by taking the ratio of the 2 band output and having a decision circuitry to discriminate between the different classes based on rationing. The feature identification code and video information from anyone of the cameras is transmitted.

Table 2-4:Rohini Smart Sensor Specifications

Parameter	Value
Resolution	1 km(Nominal)
Spectral bands Channel-1 Channel-2	 0.65±0.05 microns 0.85±0.05 microns
Swath	25 km
Overlap	30%
Optics size	Focal length 25mm, f/1.4 system
Memory	140 kbits
Power	4 watts
Weight	3 kg

The Rohini satellite carrying this payload was launched on April 17, 1983 from the Indian launch station at Sriharikota. The camera functioned normally as planned and it was possible to establish the possibility of limited feature identification on board. Water bodies, biomass, bare land and clouds can be easily identified with onboard processing.

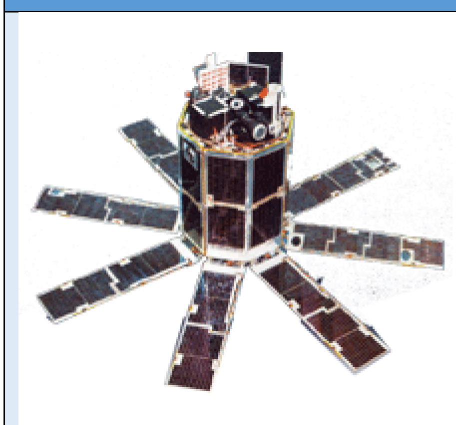
2.2.3 Stretched Rohini Satellite series

Stretched Rohini Satellite series had four satellites of 150 kg class namely

- SROSS-1
- SROSS-2
- SROSS-C1
- SROSS-C2

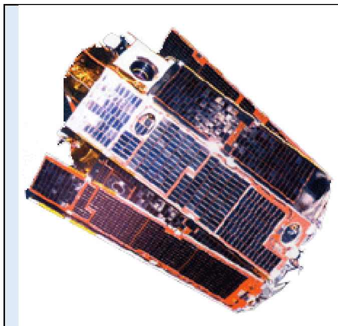
2.2.3.1 SROSS-1

The satellite was launched onboard the first developmental a flight of ASLV. It did not reach the orbit.

SROSS-1	Mission	Experimental
	Weight	150 kg
	Onboard power	90 Watts
	Communication	S-band and VHF
	Stabilization	Three axis body stabilized (biased momentum) with a Momentum Wheel and Magnetic Torquer
	Propulsion system	Monopropellant (Hydrazine based) Reaction control system
	Payload	Launch Vehicle Monitoring Platform(LVMP), Gamma Ray Burst (GRB) payload and Corner Cube Retro Reflector (CCRR) for laser tracking
	Launch date	March 24, 1987
	Launch site	SHAR Centre, Sriharikota, India
	Launch vehicle	Augmented Satellite Launch Vehicle (ASLV)
	Orbital life	Not 27ealized
	Mission	Experimental
	Weight	150 kg

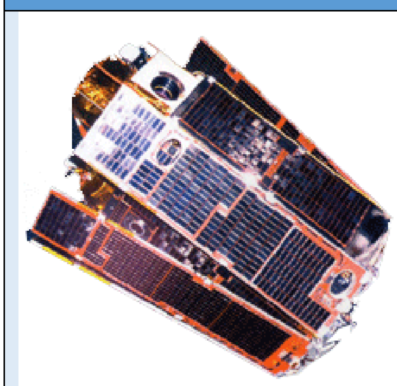
2.2.3.2 SROSS-2

SROSS-2	Mission	Experimental



	Weight	150 kg
	Onboard power	90 Watts
	Communication	S-band and VHF
	Stabilization	Three axis body stabilized (biased momentum) with a Momentum Wheel and Magnetic Torquer
	Propulsion system	Monopropellant (Hydrazine based) Reaction Control System
	Payload	Gamma Ray Burst (GRB) payload and Mono-ocular Electro-Optic Stereo Scanner (MEOSS) built by DLR, Germany
	Launch date	July 13, 1988
	Launch site	SHAR Centre, Sriharikota, India
	Launch vehicle	Augmented Satellite Launch Vehicle (ASLV)
	Orbit	Not realised

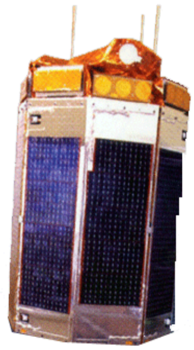
2.2.3.3 SROSS-C



SROSS-C	Mission	Experimental
	Weight	106.1 kg
	Onboard power	45 Watts
	Communication	S-band and VHF
	Stabilization	Spin stabilized with a Magnetic Torquer and Magnetic Bias Control
	Payload	Gamma Ray Burst (GRB) experiment & Retarding Potential Analyser (RPA) experiment
	Launch date	May 20,1992
	Launch site	SHAR Centre, Sriharikota, India
	Launch vehicle	Augmented Satellite Launch Vehicle (ASLV)
	Orbit	267 x 391 km
	Mission life	Two months (Re-entered on July15,1992)

2.2.3.4 SROSS-C2

SROSS_C2	Mission	Experimental
	Weight	115 kg
	Onboard power	45 Watts
	Communication	S-band and VHF



RCS	Monopropellant Hydrazine based with six 1N thrusters
Payload	Gamma Ray Burst (GRB) & Retarding Potential Analyser (RPA)
Launch date	May 04,1994
Launch site	SHAR Centre, Sriharikota, India
Launch vehicle	Augmented Satellite Launch Vehicle (ASLV)
Orbit	430 x 600 km.
Inclination	45 deg.

Configuration of SROSS C2 is SROSS C1

2.2.3.5 Mission Objectives

- To monitor celestial gamma ray bursts in the energy range 20-3000 KeV
- To measure temporal variations of gamma ray burst with high time resolution (2 ms, 16ms and 2556 ms) to search for periodicities in the emitted radiation.
- To measure temporal evolution of burst energy spectra to search for cyclotron lines and features in the energy range 20-100 KeV and possible red-shifted annihilation radiation in the energy range 400-500 KeV.
- To study the characteristic features of the thermal structure of the equatorial and low latitude ionosphere
- To study the effect of magnetic storms and Spread-F on thermal structure
- To Study the behavior of electron density anomaly in the low latitude region

2.2.3.6 Salient features of SROSS-C

Sub system		Features
Structure		Aluminum frames and honeycomb decks
Thermal		Using passive elements and 2W heaters. Temp. limit is 0 to 40 deg.
Power	Solar Panel	Four units of body mounted panels (each unit with two panels bonded at 135 deg. Along the vertical edge.
	Battery	18 AH Ni-Cd cells connected in series.
	Electronics	DC/DC converter, Under/Over volt detector circuits, E/N logic Battery Voltage control Logic
TTC	Telemetry	Format-1 mode and Dwell mode with reentrancy. PROM based main system and microprocessor based (RCA 1802) Redt. system 256 BPS, PCM/PSK/PM, 2245.68 MHz

	Telecommand	Microprocessor (RCA 1802) Based Main Decoder and Hardware redt. Decoder Unit.100 bps, S-Band PCM/FSK/FM/PM, 2067.897 MHz
AOCS	Sensors	Magnetometer, Twinslit Sun sensor
	Stabilisation	Spin stabilisation
Mass		106.1 kg

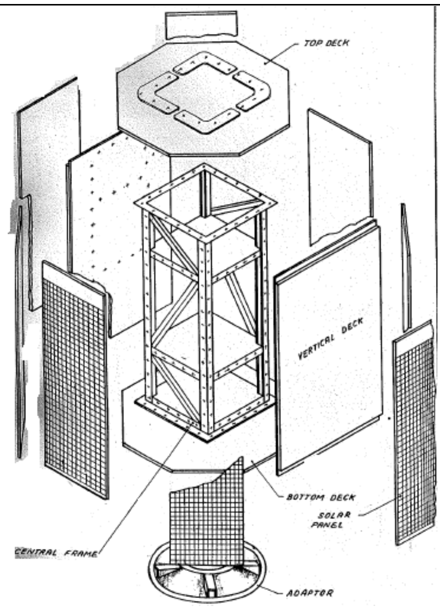
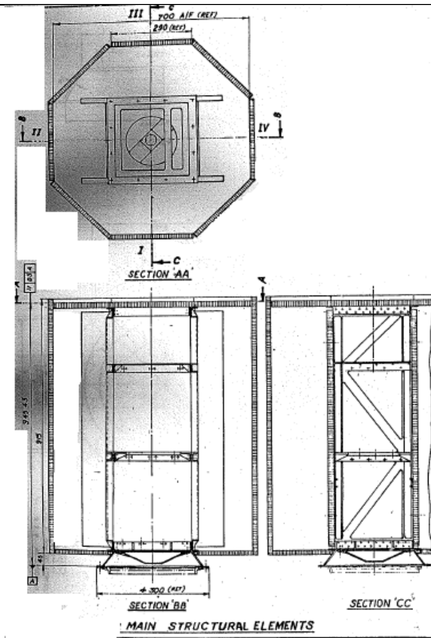
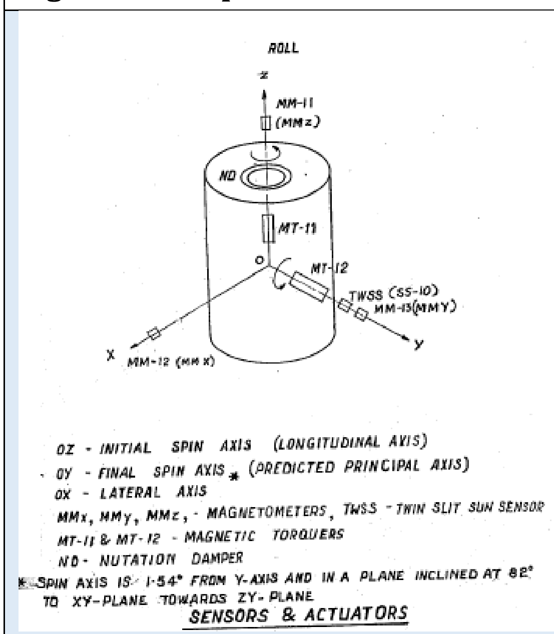


Figure 2-3: Exploded view of SROSS-C1





SROSS-C1 Sensors & Actuators

2.2.3.7 Salient features of Payloads

Payload consists of two sensors namely

- Gamma Ray Burst Detection Experiment
- Retarding Potential Analyser Payload

2.2.3.7.1 Gamma Ray Burst detector

The gamma ray burst payload consists of a main and a redundant scintillation detector viewed by separate photomultiplier tubes and powered by independent high voltage supplier. A common microprocessor based (RCA CDP1802) electronics system process the signals from either of the detectors. The main detector consists of a 76 mm diameter and 12,5mm thick CsI(Na) scintillator optically coupled to an EMI 9758 NA PMT. The redundant detector is also CsI(Na) crystal and has a diameter of 38mm and a thickness of 12.5 mm. It is viewed by RCA 7151Q tube. The scintillator is coupled to the PMT. By means of DC-93-500 potting compound.

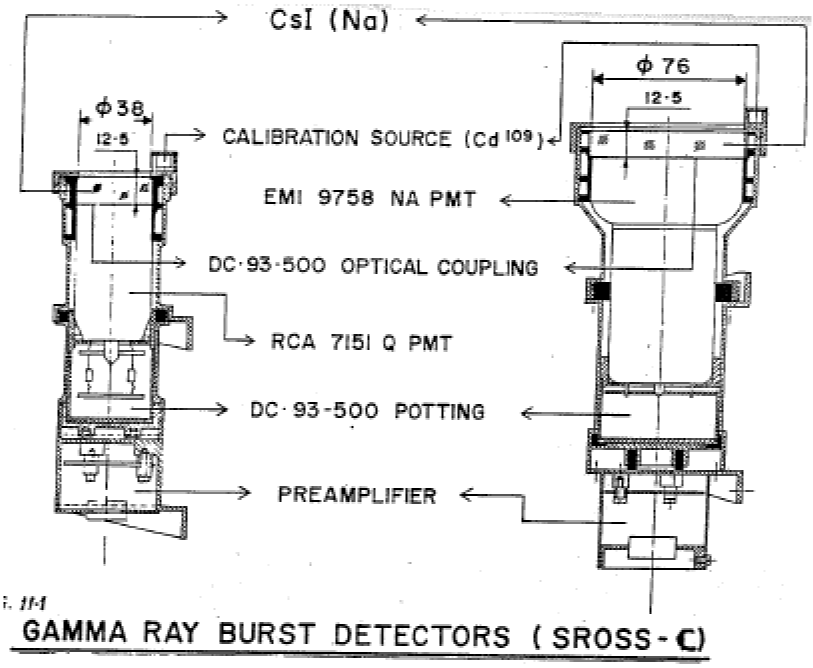


Figure 2-4 Gamma Ray Burst detector

2.2.3.7.2 Retarding Potential Analyser (RPA)

The RPA experiment is proposed to investigate the characteristics and energies of the equatorial and low latitude ionosphere and thermosphere which is an important element in understanding the sun-earth relationship and the effects of dynamics, turbulence and storms on the thermal behavior. It indents to study characteristics variation of electron density over the equator and around it (±15 deg latitude). This involves measuring plasma parameters like density and temperature to characterize the ionosphere. It also indented to identify understand and estimate various energy deposition and loss process to derive thermospheric temperature.

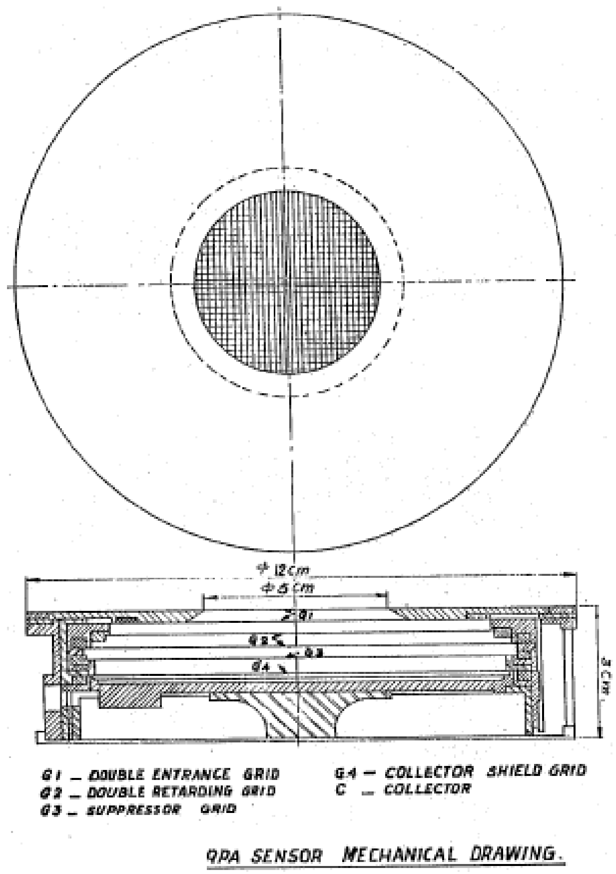


Figure 2-5 Retarding Potential Analyser (RPA)

3. Land and Water Resource Observation Series

3.1 IRS-1A & 1B

3.1.1 Introduction

The successful launch and operation of Bhaskara-I and II satellites provided experience in setting up of ground-based data reception and processing systems, gaining experience in over-all mission management, receiving data from other satellites like LANDSAT and activities related to data analysis, interpretation & utilization.

The experience gained in conceptualisation and implementing a space segment with necessary ground based data reception, processing and interpretation system, and integrating the satellite based remote-sensed data with conventional data systems for resource management, provided a way for a programme for operationalising the remote-sensing system for the country. The evolution of National Natural Resources Management System is the outcome of all the above efforts and IRS-1A mission is the first step in such an operational resources management system for the country.

IRS-1A, the first of the series of indigenous state-of-art operating remote sensing satellites, was successfully launched into a polar sun-synchronous orbit on March 17, 1988 from the Soviet Cosmodrome at Baikonur. *IRS*-1A carried two cameras, LISS-I and LISS-II with resolutions of 73 metres and 36.25 metres respectively with a swath of about 140 km during each pass over the country. Mission completed during July 1996 after serving for 8 years and 4 months.

IRS-1B, with some improved features compared to its predecessor like gyro referencing for better orientation sensing, time tagged commanding (IRS-1A) facility for more flexibility in camera operation and line count information for better data product generation, was launched on 29.08.1991. Mission completed on December 20, 2003 after serving for 12 years and 4 months.

3.1.2 Mission Objectives

The main objectives of IRS-1A mission are

- To design, develop and deploy a three axis stabilised polar sun- syncronise satellite carrying near state-of-art-multiple solid state pushbroom cameras operating in visible and near infrared bands for aquiring imageries for each resources applications on an operational basis.
- To establish and routinely operate ground based systems for spacecraft data reception, recording, processing, generation of data products, analysis and archival as well as mission control facilities.
- To use the data from IRS in conjunction with supplementary/ complementary information from other sources for survey and management of resources in imporatnt areas such as agriculture, geology and hydrology in association with the user agencies,

that will additionally enable characterisation of a future operational system for the country at the optimum level.

3.1.3 Orbit Details

IRS-1A was launched into a polar sun synchronous orbit at an altitude of 904 km. In the sun-syncronous orbit, the orbit plane rotates at the same rate as the mean rotation rate of the earth around the sun (0.9856 deg/day). Thus the satellite passes over a particular latitude approximately at the same local time. It enables the ground illumination conditions at sub-satellite regions to be constatnt throughout the mission. The equatorial crossing time of the descending node for IRS-1A is around 10.25 AM.

As the orbital period of IRS-1A is nearly 103 minues, with the satellite completing 14 orbits/day, each successive orbit is shifted westward over earth's surface by 25.798 degree of longitude, corresponding to 2872 km at equator. The satellite's path is shifted by 1.17 deg longitude to the west every day corresponding to 130.84 km at the equator. The satellite completes one coverage cycle of the Indian subcontinent in 22 days(307 orbits)

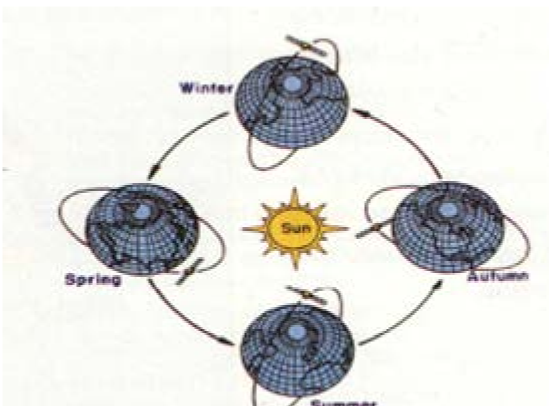


Figure 3-2 Sun Synchronous orbit

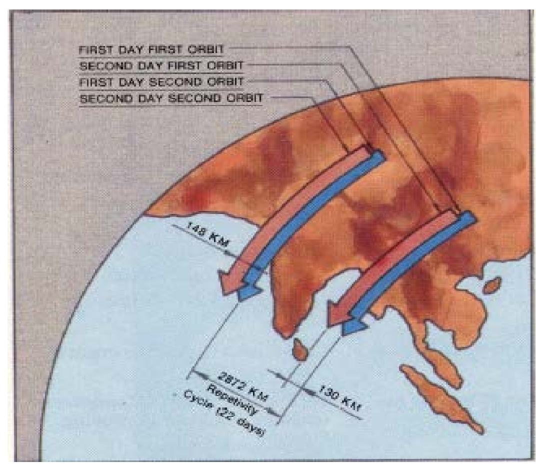


Figure 3-1Swath coverage

Table 3-1:Orbit details

Parameters	IRS-1A	IRS-1B
Altitude	904 km	904 km
Inclination	99.028 deg	99.028 deg
Eccentricity	0.008	0.008
Equatorial Crossing Time	10.30 A.M	10.30 AM
Orbital period	103.192 min.	103.192 min.
Recurrence	14 orbits/day	14 orbits/day
Repetition cycle	22 days (307 orbits)	22 days (307 orbits)
Daily shift at equator	130.54 km westward (1.17 deg)	130.54 km westward (1.17 deg)

Node	Descending	Descending

3.1.4 Salient Features of Spacecrafts

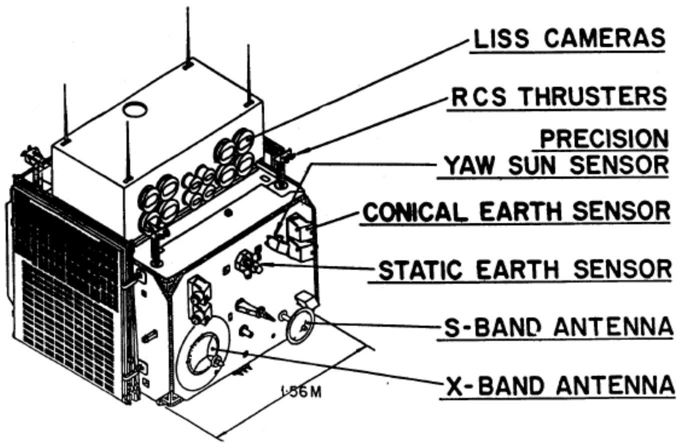


Figure 3-3 Stowed mode view of IRS-1A

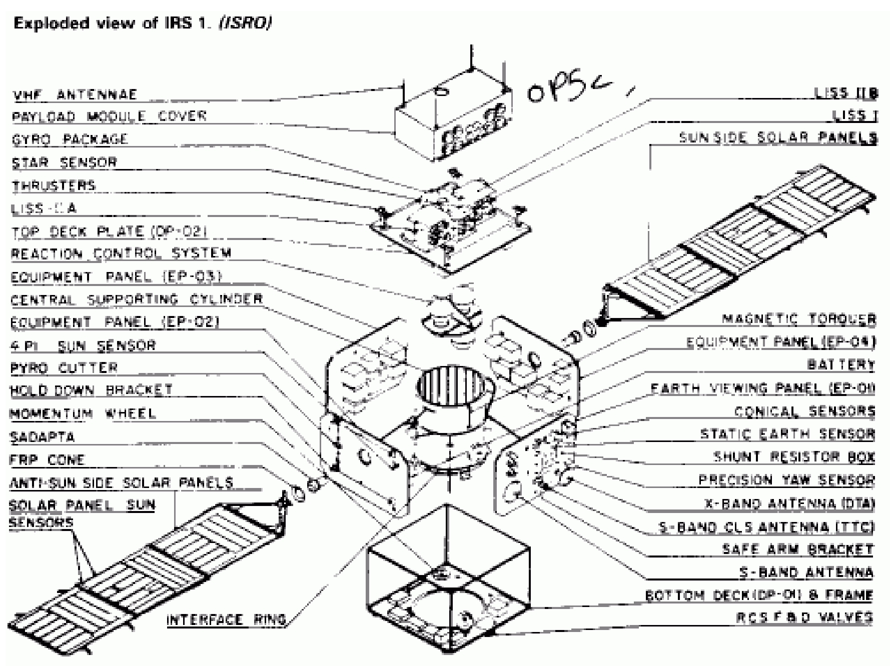


Figure 3-4Exploded view of IRS-1A

Table 3-2:Salient features of IRS-1A & 1B

Parameter		IRS-1A	IRS-1B

Mission		Launched as experimental satellite later declared as Operational.	
Mass		975 kg	
Structure		Aluminium and aluminium honeycomb structure	
Thermal	Components	Passive control using tapes , OSR, MLI Blankets and semi-active/active control using proportionate temperature controller and heaters	
	Temp. Range	20±5 deg.C range for imaging sensors electro-optics 5±5 deg. C for Chemical Batteries 0 to 40 deg.C for electronic packages	
Power	Solar Array	8.5 m^2 area, deployable and sun tracking panel, Power generation at EOL is 620 Watts (Totally 6 panels)	
	Battery	Two Ni-Cd batteries of 40 AH capacity each	
Communication	Telemetry	House Keeping(HK) information in S-Band; PCM/PSK Real Time rate 256 bps and play back rate 4 Kbps; Onboard storage capacity of 98 minutes of HK data	
	Telecommand	S-Band : PCM/PSK/FM/PM, and VHF : PCM/FSK/AM	
	Tracking	Facility for ON/OFF and Data commands S-Band tone ranging and two way Doppler	
Attitude and Orbit Control (AOCS)	Attitude sensors	IR Horizon sensors(Conical and Static), Star sensor, sun sensors, Dynamically Tuned Gyros (DTG)	
	Attitude control	Reaction Wheel(three 5NMS and one 10 NMS), Magnetic torquers, Hydrazine thrusters(16 1N)	
	Orbit Control	Monopropellant hydrazine thrusters	
	Orbit-Determination accuracy	1 km	
	Attitude Determination Accuracy	0.1 deg.	
Payload	LISS-1 LISS-2A and LISS-2B	(72.5 meter resolution), (36.25 meter resolution)	
Launch date		March 17, 1988	August 29, 1991
Launch site		Baikanur Cosmodrome ,Kazakhstan	
Launch vehicle		Vostok-II	

3.1.5 IRS-1A/1B Payloads

IRS missions envisage primarily meeting the specific Indian application needs in the areas of agriculture, hydrology, and geology. Hence the basic mission characteristics like spectral bands and resolutions, spatial and radiometric resolutions; repetivity and choice of local time have been arrived keeping these applications in view.

It is well known that to increase the accuracy of interpretation, the information has to be collected in more than one spectral band. A number of studies and experiments with Landsat data showed that four spectral bands covering visible and near infrared wavelength regions are adequate for most of the applications. Thus payloads should have multi spectral imaging capability, with four spectral bands in the visible and near infrared regions of the electro-magnetic spectrum.

Table 3-3: Spatial resolution and repetivity considerations

Application	Spatial Resolution	Repetivity
Agriculture	40-70 meters resolution	Weekly / monthly repetivity Soil classification needs seasonal considerations also
Hydrology	40-100 meters resolution	Soil moisture study for penetration beyond surface prefers microwave
Geology	100-150 meters resolutions	Repetivity can be monthly and more
Coastal studies	100-150 meters resolution; sea food study needs 70-100 meters resolutions	Weekly / monthly repetivity, for coastline delineation, yearly repetivity sufficient
Land use planning	80 meters resolution	Yearly repetivity

Table 3-4: Spectral resolution consideration

Spectral Bands	Characteristics of bands
0.45-0.52	Strong relationship between spectral reflectance in this region and plant pigment and has comparatively higher penetration in water. This band is useful for mapping suspended sediments/water quality and various studies related to coastal region.
0.52-0.59	Centered on the first local maxima of the vegetation reflectance, useful for vegetation discrimination and the study of senescence rate of leaves. Also sensitive to ferric iron oxides.
0.62-0.68	Centered around the chlorophyll absorption band of vegetation and, useful for identification of plant species. Greater soil contrast is found in this region. The upper end is limited to 0.68 to avoid the atmospheric absorption at 0.69 microns
0.77-0.86	Shows high reflectance for healthy vegetation and useful for green biomass estimation and crop vigor studies. Water absorption in this region clearly

	demarcates land water boundary. The upper end is limited to 0.86 microns to avoid the broad water vapor absorption band centered around 0.92 micron. In addition, this also helps to improve the Modulation Transfer Function (MTF) of this band since CCDMTF falls fast as wavelength increases in the near infrared region

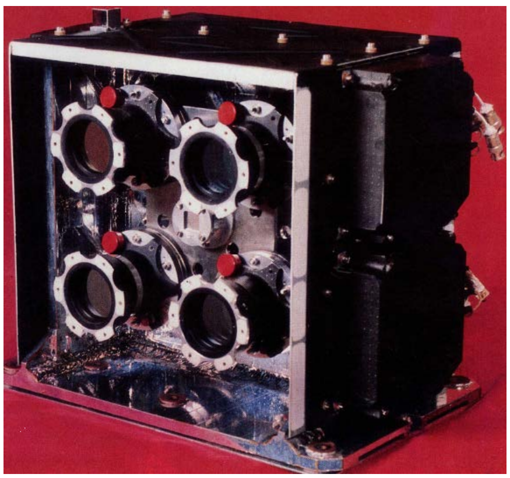


Figure 3-5 LISS-I

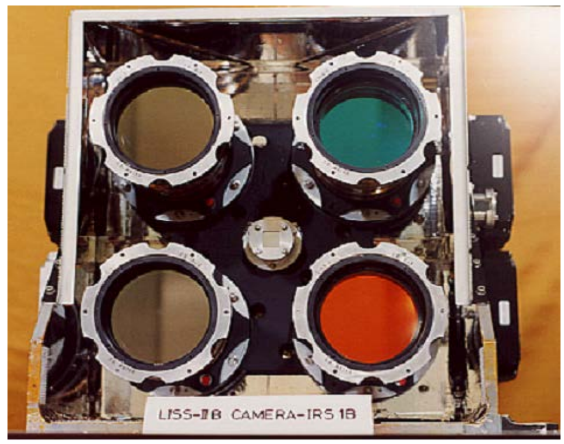


Figure 3-6 LISS-II

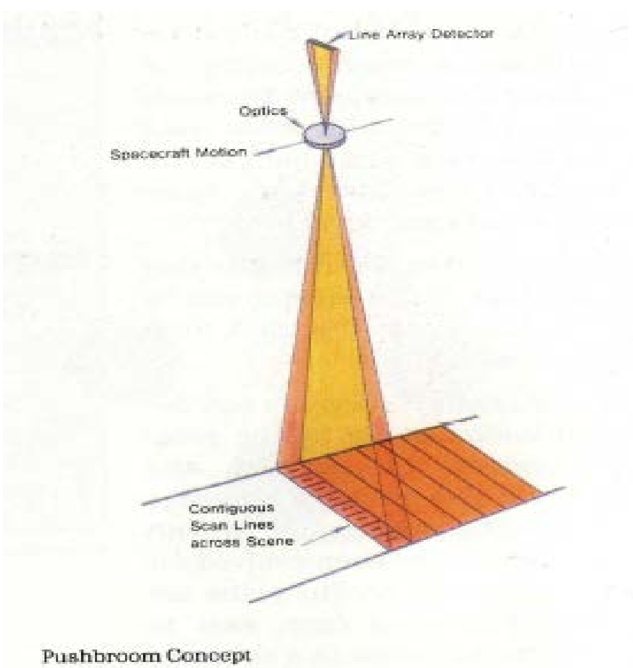


Figure 3-7 Push Broom Concept

The IRS-1A cameras operated in four spectral bands which are mentioned in the table 4.4. Each Band has separate optical system, spectral filters, thermal filters and detector.

The payload system of IRS-1A is a Linear Imaging Self-Scanning Sensor (LISS) working on the 'push-broom scanning' concept. In this mode of operation, each line of the image is electronically scanned by a linear array of detectors (Charge Coupled Devices (CCD)) and successive lines of the image are produced as a result of satellite's forward motion.

The payload system consists of two solid state cameras operating in four spectral bands in the visible and near- IR range using Charge Coupled Devices (CCD) linear arrays as sensors. There are two cameras, one is called LISS-1, and the other one is called LISS-II (It has two modules, IRS-IIA and IRS-IIB). LISS-I provide geometrical Instantaneous Field of View (IGFOV) of 73 meters and cover a swath of 148 km on ground, while LISS-II provides an IGFOV of 36.5 meters and individual swath of 74 km each. The combined swath of both LISS-II cameras is 145 km with a 3 km side lap between them.

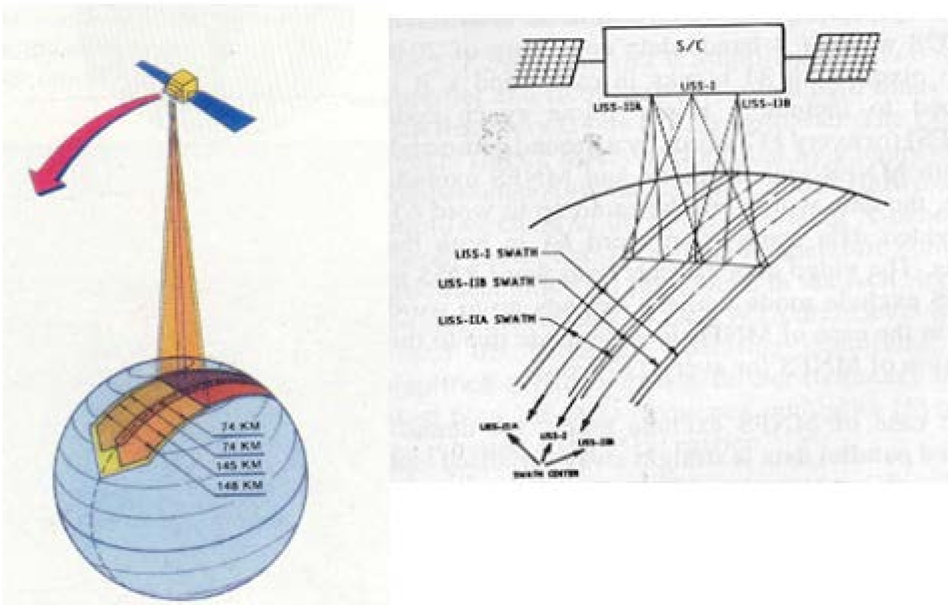


Figure 3-8 IRS-1A/1B Swath Details

The data handling system, consisting of baseband and RF modules, receives digital data from payload, formats it and after modulation transmits the data to ground station as a PCM stream LISS-I data is transmitted through a BPSK modulator in S-band at 5.2 MBPS and the data from both LISS-II cameras is transmitted through a QPSK systems in X-band at 10.4 MBPS.

Table 3-5:Payload specification

Payload specification		
Optics	Refractive , F/4.5	Refractive , F/4.5
Equivalent Focal Length (EFL)(mm)	162.2	324.4
Spectral Bands(um)		
Band-1	0.45 – 0.52	0.45 – 0.52
Band-2	0.52 - 0.59	0.52 - 0.59
Band-3	0.62 – 0.68	0.62 – 0.68
Band-4	0.77 – 0.86	0.77 – 0.86
Field of View	9.4 deg.	4.7 Deg (each)
Detector		
CCD	Linear array	Linear array
No. of pixels	2048	2048
Pixel size (um)	13 x 13	13 x 13
System		

Geometrical IGFOV (meters)	73	36.5
Angular IFOV (microradians)	80	40
Swath (km)	148	74 each (145 combined)
Integration time (m sec)	11.2	5.6
No. of radiometric levels	128	128
Data rate (Mbps)	5.2	10.4 x 2
Noise equivalent reflectance (NEdP)	<1 %	<1%
Signal – To - Ratio(SNR)	>127 for all bands at saturation level exposure	
Square wave response (SWR) at 40 Lines per millimeter (lpmm) Band -1 Band -2 Band –3 Band - 4	 >40 >40 >30 >20	 >40 >40 >30 >20
Band to Band registration (Pixel)	+/- 0.25	+/- 0.25
Operating temperature range EO module Electronics (PLE)	 20±5 deg. C 0 to 40 deg. C	 20±5 deg. C 0 to 40 deg C
Power(W) Imaging mode Cal. Mode	 34.2 37.9	 34.2 x 2 37.9 x 2
Mass(kg) EO Module Electronics Power supply	 27.50 4.48 6.44	 70.00 x 2 4.41 x 2 6.44 x 2

3.1.6 Ground Segment

The IRS-1A Ground segment controlled and monitored the satellite throughout the mission and performed the image data reception, processing, generation/dissemination and archival of data products.

The major elements of IRS-1A ground segment were

- Telemetry, Tracking and Command (TTC) network
- Spacecraft Control Centre (SCC)
- Data Reception System
- Data Products System

Ground segments location and functionality

Element	Location	Functions

Telemetry, Tracking & Command(TTC)	ISTRAC stations at Bangalore and Lucknow (Selective Support from External stations)	• Satellite Health data reception and recording • Spacecraft commanding and tracking
Spacecraft Control Centre (SCC)	ISTRAC, Peenya Bangalore, India.	• Network coordination and control • Spacecraft Operations • Spacecraft health analysis and control • Orbit and attitude determination • Communication links
Data Acquisition	National Remote Sensing Agency (NRSA), Shadnagar, Hyderabad, India. (Now it is National Remote Sensing Centre(NRSC).	• Reception and recording of Image data • Quick-look imagery and display. • Ancillary data generation for further processing of data.
Data Processing, Dissemination and Archival	NRSA, Balanagar, Hyderabad, India Space Application Centre (SAC Ahmedabad, India.	• Generation and distribution of Browse and Standard products. • Generation of precision and special products. • Data quality evaluation.

3.2 IRS-1E

3.2.1 Introduction

IRS-1E satellite, derived from the engineering model of IRS-1A incorporating a Monocular Electro-Optical Stereo Scanner developed by DLR, Germany, and a LISS-I camera similar to that on IRS-1A, could not be placed into orbit by the PSLV-D1 launched in September 1993.

The mission was not realised due to problems faced by Launch Vehicle. It was the first development flight of PSLV. IRS-1E was carrying LISS-1 and MEOSS payload.

3.2.2 Mission Objectives

The spacecraft was realised as a payload for the First developmental flight of PSLV.

3.2.3 Orbital Details

Parameter	Value
Orbit	**Not Realised**
Power	415

Mass	846
Launch Date	September 20, 1993
Launch Vehicle	PSLV-D1

3.3 IRS-P2

3.3.1 Introduction

IRS-P2 was the fourth in the series of Indian Remote Sensing Satellites. It was launched into the sun synchronous orbit of 817 km on October 15, 1994. This satellite is the first Spacecraft successfully orbited onboard by the second developmental flight of PSLV.

3.3.2 Mission Objective

The mission objective of IRS-P2 is to be the payload of the second developmental flight of PSLV

3.3.3 Orbit Details

Parameter	Values
Orbit	Polar sun synchronous Orbit
Altitude	817 km
Inclination	98.68^0
Repetivity	24 days
Orbits/cycle	
Equatorial crossing time	10.30 AM
Launch date	October 15, 1994
Launch site	SHAR Centre, Sriharikota, India
Launch vehicle	PSLV-D2
Mission completed on	1997

3.3.4 Salient Features of Spacecraft

Table 3-6:Salient features of IRS-P2

Parameter		IRS-P2
Payload	LISS-II*	LISS-II* was achieved by mounting two CCDs per optical lens system in staggered mode (Shown in fig.)
Data Handling		10.4 Mbps
Structure		Aluminium and aluminium honeycomb structure 4 vertical and 2 horizontal Al Honeycomb panels.
Thermal	Components	Passive control using tapes , OSR, MLI Blankets and semi-active/active control using proportionate temperature controller and heaters
	Temp. Range	20$\pm$5 deg. C for imaging sensors electro-optics 5$\pm$5 deg. C for Chemical Batteries

		0 to 40 deg. C for electronic packages
Mechanism	Solar Panel	Solar Panel deployment and drive mechanism
Power	Solar Array	6.424 m^2 area,(2 x 2 panels) deployable and sun tracking panel, Power generation at EOL is 620 Watts (Totally 6 panels)
	Battery	42V, Ni-Cd(2), 21 AH, 28 cells each, 27.64 kg
	Electronics	TCR, domestic regulators, battery individual cell monitoring, K relay emergency Relay Bus parallel relays, DC/DC Converters
Communication	Telemetry	House Keeping (HK) information in S-Band; PCM/PSK Real Time rate 256 bps and play back rate 4 Kbps; Onboard storage capacity of 98 minutes of HK data
	Telecommand	S-Band : PCM/PSK/FM/PM, and VHF : PCM/FSK/AM
	Tracking	Facility for ON/OFF and Data commands S-Band tone ranging and two way Doppler X-band beacon
Attitude and Orbit Control (AOCS)	Attitude sensors	IR Horizon sensors(Conical and Static), Star sensor, Yaw sun sensors, Dynamically Tuned Gyros (DTG)
	Actuators	Reaction Wheels (three 5NMS and one 10 NMS), Magnetic torquers, Hydrazine thrusters(16 one Newton)
	Orbit-Determination accuracy	1 km
	Attitude Determination Accuracy	0.1 deg.
Mass	Spacecraft	975 kg
	Payload	98 kg

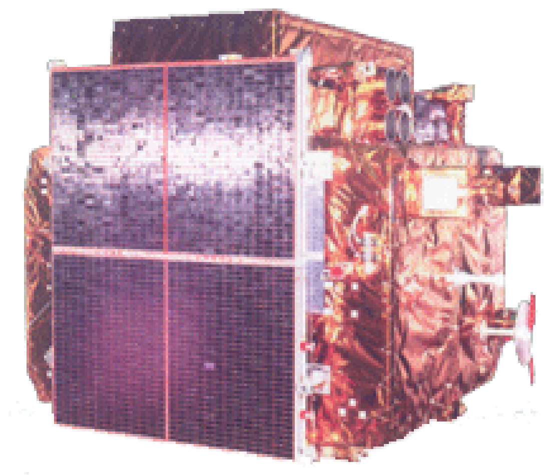


Figure 3-9 IRS-P2 Stowed Mode

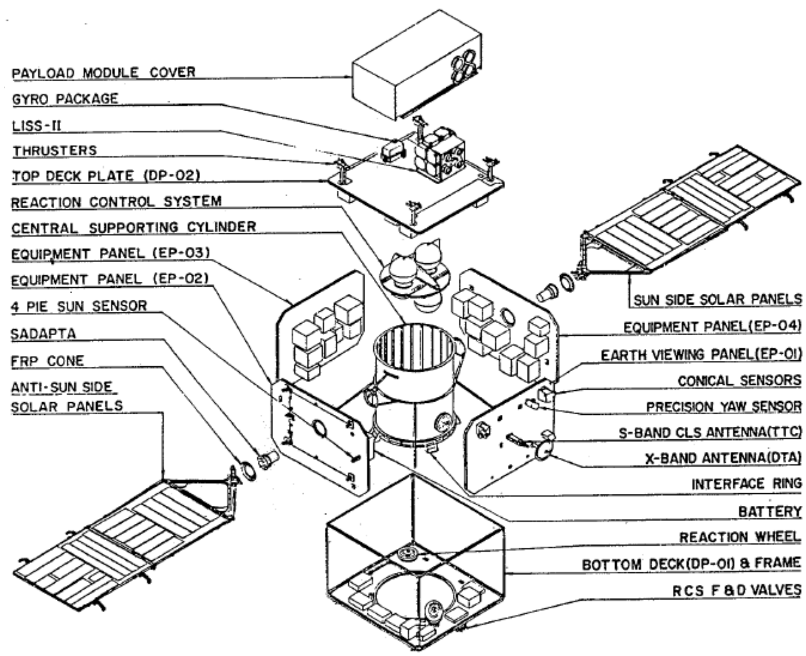


Figure 3-10: Exploded view of IRS-P2

3.3.5 IRS-P2 Payload

The LISS-II payload is a solid state camera operating in four spectral Bands in the visible and near IR range using 2048 elements Linear array CCDs as Sensors. Unlike in IRS-1A/1B satellites in which the LISS-II camera was made of two separate electro optical modules, in this camera two CCDs per band are placed in the focal plane of the same optics in a staggered configuration. These are designated as LISS-IIA and LISSIIB. Each camera provides an IGFOV of

32.74 meters and individual swath of 67 km the along track separation between the two CCDs is above 62 km on ground which will result in a combined swath of 128 km.

The data handling system, consisting of Base band and RF Modules, receives the digital data from Payload, formats it and after modulation transmits the data to ground station as a PCM stream.

Table 3-7 Features of IRS-P2 Payload

Parameter	Value
OPTICS	
Type of Optics	Refractive F/4.5
Equivalent Focal Length(EFL)mm	324.4
Spectral Bands (Microns)	Band1 : 0.45 - 0.52 Band2: 0.52 - 0.59 Band3 : 0.62 - 0.68 Band4: 0.77 – 0.86
Field of view	±5.2 degs
Detector	
Detector Type	CCD 143 Linear Array
Number of Pixel	2048
Pixel Size (Microns)	13 x 13
System	
Geometrical IFOV (m)	32.74
Along Track Sampling (m)	37.24
Angular IFOV (microradians)	40
Swath (km)	67 km each CCD (128 km Combined)
Integration Time (ms)	5.6
Quantisation	7 Bits
Data Rate	10.2 Mbps
Signal to Noise Ratio (SNR) @ saturation exposure	>128
Square Wave response SWR @ 40 lp/mm	Band1: >0.4Band2: >0.4 Band3: >0.3Band4: >0.2
Band to Band registration(Pixels)	Less than ± 0.25
Operating Temperature Range	EO module 20±5 deg C Electronics0 to 40 deg. C
Power in watts	Imaging mode :32 Calibration mode :34
Mass (kg)	72

3.3.5.1 Payload Configuration

The payload consists of three major elements

- Electro-optical Module
- Payload electronics
- Payload Power Supply

3.3.5.1.1 Electro optical Module

The EO module contains imaging optics including the spectral band pass filters and neutral density filters (ND), CCD detectors and detector electronics. The four band assembles in the camera use refractive optical systems and these are coupled to the detectors through Invar housings. These four single band assemblies are mounted on a welded aluminum bracket with their optical axes parallel to each other. To minimize the variation of BBR with temperature gradients, the four band assemblies are coupled through Invar plates at lens end, detector end and middle flanges. Two CCDs are mounted in the focal plane of each lens separated by a distance (in along-track direction) of about 25 mm and a gap of 4 pixels (in the across track direction). Two LEDs per CCD mounted at an angle of 60 deg are used for on board calibration. DE packages mounted on EO module Houses the Bias Voltage generator, clock driver as well as pre-amplifier circuits for the operation of each CCD.

The payload electronics and payload power supply are similar to IRS-1A/1B.

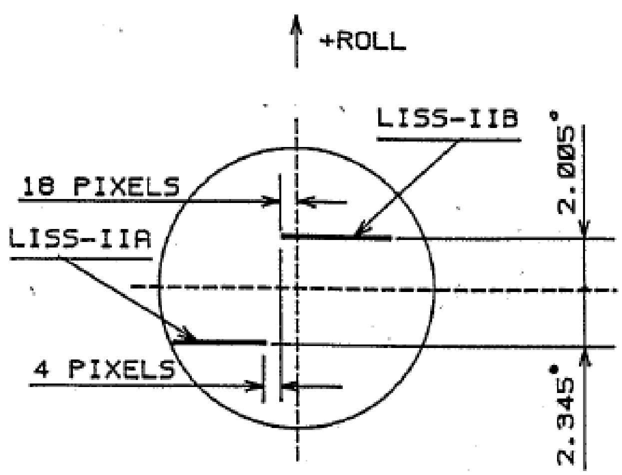


Figure 3-11: CCD arrangement in detector plane as seen from detector Plane

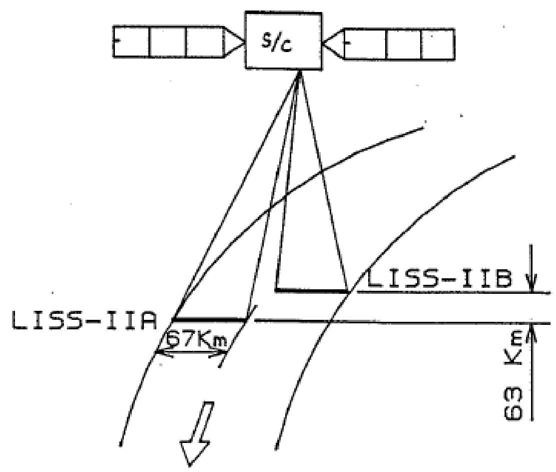


Figure 3-12: Swath coverage of LISS-IIA and LISS-IIB of IRS-P2

3.4 IRS-1C &1D

3.4.1 Introduction

IRS-1C is the second generation Remote sensing operational satellite developed by ISRO that carried three distinct and mutually complementary imaging payloads. The combination of payloads enhanced the capabilities of IRS-IC as compared to IRS- IA/1B in terms of spatial resolution, provision of an additional spectral band, ability to acquire stereoscopic images and inclusion of a wide field sensor for improved temporal resolution. IRS-1D was the follow-on mission.

3.4.2 Mission Objective of IRS-1C and 1D

Mission objectives of IRS-1C and 1D are as given below

- To design develop launch and operate state of art three axis body stabilised satellite for providing continued space based remote sensingt services to the user community with enhanced resolution capability compared to IRS-1A/1B
- Further develop new areas of user applications to take fiull advantages of the enhanced resolution and capacity of IRS-1C/1D spacecraft.

3.4.3 Orbit Details

IRS-1C and 1D were launched into polar sun syncronous near circular orbit to ensure ground illumination conditions are nearly the same for imageries collected on different days. Local time equtorial crossing was chosen 10.30 AM based on application needs of the users.

Table 3-8: Orbital parameters of IRS-1C/1D

Parameter	IRS-1C	IRS-1D
Orbit	Polar sun synchronous	Polar sun synchronous
Altitude	817 km	740 x 817 km
Inclination	98.69 deg	98.6
Eccentricity	0.0004	
Period	101.35 minutes	101.35 minutes
Local Time	10.30 A.M	10.30 A.M
Repetivity Cycle	24 Days (For Liss-3) 5 Days (for Pan 5 Days (for PAN revisit)	24 Days (For Liss-3) 5 Days (for Pan 5 Days (for PAN revisit)
Distance between adjacent Traces	117.5 km	117.5 km
Minimum Picture Overlap for LISS-3	22.5 km	22.5 km
Off Nadir coverage +/- 26 deg (for PAN)	398 km	398 km
Distance between successive Ground tracks	2828 km	2828 km
Ground Trace velocity	6.65 km/s	6.65 km/s

3.4.4 Salient features of IRS-1C/1D

Though most of the systems were fabricated similar to IRS-1B, based on the onboard experiences of IRS-1A and IRS-1B satellites, and in view of launching the satellite using Indian launch vehicle (PSLV), some modifications/ improvements were carried out in IRS-1D spacecraft. They are given in the following table

Table 3-9. Salient features of IRS-1C/1D

Subsystem	IRS-1A/1B	IRS-1C/1D
Structure	Al. honeycomb structure with central load bearing Al. Cylinder	Shear webs added to increased the frequency to cater to PSLV launch. CFRP cylinder (370 mm height 930 mm dia) for thermal isolation of payload deck incorporated.
Thermal	Passive, Semi-active with heaters	Experience gained from IRS-1A for evolving IRS-1C thermal design, Payload module had new configurations and power dissipation in IRS-1C was more and hence a new design and analysis made.

Mechanism	Solar Panel deployment mechanism	Solar panel deployment mechanism used as it is 'PAN' camera deployment and steering mechanism newly developed.
Power	Solar Panel	9.636 m^2, 6 panels 1.1 x 1.46 m^2 (Each) BSR (SCA) 813 watts (10% area increased)
	Battery	2 batteries, 42V, 28 Cells, Ni-Cd 21 AH
	Power electronics	More efficient power electronics developed. Additionally CUK type of DC/DC converters developed.
TTC	S-Band transponder	S-band transponder used as it is
	PROM based telemetry	Modified to meet mission specific requirements
	Telecommand system (418 ON/OFF, 21 data commands)	Modified to meet mission specific requirements including time tag commands (704 ON/OFF and 46 data commands)
Data Handling	S-Band for LISS-1(5.2 MBPS)	PAN and LISS-3 data in two independent X-band chains. QPSK modulator developed.
	X Band for LISS_2(10.4 x 2 MBPS) with 20 watts TWTA	40 watts TWTA used. 84 MBPS and 42 MBPS data Handling system Developed.
Sensors	Sun sensors (4PI, TWSS, FSS, PYS)	PYS improved to reject any spurious signals
	Conical earth sensors	Same earth sensors used but with improvements to overcome dazzle problems and reduce the systematic errors.
	Dry tuned gyro for yaw control. Two gyros in a cluster	Three gyros in a cluster. All three axis to be controlled by gyro.
	Star sensor based on linear CCD	Star sensor based on area array CCD (for providing attitude information about all three axes)
RCS	Mono propellant 1 Newton system. Tanks for 80 kg Capacity.	IRS-1A system retained. In addition latch valves, filters, thermocouples bed heaters developed. 11 Newton thruster developed for IRS-1C and incorporated in the configuration for orbit correction purpose
Reaction Wheel	5 NMS wheels	Same wheels retained. However improvements made with an add-on-package for dynamic friction compensation

SADA	Developed for IRS-1A provided 5000 gm.cm torque	Same used But improvements made to increase the tongue margin
AOCE	Hardwired system PWPFM controller	Hardwired system as a backup only for processor based linear controller.
	KF Used	Improved KF used
PYRO	Developed for IRS-1A	Same retained
Payloads	LISS-I and LISS-II	LISS-III, PAN and WiFS
Reliability goal	0.75 at the end of 3 years	0.75 at the end of 3 years

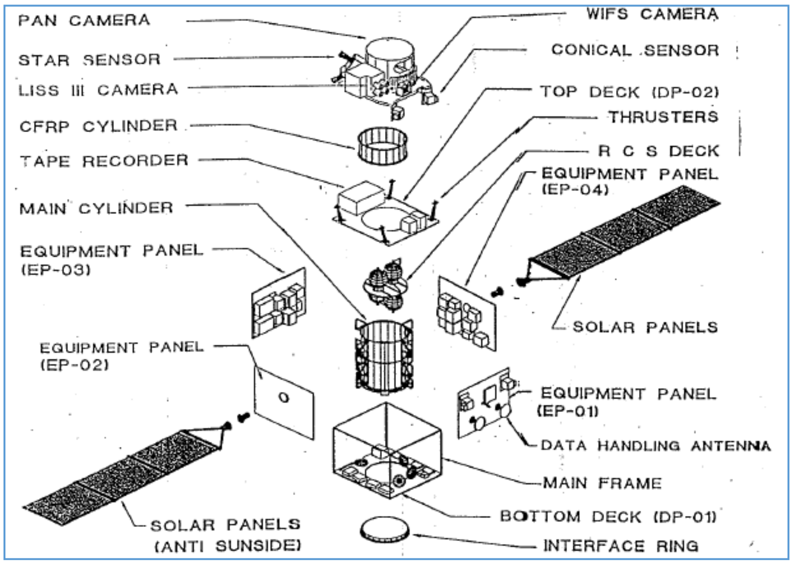


Figure 3-13 IRS-1D Exploded View

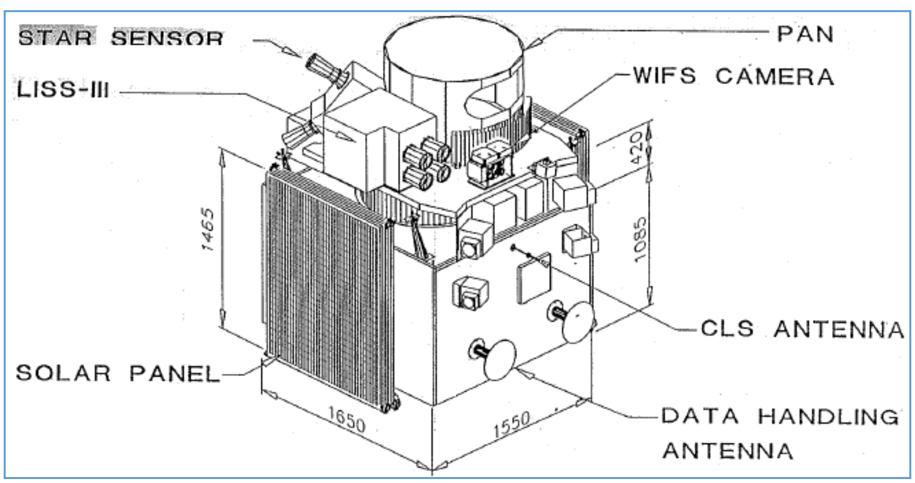


Figure 3-14 IRS-1D Stowed View

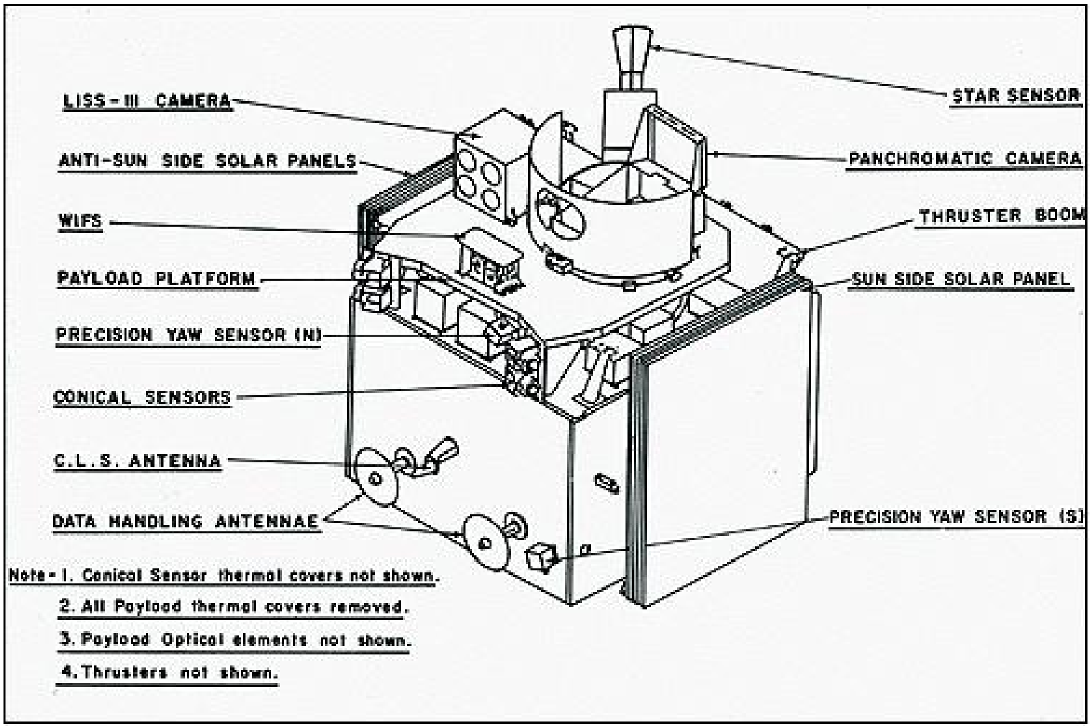


Figure 3-15. Stowed mode view of IRS-1C

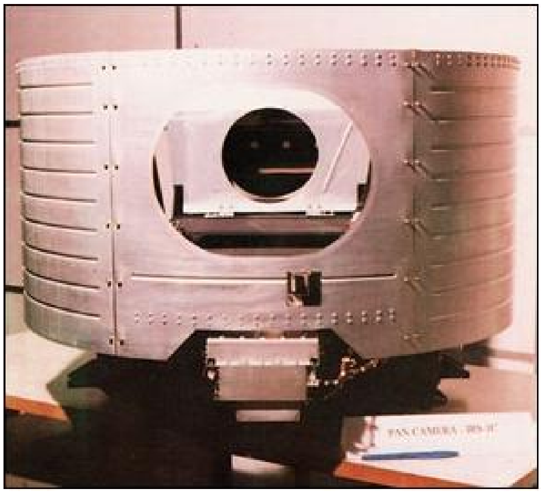
3.4.5 IRS-1C/1D Payloads

The payload system of IRS-1C/1D consists of three cameras namely

- Panchromatic camera (PAN),
- Linear Imaging Self Scanning sensor (LISS-III) and
- Wide Field Sensor (WiFS)

All cameras operate in the push-broom scanning mode employing linear array charge coupled devices (CCD).

3.4.5.1 Panchromatic Camera (PAN)



The PAN camera provides a spatial resolution of 5.8 meters at nadir and operates in a single (0.5-0.75) panchromatic spectral band. This camera covers a ground swath of 70 km which is steerable upto 26 deg. from nadir in the across track direction. This off nadir viewing provides the capability to revisit any given site with a maximum delay of five days. The major specifications of the IRS-IC PAN camera are given in table 3.10

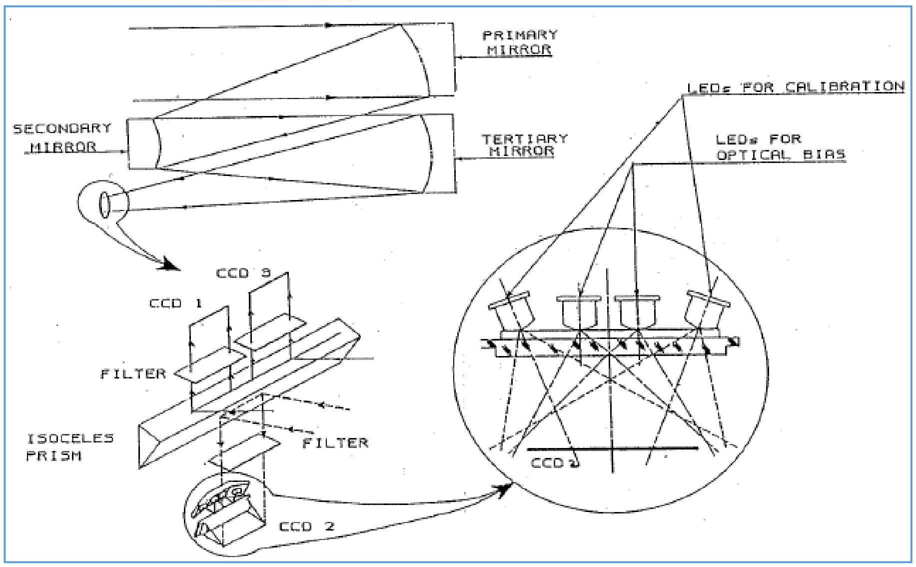


Figure 3-16 CCD Arrangement in IRS-1C/1D PAN camera

3.4.5.1.1 Optical Design of PAN

The PAN camera uses an all reflective off axis telescope, while LISS-III and WiFS are realised using refractive optics. The PAN optical system is a 980mm focal length (f/4.5) unobscured three mirror system i.e there is no obstruction to the incoming beam by any part of the optical system. The optical design features an off axis primary hyperboloid mirror, a spherical secondary mirror and an off axis ellipsoidal tertiary mirror. By using off axis sections of conic surfaces, obstruction of the incoming radiation is avoided resulting in higher modulation transfer function for a given aperture. Since the image format (85mm) is too large to be covered by a single CCD, an arrangement of 3 CCDs is used to cover the full swath. A prism with two reflecting sides is placed slightly ahead of the image plane. The light rays from the tertiary mirror falling on the sides of the prism are reflected out in opposite directions. The prism angles are so configured that the light rays from 0.3 deg. of nadir, along track, form two image lines on either side of the prism. These two image lines when projected on ground are separated by 8.6 km. One of the image lines is covered by two CCDs with a gap corresponding to the coverage by one CCD between them. The second image line is imaged by a single CCD which is centrally located.

The telescope mirrors are fabricated out of zerodur and are mounted in multi bladed mirror mounts using an appropriate glue in such a way that surface deformation on the mirrors do not occur. The super invar mirror mounts have been designed to withstand storage temperature and mechanical loads generated during the launch. The use of zerodur mirrors with the invar structure reduces the drop in MTF due to temperature variation within the operating temperature of 17-23°C. Further, the mirror surface does not show any non elastic behavior in the storage temperature range of -30°C to +60°C. Baffles in the optical assembly have been designed to reduce out of field radiation and reduce the drop in MTF from stray light. The baffles have been located near secondary and tertiary mirror mounts. The design value of MTF is greater than 0.6. In practice after taking to account the fabrication, tolerance, alignment etc., it is possible to realize MTF of 0.5.

Table 3-10: Characteristics of PAN

Parameters	Parameters
Instantaneous Geometric field of view * (meters)	5.8
A)Swath* (km) B)Swath Steering Range (degree) C)step Size (Degree)	70 ± 26 ± 0.09
Spectral Band (micron)	0.50-0.75
Camera SWR (At Nyquist frequency)	0.20
Quantization (bits)	6
System Noise	1 LSB
Saturation radiance (nominal) (MW/Cm2- STR-micron)	47
Detector	3 x 4096 pixel CCD (7 x 7 micron)
Size of EO Module (Envelope) (mm)	605 (R) X 903 (P) X 861 (Y)
Weight (kg) EO Module PLE Package	105 (without PSM) 20
Power (W) Imaging Mode CAL Mode	55 65
Data Rate (MBPS)	84.9

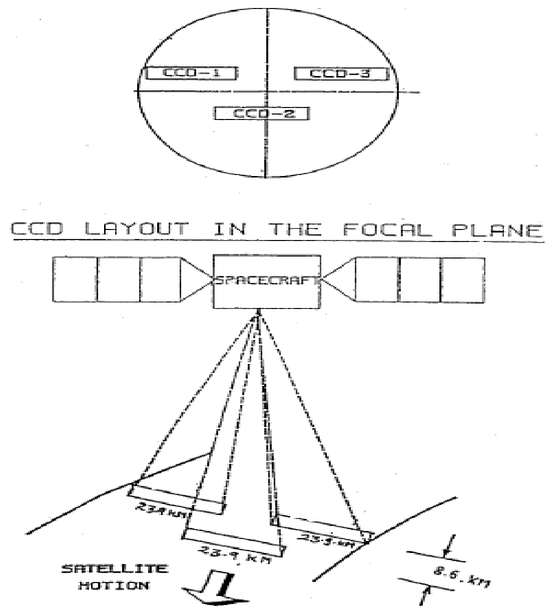


Figure 3-17 Swath coverage of IRS-1C/1D

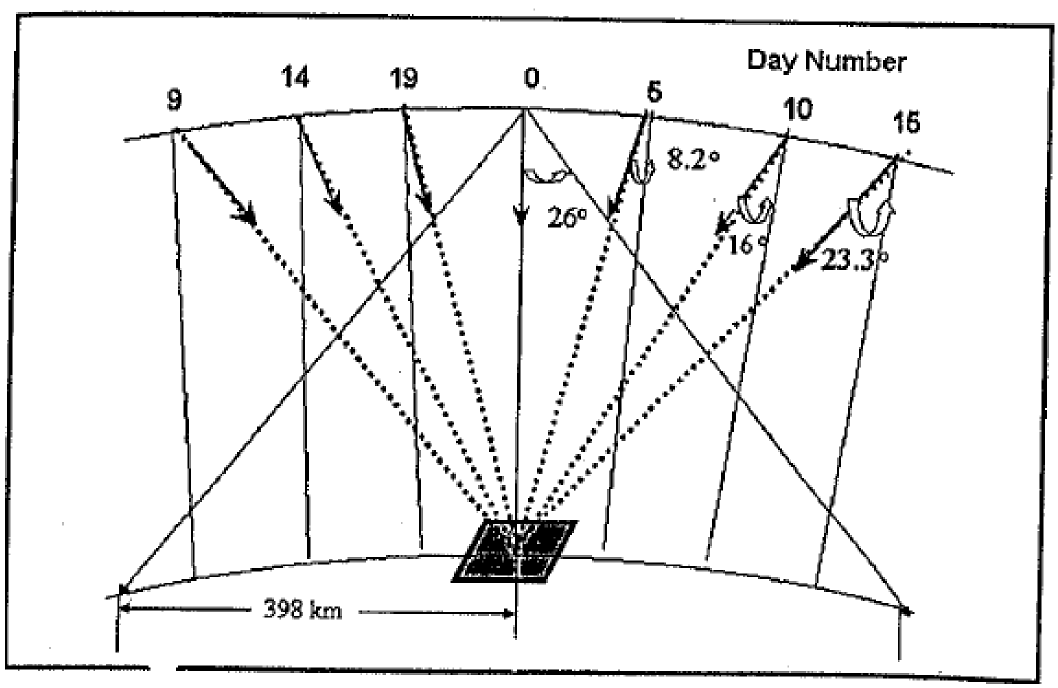


Figure 3-18 Pan Off nadir viewing capability Swath coverage of PAN

3.4.5.2 LISS-3 Camera

The LISS-3 camera is a multispectral imaging system operating in four spectral bands, three in the visible and Near IR (VNIR) region which are identical to B2, B3 and B4 of IRS- IA/1B and one in short wave infrared (SWIR)-band B5. LISS-3 provides a ground resolution of 23.5 m in VNIR and 70.5 m in SWIR with a swath of 141 km and 148 km respectively for VNIR and SWIR.

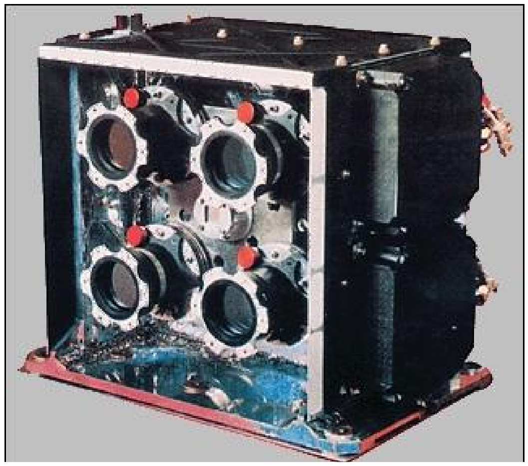


Figure 3-19 IRS-1C/1D LISS-3 Camera

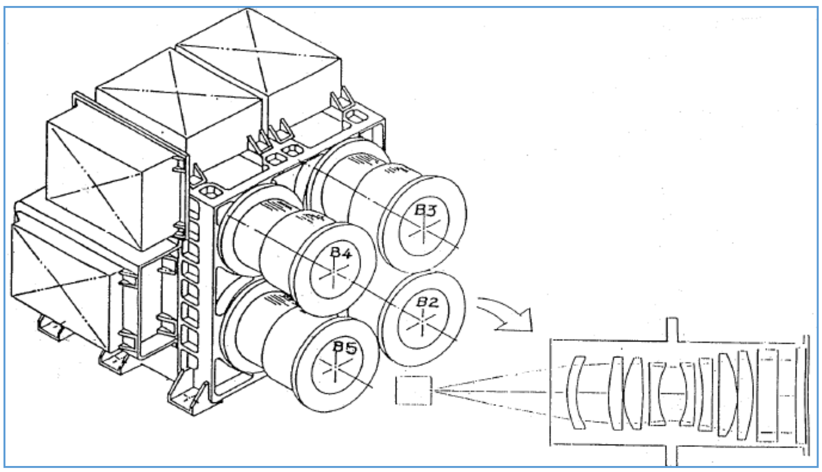


Figure 3-20 LISS-3 EO Module

All the four lenses of LISS III is similar. The lens design is derived from double Gauss concept. The design is optimized separately for each spectral band to obtain the best MTF performance. The design features a very low sensitivity of EFL, FD and collinearity to temperature variation. The lenses for bands B2, B3 and B4 having an f/no. of 4.35, and a focal length of 347.5 mm operate at 50 lp/mm whereas band B5 has focal length of 301 .04 mm with f/no. 4.35 and operates at 20 lp/mm .

To minimize the impact of surface reflections, each surface of the optical elements carries antireflection (AR) coatings. The AR coatings have been provided on all lens elements, thermal filter and outer surfaces of the interference filter. The lens is purged with dry Nitrogen and sealed with a membrane. After the launch, when the differential pressure is more than 500 mbar the membrane ruptures and allows the evacuation of the lens assembly. The assembly of the camera takes into account the change in focal length from laboratory environment to the vacuum conditions in orbit.

Hard coated four cavity interference filters have been used in these lenses, for spectral selection. The thermal filter made of fused silica, makes an angle of one degree with the optical axis of the lens assembly to avoid ghost images at the CCD plane. It has a provision to allow rotation of this angle around the optical axis to take into account the CCD detector orientation with respect to the mounting holes of the flange.

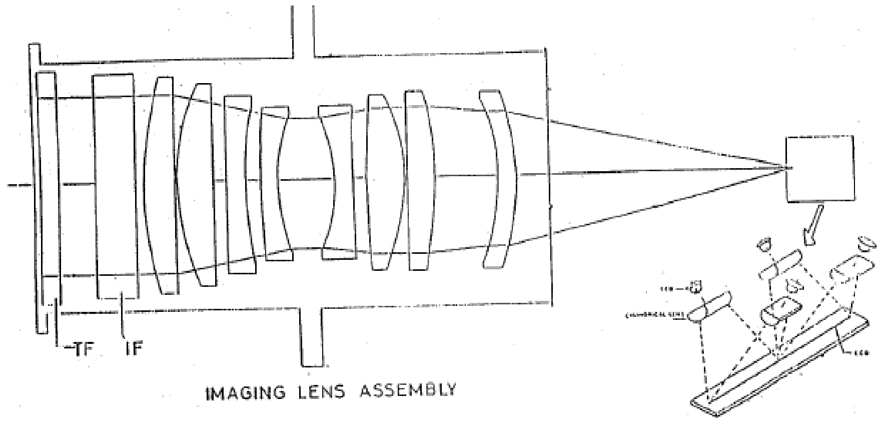


Figure 3-21 Calibration LEDs arrangement In LISS-3

Table 3-11 Characteristics of IRS-1C/1D LISS-3 Payload

Parameters	Parameters
Instantaneous Geometric field of view * (meters)	23.5 B2, B3, B4 70.5 SWIR(B5)
Swath* (km)	>141
Spectral Band (micron)	B2 0.52-0.59 B3 0.62-0.68 B4 0.77-0.86 B5 1.55-1.70
Camera SWR (At Nyquist frequency)	B2 40; B3 40; B4 35; B5 30
Quantization (bits)	7
System Noise	>1 LSB
Saturation radiance (nominal) (MW/Cm2- STR-micron)	B2 29± 1.5 B3 28 ± 1.5 B4 31 ± 1.5 B5 3.5 ± 0.3
Detector	10 x 7 micron 6000 element CCD for Visible 26 x 26 micron 2100 element CCD for NIR
Size of EO Module (Envelope) T(mm)	455 (R) X 522 (P) X 500 (Y)
Weight (kg) EO Module camera	76.5 95
Power (W) Imaging Mode CAL Mode	74 78

Data Rate (MBPS)	B2,B3,B4 35.8 B5 1.4

3.4.5.3 WiFS Camera

The WIFS camera has a spatial resolution of 188 meters covering a swath of 804 km. This wide swath coverage results in a repeatable observation of the same ground location after every 5 days. The WIFS operates in two spectral bands B3 and B4 of LISS-III (0.62 - 0.68 and 0.77 - 0.86).

Figure 3-22 IRS-1C/1D WiFS Camera

For the WiFS camera, the total field to be covered was 52°. If this was realized using single lens for each band, due to the large variation in the incidence angle at the interference filter, there will be considerable shift in the band edge positions over the field of view. To minimize the above effect, the total FOV is realized by two lenses for each band. The two lenses are mounted with their optical axes canted 13° on either side of nadir. The basic optical design is similar to LISS III except for the focal length of 56 mm.

Table 3-12:Characteristics of WiFS Camera

Parameters	Parameters
Instantaneous Geometric field of view * (meters)	188
Swath* (km)	804
Spectral Band (micron)	B3 0.62-0.68 B4 0.77-0.86
Camera SWR (At Nyquist frequency)	B3 >34 B4 >20
Quantization (bits)	7
System Noise	<1 LSB

Saturation radiance (nominal) (MW/Cm2- STR-micron)	B3 28 ± 1.5 B4 31 ± 1.5
Size of EO Module (Envelope) (mm)	250 (R) X 335 (P) X 170 (Y)
Weight (kg) EO Module camera	18 23
Power (W)	28
Data Rate (MBPS)	2.1

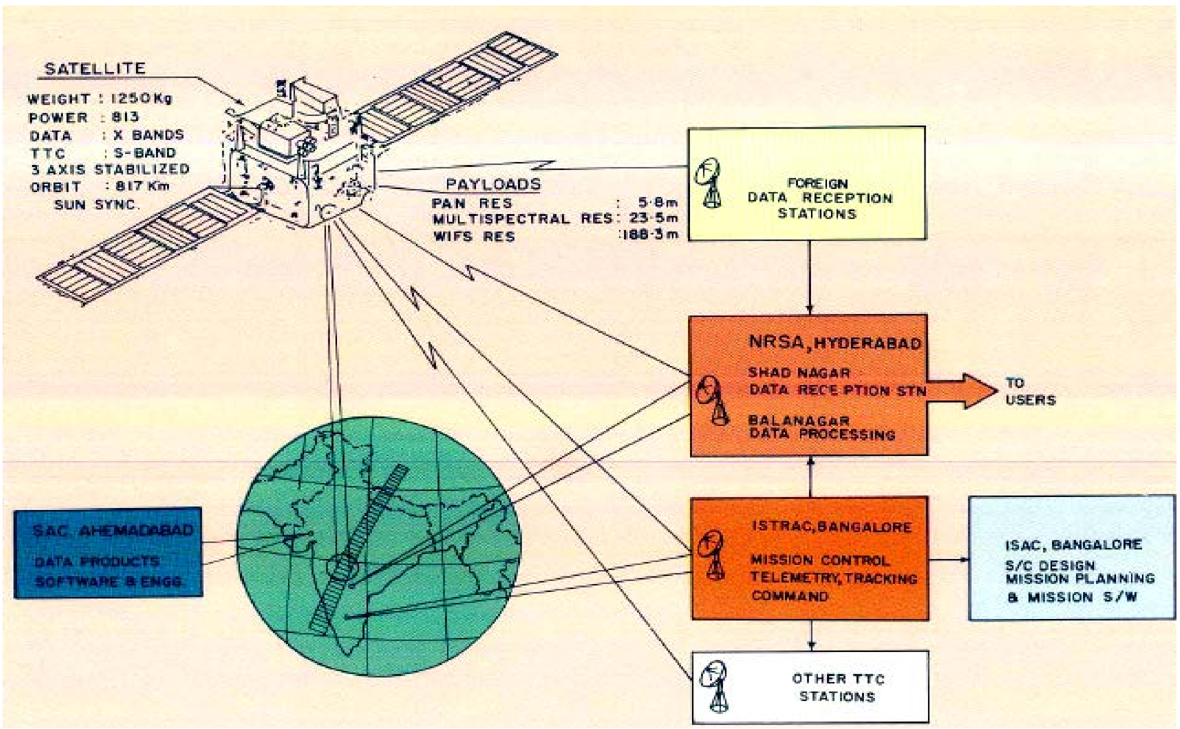


Figure 3-23 IRS-1C/1D Mission Elements

3.5 IRS-P6 (Resourcesat-1)

3.5.1 Introduction

IRS-P6 is the continuation of IRS-1C/1D missions with enhanced capabilities. Panchromatic camera of IRS-1C/1D is improved to Multispectral by using three 12K detectors. The spatial resolution of AWIFS is improved to 56 m from ~188m.

3.5.2 Mission Objective

Mission objectives of the IRS-P6 are as given below

- To provide continued Remote Sensing data services on an operational basis for integrated land and water resources management at micro level with enhanced multi spectral/ spatial coverage and stereo imaging
- To further carry out studies in advanced areas of User applications like improved crop discrimination, crop yield, crop stress, pest/disease surveillance, disaster management etc.

3.5.3 Orbit Details

The IRS-P6 is with payloads similar to the IRS-1C/1D. Choice of the orbit is same as that of IRS-1C i.e Sun synchronous orbit at an altitude of 817 Km.

Table 3-13 Orbit details of IRS-P6

Parameter	IRS-P6
Orbit	Polar sun synchronous circular
Altitude	817 km
Inclination	98.69 deg
Eccentricity	0.0004
Period	101.35 minutes
Local Time	10.30 A.M
Repetivity Cycle	24 Days (For LISS-3) 5 Days (for AWifs) 5 Days (for LISS-4 revisit)
Distance between adjacent Traces	117.5 km
Minimum Picture Overlap for LISS-3	22.5 km
Off Nadir coverage +/- 26 deg (for PAN)	398 km
Distance between successive Ground tracks	2820 km
Ground Trace velocity	6.65 km/s
Liss-4 Coverage with steering of ± 26 Deg	± 398 km

3.5.4 Salient features of IRS-P6

The S/C mainframe is of IRS-1C/1D -P3 heritage. The S/C structure consists of two modules, the main platform and the payload module. The main platform is built around a central load bearing cylinder of 915 mm diameter and consists of four vertical panels and two horizontal decks. The bottom of cylinder is attached to an interface ring which interfaces with the launch vehicle. The vertical panels and the horizontal decks carry various subsystem packages. Various attitude sensors, SPS (Satellite Positioning System) and data transmitting antennas are mounted on the outside surfaces of the equipment panels and the bottom deck. Two star trackers are mounted with skewed orientation on the top deck. The payload module in turn is comprised of a two-tier system, the payload module deck and the rotating deck. The payload module deck accommodates LISS-3, AWIFS-A and AWIFS-B camera modules.

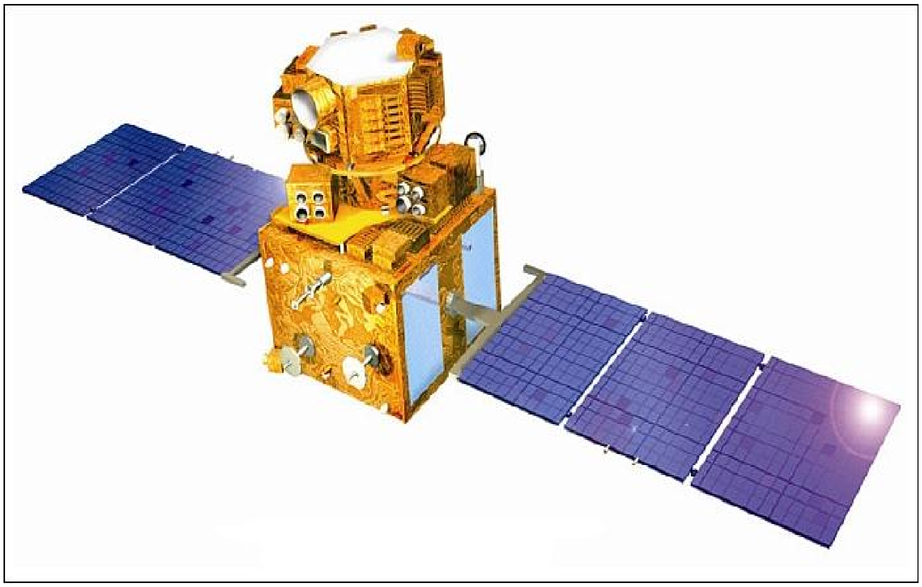


Figure 3-24Illustration of the IRS-P6 spacecraft

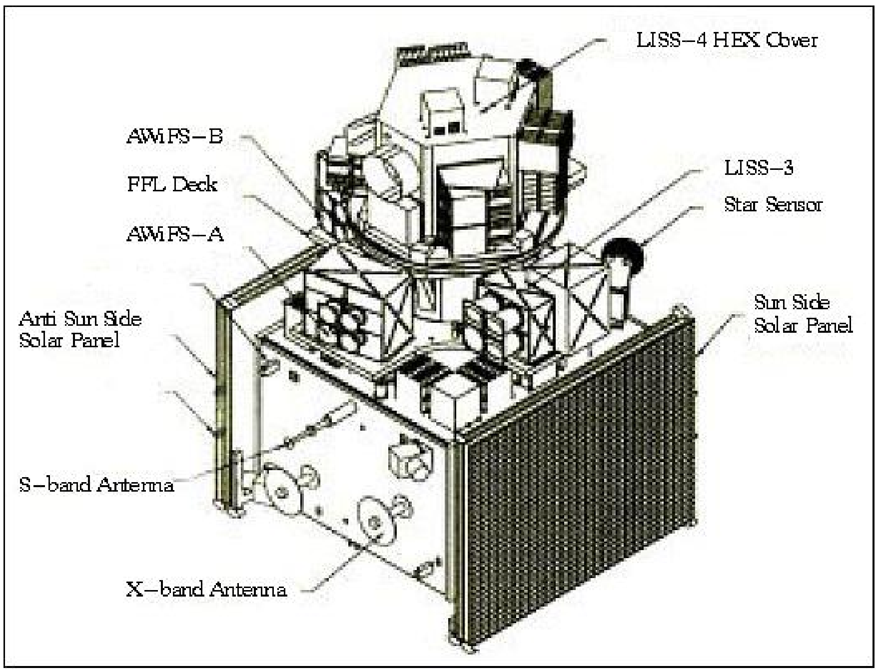


Figure 3-25 Isometric view of the IRS-P6 spacecraft in launch configuration

Table 3-14 Salient features of IRS-P6

Subsystem		IRS-P6
Structure		Shear webs added to increase the frequency to cater to PSLV launch. CFRP cylinder (370 mm height 930 mm dia) for thermal isolation of payload deck incorporated.
Thermal	Control	Temperature control is with passive techniques using Paints, multilayer blankets, Optical solar Reflector, and active thermal elements like heaters also. Heat pipe radiator panel is used to maintain the temperature of LISS-4 detector head assembly.
	Limits	All electronics packages 0-40degC, Battery 0-10 degC, Payload EO modules : 17 to 23
Mechanism	Solar Panel	Solar panel deployment mechanism and Drive Mechanism
	LISS-4	Deployment and steering mechanism for the LISS-4 Payload to cover +/- 26 deg. w.r.t. roll.
Power	Solar Panel	Sun tracking, rigid,15.12 M^2 6 panels 1.4 x 1.8 m^2 (Each), 1250 W at EOL, BSR(SCA)
	Battery	2 batteries, 28 to 42V, 28 Cells, Ni-Cd 24 AH
	Power Electronics	PWM TCR, FCL, 10 Strings
TTC	Telemetry	1024 words/frame, 2250 MHz2071, storage: 6.29 x 10^6 Bits PCM/PSK/PM, 16 Kbps
	Telecommand	PCM/FSK/FM/PM 2071.875 MHz
Data Handling		The payload data are transmitted in X-band at a data rate of 105 Mbit/s. The BDH (Baseband Data Handling) system consists of two separate chains, one for LISS-3 and AWiFS data, and the second chain for LISS-4 data. The LISS-4 data are transmitted on carrier-1 at 8125 MHz and LISS-3 + AWiFS data are transmitted on carrier-2 at 8300 MHz.
Data Transmission		40 watts TWTA used. 84 MBPS and 42 MBPS data Handling system Developed.
AOCS	Spec.	Pointing Accuracies: Yaw: $\pm$ 0.05^o Roll: $\pm$ 0.05^o Pitch: $\pm$ 0.05^o (3 sigma); Drift rate : 5 x 10^{-4} deg/sec (3 sigma)
	Sensors	Earth sensor(1), DSS(2), Star Sensors(2), 4Pi SS(4), Magnetometer (2) IRU(3 DTG), GPS
	Actuators	Reaction Wheels, 5 NMS(4 in tetrahedral), Magnetic Torquers (2) , 1N Thrusters(8) 11 N Thruster(4) Fuel (100 kg)
SADA		Improved SADA used to increase the torque margin

AOCE		Hardwired system as a backup only for processor based linear controller. Improved KF used
Payloads		LISS-III, PAN and AWiFS
Mass		1360 kg

3.5.5 IRS-P6 Payloads

Resourcesat-1 carries three payloads.

- A high resolution linear imaging self-scanner (LISS-IV)
- A medium resolution linear imaging self-scanner (LISS-III)
- AWiFS (Advanced Wide Field Sensor).

3.5.5.1 LISS-4 (Linear Imaging Self-Scanning Sensor-4):

The LISS-4 multispectral high-resolution camera is the prime instrument. LISS-4 is a three-band pushbroom camera of LISS-3 heritage (same spectral VNIR bands as LISS-3) with a spatial resolution of 5.8 m and a swath of 70 km. LISS-4 can be operated in either of two support modes:

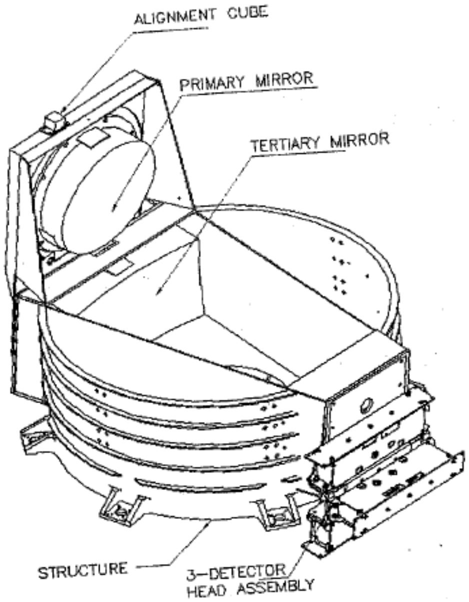
Multispectral (MS) mode: Data is collected in 3 bands corresponding to pre-selected 4096 contiguous pixels with a swath width of 23.9 km (selectable out of 70 km total swath). The 4 k detector strip can be selected anywhere within the 12 k pixels by commanding the start pixel number using the electronic scanning scheme.

Mono mode: Data of the full 12 k pixels of any one single selected band, corresponding to a swath of 70 km, can be transmitted. Nominally, band-3 data (B3) are being observed and transmitted in this mode.

LISS-4 has ±26º steering capability in the cross-track direction which provides a 5-day revisit cycle. The optoelectronic module of LISS-4 is identical to that of the PAN camera of IRS-1C/1D. The CCD array features 12,288 elements for each band. The instrument has a mass of 169.5 kg, power of 216 W, and a data rate of 105 Mbit/s. The detector temperature control is implemented using a radiator plate coupled to each band CCD through heat pipes and copper braid strips.

Figure 3-26 LISS-4 camera

The LISS-4 camera is realized using the three mirror reflective telescope optics (same as that of the PAN camera of IRS-1C/1D) and 12,288 pixels linear array CCDs with each pixel of the size 7 µm x 7 µm. Three such CCDs are placed in the focal plane of the telescope along with their individual spectral bandpass filters. An optical arrangement comprising an isosceles prism is employed to split the beam into three imaging fields which are separated in along track direction. The projection of this separation on ground translates into a distance of 14.2 km between the B2 and B4 image lines. While B3 is looking at nadir, B2 is looking ahead and B4 is looking behind in the direction of velocity vector. Detector type: THX31543A of Thomson.



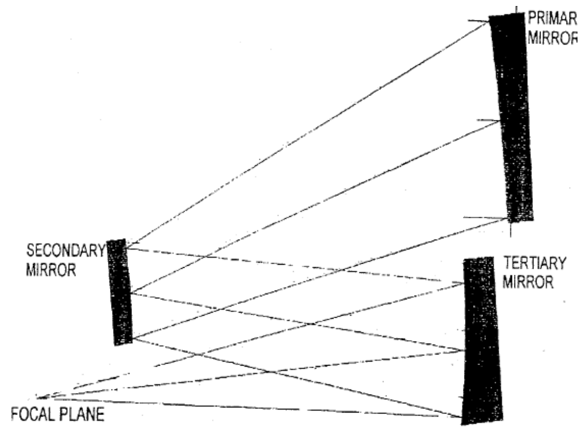


Figure 3-28 Optical Schematic of LISS-4

Figure 3-27 LISS-4 Payload Schematic

LISS-4 calibration: An in-flight calibration scheme is implemented using LEDs (Light Emitting Diodes). Eight LEDs positioned in front of the CCD (without obstructing the light path during imaging). These LEDs are driven with a constant current and the integration time is varied to get 16 exposure levels, covering the dynamic range in a sequential manner. This sequence repeats in a cyclic form.

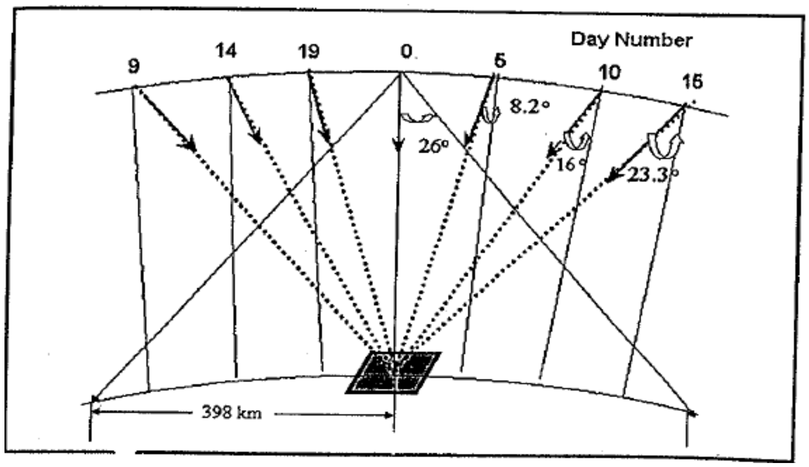


Figure 3-29 Possible Image coverage due to Steering of IRS-P6 LISS-IV

3.5.5.2 LISS-3 (Linear Imaging Self-Scanning Sensor-3):

LISS-3 is a medium-resolution multispectral camera. The pushbroom instrument is identical to LISS-3 on IRS-1C/1D (with regard to lens modules, detectors, and electronics) in the three VNIR bands, each with a spatial resolution of 23.5 m. The resolution of the SWIR band is now also of 23.5 m on a swath of 140 km. The optics design and the detector of the SWIR band are modified to suit the required resolution; B5 uses a 6,000 element Indium Gallium Arsenide CCD with a pixel size of 13 µm. The SWIR CCD is a new device employing a CMOS readout technique for each pixel, thereby improving noise performance. The VNIR CCD array features 6,000 elements for each band. The instrument has a mass of 106.1 kg, a power consumption of 70 W, and a data rate of 52.5 Mbit/s.

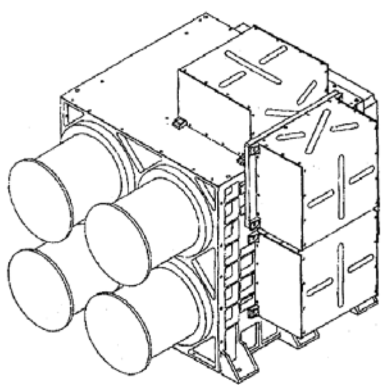



Figure 3-30 IRS-P6 LISS III Payload

The in-flight calibration of the LISS-3 camera is carried out using 4 LEDs per CCD in the VNIR bands and 6 LEDs for the SWIR band. These LEDs are operated in pulsed mode and the pulse duration during which these LEDs are ON is varied in specific steps. Each LED has a cylindrical lens to distribute the light intensity onto the CCD. Each calibration cycle consists of 2048 lines providing six non-zero intensity levels.

3.5.5.3 AWiFS (Advanced Wide Field Sensor):

AWiFS is a wide-angle medium resolution (56 m) camera with a swath of 740 km (FOV=±25º) of WiFS heritage. The pushbroom instrument operates in three spectral bands which are identical to two VNIR bands (0.62 - 0.68 µm, 0.77 - 0.86 µm) and the SWIR band (1.55-1.70 µm) of the LISS-3 camera. The AWiFS camera is realized using two separate optoelectronic modules which are tilted by 11.94º with respect to nadir. Each module covers a swath of 370 km providing a combined swath of 740 km with a side lap between them. The wide swath coverage enables AWiFS to provide a five-day repeat capability. The optoelectronic modules contain refractive imaging optics along with band pass interference filter, a neutral density filter and a 6000 pixels linear array CCD detector for each spectral band.

Figure 3-31 Illustration of the AWiFS-A camera

The in-flight calibration is implemented using 6 LEDs in front of each CCD. For the VNIR bands (B2, B3, B4), the calibration is a progressively increasing sequence of 16 intensity levels through exposure control. For the SWIR band, the calibration sequence is similar to that of LISS-3 through a repetitive cycle of 2048 scan lines.

Table 3-15 Summary of the IRS-P6 instrument parameters

Parameter/Instrument	LISS-4	LISS-3	AWiFS
Spatial resolution or IFOV (Instantaneous Field of View)	5.8 m	23.5 m	56 m (nadir) (70 m a swath edge)
Spectral bands (µm)	B2: 0.52-0.59, (green) B3: 0.62-0.68, (red) B4: 0.77-0.86 (NIR)	B2: 0.52-0.59, (green) B3: 0.62-0.68, (red) B4: 0.77-0.86, (NIR) B5: 1.55-1.70 (SWIR)	B2: 0.52-0.59, (green) B3: 0.62-0.68, (red) B4: 0.77-0.86, (NIR) B5: 1.55-1.70 (SWIR)
Swath width	23.9 km in MS mode 70 km in PAN mode	141 km	740 km

Detector line arrays x No of elements	1 x 12,288 PAN mode 3 x 12,288 MS mode	4 x 6,000	4 x 2 x 6,000
Data quantization	10 bit (selected 7 bit are provided to the data handling system)	7 bit (VNIR), 10 bit (SWIR)	10 bit
Square wave response at Nyquist	> 0.20	B2> 0.40, B3> 0.40 B4> 0.35, B5> 0.20	B2> 0.40, B3> 0.40 B4> 0.35, B5> 0.20
Power consumption	216 W	70 W	114 W
Instrument mass	169.5 kg	106.1 kg	103.6 kg
Date rate	105 Mbit/s	52.5 Mbit/s	52.5 Mbit/s

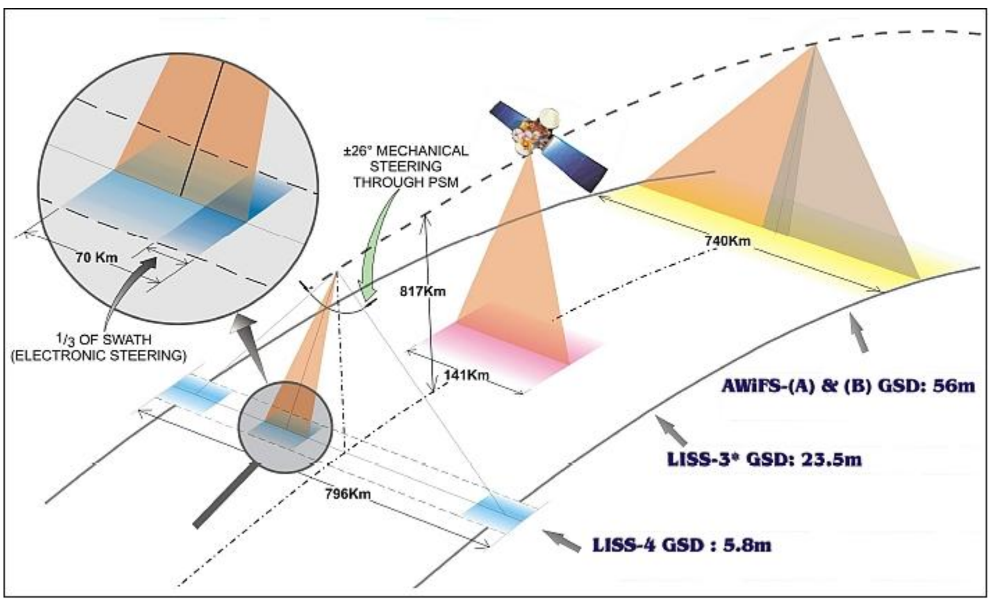


Figure 3-32 IRS-P6 Three tier imaging and swath coverage

3.6 Resourcesat-2

3.6.1 Introduction

Resourcesat-2 is a follow on mission to provide service continuity to the Resourcesat-1 users. Hence Resourcesat-2 payload systems were conceived around IRS-P6 with certain improvements in payload electronics. Resourcesat- 2 spacecraft is configured with improved features like 70 km Mx data, Enhanced SSR memory, new data handling system, 10/8 channels SPS, indigenously developed star sensor, AOCE with Mil-1553 interfaces. Apart from Resourcesat -1 payloads, a new payload called Hosted Indian payload from COMDEV, Canada for automatic identification of ship was also flown.

3.6.2 Mission Objective

Mission objective of Resourcesat-2 are

- To provide continued remote sensing data services on an operational basis for integrated land and water resource management at a micro level with enhanced multispectral/ spatial coverage and stereo imaging.
- To further carryout studies in advanced areas of user applications like improved crop discrimination, crop yield, crop stress, and pest / decease surveillance and disaster management etc.

3.6.3 Orbital Parameters

Table 3-16 Orbit details of Resourcesat-2

Parameter	Resourcesat-2
Orbit	Polar sun synchronous circular
Altitude	817 km
Inclination	98.69 deg
Eccentricity	0.0004
Period	101.35 minutes
Local Time	10.30 A.M
Repetivity Cycle	341 Orbits 24 days for LISS-3 5 days for AWiFS 5 days Revisit (LISS-4)
Distance between adjacent Traces	117.5 km
Off Nadir coverage +/- 26 deg	398 km (for LISS-4)
Distance between successive Ground tracks	2820 km
Ground Trace velocity	6.65 km/s

3.6.4 Salient features of Spacecraft

Table 3-17 Changes with respect to Resourcesat-1(IRS-P6)

Subsystem	Changes	
Mission modes	LISS-3 Operation remain same LISS-4 multispectral data of 23.5 km swath possible real time for India, AWiFS Operations remains same	
Payload	LISS-3, LISS-4 & AWiFS .Electronics miniaturized using FPGAs and MLBs with Interfaces remaining same with other subsystems.	
BDH	BDH formatter design changed to transmit 23.5 km Mx data in Indian region.	
SSR	Imported SSR is replaced by indigenous SSR. There are two SSRs, one for LISS-4 (200 Gb) and one for LISS-3 & AWIFS	
X-Band Systems	Similar to IRS-P6 with data transmitted by TWTA	
Power	Similar to IRS-P6 payload. DC/DC converters were C'UK type in IRS-P6 and Push-pull Type in Resourcesat-2	
Telemetry	Similar To IRS-P6	
AOCE	Hardware is similar to IRS-P5 with software similar to IRS-P6, with additional two DHRT cards for incorporating additional RT for SSR and Processing Software for computing ampere hour information	
Telecommand	Hardware similar to IRS-P6 with improvement in S/W based on experiences of IRS-P6	
TTC RF	Similar to IRS-P5 due to obsolete components of IRS-P6	
SPS	SPS was without 1553 interface in P6.	
Sensors	All are similar to P6 except Star sensor which is similar to P5 Star sensor with 1553 interface.	
IISU Elements	Reaction Wheels	Hall effect sensor used for position sensing instead of inductive sensor and associated electronics. No commutative electronic inside the wheel Mass 3.8 kg (4.2 kg for P6)
	IRU	All Specifications are similar to P6 with design and configuration similar to carto-2
	SADA	Configuration is similar to P5 with stepper motor torgue increased to 4 NM from 1 NM specification of IRS-P6. All Driving modes and schemes similar to P5
	PSM	Payload steering Mechanism and electronics similar to P6

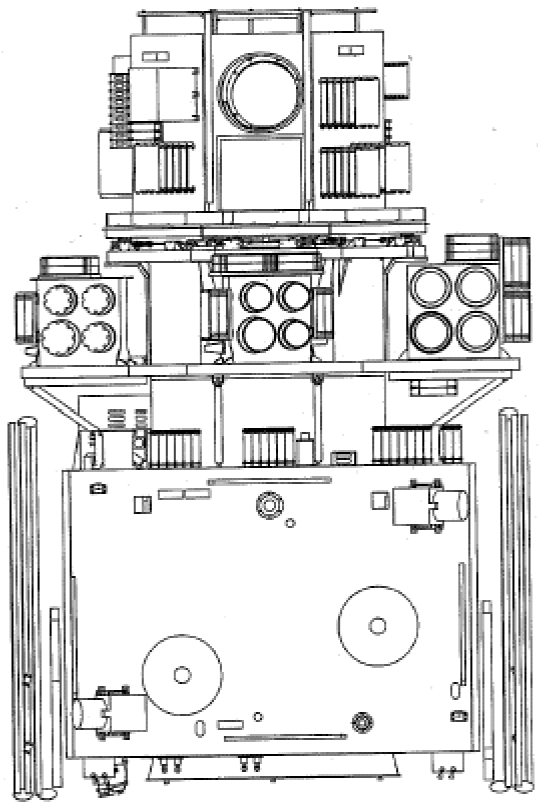


Figure 3-33 Stowed view of Resourcesat-2

Table 3-18 Salient features of Resourcesat-2

Subsystem		Resourcesat-2
Structure		Similar to IRS-P6.
Thermal	Control	Temperature control is with passive techniques using Paints, multilayer blankets, Optical solar Reflector, and active thermal elements like heaters. Heat pipe radiator panel is used to maintain the temperature of LISS-4 detector head assembly.
	Limits	All electronics packages 0-40degC, Battery 0-10 degC, Payload EO modules : 17 to 23
Mechanism	Solar Panel	Solar panel deployment and Drive Mechanism
	LISS-4	Hold down-release and steering mechanism for the LISS-4 Payload to steer +/- 26 deg. w.r.t. roll.
Power	Solar Panel	Sun tracking, rigid,15.12 M^2 6 panels 1.4 x 1.8 m^2 (Each), 1250 W at EOL, BSR(SCA)
	Battery	2 batteries, 28 to 42V, 28 Cells, Ni-Cd 24 AH
	Power Electronics	PWM TCR, FCL, 10 Strings

TTC	Telemetry	1024 words/frame, 2250 MHz, storage: 6.29 x 10^6 Bits PCM/PSK/PM, 16 Kbps
	Telecommand	PCM/FSK/FM/PM 2071.875 MHz
Data Handling		In X-band at a data rate of 105 Mbps. The BDH has two separate chains, one for LISS-3 and AWiFS data, and the second chain for LISS-4 data. The LISS-4 data on carrier-1 at 8125 MHz and LISS-3 + AWiFS data on carrier-2 at 8300 MHz.
Data Transmission		40 watts TWTA used. 210 MBPS (52.5 x 4) data Handling system Developed.
AOCS	Spec.	Pointing:: Yaw: $\pm$ 0.05° Roll: $\pm$ 0.05° Pitch: $\pm$ 0.05° (3 sigma) Drift rate : 4.8 x 10^{-5} deg/sec (3 sigma)
	Sensors	Earth sensor(2), DSS(2), Star Sensors(2), 4PiSS(4), Magnetometer (2) IRU(3 DTG), SPS (10/8 Channel)
	Actuators	Reaction Wheels, 10 NMS (4 in tetrahedral), Magnetic Torquers (2) , 1N Thrusters(8) 11 N Thruster(4) Fuel (100 kg)
SADA		**Configuration is similar to P5 with stepper motor torque increased to 4 NM from one NM specification of IRS-P6.**
Payloads		LISS-4, LISS-III*, AWiFS and COMDEV
Mass		1250 kg

3.6.5 Resourcesat-2 Payloads

Resourcesat-2 payload system consists of Four Payloads namely

1. Linear Imaging Self Scanning Sensor-4 (LISS-4)
2. Linear Imaging Self Scanning Sensor-3 (LISS-3)
3. Advanced Wide Field Sensors (AWiFS)
4. Automatic Identification of Ship (AIS)

3.6.5.1 Linear Imaging Self Scanning Sensor (LISS-4)

3.6.5.1.1 Introduction

LISS-4 is a high resolution multi-spectral camera with three spectral bands namely B2, B3 and B4 similar to those of LISS 3* and AWiFS camera. This camera operates in three spectral bands B2 (0.52-0.59 µm), B3 (0.62-0.68 µm) and B4 (0.77-0.86 µm). The ground resolution of LISS-4 will be 5.8 m with the swath of 70 km from an altitude of 817 km.

The three spectral bands are realized using field-splitting technique near the focal plane. The final selection of the spectral bands is achieved by using appropriate band pass filters in front of the detectors.

3.6.5.1.2 <u>LISS-4 Specifications</u>

Optical system	
Type	Off axis unobscured three-mirror telescope
Focal length	980mm
F-number	4.0
Spectral bands	B2(0.52-0.59µm) B3(0.62-0.68µm) B4(0.77-0.86µm)
Field of view (FOV) Across track Along track	 + 2.5º +0.4º & -0.6º
Telescope MTF	>40% at 70 lp/mm
Optical efficiency	0.6
Detector (CCD)	
No of pixels	12000
Pixel size	7µm x 7µm
No of output ports	8
Separation between odd - even rows	35µ (5 scan lines)
System	
IGFOV (m) Across track Along track	 5.83 5.82
Swath (km)	70.0 (Mono & Mx mode)
Integration Time (ms)	0.8777142
Quantization	10 bits (7 bits transmission to BDH after DPCM)
SNR (at saturation)	>128
SWR (%)	>20
BBR (pixel)	≤+ 0.25
Saturation Radiance (mw/cm^2/sr/µm)	B2 53 B3 47 B4 31.5
Raw Bus Power (W) Imaging mode Calibration mode	 126 (all bands) 127.7 (all bands)
Size (p x r x y) (mm)	742 x 596 x 888
Weight (kg) EO module Camera	 95 104

3.6.5.1.3 LISS-4 System configuration

Electro optical module (EOM)

The Electro optical module (EOM) of LISS-4 consists of three mirror assemblies, focal plane splitter optics and Detector Head Assembly (DHA) at specified locations. The telescope is a three mirror off-axis reflective system (similar to IRS-1C/1D PAN telescope).

Optical System

The optical system of LISS-4 consists of three mirrors unobscured off-axis telescope (an off-axis concave hyperboloid primary mirror, a convex spherical secondary mirror, and an off axis concave oblate ellipsoid tertiary mirror), focal plane splitter assembly, and band-pass filters. A 245 mm diameter primary mirror collects the radiation from earth and reflects it on the secondary mirror. The beam reflected from secondary mirror falls on to tertiary mirror, which focuses the beam on to the detector. Three focal planes are realized by splitting the field in the along track direction using an isosceles reflecting prism with a slot. The beam corresponding to B3 is transmitted through the slot while the B2 and B4 are reflected by prism sides. The placement of band pass interference filter in front of CCD ensures the selection of required band.

The telescope has an effective focal length of 980 mm and covers a field of view of ± 2.5º in across track and + 0.4º and -0.6º in the along track directions.

Detector Head Assemblies (DHA)

LISS-4 payload has three detector head assemblies (DHA) corresponding to B2, B3 and B4 respectively. LISS-4 DHA for Resoucesat-2 is identical to Resoucesat-1 in terms of interfaces and basic philosophy, but has been realized with reduced size and number of components. Also improvement has been carried out in thermal interface.
Each DHA consists of

- 12K linear array CCD
- Bias voltage generating circuits
- Clock driver circuits
- Heaters and thermistors for thermal control
- LEDs for On-board Calibration

The 12K element linear CCDs of Thomson make (TH31543) are used for each spectral band. Each CCD has a pixel size of 7 µm x 7 µm. The Odd and Even pixel rows are arranged in a staggered mode separated by 35 µ (equal to 5 scan lines). Each CCD gives analog data on eight output ports - four for odd pixels and four for even pixels. Each port provides data for 1520 pixels including 20 pre-scan/ white reference pixels. CCD has in-built anti-blooming and integration control.

DHA receives +18.3 V DC regulated voltage from power package and generates various bias voltages required for CCD operations using series regulator. DHA receives clock signals at a TTL levels from timing and control logic circuits of Camera Electronics (PLE12/13/14) and conditions them to suitable voltage levels and drives the required capacitive loads of CCD using clock driver circuits. CCD requires a total of 16 clocks for its operation.

Video data obtained from eight video ports are given to video processor circuit (PLE12/13/14 of camera electronics).

Each band DHA consists of two identical PCBs and each PCB caters to electrical requirements of four ports. DHA also receives +5.6 V regulated DC voltage from spacecraft to be applied to heaters to maintain CCD temperature.

The CCD temperature increases considerably whenever DHA is powered. In order to control the temperature excursion in the CCD, heaters are placed near the CCD, which are switched ON whenever DHA is switched OFF and vice versa. This ensures minimum change in CCD temperature at any time. CCD temperature needs to be maintained within 20±2 ºC, hence DHA is cooled. To achieve temperature control, control heaters are provided on the DHA. This is realized using a copper braid whose ends are terminated with copper blocks. A compensatory heater of 1.8 W (equal to CCD dissipation) is switched ON and OFF as complementary to LISS-4 OFF and ON respectively.

3.6.5.2 Linear Imaging Self Scanning Sensor (LISS-3*)

3.6.5.2.1 <u>Introduction:</u>

The LISS-3* Camera is a medium resolution multi-spectral camera operating in four spectral bands - B2, B3, B4 in Visible - Near infrared Range (VNIR) and B5 in Short Wave Infrared Range (SWIR) . This camera is similar to the LISS-3* of Resoucesat-1. LISS-3* will have four spectral bands with independent optical assemblies and a linear array detector for each channel providing identical IGFOV of 23.5 m. All bands will provide 100% albedo coverage with 1023 levels of quantization. It may be noted that Resoucesat-1 could provide limited albedo coverage with 127 radiometric levels and four selectable sliding gains for VNIR bands. The block schematics of LISS 3* (VNIR&SWIR) payload is shown in figure 3.6 and 3.8 resp.

The SWIR band is designed using a new custom built 6000 element Indium Gallium Arsenide (InGaAs) CCD. Based on the experience of IRS-1C/1D in-orbit performance, certain improvements have been incorporated in the LISS-3* design. LISS-3* SWIR band has improved performance in terms of resolution with 23.5 m compared to 70 m in IRS-1C/1D. The focal length of the SWIR band is modified to meet the improved resolution.

3.6.5.2.2 <u>LISS3* Specifications:</u>

	B2, B3 & B4	B5
Optical System		
EFL (mm)	347.5±0.3	451.75±0.3
F-number	<4.5	<4.5
FOV (deg)	+5	+ 5

Spectral bands	B2(0.52-0.59µm) B3(0.62-0.68µm) B4(0.77-0.86µm)	B5 (1.55 -1.70 µm)
Field of view (FOV) Across track Along track	 + 2.5º +0.4º & -0.6º	
Telescope MTF	>40% at 70 lp/mm	
Optical efficiency	0.6	
Detector (CCD)		
No. of pixels	6000	6000
Pixel size	7µm x 10µm	13µm x 13µm
No. Of output ports	2	2
Detector	Si	InGaAs
System		
IGFOV (m)	23.5	23.5
Along track sampling (m)	22.0	22.11
Swath (km)	141	141
Integration Time (ms)	3.32	3.32
Quantization bits	10 bit	7 bit
SNR (at saturation)	>128	>128
SWR (%)	B2 >30 B3 >30 B4 >20	B5 >20
BBR (pixel)	≤+ 0.25	≤+ 0.25
Saturation Radiance ($mw/cm^2/sr/\mu m$)	B2 53 B3 47 B4 31.5	B5 7.5
Raw Bus Power (W) Imaging mode Calibration mode	 72.4 (All Bands) 74.9 (All Bands)	
Size (P x R x Y) (mm)	493x470.5x626.31	
Weight (kg) EO module Camera	 73.2 74.9	

3.6.5.2.3 System Configuration

Each band has individual optics, DHA and camera electronics (CE). Four identical DHAs, one each per band forms part of the EO module. The major constituents of the payload are described below.

Optics: The LISS-3* camera uses refractive optics for all four spectral bands. The collecting optics consists of 8 refractive lens elements with the interference filter and the thermal filter. The optical configuration consists of a multi-element lens assembly. All the lens elements have spherical surface profile. Lenses of all four bands are developed by LEOS.

Detectors: For the 3 VNIR bands, 6000 elements devices (CCD 191A) with a pixel size of 10 μX7 μ on a pixel pitch of 10 μ with two video output ports is used. For the SWIR band, a 6000 element staggered array device with a pixel size of 13 μX13 μ on a pixel pitch of 13 μ and line pitch of 26 μ with two video output ports is used.

Detector Head Assembly for VNIR: DHA houses linear array CCD-detector, PCB, onboard calibration LEDs and mechanical mount. LISS 3* will have three VNIR band DHAs.

Detector: For the 3 VNIR bands, 6000 elements devices (CCD 191A) with a pixel size of 10 μ X 7 μ on a pixel pitch of 10 μ with two video output ports is used.

Description of CCD 191A Device:

The CCD 191A has 6000 photosensitive elements each of size 10μm along the array and 7 μm perpendicular to array length. The device has two video output ports and packaged in a custom build 40 pin DIP ceramic package. CCD 191A is fabricated using advanced n-channel isoplanner buried channel technology.

The photosensitive elements accumulate charges during integration period. These accumulated charges are transferred to two analog shift register using transfer clocks. Analog shift register transport these charge packets sequentially, with the help of 2-phase shift clocks, to charge amplifier where charge to voltage conversion takes place and three levels analog voltage signal is available on each port. The true video information is carried out by taking difference of reference level and video level. Both the ports start with 10 prescan reference pixels followed by 3000 photo-responsive pixels. The ports here correspond to the even and odd pixels respectively.

Anti-Blooming control: In order to take care of the super saturation problems seen in IRS-1C/1D in-orbit, Anti-blooming control (ABC) feature of CCD191A is incorporated in the VNIR bands of LISS3*. This feature is used to arrest the Raw-Bus current increase when camera is exposed to higher illumination compared to its saturation settings. The ABC is implemented by proper setting of 'Integration Control' bias voltage (V_{IC}) available on the CCD pin and is adjusted by trimming the bleeder resistors in the bias supply circuit in the DE package. With this modification, the performance was verified with 100% saturation radiance input and raw bus current was monitored in all 4 Gains. It is seen that the current increase is about 4 mA (max) as against 160 mA increase seen without this ABC feature.

Onboard Calibration: Calibration assembly consists of LED wired on PCB and mounted on LED holders. Four LEDs connected in series mounted on LED-holder. Two such holders one on each side of CCD are mounted on DHA plate/flange. All LEDs are connected in series. A DC current of 16±1mA is passing through LEDs. Calibration levels are generated using exposure control feature of the CCD. Onboard calibration is to be carried out using 8 LEDs (02 sets of 4 LEDs) covering the complete array. LED profile depends on the LED mounting geometry. 16 levels will be generated using integration control feature. The LED intensity is expected to vary with temperature.

Mechanical Mount: The PCB is mounted on the mechanical mount made from lnvar and back cover of aluminium. The calibration LED assembly is also mounted on the same mount.

Detector Head Assembly for SWIR: The detectors used for SWIR channels in RS-1/2 are of type TH31906. This detector uses modular approach. Each module contains 600 photodiodes arranged in staggered fashion and CMOS multiplexers on either side of the array for even and odd pixels readout. A total of 10 such modules are butted together to form a linear array of 6000 pixels. Two consecutive pixels are lost at each splice due to butting. Photosensitive area of each pixelis13 µmX13 µm with 13 µm pitch along the length of the array. The odd and even lines of the staggered configuration are separated by 26 µm. the photodiodes and CMOS MUX are glued on a 2 mm thick co-fired ceramic plate. Electrical interconnections are provided by either gold coated tracks or wires.

The photodiodes in TH31906 are operated under near zero bias condition. This is ensured by placing a suitable resistance between VREF and ADJREF pins.

In order to avoid reflections from the focal plane which manifests itself as ghost image after re-reflection from interference filter which is part of camera optics, most of the focal plane is masked and 1 mm wide and 83 mm long slit which is 1mm above the surface of photodiode die exposes photodiodes to the incident radiation. But reflections from the edges of this slit cause some extra illumination in few of the pixels. In order to avoid this, two external slits of appropriate dimensions are placed near optics exit so that the edges of the slit on mask are properly shadowed.

3.6.5.3 Advanced Wide Field Sensor (AWiFS)

3.6.5.3.1 <u>Introduction</u>

The Advanced Wide Field Sensor (AWiFS) camera will be catering to the high temporal resolution requirement of RS-2 mission with revisit period of 5 days. It has IGFOV of 70m from an altitude of 817 km. The AWiFS camera consists of four spectral bands, three in the visible and in near IR (VNIR B2, B3 and B4) and one in the short wave infrared (SWIR B5) similar to AWiFS of Resoucesat-1.

AWIFS is configured as a set of two identical camera modules i.e. AWIFS-A &AWIFS-B. Each camera consists of four lens assemblies, detectors and associated electronics pertaining to the four spectral bands B2, B3, B4 and B5. The two cameras are combined to generate the required field of vie commensurate with the desired swath. The imaging concept is based on push broom scanning that uses a linear array CCD placed in the focal plane of the optics. The 4 spectral bands are realized using independent refractive optical assemblies. To generate the required swath along with the desired overlap of 150±20 pixels, the two EO modules will be mounted on the spacecraft deck such that they are squinted with respect to nadir by ±11.84º. The field of view of each lens assembly is ± 12.5º. In nutshell, total field coverage of 47.94º is shared equally by two optical heads for each of the four bands.

3.6.5.3.2 AWiFS Specifications

	B2, B3 & B4	B5
Optical System		
EFL (mm)	139.5 + 0.15	181.35 + 0.2
F-number	<5.0	<5.0
FOV (deg)	±12.5	±12.5
Spectral Bands	B2(0.52-0.59µm) B3(0.62-0.68µm) B4(0.77-0.86µm)	B5 (1.55 -1.70 µm)
Detector (CCD)		
No. of pixels	6000	6000 * 8
Pixel size	7µm x 10µm	13µm x 13µm
No. of output ports	2	2
Detector	Si	InGaAs
System		
IGFOV (m)	56 (@ nadir), 70(off-nadir)	56 (nadir) 70(off-nadir)
Along track sampling (m)	66	66
Swath (km)	740	740
Integration Time (ms)	9.96	9.96
Quantization bits	12(MLG)	12(MLG)
SNR (at saturation)	> 512	> 512
SWR (%)	B2 >30 B3 >30 B4 >20	B5 >20
BBR (Pixel)	≤± 0.25	≤± 0.25
Saturation Radiance ($mw/cm^2/sr/\mu m$)	B2 53 B3 47 B4 31.5	B5 7.5
Calibration levels	16	6 non-zero
Raw Bus Power (W) Imaging mode Calibration mode	 124.1(All Band AWiFS-A&B) 126.7(All Band AWiFS-A&B)	
Size (P x R x Y) (mm) AWiFS-A	 471 x 410 x 316	

AWiFS-B	418 x 410 x 316
Weight (kg) EO module Camera	 28(AWiFS-A), 25.5 (AWiFS-B) 55.8(AWiFS-A + AWiFS-B)

3.6.5.3.3 System Configuration

Each band has individual optics, DHA and camera electronics (CE). Four identical DHAs, one each per band forms part of the EO module. The major constituents of the payload are described below.

Optics: The optics of AWiFS camera consists of two optical heads (two lens assemblies) for each of the four spectral bands to cover the full swath. Each lens assembly comprises a Thermal Filter, interference Filter, and 8 lens elements. In view of the required geometric/radiometric performance, f/5 system is employed for both VNIR and SWIR bands. All the lens elements have spherical surface profiles. The optics for the same is being developed at LEOS.

Detectors: For the 3 VNIR bands, 6000 elements devices (CCD191A) with a pixel size of 10 µX7 µ on a pixel pitch of 10 µ with two video output ports is used. For the SWIR band, a 6000 element staggered array device with a pixel size of 13 µ X 13 µ on a pixel pitch of 13 µ and line pitch of 26 µ with two video output ports is used.

Detector Head Assembly for VNIR: DHA houses linear array CCD-detector, PCB, onboard calibration assembly and mechanical mount. AWiFS-A & B will have three VNIR band DHAs.

Detector: For the 3 VNIR bands, 6000 elements devices (CCD 191A) with a pixel size of 10 µX7 µ is used.

Onboard Calibration Assembly: Calibration assembly consists of LED wired on PCB and mounted on LED holders. Four LEDs connected in series mounted on LED-holder. Two such holders one on each side of CCD are mounted on DHA plate/flange. All LEDs are connected in series. A DC current of 16±1 mA is passing through LEDs. Calibration levels are generated using exposure control feature of the CCD.

Mechanical Mount: The PCB is mounted on the mechanical mount made from lnvar and back cover of aluminium. The calibration LED assembly is also mounted on the same mount.

Detector Head Assembly for SWIR: The detectors used for SWIR channels in RS-1/2 are of type TH31906. This detector uses modular approach. Each module contains 600 photodiodes arranged in staggered fashion and CMOS multiplexers on either side of the array for even and odd pixels readout. A total of 10 such modules are butted together to form a linear array of 6000 pixels. Two consecutive pixels are lost at each splice due to butting. Photosensitive area of each pixelis13µmX13µm with 13µm pitch along the length of the array. The odd and even lines of the staggered configuration are separated by 26µm. the photodiodes and CMOS MUX are glued on a 2mm thick co-fired ceramic plate. Electrical interconnections are provided by either gold coated tracks or wires.

The photodiodes in TH31906 are operated under near zero bias condition. This is ensured by placing a suitable resistance between V_{REF} and ADJ_{REF} pins.

In order to avoid reflections from the focal plane which manifests itself as ghost image after re-reflection from interference filter which is part of camera optics, most of the focal plane is masked and 1mm wide and 83mm long slit which is 1mm above the surface of photodiode die exposes photodiodes to the incident radiation. But reflections from the edges of this slit cause some extra illumination in few of the pixels. In order to avoid this, two external slits of appropriate dimensions are placed near optics exit so that the edges of the slit on mask are properly shadowed.

Camera Electronics: Each camera and detector have independent camera electronics to cater to various functional and performance requirements. AWiFS-A accommodates hardware for Detector 1 of all bands and similarly AWiFS-B covers hardware for Detector 2 of all bands. In RS-2, all four bands use Multi Linear Gain (MLG) technique to provide 12 bit performance retaining 10 bit hardware interface. RS-2 camera electronics (CE) hardware is realized using passive SMDs, FPGAs, double sided or multi-layered PCBs and tray packaging. This has resulted in improvements with respect to size, weight and power. Similar approach has been adopted for SWIR electronics.

VNIR: Camera electronics is custom designed for Resourcesat-2 AWiFS camera. The system is configured to meet the functional and performance requirements with minimum hardware complexity. The salient features of AWiFS (VNIR) camera electronics are

- Separate camera electronics for each detector
- Separate detector-drive electronics for each detector
- Separate timing logic for each detector without redundancy(like in Resoucesat-1)
- Cross coupling of BRC and WLS
- Hot redundancy for data and telemetry

The electronic system design of Resourcesat-2 maximally uses the subsystems and circuit blocks designed and developed for Resourcesat-1, thereby improving reliability. Camera electronics is modular for each detector. The functional blocks of camera electronics consists of

- Bias generator
- Clock driver
- Cal LED Driver
- Video Processing Electronics
- Timing and Calibration Logic
- Telemetry and Telecommand Interface

Bias generator: Bias generator circuit consist of linear regulators and capacitor multiplier filters, which provides 4 regulated low noise bias voltages for detector operation. The circuit also incorporates short circuit protection for VDD. The circuit configuration is same as that used in OCM.

Clock Drivers: Detector electronics receives 9 clocks from payload electronics package for its operation in phased read out mode. Clock driver translate these TTL signals to MOS level with adequate capability to drive capacitive loads for realizing fast rise/fall times. The high level required by photo gate, transfer and reset clocks are typically 15 V and 10 V for transport and integration control clocks. Accordingly, two linear regulators are used to generate low noise supply voltages required for 15 V and 10 V.

CAL LED Driver: Calibration requires a light source in front of the CCD. IRS payloads uses solid-state LED based source. To drive LEDs a low noise constant current is generated. A regulator ie wired with LEDs in the feedback loop. The output gain resistors and the current deciding resistor at the inverting input of the error amplifier control the required current.

3.6.5.4 Automatic Identification of Ship (AIS)

3.6.5.4.1 <u>Introduction:</u>

COMDEV AIS Payload was flown in Resourcesat-2 as an experimental payload for ship surveillance in VHF band to derive the position, speed, start point and end point of ships. The VHF antenna which is going to be provided by CMG, ISAC consists of four orthogonally polarized monopole antennas (one is left hand circular polarized and another right hand circular polarized), placed at the edges of EP-01 panel , receive data from ships which may have horizontal or vertical movement because of sea tides. The data received from ships is stored in the onboard memory (4 GB) and it is transmitted through QPSK modulated S-Band carrier (2280 MHz) to ISTRAC and Canadian stations. The S-Band transmitter and Antenna will be supplied by ISRO. VHF data (160-162 MHz) is uplinked to AIS payload at 2.5 Mbps and stored in onboard memory of 4 GB which can simultaneously record and playback the data. AIS receiver will transmit data to S-Band transmitter on TTL interface without differential encoding with QPSK modulation. With QPSK, download data will be 16 Mbps data rate.

The AIS is a shipboard broadcast system. The AIS will improve security by increasing the Coast Guard's awareness of vessels in the maritime domain, especially vessels approaching ports. The AIS corroborates and provides identification and position of vessels not always possible through voice radio communication or radar alone.

The payload developed by COMDEV was flown onboard NTS-1 satellite Launched by PSLV C9 in April 2008. NTS-1 is a dedicated VHF Radio (160-162 MHz range) for the collection and monitoring AIS signals from space.

Ships can be identified anywhere in the Oceans/seas by receiving AIS signals. Unidentified ships, which can pose threat to security, can be figured out by spotting all the ships through radar network and isolating the unidentifiable ships near coastal zones.

AIS antennae Operate in the VHF maritime band (160 -162 MHz) standardized digital communication protocols. Each station transmits and receives over two radio channels to avoid interference problems. Transmissions use 9.6 Kb GMSK/FM modulation. Uses Self-Organizing

Time Division Multiple Access (SOTDMA) technology to meet this high broadcast rate and ensure reliable ship-to-ship operation. Each station determines its own transmission schedule (slot), based upon data link traffic history and knowledge of future actions by other stations.

- The data received from satellite is processed offline at Bangalore and Canada.
- COMDEV agreed to provide ships data, processed at Bangalore and Canada, for Indian region and Indian ships data throughout the world to ISRO.

3.6.5.4.2 Payload Description

The AIS payload is designed to perform AIS signal receive, store and forward functions covering the two AIS frequencies of 161.975 MHz and 162.025 MHz. The payload is comprised of

- Two AIS Receive antennas, circular polarization (one left handed, one right handed).
- Two RF cables between the AIS antennas and the AIS Receiver.
- An AIS Receiver provided by COMDEV that provides the power conditioning, RF front end, digital control, data storage, data conditioning and data transmission to the S-Band downlink
- ISRO's cables to carry the AIS data signal from the AIS Receiver to the S Band Transmitter.
- An S Band Transmitter provided by ISRO, 16 Mbps QPSK (the transmitter will receive two signals at 8Mbps each, used as I and Q signals for the modulation).
- An S Band Antenna provided by ISRO

3.7 Resourcesat-2A

3.7.1 Introduction

Resourcesat-2A (RS-2A) mission is a continuity mission of Resourcesat series (RS-1, RS-2) to provide continuity of operational services of AWiFS, LISS-3 and LISS-4 payloads. Primarily the spacecraft configuration of RS-2A is similar to RS-2 with few incremental changes in some of the systems to take care of the in-orbit observations of RS-2 and improve the overall mission operations management.

The payload system comprises of three optical remote sensing cameras, viz., LISS-4, LISS-3 and AWiFS. LISS-4 provides 5.8m resolution in three bands with 70 km swath, LISS-3 provides 23.5m resolution in four bands with 140 km swath and AWiFS camera provides with a spatial resolution of 56m in four bands with 740 km swath.

A new payload Solid state C Band Transponder (SCBT) is added in Resourcesat-2A to aid the calibration of C-Band Radars at SDSC, SHAR. HIP (Hosted Indian Payload) payload from COMDEV, Canada flown in Resourcesat-2 is not planned in Resourcesat-2A.

Resourcesat-2A will cater to several applications in the areas of natural resources monitoring and management like

- Discrimination of multiple crops and their condition assessment

- Precision farming and sampler for crop yield estimation
- Discrimination of forest types & forest inventory monitoring
- Surface soil wetness zonation
- Regional land use/ land cover mapping
- Snow-cloud discrimination & Glacier inventory
- Flood and drought monitoring, damage assessment and mitigation management
- Coastal zone mapping

3.7.2 Mission Objectives:

Following are the main objectives of the Resourcesat-2A mission

- To provide continuity of on-going services of Resourecesat-2 and ensure in-orbit redundancy of the satellite.
- Increased frequency of observations in tandem with Resourcesat-2.
- To explore new application areas in Land and Water Resources monitoring & management.

3.7.3 Orbit Parameters

Parameter	Value
Orbits/ Cycle	341
Repetivity (LISS-3)	24 days
Revisit(LISS-4 & AWiFS)	5 days
Altitude	817 km
Semi Major Axis (Mean)	7195.11 km
Inclination (With Bias)	98.719 deg
Inclination Bias	0.02 deg
Period	101.35 minute
Distance between Adjacent Traces	117.5 km
Distance between Successive Tracks	2828 km
Ground Track Velocity	6.65 km/sec
LISS-4 Coverage with Steering of ± 26 deg.	±398 km

Deployed configuration of Resourcesat-2A is shown in figure 3-34, stowed configuration of Resourcesat-2A is shown in figure 3-35.

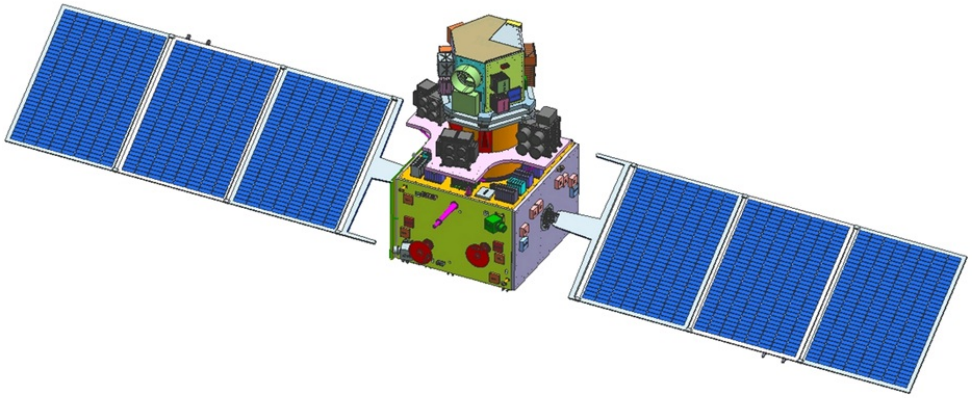


Figure 3-34 Deployed configuration of Resourcesat-2A

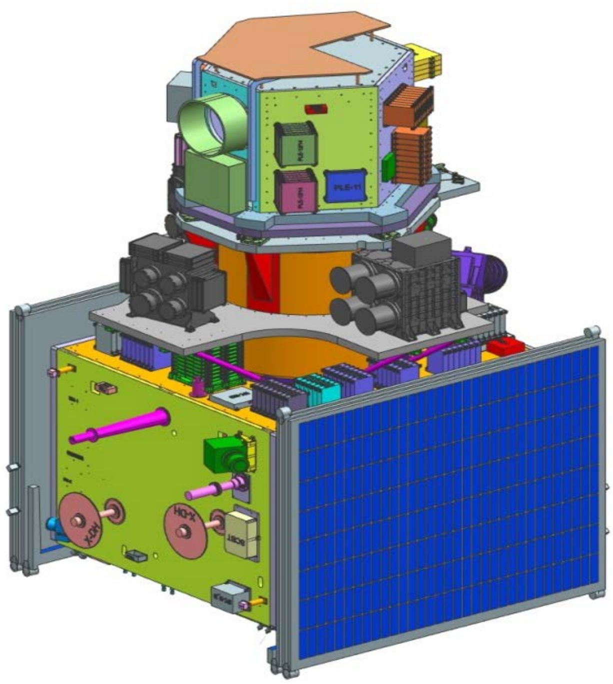


Figure 3-35 Stowed configuration of Resourcesat-2A

3.7.4 New Features/ Improvements w r to Resourcesat-2

- Reuse of the uplink & Downlink (TTC) frequencies for Resourcesat-2 and Resourcesat-2A by phasing. Resourcesat-2A is phased with Resourcesat-2 to get better revisit (12 days for LISS-3 and about 2-3 days for L4 MX, AWiFS)
- Moon-imaging based radiometric calibration has a number of advantages over other existing methods of vicarious calibration because, the reflectance properties of moon surface do not

change over a large time period (years), hence making it a stable source, also there are no disturbances related to atmosphere. In addition to LED based calibration added Moon Calibration for radiometric calibration of LISS-3 & AWiFS using moon-imaging to improve the radiometric calibration of detector. Moon Imaging is planned once in six months for AWiFS and LISS-3.

- Due to change in AWiFS SWIR CCD, Multi Linear Gain (MLG) Scheme is removed for SWIR Band. AWiFS SWIR band 14 bit data (without MLG) will be formatted by BDH (discarding 2 LSB data).
- Out-of-Plane & In-Plane maneuver sequencer (Rotation Operation & Yaw Steering Operation Enable / Disable), operation made automated to take care of automatic orbit maneuver with reduction in ground commands.
- Auto-heater, Contingency heater configuration modified to avoid single point failure by grouping appropriately.
- Limitation of sending real time (RT) command when Telecommand Processor (TCP) is executing the sequence of command. This limitation is overcome by the modification in the processor card.
- The AWiFS A & B payload requires the overlap requirement of 150±20 pixels. To achieve the overlap, PPL Deck Payload Interface inserts potting process modification to achieve better positional accuracy.
- Event Based Commanding provision for unforeseen observation and onboard correction. This feature will monitor the parameters and based on the event occurrence, initiate the necessary action without ground intervention.
- DPCM bypass mode with RICE compression for LISS-4 and LISS-3 Payload implemented at Data Handling System.
- Playback with Segment Erase and append data into a single file features are added in SSR to improve payload operational efficiency.
- Thrust on Mission mode of operation during the spacecraft level testing by participation of ISTRAC, Mission and NRSC.
- Operational constraints of PSM rotation Disable, SADA rotation enable, SSR Stop commands addressed and included in Payload macros at appropriate place.
- Macro based payload sequencer instead of fixed sequencer to provide flexibility for Mission operations to meet the following requirements
  - NRSA Data Center (NDC) Requirement
  - AWiFS SWIR requirement
  - Independent Payload operation
  - Flexibility Provision
  - Future requirement

3.7.5 Resourcesat-2A Payloads

3.7.5.1 Linear Imaging Self-Scanning Sensor (LISS-4)

3.7.5.1.1 <u>Changes/Improvement with respect to Resourcesat-2</u>

✶ **Presence of streaks in the images of LISS-4 of RS-**2

- An add-on card is incorporated to prevent blooming in LISS-4 DHA by changing PHI-A and PHI-P clock levels.
- After incorporating, the subsystem was tested at qualification level and simulated.
- The approved changes have been implemented in all FM DHAs.

✶ **Inclusion of DPCM Bypass Mode (10 bit data to BDH)**

- RS-2 LISS-4 also had the provision for DPCM bypass at payload level; but was not available on board.
- LISS-4 P/L has 10 bit data interface with BDH. Data is not DPCM encoded. At BDH this data undergoes lossless compression before transmission.
- Mode is Telecommand selectable.
- Implementation of this mode requires new design at BDH; one additional TC and one TM; additional on board harness in AIT.

✶ **Optical Telescope Assembly**

- Modification of MFD's due to change in mirror thickness
- Modification of stray light cover for better WFE performance of camera.

✶ **DHA**

- Usage of cover to close the gap between DHAs & hub of LISS-4 using long washer screws to prevent contaminations.
- Addition of screw in LISS-4 DHA assembly near 3W3 video connector (3 DHAs).
- Usage of pit in PCB mount of DHA of LISS-4 to accommodate spiral joint of heater wires (3 DHAs).
- Usage of the black paint inside LISS-4 DHA hub to reduce stray light.
- Usage of special connectors (2 pieces) for band-2 DHA of LISS-4 near base plate for ease of temp sensors.

✶ **Moon Calibration provision for the spatial characterization of LISS-4 sensor.**

3.7.5.2 Linear Imaging Self-scanning Sensor (LISS-3*)

3.7.5.2.1 <u>Changes/Improvement with respect to Resourcesat-2</u>

The following is the list of changes carried out in various subsystems with respect to Resourcesat-2 and all changes have been approved by CMRB-II.

Camera Electronics

- Modification of FPGA logic to accommodate 101 dummy pixels instead of 100 dummy pixels for SWIR.
- Implementation of DPCM Enable *I* Disable tele command.
- Increase in operating calibration LED current from 0.9 mA & 1.1 mA to 1.1 mA & 1.3 mA respectively for SWIR.
- Implementation of two capacitors in series for BCD card of VNIR.
- Continuous 168 hours burn-in of new T & E ed logic card of PLE-22 of LISS-3* {VNIR) as per T & E Board's recommendation.

Optics

- Implementation of a tilt of 1° (about pitch axis) in the interference filter within lens assembly for each bands.
- Placement of an aperture plate at the rear end of lens assembly for VNIR bands, thereby allowing a field angle of ±1° in the along track direction.

DHA

- Placement of an aperture slit near to the focal plane over VNIR 6K CCD window for blocking unwanted reflecting metallic zones on either side of photosensitive area.
- Addition of mounting screw near to the detector for VNIR bands (B2, B3 & B4).

Mechanical

- Addition of button at both long ends of CCD on Device mount of DHA card of LISS-3 (VNIR).

Moon Calibration provision for radiometric calibration of LISS-3 and AWiFS using moon-imaging.

3.7.5.3 Advanced Wide Field Sensor (AWiFS)

3.7.5.3.1 Changes/Improvement with respect to Resourcesat-2

- **Presence of secondary reflections in the images of AWiFS of RS-2**
  This happens due to internal reflections. This is overcome by tilting the interference filter by 1 degree (Implemented in Lens assembly) and placing of field stop near the detector (Implemented in DHA). Further an aperture plate is introduced at the rear end of VNIR lens assemblies for controlling background/stray light. This was experimentally validated.

- **Camera Electronics**

1. Redesign of FPGA logic of VPL card of AWiFS (VNIR) CE to overcome usage of extra dummy card (6 packages).
2. Usage of two RM1206 style resistors in parallel in place of one RM1206 style in BCD card (6 packages)

3. Usage of two polarized capacitors in series in place of single capacitor in all bias & clk driver card AWiFS (VNIR) CE (6 packages)

List of changes for AWiFS SWIR chain

Optics

- Implementation of a tilt of 1° (about pitch axis) in the interference filter within lens assembly for each bands.
- Placement of an aperture plate at the rear end of lens assembly, thereby allowing a field angle of ±1° in the along track direction.

DHA

- New 6000×8 pixels SWIR Detector is used instead of 6000×1 SWIR Detector.
- Pixels are true linear instead of staggered configuration of RS-2.
- DHA consists of DPE card and Detector connected by link wires instead of soldered detector on DHA PCBs in RS-2.
- Detector output is in the form of 14 bit serial digital data instead of analog output in RS-2 detectors.
- 25 pin D-Type connector will be used for heater, thermistor interface instead of 15-pin D-Type in RS-2.

Electronics

- Completely New Design except temperature controller.

3.7.5.4 Solid State C-Band Transponder (SCBT)

3.7.5.4.1 Introduction

Tracking is one of the most important mission requirements of any launch vehicle for range safety decision making during the initial phase of ascend and also to locate the position of the launch vehicle during flight. C Band Radars at SHAR in conjunction with C Band Transponders (CBT) onboard the launch vehicle provides the real time trajectory data for range safety decision making.

Tracking is part of range safety and hence range accuracy is very important. To achieve the intended specifications of the Radar, calibration is regularly carried out. This is to eliminate the bias and dynamic errors of SHAR Ground Radars. Usually bore sight calibration is employed, which is a static calibration technique. Radar calibration based on satellites is more precise. The calibration is carried out when the satellite is visible over SHAR horizon. The visibility period is approximately 20 minutes.

ISRO has already implemented satellite calibration methods by sending C Band transponders onboard IRS P3, P5 and SARAL. This technique could achieve accuracy close to the specifications. Magnetron based Mini CBTs were flown in IRSP3 & IRSP5. These transponders successfully completed the life time after providing the intended functions. Solid state CBT, flown

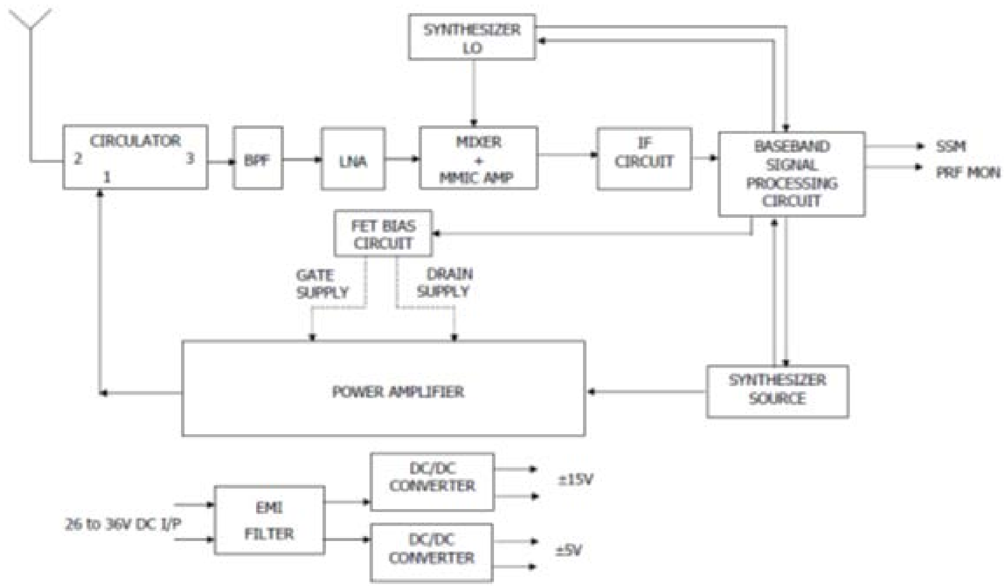


Figure 3-36 Block Diagram of SCBT

in SARAL spacecraft is still providing continuous support to SHAR for calibration activities. Further requirement has come from SHAR for another Transponder to be operationalized in satellite.

3.7.5.4.2 System Description

The SCBT consists of receiver, transmitter and interface circuits. The receiver accepts the interrogating signal from the radar and generates the detected pulse, on pulse-to-pulse basis, which is then delayed, processed and retransmitted at a different frequency.

The RF input from the antenna is fed to the C-Band circulator. The circulator is used to couple the transponder to the same antenna, which transmits and receives signal to and from ground tracking radars. The circulator also provides sufficient isolation between the transmitter and receiver and protects the receiver from the transmitter high power.

The received pulse modulated RF signal from the radar, after band limiting is down converted by the mixer to a level suitable for driving the IF logarithmic amplifier. The log amplifier detects the RF signal and produces a pulse, whose amplitude is proportional to the input RF signal level. This pulse is then applied to a threshold detector, which produces a TTL compatible output above the set threshold.

The output of the threshold detector is fed to base band signal processing circuit. The FPGA based base band signal processing circuit generates gate pulse for FET gate bias circuit and provides the internal delay and the inhibition to the Transmitter. PRF monitoring circuit, which is a frequency counter implemented in FPGA is also included along with base band processing, which is transmitted as telemetry data. The encode pulse and the latch required for signal strength monitoring is provided by this circuit.

The function of the Signal Strength Monitoring circuit is to monitor the strength of the received signal and transmit it as telemetry data. The pulsed signal strength measurement technique is a unique method to measure pulsed signal strength in which, the signal is first sampled and digitally processed. It is then stored and converted to analog form and telemetered.

Radar Calibration is carried out regularly before each launch to eliminate the bias and dynamic errors of SHAR Ground Radars. Tracking is done when the satellite is visible over the SHAR horizon. The tracking data is then processed offline to obtain the bias values for aiding calibration. Period of operation is approx 15 min during each switch ON. There is no predetermined time for switching 'ON'. It can be switched 'ON' as and when required.

3.7.5.4.3 SCBT Specifications

Receiver	
Type of Receiver	Super heterodyne
Centre frequency	5660MHz ± 2MHz
Sensitivity (stable output without missing pulse)	-70dBm min
Dynamic Range (input signal level)	0dBm to -70dBm
IF Bandwidth (3dB)	9MHz ± 2MHz
Image Rejection	60dB (min)
Pulse width	1µS ± 0.1µS
Spurious rejection for frequencies b/w 10 MHz to 10GHz (except for the range of f_0 ± 300MHz)	60dB min
PRF (Pulse repetition frequency) 1R 2R 3R	 585.5Hz 1171Hz 1756Hz
Maximum input signal	0dBm
Input impedance	50Ω
Transmitter	
Frequency	5510MHz/5790MHz (preset)
Frequency Settability	± 2 MHz

Frequency Stability	± 2 MHz under all operating conditions
Power (peak)	120 Watts (min)
Pulse width	1 ± 0.1μS
Pulse Rise time (10% to 90%)	≤200nS
Pulse Fall Time (90% to 10%)	≤100nS
Output Impedance	50 Ω
Transmitter Spectrum	Side lobe should be min. 10dB down from the main lobe
Transponder	
Transponder delay	5 to 35μS in steps of 5μS, 64μS max (preset)
System Delay Variation	100ns maximum, 0dBm to -65dBm
Transponder inhibition time	225μs ± 5μs (preset)
Power	28 ± 1V DC 0.5A(max) (operating current) 2.5A(max) (surge current)
UT	310 ± 30mA
0.5R	330 ± 30mA
1R	350 ± 30mA
1.5R	370 ± 30mA
2R	390 ± 30mA
3R	430 ± 30mA
Telemetry Output	
PRF monitoring (1R, 2R, 3R)	0V to 5 Volts; 3 discrete levels to Distinguish 1R, 2R, 3R
0.5R	0.5 ± 0.1V
1R	1 ± 0.2V
1.5R	1.5 ± 0.2V
2R	2 ± 0.2V
3R	3 ± 0.2V
Signal strength monitoring (-30 to -65dBm)	0 to 5 Volts
Mechanical	
Size	214mm x 174mm x 56mm
Weight	2 kg
Material	Aluminium B51SWP
Mounting holes	5.5mm dia, 6 No.s of through holes
Finish	All inner surfaces, mating surfaces with top deck,

	connector mounting surfaces and bottom surface is gold plated. All other surfaces are anodised black. No hermeticity.

4. Ocean and Atmospheric Observation Series

4.1 IRS-P4 (Oceansat-1)

4.1.1 Introduction

The oceans occupying more than two-third surface area of Earth, have great influence on the global climate, affecting the economy and day-to-day life of people. As the measurement of the oceanic parameters by conventional methods using ships, buoys and other in-situ methods is difficult and expensive, remote sensing method which give frequent, accurate updates and economical is preferred.

IRS-P4 in the series of Indian Remote Sensing Satellites (IRS) was designed to serve the applications in the area of oceanography. Ocean Colour Monitor (OCM) and Multi-frequency Scanning Microwave Radiometer (MSMR) were the two payloads. The OCM operated in the visible and near infra-red bands and MSMR in Microwave bands. Both the payloads are configured to serve the application areas related to oceanography. Accordingly the satellite is called Oceansat-1.

These instruments were used to sense such important geophysical parameters as, chlorophyll content, yellow substance and suspended sediments in ocean waters; sea surface temperature, sea surface winds, water vapour in an atmospheric column, identifying the potential fishing zones, coastal zone management, ship routing, operations of offshore oil rigs and water content in clouds.

The 720 km altitude orbit was selected to achieve systematic coverage of the whole globe in two days considering the swaths of 1400 km. The satellite mainframe derives its heritage from the earlier IRS mission. The data from both the payloads were received and processed by National Remote Sensing Agency (NRSA) at Hyderabad.

4.1.2 Mission Objective

Mission objective of IRS-P4 are as follows

- To gather data for oceanograpics, land (vegetation dynamics) and atmospheric applications.
- To develop new application areas, using IRS-P4 data as complimentary / supplimentary to the data from already operating remote sensing satellites.
- To provide opportunity for conducting technological / scientific experiments that are of relevance for future developments.

4.1.3 Orbit Details

Table 4-1 Orbit details of IRS-P4

Parameters	Values
Orbit	Polar Sun-synchronous

Altitude (km)	720
Inclination (Deg)	98.27°
Equatorial Crossing Time (ECT)	12.00 noon (descending node)
Orbital period (Min)	99.31
Distance between adjacent orbital traces (km)	1382
Distance between successive ground traces (km)	2764
Repetivity	2 days (29 orbits)

4.1.4 Salient features of Spacecraft

Table 4-2 Salient features of IRS-P4

Subsystem		IRS-P4
Structure		Aluminum / aluminum honey comb with CFRP elements for MSMR payload structures
Thermal	Thermal control	Passive/ semi active thermal control with paints, MLI blankets, OSR and close loop temperature control
	values	All electronics 0-40degC, Battery 0-10 degC, OCM 15$\pm$2 deg, MSMR 10-30 degC
Mechanism	Solar panel	Solar panel hold down and deployment mechanism Sun pointing through SADA
	OCM	Hold down and tilt mechanism
	MSMR	Payload antenna scanning mechanism
Power	Solar panel	9.636 m^2, 6 panels 1.1 x 1.46 m^2 (Each) BSR (SCA) 800 watts (EOL)
	Batteries	2 x 21 AH, 42V, 28 Ni-Cd Cells,
	Electronics	More efficient power electronics developed. Two raw buses (28-42V) supplying power to all subsystems. Modular type of DC-DC converters for payload and data handling
TTC	Telecommand	PCM/FSK/FM/PM Modulation, Time tag command facility Conventional systems backed by microprocessor based, both for main and redundant.
	Telemetry	ASIC based telemetry system. Storage capacity of 4 orbits, PCM/PSK/PM modulation
	Transponder	Uplink frequency 2028.70 MHz Downlink frequency 2203.20 MHz
Data Handling		Data rate 2 X 10.4 MBPS Transmission frequency: X-band 8350 MHz Modulation QPSK

		Recording facility: global data for MSMR and 10 minutes average data anywhere in the orbit for OCM
AOCS	Specification	Pointing accuracy: Pitch : $\pm 0.15^{o}$ Roll: $\pm 0.15^{o}$ Yaw: $\pm 0.20^{o}$ Drift rate: 3.6×10^{-4} ^{o}/s
	Sensors	Conical Earth sensor (2), Dual cone earth sensor(1), PYS (2), 4 pi sun sensors(4), Magnetometers(2), IRU
	Actuators	Magneto torquers(2), Reaction wheels(4), 1 N thrusters (8) and 11 N thruster (1)
	AOCE	1750 architecture based microprocessor system for main and redundant
	Orbit	Orbital Accuracy 100-150 m (Autonomous mode using SPS)
Reliability goal		0.75 at the end of 3 years
Mass		1050 kg

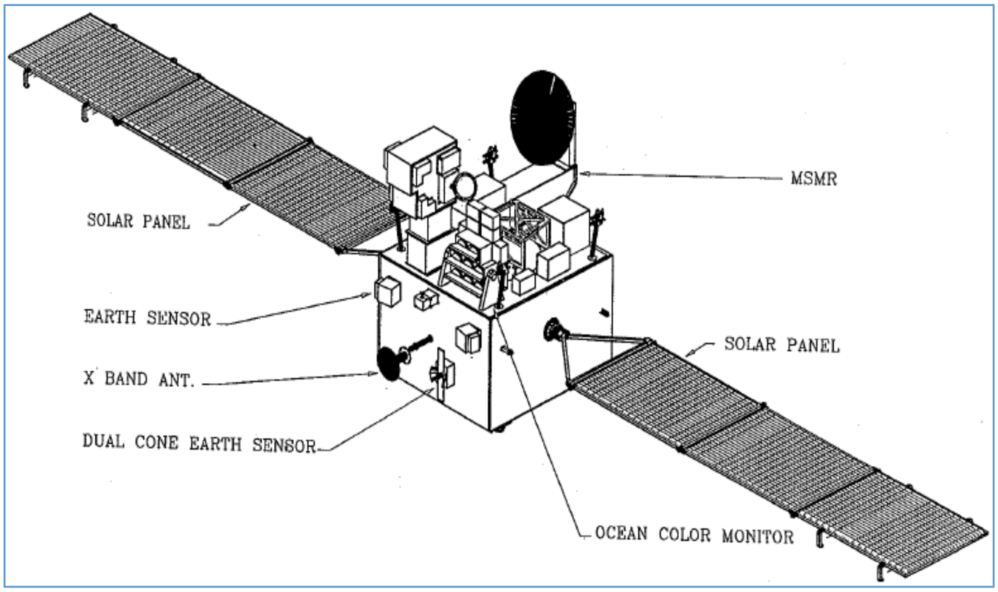


Figure 4-1 Deployed view of IRS-P4

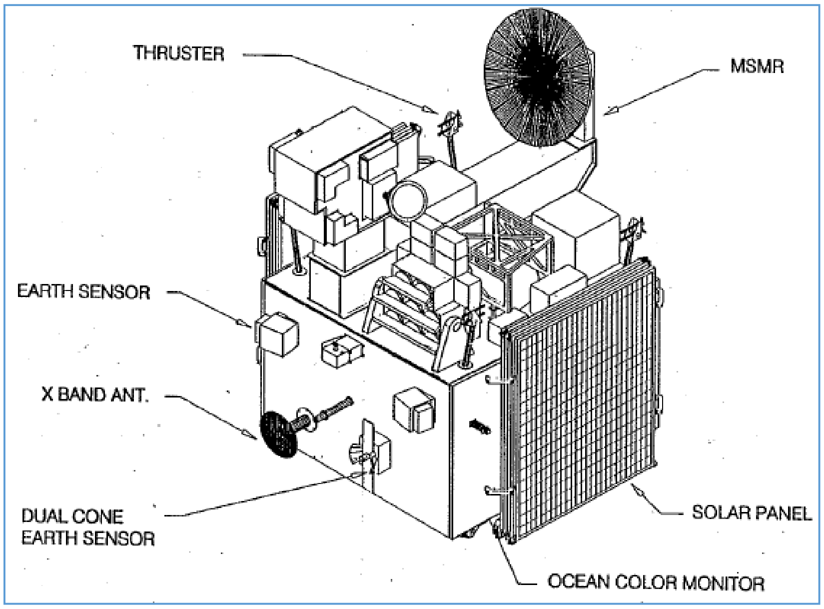


Figure 4-2 Stowed mode view of IRS-P4

4.1.5 IRS-P4 Payloads

4.1.5.1 OCM Payload

OCM operates in eight spectral bands. The imaging principle of OCM is based on push-broom technique which is the same as for the Linear Imaging Self-Scanner (LISS) cameras used in earlier missions. There is separate refractive optics for each band. Each band has a linear charge coupled devices (CCD) array in the focal plane of the optics as the detector. The detector outputs are processed by the payload electronics which provide serial digital data stream of each band to the data handling system.

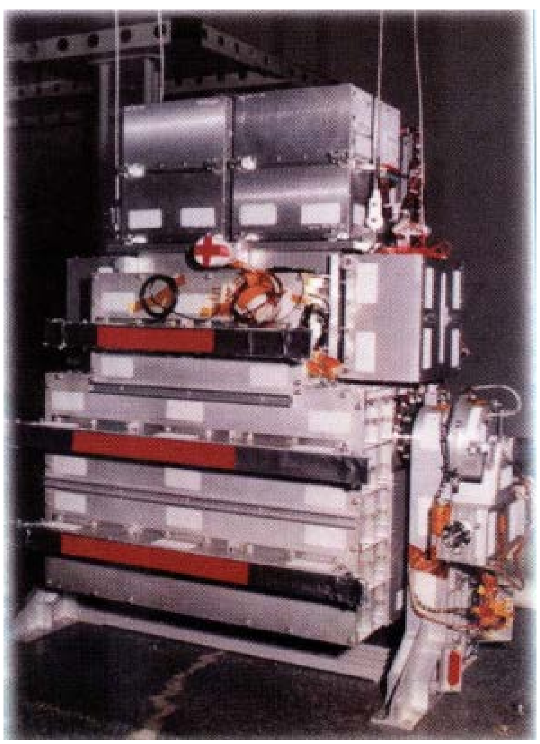


Figure 4-3 OCM Payload

Monitoring the colour of the ocean water leads to the information on the phytoplankton concentration, suspended sediments and yellow substance. OCM characteristics like observation bands and their bandwidths, spatial resolution, etc. are dictated by these water constituents. Additionally, the applications of OCM data for land-based applications, where frequent information is required on regional scale, are also kept in view while choosing the OCM parameters. The OCM is characterized by coarse spatial resolution, eight narrow spectral bands, high radiometric resolution, large field of view (± 43° providing a swath of 1420 km). Designing OCM for low ocean surface radiance and wide FOV were some of the challenges in its realization.

While only about 20% of the signal received by the OCM optics in the orbit comprises ocean radiance, 80% is the contribution from intervening atmosphere. Thus, to extract information on ocean colour, the contribution from atmosphere needs to be eliminated, and, therefore, accordingly correction is carried out by using data from band 7 and 8. Ocean radiance being low, 12 noon has been chosen as the time of equatorial crossing for descending pass to maximize the signal. This has an associated phenomenon of sun glint entering into the field of view of OCM, time of which is a function of season and latitude. To get over the problem of sun glint, a provision to tilt the OCM payload by ± 20° in steps has been provided. Its position can be fixed according to the latitude of observation and season. Tilt mechanism ensures a glint-free observation anywhere on the globe.

Table 4-3 IRS-P4 OCM specifications

Parameters	Specifications	
IGFOV	360m (across track) X 252m (along track)	
Swath	>1420 km	
Repetivity	2 days (29 orbits)	
Quantisation	12 bit	
Spectral range	402-885 nm	
Spectral bandwidth	20 nm (B1-B6) 40 nm (B7, B8)	
SNR @ saturation radiance	>512	
Spectral bands (microns)	Spectral Bands B1: 402-422, B2: 433-453, B3: 480-500, B4: 500-520, B5: 545-565, B6: 660-680, B7: 745-785, and B8: 845-885	Saturation radiance B1: 35.5, B2: 28.5, B3: 22.8, B4: 25.7, B5:22.4 , B6: 18.1, B7: 9.0, B8: 17.2
Integration time (ms)	34.75	
Detector	CCD191A	
Number of pixel	Total : 6000 Used: 3730	
Video readout rate/port	86.6 KHz	
Data rate / band	2.08 Mbits	
Total data rate generated	16.64 Mbits	

Camera MTF @ Nyquist frequency	>0.2
Size (mm) E-O module	701 (R) x 527 (P) x 420 (Y)
Weight(kg)	EO module: 64 and camera: 78

E.O Module: The EO module consists of Imaging lens assembly, EOM Structure, Detector head assembly, Detector electronics and payload tilt mechanism.

Optical system: It consists of eight spectral bands in visible and near infrared region having spectral bands between 0.4 um and 0.885 um with 20nm band width for bands B1 to B6 and 40 nm bandwidth for B7 & B8. Each band consists of its own collecting optical system and a linear array detector (CCD). The optical system consists of 10 refractive lens elements, a thermal filter in front and interference filter at back end close to the detector. The rear surface of the first lens is aspherical. A "telecentric" optical system is selected to provide minimum distortion, uniformity of illumination and good MTF over wide field angle which provides two days repetivity. The optical system is composed of a divergent component at the front end and a convergent group at the back end. This configuration gives longer back focal length than effective focal length and the main ray for each FOV goes out parallel to the optical view. The maximal angle of ray allowed to reach the focal plane is just 7 deg. This allows placing the band pass filter behind the optical system just in front of the CCD.

Table 4-4 IRS-P4 OCM Optics specifications

Parameters	Values
Equivalent focal length (EFL) (mm)	20.0+ 0.1
F-number	4.3 for B1 & 62 4.5 for B3 to 68
Field of View (degrees)	> + 43 (86 deg total)
Clear working distance (mm)	>16
Distortion	<+0.02%

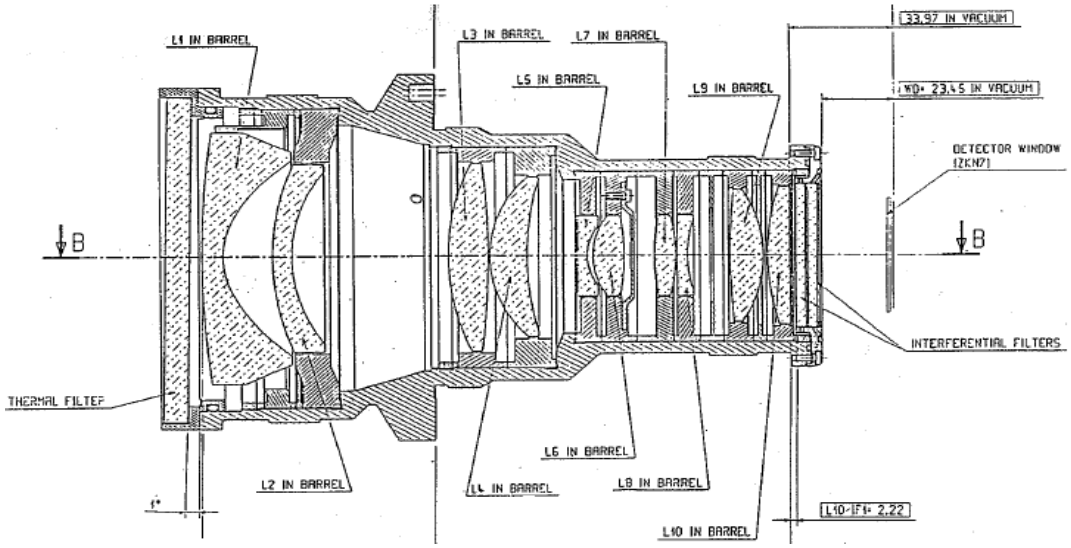


Figure 4-4 OCM Lens assembly

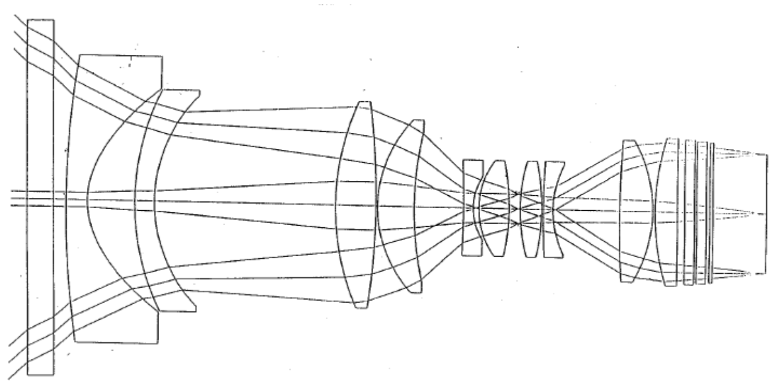


Figure 4-5 Optical Ray trace Diagram of OCM

EOM structure: The main structure of EO module is made out of single block of Al. Alloy 6061 material. This material is selected for its matching coefficient of thermal expansion, which helps in maintaining the separation between the lens focal plane and detector within ±2.0 micron over a temperature variation of 15± 2°C. Eight DE boxes are mounted on a support structure of four DE mounting's which are mounted on the EOM main structure.

Four thermal covers fitted on the EOM will cover the EO module on +ve Yaw, -ve Yaw, +ve pitch and -ve pitch direction. Thermal cover is black painted on its inside surfaces and covered by thermal blanket outside. Auto-control heaters are mounted on the inside surface of thermal cover. Lens side and detector side thermal covers have one cut-out for viewing and wire harness. A common hood with a slit aperture is placed in front of each row of the lenses. These hoods limit the Field of View of the lenses to ± 45° along pitch axis and + 2° along the roll axis.

OCM Electronics: The OCM electronics is modular, takes into account the reliability and testability, satisfies the mission goal that no single point failure shall lead to non-availability of two or more bands data. It has separate electronics for each band without any redundancy. But cross coupling exists between camera electronics and BDH. OCM Electronics consists of Detector Head, Detector Electronics

Detector Head: A 6000 element 7 x 10 um pixel size linear array CCD (CCD191A same as that used in IRS-1C/1D LISS-3 VNIR bands) is used as detector. This detector needs four bias voltages and nine clocks for its operations. The detector electrical interfaces, voltage levels are similar to IRS-1C except the readout speed. In IRS-1C to meet the high readout rate (866 KHz per port), two shift registers of the detector are read out simultaneously but in OCM phased readout mode of CCD operation is implemented like in CCD 143A of IRS-1A. This reduces 8 video processors.

Each lens assembly has different back focal length. Suitable spacers are used to place the Detector in focal plane. Considering the variation of the focal length with reference to temperature the most matching material is found to be Aluminium. However CCD is made out of ceramic which has very low Coefficient of Thermal Expansion (CTE). Hence Invar material is chosen for CCD Holder. Thermal stability among these two dissimilar materials will be achieved by using a dowel screw at one end and free screw at other end. In addition to these two LED holders are located on detector head. Each LED holder would accommodate two LEDs.

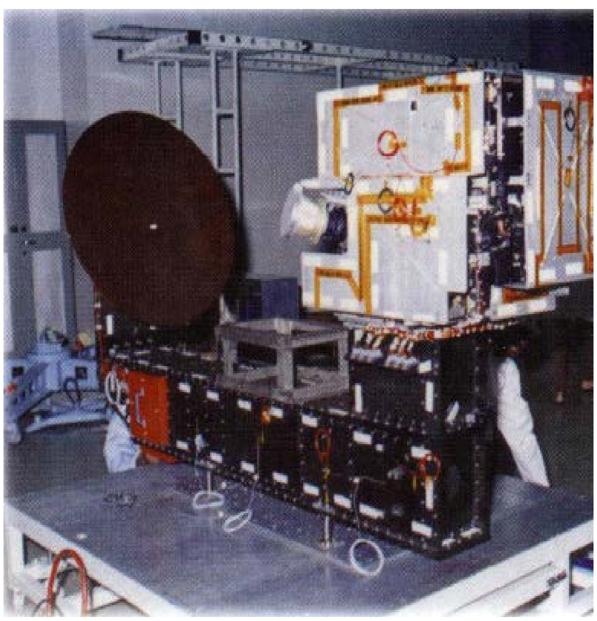
Detector Electronics: The detector electronics consists of bias generator and clock drivers located on the Electro Optic Module. The configuration of these circuits are similar to IRS-1C/1D except for additional drivers for reset clock and integration control, two phased readout and exposure control.

Calibration: Four LEDs of type HP 1 N6092 are mounted on the detector mount. Their optical axis is at 71deg from the normal due to the limited space between the detector and the imaging optics. In view of this large angle, the LEDs illuminate a larger photosensitive area compared to the imaging mode in the lateral direction of the detector array.

Table 4-5: Comparison of IRS-P4 OCM and SeaWiFS Parameters

Parameters	OCM	SeaWiFS
Band 1	404-424 nm	402 – 422 nm
Band 2	432-452	433- 453
Band 3	479-499	480-500
Band 4	502-522	500-520
Band 5	547-567	545-565
Band 6	660-680	600-680
Band 7	748-788	745-785
Band 8	847-887	845-885
Quantization bits	12	10
Sensor type	Pushbroom linear array CCD	Whiskbroom scan mirrors
Orbit Type	Sun synchronous	Sun synchronous
Altitude	720 Km	705 Km
Equator Crossing	Noon +20 min, Descending	Noon +20 min, Descending
Tilt	Along track, +20,0,20	Along track +20, 0 -20
Swath	1420 Km	2801 Km
Spatial Resolution	0.36 Km along track 0.236 Km cross track	1.1 Km
Revisit	2 days	1 Day

4.1.5.2 Multi-frequency Scanning Microwave Radiometer



It is a day-night-all weather sensor, designed to measure sea surface temperature, sea surface wind speed, atmospheric water vapour and liquid water content in the clouds. Four microwave frequencies, in both horizontal and vertical polarizations, have been chosen which are sensitive to these geophysical parameters.

MSMR has 862 mm x 800 mm off-axis parabola as the antenna reflector, and a corrugated feed to receive the emitted radiation from earth and its atmosphere. The antenna reflector is rotated at 11.16 rpm to get a circular scan of 1360 km width at the earth's surface, and 49.7° constant incidence angle at the beam centre. The feed meets the requirements of multi-frequency and multi-polarization operation. It is characterized by high-polarization purity, high-beam efficiency and low-ohmic losses. The receiver following the feed is a Dicke receiver which switches its inputs between incoming signal, reference load and cold-sky calibration horns.

MSMR was fully calibrated on ground for various return losses and receiver parameters. Various challenges in MSMR design included the stringent alignment stability requirement of 0.01° during launch and over a wide temperature range, antenna steering mechanism, and feed and a sensitive receiver.

The MSMR is a dual polarised radiometer system and is designed to estimate and monitor geophysical parameters related to the land, the ocean and the atmosphere. The frequencies and polarisation for MSMR have been arrived at by considering the applications like atmospheric water vapour, Sea Surface Temperature (SST), over oceans, ocean surface winds, cloud liquid water, snow/ice coverage etc.

Table 4-6:Specifications of IRS-P4 MSMR

Specifications	Values
Swath	1360 km
Repetivity	2 days (29 orbits)
Frequencies	6.6GHz(V&H) 10.65 GHz (V & H) 18GHz(V&H) 21 GHz(V&H)
Temperature	Better than 1°K

MSMR consists of following systems

- Antenna
- Receiver
- Data acquisition and control system (DACS)
- Analog & Digital telemetry sub-systems(ADTMS)

4.1.5.2.1 MSMR antenna system

The MSMR is configured with a scanning antenna system which consists of an offset parabolic reflector with a 80 cm diameter collecting aperture and a multifrequency feed assembly. The antenna reflector is mechanically rotated with a constant angular velocity for scanning the antenna beam across the satellite trace in order to give the required swath of 1360 km
MSMR antenna consists of following subsystems

- Offset parabolic reflector
- Multifrequency Dual Polarised Feed
    - Multifrequency Ortho mode Transducers
    - Calibration Horn for 6.6 & 10.65 GHz
    - Calibration Horn for 18 & 21 GHz
    - Support structure
    - Antenna Scan Mechanism (ASM)

Table 4-7:Specifications of MSMR Antenna

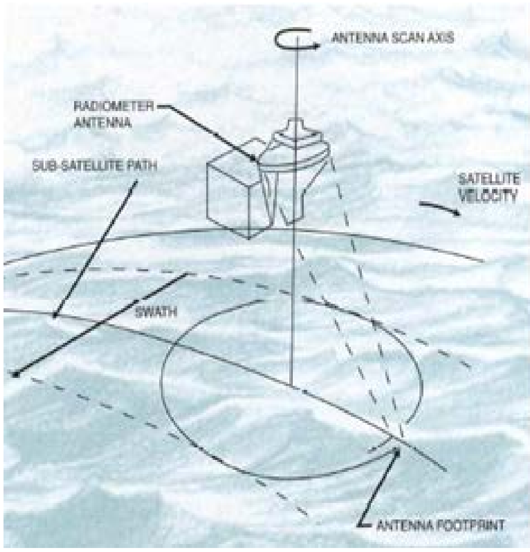
Frequency(GHz)	6.6	10.65 18	21	6.6
Bandwidth(MHz)	+ 112	+ 112 + 160	+ 170.5	+ 112
Beamwidth	42° ± 0.2°	2.6° ±0.15°	1.6° ± 0-1°	1.4° ± 0.1°
Polarisation	V&H	V&H	V&H	V'&H
Cross Pol.(dB)	< - 23	< -23	< -23	< - 23
Return loss(dB)	<-17	<-17	<-17	<-17
Beam efficiency	90%	90%	90%	90%
Scan offset angle	43.32°	4332°	43.32°	43.32°

Reflector: The offset parabolic reflector is of elliptical shape (862mm x 800 mm) and renders a projected diameter of 800 mm aperture. The offset reflector is having F/D=1.8 and an offset angle of 43.32°. This helps in achieving a clear field of view as well as 50° the Earth incidence angle of the beam. The reflector is having embedded copper mesh on the reflecting surface to enable the operation at 18 & 21 GHz of the system. The overall RMS variation of surface accuracy is 0.10 mm over the elliptical size of (8 62.0 X 862.8) the reflecting surface. The reflector is fabricated out of CFRP sandwich of aluminium-honey comb.

Feed: The feed consists of Horn and Ortho Mode Transducers (OMT). Corrugated horn has been used because of the pattern symmetry, low cross polarisation and low side lobe levels are achievable with this type of horn.

Support structure: In MSMR payload antenna, reflector and multifrequency feed are at a distance of 1667 mm apart. The phase centre of the feed being 337 mm from the aperture of the feed, the resulting seperation of 1330 mm between the reflector and the feed aperture plane has been considered in achieving overall 2029 mm length of CFRP support structure. The support structure supports the view of Scanning Mechanism (ASM) and the reflector at one side and the feed, front end electronic packages, two number of calibation horns which are interconnected with each other by plumbing waveguides and RF cables on the other side.

Antenna Scanning Mechanism: The spatial resolutin of MSMR is decided by the foot print of the antenna beam. Circular scan is adopted for the MSMR because it provides more integration tme and least torque requirement due to continuous constant angular rotation. In additon to this, due to non reversal of angular momentum, the scan mechanism will have minimal effect on the satellite attitude. The scan geometry of constant incident angle of 50 degrees makes the elliptical footprint with larger axis across the scan direction. The 3 dB foot print in scan direction is decided by the slant range and the antenna 3 dB beamwidth.



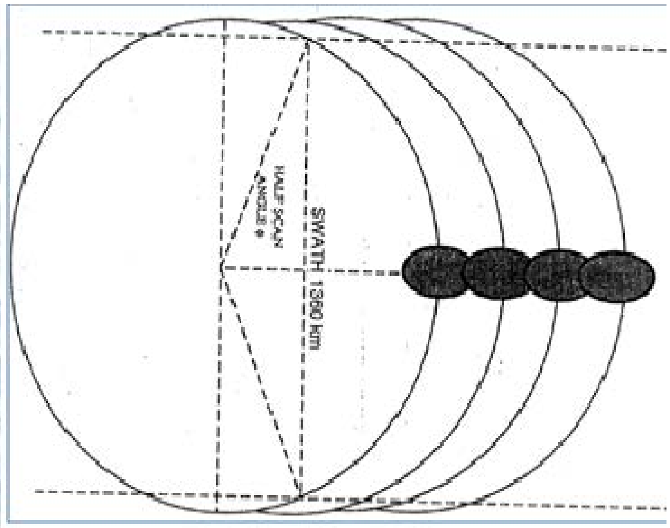


Figure 4-6 MSMR Scan Path illustration **Figure 4-7 Swath generation by MSMR**

Frequency (GHz)	Beam Foot Print (km)		Cell dimention (After ground processing) km^2
	Along Scan	Across scan	
6.6	77	119	120 X 120
10.65	47	73	80 X 80
18	30	46	40 X 40
21	25	39	40 X 40

The scan period is fixed in such a way that 10 % overlap is provided for the smallest footprint ie. 21 GHz channel. This corresponds to an angular scan speed of 11.173 RPM (5.37 seconds per rotation). The integration time corresponding to 11.173 RPM for various frequencies are listed below. The onboard integration time implemented corresponds to half that for 21 GHz channel which provides smallest foot print. The temperature sensitivity of MSMR depends upon receiver predetection bandwidth, Integration time, type of receiver, antenna and receiver temperature and gain measurement accuracy.

Table 4-8 MSMR Integration time

Frequency (GHz)	Integration Time (oveall in msec)	Integration time implemented (msec)
6.6	96	18
10.65	60	18
18	36	18
21	32	18

4.1.5.2.2 MSMR Receiver

MSMR payload has six receiver chains, catering to 21 GHz-V pol, 21 GHz - H pol, 18 GHz- V pol, 18 GHz-H pol, 10.65 GHz and 6.6 Ghz bands. For 6.6 and 10.65 GHZ bands single receiver is used to collect both polarisations using a polarisation select switch at the input.

Table 4-9 Specifications of MSMR Receiver

Frequency (GHz)	6.6	10.65	18	21
No of channels	1(V/H)	1(V/H)	2(V/H)	2(V/H)
Predetection bandwith (MHz)	100	100	150	150
Noise Figure of receiver (dB)	4	4	4.5	4.5
Dicke clock	1 KHz			
Integration Time	18 msec			
Input dynamic range	2.7 degK-330 degK			
Output signal level	0-10V			
Sensitivity	~1degK			
Receiver stability	0.01db			

The function of radiometer is to measure the noise power incident at the antenna. In MSMR dicke type configuration is used for the receiver. In dicke type radiomater, a SPDT switch used to periodically switch the receiver input between the antenna and a constant noise sourse (Tref) at a switching rate higher than the highest significant spectral component in the gain variation spectrum.

Local Oscillators (LO): While dielectric Resonators are used for 6.6 GHz and 10.65 MHz oscillators, Gunn diodes mounted in short circuited half guide wavelength cavity used for 18 GHz and 21 GHz oscillators. Schottky barrier diode is being used as the device for detection. The diode is biased and designed arround flat detector configuration to achieve the required banwidth.

Precision baseband processing subsystem (PBPS): This lies between the RF front end and the quantiser of the DACS (Data Acquisition and control System), forming tail end of receiver. PBPS has to generate a DC signal demodulation, and converts it to a format suitable to the quantiser.

4.1.5.2.3 Data acquisition and control subsystem

The Data Acquisition and Control Subsystem (DACS) is the tail end of the MSMR payload. DACS carries out data digitisation, timing sequence generation and control signal generation for the MSMR payload electronics. The on-board integration time selected is 18 msec. The MSMR data will be digitised in 12 bits/sample to achieve digitisation accuracy better than 0.1 deg K over the specified range of antenna temperature. 12 bit data of each radiometer channel is serialised at 8 KHz rate, multiplexed and transferred to on-board baseband data handling system (BDH). Alongwith serial data, an additional strobe at every 12th bit is also provided to spacecraft BDH.
The six channel MSMR sensor and calibration data are digitised with 12-bit resolution to achieve the required accuracy for specified range of antenna temperature. Uniform onboard integration

and sampling intervals of 18 msecs amd 9 msecs. have chosen for all the six channels to reduce overall MSMR hardware complexity. This avoids signal aliasing and enables further integration and averaging operations in the ground processor. The table gives the polarisation switching sequence & sampling interval details for the different MSMR channels. The total DACS data rate is about 5.6 Kbps.

Frequency	Polarization		Sampling Interval (msec)	
	Even cycle	Odd cycle		Even cycle
21.0 GHz	V	V	9	g
21.0 GHz	H	H	9	9
18.0GHz	V	V	9	9
18.0 GHz	H	H	9	9
10.65 GHz	V	H	9	9
6.6 GHz	V	H	9	9

The MSMR Antenna Scan Mechanism (ASM) provides an anti-clockwise scanning of antenna footprint on ground. A scan Start pulse which corresponds to the angular position of - 90 Deg with reference to roll axis in each circular scan cycle, is provided by BDH to DACS. The first half cycle from the SCAN START in each scan cycle is utilised for sensor data collection and the remaining period is utilised for calibration sequence and collection of temperature information from Analog & Digital Telemetry Subsystem (ADTMS).

DACS acquires the multichannel radiometer data and also generates the timing and control signals required for Precision Base Band Processing Subsystem (PBPS), ADTMS and data transfer to BDH.

DACS gets analog inputs from PBPS. Differential amplifiers are used in front of DACS input to take care of ground potential diff. between PBPS and DACS packages.

The timing and sequence generator receives 8 kHz clock and SCAN START from BDH. All timing windows and clock signals required to acquire the sensor, calibration and data acquisition slot for ADTMS are generated with reference to the SCAN START.

Parameter	Specification
No. Channels	6
Analog Input	0 to 10
AID Resolution	12 bits
Sampling period	9 msec.
Total Cycle duration	5376 ms
Data words per cycle	j 2521
Data rate	~ 5.6 Kbps
Digitizer	± I LSB rms

The time sequencer generates various timing waveforms with reference to the Scan Start pulse signal from BDH. In addition to these it generates Dickie switching clock (1 KHz) and sampling clock (9 ms). All the timing windows are generated with 9 msec resolution and realised using a programmable synchronous counter chain.

4.1.5.2.4 Analog & Digital telemetry subsystem (ADTMS)

The MSMR payload has several systems, the physical temperatures of which form the part of the system calibration data. The knowledge of the absolute temperatures of these is necessary in order to model/calibrate the system functionally. The analog & Digital Telemetry subsystem (ADTMS) of the MSMR payload is meant for this precision temperature monitoring application. Temperature is monitored at 51 points to an accuracy of + 0.1 deg. C. Thermisters and platinum Resistance Devices are used as temperature sensors. The voltages sensed are quantised using a 12 bit ADC. The total ADTMS data stream consists of 64 words of 12 bits each which are transferred to S/C data handling unit using a synchoronous serial transmission philosophy. This transfer is carried out in the time slot of MSMR pay!oad at the basic clock frequency of 8 KHz. The digital interfaces for ADTMS are the basic clock input. ADTMS acquisition slot. 9 ms sampling clock from DACS and data, strobe and gate lines to the spacecraft.

4.1.5.2.5 Calibration

In MSMR two point internal calibration approach has been utilised by using dedicated horn antennas viewing the cold space (2.7 deg K) and a black body at a high temperature will be used for the hot reference. As the circular conical scan method is used in MSMR, the fore half period is utilised for data collection whereas the aft half period will effectively be utilised for cold and hot calibration sequences. During the aft half cycle, two sets of internal calibration are envisaged just before and after data collection cycle.

4.1.5.2.6 MSMR Interfaces

Power interface : There are 10 DC-DC converter along with their 5 filter/regulator modules All connection with the power packages will be through shielded twisted wire pairs on D-Type connectors.

Subsystems	Power dissipation(W)
Feed front end assembly (FEFA)	38.5
PBPS	18.75
DACS	7.75
ADTMS	10
Antenna scan mechanism (ASM)	3.5
Power subsystem (All MPC+MPR)	22.5
Total	101

BDH Interface: The DACS and ADTMs have interfaces with BDH. The 12 bit MSMR sensor and calibration data from DACS is sent in a bit serial format to the BDH system. At every 12th bit a strobe is sent. Same way the ADTMS send s 12 bit analogue telemetry (ATM) data in serial format along with data validating strobe at every 12 th bit.

4.2 Oceansat-2

4.2.1 Introduction

Oceansat-2 spacecraft provides continuation of services of IRS-P4 with enhanced application areas. Oceansat-2 is a 3-axis stabilized spacecraft basically derived from I-1.5K IRS bus with proven mainframe systems.

Oceansat-2 carries three payloads - Ocean Colour Monitor (OCM-2), Ku-band Scatterometer and Radio Occultation Sounder for Atmosphere (ROSA). OCM is a multi-spectral optical camera, providing ocean colour data with repetivity of two days. It provides a ground IFOV of 360 m in across track and 246 m in along track directions covering a swath of 1420 km. The camera can be tilted by ± 20 ° with respect to nadir in the along-track direction.

The Ku band pencil beam scatterometer is an active microwave sensor for measurement of wind speed and wind direction. It consists of a parabolic dish antenna of 1m diameter that is continuously rotated at 20.5 rpm using a scan mechanism with the scan axis along the +Yaw axis. This antenna is arrested during launch and released in orbit for the operations.

The ROSA is a GPS Receiver for atmospheric sounding by radio occultation. The GPS receiver determines position, velocity and time using GPS signals. Besides, ROSA receives RF signals from the 'rising' GPS satellites near Earth's horizon through its occultation antenna and from the excess phase delay and Doppler measurements, atmospheric parameters (Temperature, humidity, pressure) can be derived.

During operational phase of the spacecraft, Scatterometer and ROSA payloads are continuously ON and OCM will be switched ON during sun-lit passes over oceans as per user requirements.

4.2.2 Mission Objective

The mission objectives of Oceansat-2 are

- To design, develop, launch and operate a state of art 3 axis body stabilized satellite providing ocean based remote sensing services to the user communities
- To develop remote sensing capabilities with respect to Ocean resources
- To establish ground segment to receive and process the payload data.
- To develop related algorithms and data products
- To serve in well-established application areas and also to ensure the mission utility.

4.2.3 Orbit Details

Parameter	Value
Type	SSPO

Altitude (km)	720
Inclination (Deg)	98.28
Period (Min)	99.31
Local Time	12.00 Noon ± 10 min
Repeat Cycle	2 Days
Distance between adjacent traces	1382 km.
Distance between successive ground track	2764 km
Ground Track Velocity	6.7818 km/s

4.2.4 Salient features of Oceansat-2

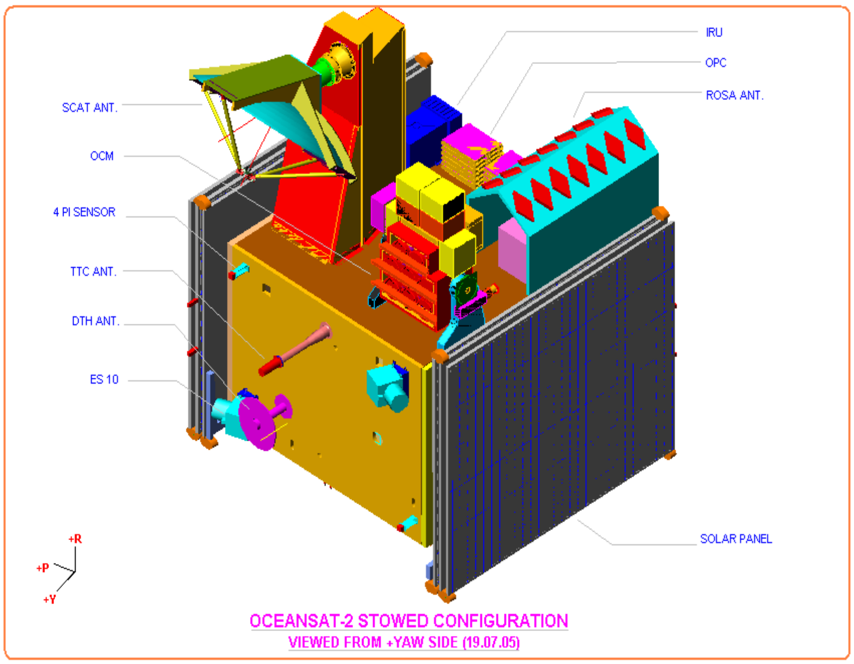


Figure 4-8 Stowed configuration of Oceansat-2

Subsystem		Oceansat-2
Structure		CFRP- Aluminum honeycomb sandwich cylinder with aluminum honeycomb panels.
Thermal	Control	Temperature control is with passive techniques using Paints, multilayer blankets, Optical solar Reflector. Active thermal elements like heaters.
	Limits	All electronics packages 0-40degC, Battery: 5±5 degC, OCM :15± 2 ^{o} C
Mechanism	Solar Panel	Solar panel deployment mechanism and Drive Mechanism
	OCM	OCM Hold down and release & OCM Tilt

	Scatterometer	SCAT antenna hold-down & release
Power	Solar Panel	Sun tracking, rigid,15.12 m^2 6 panels 1.4 x 1.8 m^2 (Each), 1360 W at EOL, BSR(SCA)
	Battery	2 batteries, 28 to 42V, 28 Cells, Ni-Cd 24 AH
	Power Electronics	Two raw buses (28-42V), PWM TCR, FCL, 10 Strings
TTC (BMU)	Telemetry	1024 words/frame, 2250 MHz, storage: 6.29 x 10^6 Bits PCM/PSK/PM, 16 Kbps
	Telecommand	PCM/PSK/PM,4Kbps, .2071.875 Mz Time tag command facility
Data Handling		Data rate 42.4515 MHz
Data Transmission		QPSK Modulated Transmission , Transmitted at 8300 MHz.
AOCS (BMU)	Spec.	Pointing Accuracies: Yaw: $\pm$ 0.15^o Roll: $\pm$ 0.1^o Pitch: $\pm$ 0.1^o (3 sigma) Driftrate : <3.0 x 10^{-4} deg/sec (3 sigma) : Pointing accuracy 100-150 m
	Sensors	Earth sensor(1), DSS(2), 4Pi SS(4), Magnetometer (2) IRU(3 DTG), GPS
	Actuators	Reaction Wheels, 5 NMS(4 in tetrahedral), Magnetic Torquers (2) , 1N Thrusters(8) 11 N Thruster(4)
Payloads		OCM, Scatterometer, ROSA
Reliability goal		0.75 at the end of 5 years
Mass		975 kg

4.2.5 Oceansat-2 Payloads

There are two main payloads in Oceansat-2, namely, an Ocean Colour Monitor (OCM) and Ku-band Pencil beam Scatterometer. In addition, a piggy-back payload called the Radio Occultation Sounder for the Atmosphere (ROSA) developed by the Italian Space Agency will also be flown on-board Oceansat-2. While the OCM provides data on the bio-physical properties of global oceans like Chlorophyll concentration, suspended sediments, algal blooms etc., the Scatterometer provides data from which the surface Wind velocity (both speed and direction) over ocean surface will be derived. ROSA is an atmospheric sounder and provides data on Temperature and Humidity profiles in the troposphere as well as space weather.

A brief description of these payloads and their interfaces with other systems is given in the following paragraphs.

4.2.5.1 Ocean Color Monitor (OCM)

The Ocean Color Monitor (OCM) is a solid state CCD camera and operates in eight narrow spectral bands with 360m along-track and 240m across-track ground resolution covering a swath of 1420 km from 720 km altitude. All the eight bands are in Visible and Near Infrared region having spectral bands between 0.4µm and 0.885µm. Since the local time of pass for Oceansat-2 is 12noon, provision is made to tilt the Electro-Optics module about the Pitch axis by ±20º with reference to nadir to avoid the sun glint from sea surface. During launch the EO Module will be held by hold-down mechanism, which will be released in-orbit using a pyro cutter.

4.2.5.1.1 Specifications of OCM

Parameters	Specifications	
IGFOV	360m (across track) X 252m (along track)	
Swath	>1420 km	
Repetivity	2 days (29 orbits)	
Quantisation	12 bit	
Spectral range	402-885 nm	
SNR @ saturation radiance	>512	
Spectral bands (microns)	Spectral Bands B1: 402-422, B2: 433-453, B3: 480-500, B4: 500-520, B5: 545-565, B6: 610-630, B7: 725-755, and B8: 845-885	Saturation radiance B1: 35.5, B2: 28.5, B3: 22.8, B4: 25.7, B5:22.4 , B6: 18.1, B7: 9.0, B8: 17.2
Integration time (ms)	34.75	
Detector	CCD191A	

Number of pixel	Total : 6000 Used: 3730
Video readout rate/port	86.6 KHz
Data rate / band	2.08 Mbits
Total data rate generated	16.64 Mbits
Band-to-band registration	<+ 0.25 Pixel
Camera MTF @ Nyquist frequency	>0.2
Size (mm) E-O module	701 (R) x 527 (P) x 420 (Y)
Weight(kg)	EO module: 64 and camera: 78
Power (Regulated)	Imaging mode : 130W Calibration mode: 132W

The OCM payload consists of the following systems

- Electro-Optics Module (EOM)
- Payload Electronics (PLE)
- Power converters and regulators (OPC / OPR)

4.2.5.1.2 Electro-Optics Module (EOM)

The EO Module consists of imaging lens assemblies, EOM Structure, Detector head assembly; Detector electronics and payload tilt mechanism

Optical system: It consists of eight spectral bands in visible and near infrared region having spectral bands between 0.4µm and 0.885µm with 20nm band width for bands B1 to B6 30nm bandwidth for band B7 and 40nm bandwidth for B8. Each band consists of its own imaging Lens assembly and a linear array detector (CCD). The optical system consists of 10 refractive lens elements, a thermal filter in front and interference filter. To cover the wide field-of-view (± 43º), the first lens element is realized with parabolic surface. A "tele-centric" optical system is selected to provide minimum distortion, uniformity of illumination and good MTF over the wide field angle. The optical system is composed of a divergent component at the front end and a convergent group at the back end. This configuration gives longer back focal length than effective focal length and the main ray for each field of view goes out parallel to the optical view The maximum angle of ray allowed to reach the focal plane is just 7 deg. This allows to place the band pass filter behind the optical system just in front of the CCD.

Transmission of the lens is improved by providing anti-reflection coating. The band pass function is achieved by using an interferential filter located after the lenses, at the end of the objectives. This filter has two substrates, each one having two faces coated.

Optical system specification

Parameter	Value
Equivalent focal length (EFL) (mm)	20.0±0.1

F-number	0.3 for B1 & B2 0.5 for B3 to B8
Field of View	± 43º (86º total)
Clear working distance (mm)	>16
Distortion	< ±0.02%
MTF(@ 50 lp/mm)	> 0.6

EOM Structure: The main structure of EO module is made out of single block of Al. Alloy 6061 material. This material is selected for its matching coefficient of thermal expansion, which helps in maintaining the separation between the lens focal plane and detector within ± 2.0μ over a temperature variation of 15 ± 2º C. Four thermal covers fitted on the EOM will cover the EO module on +ve Yaw, -Ve Yaw, +ve pitch and –ve pitch direction. Thermal cover is black painted on its inside surfaces and covered by MLI blanket outside. Auto-control heaters are mounted on the inside surface of thermal cover. Lens side and detector side thermal covers have one cutout for viewing and wire harness. A common hood with a slit aperture is placed in front of each row of the lenses. These hoods limit the Field of View of the lenses to ±45º in pitch-yaw plane and ±2º in roll-yaw plane w.r.t the optical axis. The interface for the Tilt mechanism gimbal shaft is provided on the Pitch axis sides of EOM. The EOM is held-down at an angle of –23º in the launch configuration.

4.2.5.1.3 OCM Electronics

The OCM electronics is modular and satisfies the mission goal that no single point failure shall lead to non-availability of two or more bands data. It has separate electronics for each band without any redundancy. But cross coupling exists between camera electronics and BDH.

OCM Electronics can be functionally divided into following three portions

- Detector Head Assembly
- Detector Electronics
- Video processing Electronics

Detector head assembly: A 6000 element of 7 x 10μm pixel size linear arrays CCD (CCD191A same as that used in IRS-P4) is used as detector. Out of 6000 active pixels available in the CCD, 3730 central pixels are used for imaging. For the dark current estimation and subtraction from video data, 150 pixels on either edge of the CCD are used.
Each band lens assembly has different back focal length. Suitable spacers are used to place the Detector in the focal plane. Considering the variation of the focal length with reference to temperature the most matching material is found to be aluminum. However CCD is made out of ceramic, which has very low coefficient of Thermal Expansion (CTE). Hence Invar material is chosen for CCD Holder. Thermal stability among these two dissimilar materials will be achieved by using a dowel screw at one end and free screw at other end. Two LED holders are located on the detector head. Each LED holder would accommodate two LEDs used for on-board calibration.

4.2.5.1.4 Calibration

Four LEDs of type HP 1N6092 are mounted on the detector mount. Their optical axis is at 71º from the normal due to the limited space between the detector and the imaging optics. In view of this large angle, the LEDs illuminate a larger photosensitive area compared to the imaging mode in the lateral direction of the detector array. Sixteen distinct calibration levels will be set using digitally altered exposure time method. The signal range coverage would be about 70% of the full range. The expected non-uniformity of the illumination using calibration LEDs is better than ± 50% with respect to the array mean calibration count.

4.2.5.2 Ku-Band Scatterometer

4.2.5.2.1 Introduction

The main objective of the Scatterometer Payload onboard OCEANSAT-2 is to gather the information about the near surface winds over oceans at a global level. The Scatterometer wind field measurements form a very important input to the global weather forecasting system.

Figure 4-9 Scatterometer

The near surface wind mostly modulates the capillary waves on the ocean surface whose wave length is of the order of centimeters. Previous missions carried Scatterometers operating in both C-band and Ku-band frequencies. The examples of the C-band Scatterometer are ERS-1 & 2, and the recent Advanced Scatterometer (ASCAT) onboard METOP satellite. Ku-band missions include SeaSAT, NSCAT and SeaWinds Scatterometer on QuikScat and ADEOS-II satellites. Looking at the already established sensitivity of the Ku-Band frequencies to the wind vector and the wide applications and significant research being carried out using the data from NSCAT and SeaWinds and the available primary allocation, Ku-band was chosen for the Oceansat-2 mission.

Characteristics of Scatterometer

Parameter	Inner beam	Outer beam
Satellite Altitude	720 km	
Principal Axis Pointing angle	46°	
Frequency	13.515625GHz	
Wavelength	0.0221965 m	
Wind Speed	4 – 24m/s with accuracy of 10 % or 2 m/s whichever is higher	
Wind direction	0 – 360ºwith accuracy of 20º deg.	
Swath (qualified)	1400 km	
Polarization	HH	VV
Basic Sigma naught cell (Center)	26 km x 6.8 km	31 km x 5.8 km
Kp over Basic sigma naught Cell without error (4 m/s cross wind)	1.32-3.49 over 5 cells around beam centre	1.89-3.02 over 5 cells around beam centre
Kp over Basic sigma naught Cell without error (24m/s cross wind)	0.28-0.3 over 5 cells around beam centre	0.3-0.33 over 5 cells around beam centre
Slant Range(km) 1208	1031	1208
One Way 3dB Foot Print Az(km) x El(km)	26 X 46	31 X 65
Along Track Spacing (km)	19.7	19.7
Along Scan Spacing (km)	16.3	21
Across Scan Overlap (Varies with Azimuth position)	28% to 48%	29% to 71%
Along Scan Overlap	41%	34%
Cell Center Doppler (Excluding Earth rotation) -0.25 Deg Pointing Error (KHz) 0.00 Deg Pointing Error (KHz) +0.25 º Pointing Error (KHz)	 ± 455.69 ± 457.86 ± 460.03	 ± 510.54 ± 512.47 ± 514.38
Earth Rotation Doppler (Equator)	± 29KHz	± 31KHz
σ_0Parameters (dB) 4m Cross Wind (Qualified) 24m Up Wind (Qualified)	 -31.3 -10.9	 -29.8 -12.4
Antenna Specifications Antenna Diameter Angular Separation of Feeds Peak Gain Beam width (Al. Scan. X Ac. Scan)	 1m 6.76º 39.5dBi 1.47º x 1.67º	
Transmitter Specifications		

Transmit Power Transmit Duty Cycle Transmit PRF for system(Nominal) Transmit PulseWidth Transmit Modulation Transmit Chirp Bandwidth	100W 27% 193Hz 1.35ms LFM 400KHz
Receiver Specifications Receiver Type O/P Bandwidth Gain control (controlled by 6-bit gain control telecommand) Receiver Pathloss (dB) Noise figure (dB) Input Noise Power Receiver output	Single Channel Output on IF of 15.625MHz 15.625MHz ±800KHz 45dB to 109dB 3 3 -109dBm over 1600KHz bandwidth 500m V p-p across 50 Ω resistance

Power(dBm) Spread Over 400KHz Chirp bandwidth	Signal	Signal+Noise	Signal	Signal+Noise
4m Cross Wind (qualified) 24m Up Wind (qualified)	-141.17 -120.82	-115.18 -114.51	-140.77 -123.25	-115.18 -114.51
Receive Window	2.097ms (4096 sample points)			
Window Start time	6.73ms		7.80ms	
PRF	193Hz (nominal)			
Quantization	8 bits I + 8 bits Q (including sign bit)			
Sampling Frequency	1.953125MHz			
Noise Bandwidth	1245KHz			
Processing Bandwidth	305KHz (94KHz Foot print bandwidth with ±0.25 error in pointing, 58KHz earth Doppler and remaining is the margin)		305 KHz (153KHz Foot print bandwidth with ±0.25 error in pointing, 62KHz earth Doppler and remaining is the margin)	
Measurement bandwidth	9.54KHz (nominal)			
Output Processed Data rate (only)	330Kbps (nominal mode)			
Output Raw Data rate (only)	13.6Mbps			
Elevation pointing (Inclusive of attitude error)	+/- 0.25º			

4.2.5.2.2 Antenna Sub-System:

The 1 mtr dia antenna reflector is a prime focal parabolic dish with a focal length of 0.4 meter. The vertex of the dish is offset from the mounting interface by 180 mm (approx.). The reflector dish is fixed at a tilt of 46^0 about the + yaw axis in the yaw-wave-guide plane. In the launch condition, the reflector assembly is rotated by 30^0 clockwise, about the yaw axis (when viewed from the positive yaw axis). Three CFRP tubes (called Spars) support the feed horns of the antenna. The wave-guide is to be routed along one of the three Feed Support tubes. Two feed horns are used for generating the two beams (Outer & Inner) with different polarizations at specified look angles. The Antenna is attached to the Scatterometer scan mechanism to mechanically spin the parabolic reflector along with the feeds around scan axis coinciding with the +ve yaw axis of the satellite. The rear side of the antenna is white painted and the feed assembly is covered with MLI blanket as per the Thermal design. Major specifications of the Antenna reflector are given below.

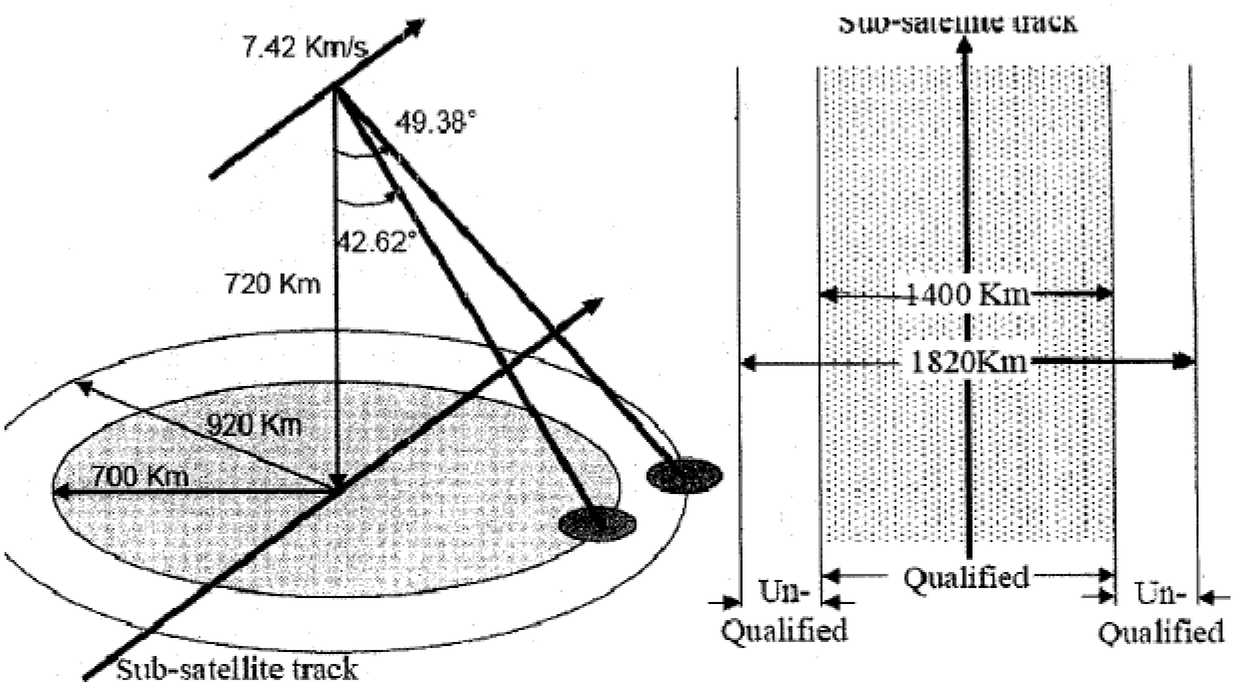


Figure 4-10 Scatterometer Swath coverage

Specifications of Scatterometer antenna reflector

Parameter	Value
Diameter of dish	1000mm
Outer diameter of dish	1014mm (max. permitted)
Focal length	400mm
Shape	Axis-symmetric paraboloid
Angle of tilt (With respect to spin axis)	46.0 ± 0.01^0

RMS error	0.1mm
Mass	<10 kg
No. of feed support tubes (FST)	3
Diameter of FST	20mm (max. permitted)
Feed bracket mass	0.4 kg
Feed adjustment (along reflector axis)	± 10mm

4.2.5.2.3 Dual Channel Wave guide Rotary Joint

Space qualified dual channel microwave rotary joint for Ku-band has been developed in-house for pencil beam scanning Scatterometer payload onboard Oceansat-2 after performing electrical, mechanical, thermal and environmental qualifications. IISU has been responsible for fabrication assembly and qualification of bearing assembly for the rotary joint. The conical scanning is effected by mechanically rotating the Scatterometer antenna about the yaw axis at 20.5 rpm using a scan mechanism. The two feeds of the reflector antenna for Scatterometer payload are fed with microwave signal through the dual channel rotary joint. The stationary part of the joint is mounted to the satellite deck through waveguide plumbing. The specifications of the Rotary joint are given below.

Specifications for the Dual Channel Rotary Joint

Parameter	Value
Frequency	13.515625GHz
Bandwidth	± 25MHz
Return Loss (max)	Ch1: >19dB Ch2: >19dB
Insertion loss (max)	Ch1: 0.35dB Ch2: 0.35dB
Isolation	40dB
Insertion loss – variation within a single scan	Ch1: ± 0.05dB Ch2 : ±0.05dB
Peak power	Peak 140W, 34%DC
EMI/EMC Results Test type – RE102 MIL-STD 461 E	67 dB µVolt/m

4.2.5.2.4 Scatterometer Scan Mechanism (SSM)

The Scatterometer Scan Mechanism is used to rotate the Antenna reflector along with its Feeds and waveguide assembly and Rotary joint at a constant rate of 20.5rpm. It consists of a brushless DC motor and its Drive Electronics.

The functional requirements of Scatterometer Scan Mechanism are

- To provide necessary mechanical interface and rotate antenna, feed system, which weighs 10 kg, at 20.5 rpm about the precision axis of rotation with scan stability of 0.1%.
- Provide necessary mechanical interface for microwave rotary joint stator & rotor part
- To accommodate the rotary joint stator part.
- Provide accurate scanning position information with accuracy of 0.01º on interrogation at PRF
- Provide mechanical interface to payload deck.
- Define scatterometer relative orientation and position w. r. to payload reference frame/bus reference frame.
- Precise static and dynamic balancing of rotating parts.
- Provide redundancy in drive motor, angular sensor and control electronics.

The SSM consists of a precision bearing unit with static reservoir, brushless DC outer rotating drive motor assembly, optical encoder assembly, cube assembly (both in stator part and rotor part) for optical alignment, housing, motor rotor housing, hollow central shaft and balancing arms.

Bearing Unit Assembly: The mechanism is outer rotating one. SSM consists of precision bearing unit assembly, which includes static central shaft that locates inner race of face to face mounted two pairs of duplex angular contact ball bearings. Outer races are axially supported by a single housing. Static reservoirs using nylasint are provided at both bearing pairs. This forms the bearing unit assembly. The criticality is the designing and achieving of labyrinth seal so as to avoid lubricant leakage as well as sufficient clearance between stator and rotor.

Encoder Assembly: Custom built 17 bit absolute optical encoder is used as angle sensor. It consists of three parts such as encoder rotor, encoder stator, and electronics on stator part. Configuration design ensures proper axial gap between stator and rotor disk of encoder.

The C.G. offset and cross inertia of Scatterometer rotor has to be balanced for static and dynamic balancing. Balancing provisions are provided in rotating parts in two balancing planes sufficiently apart. Locations are identified in other two orthogonal directions for minor cross inertia unbalance correction. Static & dynamic balancing of QM SSM with antenna has been done using 4 component Kistler dynamometer. The residual unbalance and the location of unbalance w. r. t. scan start are obtained. S/C level drift rate performance analysis has been carried out using the above values and the results are satisfactory.

Pennzane lubricant oil with its matching grease (Rheolube 2000/ MAPLUB) is used for SSM. To ensure lubricant availability, the amount of oil that will be lost by evaporation over the mission life is compared to the oil quantity available in the bearings. The rate of lubricant mass loss per unit surface area of oil is calculated to be 4.0mg/ cm^2/year. For 200m radial gap between stationary shaft and rotating retainer, by providing a labyrinth passage estimated opening is $1cm^2$ on each side of the bearings. Hence the expected loss from each bearing during the 5 year life is only 56mg. Oil retained by each bearing cage is more than 450mg. Hence, factor of safety = 450 / 56 = 8.

4.2.5.3 ROSA (Radio Occultation Sounder for the Atmosphere)

4.2.5.3.1 Introduction

The ROSA Receiver is a GPS Receiver for space borne applications, specifically conceived for atmospheric sounding by radio occultation, which is able to determine position, velocity and time using GPS signals.

The ROSA, besides providing real-time navigation data, is able to accurately measure pseudo ranges and integrated carrier phase (raw data), to be later processed on ground for scientific purposes.

The ROSA processes the received GPS signals in both the L1 and L2 frequency bands, allowing compensation of ionospheric delays. A codeless tracking scheme is included, in order to process the encrypted P(Y) signals transmitted in the L2 frequency band.

The instrument is equipped with one hemispherical-coverage antenna that is mounted with bore-sight direction equal to the Zenith direction and is used to track the GPS signals for navigation purpose and for Precise Orbit Determination (POD). In addition, a directive Velocity antenna is mounted on the Oceansat-2 spacecraft. This antenna is oriented in such a way to be able to track signal from GPS satellites in Earth occultation (rising).

Sixteen (4 AGGA chips) dual-frequency channels are available in the ROSA Receiver, and can be freely assigned to any combination of satellites. ROSA is provided with a MIL-STD-1553 communication interface over which telecommand, telemetry and measurement data are exchanged. The Receiver digital section is based on an ADSP 21020 processor and four AGGA-2a channels ASIC.

4.2.5.3.2 Main elements of ROSA Payload

Zenith pointing Hemispherical Antenna and LNA: Signals from this antenna can reach (through the RF/IF section) all the Receiver AGGA-2a HW channels. GPS signals received by this antenna are used to produce raw data measurements that are post-processed on ground for Atmospheric Sounding purposes and they are used also to compute the on-board real-time navigation and time solutions.

Velocity Antenna and LNA: This antenna is used only for Atmospheric Sounding applications to produce raw data measurements from satellites that are rising behind the Earth horizon. The Antenna is composed by two linear arrays, each pointing to a semi-plane (Velocity-Left, Velocity-Right). The RF/IF paths coming from Velocity arrays are connected to all AGGA-2a chips.

RF/IF sections: Five RF/IF sections (one for each antenna path) compose the Receiver front-end and include filtering and down conversion for the L1 and L2 frequencies. A 10 MHz reference

OCXO oscillator is used in the frequency synthesizer from which all the Local Oscillators, and also clocks used internally to the Receiver, are derived.

Digital Section: This section includes the Signal Processing HW(AGGA-2a channels), the CPU Module that controls via SW all the Receiver functions and the Communication Module that handles communication with the external Host (On-board Computer or Test Equipment). Four AGGA-2a chips are mounted on ROSA, each AGGA being composed of 4 complex (12 single) channels.

Power Management: ON/OFF commands are required from spacecraft to switch ON/OFF ROSA receiver. These commands issued by the Spacecraft and to execute these commands DC/DC Converter provides the secondary voltages to the Receiver, starting from the Spacecraft primary line.

4.2.5.3.3 Navigation Antenna

The Navigation Antenna is dedicated to acquire the GNSS signals to determine with precision the orbit of the satellite where there is installed the ROSA Instrument.

The navigation solution, from which depends the orbit determination, is fundamental in this application, because the position's knowledge in the time is essential to trace all the occultation events during the observation phase of the Instrument. Its main features are summarized by the following information.

Specifications:

Parameter	Value
Frequency range	L1→ 1560 - 1590MHz L2 → 1212 - 1242MHz
VSWR	1.5:1
Gain	-5dBic at Zenith 4dBic at 5° elevation above the horizon
Polarisation	RHCP
Radiation Pattern	Omni-directional – Azimuth Hemispherical - Vertical
Weight	0.23 kg

4.2.5.3.4 Radio Occultation Antenna

The Radio Occultation Antenna is a special Antenna designed and developed for this application. Its main purpose is to acquire and amplify the signal from the high atmospheric layers (ideally up to 600 km) to ground earth surface (ideally 0 km). The major part of the gain of this Antenna (about 12dB) is concentrated on angle of view of the lower atmospheric layers (under 100 km) where the signal is weaker due to the atmospheric absorption, refractivity, and multipath effects.

The main functional and performance features of this Antenna could be summarized in the following information:

Frequency bands: The R.O. antenna shall operate in two L frequency bands:

L1 = 1565.19 - 1613.86MHz & L2 = 1217.37 – 1256.36MHz

VSWR: For both L1 & L2 bands the VSWR shall be ≤1.4:1 (return loss ≥15.5 dB)

Polarisation: Circular Right Hand Polarisation is requested for both L1 and L2 frequency band.

Axial Ratio: The 3.5dB value can be considered only in the baseline coverage region; outside the axial ratio will be limited as much as possible.

Gain inside the baseline coverage region: The minimum value of 12dBi for both L1 & L2 bands is critical related to the requested antenna dimension.

Gain inside the extended coverage region: The constraint of the minimum gain of -3dBi in the extended coverage reduces the minimum gain value of the baseline coverage region.

Azimuth Gain Ripple: The antenna gain ripple shall not vary by more than 2dB for any azimuth variation inside the baseline coverage region (0-100 km). For the extended region the gain ripple will be minimized.

Functional Description

The main operations are:

- The ROSA Receiver performs the following main operations:
  - Allocates HW channels to GPS satellites
  - Receives L1/L2 C/A and P signals from GPS satellites
  - Acquires and maintains Code Lock and Carrier Lock, demodulates and decodes data
  - message and recovers Navigation Data from each received GPS satellite
    When at least 4 GPS satellites are in view, performs position, time and velocity calculation based on a Least Squares algorithm (ECEFSPS solution)
- In parallel, performs Filtered Navigation Solution (ECINKF solution); the Kalman filtered solution is able to propagate the solution also in absence of GPS measurements
- Uses calculated position information to establish geometrical line of sight information of each acquired GPS satellite with respect to the Receiver platform, maintains a tracking list of visible satellites and performs occultation events prediction
- For observation events, when Carrier Lock is not possible, performs Open-Loop high-rate sampling of raw observables for carrier reconstruction on ground
- Monitors and maintains Receiver Health & Status

4.3 Megha-tropiques

4.3.1 Introduction

Megha-Tropiques is an Indo-French Joint Satellite Mission for studying the water cycle and energy exchanges in the tropics. In the early 1990s, France wanted a 'Tropiques' satellite while India wanted a 'Climatsat' satellite. They merged the two ideas, resulting in a joint venture Megha-Tropiques. The name chosen for the satellite, Megha-Tropiques, reflected the mission's goals. 'Megha,' the Sanskrit word for clouds, underscoring a key focus of the satellite, and the French word 'Tropiques' denoting its concentration on the tropical region.

4.3.2 Mission Objective

The main objective of the Megha-Tropiques mission is to study the convective systems that influence the tropical weather and climate. The tropical region is the domain of monsoons, tropical cyclones. It is also characterized by large intra seasonal inter annual variations, which may lead to catastrophic events such as droughts or floods. Any change in the energy and water budget of the land-ocean-atmosphere system in the tropics has an influence on global climate. Objectives can be stated briefly as given below.

- To provide simultaneous measurements of several elements of the atmosphere water cycle, water vapour, clouds, condensed water in clouds, precipitation and evaporation.
- To measure the corresponding radiative budget at the top of the atmosphere
- To ensure high temporal sampling in order to characterize the life cycle of the convective systems and to obtain significant statistics

4.3.3 Orbital Parameters

Table 4-10: Orbital Parameters of Megha-Trophique

Parameter	Value
Orbit	Near circular/ equator
Altitude (km)	865.5 km
Inclination (deg)	20 Deg
Orbit Plane regression	6.01 deg/day
Apparent Sun Angle	52 days/Cycle
Orbit perigee	within +/- 10 km
Distance between successive orbit (km)	2892
Orbital Period(min)	101.91
Number of period/day	14.13
Launch vehicle	PSLV-C18

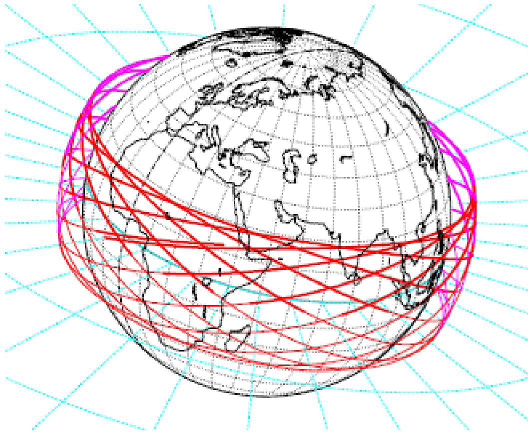


Figure 4-11 One day orbit pattern of Megha-Tropiques

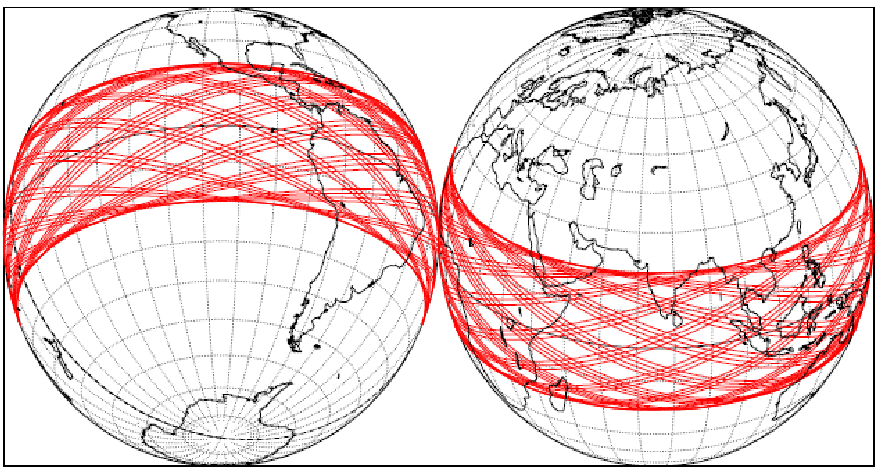


Figure 4-12: 3.5 day Orbit patterns of Megha-Tropiques

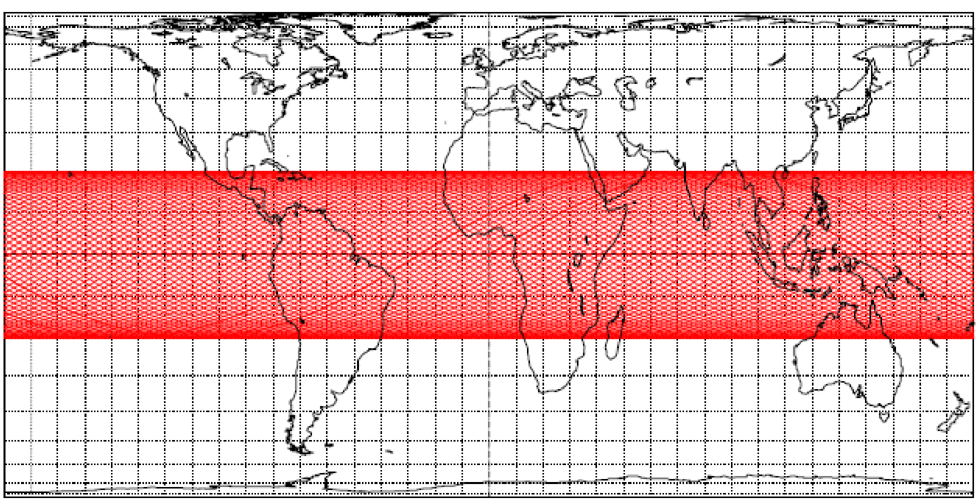
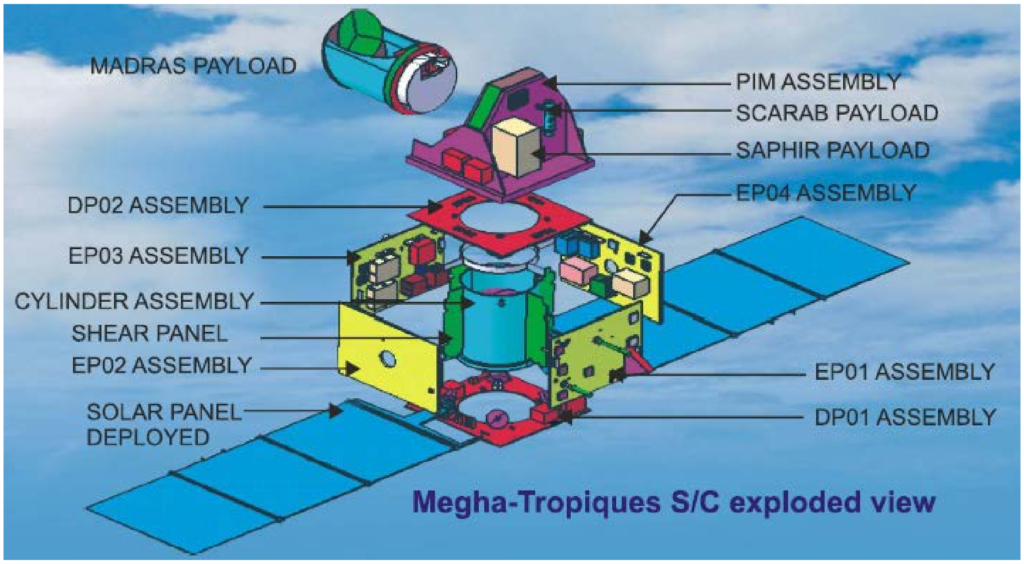


Figure 4-13 Orbital pattern of Megha- Trophiques

4.3.4 Salient features of Satellite

Subsystem		Megha Tropiques
Structure		IRS Bus structure with Separate Payload Interface Module (PIM) for Payload mounting.
Thermal	Control	Temperature control is with passive techniques using Paints, multilayer blankets, Optical solar Reflector, and active thermal elements like heaters.
	Limits	All electronics packages 0-40degC, Battery 0-10 deg ,
Mechanism	Solar Panel	Solar panel deployment mechanism and Drive Mechanism
	HRDM	Hold Down Release Mechanism for MADRAS payload
	ROSA	ROSA antenna deployment mechanism
Power	Solar Panel	Rigid, deployable, Sun Pointing, CFRP Faceskin, 15.12 m^2 , 6 panels 1.4 x 1.8 m^2 (Each), 58.8 kg, 50 mic. Kaptan insulator, 133 cells in series, 35 in parallel 10 string. 1180 W at EOL, GaInPas(In)GaAs/Ge ITJ
	Battery	2 Ni-Cd, 24 AH batteries, 28 to 42V, 28 Cells/bat., (32 kg)
	Power Electronics	2 buses, PWM TCR, FCL, 10 Strings , AH Meter
OBC	Telemetry	2250.00 MHz , PCM/PSK/PM, RT: 4 Kbps, PB: 16 Kbps
	Telecommand	PCM/PSK/PM, 2071.875 MHz
	Computer I/F	ASIC based processor interface logics
Data Handling	BDH	BDH & SSR in a single package with a storage capacity of 16 Gbits
Data Transmission		Payload data transmission through S-Band (2280 MHz) at a data rate of 5.2 Mbps
AOCS	Spec.	Pointing Accuracies: Yaw: $\pm$ 0.05^o Roll: $\pm$ 0.05^o Pitch: $\pm$ 0.05^o (3 sigma) Driftrate : $\pm$3 x 10^{-2} deg/sec (3 sigma)
	Sensors	Star Sensors(2), 4Pi SS(4), Magnetometer (2), IRU(3 DTG), SPS
	Actuators	Reaction Wheels 5 NMS(4 in tetrahedral), Magnetic Torquers (2) , 1N Thrusters(8) 11 N Thruster(4) Fuel (131 kg) Dry Mass(36 kg)
	Orbit	10/8 channels SPS
SADA		SADA with Micros stepping (Unified SADA)
Payloads		MADRAS, Saphir, SCARAB, ROSA
Mass		998 kg

The Megha Tropiques satellite can be divided into two main parts Main Bus and PIM(Payload Interface Module.



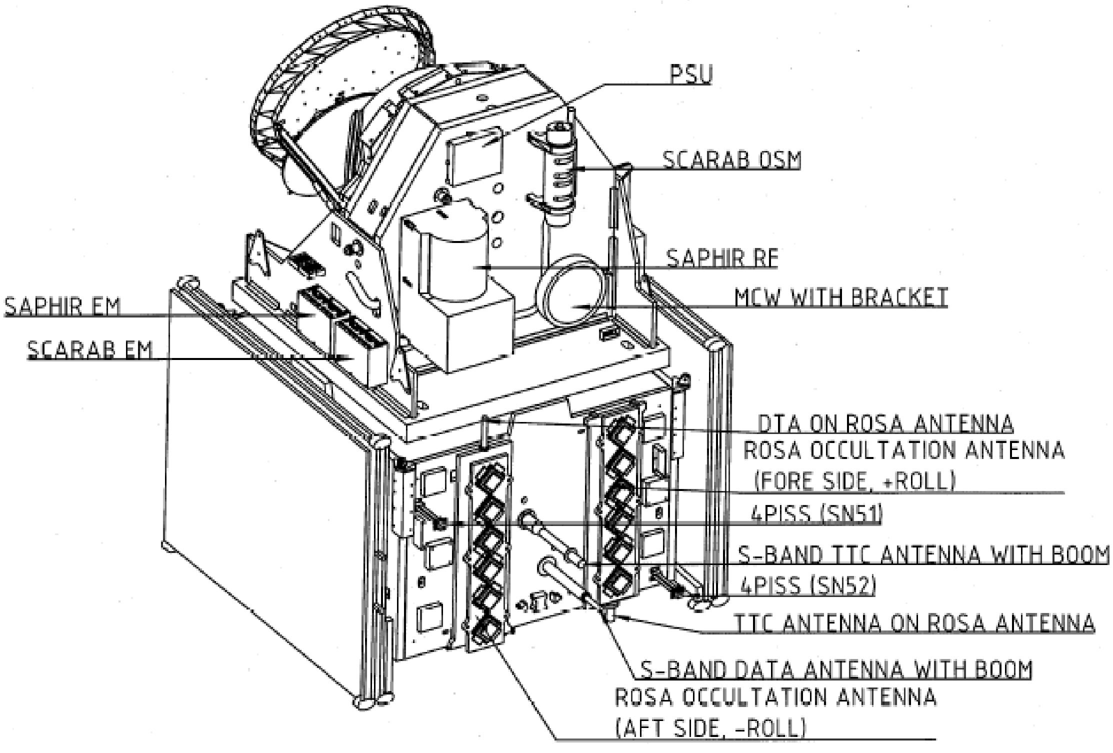


Figure 4-14 Stowed configuration of Megha tropiques

4.3.5 Payloads

Megha-Tropiques satellite carried 4 scientific passive instruments, ie :

MADRAS: (Microwave Analysis and Detection of Rain and Atmospheric Structures) A multi–channel self-calibrating microwave imager mainly aimed at studying precipitation and cloud properties.

SAPHIR: (Soundeur Atmospherique du profil d'Humidite Interopicale par Radiometric) A microwave instrument used to retrieve water vapour vertical profiles.

SCARAB: (Scanner for Radiation Budget) An optical radiometer devoted to the measurement of outgoing radiative fluxes at the top of the atmosphere.

ROSA: (Radio Occultation Sounder for Atmosphere Payload) A GPS Receiver specifically conceived for atmospheric sounding by radio occultation, which is able to determine position, velocity, and time using GPS signals.

Combining the information from these different payloads, the following parameters can be derived: size of convective cells, cloud cover, water vapour profiles, deep cloud water content, rain rate, cloud ice content and radiative fluxes, humidity content at the top of the atmosphere.

4.3.5.1 Microwave Analysis and Detection of Rain and Atmospheric Structures (MADRAS) Payload

The MADRAS instrument is a 9 channel self-calibrating microwave imager. The payload is jointly developed by ISRO and CNES. The payload scans the Earth +/65 deg, with onboard angle of +45.05 deg in the along track direction. The rotating part of MADRAS has a mass of about 100 kg. MADRAS is nominally a scanning payload. A **stationary mode** is defined for the payload, where the MADRAS is pointed a specific angle (within +/- 65 deg) continuously. IISU, SAC and ISAC have developed the MSM/MCE, MBE-R/MBE-S and HDRM components of MADRAS respectively, whereas MARFEQ-A & B have been developed by CNES.

Figure 4-15 MADRAS Payload

The MADRAS RF front-end consisting of the entire RF units from 18 GHz – 157 GHz including the antenna, feed cluster, and on-board calibration is designated as MARFEQ (MADRAS RF Equipment).

MARFEQ-A is the mobile part of MADRAS. It includes a structure supporting the main reflector associated to the horns located at the focal point of the parabola. Behind each horn one or several receivers allows the detection of the RF signals

- Main Reflector made from a CFRP dish (projected diameter 650 mm),
- Feed Cluster and Front Ends Assembly constituted by aluminum RF elements with their supporting structure and thermal hardware

- Back Ends, Low Frequency Receivers and Interface Electronic Unit,
- Related waveguides, coaxial cables and harness,
- CFRP structure which supports the above elements and their thermal hardware and which interfaces with the Scan Mechanism Rotating part through a titanium cylinder and with the Hold Down and Release Mechanism through 6 titanium integrated fittings.
- MBE(R) will be accommodated on the lower part of MARFEQ-A Deck.

MARFEQ B is mounted on the fixed part of the instrument. It allows the calibration of the receivers at each rotation. It is constituted of a mirror allowing a cold calibration and a black body allowing the hot calibration. This equipment contains only accurate thermistors to measure the physical temperature of the black-body.
The MARFEQ fixed part (MARFEQ-B) consists of:

- Cold Calibration Reflector made from a CFRP dish (projected diameter 285mm),
- Hot Calibration Target
- Aluminum structure which supports the above elements and interfaces with the Scan Mechanism fixed part.

4.3.5.1.1 Hold Down and Release Mechanism (HDRM)

This is necessary to protect MSM bearings from launch loads, since the rotating elements of MADRAS high. The mechanism will rigidly hold the MARFEQ at six locations. Once the spacecraft is in orbit, the mechanism will release the MARFEQ, to enable scanning. MSM, MCW and MCE form MADRAS Mechanism and Momentum Compensation System (MMCS). MMCS has four modes of operation, viz.

- Run-up mode
- Scanning mode
- Pointing mode
- Run down mode.

4.3.5.1.2 MADRAS Momentum Compensation System (MMCS)

MMCS as a part of MADRAS payload consists of three elements such as;

MADRAS Scan Mechanism (MSM) : Scan Mechanism consisting of precision angular contact ball bearing assembly, Diaphragm assembly for hold down compliance, drive motor, optical encoder, PSTD for transfer of power and signal from and to Marfeq A/MBE(R) and MBE (S). The nominal speed of the mechanism is 24.14 rpm with scan stability of +/- 0.1%.

Momentum Compensating Wheel (MCW): Momentum Compensating Wheel MCW consists of precision ball bearing assembly, flywheel, brushless iron less DC motor in a hermetically sealed casing. The MCW (Momentum Compensative Wheel) generates counter momentum such that the residual momentum is very small and tolerable by the spacecraft. MCW consists of a wheel with a low mass but high rotational speed to generated compensative momentum.

MADRAS Control Electronics (MCE): The MADRAS Control Electronics (MCE) containing all the electronic functions as management of the MSM, MCW, commutation electronics, power supply, mechanisms command and control, interface with MBE.

MCE (MSM and Momentum Compensating Electronics) is an integrated control electronics package for MSM. Rotating mass of the payload (100 kg) generates a large momentum about its axis of rotation, which can destabilize the platform.

4.3.5.1.3 Payload Characteristics

MADRAS channel definitions

Channel No.	Frequency	Polarization	Pixel size	*Bandwidth*	Science Parameters
M1	18.7 GHz	H+V	40 km	±100 MHz	Rain above oceans
M2	23.8 GHz	V	40 km	±200 MHz	Integrated water vapour
M3	36.5 GHz	H+V	40 km	±500 MHz	Liquid water in clouds, rain above sea
M4	89 GHz	H+V	10 km	±1350 MHz	Convective rain areas over land and sea
M5	157 GHz	H+V	6 km	±1350 MHz	Ice at cloud tops

Parameter	**Value**
Scan type	Conical scanning at constant speed
Onboard look angle (w.r.t. Nadir)	45.05°
Incidence angle	53.5°
Maximum scan angle (cross track)	±65°
Scan mechanism speed	24.14 rpm⇔ 144.84 °/sec ⇔ 2.4855 sec/revolution ⇔ 0.4023 cycles/sec = 0.4023 Hz.
Dwell time (Channel wise)	16.8 millisec (18.7, 23.8, 36.5 GHz) 4.2 millsec (89 GHz) 2.5 millisec (157 GHz)
Swath	1700 km
Dynamic range of radiometer Brightness temperature Scan mechanism stability	 3 °K to 320 °K 0.1% of the rate

Data rate: **The total data rate of MADRAS is 33.858 Kbps.**

Type of data	**Size**	**Rate**

Science TM	5248 words	2.48sec

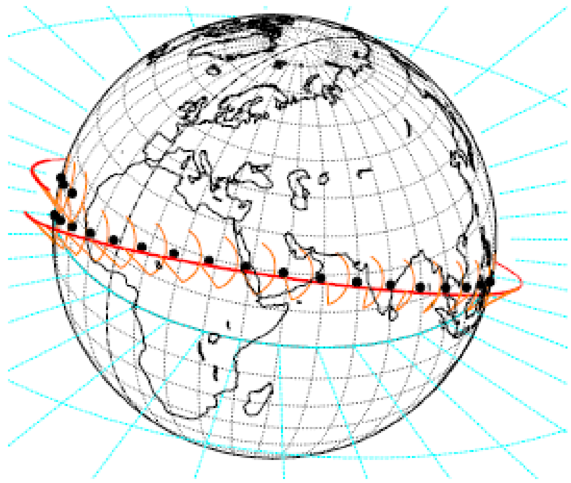


Figure 4-16 MADRAS Swath coverage pattern

4.3.5.2 SAPHIR (Soundeur Atmospherique du Profil d'Humidite Intertropicale par Radiometrie) Payload

SAPHIR is a Microwave instrument for the retrieval of water vapour vertical profiles and scanning millimeter wave humidity sounder. It scans the Earth in a nadir plane symmetrically with respect to the local vertical with a scan angle of ±42.96 deg. It uses narrow channels close to a water vapour absorption band at a frequency of 183 GHz. Six channels would allow to retrieve information about six atmospheric layers from the Earth surface up to 12 km height. The horizontal resolution is 10 km. The 6 channels are in the range of 183.31 ±0.2, ±1.1, ±2.8, ±4.2, ±6.8, ±11.0 (GHz).

The instrument is composed of two packages linked by a dedicated harness. The packages are

The **RF Unit** (6 Passive microwave channels) contains the antenna, the frontend, IF processor, the scanning with the shroud and the calibration target.

The **Electronic unit** (EU) containing all the electronic functions as management of the equipment, power supply, mechanisms command and control interface with satellite processor.

Following are highlights of its operation. Scans Earth's atmosphere and switches between the calibration sources of cold sky and hot target

- During each scan period the antenna performs one complete rotation in order to scan the Earth over an angle of +/- 42.96 deg and performs hot and cold calibration
- During Earth scanning of +/- 42.96 deg, in nominal mode the angular speed is constant and equal to 103.5 degree/sec.
- During the rest of scan period, in order to optimize the time dedicated to Earth's atmosphere measurements, the motor will produce half part of the time a constant and

maximum acceleration and on the other half part of the time a constant maximum deceleration. The current values of acceleration and deceleration [20] are 1666 deg/sec^2 and 1666 deg/sec^2.

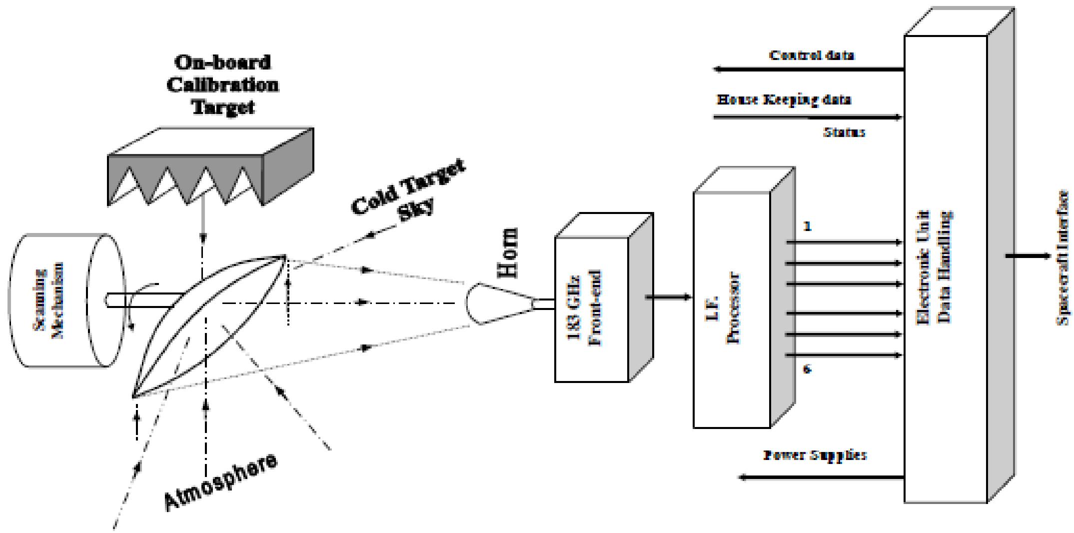


Figure 4-17 Schematic of SAPHIR Payload

SAPHIR channel definitions and characteristics

Channels	Central nominal frequencies (GHz)	Nominal Bandwidth (MHz)	ΔT (Sensitivity)	
			Req.	**Goal**
S1	183.31 ± 0.2	200	2 °K	1 °K
S2	183.31 ± 1.1	350	1.5 °K	0.7 °K
S3	183.31 ± 2.8	500	1.5 °K	0.7 °K
S4	183.31 ± 4.2	700	1.3 °K	0.6 °K
S5	183.31 ± 6.8	1200	1.3 °K	0.6 °K
S6	183.31 ± 11	2000	1.0 °K	0.5 °K

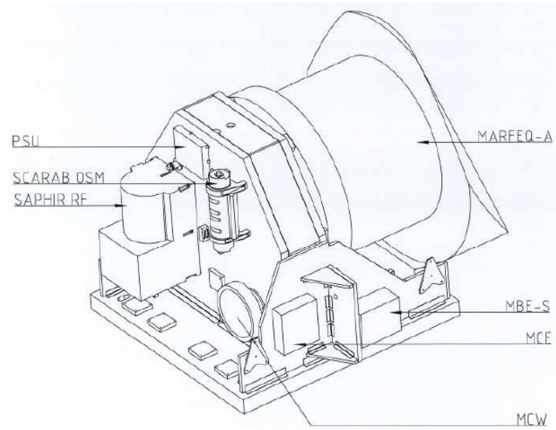


Figure 4-18 Payload Interface Module

Parameter	Value
Scan type	Conical scanning at constant speed
Polarisation	Variable along swath
Incidence angle	Variable along swath
Maximum scan angle(cross track)	± 42.96°
Scan mechanism speed	36.63 rpm⇔ 219.78 °/sec ⇔1.638 sec/revolution ⇔ 0.6105 Hz
Scan rate during Earth viewing phase	103.5 °/sec
Dwell time	6.406 msec
Swath	1705 km
Nadir spatial resolution	10 km
Dynamic range of radiometer Brightness temperature	4 °K to 313 °K

Data Rate:

The total data rate of SAPHIR is 12.487 Kbps. The different kinds of data coming to BDH from SAPHIR instrument are given in the table below.

Type of data	Size	Rate
Science TM	1216 words	1.64 sec
Aux data	64 words	19.6 sec

4.3.5.3 ScaRaB (Scanner for Radiation Budget) Payload

Radiometer devoted to the measurement of outgoing radiative fluxes at the top of the atmosphere. Measures radiation fluxes in four channels in the range of 0.5 to 0.7μm, 0.2 to 4 μm, 0.2 to 50 μm and 10.5 to 12.5 μm spectral bands; in Visible, Solar, Total and IR Windows. It consists of (a) Optical Sensor Module (including scanner and calibration devices) and (b) Electronic Module.

The optical sensor module (OSM) can be divided in two parts

- A rotating part with mechanism, four detectors, two choppers, an internal electronics and a filter wheel.
- The external structure, with the casing, the two feet and the calibration module (CalM) formed by three black body simulators and a lamp.

The Electronic Module (EM) containing all the electronic functions as management of the equipment, power supply, mechanisms command and control interface with satellite processor.

Following are the highlights of its operation.

- During each scan period the rotor performs one complete rotation in order to scan the Earth over an angle of +/- 48.91° and performs calibration on cold space.
- Produce part of time some acceleration and part of time some deceleration. In the table, “te” represents ‘elementary time’ = pixels sampling period = 62.5ms. The total duration for one full scan is 6 sec.

4.3.5.3.1 Scanning sequence in nominal mode

Function	Angle	Typical Duration	Type of movement
Earth/Atmosphere Scanning	-48.91° to +48.91°	51Xte = 3.1875 sec	Constant speed
Switching period	+48.91° to -74.35°	30Xte = 1.875 sec	Acceleration/ Deceleration
Stop on space view	-74.35°	6Xte = 0.375 sec	Stop(fixed position)
Switching period	-74.35° to -48.91°	9Xte = 0.5625 sec	Acceleration/ Deceleration
Total Period		**96*te = 6 sec**	

In the background of the discussion above on the working of payloads, it is evident that the payloads’ scanning are asynchronous. Further SAPHIR has acceleration and deceleration before it scans the Earth portion. Similarly in case of SCARAB, in addition to acceleration and deceleration, it stops to view deep space for a finite amount of time. This is likely to cause disturbance on the platform with impact on spacecraft control and eventually Data Products Generation. During this exercises, it emerged that platform rates achievable are of the order 10^{-2}deg/sec.

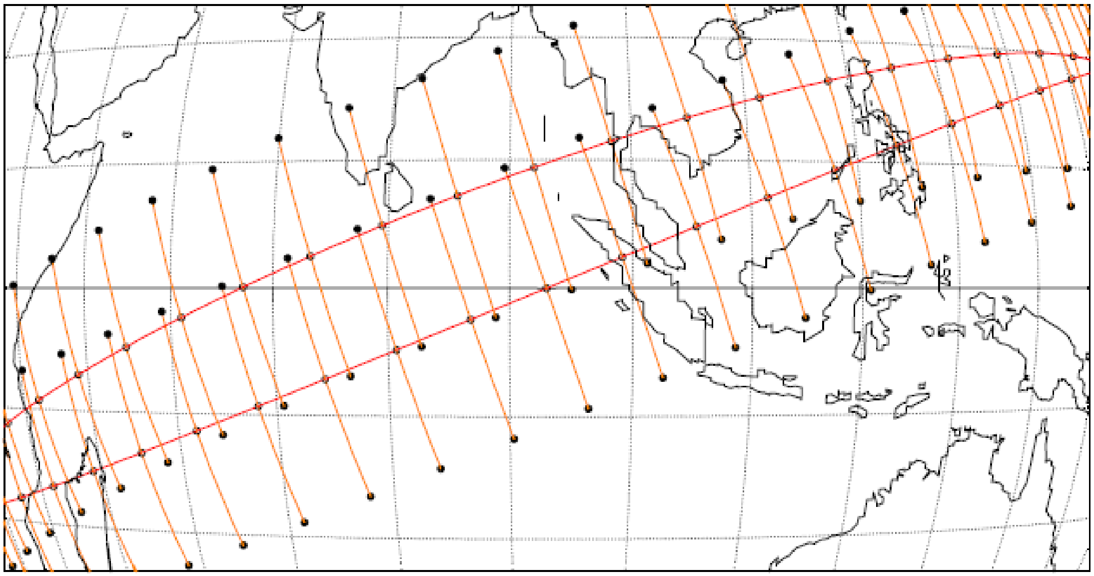


Figure 4-19: Scarab Scanning lines and Overlap

ScaRaB channel definitions and characteristics

Channels	Wavelength	*Radiometric Resolution (Noise)*	Signal Dynamics (Max.)
Sc_1 – Visible	0.5 to 0.7 μm	< 1 $W.m^{-2}.sr^{-1}$	120 $W.m^{-2}.sr^{-1}$
Sc_2 – Solar	0.2 to 4 μm	< 0.5 $W.m^{-2}.sr^{-1}$	425 $W.m^{-2}.sr^{-1}$
Sc_3 – Total	0.2 to 50 μm	< 0.5 $W.m^{-2}.sr^{-1}$	500 $W.m^{-2}.sr^{-1}$
Sc_4 – IR Window	10.5 to 12.5 μm	< 0.5 $W.m^{-2}.sr^{-1}$	30 $W.m^{-2}.sr^{-1}$

Scan type : Cross track scanning at constant speed

Parameter	**Value**
Scan Angle (across track)	± 48.91°
Scan mechanism speed	10 rpm⇔ 60 °/sec ⇔ 6 sec/revolution ⇔ 0.17 Hz
Dwell time	62.5 msec
Swath	2242 km
Nominal nadir spatial resolution	40 km

Data rate:

The total data rate of SCARAB is 853.333 Kbps. The different kinds of SCARAB data coming to BDH are given in the table below.

Type of data	**Size**	**Rate**
Science TM	256 words	6 sec
Aux data	64 words	6 sec

4.3.5.4 ROSA

The ROSA is a 16-channel dual-frequency GPS (Global Positioning System) receiver for space borne applications, specifically used for atmospheric sounding by radio occultation and determines position, velocity and time using GPS signals. The ROSA processes the received GPS signals in both the L1 and L2 frequency bands, allowing compensation of ionospheric delays. A codeless tracking scheme is included, in order to process the encrypted P(Y) signals transmitted in the L2 frequency band.

The ROSA, besides providing real-time navigation data, is able to accurately measure pseudo-ranges and integrated carrier phase (raw data), to be later processed on ground for the scientific purposes of retrieval of atmospheric parameters such as Humidity, Pressure and Temperature profiles between 0 and 100 km height above the Earth surface. These profiles can be used in meteorological and climatologic forecast with a vertical resolution much higher than that obtainable with measurement based upon microwaves or infrared techniques.

ROSA payload on Megha-Tropiques will supplement / complement the mission objectives for the atmospheric studies. ROSA on Megha-Tropiques spacecraft has a fore and an aft antenna facilitating occultation measurements in both velocity and anti-velocity directions of the spacecraft thus allowing a large number of observations. The Navigation antenna looking along the spacecraft's zenith direction facilitates the precise orbit determination (POD).

GPS ROSA, raw data and the products are generated at ISSDC Bangalore and are also archived for further use by application scientists.

ROSA Specifications

Features	Specification
Dual Frequency operation:	Receiving Frequencies: L1 [1575.42 MHz] C/A-Code signal L1 [1575.42 MHz] P-Code signal L2 [1227.60 MHz] P-Code signal
Bandwidth:	±10 MHz nominal
Number of Dual-Frequency Channels:	16 Dual-Frequency channels. Allocating channels to POD or Occultation is managed automatically only by onboard software in order to optimally share the hardware resources (channels).
Measurement rate:	Navigation/POD: 1 Hz sampling data rate (for both code phase and carrier phase) Carrier Phase measurements for Occultation/space weather channels:

	(a)Observation (Close loop): 1 Hz, 10 Hz and 50 Hz sampling data rate depending on altitude of the tangent point. (b)Occultation (open loop): 100 Hz sampling data rate only in lower troposphere.
Measurement accuracy:	Pseudo range:< 50 cm Carrier phase:< 5 mm Bending angle: Better than 1 μrad
On-board POD software with Satellite positioning accuracy:	< 30 metre (Real-time 3D-3σ solution)
Receiver Power consumption:	45 Watts (Operating Mode)
Receiver operating voltage:	+28V to +42V DC (37 V nominal)
Navigation input signal levels:	L1-CA: -127 dBm (minimum) L1-P: -130 dBm (minimum) L2-P: -133 dBm (minimum)
Radio Occultation input signal range:	L1-CA: -130 dBm to -142 dBm, -132 dBm (nominal) L1-P: -133 dBm to -145 dBm, -135 dBm (nominal) L2-P: -136 dBm to -148 dBm, -138 dBm (nominal)
Interfaces with satellite platform:	House-Keeping Telemetry, Telecommand and Science Telemetry interface: Mil-Std-1553B Science Telemetry format: Space Packet as per CCSDS 133.0-B-1 Pulse Per Second (PPS) signal interface: RS-422 Number of PPS output signals:2
Receiver Mass:	9.2 kg
Receiver Dimension:	290.6 mm x 334.6 mm x 207.7 mm
Receiver operating temperature:	-10 °C to +45 °C

The total data rate of ROSA varies from 11.264 Kbps to 113.664 Kbps.

Type of data	Size	Rate
Navigation and Observation data	704 to 7104 words	1 sec

Dual-Frequency ROSA antenna specifications

Features	Specification
FOV (Field of View):	Azimuth FOV (referred to orbital plane): ± 30° Elevation FOV (referred to local zenith) • Baseline coverage region: 116.7° to 118.3°

	•Extended coverage region: 90° to 116.7°
Antenna Gain:	Gain inside the coverage region: ≥ 12 dBi for both L1 and L2 band Gain inside the extended coverage region: ≥ -3 dBi for both the L1 and L2 band
Polarization:	Right Hand Circular Polarization (RHCP)
VSWR:	1.4:1
Mass:	2.5 kg
Dimensions:	1050 mm x 280 mm x 80mm (Single panel patch array)
Operating temperature:	-80 °C to +100 °C

Dual-Frequency Navigation/POD antenna specifications

Features	Specification
FOV (Field of View):	± 75° (referred to local zenith)
Antenna Gain:	5 dBi (at zenith) 4 dBi (at 5° elevation above the horizon)
Polarization:	Right Hand Circular Polarization (RHCP)
VSWR:	1.5:1
Mass:	0.138 kg
Dimensions:	127 mm x 49 mm
Operating temperature:	-70 °C to +80 °C

4.4 SARAL (Satellite for ARGOS and ALTIKA Payloads)

4.4.1 Introduction

The first mission for the Mini- satellite bus named as 'IMS-2' is being designed and developed for a mini satellite series in the weight range of 400 to 450 kg capable of carrying payloads up to a weight of 200 kg. The first satellite in the series is envisaged to carry two payloads of CNES called ALTIKA & ARGOS. ALTIKA mainly consists of an altimeter useful for ocean topography and ARGOS is a data collection platform to collect weather information from ocean buoys for weather predictions. The first mission with these payloads is called 'SARAL' (Satellite for ARGOS and ALTIKA Payloads) mission. It is planned to be launched onboard PSLV in 2010.

4.4.2 Mission Objective

- To design and develop a satellite bus (IMS-2 bus) in the weight range of 400 kg including payloads up to a mass of 200 kg.
- The first bus IMS-2 will carry two payloads namely ARGOS and ALTIKA from CNES. The first mission will be called the "SARAL" (Satellite with ARGOS and ALTIKA Payloads) mission
- To develop required ground processing software for Altika payload and utilize the payload data within India for ocean related applications

4.4.3 Orbit Details

Parameter	Value
Orbit type	Sun Synchronous Orbit (SSO)
Orbit Radius (km)	7159.5
Mean altitude (km)	799.8
Inclination (deg)	98.55
Eccentricity	$1.165\ 10^{-3}$
Orbit period	100.59
Local time at ascending node	6.00 AM
Cycle	35 days

4.4.4 Salient Feature of Spacecarft

The bus design is being done to miniaturize the currently proven operational main frame sub-systems. These can be used for the mini satellite series and also it is planned to identify the new technologies required for accommodating futuristic requirements of payloads and of mini satellites. The SARAL mission envisages

- A small satellite bus to carry two payloads namely ARGOS and ALTIKA from CNES.
- The data reception and analysis facilities for data from these payloads.
- To explore the utilization of this data within India by involving the user agencies

Table 4-11 Salient features of IMS-2 Bus (SARAL)

Subsystem		SARAL
Structure		Cuboid, Aluminum , Honey-comb panels with CFRP/AL face skin
Thermal	Control	Temperature control is with passive techniques using Paints, multilayer blankets, Optical solar Reflector, and active thermal elements like heaters also..
	Limits	All electronics packages 0-40degC, Battery 0-10 deg ,

Mechanism	Solar Panel	Solar panel deployment mechanism and drive mechanism
Power	Solar Panel	Deployable Solar Panels with ITJ cells Four panels (two on either side in Roll), ~700 watts
	Battery	Single Li Ion battery - 21AH, 350w continuous
	Power Electronics	
RF Systems	Telemetry	1024 bits, 2245.68 MHz , storage: 6.29×10^6 Bits PCM/BPSK/PM, RT: 4 KBPS, PB: 64 KBPS 1W for S-Band
	Telecommand	FM/PSK/PCM, 2067.897 MHz, 4 KBPS
	DH	QPSK for X-Band, 8 W for BDH
Data Transmission	BDH	
	SSR	Mission specific (~32Gb SSR)
AOCS	Spec.	Pointing Accuracies: Yaw: $\pm 0.05^o$ Roll: $\pm 0.05^o$ Pitch: $\pm 0.05^o$ (3 sigma) Driftrate : 5×10^{-5} deg/sec (3 sigma)
	Sensors	Four pi (4Heads), Magnetometer (Two- tri axial), Star sensor, miniaturized IRU.
	Actuators	RCS 4+4 thrusters of 1N & 0.2N, monopropellant, 30 kg single tank, Four 5NMS reaction wheels in tetrahedral Configuration, two magnetic torquers $20Am^2$
	Orbit	GPS : Miniaturized SPS with dual frequency
Mass		<400 kg

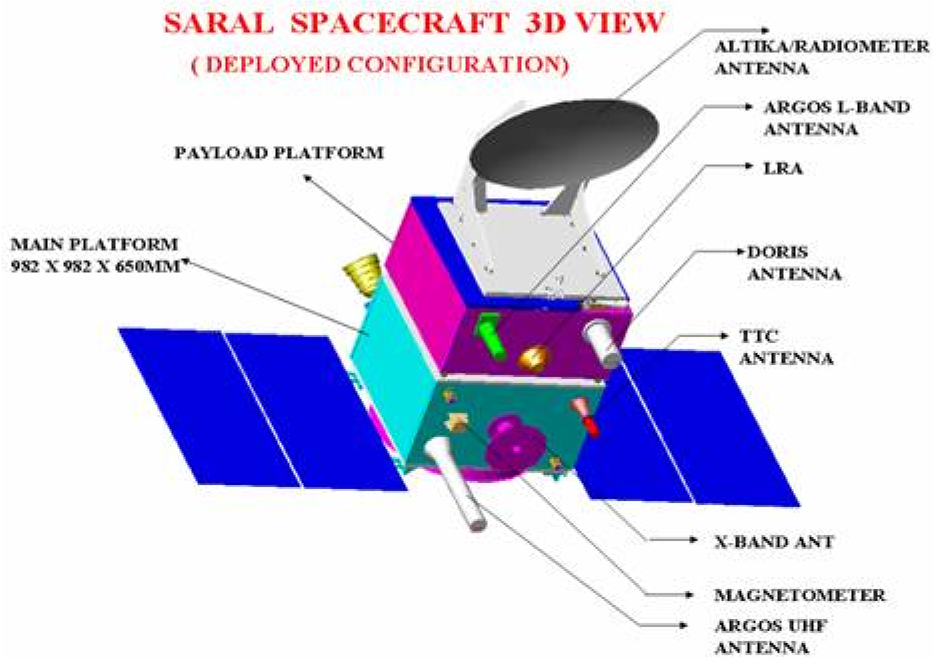


Figure 4-20 Exploded view of SARAL

4.4.5 SARAL Payloads

4.4.5.1 ARGOS PAYLOAD

ARGOS is a global satellite system dedicated to science applications such as metereological observation and environment study and protection, which is operated by the French National Space Reasearch Centre (CNES) and the National Oceanic and Atmospheric Administration (NOAA) of USA.

It consists of unattended data collection platforms distributed over sea and land which transmit data to the satellites, the ARGOS space equipment which receives the data transmitted by the data collection platforms and transmits them to the ARGOS ground station, and the ATGOS receiving sationas are located in Fairbanks and Wallops in USA, and Swalbard in Norway. The global processing centres are in Toulouse, France and lorgo, USA, ArGOS also provides determination of the location of the platforms in addition to relaying their data. Currently there are more than 16000 data collection platforms and 900 Users from 70 countries.

Typical uses of the ARGOS system are environmental minitoring, oceanography, meteorology, monitoring of vocanoes and water bodies, tracking of animals and birds, fleet management, etc.

SARAL is the joint satellite mission in which ISRO would provide the satellite bus and CENS would provide the ALTIKA Ka band altimeter and associated instruments, and ARGOS space equipment. By placing the ARGOS instrument on this satellite, ISRO will become a member of the international ARGOS Operations Committee and also come into contact with other users of

ARGOS around the world for exploring the possibility of scientific sooperation in using data collected by the ARGOS data collection platforms.

The new Advanced Data Collection System (A-DCS) insrument is designed to meet the requirements of the third generation of the Argos system defined in the Argos-3 mission specification.

A-DCS (ADVANCED DATA COLLECTION SYSTEM) developed by Alcatel for CNES includes:

- UHF Transmitter (465.98 MHz, 5W, 400bps)
- L- Band Transmitter (1.7 GHz, 1W, 7.5 kbps)
- Antenna, UHF Diplexer, Receiver Processing Unit
- Data Interface Unit

4.4.5.2 ALTIKA PAYLOAD

The aim of ALTIKA mission is to provide altimetric measurements designed to study ocean circulation and sea surface elevation. The applications include Marine meteorology and sea state forecasting, Operational oceanography, seasonal forecasting, climate monitoring and climate research.

ALTIKA Payload includes Following Instruments

Parameter	Value
A Ka Band altimeter (35.5 - 36 GHz):	The advantages of Ka band altimeters are It has negligible Ionospheric effects and Better vertical Resolution. The Ka Band altimeter is a compact lightweight instrument easier to accommodate on a wide range of satellite buses.
A dual frequency radiometer (24 / 37 GHz)	Required for tropospheric correction Derived from Madras (Megha-Tropiqus) Development
LRA (Laser Retro-reflector Array)	Useful for orbitography and system calibration
DORIS (Doppler Orbitography and Radio positioning Integrated by Satellite)	For adequate orbitography performance in low earth orbit Enable to have similar performance as reference missions like T/P, Jasan, ENVISAT Required for mean sea level analysis and coastal/inland applications

4.5 ScatSat-1

4.5.1 Introduction

ScatSat-1 is a continuity mission for Scatterometer payload (Scat-1) on-board Oceansat-2 spacecraft. This mission will provide continuity of weather forecasting services to the user communities, as Scat-1 payload was declared non-operational. The data from this payload was being used by many national and international users. Hence, demand for a new satellite with only Scatterometer has come up.

ScatSat-1 will carry Ku-Band Scatterometer named as Scat-2 payload (Oceansat-2 Scatterometer was Scat-1) similar to the one flown on-board Oceansat-2 but with some enhanced features. In order to take care of Oceansat-2 Scatterometer issues, like thermal and component related, cross-couplings at crucial places and improved thermal design is addressed in ScatSat-1 (Scat-2). Scatterometer is an instrument working on the principle of back-scattered energy and is used to measure the wind velocity (speed and direction) over the ocean.

The payload is configured on Indian Mini Satellite-2 Bus (IMS-2). The first mission of IMS-2 bus is SARAL. IMS-2 Bus is evolved as a standard bus of 450 kg class which includes payload capability of around 200 kg. IMS-2 development is an important milestone as it is envisaged to be a workhorse for different types of remote sensing applications. The IMS-2 Bus is designed with modularity, miniaturization and standardization. It is an operational class Satellite Bus with complete redundancy in mainframe systems. ScatSat-1 is the second spacecraft configured on IMS-2 Bus with some modifications. The Scatterometer payload will be accommodated on top deck top side of the bus.

4.5.2 Mission Objectives

The mission objectives of ScatSat-1 are as follows:

- To provide continuity of weather forecasting services to the user communities.
- To generate wind vector products for weather forecasting, cyclone detection and tracking.

4.5.3 Orbit Details

The orbit details for ScatSat-1 are given in the table 4-12 below:

Table 4-12 Orbit details of ScatSat-1

Parameter	ScatSat-1
Spacecraft Mass	371 kg
Payload Mass	80 kg (without structure support)
Spacecraft Size	2185 (Y) x 1253 (R) x 2109 (P) mm^3 (Stowed) 2185 (Y) x 4418 (R) x 2109 (P) mm^3 (Deployed)
Average Power Generated	280 W

Average Payload Power	120 W
Altitude	720 km
Orbit	Sun Synchronous Orbit SSO
No. of orbits/day	14 + 1/2
Inclination	98.10° / 98.28°
Local time	9.20 am
Stabilization	Three Axis stabilization
Launch Date	26 September 2016
Launch Site	Shriharikota
Orbital Period	99 min 11 sec
Swath	Inner: 1400 km (both HH & VV beams available) Outer: 1400 - 1800 km (only VV beams available)

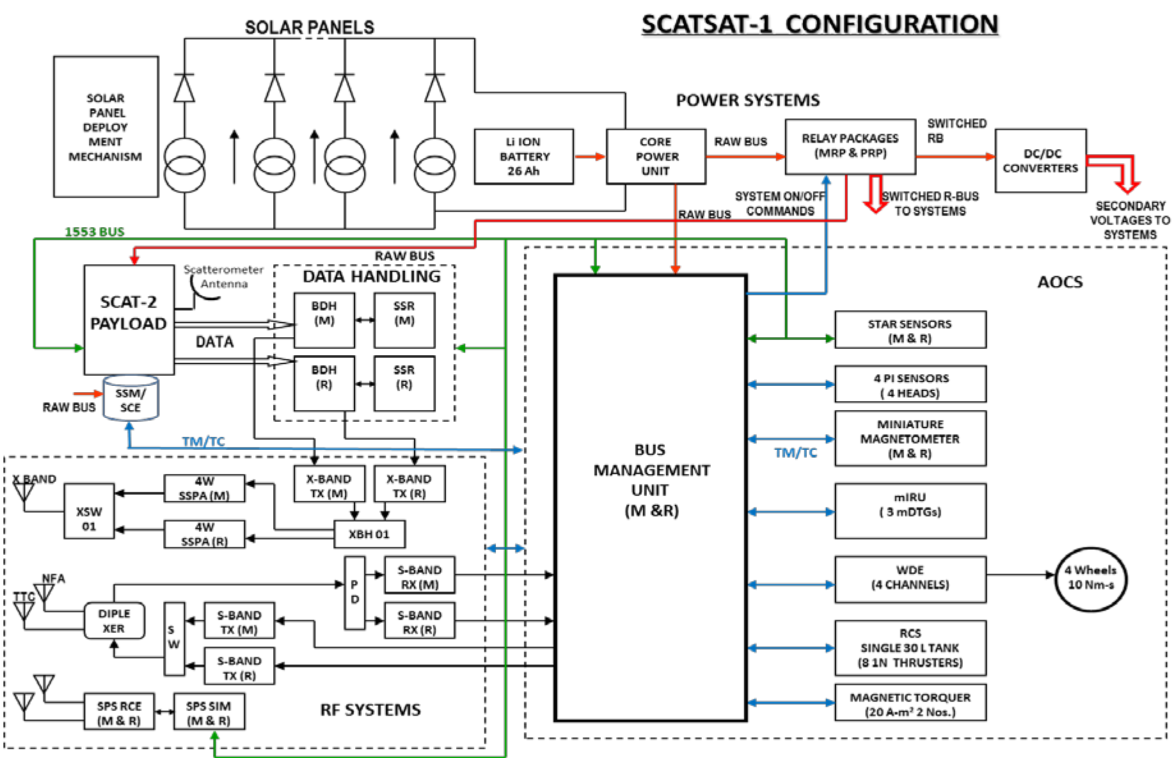


Figure 4-21 ScatSat-1 Configuration

4.5.4 Salient features of ScatSat-1

Table 4-13 Features of ScatSat-1 Systems

Subsystem		Specification
Mass		371 kg (Mainframe: 266 kg, Payload: 105 kg)

Payload		Scat-2
Structure		Cuboid built by 4 vertical panels & 2 horizontal panels
Thermal Control		MLI, Thermal coatings, Diffuser plates, OSR along with auto controlled heater system
Mechanisms		1. Paraffin Actuator Based Hold down and Release Mechanism for Solar Panel Deployment 2. Pyro Based Hold down and Release Mechanism for Scatterometer Antenna Deployment
AOCS	Specification	1. Pointing : ± 0.1° (3σ) 2. Drift Rate : ± 3.0 x 10^{-04} º /s (3σ)
	Sensors	Mark 2 type Star Sensor (M&R), IRU (3 DTGs), Magnetometer (M&R), Four 4 Pi Sun Sensors
	Actuators	1. Four nos. of 10 Nm-s Reaction Wheels (10 Nm-s Angular Momentum, 0.05 Nm Reaction torque) 2. 20 A-m^2 Magnetic Torquers along Roll and Pitch, 3. RCS is monopropellant type with single 21 kg tank, 4. 8 nos. of 1 N Thrusters
Power system	Solar Panels	2Sx2P (753 W at EOL)
	Battery	28 Ah
	Electronics	Core power package (MPW11)
RF system	Telemetry	TM Data Rate: 8 kbps (RT) (After channel coding); Frequency: 2255.14 MHz Modulation: BPSK, RF Power: 100 mW
	Telecommand	TC Data Rate: 4 kbps, Demodulation: FM/PSK/PCM Frequency: 2088.38 MHz
	Payload Data System	Data Rate: 64 Mbps (After channel coding), QPSK, 4 W SSPA, Frequency: 8300 MHz
	SPS	12 Channels, L1 & C/A

SSR		32Gb

4.5.5 SactSat-1 Payload

ScatSat-1 is a dedicated spacecraft configured to carry Scatterometer Payload named Scat-2 (Oceansat-2 Scatterometer is called Scat-1/OSCAT). Scatterometer Payload consists of Digital Sub-System (Payload Controller + Digital Chirp Generator + Data Acquisition & Compression System), Frequency Generator, Receiver, Transmitter, Travelling Wave Tube Amplifier, Front End Switch Assembly, Scatterometer Scan Mechanism and Ku-band reflector antenna. Scatterometer Scan Mechanism along with its control electronics was delivered by ISRO Inertial Systems Unit (IISU), reflector by CMSE and payload structure by ISAC.

4.5.5.1 Scatterometer

Ku-Band pencil beam Scatterometer is an instrument working on the principle of back-scattered energy and is used to measure the wind velocity (speed and direction) over the ocean.

Wind flow over the ocean surface generates and amplifies surface waves of centimeter wavelength. Changes in wind velocity (speed and direction) cause changes in ocean surface roughness, which in turn affect the radar cross-section and hence the magnitude of the backscattered power. The normalized radar cross section ($\sigma°$) is estimated from the measured backscattered power to estimate the wind speed. The $\sigma°$ is a function of wind speed, relative wind direction, incidence angle, polarization and radar frequency.

Knowledge of wind velocity (speed and direction) over oceans is critical for the understanding and prediction of many oceanographic, meteorological and climate phenomena.

Ku-Band pencil beam Scatterometer Payload was launched on 23rd September 2009 on Oceansat-2 satellite along with other two payloads namely Ocean Colour Monitor (OCM) & Radio Occultation Sounder for Atmospheric studies (ROSA) from SHAR launch pad using PSLV-14 launcher. It was launched into a near polar sun-synchronous orbit of 720 km altitude with the local time of equatorial crossing in the descending node at 12 noon ± 10 minutes. Based on the large amount of data received and analyzed, it is proven that they are consistent and of immense use to the scientists and user agencies engaged in ocean studies, wind vector retrieval, weather forecasting and climate studies. Scatterometer is a global mission and payload is designed for five years life.

Incidentally, the Ku-band Scatterometer on Oceansat-2 was the only sensor of its kind anywhere in the world and proving its value to the scientific community. The downloaded semi-processed data from the satellite sensor is further processed at NRSC, Hyderabad and ground derived wind vectors were being posted daily at 12 hourly intervals i.e. 5 AM and 5 PM on the internet for utilisation by the user community. Due to certain anomalies in the Traveling Wave Tube Amplifier (TWTA) in the Scatterometer Main chain, the Scatterometer ended its mission pre maturely on 2nd April 2014. As the Indian and global meteorological agencies were utilising the Scatterometer operationally, currently there is a gap of such data. Keeping in view, the user interest and requirement for providing timely updates of sea surface wind vector products, it is proposed to have a repeat mission of Scatterometer similar to Oceansat-2 but with enhanced features as early as possible to ensure the data availability.

Parameter	Value
Payload Mass	Without structure support: 80 kg With structure support 105 kg
P/L Module Dimensions	1823 (Y) X 1025 (R) X 1276 (P) mm^3
Power	120 W
TM-TC Interface	MIL-STD-1553B
Data Interface	LVDS Serial
Data Rate	Processed Data Mode: Mode 1: 293.36 to 725.68 kbps Mode 2: 367.47 to 8181 kbps Raw Data Mode: Mode 1: 13.58 to 14.03 Mbps Mode 2: 13.28 to 21.1 Mbps
Frequency	13.515625 GHz ± 50 MHz
Polarization	HH (Inner beam) VV (Outer beam)
Pointing angle	Principle Axis: 46^o Inner (HH) Beam: 42.62° Outer (VV) Beam: 49.38°
Swath	Inner: 1400 km (both HH & VV beams available) Outer: 1400 - 1800 km (only VV beams available)
One way 3 dB foot print Azimuth (km) x Elevation (km)	26 x 46 31 x 65
Scan Rate	20.5 rpm
Transmit PRF	186 – 200 Hz (193 Hz Nominal)
Wind Speed Range	3 to 30 m/s (10.8 to 108 km/hr)
Wind Direction Range	0° to 360°
Wind Speed Accuracy	1.8 m/s rms or 10% whichever is higher
Wind Direction Accuracy	200 rms
Wind Vector Cell (grid) Size	25 km x 25 km

The block diagram of Ku-band Pencil beam Scatterometer operating at 13.515 GHz is shown below. The payload is configured with 100% cold redundancy except for mechanical elements like antenna, scan mechanism and rotary joint. It consists of a parabolic dish antenna of 1m diameter which is offset mounted at an angle of 46° with respect to the Yaw axis (earth viewing axis). This antenna is continuously rotated using Scatterometer Scan Mechanism (SSM) at 20.5 rpm with the scan axis along the +ve Yaw axis. Two offset feed horns at the focal plane of the antenna are used to generate Inner beam and Outer beam which will conically scan the ground surface. The Frequency Generator subsystem provides coherent RF and master reference signals needed for the instrument and up converted 13.515 MHz ± 200 kHz frequency signal as input to transmitter subsystem. The chirp modulated signal is amplified using TWTA and is transmitted at a pulse repetition frequency of 193 Hz (nominal). The inner beam operates with HH polarization and outer beam with VV polarization. The earth viewing geometry is shown in fig. 2. The backscattered energy of the transmitted RF pulse from the ocean surface is received back at the antenna and after on-board Doppler compensation and range compression; the digitized data is transmitted to the ground. The normalized radar cross-section referred as Sigma-naught (σ^o) is calculated on ground from this echo data and the Wind vector is derived from (σ^o) using a Geophysical Model Function (GMF). The Receiver subsystem is designed with -109 dBm sensitivity and 109 dB gain. The On-board payload controller receives the tele-commands from OBC over the MIL-STD-1553B bus and configures the instrument for data acquisition. Due to relative motion of the satellite with respect to earth, a Doppler shift of the echo return signal is imparted which needs to be estimated on-board with the knowledge of antenna scan position, spacecraft velocity and beam pointing angle. The Scan Mechanism subsystem provides the angle encoder interface of the rotating antenna system and the antenna position information is used in real time by the Payload Controller for computation of Doppler Frequency (f_{dc}). The data acquisition and compression subsystem (DACS) receives the receiver output analog signals and does single digitizer I-Q demodulation, range signal processing and formatting. The processed and Formatted DACS data is sent to BDH subsystem through LVDS interface. On-board calibration is done once every PRF, which gives a measure of the product of instantaneous transmitter power and receiver gain. Hence, drifts in these parameters during the operational life of the instrument can be known and corrections can be applied on ground in computation of wind vectors.

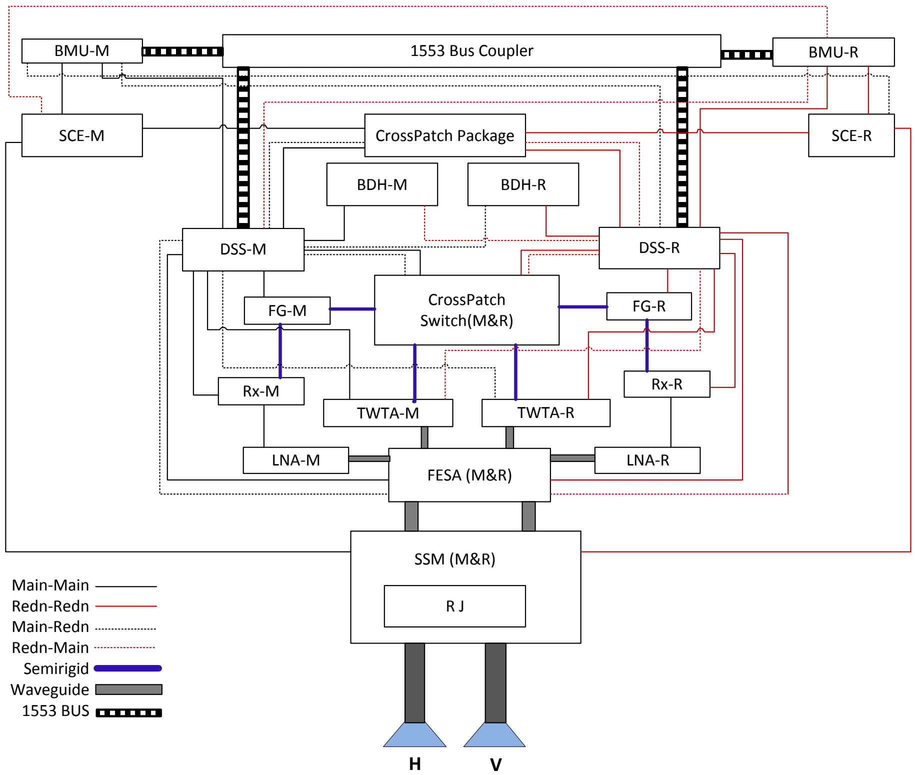


Figure 4-22 Block Diagram of the Scatterometer System

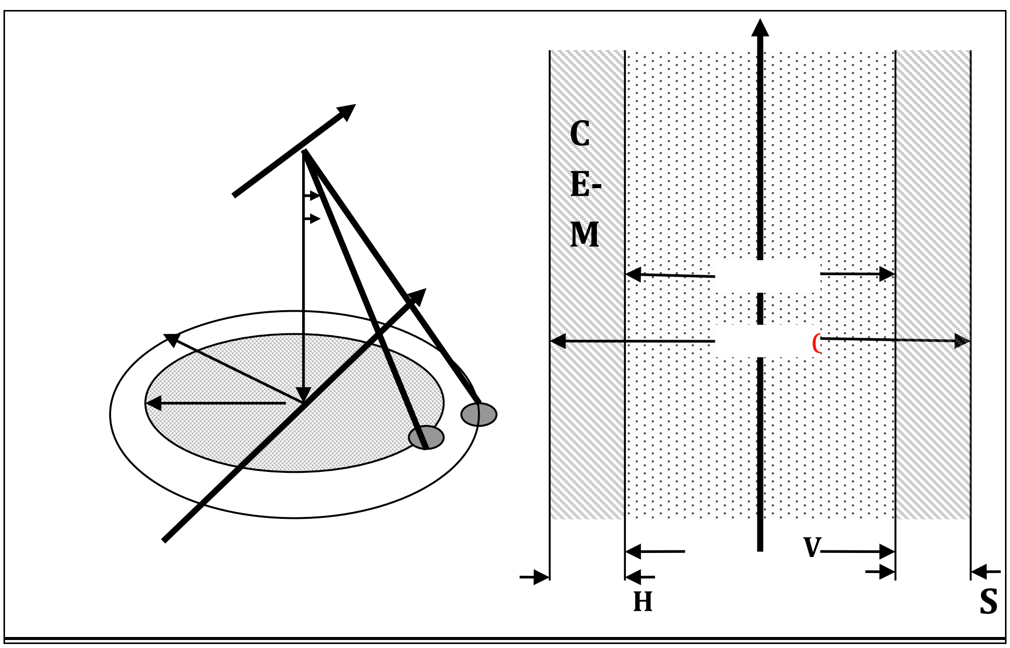


Figure 4-23 Earth Viewing Geometry for Scatterometer

4.5.5.1.1 Scatterometer Scan Mechanism

Scatterometer Scan Mechanism along with Rotary Joint (RJ), bearing unit assembly, lubrication system and SSM Control Electronics (SCE) will be delivered by ISRO Inertial Systems Unit (IISU) and all other payload elements will be delivered by Space Applications Centre (SAC).

Scatterometer Scan Mechanism (SSM) has following functional requirements:

- Provide necessary mechanical interface and rotate reflector with feed which weighs 11 kg at 20.5 rpm about the precision axis of rotation with a scan stability of +0.1%.
- Provide necessary mechanical interface for Rotary Joint, Scatterometer Interface Module (SIM) structure & reflector
- Define reflector and RJ relative orientation and position
- Provide accurate scanning position information for the payload
- Incorporate features to hold down during launch conditions in order to protect the precision ball bearings of mechanism & meeting the stiffness requirement

Rotary Joint (RJ) has the following functional requirements:

- It transfers microwave between rotating and non-rotating parts of SSM Control Electronics (SCE) has the following functional requirement;
- Drive control electronics unit for SSM and rotate SSM at a speed of 20.5 rpm with scan speed stability better than 0.1 %.
- Scan position is provided on interrogation depending on PRF. The angular position of the mechanism should be made available with an accuracy of 0.01 degree with respect to a scan start position on receiving data request from the Pay Load Controller (PLC).
- Provide scan start and angular position data during scan to PLC.
- Electrical interface for power, telemetry and Telecommand signals
- The commutation electronics for SSM drive motor
- Provides scan start data to BMU

5. Cartographic Satellite Series

5.1 Technology Experiment Satellite (TES)

5.1.1 Introduction

The TES (Technology Experimental Satellite) is the first high resolution (<1 m) satellite launched by ISRO. It was launched to demonstrate more than eleven new technologies developed by various design groups across the centres

Critical technologies tested in the TES are given below

- Attitude and orbit control system (AOCS) for step and stare imageries in desired direction.
- Two mirror on-axis optics (RC Type) for payload (providing <1m nadir resolution at 560 km altitude)
- X-band phased array antenna (PAA) with two beam generation capability for payload data transmission
- Single surface tension propellant tank of large capacity RCS tank
- High torque reaction wheels : 0.1 nm and 10 NMS
- Standardized PW, TM, TC system
- Tetrahedral wheel configuration which provides 0.23 NM torque and 23 Nm sec. Angular momentum capacity about each axis.
- Improved satellite positioning system
- Two Advanced solid state recorder with 32 GB each for storage of 6 mins of payload data
- Data security by encryption technique (encryption by stream ciphering scheme inclusion/exclusion option and key changing provision.
- Honeycomb type central cylinder

5.1.2 Mission Objective

The mission objectives of TES are

- To design and develop a technology experimental satellite incorporating a set of critical technologies
- To provide on-orbit demonstration and validation of these technologies for future enhanced capability missions, and also
- To provide hands on experience in complex mission operations like step and stare maneuvers and onboard earth rotation compensation etc.

5.1.3 Orbital Parameters

Parameter	Normal Orbit	Special orbit 1	Special orbit 2
Altitude (km)	560	410	501
Repeat Cycle(Days)	1	2	5

No. of orbits per cycle	15	31	91
Inclination (Deg)	97.65	97.08	97.45
Ground trace Velocity (km/s)	6.97	7.2	7.04
Decay rate (m/day)	61 to 27	410 to 232	125 to 61
Orbital Time (min)	96	92.9	94.73
Local Time (descending) AM	10.30	10.30	10.30

5.1.4 Salient Features of Spacecraft

Subsystem		TES
Structure		Aluminum / Aluminum honey comb elements, Cuboid main frame similar to IRS-P4
Thermal	Thermal control	Passive/ semi active thermal control with paints, MLI blankets, OSR and close loop temperature control
	Thermal Limits	All electronics 0-40deg C, Battery 0-10 deg C, PAN : 20$\pm$3 deg C
Mechanism	Solar panel	Solar panel hold down and deployment mechanism similar to 1A/1B Sun pointing through SADA
Power	Solar panel	Rigid ,Sun tracking, uncanted, 9.636 m^2, 6 panels 1.1 x 1.46 m^2 ,(Each), BSR (SCA) , 800 watts (EOL)
	Batteries	2 batteries, 28-42V, 28 Cells, Ni-Cd, 21 AH
	Electronics	More efficient power electronics developed. Two raw buses (28-42V) supplying power to all subsystems. Modular type of DC-DC converters for payload and data handling
TTC	Telecommand	Conventional systems backed by microprocessor based, time tagged and payload sequencer both for main and redundant.
	Telemetry	ASIC based telemetry system. PCM/PSK/PM modulation
	Transponder	Uplink frequency 2028.70 MHz Downlink frequency 2203.20 MHz
Data Handling		Data rate :2 X 42.4515 Mbps Transmission frequency X-band 8150 and 8350 MHz Modulation : QPSK Recording facility: 2 x 32 GB (SSR)
AOCS	Specification	Pointing accuracy Pitch : +0.15^{o}Roll: +0.15^{o} Yaw: +0.20^{o}

		Drift rate: 3 X 10-4 °/s
	Sensors	Earth sensor (2+1), PYS (1), 4 pi sun sensors(4), Magnetometers(2), IRU
	Actuators	Magneto torquers(2), Reaction wheels(4), 1 N thrusters (8) and 11 N thruster (1)
	AOCE	1750 architecture based microprocessor system for main and redundant

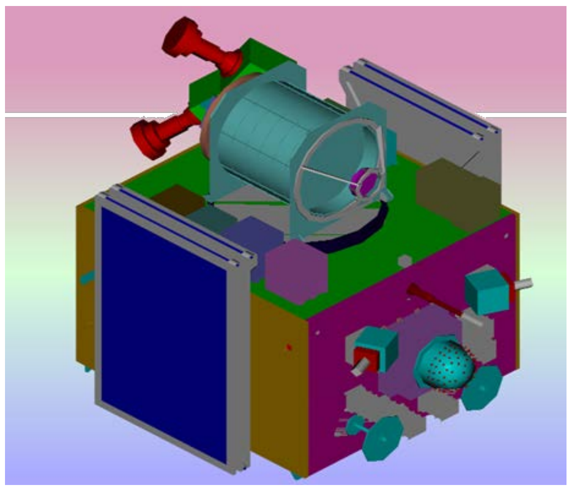


Figure 5-1 Stowed View of TES

5.1.5 TES Payload

The Technology Experiment Satellite (TES) carries one PANchromatic camera called PAN-TES. This camera works on the 'push-broom scanning' concept using linear array Charge Coupled Devices (CCD) as sensors. Four 4K, 7um x 7um are used to cover a swath of about 13.5 km at Nadir. In this mode of operation, each line of the image is electronically scanned and contiguous lines are imaged by the forward motion of the satellite. The improved along track resolution is achieved by step and stare method.

The PAN-TES camera is a high resolution camera with Instantaneous Geometrical Field of View (IGFOV) of better than 3 meters. Totally this camera covers a swath of better than 13.5 km. The satellite is agile and can be rotated to +/- 45 deg w.r.t pitch axis and +/- 26 deg w.r.t roll axis. The focal length of 3920 mm provides an across track IGFOV of better than 3 meter at nadir view from 560 km altitude. The pitch bias and rate enable the camera to provide better than 3 meter along track resolution. The capability of having maximum +/- 26 deg. bias w.r.t roll axis provide 5 days revisit of the same location as well as stereo viewing capability in across the track direction.

5.1.5.1 System Configuration

The PAN-TES camera had three elements. They are

- Electro-optics module (EOM)
- Payload electronics
    - Detector electronics
    - Payload electronics packages
- Payload power supply
    - Payload power converters
    - Payload power regulators

5.1.5.1.1 Electro-Optics Module (EOM)

PAN-TES camera is a single band camera covering the spectral range from 0.5 to 0.85 microns wavelength. The EOM contains

- Imaging Optics
- Detector Head assembly
- Detector electronics

The imaging optics is an Ritchy-chretian (RC) type reflective system with three field correction lens covering a FOV of +0.85 deg. The optical system has an F/no of 7 and effective focal length of 3920 mm. The two mirror system is chosen because of its compactness. The use of hyperboloids for both mirrors allows simultaneous correction of third order spherical aberration and third order coma. The lenses extend the FOV of the telescope by reducing the Field aberrations and give a flat image. The optical design of the telescope features an on-axis concave hyperbolical primary mirror and a convex hyperboloid secondary mirror and three spherical field correcting lens elements(for extending the FOV of telescope) The lenses are housed in a barrel with an appropriate flange and are refereed as lens assembly. Bothe primary and secondary mirrors are coated with enhanced aluminum coating, to avoid the oxidization of aluminum a protective layer of MgF2 coated on the aluminum coating

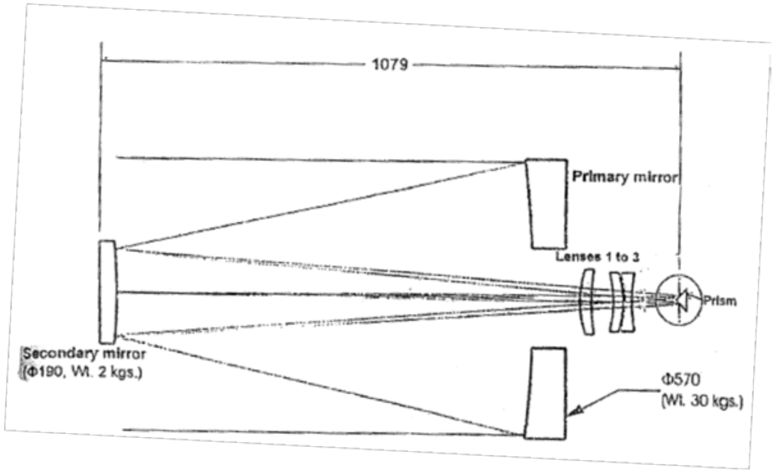


Figure 5-2 Optical Schematic of TES PAN Payload

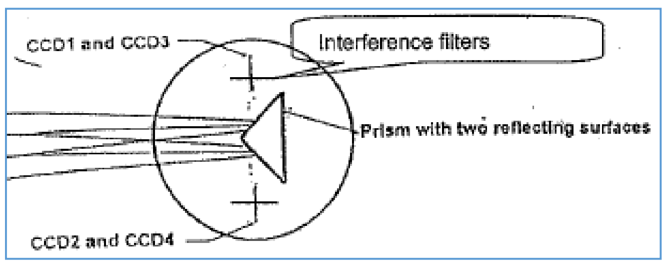


Figure 5-3 Multiple focal plane generation

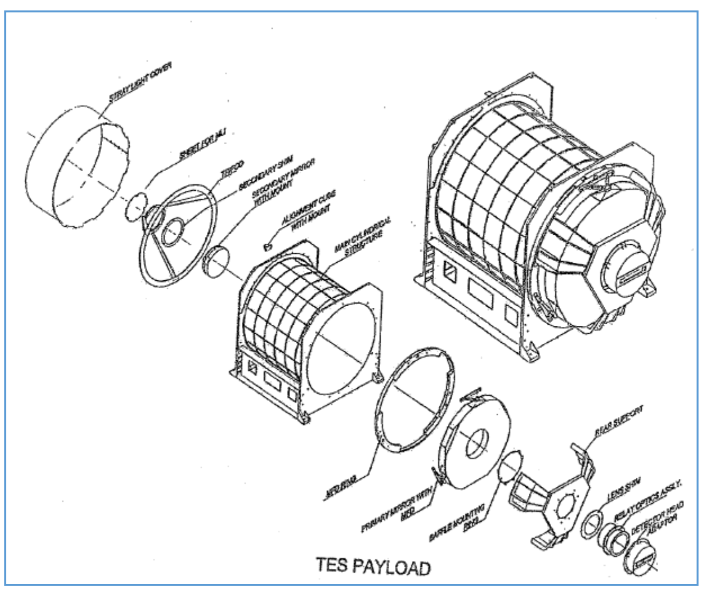


Figure 5-4 Exploded View of TES Payload

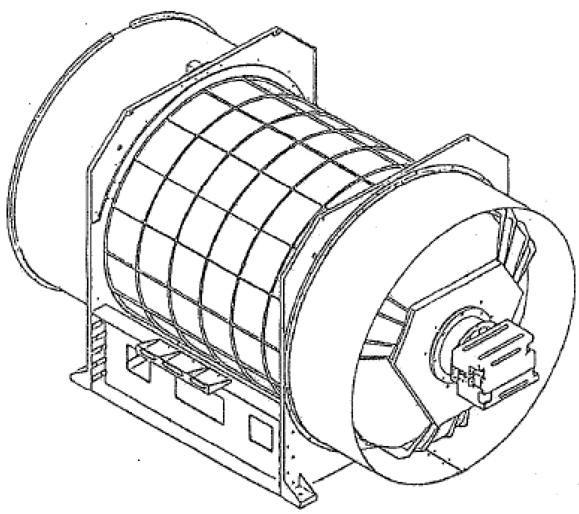


Figure 5-5 TES PAN Camera and Detector Head

5.1.5.1.2 PAN TES Specification

OPTICS	Value
Type	RC Type
Primary mirror	
Material	Zerodur
Diameter (mm)	570 mm (Usable (560 mm)
Center thickness (mm)	65
Weight (kg)	30
Aspect ratio	1: 10
Obscuration	11.5 %
Opening radius at the center(mm)	190
Secondary mirror	
Material	Zerodur
Radius of curvature Ro	905 +/- 2 mm
Conic Constant	-5.057
Surface figure	lambda/10(rms(lambda/67)
Center thickness (mm)	35
Diameter (mm)	190
Weight (kg)	2
Field corrector	
No. Lenses	3
Max. lens diameter	128 mm
Focal length	1310 mm
Housing material	Titanium
Optical system specification	

Effective focal length (EFL)	3920 mm
Spectral Band (micron)	0.5 – 0.85
F-Number	F/7
Field of View	± 0.85 deg.
Optical system length	1068 mm
Diffraction limited MTF	0.42
Design MTF	0.39
Achieved optical system MTF(Optics level)	32
Detector	
Type	Charge Coupled Devices
Detector material	Silicon
Spectral response	0.4 um to 0.85 um
No. of pixels	4096/CCD(TH7833)
Pixel arrangement	Inline
No. Output ports	4/CCD
No. of CCD devices	4

The four numbers of 4k CCD with 7 x 7 micron size pixels were used to cover a swath of greater than 11 kilometers. The rays come out from the secondary are spilt by a isosceles prism and two image planes are created. To mount four devices a specific assembly was designed and detector 1 & 3 are mounted on one side and 2 & 4 were mounted on another side of the detector head. The CCDs are mounted on PCBs which in turn are supported by a carrier plates. Detector 1 & 3 view along nadir whereas detectors 2& 4 are shifted in the image plane in the along track direction. The actual along track distance between these two planes was 22.533 mm each detector had separate interference filter and LED Panels(consisting of four LEDs, two for optical bias and two for calibration mode operation). The earth rotation effects on the swath are taken care by adjusting the location of CCDs. Detector parameters are given below.

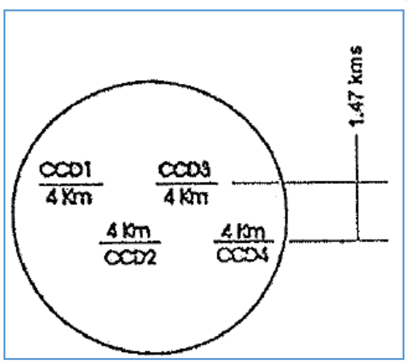


Figure 5-6: illustration of CCD projection on Ground

5.1.5.1.3 Payload Electronics

Payload electronics is similar to IRS-1D payload electronics with four chains. The payload electronics consists of

1. Detector electronics
2. Payload electronics

Detector Electronics (DE): Each DE package consists of four preamplifiers, bias voltage generators, clock drive optical bias LED drivers. Detector driver electronics supplies bias voltages and clocks required for CCDs. The two LEDs required for Optical Bias of CCDs are driven in series with a constant current drive. Designed around LM 723 regulator. The power supply lines to the DE are filtered Using Line filters before being fed to the circuits. The charge collected by the detector pixels are read simultaneously from all four ports and converted to voltages. This signal is amplified by the DE and pre-amplified signals from DE are provided to the Payload electronics (PLE) package.

Payload Electronics: The signals from the DE are amplified in the programmable gain amplifier. The three levels pulse amplitude modulated (PAM) signals of 1.2 MHz is pre-amplified in DE and is further processed in PLE. A constant DC bias is subtracted from total signal to subtract the optical bias in the summing amplifier. There are four Gain settings for the amplifier for each Band which are selectable through ON/OFF commands. The amplified signal is DC restored and digitized. The seven bit parallel data with hot redundancy is available at PLE output on separate buffers for BDH main and BDH redt.

The timing logic receives the Line start pulse (WLS repetition Rate: 0.8836 ms, pulse width 1.48 microseconds) and Bit Rate Clock (BRC) of 28.301 MHz with 50% duty cycle from baseband data handling system and generates the required clock wave forms to read out the data from CCDs The input clocks from BDH main and redundant are cross coupled with logic main and redundant and also the output signals of timing logic are cross coupled and given to BDH.

On-board calibration scheme: In calibration mode the detectors were directly illuminated by the two LEDs which were mounted at an angle of 15 deg. to the optical axis. Calibration mode operations were done during night passes. Provision to operate the individual CCDs or all CCDs together in Cal Mode was provided.

System Spec.

Parameter	Value
IGFOV (m)	< 3meters
Swath km	better than 13 km
Integration time (msec)	0.883
Quantization level	128 (7 bits)
Number of gains	4
Signal to noise ratio	> 128 (at saturation

5.1.5.1.4 Step and Stare method

A new imaging method called Step and Stare method implemented first time in this mission. In this method the ground trace is slowed down by changing the look angle continuously and the along track resolution is improved.

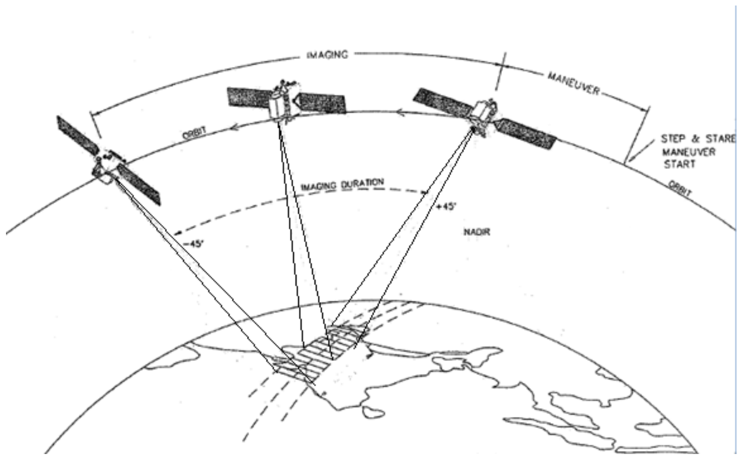


Figure 5-7 Step and Stare method of TES

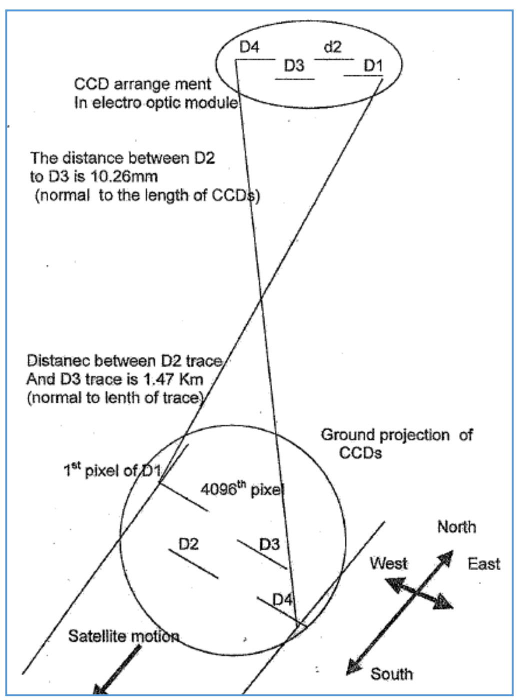


Figure 5-8: Ground Projection of Detector

5.2 IRS-P5 (Cartosat-1)

5.2.1 Introduction

IRS-P5 is a first spacecraft designed to acquire stereoscopic Imageries. The objectives of the IRS-P5 mission are directed at geo-engineering (mapping) applications, calling for high-resolution panchromatic imagery with high pointing accuracies. The spacecraft features two high-resolution panchromatic cameras for in-flight stereo imaging. Hence, IRS-P5 is also referred to as **Cartosat-1.** The data products are intended to be used in DTM (Digital Terrain Model)/DEM (Digital Elevation Model) generation in such applications as cadastral mapping and updating, land use as well as other GIS applications.

5.2.2 Mission Objective

Followig are the mission objectives

- To design and develop an advanced 3-axis body stabilised remote senisng satellite for providing the enhanced spatial resolution (better than 2.5 m) with stereo imaging capability for the cartographic applications.
- To further stimulate new areas of user applications in the areas of cartographic applications; urban management; disaster assesment, relief planning and management; environmental assesment and other GIS applications.

5.2.3 Orbit Details

Following factors were considered for finalizing the orbit.

- A marching orbit
- Early revisit of adjacent path
- A faster revisit to cover the region of interest

Two orbits were selected for envisage two different operation modes called stereoscopic image mode and wide swath mode.

Table 5-1 Orbit Details of IRS-P5

Parameter	Stereoscopic Mode	Wide swath Mode
Altitude (km)	~618	~618
Orbit	Polar sun synchronous Orbit	Polar sun synchronous Orbit
Semi Major Axis (km)	6996.14	6996.14
Inclination (deg)	97.87	97.87
Orbital Period (min)	97.1826	97.1826
Equatorial crossing time	10.30 AM	10.30 AM
Cycle Time	126 days	131 Days
Orbits in cycle	1867	1941
Launch Vehicle	PSLV-C6	

5.2.4 Salient Features of Spacecraft

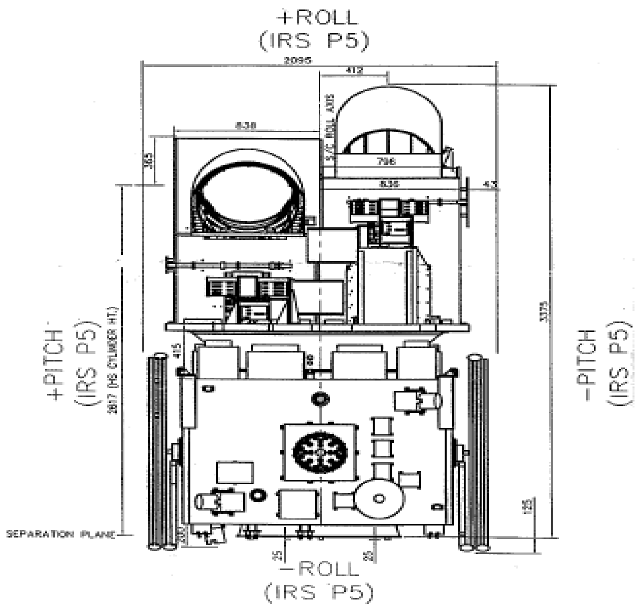
The spacecraft structure is of IRS-P6 heritage, having a size of about 2.4 m x 2.7 m (height). The structure of the spacecraft consists of the MPL (Main Platform) and the PPL (Payload Platform). The MPL consists of main cylinder assembly, four vertical panels, top deck and bottom deck. The cylinder assembly comprises of a central load bearing cylinder, satellite interface ring and top ring. The top ring of the cylinder interfaces with the top deck. All the four panels consist of a CFRP cone, PPL deck, wedges for camera mounting, bracket to mount the payload electronic package near to the Detector Head assembly, and star sensor mounting wedge. The CFRP interface cone isolates the PPL Deck and the MPL. The two cameras are encompassed within a thermal cover assembly with two hoods and anchored to the PPL deck

AOCS (Attitude and Orbit Control Subsystem): The platform is three-axis stabilized (star sensors in loop, magnetic bearing reaction wheels in tetrahedral configuration, 16 nozzles with 1 N thrusters, 4 nozzles with 11 N thrusters). The pointing accuracies are ±0.05º in all axes, attitude knowledge = 0.01º, the stability (attitude drift) is 5 x 10-5 º/s, and the ground location accuracy is < 220 m. The S/C provides a body-pointing capability in the cross-track direction to facilitate a better observation coverage of points of interest, the FOR (Field of Regard) is ±26º. The AOCS employs a MIL-STD 31750 processor.

A power of about 1.1 kW (EOL) is provided. The power subsystem of Cartosat-1 consists of six deployable solar panels, with three panels in each wing (sun side and anti-sun side), each panel of size 1.4 m x 1.8 m. A SADA (Solar Array Drive Assembly) is employed for maximum power tracking. Two NiCd batteries, each of 24 Ah capacity, provide power during the eclipse phases of the orbit. The power bus is formed by ohmic interconnection of solar array strings (current source) and battery (voltage source). There are two raw bus lines called Bus-A and Bus-B. The raw bus is essentially the battery whose voltage ranges from 28 - 42 V. Bus control is by PWM based TCR (Taper Charge Regulator).

RCS (Reaction Control Subsystem): The RCS of Cartosat-1 is a monopropellant hydrazine system using nitrogen as pressurant and operating in a blow-down mode. The RCS is used for correcting the satellite injection errors in attitude and inclination, attitude acquisition and maintenance of the desired sun synchronous orbit. Eight nozzles of 1 N and four 11 N thrusters are mounted on the bottom deck.

The Thermal control subsystem maintains the temperature of different subsystems within the specified limits using semi-active and active thermal control elements like paints, MLI (Multi-Layer Insulation) blankets, optical solar reflectors and auto-temperature controllers All the surfaces of PAN cameras are thermally treated with black paint. All MFD (Mirror Fixing Devices) are provided with black tapes. The payload CCD cold finger is connected to heat pipe by a copper braid. Each CCD has one heat pipe which runs over the thermal cover and gets attached to the sun side radiator plate and anti-sun side radiator plate respectively.



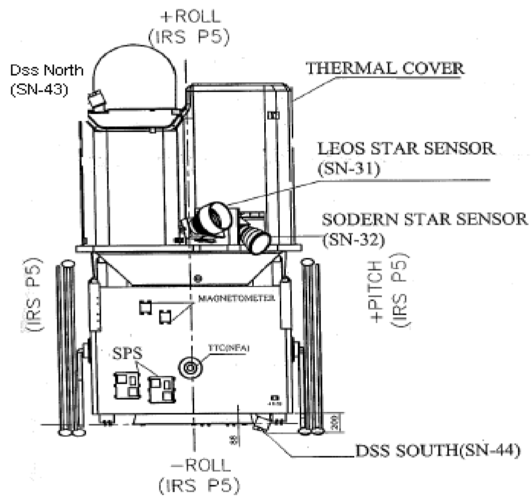


Figure 5-9 IRS-P5 Viewed from EP-01 side **Figure 5-10 IRS-P5 Viewed from EP-03 side**

Table 5-2 Salient features of IRS-P5

Subsystem		IRS-P5
Structure		Cuboid, Aluminum aluminum Honeycomb structure, Payload support structure with CFRP to separate payload platform from main bus
Thermal	Control	Temperature control is with passive techniques using Paints, multilayer blankets, Optical solar Reflector, and active thermal elements like heaters also. Heat pipe radiator panel is used to maintain the temperature of LISS-4 detector head assembly.
	Limits	All electronics packages 0-40degC, Battery 0-10 deg , Payload EO modules : 17 to 23
Mechanism	Solar Panel	Solar panel deployment mechanism and Drive Mechanism
Power	Solar Panel	Rigid, deployable, Sun tracking, CFRP Faceskin, 15.12 M^2 , 6 panels 1.4 x 1.8 m^2 (Each), 58.8 kg, 50 mic. Kaptan insulator, 133 cells in series, 35 in parallel 8 string. 1020 W at EOL, BSR
	Battery	2 batteries, 28 to 42V, 28 Cells, Ni-Cd 24 AH
	Power Electronics	2 buses, PWM TCR, FCL, 8 Strings

TTC	Telemetry	1024 bits, 2245.68 MHz , storage: 6.29 x 10^6 Bits PCM/PSK/PM, 16 Kbps
	Telecommand	PCM/FSK/FM/PM, 2067.897 MHz
Data Handling		The payload data are transmitted in X-band at a data rate of 105 Mbit/s. The BDH (Baseband Data Handling) system consists of two separate chains, one for LISS-3 and AWiFS data, and the second chain for LISS-4 data. The LISS-4 data are transmitted on carrier-1 at 8125 MHz and LISS-3 + AWiFS data are transmitted on carrier-2 at 8300 MHz.
Data Transmission	BDH	X-Band, PCM/QPSK, 2 carriers, data rate: 2 x 52.5 Mbps/carrier Carrier frequency 8125 MHz & 8300 MHz , PAA(64 elements) RHCP JPEG like compression(3.2:1)
	SSR	120 Gbit(EOL),
AOCS	Spec.	Pointing Accuracies: Yaw: $\pm$ 0.05° Roll: $\pm$ 0.05° Pitch: $\pm$ 0.05° (3 sigma) Driftrate : 5 x 10^{-5} deg/sec (3 sigma)
	Sensors	Earth sensor(1), DSS(2), Star Sensors(1 Indian, 1 imported)), 4Pi SS(4), Magnetometer (2), IRU(3 DTG), SPS
	Actuators	Reaction Wheels 5 NMS(4 in tetrahedral), Magnetic Torquers (2) , 1N Thrusters(8) 11 N Thruster(4) Fuel (131 kg) Dry Mass(36 kg)
SADA		SADA with Micro-stepping
AOCE		Hardwired system as a backup only for microprocessor based linear controller.
Payloads		PAN Aft, PAN Fore mass(250 Each)
Mass		1560 kg

5.2.5 IRS-P5 Payload

The payload instrumentation consists of two panchromatic cameras of PAN heritage as flown on the IRS-1C/D satellites. The objective is to obtain fore-aft stereo imagery with two fixed (body-mounted) instruments (i.e., a two-line stereo configuration). They are mounted with a tilt of + 26 deg. (Fore) -5 deg (Aft) from yaw axis in Yaw roll plane. Both cameras are identical in optical electrical and mechanical design. It also has off-nadir capacity up to + 22 deg by providing roll biasing in the orbit ref. frame. The discrimination of elevation differences of better than 5 m make the data particularly suitable for map-making and terrain modeling

- **PAN-F** (Panchromatic Forward-pointing Camera) featuring a fixed forward tilt of 26º.
- **PAN-A** (Panchromatic Aft-pointing Camera), it is fixed at an aft tilt of -5º.

Each camera provides a spectral range of 0.5 - 0.85 µm, a spatial resolution of 2.5 m, a swath width of 30 km, and data quantization of 10 bits. Stereo imagery is acquired with a small time difference (about 50 s) due to the forward and backward look angles of the two cameras. The major change in imaging conditions during this time period is due to rotation of Earth. An algorithm for Earth rotation compensation is being used to eliminate the delayed observations of the two cameras.

Table 5-3 Features of IRS-P5 Payload

Parameter	PAN-F Camera	PAN-A Camera
Spectral range	500 - 850 nm	
Along-track tilt angle with respect to nadir	+26º	-5º
Spatial resolution (cross-track x along-track)	2.5 m x 2.78 m	2.22 m x 2.23 m
Radiometric resolution a) saturation radiance b) data quantization c) SNR	 55 mW/(cm^2 sr µm) 10 bit 345 at saturation radiance	
Swath width (for stereo imagery)	29.42 km	26.24 km
Swath width (for monoscopic observation mode)	55 km (with swath overlaps)	
CCD array (No of arrays x No of elements) Detector element size	1 x 12,288 7 µm x 7 µm	1 x 12,288 7 µm x 7 µm
Optics: Telescope aperture diameter No of mirrors Effective focal length F number FOV (Field of View)	 50 cm 3 1945 mm f/4 ±1.08º	
Integration time	0.336 ms	
Detector	12 K CCD	
Quantization	10 Bits	
SWR @ nyquist frequency	>0.20	
SNR Signal To Noise Ratio	≥256	
MTF (Modulation Transfer Function)	cross-track = 20, along-track = 23	
Onboard calibration	Relative, using LEDs	
Data compression	JPEG algorithm, compression ratio = 3.2:1 (max)	

Data rate	105 Mbit/s (source data rate of 340 Mbit/s)
Nominal B/H ratio for stereo	0.62
Power	110 W (Per Camera)
Mass	< 250 kg (Per Camera)

Payload consists of

- Electro optical module
- Payload Electronics
- Power Electronics

5.2.5.1.1 Electro-optical Module

Each optical module consists of axis three mirror optical system and detector Head assembly consisting of 12K CCD, spectral Band filter and calibration LED

Optical system: The optical system is extended version of the panchromatic camera of IRS-1C/1D. i.e un-obscured off-axis reflective system. The focal length of the system is 1945 mm and the FOV is $\pm$1.3 deg across track and $\pm$ 0.2 deg in along track. The optical system of each PAN camera is designed with a three-mirror off-axis reflective telescope with an off-axis concave hyperboloid primary mirror, convex spherical secondary mirror and an off-axis concave ellipsoidal tertiary mirror - to meet the required resolution and swath width.
The mirrors are made from special Zerodur glass blanks. The mirrors are polished to an accuracy of lambda/80 and are coated with enhanced AlO_2coating. The mirrors are mounted to the electro-optical module using iso-static mounts, so that the distortion on the light weighted mirrors are reduced to a minimum.

Interference spectral filter

Shape: Rectangular

Dimension: 115 x 20 x 6 mm^3

Coated area: 110 x 18 mm^2

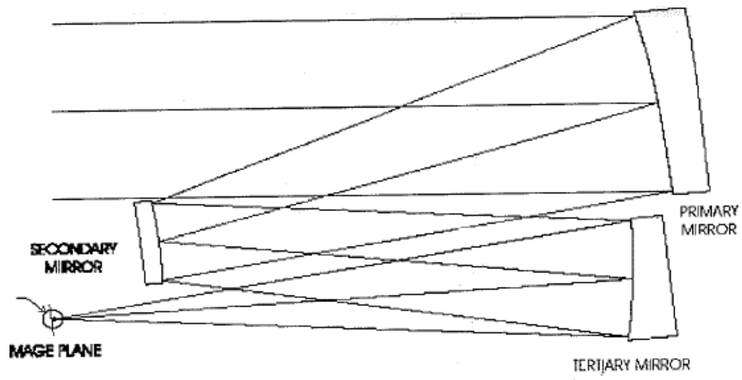


Figure 5-11 Optical schematic of IRS-P5 PAN

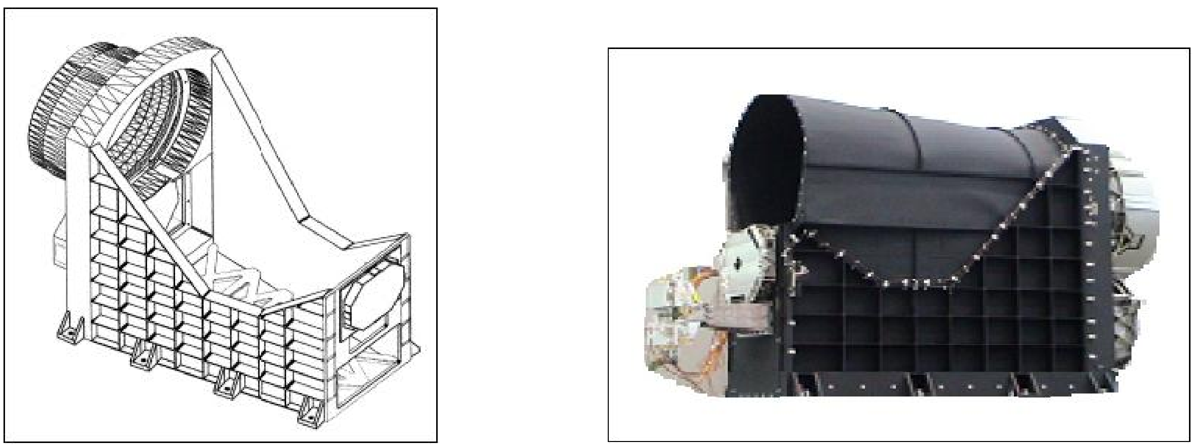


Figure 5-12 Electro optical Module of IRS-P5 Camera

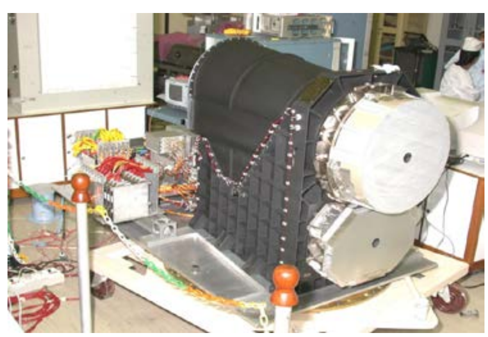


Figure 5-13 PAN camera under testing

5.2.5.1.2 <u>Detector Head Assembly:</u>

Each camera has separate DHA. The Detector is a linear CCD detector array of 12,288 pixels which is mounted in a DHA.

The DHA Consists of DHA Housing, 12K Linear CCD, CCD Holder, 16 LEDs per CCD, LED Holder, Interference Spectral Filter, cold finger, Bias voltage generating circuits, clock driver circuits and Thermal control systems.

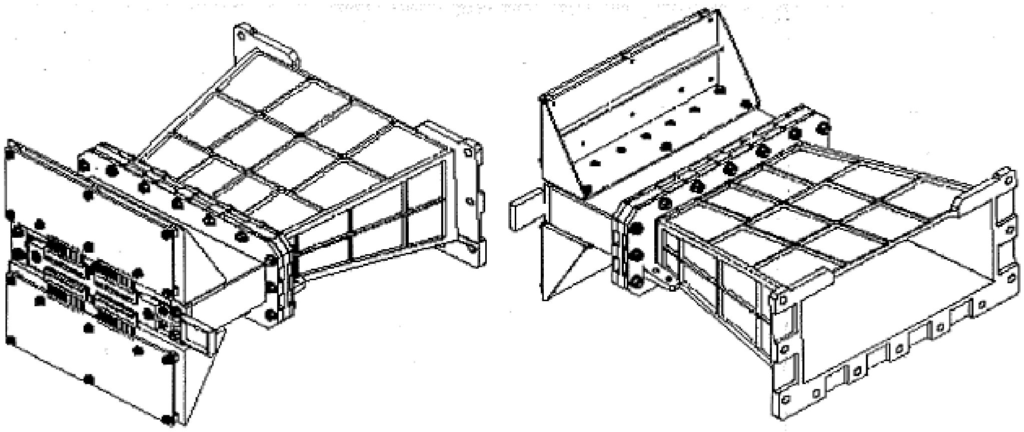


Figure 5-14 Detector Head Assembly

5.2.5.1.3 <u>Detector</u>

Each DHA uses 12 K element linear CCD Thomson make (THX31543A) with a pixel size of 7 micron x 7 micron staggered by 35 microns. Silicon is used as photo sensitive element which is sensitive upto 1.1 micron. The detector provides video data on 8 ports 4 ports for odd pixel and 4 ports for even pixels. Each port provides video data for 1520 pixels including 20 prescan pixels.

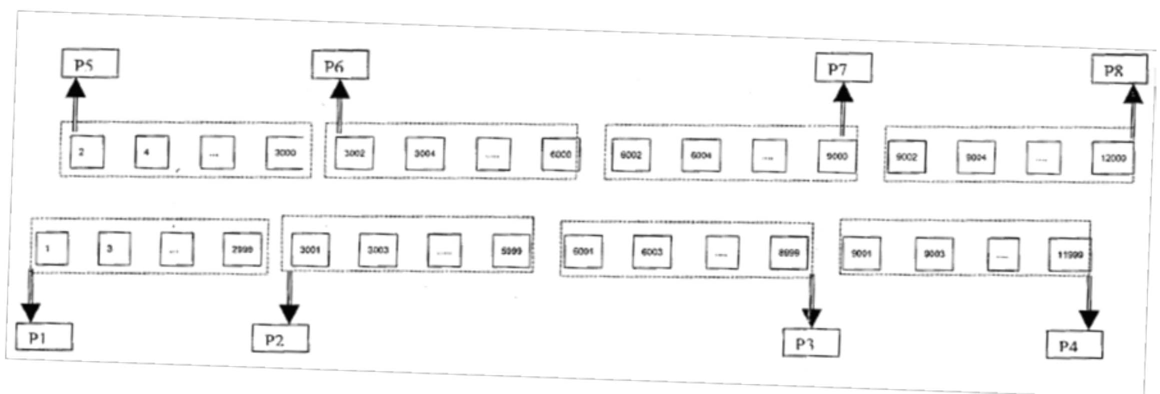


Figure 5-15: Staggered arrangement of pixels in 12 K CCD

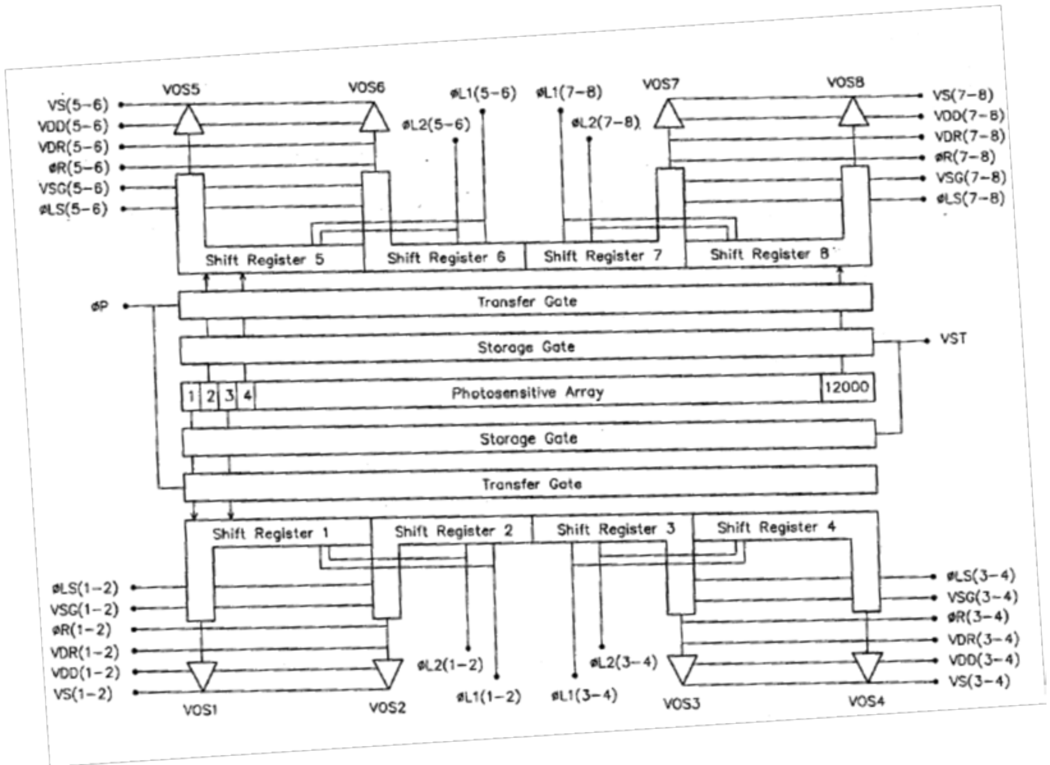


Figure 5-16 12K CCD Architecture

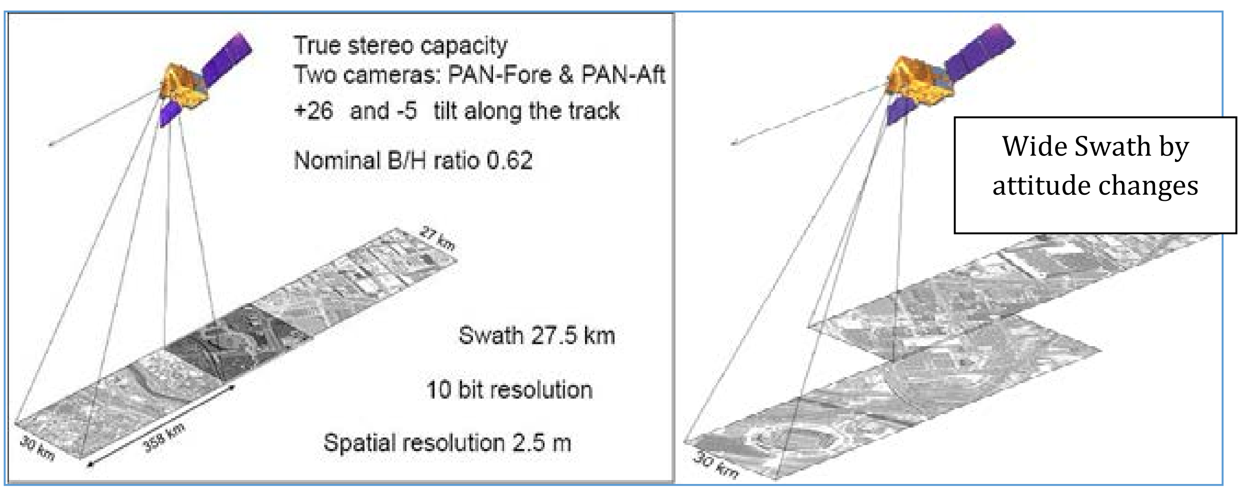


Figure 5-17: Schematic of imaging modes of IRS-P5

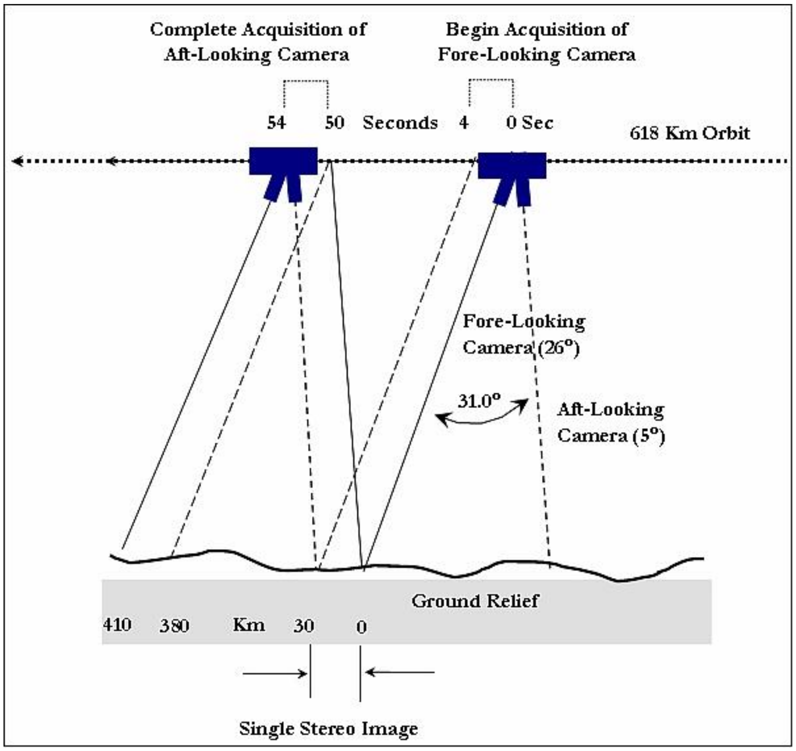


Figure 5-18: Along-track imaging geometry of the CartoSat-1 fore- and aft-viewing cameras

The imagery of the 2-line along-track stereo camera may be used for a variety of applications, among them for the generation of DEMs (Digital Elevation Models). The data is expected to provide enhanced inputs for large scale mapping applications and stimulate newer applications in the urban and rural development.

5.3 Cartosat-2/2A/2B

5.3.1 Introduction

The Cartosat-2 is a high resolution agile satellite with less inertia. This satellite is used to acquire spot images and strip images upto 200 km. The various type of image pattern possible is provide in the figure 13.4

5.3.2 Mission Objective

The main objectives of the Cartosat-2 mission are –

- To design and develop a high agility advanced satellite with a high spatial resolution of around 1.0 m in panchromatic band with an operational life of 5 years and mission reliability of 0.75.

- To meet the ever – increasing user demands for cartographic applications at cadastral level urban and rural management, coastal land use and regulation, utilities mapping and development and various other GIS applications.

5.3.3 Orbit Details

Table 5-4 Orbit details of Cartosat-2/2A/2B

Parameters	Nominal	Recurrent
Altitude (km)	630.6	560
Semi Major Axis (km)	7008.6	6938.1
Eccentricity	9999 E-004	
Inclination (Deg)	97.914	97.91
Argument Of Perigee(Deg)	90	87.19
Local Time	9 .30 A.M	9.30 AM
Revisit (Days)	4	1
Repetivity	310 days	1
Orbits/day	14.	15
Period(min)	97.446	96

5.3.4 Salient Features of Spacecraft

Table 5-5: Salient features of Cartosat-2/2A/2B

Parameter		Cartosat-2/2A/2B
Mass		678 kg (Payload: 119.5 kg)
Structure		Hexagon shaped Aluminum and aluminum honeycomb structure
Thermal	Components	Passive control using tapes , OSR, MLI Blankets and semi-active/active control using proportionate temperature controller and heaters, Detector cooling via heatpipe
	Temp. Range	20$\pm$5 deg.C range for imaging sensors electro-optics 5+5 deg. C for Chemical Batteries 0 to 40 deg.C for electronic packages
Mechanism	DGA Drive	Dual Gimbal antenna hold down and drive mechanism.
	Solar Array Deployment	Solar arrays deployment is done by solar array hold down and deployment mechanism.

Power	Solar Array	Four (2+2), 4.64 m^2 (1.45m x 0.8 m each), Rigid deployable panels, without SADA, Sun pointing. 23 cells(Triple junction) series, 60 in parallel 9 strings. 1200 W @ EOL
	Battery	42V, SAFT, NiCd, 18 AH, 2 Batteries, 28 cells in series
	Electronics	Modularized distribution Package, Mother Board-daughter Board control Package.
Communication	Telemetry	S-Band; PCM/PSK Real Time rate 256 bps and play back rate 4 Kbps;
	Telecommand	S-Band : PCM/PSK/FM/PM, and VHF : PCM/FSK/AM
	Tracking	Facility for ON/OFF and Data commands S-Band tone ranging and two way Doppler X-band beacon
BMU (AOCE + TM/TC)	Telecommand	4 KBPS PSK Modulated with 44 KHz Sub carrier.
	Telemetry	4 KBPS with 32 KHz/128 KHz sub carriers
	Attitude/Orbit sensors	Star sensor(2), 4 PI sun sensors(4), Dynamically Tuned Gyros (DTG)(3), Magnetometers(2), SPS for orbit determination)
	Attitude control	15 NMS, 0.3 Nm RW (4) mounted in tetrahedral configuration, Magnetic torquers(3), Hydrazine thrusters(8 one Newton) 63 kg Fuel
	Orbit Control	Monopropellant hydrazine thrusters
	Orbit-Determination accuracy	<40 meters
	Attitude Determination Accuracy	40 Arc sec along bore sight of SS 10 Arc sec across bore sight
Data Handling		

Payload	Panchromatic	~1 m resolution RC type optics
Mass	Bus Payload	C2:678 kg (Bus+ Payload) Payload:119.5 kg

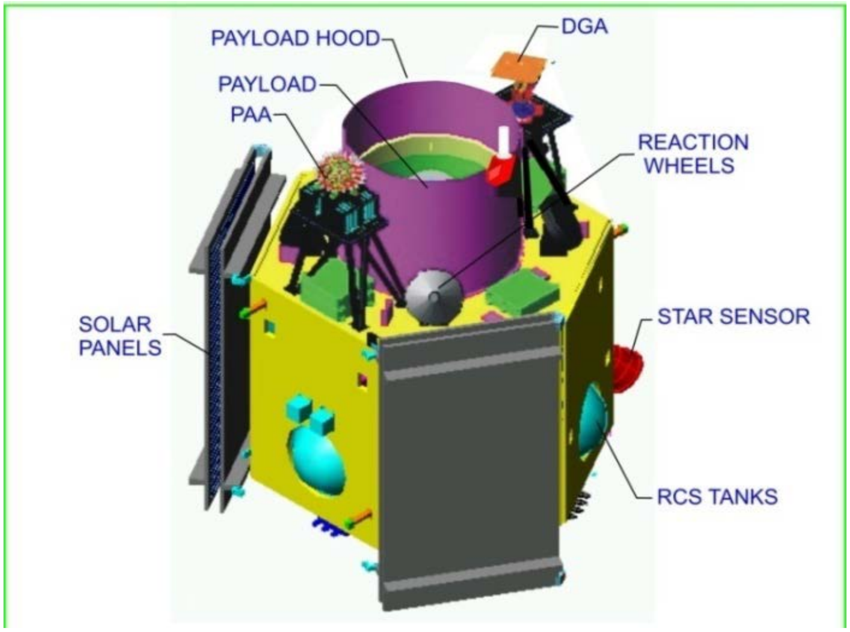


Figure 5-19 Stowed Configuration- Cartosat-2/2A/2B

Figure 5-21 Cartosat-2 Exploded view

5.3.5 Cartosat-2/2A/2B Payload

Optical system is a modified RC system consisting of two-mirror RC type telescope, three lenses, a window and a band pass filter. Field correcting optics consisting three lenses elements is used to correct the aberrations at the larger field of view (+/- 0.5 deg.) and also to flatten the image. A band pass filter placed close to the CCD defines the band shape. The camera operates in the spectral band of 0.5 – 0.8 μm using 12000 elements CCD array. The CCD covers a swath of about 9.6 km.

The two CCDs are located within the focal plane along with band pass filters and calibration system using LEDs. Two independent chains of Camera Electronics are planned to cater to two CCDs.

5.3.5.1 Panchromatic camera specifications:

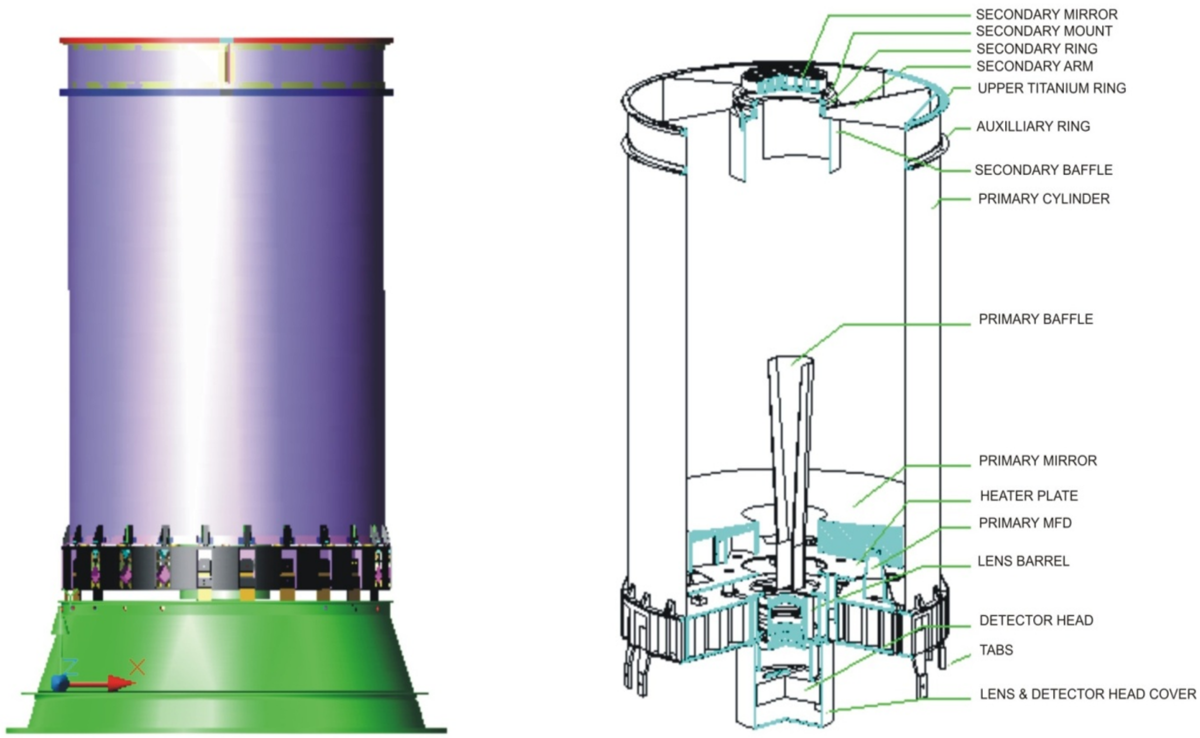
Camera	Value
Resolution	~1 m
Swath	~9.6 km
Spectral Band	0.5 to 0.8 M
Detector	12 K Linear Array CCD
Optics Type	RC type
Optics	F/8, 5.6m Focal Length
Spectral Band	0.5 to 0.85 μm
FOV	± 0.43 Across track) ± 0.2 (along Track)
Size of the Primary Mirror	700mm
Size of the Secondary Mirror	199mm
CCD	
No. Of Pixels /Detector	12000
Pixel Size	7 μm X 7μm
No. Of Output Ports / Detector	8
System	
IGFOV	~1 M (for non- tilt conditions)
Integration time	366μsec
Radiometric Quantization	10 bits
Quantization Levels	1024 (for 10 bits)
SNR (at saturation) 55 Mw/Sr/μM 10 Mw/Sr/μM	 180 80
Camera Size	760mm (dia) X 1600mm (height)
Camera Weight	~ 120 kg
Power	< 60 W

5.3.5.2 Payload Configuration

The payload consists of a Telescope-having an obscured two-mirror system with field correcting optics, two CCDs located within the focal plane along with the band pass filters and

calibration system using LEDs. The payload is a single panchromatic camera (0.5 to 0.8 microns) with a spatial resolution of around 1m and swath of 9.6 km. Two CCD's – One main and one redt. have been provided. The main CCD interfaces with the BDH (M) and RF (M), while the redt. CCD interfaces with BDH (R) and RF (R). The camera is mounted on a highly agile platform capable of being steered across and along the track to provide spot imageries of the desired locations.
The camera system requirements are as follows:

- Provide images with ground projection of better than 1 m in panchromatic band.
- Provide swath of about 10 km.
- Cover 100% albedo for an observation time of around 09.30 AM
- Configuration shall have low moment of Inertia.



PAYLOAD ASSEMBLY

Two independent chains of Camera electronics to cater to two CCDs and are planned to be located close to the detector. The CCDs are mounted in two identical DHAs (Detector Head Assembly) and are configured to have cold redundancy and one of them will be on at a time.

Each DHA consists of:

- 12 K Linear array CCD.
- Bias voltage generating circuits
- Clock driver circuits
- LEDs for onboard calibration

- Heaters and thermistors for thermal control.

5.3.5.2.1 Detector

Detector is a 12 K element linear charge coupled device THX31543A with a pixel size of 7 µ x 7µ staggered by 35 µ. It provides video data on 8 ports; four ports for odd pixels and four ports for even pixels. Each port provides video data for 1520 pixels including 20 pre-scan cells. CCD has anti blooming and integration control and the Integration time selected is 366 µs.

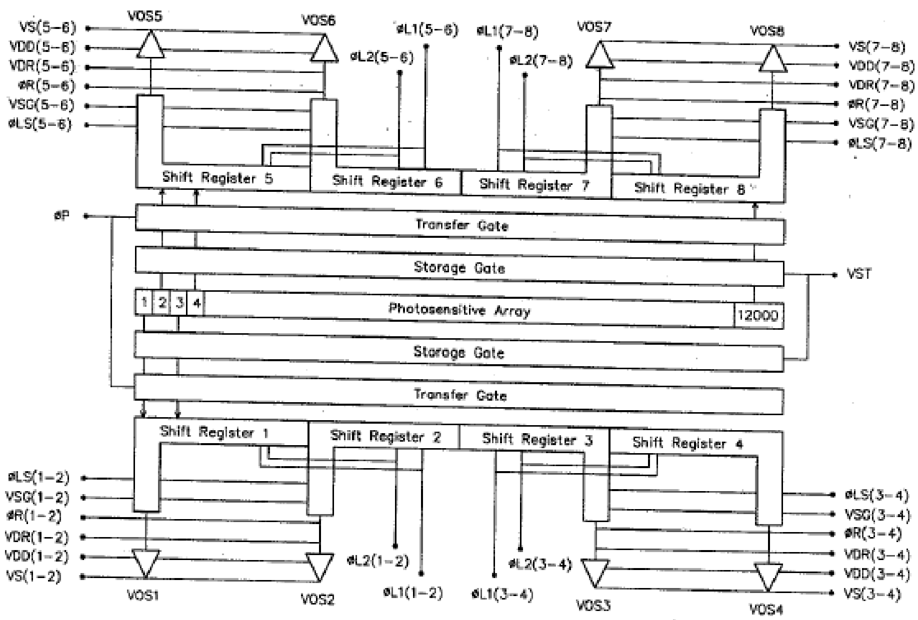


Figure 5-22 12K CCD architecture

5.3.5.2.2 CCD Drive

CCD requires a total of 20 bias voltage lines for its operation. These are generated using the series regulators with an input supply of 18 V.CCD needs a total of 20 clocks for operation. DHA receives clock signals at TTL level from timing and control logic circuits and are conditioned to suitable voltage levels to drive the required loads of CCD.

5.3.5.2.3 Calibration:

There is provision for in-flight calibration. Total eight LEDs are provided in each DHA as two sets of four LEDs in series.

5.3.5.2.4 Heaters and thermistors for CCD temp control

CCD temperature variation is required to be controlled to a narrow range of about 20 $\pm$ 2 deg. C to minimise the impact of photo response variation on radiometer. The heaters are put ON whenever CCD is OFF to minimise temperature variation near the CCD. These heaters are mounted on cold fingers. DHA also requires to maintained at a nominal setting of 20 deg. C and control range of $\pm$ 2 deg. C; for this DHA requires to be cooled. This is achieved by attaching copper braids to the cold finger used for holding the CCD. The copper braid in turn is attached to radiator plate through heat pipes. There are control heaters along with controller to maintain the DHA at 20 $\pm$ 2 deg.

5.3.5.2.5 Camera Electronics:

The camera electronics consists of:

- Port wise video processing chain
- Timing and control logic
- Exposure control logic for imaging and calibration modes
- Clock distribution circuit
- Calibration drivers.

Camera electronics provides necessary clocks for detector operation and constant current for calibration LEDs. It receives video signal from detector, processes and digitises it, extract true video and provide it to BDH subsystem for further processing. Fig: 3.1.6 Shows the Block Schematic of Camera Electronics.

5.3.5.2.6 Video Processor

It receives analog video from DHA, amplifies the signal, restricts the bandwidth, and extracts digitised data corresponding to true video by converting analog data. After processing the data it transmits the same to the Base Band Data Handling System.

It also takes care of saturation and low-level noisy conditions. All the ports are read out simultaneously. Considering the requirements of processing speed (readout rate: 4.2MHz/port), data rate (336 Mbps/detector) and accuracy and other factors the video chain is configured separately for each chain.

5.3.5.2.7 Logic:

Control logic generates various signals required for CCD operation and video processing. The function is realised by three circuits:

a)Timing and control logic: This logic receives Bit Rate Clock and Wider Line Start Pulse (WLS) from BDH subsystem and generates control signals required to read out of signals from the CCD, signal processing activities of video processing and latching of port data and basic clocks for calibration logic circuit.

b) Exposure control Logic: On board calibration is used to monitor degradation, if any, of the detectors. Here the CCD is continuously illuminated with a constant light using 8 LEDs in 2 sets and the integration time is varied to obtain various calibration levels.

ECL generates integration clock with 16 exposures widths in calibration mode a single exposure width in image mode.

In imaging mode, the residual charges, if any, at photo sites are drained into sink for a small duration to result in better radiometric accuracy. The circuit takes line rate and pixel rate/16 control pulses from timing logic and generates integration control pulses. In imaging mode the width of this clock is fixed. When the calibration mode is selected, the mode status is used to initiate the counter and vary the width of integration clock at line rate. The logic generated 16 widths in a cyclic form.

c) Clock distribution: Clock distribution receives all the clock outputs generated in Timing and Exposure logic and distributes them to DHA subsystem and video processor packages according to the requirements. Calibration Driver: Calibration driver provides constant DC current for 8 calibration LEDs located in two groups at Detector Head Assembly. The LEDs continuously glow and provide illumination on detector in calibration mode.

A constant current of 16 $\pm$ 1mA is passed through each set of 4 regulator. The circuit is powered only when calibration mode is selected.

5.3.5.3 Data Handling

Cartosat-2 is the Advanced Remote Sensing spacecraft configured with a high-resolution camera on an agile platform. The imaging modes/profiles are new this mission so as to use the imaging time more efficiently with respect to the coverage. At 630 km sun synchronous orbit, Cartosat-2 carries a high spatial resolution Panchromatic camera with an operational life of 5 years. The payload consists of two 12K element linear CCDs. The video data is quantised to 10 bits. The total data rate per ports of the CCD works out to be 336MBPS. The video data at this rate from the 8 ports of the CCD video processor electronics required to be formatted and transmitted to the ground by DH system through the X-band carrier. The video data is also to be suitably encrypted and additionally stored in a SSR for later playback and transmission. CCD1 data is to be transmitted through the main DH chain and the CCD2 data is to be transmitted through the redundant DH chain.

5.3.5.3.1 Data Interface Package

Data Interface Package receives eight ports of data each at 4.2 Mpixels/sec from Payload Electronics. Each port data bus is 10 bits parallel. The JPEG like compression system requires data in a 8x8 block form. There is also a requirement to bypass the compression and transmit the original data in bypass.

Electrical Specifications

No. of input ports	8 (4I + 4Q)
Port data rate	4.2M words/Sec
Port data format	10 bit parallel
No. Of Pixels/Port	1504 (2 prescan+1500 Valid video+2 post scan)
Integration time	~ 365.71µsec
Output data to DCS	2 ports, LVDS, 10 bit parallel @ 17.5 MBPS

Output data to BDH	2 ports, TTL 10 bit parallel @ 5.25 MBPS
Power	~ 10W

The order in which video pixels of each port are coming out from payload is as shown below:

Odd Ports

Port1: Prescan 19 &20,, Pixel#1, #3,......,#2997, #2999, post scans 1&2

Port2: Prescan 19 &20,Pixel#3001, #3003,......,#5997, #5999 post scans1&2

Port3: Prescan 19 &20,, Pixel#8999, #8997,......,#6003, #6001, post scans1&2

Port4: Prescan 19 &20, Pixel#11999, #11997,......,#9003, #9001, post scans1&2

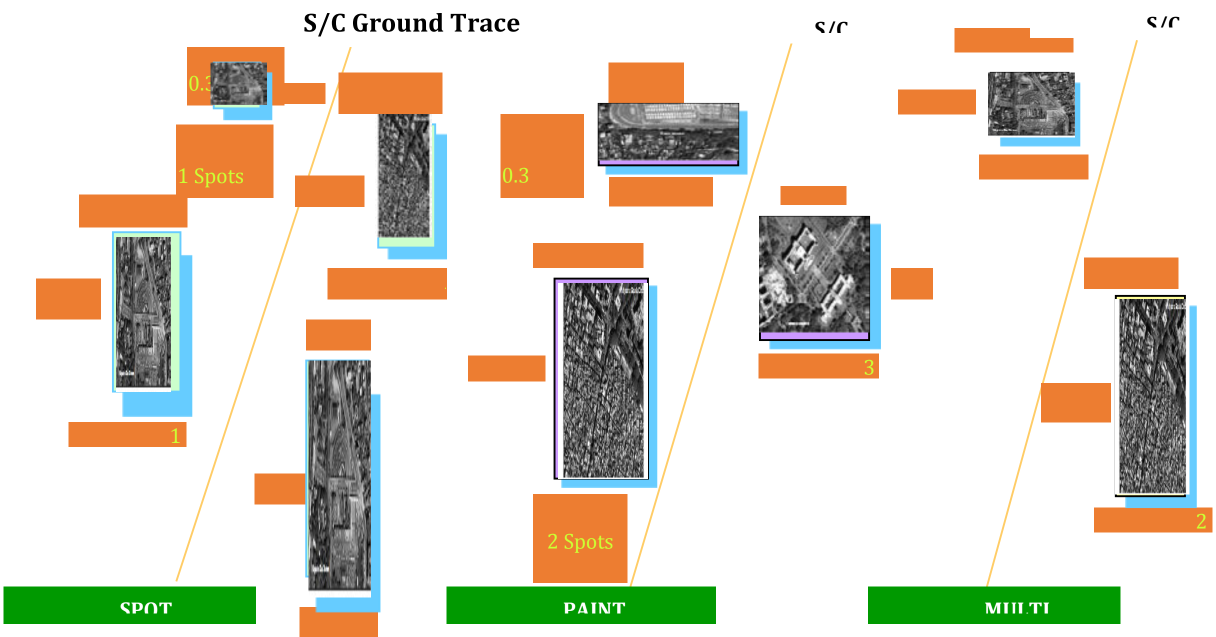
Even Ports

Port5: Prescan 19 &20, Pixel#2, #4,......,#2998, #3000, post scans1&2

Port6: Prescan 19 &20, Pixel#3002, #3004,......,#5998, #6000, post scans1&2

Port7: Prescan 19 &20,Pixel#9000, #8998,......,#6004, #6002, post scans1&2

Port8: Prescan 19 &20,Pixel#12000, #11998,......,#9004, #9002, post scans1&2



Strip Length (km)	No of targets	Total (sq. km)
9.6	10	922
28	8	2150
50	6	2880
290	2	5568

Figure 5-23 Image pattern of Cartosat-2

5.4 Cartosat-2S (2C/2D/2E/2F)

5.4.1 Introduction

Cartosat-2 Series Satellites are high resolution remote sensing satellite configured with panchromatic camera and a 4-band multispectral camera operated in '*Time Delay Integration (TDI)*' mode. It provides scene specific imageries of 0.64m spatial resolution in panchromatic camera and better than 2m in multispectral camera with a swath of 10 km. This three-axis stabilized mission provides spot images of the desired location and has the capability of a long track steering to provide stereo spot imageries and across track steering up to + 26° to enhance the range of spot imageries. Cartosat-2 Satellites also carries two Event Monitoring Cameras Ev1 (0.5m Monochrome) and Ev2 (0.4m RGB) to provide video imagery of pre-selected site with sub meter sampling to build the capability for real time monitoring of ground events. Cartosat-2 Satellites will meet the information requirements of the user community primarily for urban and rural planning, micro-watershed development and geo-engineering applications for an operational life of 5 years.

The Cartosat-2 Satellites are envisaged to provide a highly agile 3- axes stabilized platform. Thus, the satellite provides a 5 day re-visit capability using its agility.

The spacecraft nominal altitude is 505 km. The overall mass of the satellite is around 715 ± 5 kg and the power generation capability is around 986 watts (BOL).

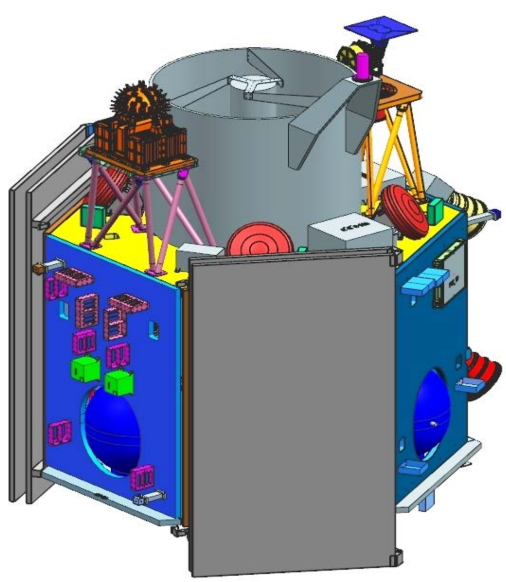


Figure 5-24 Cartosat-2S Remote Sensing Satellite

5.4.2 Mission Objectives

The main objectives of the Cartosat-2S mission are:

- To provide assured continuity of data services for user community.
- To provide high resolution images in Panchromatic and Multispectral bands with an operational life of 5 years.

5.4.3 Orbit Details

Cartosat 2C, 2D, 2E and 2F comparison.

Table 5-6: Orbit details of Cartosat satellites

Parameter	Cartosat 2C	Cartosat 2D	Cartosat 2E	Cartosat 2F
Spacecraft Mass	738 kg	714 kg	711 kg	710 kg
Payload Mass	120 kg	120 kg	120 kg	120 kg
Spacecraft Size (Yx Px R)mm	2435 x 2300 x 1620	2435 x 2300 x 1620	2435 x 2300 x 1620	2435 x 2300 x 1620
Average Power Generated	986 W (BOL)	986 W (BOL)	986 W (BOL)	986 W (BOL)
Average Payload Power	<70 (PAN) <130 (MX)	<70 (PAN) <130 (MX)	<70 (PAN) <130 (MX)	<70 (PAN) <130 (MX)
Altitude	505 km	505 km	505 km	505 km
Eccentricity	9.999 x 10^{-04}	9.999 x 10^{-04}	9.999 x 10^{-04}	9.999 x 10^{-04}
Orbit	LEO	LEO	LEO	LEO
No. of orbits/day	15 17/93	15 17/93	15 17/93	15 17/93
Inclination	97.48°	97.46°	97.44°	97.47°
Local time	9 hrs 30 min	9 hrs 30 min	9 hrs 30 min	9 hrs 30 min
Stabilization	3 Axis	3 Axis	3 Axis	3 Axis
Launch Date	Jun 22, 2016	Feb 15, 2017	Jun 23, 2017	Jan 12, 2018
Launch Site	SHAR	SHAR	SHAR	SHAR
Orbital Period	94.72 min	94.72 min	94.72 min	94.72 min
Swath	10 km	10 km	10 km	10 km

5.4.4 Salient features of Cartosat Systems

The overall mission elements of Cartosat-2S mission are essentially the following:

- The payload system with Panchromatic and Multispectral cameras that can simultaneously image the area of interest with TDI mode of imaging.
  - The payload consists of light weighted primary and secondary mirrors and CFRP housing.
  - PAN and MX TDI detectors with optical butting and spectral filters.

  - The video processing electronics and associated powering circuits.
- The camera is operated in continuous mode with enhanced imaging efficiency.
- The spacecraft can be biased along and across the track up to ±26° respectively to enhance range of spot imageries and to image the area of interest.
- CCD is mounted along pitch axis to maximize power generation during payload operation.
- A suitable low orbit has been chosen and maintained to meet the various requirements of the payload, spacecraft bus and data users.
- Appropriate maneuvers during an orbit will be carried out to optimize between solar array sun pointing for adequate power generation and payload axis pointing towards earth to provide effective thermal control for payload and to point at regions as required for imaging.
- On-Board Computer performs various functions such as sensor processing, command & house-keeping, Attitude & orbit control, Thermal management and payload sequencer execution.
- High accuracy star sensors (3nos.) and gyros for attitude reference
- Magnetic torquers, Reaction wheels and RCS Thrusters as actuators
- Power system with
  - AZUR and ZTJ solar cells on high stiffness CFRP substrates as power generation
  - Li-Ion battery for energy storage
- RF system comprising the S-Band system for TTC functions and SSPA based X-Band system for payload data transmission
- A 12 channel GPS SPS (M &R) is in house make to provide spacecraft position data for attitude determination to support data products generation system.
- 12 Monopropellant RCS system for orbit correction and momentum dumping and for initial acquisition of spacecraft
- The Data Handling system performs DWT based compression, formatting, encryption and encoding before storing the data in SSR or Data Transmission.
- Onboard Solid State Recorder for recording the P/L data, which will be played back subsequently.
- A ground TT&C network capable of operating and controlling spacecraft as per guidelines of spacecraft control centre which will house the required hardware and software elements
- Ground system for payload data reception, data processing, data product generation, data dissemination and archival

Table 5-7 Features of Cartosat Systems

Subsystem	Specification
Mass	715 ± 5 kg
Payloads	PAN, MX & 2 EvM Cameras
Structure	Hexagonal Prism Shaped

Thermal Control	Optical Solar Reflectors (OSR), Multi-layer Insulation blankets (MLI), Quartz wool blanket, isolation spacers, and thermal interface materials.
Mechanisms	Solar Panel – Accordion Type of Deployment (CCL, Hinges & Spring Actuators) DGA – 2 orthogonal drive modules
AOCS	
Pointing Accuracy	Rate (3 σ) over 2 Hz Band : 2.0e-04°/s DC Rate : ≤ 2.0e-05°/s (one tenth of 3 σ rate)
Sensors	3 Heads Mk2 Star Sensors Two 4-PI Sun Sensors Mini 2 Two Magnetometer IRU with 1553 Interface
Actuators	Eight canted 1N thrusters for attitude corrections as well as OM operation Four un-canted 1N thrusters for OM Operation Four Reaction Wheels (0.3 Nm Torque and 15.0 Nms @3300rpm) mounted in tetrahedral configuration about +Yaw axis, Tacho & 16Bit TCS Interface Three Magnetic Torquers of 20.0 A-m2 Capacity
Power System	
Solar Panels	2*2 | Azur & ZTJ cells | 900W (EOL)
Battery	36*2 Ah Li-ion | 10Sx24P
Electronics	Bus Voltage – 30-42 V Peak Load Current - <12A@37V
TT&C	
Telemetry	S-Band |4 kbps (RT), 16 kbps (PB) | PCM (NRZ(S))
Telecommand	S-Band |4 kbps | PCM/PSK
RF System	
Payload Data System	X-Band | 320 Mbps | QPSK
SPS	12 Channel | L1 (1575.42 MHz) | C/A Code
SSR	**200Gb**

5.4.5 Cartosat-2S Payload

Cartosat-2 Satellites uses similar imaging optics and mainframe of Cartosat-2 series with enhancement in focal plane to cater to the mission objectives for improving spatial resolution in panchromatic band and incorporating multi-spectral capability for host of civilian applications. Also, it shall provide continuous imaging using Time Delay and Integration (TDI) concept which facilitates faster image acquisition by avoiding staring time. The camera is mounted on a highly agile satellite platform capable of being steered across and along the track to provide spot imageries of the desired locations. In order to achieve 0.64 meter resolution with Cartosat-2A/2B optics, the nominal satellite altitude required is 505 km. The high resolution PAN and multi-spectral camera along with the camera electronics and Two experimental "event monitor" (EvM) cameras have been accommodated within the available telescope field of view. The PAN and MX camera specifications are given in table below. Major features of EvMs are given in Table 5-8 & Table 5-9.

Table 5-8 PAN and MX Camera specifications of Cartosat-2 Satellites

Parameter	Cartosat-2 Satellites	
	PAN	MX
Ground projection at nadir (m)	0.64	<2
Swath (km)	10	10
Spectral bandwidth (μm)	0.45-0.9	B1: 0.45-0.52B2: 0.52-0.59 B3: 0.62-0.68B4: 0.77-0.86
SWR (%) @ Nyquist freq.	>10	>20
Saturation Radiance (mW/cm2-str-μm)	55	B1: 53B2: 53 B3: 47B4: 31.5
Quantization (bits)	11	11
SNR	180 (TDI=12 stages) at 100% Albedo 180 (TDI=80 stages) at 15% albedo 50 (TDI=12 stages) at 15% Albedo	Band-1/2/3/4: 300 (TDI=15) at 100% albedo Band-1/2/3/4: >60 (TDI=15) at 15% albedo Band-1/2/3/4: > 120 (TDI=45) at 15% albedo
Size of EO module envelope (mm) + Roll Pitch Yaw	775, 775, 1415	
Weight of EO module	120 kg	
Unregulated power (W)	<70	<130
Data rate (Gbps)	2.4	1.5
Equatorial crossing time	09:30 AM (descending node)	

Table 5-9 Major Features of Event Monitor Cameras

PARAMETER	EvM-1		EvM-2
	MODE-1	MODE-2	
Number of pixels in a frame	Across-track: 2048 Along-track: 2048	Across-track: 2048 Along-track: 82	1600 * 1200

Bands	Monochrome		3 (RGB)
GSD (m) from 500 km altitude	0.5		0.4 (2*2 pixels of 0.2m)
Swath (m2) from 500 km altitude	1006 * 1006	1006 * 40	314 * 236
Frame rate (fps)	28	375	15
Data rate (Mbps)	1280		~230Mbps
Unregulated power (W)	5.8		2.4

Payload consists of telescope, focal plane assembly and camera electronics housed in mechanical systems as shown in Figure 5-25 & Figure 5-26.

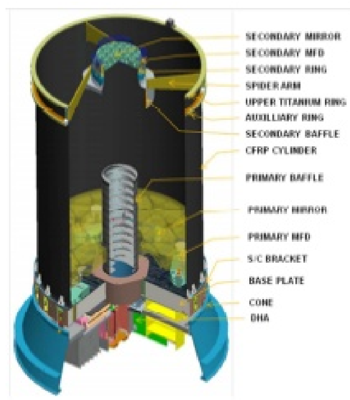


Figure 5-25 Isometric View of Electro Optic Module

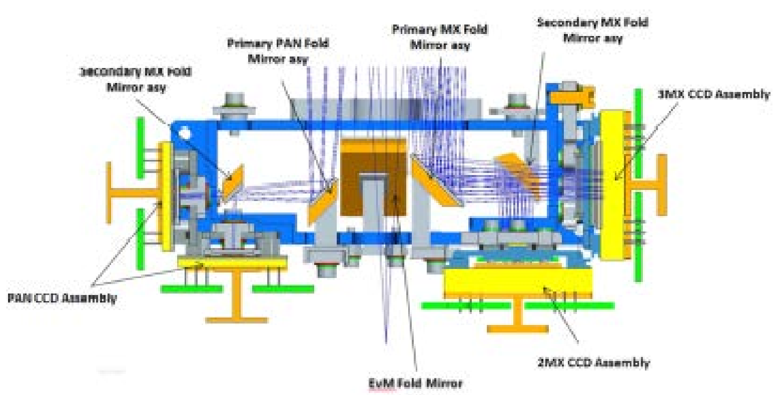


Figure 5-26 Fold mirror assembly

5.4.5.1 Optical System (Telescope)

It is same as Cartosat-2A/2B and is a modified RC system with two-mirror configuration and field correcting optics. It comprises of primary mirror (PM) (concave hyperboloid mirror of aperture 700mm and radius of curvature of 2585mm), secondary mirror (SM) (convex hyperboloid mirror of aperture 199mm and radius of curvature of 850.4mm) and field correcting optics (FCO). Field correcting optics (FCO) has been modified to increase the field of view from to ±0.5° to ± 0.6° to retain same swath at reduced spacecraft altitude. The focal length (5600mm) and the F-number (f/8) of the telescope remain same as in the earlier Cartosat-2 series. Focal plane uses split field configuration based on field separation by using multiple fold mirrors so as to accommodate PAN, MX and EvM detectors. The optical system caters to spectral range of 450nm to 900nm. The band characteristics of multi-spectral bands will be similar to the IRS series of payloads. In Mx chain spectral selection is carried out using four band strip filter commensurate with detector architecture mounted directly on the detector package. Optical schematic of the telescope is shown in Figure 5-27.

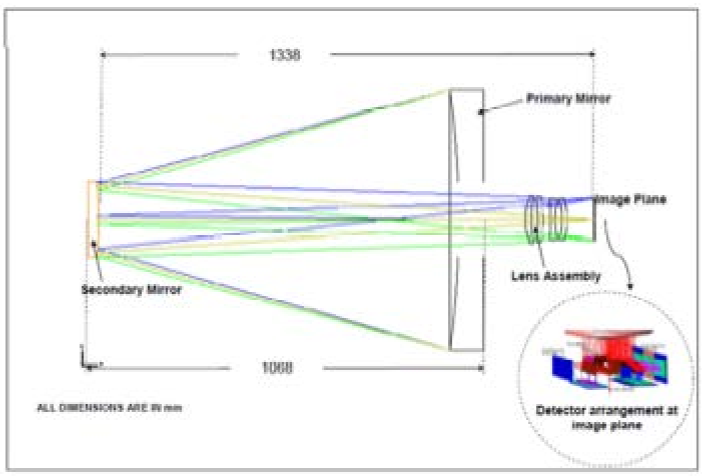


Figure 5-27 Optical Schematic of Telescope

Physical size of the detectors is almost twice the active size and therefore when placed adjacent to each other, results in discontinuous swath and also exceeds the field of view of ±0.6°. This is overcome using optical butting technology. The optical butting concept is shown in Figure 5-28.

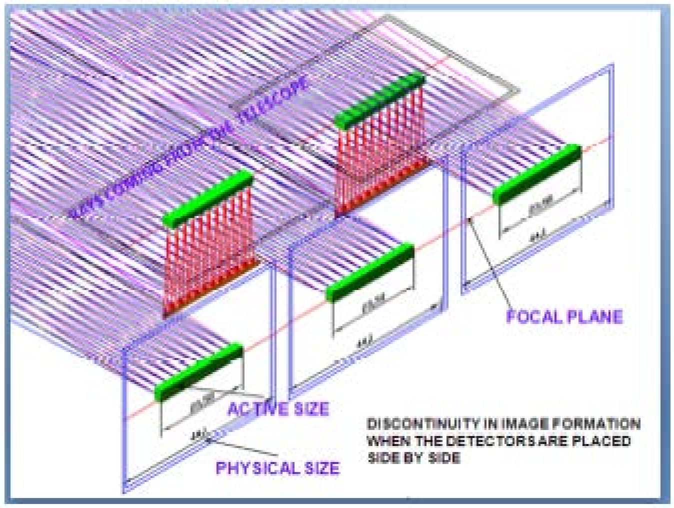


Figure 5-28 Optical Butting Configurations

5.4.5.2 The Detection Systems

It is located in the focal plane. The detection system is designed with TDI detectors which facilitates continuous imaging capability and high SNR for low-illumination scenes. PAN channel uses two 8k TDI CCDs and MX has five 1.3k TDI CCDs to achieve full swath.

Realization of 0.64m GSD in PAN with 10 km swath is with 7μm pixel size and >15000 pixels. This is achieved by use of 8K TDI detector. The 8K TDI array has 80 stages with 4, 8, 12, 16, 24, 32, 48, 64, 80 stage selections and lateral anti-blooming feature.

Realization of better than 2m GSD for MX is achieved by use of 1.3K Quad- TDI detector having pixel size of 17.6μm and four TDI arrays within single package, which can cater to four different spectral bands, is selected. It provides 1.57m ground projection.

Onboard calibration is planned to be carried out using stellar imaging, lunar imaging and ground sites similar to that of Cartosat-2 series.

5.4.5.3 Payload electronics (PE)

It consists of high speed miniaturized front-end electronics for detector drive and video processing; high speed timing and control logic electronics and low noise regulated payload power supply. Modular configuration is adopted for payload electronics. Noise performance of payload electronics is aimed to achieve high dynamic range at system level. 11-bits quantization is implemented. PE for Cartosat-2 Satellites is a similar to Cartosat-2 Series, based on TDI detectors with miniaturization and Low power dissipation using state of the art components like AFEs, FPGA, LVDS, SerDes, SMD resistors/capacitors and micro/nano connectors to meet size, weight and power goals.

Main functions of payload electronics are:

- Provide required detector drive signals (Bias and clocks)
- Perform analog processing and digitization of detector output
- Timing and control signal generation
- Power supply for all payload electronics systems
- Interface to data handling, Tele-command, Telemetry and Raw bus

All these above functionalities are realized through three building blocks namely,

- Front End Camera Electronics (FCE)
- Digital Control & Processing Electronics (DCPE)
- Payload power supply (PPS)

FCE mainly comprises of all the analog mixed electronics system near the focal plane assembly of the telescope along with detector.

DCPE is a complete digital system generating all logic and control signals for detector & FCE. **PPS** will have DC/DC converters, regulators and distribution circuitry for providing low noise multi voltage level supply lines for FCE and DCPE. DCPE and PPS will have interfaces to BDH, OBC and Spacecraft Raw bus.

PE is custom designed and independent both for PAN and MX payloads. Modular approach is followed for complete PE chain with minimum cross coupling and with available real estate and low power.

Each MX detector has four arrays. Out of four arrays, two arrays of a detector are handled by single electronics module. Hence total 10 modules are required for 20 arrays of five TDI detectors. Each chain is powered individually and there will be total 10 PPS Units (PPSU) required to power MX camera. Each PPS delivers approximately 7W power. Two PAN detectors are handled by two independent electronics modules comprising of FCE, DCPE and PPS unit. PE block diagram is shown in Figure 5-29.

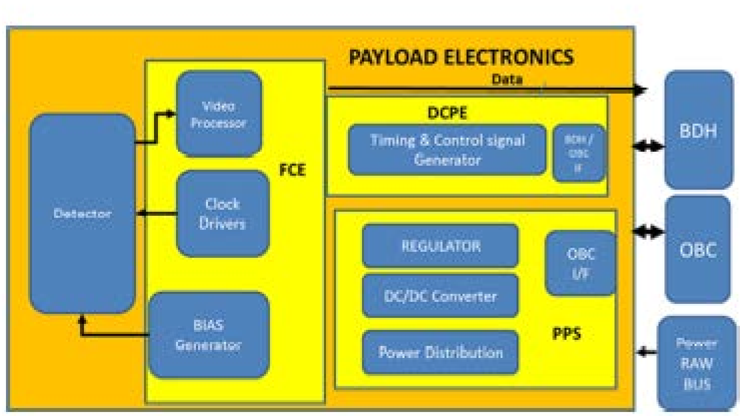


Figure 5-29 Block diagram of Camera Electronics Configuration (per module)

5.4.6 Comparison of Cartosat-2 Series Satellites

Table 5-10 Changes in Carto 2D w.r.t Carto 2C

S.No.	Cartosat 2C	Cartosat 2D
Integration		
1	Mk I SS – 2 no's & Mk II SS – 1 no	All are Mk II SS. HX1750 based CPU Card has been used to provide better timing margins.
2	X-Band Transmitter main mounted top deck outside, converter mounted on RP-05 panel.	X-Band Transmitter main & redundant, converter mounted top deck outside.
3	EP-04 panel is split as 2 panels.	EP-04 panel is single as continuous.
4	Latch valves are MPLV type	Latch valves are SFLV & MPLV type (Each 2 no's)
5	Magnetometer	Mini Magnetometer. ASPC Card Modification to accommodate Mini Magnetometer.
Reaction Control System		
6	Tayco make heaters – 4 no's	Indigenous thruster bed for 2 nos. of non-canted thrusters Y1, Y2.
OBC		
7	**Modifications in EID** o To accommodate additional heaters(Thermal Requirement) ▪ 8 heaters added(RW, BDH, SSR related) ▪ Payload heater(Cold Finger 4(M)) shifted to OBC-10) o TM & TC requirements of Mark-II star sensors	

	▪ Two Nos of PII pulse commands reduced for SS1 & SS2 o TM & TC requirements of Mini-Mag ▪ TM Requirements of Mini-Mag • 4 Nos of Analog Monitoring for +/-10V Regulator Monitoring ▪ TC Requirements of Mini Mag • 4 PII Pulse commands for Mini-Mag supply On/Off
8	OBC Software modifications based on Cartosat-2C observations & Mission/ISTRAC Requirements.
Mechanism	
9	• DGA mechanism for CARTOSAT 2D and 2E will have only one Resolver per Drive Module. • Redundancy with in DGA removed as PAA is main Antenna and DGA is Redundant.
Thermal	
11	• EP03/EP05 panel – PW-12/22 (Diode) packages are removed and diodes are shifted to solar panels rear side. • EP04 split panels are made into single panel to improve the structure stiffness • RP03 – Mark 1 star sensor replaced with Mark-2 star sensor • RP04 – Mark 1 star sensor replaced with Mark-2 star sensor • Top deck layout – XBS-11/12 package is stacked on top deck outside; XTC shifted fromRP05 to top deck adjacent to XBS package. PEC-10/20 relocated from top deck outside to top deck inside below XBS. • PAA MMICs – Eight 20 dbm and eight 26 dbm MMICs were used in Cartosat-2C. Presently, six 20 dbm and ten 24 dbm MMIC are used. • SSPA – Change in the make of 2W SSPA from GAETEC to Astra. • SPS – One imported and one Indigenous. Change in the SPC configuration • Payload metering cylinder is indigenous. • Additional Heaters For –REACTION WHEELS, BDH & SSR
SSR	
12	• Cross coupling in the playback path has been removed and the modules are now shifted to the data controller. • Data transitions have been increased in the data to remove data redundancy. • Addition of processor write signal to logic.
13	SSR software has been changed in 2D w.r.t 2C in terms of bug fixes and mission requirements.

Table 5-11 Changes in Cartosat-2E w.r.t Cartosat-2D

S.No.	Description
RCS	
1	Indigenous thruster bed heaters in place of Tayco make heaters for all 4 nos. of non-canted thrusters Y1, Y2, Y3 & Y4.
SPS	
2	• SPS Main SPS (R) are in- house SPS. • DC-DC converters SPC-13 (or SPC-41) & SPC-23 are stacked. Mounting provision to be explored on EP-03 outside near to SPS- 10/20. • SPS computes PVT based on 12-channels measurement. Existing SPS-PB format (184 bytes per frame) provides 10-channel data in HK & PB. • SPS-PB data frame size to be increased from 184 bytes to 210 bytes. It will provide the full data of 12-channel receiver. Channels 11 & 12 data also to be used for OD. • PB data acquisition at checkout and ISTRAC to be updated for increased frame length. • Ground processing software of checkout & FDG to be modified to accommodate the SPS-PB format change. • SPS-PB download time will increase from 134 seconds to 153 seconds for 1454 frames. • TM & Aux data will be available for 10-chaneels hence no change in on-board OBC software. Remaining two channel data will be available in MIL 1553-B output. • SPS PB storage logic will be modified in SPS RCE software based on CARTO-2D observation. PB storage will continue from current storage location in case of S/W reset occurs. Stored data will not be erased. • Storage will continue without giving storage on command. • On-board position loss observation will be taken care by modified frame sync software module. • RF front end housings SPS-11 & SPS-21 are identical to CARTOSAT- 2D. • Since in-house receiver needs one RF input, Power divider not required. • Filter & PD cards need not to be mounted in top housing of SPS-11 & SPS-21.

Table 5-12 Changes in Cartosat-2F w.r.t Cartosat-2E

S.No.	Description
RCS	
1	• Indigenous thruster bed heaters in place of Tayco make heaters for one block of thrusters R1, R2, R3, R4 & Y1, Y2. • All four latch valves are of SFLV type. • Use of indigenous Transition tube.
Battery	
2	• LG make battery (2 Nos) with a capacity of 50.4 Ah in place of 36 Ah ABSL make battery. Allowed DOD changed to 15%.
X-Band System	
3	• Configured with in-house X-Band Transmitter (M&R). Heritage from C2C.
SPS	
4	• SPS-10, L1 frequency GPS receiver identical to CARTOSAT-2E. • SPS -210 is GPS + NAVIC BASED receiver with messaging facility. • SPS-211 in place of SPS-21. SPS-211 has L1 & L5 dual band antenna. • SPS-20 EID modified as per SPS-210 • SPS-21 housing modified to accommodate L1 + L5 antenna. • SPS-210 PB data increased from 1454 frames to 1600 frames. • SPS-PB frame size increased from 212 bytes per frame to 360 bytes per frame. • SPS-PB data acquisition software updated and checkout and ground station for increased frame length. • SPS Filter package accommodated on EP-04 panel.
Thermal	
5	• Additional Temperature sensors and Heaters are provided for NavIC based SPS.
Software Changes	
6	• 1553 Interface with SPS to receive NAVIC based command. • Additional words in SA#7 (32 instead of 15 Only SPS2) • Command Encryption with Onboard Decryption Software for NAVIC based commanding • Command Validation and decoding • Telemetry database updation for NAVIC based parameters • ATC database modification for additional heater requirement for SPS

6. Microwave Remote Sensing Satellite Series

6.1 RISAT-1

6.1.1 Introduction

RISAT is the first microwave satellite designed and fabricated by ISRO. This mission will facilitate data collection in day/night and in all weather conditions.

6.1.2 Mission Objective

- To Develop a multimode, agile SAR payload operating in ScanSAR, Strip and spot modes to provide images with coarse, fine and high spatial resolutions respectively
- To develop and operate a compatible satellite to meet the mission requirements operating in three axis stabilized mode in 536.38 km circular sun synchronous orbit.
- To establish ground segment to receive and process SAR data.
- To develop related algorithms and data products to serve in well-established application area and also to enhance the mission utility.

6.1.3 Orbital Parameters

The guiding parameter for the orbit selection for RISAT is achieving a global coverage in a systematic way for a given swath. In interferometric applications modes, the presence of atomic oxygen and atmospheric drag has also been kept in view.

Parameters	ScanSAR Mode	Medium Resolution Mode	STRIP MAP Mode	Interferometer Mode
Altitude(km)	536.38	536.38	536.65	526.9
Inclination	97.554^{o}	97.554^{o}	97.555^{o}	97.52^{o}
Repeat cycle	377 orbits in 25 Days	377 orbits in 25 Days	2096 orbits in 139 Days	136 orbits in 13 Days
Orbit Period (Minutes)	95.4907	95.4907	95.542	95.294
Path-to-path Distance(km)	212.6	106.3	19.12	294.7
Swath (km)	223	115	25	25
Local Time: 6.00- Hrs +/- 5 min (Descending)				

6.1.4 Salient Features of the spacecraft

Parameter		RISAT-1
Structure		Single cylinder of 2.77 m height. The bottom side of the cylinder has a truncated triangular structure to accommodate major bus elements and hold the SAR antenna. At the top side of the cylinder a cuboid structure is designed to accommodate solar panels, sensors antenna etc.
Thermal	Components	Passive control using tapes , OSR, MLI Blankets and semi-active/active control using proportionate temperature controller and heaters(No eclipse in winter; max 22 min eclipse in summer) Less albedo load due to 6 AM/6PM equatorial crossing time
	Temp. Range	20±5 deg.C range for imaging sensors electro-optics 5+5 deg. C for Chemical Batteries 0 to 40 deg.C for electronic packages
Mechanisms	Payload	SAR Hold down &deployment Mechanism
	Solar array	Solar Array Hold down &Deployment Mechanism
Power	Solar Array	2 x 3 Al honeycomb sandwiched between CFRP faceskin. Panels will be along Roll axis.
	Battery	NiH_2 70 AH , 42 cells
Communication	Telemetry	2250.00 MHz , 4 Kbps with 32 kHz subcarrier/128 kHz sub-carrier(RT & Dwell) 16 kbps with 128 kHz sub-carrier (PB)
	Telecommand	2071. 875 MHz PCM/BPSK/PM Sub carrier frequency 44 kHz & 4 kbps
Attitude and Orbit Control (AOCS)	AOCS Spec.	Pointing accuracy : 0.05 Deg (3 sigma) (All Axis) Drift Rte: 3.0 e-4 deg/sec (3 sigma)
	sensors	Star sensors(2), Earth Sensor(2), Two Axis DSS, 4-Pi sun sensor(2), Tri-axial Magnetometer(1), IRU(3) with 1553 interface 10 channel S/A code SPS
	Attitude control	11 N canted thrusters(8), Center 11 N thruster(1), four Reaction Wheels(50 NMS), two 60.0 A-m_2Magnetic torquers,
	Modes Operation	Sun Acq, IAC mode, SKF,SSQ,ES-DTG, ES-Q, Normal Mode

Payload	SAR	Operating in C-Band 5.35 GHz. **Operation modes** Fine Resolution Strpmap Mode-1 (FRS-1) Coarse Resolution ScanSAR Mode(CRS) Medium Resolution ScanSAR Mode(MRS) Fine Resolution Stripmap Mode-2(FRS-2) High Resolution Spotlight Mode (HRS) Circular polarimetric Modes (C-HRS, C-FRS-1, C FRS-2, C-MRS, C-CRS) Mass 950 kg approx
Mass		1850 kg

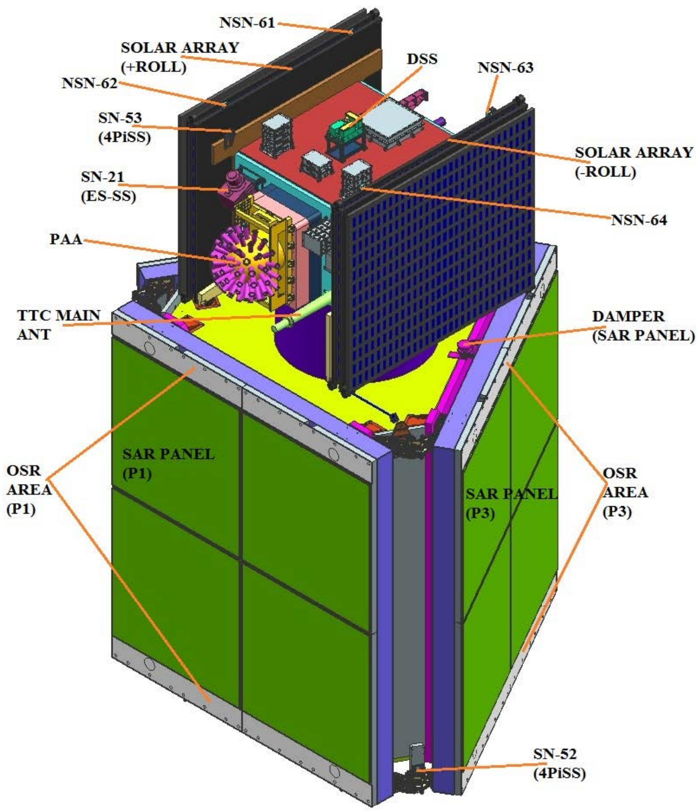


Figure 6-1 Stowed view of RISAT-1

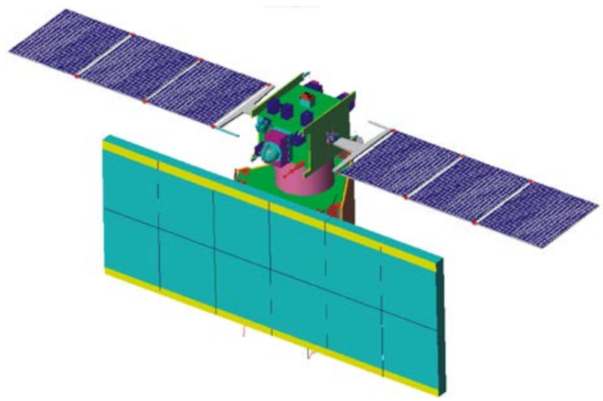


Figure 6-2 On-orbit view of RISAT

6.1.5 Payload

The C-Band Synthetic Aperture Radar (SAR) is the payload of RISAT. Radar backscattering depends upon the sensor parameters such as frequency, polarization and incidence angle, dielectric constant roughness and geometry of the target. In RISAT, SAR Payload will be operating in C-band (5.35 GHz) with both co-and cross- polarization, which will meet most of the resource applications and also enable achieving high resolution capability. The SAR payload is based on active phased array antenna technology, which will provide multimode capability.

6.1.6 Modes of Operation

The proposed SAR will operate in the following basic modes:

Fine Resolution strip map Mode-1 (FRS-1): This is the conventional mode of SAR. In this the orientation of the antenna beam is fixed with respect to flight path so that a strip of constant swath (25 km) is illuminated along the flight direction. The indented resolution is 3m for FRS-1 mode.

Coarse Resolution ScanSAR Mode (CRS): The scanSAR mode allows increasing the swath. This is achieved by periodically stepping the antenna beam to the neighboring subswaths(in range direction). In the CRS mode of RISAT there will be 12 beams. These results, total swath in CRS mode would be 223 km. the resolution offered in this mode will be 50 m.

Medium Resolution Stripmap Mode-2 (FRS-2): This is a 6 beam scanSAR mode, similar to the CRS mode, providing a resolution of 25 m over a swath of 115 km.

Fine resolution Stripmap Mode-2(FRS-2): This mode has quad polarization capability. Philosophically, this mode is a hybrid strip map and scanSAR. In this case the beam orientation is kept fixed with respect to the flight path and a strip of constant swath width is covered. Part of the aperture time the beam polarisation is switched from V-transmit to H-transmit, and vice-versa. Hence, this mode would be used for polarimetry, as we can have all the four combinations of polarization viz. VV, VH, HH,HV.

High Resolution Spotlight Mode (HRS): In the spotlight mode, the antenna beam is oriented continuously to illuminate a particular spot on the ground. This method increases the target aperture time which results in improved azimuth resolution (1m) The improved resolution is obtained at the cost of azimuth coverage.

Circular Polarimetric Modes (C-HRS, C-FRS-1, C-FRS-2, C-MRS, C-CRS): All the modes mentioned above can be operated in hybrid-circular polarization. This is achieved by transmitting H & V polarized signals simultaneously but with a relative phase-shift of 90^o. Hence, the transmit signal is in circular polarization and the receive signal is in linear (Dual-pol) – this makes it a hybrid-circular polarisation operation. To keep the average power-requirements same as the original specifications, the pulse-width is reduced to half.

Major Mission Parameters for Space borne High Resolution SAR

Parameter	Value	
Altitude	536 km	
Orbit	Sun Synchronous (6 A.M/6 PM equatorial crossing)	
P/L operating frequency	C-Band (5.35 GHz)	
Polarisation	Single/Dual/Quad-polarisation Hybrid circular polarimetry (Transmit circular, receive linear)	
Antenna	Microstrip Active antenna 6m x 2m	
Peak Gain	43.1 dBi	
Total no. Beams	64 on each side of the flight track (Total 128)	
On board storage	SSR with 240 GBits	
No. of TR Modules	288 pairs	
Transmitter power per TRM	10 W (Ave.)	
Antenna peak power	2.88 kW	
Average DC Input Power	3.92 kW	
Range Compression	On Ground	
Pulse Width	20 micro sec/10 microsec (10 microsec for circular polarization)	
Antenna Roll Bias(deg)	36	
Range Coverage(km)	107 - 659	
Look Angle (Deg	11.28 – 49.09	
Incidence Angle	12.25 – 55.02	
Doppler BW (Hz)	2532.23	
	FRS-1/FRS-2/ MRS/CRS	HRS
PRF(Hz)	2800-3200	3000-7000

Worst (dB) Considering both qualified and unqualified regions (100 km – 700km)		-16.81 @ 25 km	-15.82
Swath		25/25/115/223	10
Slant range resolution(m)		2/4/8/8	0.7
Ground range resolution(m)		FRS1: 9.4 – 2.4 FRS2: 18.8 – 4.9 MRS: 37.7-9.8 CRS: 37.7.9.8	3.3-0.85
Azimuth Resolution(m)		3/9/21-23/41-55	1
Chirp bandwidth(MHz)		75/37.5/18.75/18.75	225
Sampling frequency(MHz)		83.3/41.67/20.83/20.83	250
Data Window (micros)@ nominal earth radius of 6371 km		63-184 (@30 km Swath)	80-165(@10 km Swath
No. Of complex samples		4864-21504/2560-12288/1280-6144/1280-6144	19072-41344
Data Compression		Onboard BAQ(6/5/4/3/2 bits)	3 – bit BAQ
Data Rate (Mbps)		6 BAQ	3 BAQ(For 100 km azimuth)
	Single pol.	176-744/-/44-213/44-213	490 – 744
	Dual Pol.	352-1488/-/88-426/88-426	980-1488
	Quad pol.	-/1756-744/-/-	
Worst-case Range Ambiguity(in dB) @ nominal PRF		-16.94@22 km -15.6@25 km -13.4@30 km	-16.0 @10 km
Worst case Azimuth Ambiguity (in dB) @ Nominal PRF		-21.47	-25.20

Active Antenna specifications of C- Band SAR

Parameter	Value	
Frequency	5.350 GHz	
Antenna Type	Printed Antenna	
Antenna Size	6m (Along Flight) x 2m (Cross Flight)	
Antenna Gain	43.1 dBi	
Antenna Bandwidth	0.5 dB over 225 MHz bandwidth around center frequency	
Side Lobe Level	Azimuth	Elevation

	- 15 dB		- 18 dB	
Cross polarization level	Better than – 23 dB			
Relative gain and phase tracking between radiating arrays of 24 elements	Gain Tracking		Phase Tracking	
	0.5 dB rms		6 deg rms	
No. of TR Modules	288, each with 10 W peak power			
Peak Power	2.88 kW			
Avg. Output power	213 W (with duty cycle of 7 %)			
Average DC Input Power (to Active antenna)	3.672 kW			
TR Module Output tracking	O/P Power Tracking		Phase Tracking	
	0.5 dB rms		6 deg rms	
TR Module Receive path Tracking	Gain		Phase	
	0.5 dB rms		6 deg rms	
Gain/Phase Quantisation	Gain		Phase	
	6 bits		6 Bits	
TR Module Bandwidth	0.5 dB over 225 MHz bandwidth around centre frequency			
Loss/Noise Figure	Tx loss	Rx loss	Mismatch Loss	Noise Figure
	0.3 dB	0.3 dB	0.6 dB	3.5 dB
No. of Antenna Beams	128			

7. Space Science and Planetary Series

7.1 IRS-P3

7.1.1 Introduction

IRS-P3 was an experimental EO (Earth Observation) mission, a follow-up mission to IRS-P2, considered to be pre-operational and served in parallel for technology evaluation and scientific methodology studies. A portion of the payload was provided by DLR (German Aerospace Center). In addition, DLR provided data reception support (Neustrelitz) and launch phase support. The secondary use of the mission is to enhance and improve the IRS mission capabilities toward operationalization and application. This mission had two payload pointing modes, ie Earth pointing and stellar/inertial pointing.

7.1.2 Mission Objective

The Mission Objectives of IRS-P3 are

- To provide the opportunity for RS application in the areas of land, atmosphere and oceanographic investigations.
- To validate new RS methods and develop affiliated application potential.
- To provide opportunity for experiments in X-ray astronomy.
- As payload for the third developmental flight of PSLV.

Objectives of Earth pointing mode

- To provide continued remote sensing data services in the areas of improved crop discrimination, crop yield, crop stress and disaster management.
- Remote sensing of ocean atmosphere system and coastal waters and to retrieve quantitative values about the co-existing c-varying water constituents like chlorophyll sediments and gelbstoff
- To provide dynamic target for calibrating PCMC radars during Indian Launch campaigns.

Objective of Stellar pointing mode

- To study periodic and aperiodic intensity and spectral variations of galactic and extragalactic X-ray sources by making pointed mode observations of specific X- ray objects.
- To discover pulsations of binary nature and quasi-periodic oscillations of X-Ray sources
- Study of light curves and spectral evolution of transient & flaring X-ray sources

7.1.3 Orbit details

Table 7-1: IRS-P3 Orbit Details

Parameter	Value
Orbit	Sun-synchronous

Altitude	817 km
Orbital Period	101.35 minutes
Orbit inclination	98.7 deg
Equatorial Crossing Time	10.30 AM
Repeat Cycle	24 Days
Launch vehicle	PSLV-D3

7.1.4 Salient features of IRS-P3

The IRS-P3 spacecraft structure was of IRS-P2 heritage. The bus design had of four vertical panels and two horizontal decks supported on a central load-bearing cylinder of 930 mm diameter and 1188 mm height. The payload was accommodated on the outer side of the upper deck, which was oriented in flight direction (Roll axis). The onboard power generation was achieved by a pair of deployable, sun-tracking, un canted solar panels (9.636 m^2), which generates a power of 873 W. Two Ni-Cd batteries (21 Ah/24 Ah) catered to the eclipse and peak load demands.

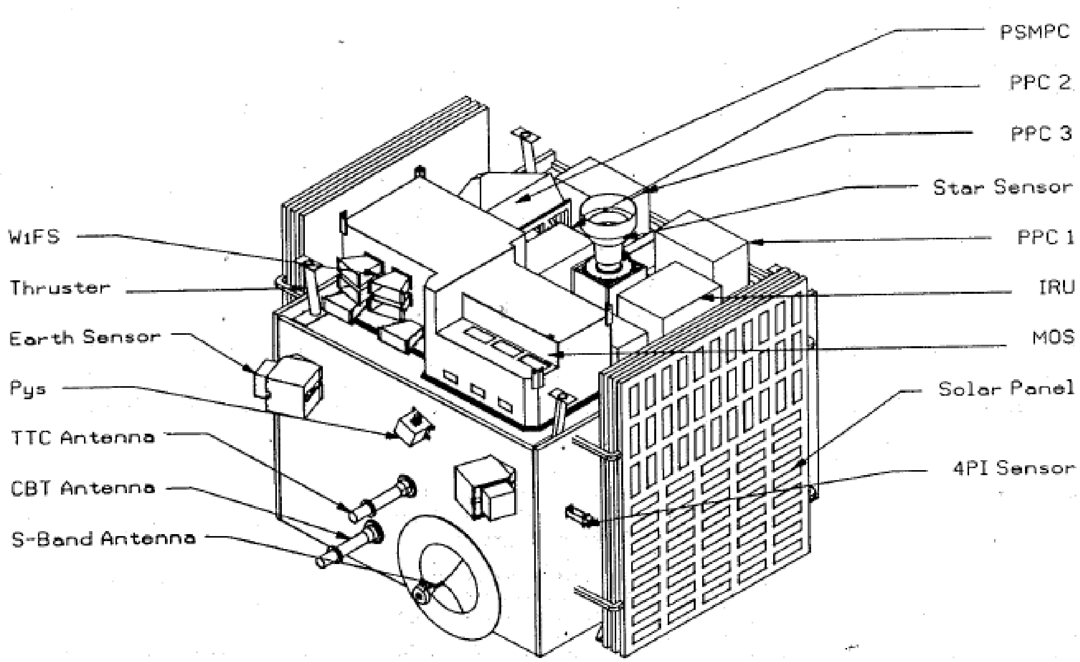


Figure 7-1: Stowed configuration of IRS-P3

The spacecraft was three-axis stabilized. The AOCS employed Earth sensors, sun sensors and dynamically tuned gyros as attitude sensors; actuation was provided by reaction wheels, magnetic torquers and an RCS (Reaction Control System). An Earth pointing accuracy of better than 0.20º in all axes and better than 0.05º in all axes for stellar pointing (X-ray observation mode) was provided. In addition to these attitude sensors, AOCS also employed a star sensor in control loop in order to maintain the attitude during stellar pointing mode. The star sensor was an area array CCD imager of 288 x 384 pixels (FOV of 6º x 8º). It worked as a star tracker with respect to a set of optical stars, identified a priori in conjunction with the X-ray package. The star sensor was mounted on positive roll axis and co-aligned with the X-ray payload's optical axis. When the spacecraft was inertially oriented and locked to a specified X-ray source, the star sensor works in a static mode. Therefore, the star sensor always locks to a specific scene about the roll axis. Total S/C mass = 922 kg, a hydrazine propulsion system (84 kg of fuel sufficient for three years) with 16 thrusters is used for orbit maintenance.

The IRS-P3 bus was derived from the flight proven IRS-1A/1B buses. New systems were the processor based attitude and orbit control system (AOCS) which was derived from INSAT-II for large angle maneuvers, the processor based telecommand system and FPGA based telemetry system. ISRO developed Ni-Cd batteries (24 Ah) were used.

Table 7-2 Salient features of IRS-P3

Subsystem		IRS-P3
Structure		Four vertical and two horizontal decks supported on a central load bearing cylinder of 930 mm dia. And 1188 mm height. Decks are made up of Aluminum/aluminum honeycomb panel. On Inner surfaces packages were mounted. Outside of Earth viewing panel carried payload data antennae, the TTC antenna and attitude sensors. The payloads were on outside of top deck
Thermal	Control	The design philosophy was maximum use of passive elements and minimum use of semi-active elements. This was achieved by extensive use of thermal control coatings, Thermal control tapes, optical solar reflectors (OSRs), multilayer insulation blankets(MLI), conductive grease etc.
	Limits	Electronic packages: between 0 to 40 deg C Battery: 5$\pm$5 Deg C
Power system	Solar Panel	Two sections feeding two batteries (Total 870 W generated) 9.7 Square Meter rigid panels. Controlled by Shunt switches. Battery charging control by dissipation less PWM taper charge regulators.
	Battery	Battery : Two 24 AH, Ni-Cd batteries comprising 28 cells each
	Electronics	370 W continuously, 40 V Bus

Telecommand		Operated in S-Band PCM/FSK/FM/PM modulation
Telemetry		System Based on PCM/PSK modulation in S-Band (2230 MHz) with dwell mode facility. One orbit data storage was implemented..
Tracking		The Tracking is provided by range and Doppler measurements using S-band TTC transponder.
Payload Data		The payload data was transmitted in S-Band 2280 MHz with BBSK modulation (Data rate 55.2 MBPS)
AOCS	Pointing Accuracy	AOCS Supported Nadir mode WiFS and MOS payloads and Stellar mode for X-Ray Payloads. Pointing accuracy : 0.2 deg. Nadir Mode 0.01 deg. Stellar mode
	Sensors	Sensors: Earth, Sun and star sensor Star sensor in loop mode was used for stellar pointing mode.
	Actuator	Actuators: Two Magnetic torquers, four Wheels, 8 one newton thrusters and one 8 newton thruster. Fuel loaded : 84 kg

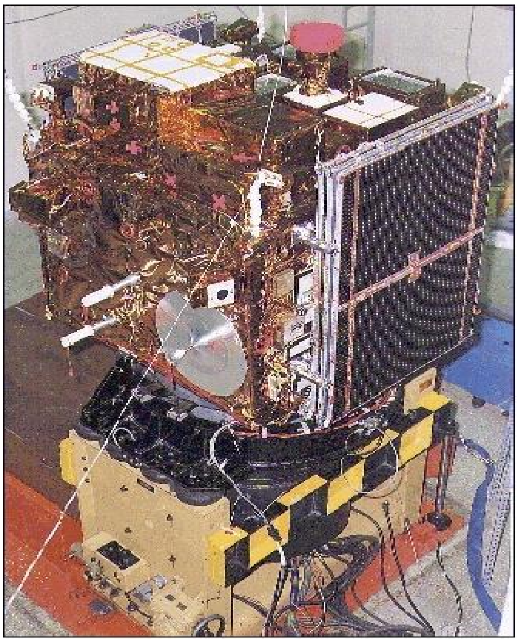


Figure 7-3 View of IRS-P3

Figure 7-2 The WiFS camera illustration

7.1.5 IRS-P3 Payloads

7.1.5.1 WiFS (Wide Field Sensor):

WiFS was an pushbroom imager of IRS-1C and IRS-1D heritage. WiFS was an extended version of 3 channels on IRS-P3: 0.62 - 0.68 µm, 0.77 - 0.86 µm, with an additional channel at 1.55-1.75 µm (SWIR). Each band had two detectors centered at a FOV of ±13.6º to achieve a swath of 770 km (repeat cycle of 5 days). The optics system consists of eight lenses with spectral bandpass and neutral density filters for each spectral band. The dynamic range in each gain was 7 bits. The absolute radiometric accuracy was better than 10% with relative in-band accuracy of 2%. The data rate for the VNIR data (2 channels) was 2.6 Mbit/s, for the SWIR data it was 1.73 Mbit/s. WiFS had a mass of 25 kg and used 50 W. The objectives of WiFS observations were to monitor the vegetation index on land and to observe the ocean surface.

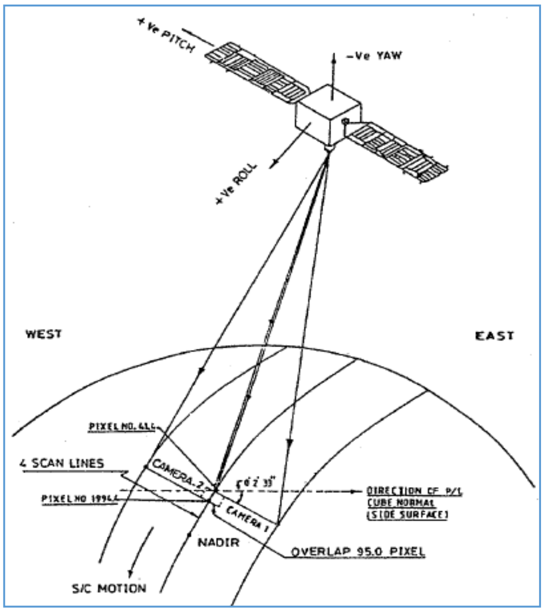


Figure 7-3: WiFS Swath coverage on earth

Table 7-3: WIFS camera specifications

Parameter	Value
Spectral bands (µm)	0.62 - 0.68, 0.77 - 0.86, 1.55 - 1.75 (SWIR)
Spatial resolution	188 m
Swath width	770 km (FOV of ±13.6º), 4096 pixels
Repetition cycle	5 days
SNR at saturation radiance	>128
Mis-registration	0.25 pixel
Data quantization	7 bit (radiometric resolution of 128 grey levels)
Integration time	28.42 ms
Data rate	2.06 Mbit/s
Mass	25 kg
Power	50 W

7.1.5.2 MOS (Multispectral Optoelectronic Scanner):

MOS is an experimental imaging push-broom spectrometer for VNIR/SWIR range observations. MOS is provided by DLR (German Aerospace Center), Berlin. The objective is to monitor the Earth's surface (surface-atmosphere interaction, ocean color, phytoplankton, regional and global distributions of man-made aerosols and their links to gaseous admixtures, spectral and spatial cloudiness characteristics, etc.) in the VNIR/SWIR region of 0.4 - 1.6 µm.

The goals of MOS payload are

- To design and build a spectral imaging instrument, dedicated for ocean colour Remote Sensing with many > 10 narrow spectral channels in the VIS-NIR range (400-1000nm)
- To separate the problem of object signature and atmospheric disturbance by independent measurements in different spectral regions and with special designed optical means.
- To make experiments to prove the instrument concept and to get experience in high spectral data handling and image processing.
- To develop algorithms and test the methodological concept with emphasis on CASe-2 coastal water
- To make measurements at different ocean/coastal regions, by satellite and synchronous ground truth to verify the algorithm or carry out its regional tuning, if necessary.

The sensor apparatus consists of three complementary instruments. MOS operation requires at least one calibration per month (with respect to the sun).

MOS-A is an atmospheric spectrometer with four narrow channels in the O_2A-absorption band at about 760 nm for the measurement of atmospheric turbidity. The data from MOS-A are used for correction of the atmospheric influence (scattering) on the multispectral data of MOS-B. In addition the O_2A-method permits the measurement of aerosol content and profile.

MOS-B is a 13-channel spectrometer in the spectral range of 408 to 1010 nm. The center wavelengths of the channels are chosen with the objective for the quantitative retrieval of ocean and coastal zone parameters. They also provide a capability for vegetation signature determination (red edge) and estimation of H_2O (water vapor) content in the atmosphere from the NIR-measurements.

MOS-C is a line camera at 1.6 µm with a bandwidth of "50 nm. The SWIR channel data is used for improved surface term and roughness estimation. In addition the data of the SWIR channel may be used for the following applications: cloud/snow/ice discrimination, cloud type discrimination, estimation of sea surface roughness, and for the improvement of atmospheric correction algorithms.

Table 7-4: Specifications of the MOS instruments

Parameter	MOS-A	MOS-B	MOS-C
Spectral range (nm)	755 - 768 nm	408 - 1010	SWIR
No. of channels	4	13	1

Wavelengths (nm)	756.7; 760.6; 763.5; 766.4 (O_2A-band)	408; 443; 485; 520; 570; 615; 685; 750; 870; 1010; 815; 945 (H_2O-vapor)	1600
Spectral resolution	1.4 nm (FWHM)	10 nm (FWHM)	100 nm (FWHM)
FOV along-track FOV across-track	0.344º 13.6º	0.094º 14.0º	0.14º 13.4º
Swath width	195 km	200 km	192 km
No. of pixels per row	140	384	299
Spatial resolution (ground pixel size)	1.57 km x 1.4 km	0.52 km x 0.52 km	0.52 km x 0.64 km
Measuring range L_{min}-L_{max} \[$\mu Wcm^{-2}nm^{-1}sr^{-1}$\]	0.1 - 40	0.2 - 65	0.5 - 18
Data quantization	16 bit		
Data rate	1.3 Mbit/s		

MOS calibration: In-orbit calibration measurements are performed using internal reference lamps (prior to each data take). In addition sun calibration measurements are performed once a month. This is achieved with a diffuser in front of the entrance optics of the sensor. The following calibration functions are performed:

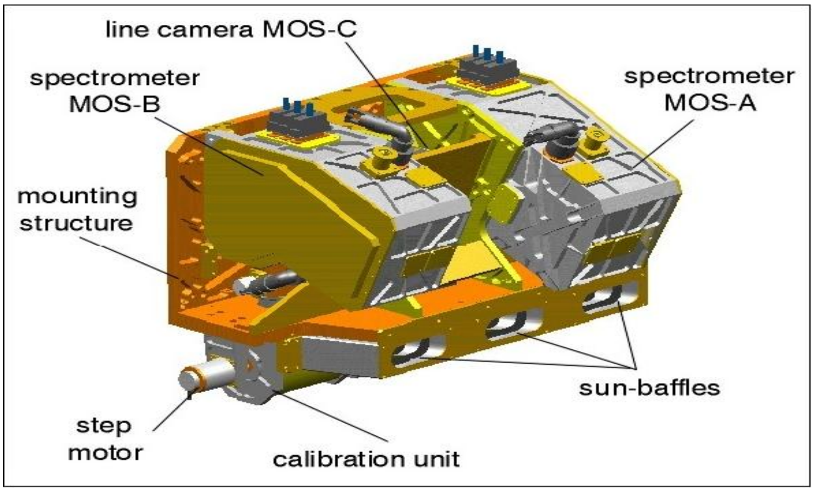


Figure 7-4 Illustration of the MOS (Modular Optoelectronic Scanner) instrument

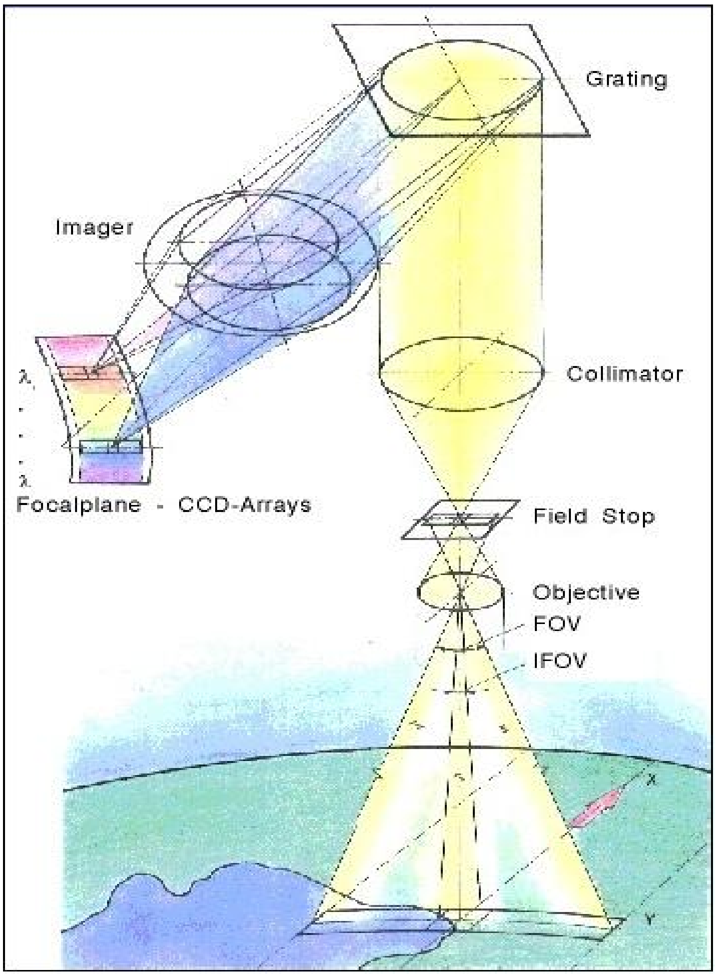


Figure 7-5 Schematic illustration of the MOS observation concept

- DSNU (Dark Signal Non-Uniformity) and PRNU (Photo Response Non-Uniformity)
- Absolute sensitivity calibration
- Linearity control
- Spectral alignment control

MOS in-orbit inter calibrations with sensors from other missions are attempted when orbital opportunities arise for a common target area or test sites. Examples are: MOS on IRS/P3 with MOS on Priroda, or with SeaWiFS on Seastar, or with OCTS on ADEOS.

Principle of the imaging pushbroom spectrometer operation: A strip (swath) of the Earth's surface is imaged through the entrance optics on the field stop. The collimator optics realizes parallel light rays falling onto the grating. The grating disperses the different “colors” that are focussed by the imager into the focal plane. Corresponding to the desired wavelength, CCD line arrays are mounted into the focal plane.

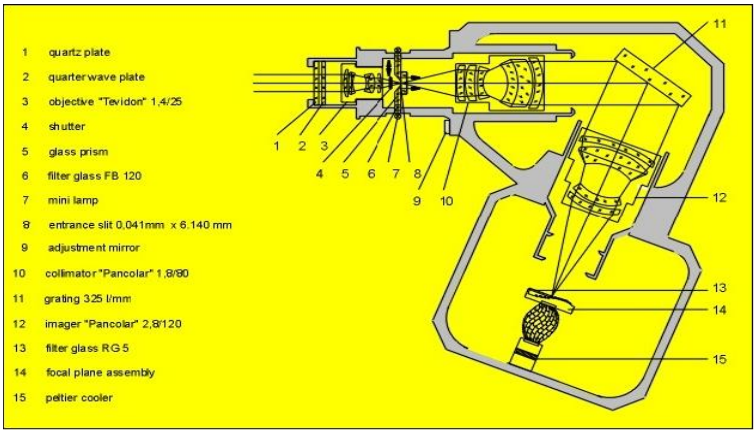


Figure 7-6 Schematic illustration of the optical block of MOS-B

Sensor Calibration

The demanded radiometric data quality was guaranteed by two on-board calibration concepts realized in hardware. 1. An internal sensitivity check 2. an external calibration to the sun (SUNCAL). The internal check is made in each block with two small filament lamps mounted besides the entrances slit. Through the auxiliary slits the lamps are illuminating the collimator optic and after dispersion at the grating are illuminated the CCD – lines in the focal plane. By powering the lamps in four high stabilized current levels and superposing of both lamps we have 16 levels of different illumination intensities for each channels in MOS-A and MOS-B. In MOS-C CCD was illuminated directly by the lamps.

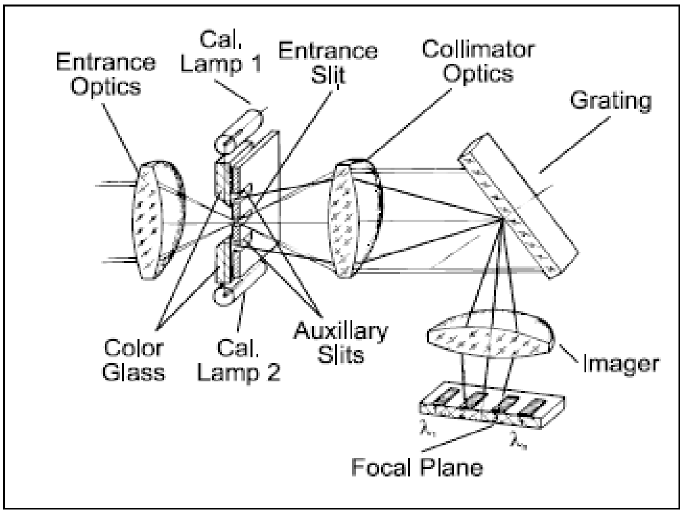


Figure 7-7.Optical Schematic of MAS-A and MOS-B

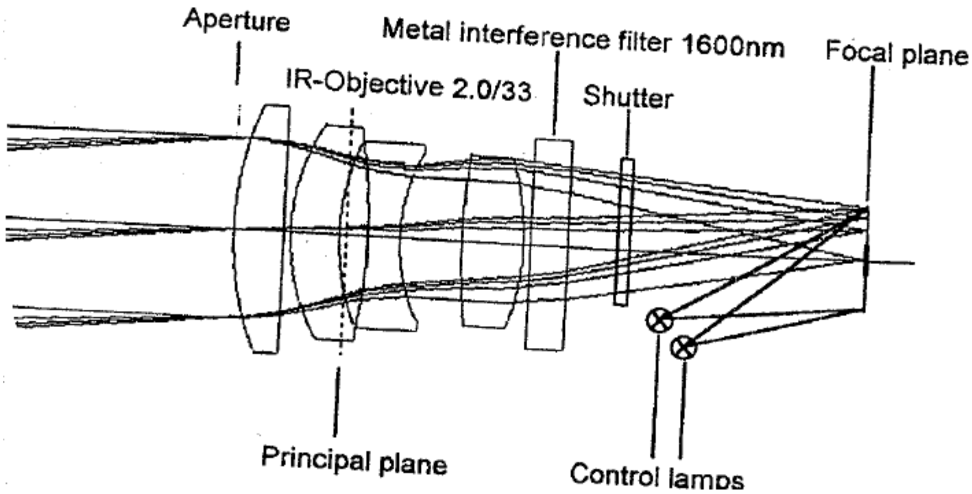


Figure 7-8 Optical Schematic of MOS-C

7.1.5.3 IXAE (Indian X-ray Astronomy Experiment):

IXAE is an ISRO/ISAC and TIFR (Tata Institute of Fundamental Research, Mumbai, India) cooperative experimental astronomy instrument package with the objective to study periodic and aperiodic intensity and spectral variations in X-ray sources. Source observation is achieved by `pointed mode observations,' employing an array of three co-aligned collimated **PPC** (Pointed Proportional Counter). The system operates in mutual anticoincidence fashion for significant reduction of background noise (cosmic rays and Compton interaction of gamma rays).

Another objective is the study of light curves and the spectral evolution of transient and flaring X-ray sources as well as long-term intensity monitoring of known binary X-ray stars and other bright X-ray sources. This is achieved by means of **XSM** (X-ray Sky Monitor), based on the principle of a pin hole placed above a position sensitive to PPC in anti-coincidence mode.

Table 7-5 PPC and XSM instrument specification

PPC		XSM	
Energy range	2 - 20 keV	Energy range	2 - 8 keV
FOV	2º x 2º	FOV	90º x 90º
No of PPC	3	Pin hole size	1 cm^2
No of layers per PPC	3	Distance to detector	16 cm
No. of anode cells/layer	18	Detector	32 proportional counters
Size of cell	1.1 cm x 1.1 cm	Detector cell size	1 cm x 1 cm x 32 cm
Entrance window	25µm , 500 Å, Al coated	Window	25 µm Mylar, Al coated
Filling gas	Ar+CH_4, at 800 torr	Filling gas	Ar+CH_4

The principle objective of the IXAE is to carry out timing studies of X-ray pulsars, X-ray binaries, and other rapidly varying X-ray sources. The XSM detects transient X-ray sources and monitors the light intensity of bright X-ray binaries. Each of the detectors (PPC, XSM) are controlled by independent microprocessor based processing electronics. A common electronics subsystem acts as an interface with the satellite bus. An oven controlled oscillator (accuracy one part in 109) provides high timing accuracy.

The PPC is a multi-cell multi-layer proportional counter array with active anticoincidence on three sides. The total geometric area is about 400 cm^2, the filling gas is 90% Argon + 10% Methane. A 25 µm aluminized mylar acts as the entrance window. The field of view is restricted to 2º x 2º using a passive collimator. The detector has a command controlled high voltage unit. The processing electronics for the PPC has an onboard memory of 512 kByte, the spectra (64 channels spanning 2 to 30 keV) and light-curves are stored onboard with the command selectable integration times.

The XSM is a planar position sensitive proportional counter with a pin-hole of 1 cm^2 positioned 16 cm above the detection plane. The FOV is 90º x 90º. The detection plane consists of 32 proportional counter cells with a resistive wire (NiCr) as the anode. Position resolution along the wires is achieved by charge division and perpendicular to the wires it is achieved by cell placement (1 cm). The energy range of the detector is 2 to 8 keV.

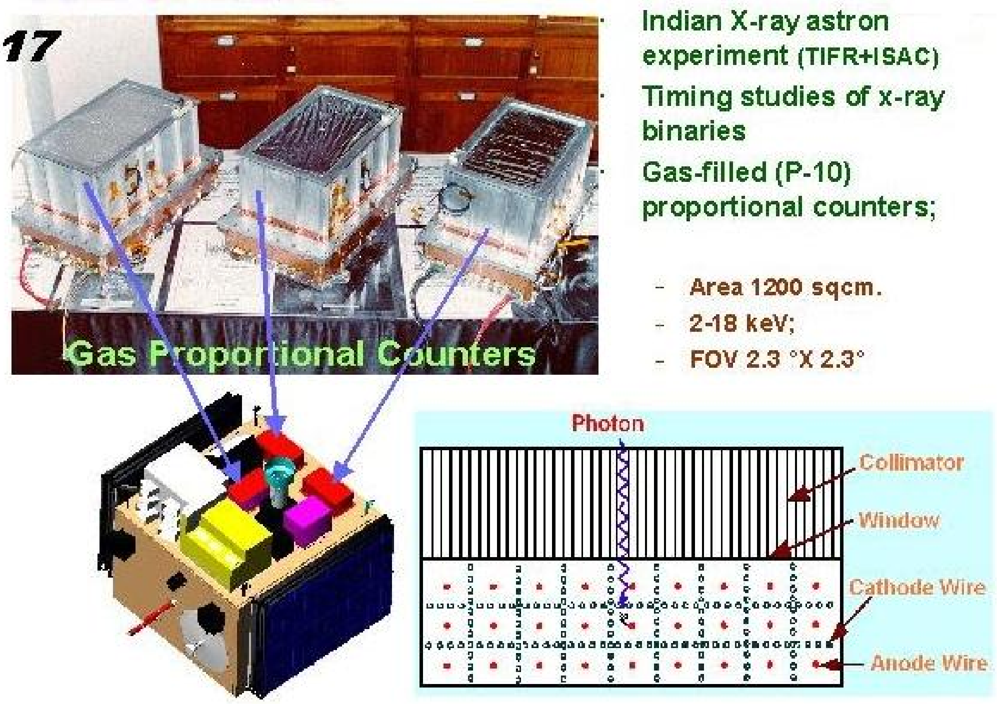


Figure 7-9 Schematic view of the IXAE instrumentation

7.1.5.4 C-Band Transponder

The C-Band transponder system consists of three portions as given below and is used for calibrating ground radars at SHAR

1. Receiving section which operates at 5.660 GHz with sensitivity of -70 dBm
2. Transmitting section which operates at 5510 MHz and 5800 MHz with a peak power of 400 watts
3. DC/DC converter which provides constant output voltages

Single antenna is used for receiving and transmitting Operations.

Receiver: The RF input received by the antenna is fed to the circulator. The signal passes from circulator to the pre-selector filter. This filter is used as selector. Output from this filter/selector is fed to a mixer. Output of local oscillator (LO) which is fundamental oscillator generates C-Band frequency is also fed into the mixer. Mixer performs as a down converter and converts C-Band signal to the IF frequency. The amplified IF signal is detected by a solid state detector, filtered by a low pass filter, am0plified and passed to digital section through a buffer. The digital circuits provide triggers to the modulator which produces a high voltages negative pulse for cathode pulsing of the magnetron.

Transmitter: The Transmitter is a mechanically tuned C-Band magnetron oscillator. The power output is provided to the unit through 4 port circulator.

DC/DC converter: The DC/DC converter provides input to CBT at 23 V +2% and 1.2 A. the initial surge current requirement under all conditions is 1.5A.

7.2 Chandrayaan-1

7.2.1 Introduction

Chandrayaan-1 is the first Indian planetary exploration mission that performed remote sensing observation of the Moon to enhance our understanding about its origin and evolution. Chandrayaan-1 was launched successfully on October 22, 2008 from SDSC SHAR, Sriharikota. The spacecraft was orbiting around the Moon at a height of 100 km from the lunar surface for chemical, mineralogical and photo-geologic mapping of the Moon. The spacecraft carries 11 scientific instruments built in India, USA, UK, Germany, Sweden and Bulgaria.

After the successful completion of all the major mission objectives, the orbit has been raised to 200 km during May 2009.

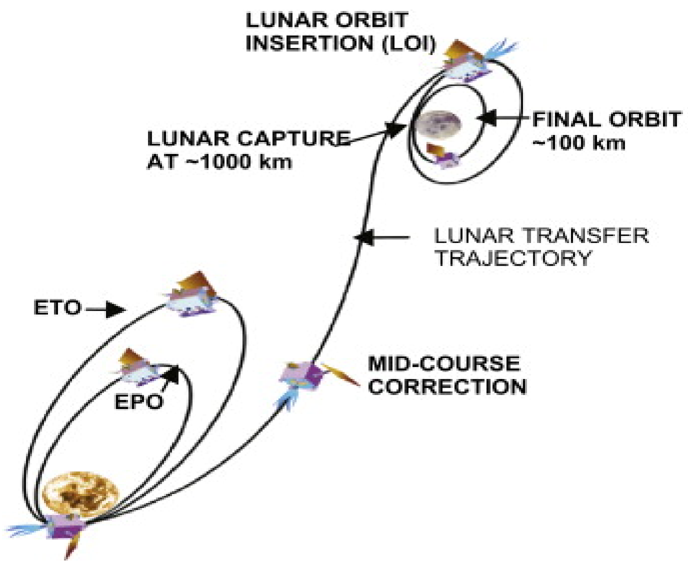
7.2.2 Mission Objectives

Mission Objectives of Chandrayaan-1 are as follows

- To realise the mission goal of harnessing the science payloads, lunar craft and the launch vehicle with suitable ground support systems including Deep Space Network station.
- To realise the integration and testing, launching and achieving lunar polar orbit of about 100 km, in-orbit operation of experiments, communication/ telecommand, telemetry data reception, quick look data and archival for scientific investigation by identified group of scientists.

7.2.3 Orbit Details

Chandrayaan-1 Mission sequence and Final orbit are shown in following picture



Parameter	Value
Mission	Remote Sensing, Planetary Science
Orbit	100 km x 100 km : Lunar Orbit

Launch Date	22 October 2008
Weight	1380 kg (Mass at lift off)
Launch Site	SDSC, SHAR, Sriharikota
Launch Vehicle	PSLV - C11

7.2.4 Salient features of Chandrayaan-1

The spacecraft design is adopted from flight proven Indian Remote Sensing Satellite bus coupled with suitable modifications specific to the lunar mission. Apart from the solar array, TTC and data transmission, that are specific to the lunar mission, other aspects of system design have flight heritage. However, some changes specific to lunar mission is also incorporated. These include extending the thrust cylinder and having an upper payload deck to accommodate MIP and few other payloads. Additionally, Chandrayaan-1 had a canted solar array since the orbit around the Moon is inertially fixed resulting in large variation in solar incidence angle. There is a need to have a gimbaled high gain antenna system for downloading the payload data to the Indian Deep Space Network (IDSN).

The spacecraft is a cuboids in shape of approximately 1.50 m side, with a liftoff mass of about 1.380 ton with bus element accounting for about 0.4 ton, payload about 0.1 ton and propellant about 0.8 ton. At lunar orbit it will be about 0.6 ton. This is a three-axis stabilized spacecraft generating about 750 W of peak power using canted single sided solar array and supported by a Li-Ion battery for eclipse operations. The spacecraft used bipropellant system to carry it from EPO through lunar orbit, including orbit and attitude maintenance in lunar orbit. The propulsion system carried required propellant for a mission life of two years, with adequate margin. The TTC communication is in the S-band. The scientific payload data that stored in a solid-state recorder is later played back and down linked in X-band through 20 MHz bandwidth by a steer able antenna pointing at DSN.

Parameter	Value
Scientific Objectives:	Simultaneous chemical, mineralogical and photo geologic mapping of the whole moon in visible, near infrared, low and high energy X-rays with high spatial resolution
Scientific Payloads	Terrain Mapping Camera-TMC Hyper Spectral Imager-HySI Lunar Laser ranging Instrument-LLRI Low Energy X-ray Spectrometer-LEX Solar X-ray Monitor- SXM High Energy X-ray /X-ray Spectrometer-HEX
Payload Weight	55 kg (Including 10 kg Announcement of Opportunity payload)
Launcher	Polar Satellite Launch Vehicle-PSLV-XL
Mission Strategy	Elliptic Parking Orbit. Trans Lunar Injection. Lunar Orbit Insertion
Lunar Orbit	100 X 100 km Circular Polar
Operational Life Time	Two Years
Spacecraft	Cuboid shape, 1.5 m side, 3-axis stabilized

Spacecraft Mass	Dry mass-440 kg, Initial Lunar Orbit Mass with propellant-524 kg
Communication System	S-Band uplink for telecommand, S-Band downlink for telemetry, X-Band for Payload data reception
Deep Space Network (DSN) Station	Location : Bangalore, Fully steerable dual feed 34m-dia antenna
Mission Control Centre	Location : Bangalore-responsible for all spacecraft operations, running of ground infrastructure
National Science Data Centre (NSDC)	Act as repository of scientific data Centre (NSDC) from experiments conducted on-board Chandrayaan-1

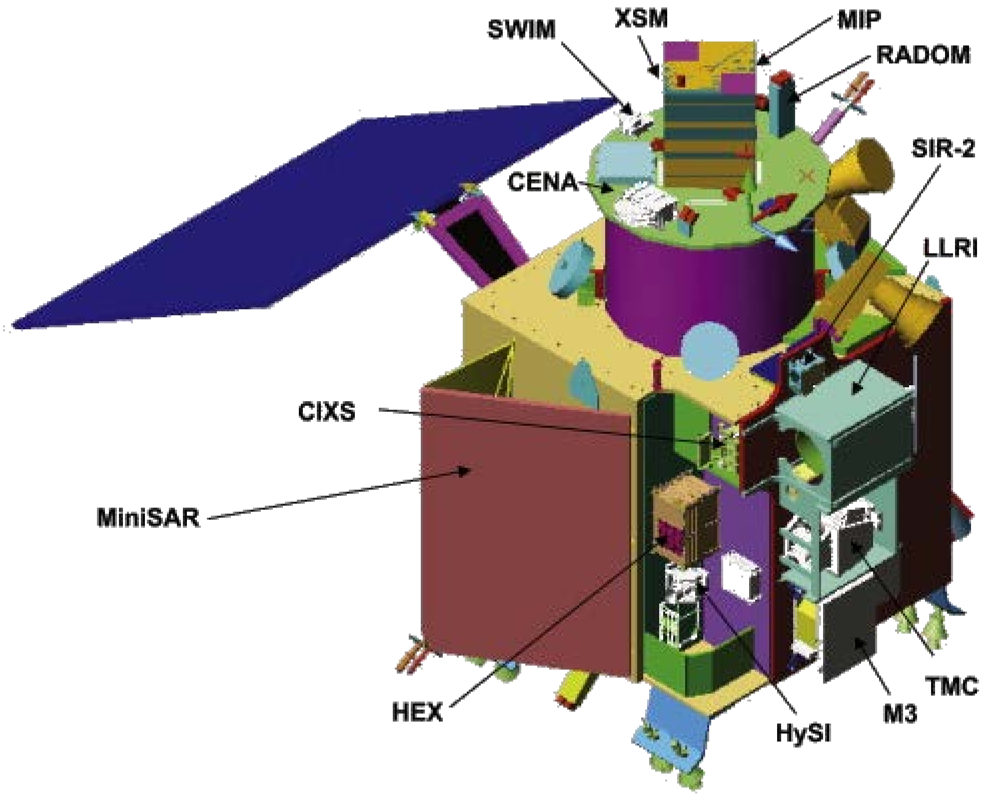


Figure 7-10 Deployed mode View of Chandrayaan

7.2.5 Chandrayaan-1 Payloads

There are 11 instruments on chandrayaan-1. Among them five are from India and six are from other space agencies.

Table 7-6 Chandrayaan-1 Payloads

Sl.No	Payload	Description	Organisation & Country
Scientific Payloads from India			
1	Terrain Mapping Camera (TMC)	camera in the panchromatic band having 5m spatial resolution and	SAC, ISRO, India

		40 km swath, to prepare a high resolution atlas of moon	
2	Hyper Spectral Imager (HySI)	imager operating in 400-900nm band with a spectral resolution of 15nm and spatial resolution of 80 m with a swath of 40 km, for mineralogical mapping	SAC, ISRO, India
3	Lunar Laser Ranging Instrument (LLRI)	for determining accurate altitude of the spacecraft above the lunar surface for topographical mapping	LEOS,ISRO,India
4	High Energy X - ray Spectrometer (HEX)	with a ground spatial resolution of approximately 20 km, for measuring 210Pb, 222Rn degassing, U, Th etc	PRL, India
5	Moon Impact Probe(MIP)	payload for exploration of the moon from close range and impacting on the moon	VSSC, ISRO, India
Scientific Payloads from abroad			
1	Chandrayaan-I X-ray Spectrometer (CIXS)	X-ray spectrometer	Rutherford Appleton Laboratory (RAL),UK
2	Near Infrared Spectrometer (SIR - 2)	Investigations of the process of basin, Maria and crater formation on the Moon	Max-Planck Institute, Lindau,
3	Sub keV Atom Reflecting Analyzer (SARA)	for imaging the Moon surface using low energy neutral atoms as diagnostics in the energy range 10eV-2keV	Swedish Institute of Space Physics
4	Miniature Synthetic Aperture Radar (Mini SAR)	for detection of water ice in the permanently shadowed regions on the Lunar poles up to a depth of a few meters	(NASA) Developed by JHU/APL and NAWC
5	Moon Mineralogy Mapper (M3)	spectrometer for characterization and mapping lunar surface mineralogy in the context of lunar geologic evolution	(NASA) Brown University and JPL.
6	Radiation Dose Monitor (RADOM)	To qualitatively and quantitatively characterize, in terms of particle flux, dose rate and deposited energy spectrum, the radiation environment in near moon space	Bulgarian Academy of Sciences

7.2.5.1 Terrain Mapping Cameara (TMC)

The terrain mapping stereo camera (TMC) in the 500–850 nm band with three linear array detectors for nadir, fore and aft viewing and has a swath of 20 km. It provide 3D image of the lunar surface with a ground resolution of 5 m with base to height ratio of one.

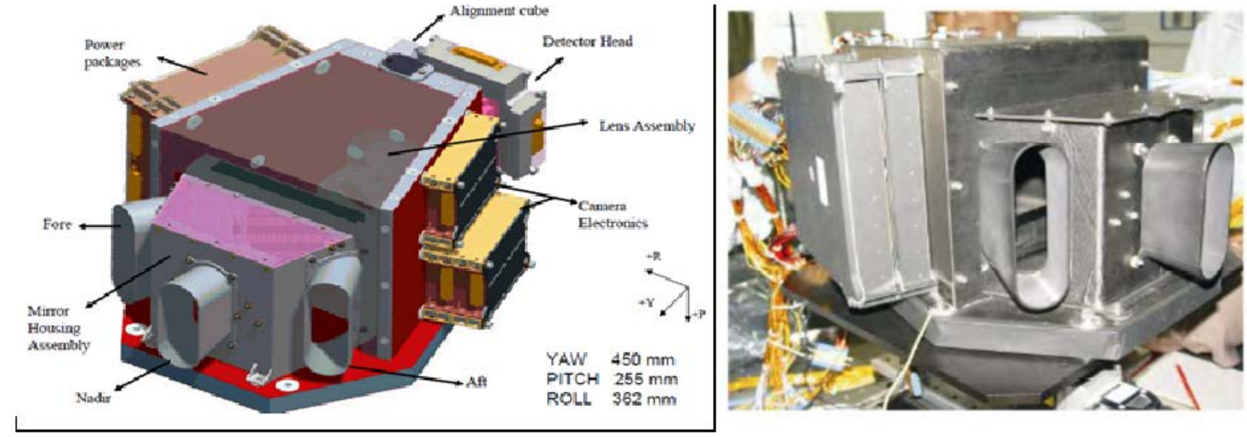


Figure 7-11 Terrain mapping Camera

Scientific Objective:

The aim of TMC is to map topography of both near and far side of the Moon and prepare a 3-dimensional atlas with high spatial and elevation resolution of 5 m. Such high resolution mapping of complete lunar surface will help to understand the evolution processes and allow detailed study of regions of scientific interests. The digital elevation model available from TMC would improve upon the existing knowledge of Lunar Topography.

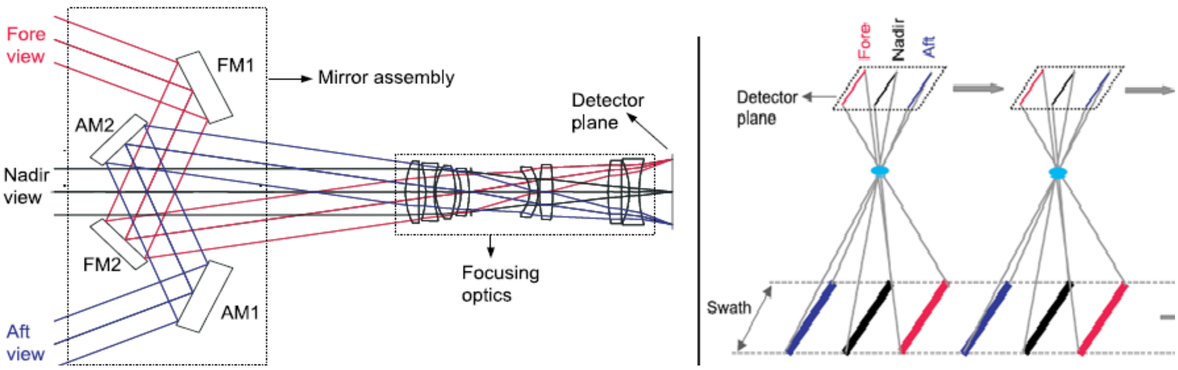


Figure 7-12 Optical Schematic and View angles of TMC

Payload Configuration Details:

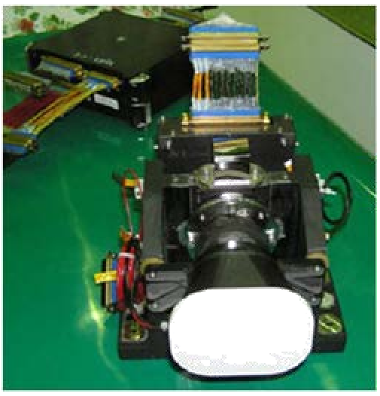
The TMC images in the panchromatic spectral region of 0.5 to 0.85 μm, with a spatial/ ground resolution of 5 m and swath coverage of 20 km. The camera is configured for imaging in the push broom mode, with three linear 4k element detectors in the image plane for fore, nadir and aft views, along the ground track of the satellite. The fore and aft view angles are ±25º respectively w.r.t. Nadir.

TMC measures the solar radiation reflected / scattered from the Moon's surface. The dynamic range of reflected signal is quite large and is represented by the two extreme targets – fresh crust rocks and mature mare soil.

TMC uses Linear Active Pixel Sensor (APS) detector with in-built digitizer. Single refractive optics covers the total field of view for the three detectors. The optics is designed as a single unit catering to the wide field of view (FOV) requirement in the direction along the ground track. The incident beams from the fore (+25°) and aft (-25°) directions are directed on to the focusing optics, using mirrors. Modular camera electronics for each detector is custom designed for the system requirements using FPGA. The data rate is of the order of 50 Mbps. The dimension of TMC payload is 370 mm x 220 mm x 414 mm and mass is 6.3 kg.

7.2.5.2 Hyper spectral imager (HySI)

The hyper spectral imager for mineralogical mapping is operating in the 400–950 nm range employing a wedge filter coupled to an area array detector. It has 64 continuous channels with a spectral resolution better than 15 nm and a spatial (pixel) resolution of 80 m with a swath of 20 km



Scientific Objective:

To obtain spectroscopic data for mineralogical mapping of the lunar surface. The data from this instrument help in improving the available information on mineral composition of the surface of Moon. Also, the study of data in deep crater regions/central peaks, which represents lower crust or upper mantle material, helps in understanding the mineralogical composition of Moon's interior.

Payload Configuration Details:

The uniqueness of the HySI is in its capability of mapping the lunar surface in 64 contiguous bands in the VNIR, the spectral range of 0.4-0.95 μm region with a spectral resolution of better than 15 nm and spatial resolution of 80 m, with swath coverage of 20 km. HySI collects the Sun's reflected light from the Moon's surface through a tele-centric refractive optics and focus on to an APS area detector for this purpose. The dispersion is achieved by using a wedge filter so as to reduce the weight and compactness of the system compared to using a prism / grating.

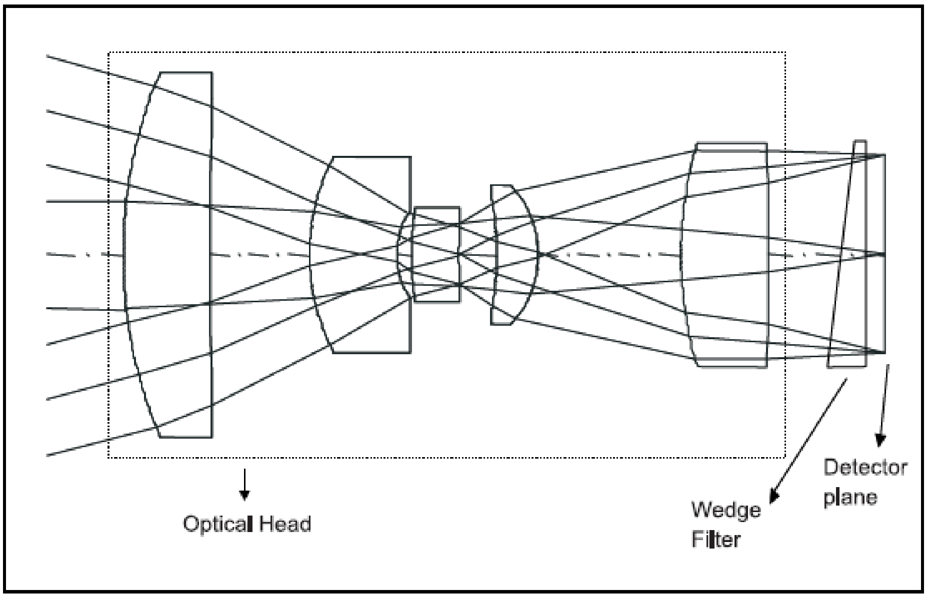


Figure 7-13 Optical Ray trace of HySI

The wedge filter is an interference filter with varying thickness along one dimension so that the transmitted spectral range varies in that direction. The wedge filter is placed in close proximity to an area detector. Thus, different pixels in a row of the detector will be receiving irradiance from the same spectral region but different spatial regions in the across track direction. In the column direction of the detector, different rows will receive irradiance of different spectral as well as spatial regions in the along track direction. The full spectrum of a target is obtained by acquiring image data in push broom mode, as the satellite moves along the column direction of the detector. An Active Pixel Sensor (APS) area array detector with built-in digitizer maps the spectral bands. The payload mass is 2.5 kg and its size is 275 mm x 255 mm x 205 mm.

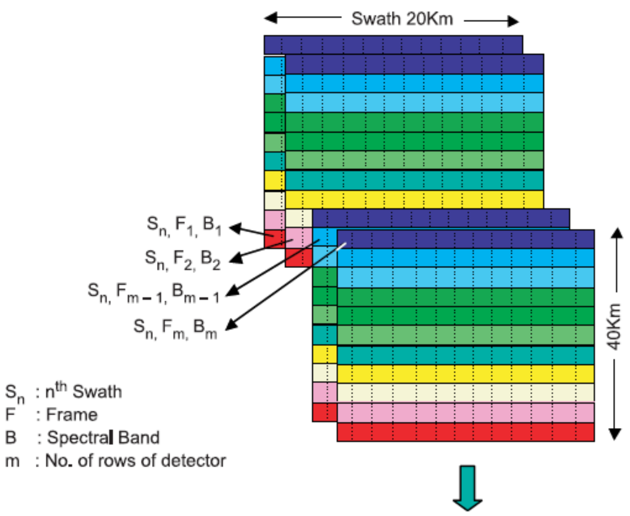
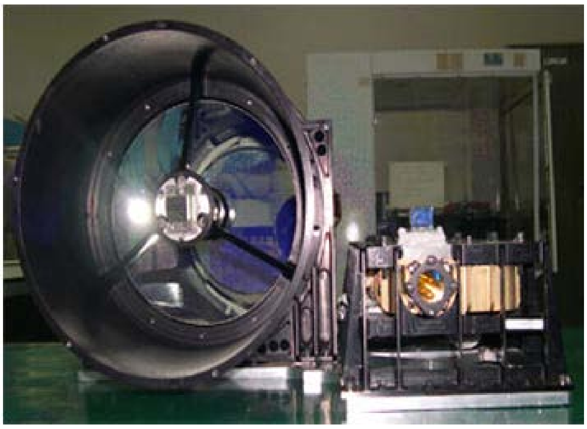


Figure 7-14 Mapping Scheme of HySI

7.2.5.3 Lunar laser ranging instrument (LLRI)

Scientific Objective:

To provide ranging data for determining the height difference between the spacecraft and the lunar surface.



The elevation map of the Moon, prepared using the laser ranging instrument carried onboard Chandrayaan-1 spacecraft helps in studying the morphology of large basins and other lunar features, study stress, strain and flexural properties of the lithosphere and when coupled with gravity studies, helps to find the density distribution of the crust

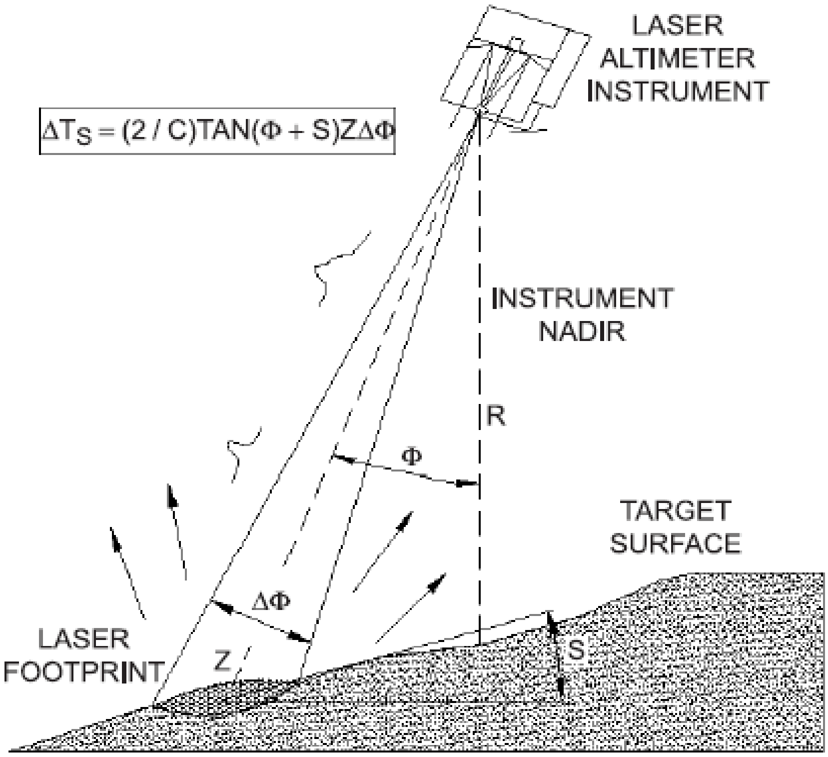


Figure 7-15: Laser pulse roundtrip – illustration

The LLRI employs an Nd–Yag laser with energy 10 mJ and employ a 20 cm optics receiver coupled to a Si–APD (Avalanche Photo Diode). It is operating at 10 Hz (5 ns pulse) and can provide a vertical resolution better than 5 m. The LLRI and TMC provide complementary data for generating a topographic map of the Moon and the LLRI, in particular, provide the first such data set for the polar region at higher than 80° latitude.

Payload Configuration Details:

LLRI works on the time-Of-Flight (TOF) principle. In this method, a coherent pulse of light from a high power laser is directed towards the target whose range is to be measured. A fraction of the light is scattered back in the direction of the laser source where an optical receiver collects it and focuses it on to a photoelectric detector. By accurately measuring the roundtrip travel time of the laser pulse, highly accurate range/spot elevation measurements can be made.

LLRI consists of a 10 mJ Nd:YAG laser with 1064 nm wave source operating at 10 Hz pulse repetition mode. The reflected laser pulse from the lunar surface is collected by a 200 mm Ritchey-Chrétien Optical receiver and focused on to a Silicon Avalanche Photodetector. The output of the detector is amplified and threshold detected for generating range information to an accuracy <5m. Four constant fraction discriminators provide the slope information in addition to range information. The different modes of operation of LLRI and the range computations from the detector output are controlled and computed by a FPGA based electronics. The processed outputs of LLRI are used for generating high accuracy lunar topography. The payload mass is 11.37 kg with base plate.

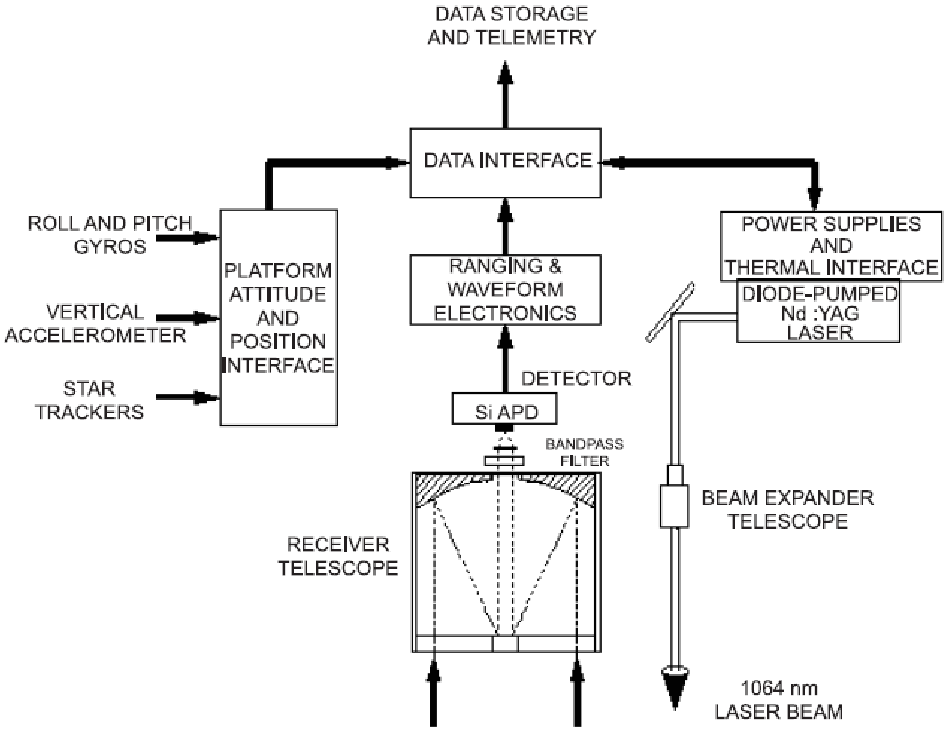


Figure 7-16: Block Schematic of LLRI

7.2.5.4 High energy X– γ ray spectrometer (HEX)

The high-energy X–γ ray (30–270 keV) spectrometer (HEX) employs CdZnTe solid-state detectors and has a suitable collimator providing an effective spatial resolution of 40 km in the low energy region (<60 keV). It employs a CsI anticoincidence system for reducing back-ground and is primarily intended for the study of volatile transport on Moon using the 46.5 keV γ ray line from ^{210}Pb decay as tracer. ^{210}Pb is a decay product of volatile ^{222}Rn and both belong to the ^{238}U decay series. The instrument has a detection threshold of <30 keV and a resolution of better than 10% at 60 keV. This instrument is to infer compositional characteristics of lunar terrain from a study of the continuum background in this energy range as well as low resolution Th and U mapping of terrains enriched in these elements

The High-Energy X-ray spectrometer covers the hard X-ray region from 30 keV to 270 keV. This is the first experiment to carry out spectral studies of planetary surface at hard X-ray energies using good energy resolution detectors. The High Energy X-ray (HEX) experiment is designed primarily to study the emission of low energy (30-270 keV) natural gamma-rays from the lunar surface due to ^{238}U and ^{232}Th and their decay chain nuclides.

Scientific Objectives:

- To identify excess ^{210}Pb in lunar polar regions deposited there as a result of transport of gaseous ^{222}Rn, a decay product of ^{238}U from other regions of the Moon. This will enable us to understand transport of other volatiles such as water to the polar regions.
- To detect other radioactive emissions, to characterise various lunar terrains for their chemical and radioactive composition on the basis of specific/integrated signal in the 30-270 keV region.
- To explore the possibility of identifying polar regions covered by thick water-ice deposit from a study of the continuum background.

Payload Configuration Details:

The geometric detector area of 144 cm^2 is realized by nine Cadmium Zinc Telluride (CZT) arrays, each 4 cm x 4 cm (5 mm thick), composed of 256 (16x16) pixels (size: 2.5 mm x 2.5 mm). Each CZT array is readout using two closely mounted Application Specific Integrated Circuits (ASICs), which provides self-triggering capability. The detector will be biased at the cathode with –550 V and the electronic charge signals are collected at the anode. A Cesium Iodide (CsI (Tl)) scintillator crystal coupled to photomultiplier tubes (PMT), will be used as the anticoincidence system (ACS). The ACS is used to reduce the detector background.

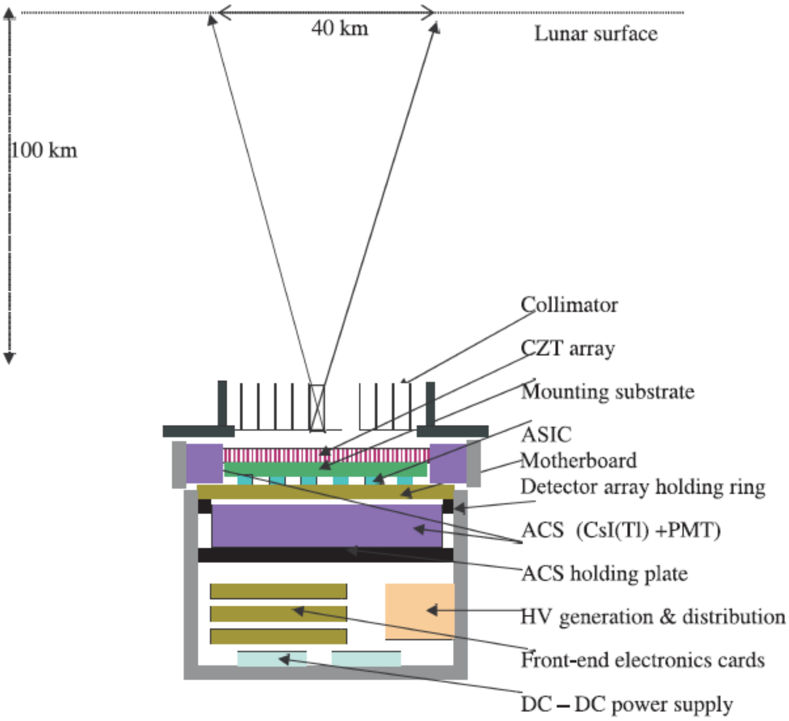


Figure 7-17 HEX Payload Transparent view

A specially designed collimator provides a field of view (FOV) of 33 km X 33 km at the lunar surface from a 100 km orbit. The spatial resolution of HEX is 33 km and the mass is 14.4 kg.

7.2.5.5 Moon impact probe (MIP)

In addition to the primary scientific payloads, an impactor carrying a high sensitive mass spectrometer, a video camera and a radar altimeter was included in this mission. The impact probe of 35 kg mass was attached at the top deck of the main orbiter. The impactor was released at the beginning of the mission after reaching 100 x 100 km lunar polar orbit and allowed to impact in a predetermined location on the lunar surface.

During the descent phase, it is spin-stabilized. The total flying time from release to impact on Moon was around 25 minutes. Apart from the video imaging of the landing site, the onboard mass spectrometer tried to detect possible presence of trace gases in the lunar exosphere.

The primary objective is to demonstrate the technologies required for landing the probe at a desired location on the Moon and to qualify some of the technologies related to future soft landing missions.

Payload Configuration Details:

There were three instruments on the Moon Impact Probe. The dimension of the impact probe is 375 mm x 375 mm x 470 mm

Radar Altimeter – for measurement of altitude of the Moon Impact Probe and for qualifying technologies for future landing missions. This is operating at 4.3 GHz ± 100 MHz.

Video Imaging System – for acquiring images of the surface of the Moon during the descent at a close range. The video imaging system consists of analog CCD camera.

Mass Spectrometer – for measuring the constituents of tenuous lunar atmosphere during descent. This instrument is based on a state-of-the-art, commercially available Quadrupole mass spectrometer with a mass resolution of 0.5 amu and sensitivities to partial pressure of the order of 10^{-14} torr.

MIP System Configuration

The Moon Impact Probe (MIP) essentially made up of honeycomb structure, which housed all the subsystems and instruments. In addition to the instruments, it comprised of, the separation system, the de-boost spin and de-spin motors, the avionics system and thermal control system. The avionics system supported the payloads and provided communication link between MIP and the main orbiter, from separation to impact and provided a database useful for future soft landing.

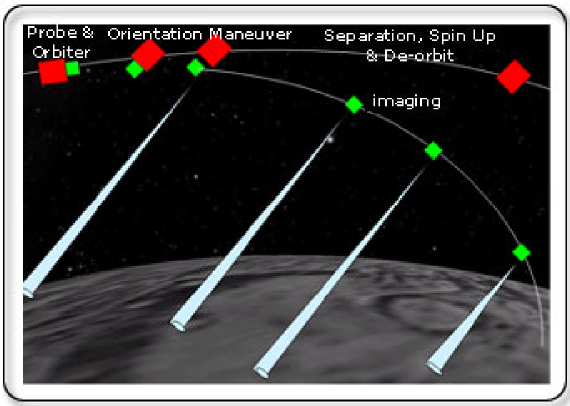


Figure 7-18 Impact Probe Mission Profile

7.2.5.6 Chandrayaan imaging X-ray spectrometer (CIXS)

Two options were considered for detection of low energy (1–10 keV) fluorescence X-rays from lunar surface; use of a thermoelectrically cooled X-ray CCD (LEX) or of a swept-charge X-ray detector (SCD) array. The final choice was SCD and CIXS, a modified version of D-CIXS instrument on board SMART-1, proposed by RAL, UK, and supported by ESA was selected in place of LEX.

This collimated LEX had a field of view of ~30 km and aimed to provide detail chemical mapping of the lunar surface for the elements, Mg, Al and Si and also for Ca, Ti and Fe during solar flare times. An X-ray solar monitor (XSM) is a part of this payload and will continuously monitor the solar X-ray flux essential for analyzing the data on fluorescence X-rays to infer absolute elemental abundance. CIXS is collaborative payload between ISRO and RAL with a group of ISRO scientists and engineers involved in various aspects of payload design and fabrication and detector characterization.

Scientific Objective:

The primary goal of the CIXS instrument is to carry out high quality X-ray spectroscopic mapping of the Moon, in order to constrain solutions to key questions on the origin and evolution of the Moon. CIXS used X-ray fluorescence spectrometry (1.0-10 keV) to measure the elemental abundance, and map the distribution, of the three main rock forming elements: Mg, Al and Si. During periods of enhanced solar activity (solar flares) events, it was planned to determine the abundance of minor elements such as Ca, Ti and Fe on the surface of the Moon.

Payload Configuration Details:

The instrument utilised technologically innovative Swept Charge Device (SCD) X-ray sensors, which were mounted behind low profile gold/copper collimators and aluminium/polycarbonate thin film filters. The system had the virtue of providing superior X-ray detection, spectroscopic and spatial measurement capabilities, while also operating at near room temperature. A deployable proton shield protects the SCDs during passages through the Earth's radiation belts, and from major particle events in the lunar orbit. In order to record the incident solar X-ray flux at the Moon, which is needed to derive absolute lunar elemental surface abundances, CIXS also includes an X-ray Solar Monitor.

The XSM sensor unit:

The X-ray Solar Monitor (XSM) was provided through collaboration between Rutherford Appleton Laboratory (RAL) and University of Helsinki. With its wide field-of-view of ± 52 degrees, XSM provides observation of the solar X-ray spectrum from 1-20 keV with good energy resolution (< 250 keV@5.9 keV) and fast spectral sampling at 16 s intervals. The onboard solar monitor acting in real time will greatly enhance the capability of CIXS to determine absolute elemental abundances as well as their ratios. The total mass of CIXS and XSM is 5.2 kg.

Heritage:

The CIXS instrument was primarily based on the D-CIXS instrument on the ESA SMART-1 mission. The hardware was developed at the Rutherford Appleton Laboratory, UK in collaboration with the ISRO Satellite Centre, Bangalore and exhibits significant improvements over the instrument flown on SMART-1.

7.2.5.7 Near infrared spectrometer (SIR-2)

SIR-2 is an upgraded, compact, monolithic grating, near infrared point spectrometer based on SIR flown on board ESA's SMART-1 mission and covered the wavelength region 0.9–2.4 µm. The instrument has a spectral resolution of 6 nm. It is a linear CCD array based instrument with a resolution of ~80 m per pixel.

Figure 7-19 Near-IR Spectrometer (SIR-2)

Scientific Objectives:

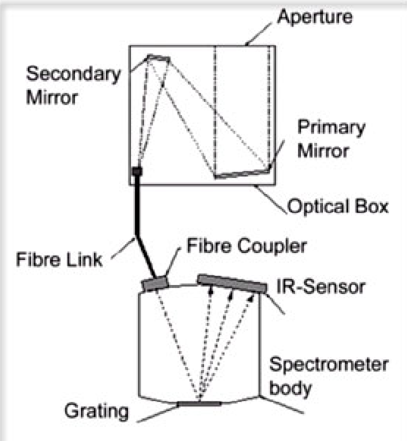
SIR-2 is to address the surface-related aspects of lunar science in the following broad categories:

- Analyse the lunar surface in various geological/mineralogical and topographical units;
- Study the vertical variation in composition of crust;
- Investigate the process of basin Maria and crater formation on the Moon;
- Explore "Space Weathering" processes of the lunar surface;
- Survey mineral lunar resources for future landing sites and exploration.

The determination of the chemical composition of a planet's crust and mantle is one of the important goals of planetary research. Diagnostic absorption bands of various minerals and ices are located in the near-IR range, thus making near-infrared measurements of rocks, particularly, suitable for identifying minerals.

Payload Configuration:

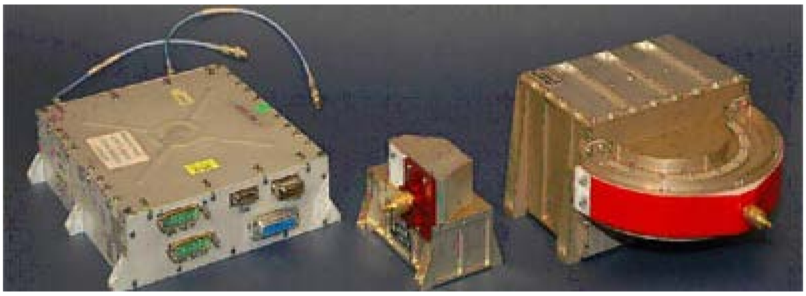
SIR-2 is a grating NIR point spectrometer working in the 0.93-2.4 microns wavelength range with 6 nm spectral resolution. It collects the Sun's light reflected by the Moon with the help of a main and a secondary mirror. This light is fed through an optical fiber to the instrument's sensor head, where it is reflected off a dispersion grating. The dispersed light reaches a detector, which consists of a row of photosensitive pixels that measure the intensity as a function of wavelength and produces an electronic signal, which is read out and processed by the experiment's electronics. The mass of the instrument is 3.3 kg and the instrument unit dimension is 260 mm x 171 mm x 143 mm.



Schematic of the SIR-2 instrument

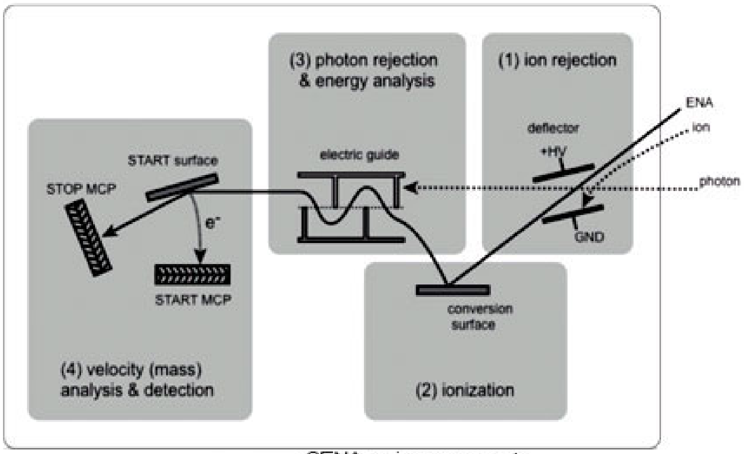
7.2.5.8 Sub-keV atom reflecting analyzer (SARA)

The SARA payload consists of two major subsystems, Chandrayaan-1 low energy neutral atom (CENA) and solar wind monitor (SWIM). CENA detects neutral atom sputtered from the lunar surface by solar wind ions. The CENA sensor has an energy range of 10 eV to 2 keV with an energy resolution of ~50% and can resolve groups of elements such as H, O, Na/Mg group, K/Ca group and Fe. SWIM is a simple ion mass analyzer consisting of a sensor and an energy analyzer that provides information on the energy and mass of the incident solar wind ions. Space Physics Laboratory, Thiruvananthapuram, is responsible for developing the data processing unit.



Scientific Objectives: SARA will image the Moon surface using low energy neutral atoms as diagnostics in the energy range 10 eV - 3.2 keV to address the following scientific objectives:

- Imaging the Moon's surface composition including the permanently shadowed areas and volatile rich areas
- Imaging the solar wind-surface interaction
- Imaging the lunar surface magnetic anomalies
- Studies of space weathering



CENA main components

The Moon does not possess a magnetosphere and atmosphere. Therefore, the solar wind ions directly impinge on the lunar surface, resulting in sputtering and backscattering. The kick-off and neutralized solar wind particles leave the surface mostly as neutral atoms. The notable part of the atoms has energy exceeding the escape energy and thus, such atoms propagate along ballistic trajectories. The SARA instrument is designed to detect such atoms with sufficient angular and mass resolution to address the above scientific objectives. SARA is the first-ever energetic neutral atom imaging mass spectrometer. Payload Configuration Details: The SARA instrument consists of neutral atom sensor CENA (Chandrayaan-1 Energetic Neutrals Analyzer), solar wind monitor SWIM and DPU (Data Processing Unit). CENA and SWIM interface with DPU, which in turn interfaces with the spacecraft. The masses of CENA, SWIM and DPU are 2 kg, 0.5 kg and 2 kg respectively, totaling the SARA mass as 4.5 kg.

Low-energy neutral atoms enter through an electrostatic charged particle deflector (1), which sweeps away ambient charged particles by a static electric field. The incoming low energy neutral atoms are converted to positive ions on an ionization surface (2), and then passed through an electrostatic analyzer of a specific ("wave") shape that provides energy analysis and effectively blocks photons (3). Particles finally enter the detection section (4) where they are reflected at grazing incidence from a start surface towards one of several stop micro channel plate (MCP) detectors. Secondary electrons generated at the start surface and the stop pulses from the stop MCP detectors preserve the direction and the velocity of the incident particle. SWIM is an ion mass analyzer, optimized to provide monitoring of the precipitating ions. Ions first enter the deflector, which provides selection on the azimuth angle, following a cylindrical electrostatic analyzer. Exiting the analyzer the ions are post-accelerated up to 1 keV and enter the time-of-flight cell, where their velocity is determined by the same principle (surface reflection), as in the CENA instrument.

7.2.5.9 Miniature synthetic aperture radar (MINI-SAR)

Multifunction miniature radar consisting of SAR, altimeter, scatterometer and radiometer operating at 2.5 GHz will explore the permanently shadowed areas near lunar poles to look for signature of water ice. The mini-SAR system will transmit right circular polarization (RCP) and receive both left circular polarization (LCP) and RCP. The SAR system has a nominal resolution of 150 m per pixel with 8 km swath.

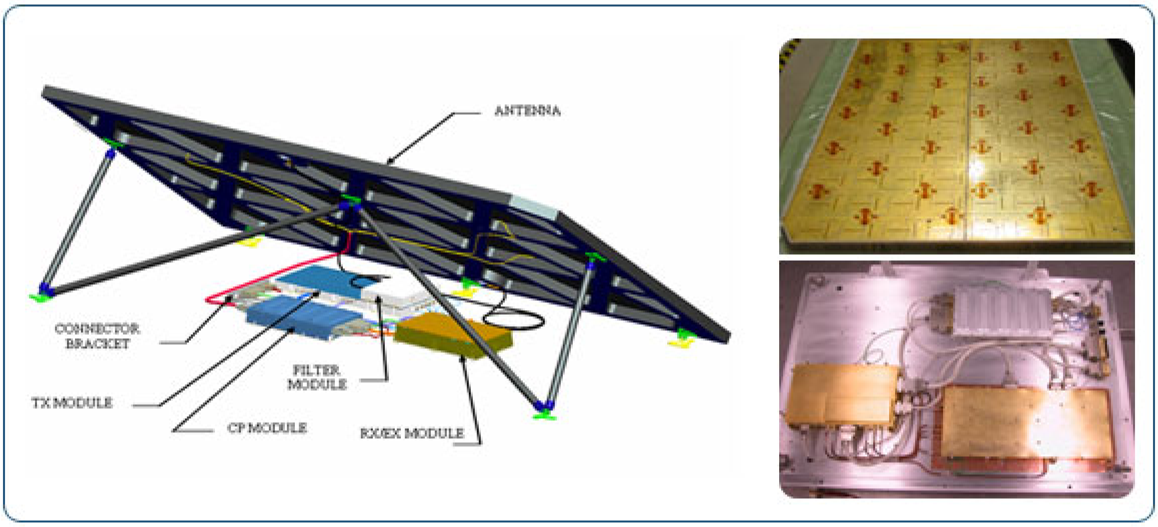


Figure 7-20: Miniature Synthetic Aperture Radar (Mini-SAR)

Scientific Objectives: To detect water ice in the permanently shadowed regions on the Lunar poles, upto a depth of a few meters. Although returned lunar samples (from earlier missions) show the Moon to be extremely dry, recent research suggest that water-ice may exist in the Polar Regions. Because its axis of rotation is perpendicular to the ecliptic plane, the poles of the Moon contain areas that never receive light and are permanently dark. This results in the creation of "cold traps", zones that, because they are never illuminated by the sun, may be as cold as 50–70^{o}K. Cometary debris and meteorites containing water-bearing minerals constantly bombard the Moon. Most of this water is lost to space, but, if a water molecule finds its way into a cold trap, it remains there forever – no physical process is known that can remove it. Over geological time, significant quantities of water could accumulate. An onboard SAR at suitable incidence would allow viewing of all permanently shadowed areas on the Moon, regardless of whether sunlight is available or the angle is not satisfactory. The radar would observe these areas at incidence angle near 45 degrees, recording echoes in both orthogonal senses of received polarization, allowing ice to be optimally distinguished from dry lunar surface. The Mini-SAR radar system can operate as an altimeter/scatterometer, radiometer, and as a synthetic aperture radar imager. Payload Configuration Details: The Mini-SAR system will transmit Right Circular Polarization (RCP) and receive, both Left Circular polarization (LCP) and RCP. In scatterometer mode, the system will measure the RCP and LCP response in the altimetry footprint, along the nadir ground track. In radiometer mode, the system will measure the surface RF emissivity, allowing determination of the near normal incidence Fresnel reflectivity. Meter-scale surface roughness and circular polarization ratio (CPR) will also be determined for this footprint. This allows the characterization of the radar and physical properties of the lunar surface (e.g., dielectric constant, porosity) for a network of points. When directed off nadir, the radar system will image a swath parallel to the orbital track by delay/Doppler methods (SAR mode) in both RCP and LCP. The synthetic aperture radar system works at a frequency 2.38 GHz, with a resolution of 75 m per pixel from 100 km orbit and its mass is 8.77 kg.

7.2.5.10Radiation dose monitor (RADOM)

RADOM is a miniature spectrometer–dosimeter that uses a semiconductor detector and measure the deposited energy from primary and secondary particles using a 256 channel pulse analyzer. The deposited energy spectrum can then be converted to deposited dose and incident flux of charged particles on the silicon detector.

Scientific Objectives:

RADOM will qualitatively and quantitatively characterise the radiation environment in near lunar space, in terms of particle flux, dose rate and deposited energy spectrum. The specific objectives are to

- Measure the particle flux, deposited energy spectrum, accumulated radiation dose rates in Lunar orbit;
- Provide an estimate of the radiation dose around the Moon at different altitudes and latitudes;
- Study the radiation hazards during the Moon exploration. Data obtained will be used for the evaluation of the radiation environment and the radiation shielding requirements of future manned Moon missions.

Radiation exposure of crew members on future manned space flight had been recognised as an important factor for the planning and designing of such missions. Indeed, the effects of ionising radiation on crew health, performance and life expectancy are a limitation to the duration of man's sojourn in space. Predicting the effects of radiation on humans during a long-duration space mission requires i) accurate knowledge and modeling of the space radiation environment, ii) calculation of primary and secondary particle transport through shielding materials and through the human body, and iii) assessment of the biological effects of the dose.

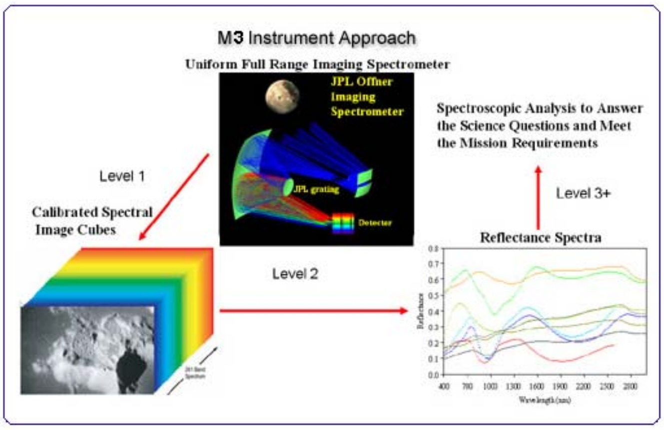
The general purpose of RADOM is to study the radiation hazards during the Moon exploration. Data obtained will be used for the evaluation of radiation environment and radiation shielding requirements for future manned lunar missions.

Payload Configuration Details:

RADOM is a miniature spectrometer-dosimeter containing one semiconductor detector of 0.3 mm thickness, one charge-sensitive preamplifier and two micro controllers. The detector weighs 139.8 mg. Pulse analysis technique is used for obtaining the deposited energy spectrum, which is further converted to the deposited dose and flux in the silicon detector. The exposure time for one spectrum is fixed at 30 s. The RADOM spectrometer will measure the spectrum of the deposited energy from primary and secondary particles in 256 channels. RADOM mass is 160 g.

7.2.5.11Moon Mineral Mapper (MMM)

The MMM (*M*3) is a high throughput push broom imaging spectrometer operating in 0.7–3.0 μm range with high spatial (70 m per pixel) and spectral (10 nm sampling) resolution. It will have a swath of 40 km. It uses a 2D HgCdTe detector array for measuring reflected solar energy in the above wavelength range.



High-resolution compositional maps by Moon Mineralogy Mapper will improve the understanding of the early evolution of a differentiated planetary body and provide a high-resolution assessment of lunar resources.

Scientific Objectives:

The primary Science goal of M3 is to characterize and map lunar surface mineralogy in the context of lunar geologic evolution. This translates into several sub-topics relating to understanding the highland crust, basaltic volcanism, impact craters, and potential volatiles.

The primary exploration goal is to assess and map lunar mineral resources at high spatial resolution to support planning for future, targeted missions. These M3 goals translate directly into the following requirements:

- Accurate measurement of diagnostic absorption features of rocks and minerals;
- High spectral resolution for deconvolution into mineral components;
- High spatial resolution for assessment geologic context and active processes;

Payload Configuration Details:

The M3 scientific instrument is a high throughput pushbroom imaging spectrometer, operating in 0.7 to 3.0 μm range. It measures solar reflected energy, using a two-dimensional HgCdTe detector array featuring.

Sampling: 10 nanometers
Spatial resolution: 70 m/pixel [from 100 km orbit]
Field of View: 40 km [from 100 km orbit]
Mass : 8.2 kg

The spectral range 0.7 to 2.6 μm captures the absorption bands for the most important lunar minerals. In addition, the spectral range 2.5 to 3.0 μm is critical for detection of possible volatiles near the lunar poles. The presence of small amounts of OH or H_2O can be unambiguously identified from fundamental absorptions that occur near 3000 nm. M3 measurements are obtained for 640 cross track spatial elements and 261 spectral elements. This translates to 70 m/pixel spatial resolution and 10 nm spectral resolutions (continuous) from a nominal 100 km polar orbit for Chandrayaan-1. The M3 FOV is 40 km in order to allow contiguous orbit-to-orbit measurements at the equator that will minimize lighting condition variations.

7.3 Mars Orbiter Mission (MOM)

7.3.1 Introduction

Mars Orbiter Mission is ISRO's first interplanetary mission to planet Mars with an orbiter craft designed to orbit Mars in an elliptical orbit. The Mission is primarily a technological mission considering the critical mission operations and stringent requirements on propulsion and other bus systems of the spacecraft. It is configured to carry out observation of physical features of Mars and carry out limited study of Martian atmosphere with five payloads finalized by Advisor Committee on Space Sciences (ADCOS).

7.3.2 Mission Objectives

One of the main objectives of the first Indian mission to Mars is to develop the technologies required for design, planning, management and operations of an interplanetary mission. Following are the major objectives of the mission.

7.3.2.1 Technological objectives:

- Design and realization of a Mars orbiter with a capability to survive and perform earth bound maneuvers, cruise phase of 300 days of travel, Mars orbit insertion/capture and on-orbit phase around Mars.

- Deep space communication, navigation, mission planning and management.
- Incorporate autonomous features to handle contingency situations

7.3.2.2 Scientific objectives:

- Exploration of Mars surface features, morphology, topography, mineralogy and Martian atmosphere by indigenously developed scientific instruments.

7.3.3 Mission Phases

Mission planning is done in conjunction with the defined mission objectives. The Mars Mission can be envisaged as a rendezvous problem, wherein the spacecraft/Mars orbiter is maneuvered to a departure hyperbolic trajectory, escapes the Earth sphere of influence (SOI) and at some epoch enters the SOI of Mars (taking into account the asynchronous movement of Earth & Mars in their respective orbits around the Sun). The SOI of earth extends up to 9, 25,000 km from the surface of the earth beyond which the perturbing force on the orbiter is due to the sun only. The orbiter cruises in its elliptical interplanetary cruise trajectory till it encounters SOI of Mars (around 5, 80,000 km from the surface of Mars).

The optimal launch opportunity is a function of the trans-Mars delta-V, the MOI delta-V, the declination of the departure V-infinity and launcher constraints. The delta-V values vary with the departure date and the transfer time since orbits of both Earth and Mars are eccentric. Interplanetary transfer trajectory geometry requires that the declination of the departure asymptote (departure V-infinity direction) must be less than or equal to the initial parking orbit inclination. The optimum launch opportunity is that which requires the minimum total delta-v for achieving the final Martian orbit satisfying the parking orbit inclination constraint.

The targeted Martian orbit by PSLV launcher with achievable MOI delta-V & taking into consideration the payload requirements is a highly elliptic orbit with Periareion altitude of 366 km and Apoareion altitude of 80000 km.

The Mars Orbiter Mission phases are classified as follows:

- Earth centric phase
- Heliocentric phase
- Mars centric phase

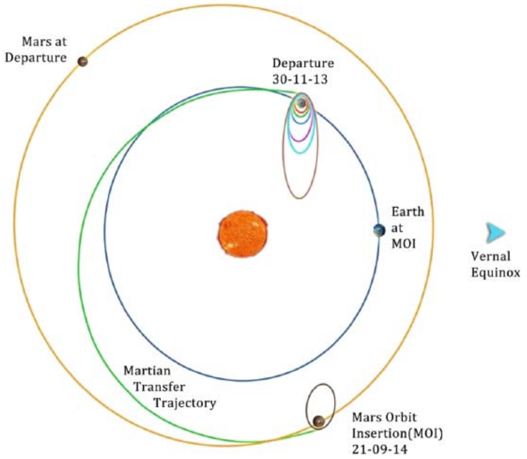


Figure 7-21 Typical trajectory of Mars Orbiter Mission

7.3.4 Scientific Instruments (Payloads)

Based on the overall science objectives and considering the constraints of the orbit, 5 experiments have been short-listed. The list of payloads along with objectives are summarized in the table 7-7.

Table 7-7 MOM Payloads

Science Theme	Payload	Primary objective
Atmospheric studies	LAP	Measure D/H
	MSM	CH4 detection
Plasma and Particle environment studies	MENCA	Map neutral in exosphere
Surface Imaging studies	MCC	Optical imaging
	TIS	Thermal Imaging

7.3.4.1 Lyman Alpha Photometer (LAP)

Lyman Alpha Photometer (**LAP**) is an absorption cell photometer. It measures the relative abundance of deuterium and hydrogen from Lymanalpha emission in the Martian upper atmosphere (typically Exosphere and Exobase). Measurement of D/H (Deuterium to Hydrogen abundance Ratio) allows us to understand the loss process of water from the planet.

Objectives of Instrument:

- Estimation of D La (@ 121.53 nm)/H La (@ 121.56 nm) ratio.
- Estimation of escape flux of H2 corona.
- Generation of Hydrogen and Deuterium coronal profiles.

Instrument description: The payload achieves its objectives using two narrowly spaced spectral lines. Since the separation between the two lines is only 0.033nm, an extremely high-resolution spectroscopic technique is necessary to separate the two lines. A standard ultraviolet spectrometer with a diffraction grating is not applicable to spacecraft measurement because of its large size and heavy weight. An optimal approach is to use the absorption cell photometer (consisting of separate hydrogen/deuterium absorption cells) by virtue of their small size, light weight and small power consumption.

LAP instrument consists of a UV detector equipped with gas filled Hydrogen and Deuterium cells, a filter, and a light baffle. The hydrogen and deuterium cells are resonance absorption cells filled with pure molecular hydrogen and deuterium respectively and contain tungsten filaments. These two cells are located between an objective lens and a detector.

When power is applied to the filaments, the hydrogen or deuterium molecules thermally dissociate into atoms that absorb hydrogen or deuterium Lyman-α incident upon the cells. The absorption cells can be regarded as a very narrow-band rejection filter for H and D Lyman-α lines with a FWHM (full width half maximum) of 0.002–0.004nm. By switching the filament turned on in the hydrogen cell and that in the deuterium cell alternately, one can measure both intensities of H and D Lyman-α emissions separately.

Figure 7-22 Configuration of LAP Payload

7.3.4.2 Methane Sensor for Mars (MSM)

MSM is designed to measure CH4 in the Martian atmosphere with ppb accuracy and map its sources. Data is acquired only over illuminated scene as the sensor measures reflected solar radiation. Since concentration of methane in the Martian atmosphere undergoes spatial and temporal variations it is necessary that global data is collected during every orbit. To retrieve methane concentration, the measured data has to be corrected for atmospheric effects. Better retrieval accuracy is obtained when atmosphere is clear. Since Mars atmosphere is so dynamic and turbulent with frequent dust storms, it is required that MSM data is acquired as often as possible since the possibility of acquiring a data set under clear atmospheric conditions at optimum illumination and viewing geometries become more.

Objective:

- To measure CH_4 in the Martial atmosphere with ppb accuracy and for mapping its sources. It may also give useful information about the origin of CH_4 whether it is biogenic or volcanic.

Instrument Description: MSM is based on Fabry-Perot (FP) etalon filters that work on the principle of multiple beam interferometry. FP etalon provides extremely narrow, evenly spaced transmission peaks within the desired spectral band when the condition of resonance is satisfied. The frequency of the spectral lines of the FP etalon proposed for MSM is chosen to coincide with the absorption spectra of the methane (CH4) gas in the specified spectral range. The input parallel beam is focused by the fore optics at the focal plane where the field stop is located (Figure 2.2). The collimator gives a collimated output which gets incident on the band pass filter subsequent to that. The collimated beam within the specified spectral range then falls on the beam splitter followed by the FP etalon. The transmitted and the reflected light from the beam splitter in conjunction with the etalon form the transmission peaks for the main CH4 and reference channel respectively. Output from the etalon is focused by the focusing optics on to an InGaAs detector.

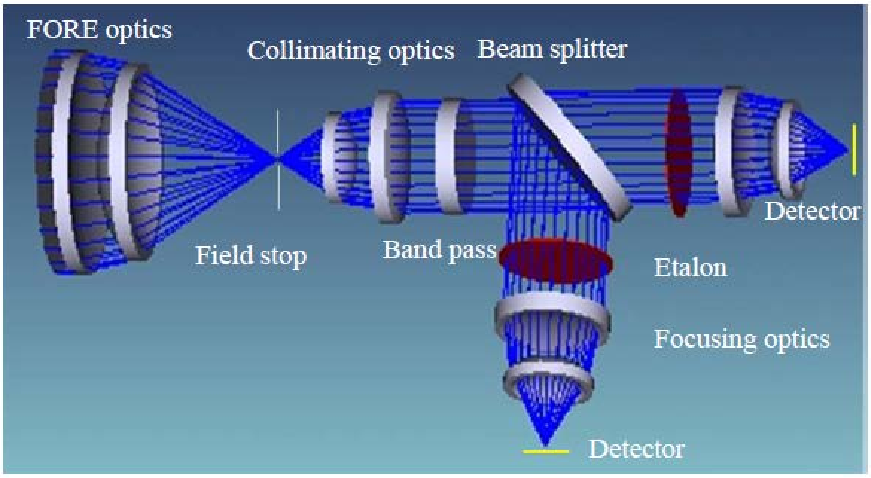


Figure 7-23 Configuration of Methane sensor for Mars (MSM)

Table 7-8 Specification of MSM

	CO_2 Channel	CH_4 Channel
Spectral Region	6408-6424cm^{-1}	6030-6090cm^{-1}
Fore Optics	Aperture=5cm, IFOV=5mrad	
FPE Filter	FSR:1.5821cm^{-1}, Finesse: 14.05	FSR : 9.995cm^{-1}, Finesse: 7.24
FPE Temp	~320K CO_2 l, ~325K Ref.	~320K CH_4, ~325K Ref
Detector	InGaAs diode, 500µm pixel	
Estimation Accuracy	<0.5 %	<1 ppb
Radiometric Quantization	14 bit	14 bit
Payload Weight	~ 3 kg	
Power & Data rate	~7.5W and < 1kpbs	

7.3.4.3 Mars Exospheric Neutral Composition Explorer (MENCA)

The MENCA (Mars Exospheric Neutral Composition Analyser) is a quadrupole mass spectrometer based payload, shortlisted for the first Indian orbiter mission to Mars. The instrument is capable of analyzing the neutral composition in the range of 1 to 300 amu with unit mass resolution. MENCA draws its heritage from the CHACE (Chandra's Altitudinal

Composition Explorer) payload flown aboard the Moon Impact Probe (MIP) in Chandrayaan-1 mission.

The low inclination of spacecraft orbit selected for the Mars mission provides an ideal opportunity to study the neutral density distribution around Phobos (the Martian closest satellite).

Objective:

- Scientific objective of the proposal is to study the neutral composition of the Martian upper atmosphere, as a function of time to understand diurnal and seasonal variations, from an orbiter around Mars.
- The temperature profile can be derived from the structure of the concentration of the neutral species to be obtained during the approach of the spacecraft from the apoapsis to the periapsis.

Instrument description: The basic instrument is a Quadrupole mass spectrometer, which operates based on path stability of ionized species from the ion source (the ambient neutrals are ionized upon impact by accelerated thermionic electrons) to a detector unit, through an assembly of a set of four conductive rods, collectively referred to as "quadrupoles". The rods are maintained at a dynamically changing relative potential condition. Mass-filtering takes place in the quadrupole rod region based on the specific charges (q/m) of the ions. At a given potential condition of the rods, a particular species (characterized by a given specific charge, m/q), is allowed to reach the detector unit. Ions of the other species, however, have unstable paths, get collected at the walls of the conductors and are neutralized.

The detector system is a combination of a Faraday cup and a channel electron multiplier. Incorporation of the electron multiplier enhances the sensitivity of the system in detecting species of very low partial pressure, since it has an inherent gain ranging between 10^3 and 10^5. The detector will respond to the transmitted ion current corresponding to a given species in terms of an analogue output current, which will be the measure of the relative abundance of that species. The analogue current will then be converted to voltage, signal-conditioned, digitized and processed within the electronics, and finally sent to the satellite telemetry unit.

The instrument will consist of two basic parts, viz. The sensor probe and the electronics. While the sensor probe houses the ion source, ion optics, the quadrupole mass filter and the detector assembly; the electronics will contain the control for the operation of the mass spectrometer, data acquisition, and telemetry and telecommand operations.

The instrument can be regarded as three pressure measuring devices all housed in a single instrument. They can measure total pressures from 760 Torr to 10^{-9}Torr with its built-in Pirani and Ion gauge with Bayard-Alpert geometry. The mass spectrometer operation, however, requires the total pressure to be less than 10^{-4}Torr. The sensor probe consists of an ionizer, quadrupole mass filter and detector assembly. The top of the sensor probe acts not only as a high efficiency ionizer for the mass filter but also as a high quality pressure gauge making use of the same emission circuit. Hence the instrument can measure total pressure starting from 760 Torr to 10^{-9}Torr through the Pirani and ionization gauges and extends the range to the partial pressure through the mass analyzer (when the total pressure is below 10-

^{4}Torr). The standard Faraday cup detector allows partial pressure measurements from 10^{-4} to 10^{-11}Torr, while the channel electron multiplier detector can detect partial pressures down to 10^{-14}Torr.

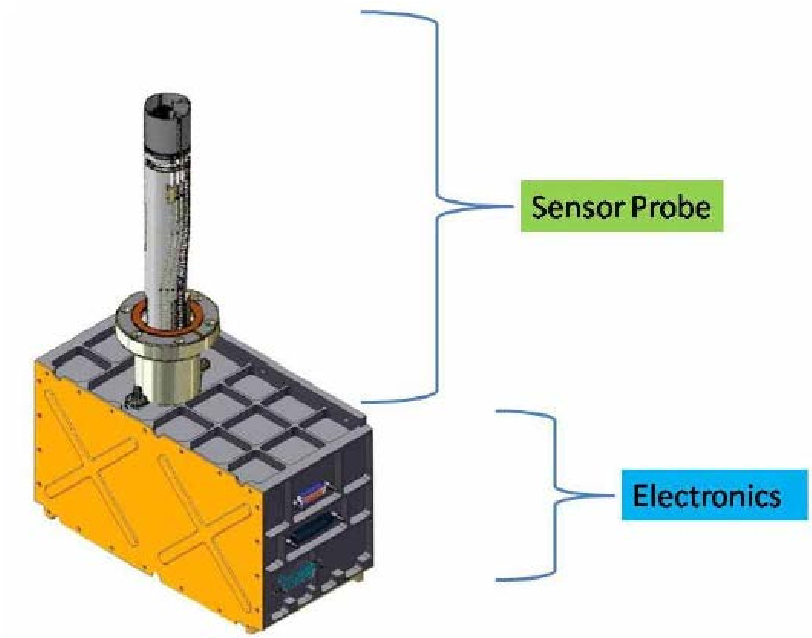


Figure 7-24 Configuration of MENCA Payload

Table 7-9 Specifications of MENCA

Parameter	Value
Mass range	1-300amu (Programmable)
Detector type	Channel Electron multiplier (along with Faraday cup)
Mass filter type	Quadrupole
Mass Resolution	Unit Mass
Dynamic range	10^{10} (with electron multiplier)
Minimum detectable partial pressure	~ 10^{-14}torr
Operating pressure	Atmosphere to UHV (ion gauge operation below 10^{-3}torr and RGA operation below 10^{-4}torr)
Power requirement	~ 35.0W (normal mode) ; 50.0 W (degas mode)
Weight	4 kg (inclusive of the DC/DC converter)
Number of packages	2 (Two) 1. MENCA payload (Includes sensor probe with electronics) 2. MENCA DC-DC converter
Overall dimensions of the instrument (without DC/DC converter)	Mechanical hardware for Electronics: 235 mm (with connector mounts) X 117mm (including screw head) X 144.5 mm ; the cylindrical sensor probe is of diameter 36 mm and length 230mm.

Footprint on the mounting plane	235 mm (without connector mount 231 mm) horizontally x 144.5 mm vertically (for the hardware of the electronics) 133.5mm X 95mm (for the probe support)
Thermal Constraints	Electronics operating temperature to be maintained between 10°C and 50°C

7.3.4.4 Mars Colour Camera (MCC)

Tri-colour MCC images give useful inputs about the surface features and composition of Martian surface. They are useful to monitor the dynamic events and weather of Mars. MCC is also useful for imaging the surface of the two satellites of the Mars – Phobos and Deimos. Moreover MCC provides the context information for other science payloads. So MCC images are to be acquired whenever MSM and TIS data is acquired.

Objective:

- To image the morphology of Martian surface with high geometrical fidelity.
- To map the morphological units, landforms, geological structures, craters, etc.
- To map Martian polar ice caps and its dynamic behaviour through seasonal variations.
- To observe and study events like dust storms, dust devils etc.
- To image Mars, its moons, asteroids and other celestial bodies from close quarters.
- To provide useful inputs for attitude determination, control and navigation of the spacecraft.
- To provide context information for other science payloads.

Instrument Description: Mars Frame Camera is a moderate resolution panchromatic camera. Since entire scene is acquired simultaneously in a Frame camera, the image is least affected by spacecraft attitude rates. Elliptical orbit of Mars mission allows imaging of localized scenes at high spatial resolution as well as synoptic view of the full globe.

To take images with a GIFOV of 25m with a frame size of 53.6 km x 53.6 km from perigee and cover the full mars disc from apogee, MCC employs a multi-element lens assembly and a 2K×2K area array detector with RGB Bayer pattern. The f/4 lens has a focal length of 105mm with a circularly symmetric field of view of ± 4.4°. The proposed detector has 2048 x 2048 elements on a pixel pitch of 5.5µ. The sensor will be driven by a custom-built electronics designed around the proposed detector. Camera electronics generates bias for detector, clock generation, data pre-processing and generation of biases from raw bus. The raw data volume is 40Mb/Frame from which Colour image will be generated at ground using standard demosaicing.

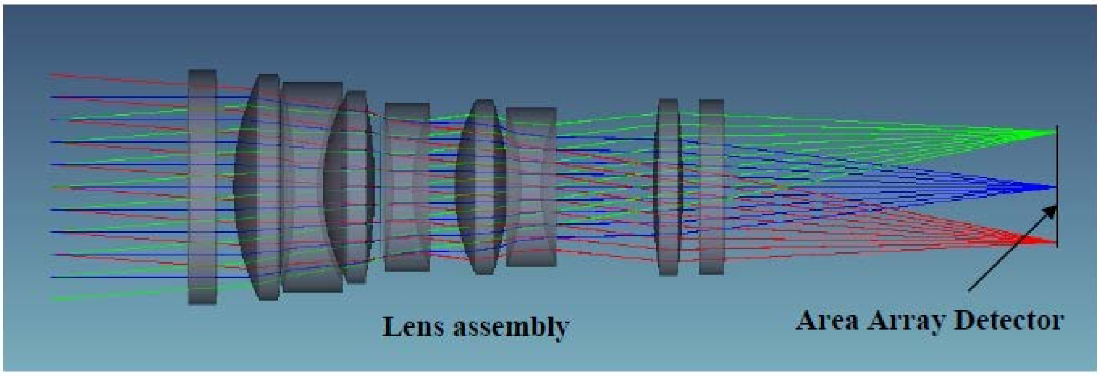


Figure 7-25 Configuration of MCC Payload

Table 7-10 Specifications of MCC

Parameter	Value
GIFOV	25m from 500 km; 4 km from 80000 km
Frame Size	53.6km x 53.6km from perigee Full Disc from Apogee
Spectral Band	Option:1 Panchromatic Band Option:2 True Color Imaging using Bayer Pattern detector
Optics	F=110mm, f/4, Field= ±4.2°
Detector	2K×2K Array, 5.5µm pixel, RGB Bayer Pattern
SNR	> 250 @ 19 % albedo &θ_{sun}=0°
Radiometric Quantization	10 bit
Data Volume/Frame	40Mb
Payload Weight	~ 1.5 kg
Power	~ 4W

7.3.4.5 TIR Imaging spectrometer (TIRIS)

Thermal Imaging Spectrometer (TIS) is one of the proposed instruments on the first Indian Mission to Mars for surface and atmospheric exploration using thermal remote sensing. Thermal Imaging Spectrometer would detect thermal emitted radiation from Martian environment in 7-14 µm thermal infrared regions using micro bolometer device.

Specific absorption features of surface mineral composition manifest in thermal emission spectra of any planet. Precise detection of specific spectroscopic features allows estimating the surface composition. Thermal imaging spectroscopy will allow studying the aerosol and its variability.

<u>Objective:</u>

- To map surface composition and mineralogy of Mars.
- To detect hot spots which indicate underground hydro thermal systems
- To detect and study the variability of aerosol/dust in Martian atmosphere.

Determining the mineralogy of Mars is an essential part of revealing the conditions of the surface and subsurface. Thermal Imaging Spectrometer will be useful in mapping these important mineral composition using spectroscopic techniques. Thermal Imaging Spectrometer data will be used for assessment aerosol turbidity in Martian atmosphere.

Instrument description: The sensor is a Thermal Infrared (TIR) grating spectrometer which makes use of an un-cooled micro-bolometer array as the detector. Unlike photon detectors like MCT which operate at cryogenic temperatures, micro bolometer does not require any cooling. So, payload size and weight can be considerably less. Also, issues associated with reliability of cryogenic coolers are absent. These factors make them ideal for low cost micro-satellite missions. But, signal to noise characteristics of micro-bolometer detectors (in terms of specific detectivity) is about two orders of magnitude less than that of MCTs. So, it is not expected to give high measurement accuracy at high spatial and spectral resolutions.

Optical configuration of TIS consists of fore optics, slit, collimating optics, grating and reimaging optics. A 120x160 element bolometer array is used as the detector. The longer axis of the detector is aligned in the cross-track direction. So, 160 detector elements in the cross-track direction defines the swath coverage of the sensor while 120 elements in the along track direction defines the spectral range. By binning pixels in the along track direction, it is possible to manipulate spectral sampling interval as well as number of spectral bands. Similarly by binning pixels in the cross-track direction, different spatial sampling intervals (GSD) can be selected.

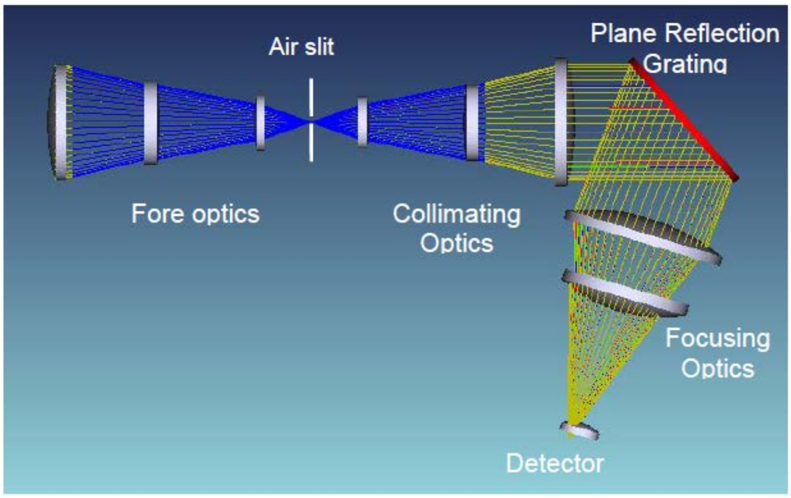


Figure 7-26 Configuration TIS Payload

Table 7-11 Specifications of TIR imaging spectrometer

Parameter	Value
GIFOV /GSD	200m-4000m (Programmable)
Type of Sensor	Imaging Spectrometer
Swath Coverage	64 km
Spectral Range	7µm – 14µm
Spectral Resolution / Sampling	25nm ×N (Programmable, N=1,2,..240)
No. of bands	Programmable, Max: 240
OPTICS	Refractive telescope, Slit, Collimator, Grating, Re-imaging optics

Aperture	6 cm
F-number	1
DETECTOR	120×160 μ-bolometer NedT = 100mK @ 50Hz, Fn=1 @300K
Quantization	14 bits
Weight	~ 4.5 kg
Power	~ 6W

7.3.5 MOM Configuration

The spacecraft configuration is derived from Chandrayaan-1, which is a balanced mix of design from flight proven IRS/INSAT bus. Compared to Chandrayaan-1, the reliability of the design is enhanced with redundant features. Modification required for Mars mission are in the areas of communication and power, propulsion system (mainly related to liquid engine restart after nearly a year) and mechanisms.



Figure 7-27 Views of Mars Orbiter Mission Spacecraft

The 390litres capacity propellant tanks used for Chandrayaan-1 accommodate a maximum of 850 kg of propellant which is adequate for the proposed Mars mission. A Liquid Engine of 440N thrust is planned to be used for orbit raising and Martian Orbit Insertion (MOI). Additional flow lines and valves have been incorporated to ensure LE 440N engine restart after 300 days of Martian Transfer Trajectory (MTT) cruise and to take care of fuel migration issues. 8 numbers of 22N thrusters are used for wheel desaturation and attitude control during manuevers. Accelerometer is used for measuring the precise incremental velocity (ΔV) and for precise burn termination. Star sensors and gyros provide the attitude control signals in all phases of mission.

Chandrayaan-1 required a single solar panel (1800 X 2150mm). However, to compensate for the lower solar irradiance (50% compared to Earth), the Mars orbiter would require three solar panels of size 1400 X 1800mm. Single 36AH Li-Ion battery (similar to Chandrayaan-1) is sufficient to take care of eclipses encountered during Earth bound phase and in Mars orbit.

The communication dish antenna is fixed to spacecraft body. The antenna diameter is 2.2m which is arrived after the trade-off study between antenna diameter and accommodation within the PSLV-XL envelope. Onboard autonomy functions are planned as the large Earth-Mars distance does not permit real time interventions. This will also takes care of on-board contingencies.

7.4 AstroSat (Astronomy Satellite)

7.4.1 Introduction

Astrosat is India's first dedicated astronomy mission, a broad spectral band Indian national space observatory. Astrosat will provide an opportunity for the Indian astronomers to carry out cutting-edge research in the frontier areas of X-ray and ultraviolet astronomy and allow them to address some of the outstanding problems in modern astrophysics.

7.4.2 Mission Objectives

- **Multi-wavelength observations:** for a wide variety of both Galactic and extra-galactic source types [AGN (Active Galactic Nuclei), binaries, flaring stars, SNRs, clusters]. Use of five co-aligned telescopes simultaneously cover the hard X-ray to visible bands
- **Broadband X-ray spectral measurements:** Emission and absorption features with medium energy resolution capability in the 0.3 – 100 keV spectral band with 3 co-aligned X-ray instruments.
- **High time-resolution studies:** Periodic, aperiodic and chaotic X-ray variability in X-ray binaries. Detect new accreting milli-sec binaries and AXPs. Study evolution of pulse and orbital periods.

Astrosat is expected to focus on high-resolution UV imaging for morphological studies of galactic and extragalactic objects, broad-band studies of X-ray sources and other multi wavelength targets ranging from nearby stars to the very distant active galactic nuclei.

Astrosat is a collaborative project of the following institutions:

- TIFR (Tata Institute of Fundamental Research), Mumbai
- ISRO (Indian Space Research Organization), Bangalore
- IIA (Tata Institute of Fundamental Research), Bangalore
- IUCAA (Inter-University Centre for Astronomy & Astrophysics), Pune
- RRI (Raman Research Institute), Bangalore
- Physical Research Laboratory, Ahmedabad
- CSA (Canadian Space Agency), Canada
- Leicester University, U.K.
- Participation of many Indian Universities and research centers.

The IUCAA (Inter-University Centre for Astronomy and Astrophysics) is an autonomous institution set up by the University Grants Commission to promote nucleation and growth of active groups in astronomy and astrophysics in Indian universities. IUCAA is located in the University of Pune campus next to the National Centre for Radio Astrophysics, which operates the Giant Meter-wave Radio Telescope.

7.4.3 Orbit Details

Launch: A launch of AstroSat is launched on 28th September 2015.
Orbit: Near-equatorial orbit, altitude = 650 km, inclination = ~ 8º, period = 100 minutes.

The LAPAN-A2 microsatellite of LAPAN, Indonesia, is a secondary payload on this flight.

7.4.4 Salient Features of Astrosat:

The spacecraft bus configuration and design have heritage and are similar to the ones earlier used IRS bus. A BMU (Bus Management Unit), similar to the one used in CartoSat-2, is selected for the integrated main bus functions including AOCS, command processing, housekeeping telemetry, sensor processing and antenna position processing.

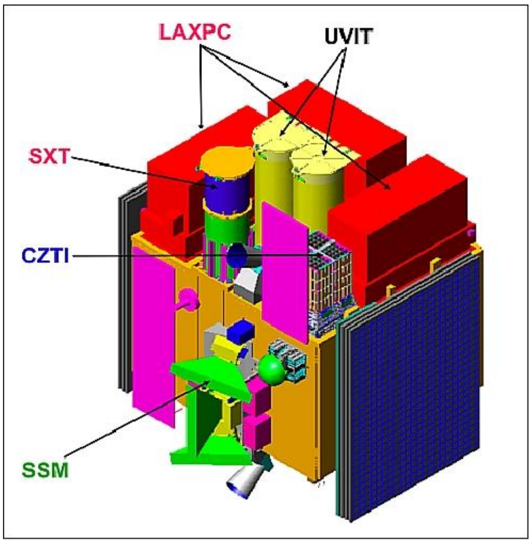


Figure 7-28 Illustration of the AstroSat spacecraft and its instruments

EPS (Electrical Power Subsystem): Two deployable solar panels with single axis rotation are used for power generation. During the full orbit, except for the eclipse period, the panels are always oriented normal to the sun in order to generate maximum power. Whenever the stellar orientation is changed the panels are reoriented. The EPS provides a power of 1250 W, the required payload power is 488 W.

AOCS (Attitude and Orbit Control Subsystem): The spacecraft is 3-axis stabilized. The attitude is sensed with two star sensors and three gyros to provide 1 arcsec pointing capability. Actuation is provided by reaction wheels and magnetic torquers for momentum dumping. The pointing accuracy is < 0.05º.

RF communications: X-band downlink of payload data at a rate of 105 Mbit/s. A solid state recorder with 120 Gbit storage capacity is used for onboard storage of data.

The AstroSat spacecraft has a launch mass of ~ 1808 kg, including 868 kg of payload mass. The expected operating life time of the satellite is five years.

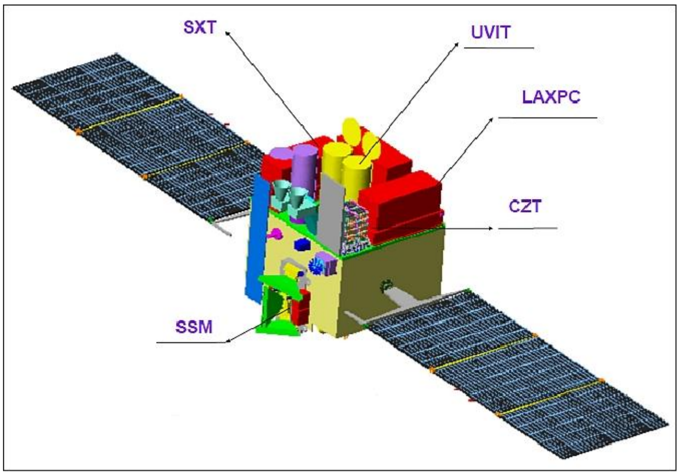


Figure 2: Deployed View of AstroSat spacecraft

7.4.5 Astrosat Payloads

AstroSat carries four co aligned astronomy payloads for simultaneous multi-band observations and one ultraviolet instrument with two telescopes. In addition, a CPM (Charged Particle Monitor) is installed for the control and operation of the sensor complement.

Table 7-12 Overview of instrument parameters

Parameter/ Instrument	UVIT/OPT	SXT	LAXPC	CZTI	SSM
Detector	UV: photon counting CCD; Opt: CCD photometer	X-ray CCD (at the local plane)	Proportional counter	CdZnTe detector array	Position sensitive proportional counter
Imaging property	imaging	imaging	non-imaging	Imaging (<100 keV)	imaging
Optics	Twin Ritchey-Chretien 2 mirror system	Conical foil (Wolter-I mirrors)	Collimator	2D coded mask	1D coded mask
Bandwidth	130-320 nm	0.3-8 keV	3-100 keV	10-150 keV	2-10 keV
Geometric area	1250 cm^2	250 cm^2	10,800 cm^2	1000 cm^2	180 cm^2
Effective area (cm^2)	60 (depends on filter)	125 @ 0.5 keV 200 @ 1-2 keV 25 @ 6 keV	6000 @ 5-30 keV	500(<100 keV) 1000 (>100 keV)	~40 @ 2 keV 90 @ 5 keV (Xe gas)

FOV	0.50º diameter	0.35º (FWHM)	1º x 1º	6 x 6 (<100 keV) 17º x 17º (>100 keV)	
Energy resolution	< 100 (depends on filter)	2% @ 6 keV	9% @ 22keV	5% @ 10 keV	19% @ 6 keV
Angular resolution	1.8 arcsec	3-4 arcmin (HPD)	1-5 arcmin in scam mode only	8 arcmin	~10 arcmin
Time resolution	10 ms	2.6 s, 0.3 s, 1 ms	10 µs	1 ms	1 ms
Typical obs. time/target	30 min	0.5-1 day	1-2 days	2 days	5 min
Sensitivity (obs. time)	21st magnitude (5σ) (1800 s)	10 µCrab (5σ) (10000 s)	0.1 mCrab (3σ)(1000 s)	0.5mCrab (3σ) (1000 s)	~30mCrab (3σ) (300 s)

7.4.5.1 UVIT (Ultraviolet Imaging Telescopes):

The UVIT instrument is a collaboration between ISRO and the Canadian Space Agency (CSA), a contract was signed in 2004. The NRC-HIA (National Research Council Canada - Herzberg Institute of Astrophysics) provides scientific and technical expertise with funding from CSA. Canada is providing the UV photon counting detector subsystem for UVIT.

The objective of UVIT is to perform imaging simultaneously in three channels: 130-180 nm, 180-300 nm, and 320-530 nm. The FOV (Field of View) is a circle of ~ 28 arcmin diameter, the angular resolution is 1.8 arcsec for the ultraviolet channels and 2.5 arcsec for the visible channel. In each of the three channels a spectral band can be selected through a set of filters mounted on a wheel; in addition, for the two ultraviolet channels, a grating can be selected in the wheel to do slit-less spectroscopy with a resolution of ~ 100 cm^{-1}.

The instrument comprises two telescopes: one is for the FUV (130-180 nm) channel, and the other is for simultaneous imaging in the NUV (180-300 nm) & VIS (320-530 nm) channels. Each of the two telescopes is a f/12 Ritchey-Chretien combination with a primary mirror of ~380 mm diameter and a plate scale of ~ 24 µm/arcsec.

The 3 channels use MCP (Microchannel Plate)-based intensified CMOS imaging detectors for the recording of imagery in either (high gain) photon counting mode or in (low gain) integrating mode in which individual photons cannot be distinguished. Typically, the photon counting mode is used for the two ultraviolet channels which have a small flux, while the integration mode is used for the VIS channel which has a high flux. Special attention has been paid to minimize the photo-cathode/MCP gap to get a spatial resolution of ~ 25 µm FWHM Full Width Half Maximum), i.e. ~ 1 arcsec on the plate scale of the telescopes, in the photon counting mode.

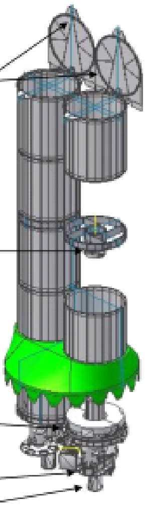


Figure 7-29 Configuration of the UVIT Assembly of two telescopes

The UV images are typically taken at ~ 30 frames/s; for specific observations, depending on the size of the selected field, images of a partial field can be taken up to a rate of 200 frames/s. The time of each frame can be tracked to an absolute accuracy of 5 ms.

The effective area of the telescope depends on the chosen channel and the filter: it is ~ 15 cm^2 for the FUV (Far Ultraviolet) channel which only use crystal filters, and it is in range 15-40 sq cm for the various filters in NUV (Near Ultraviolet) & VIS channels.

Figure 7-30 Photo of the UVIT engineering model

Table 7-13: Key parameters of UVIT

Parameter	Value
FOV (Field of View)	~28 arcmin
Selectable filters	~ 100 A and ~ 500 A

Spectroscopy	~ 100 cm^{-1} resolution in FUV/NUV
Temporal resolution/accuracy	~ 5 ms
Observing mode	Stare
Peak effective area	~ 15 cm^2 in the 130-180 nm range ~ 50 cm^2 in the NUV and VIS channels
Spatial resolution	< 1.8 arcsec in UV

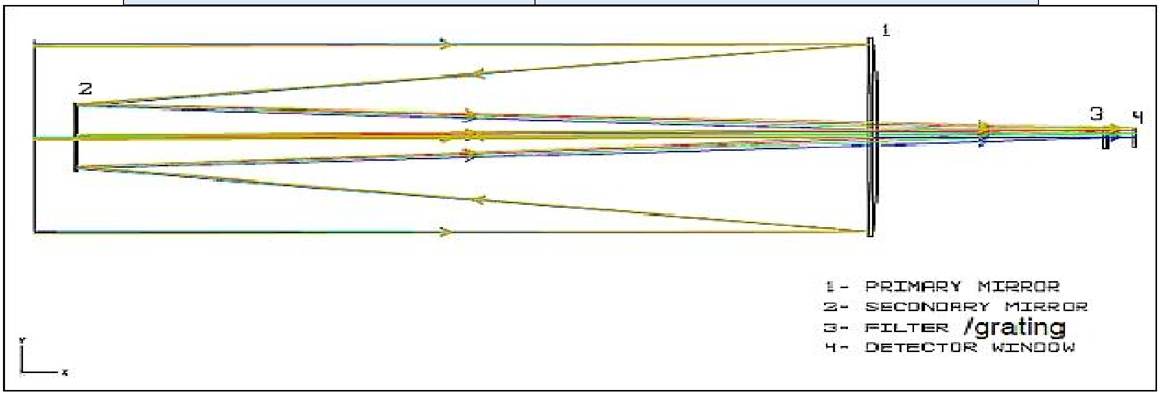


Figure 7-31: Optical layout of the FUV channel, f/12 Cassegrain, ~ 380 mm aperture

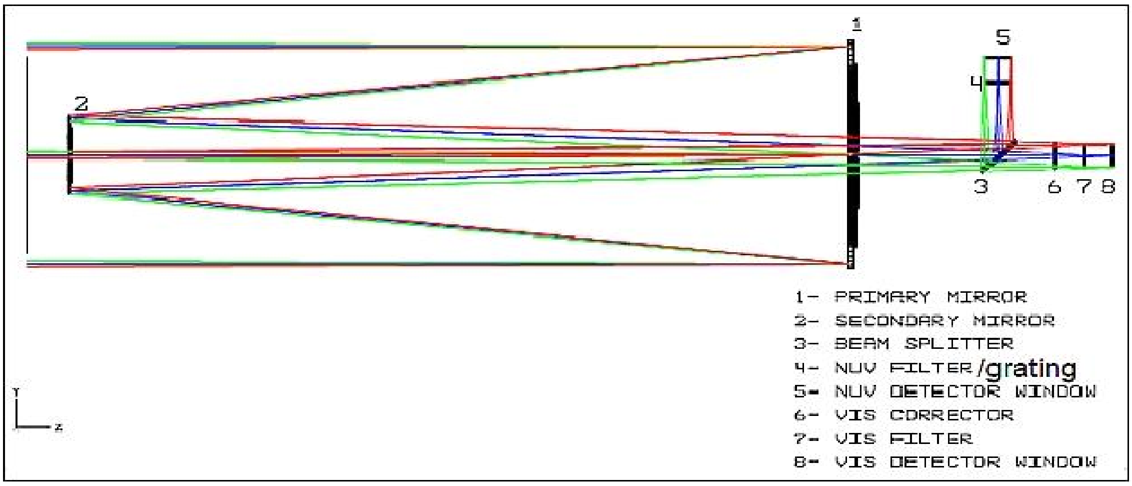


Figure 7-32: Optical layout of the NUV & VIS channels, f/12 Cassegrain, ~ 380 mm aperture

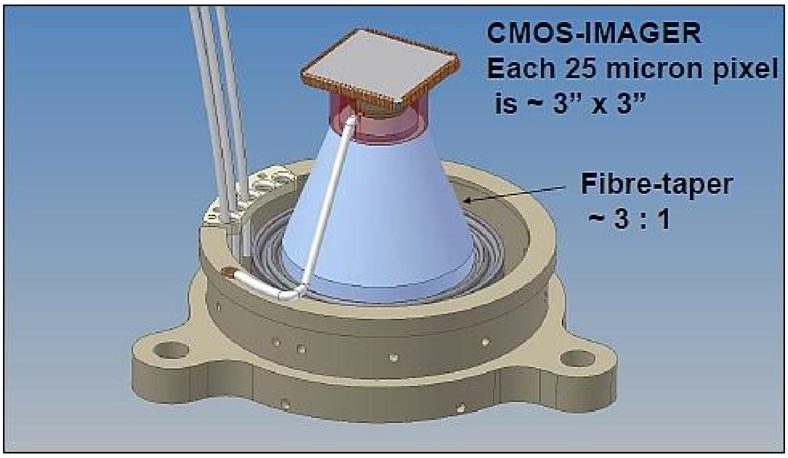


Figure 7-33: UVIT detector module

The UltraViolet Imaging Telescope (UVIT): The UltraViolet Imaging Telescope will perform imaging simultaneously in three channels: 130-180 nm, 180-300 nm, and 320-530 nm. The field of view is a circle of ~ 28 arcmin diameter and the angular resolution is 1.8" for the ultraviolet channels and 2.5" for the visible channel. In each of the three channels a spectral band can be selected through a set of filters mounted on a wheel; in addition, for the two ultraviolet channels a grating can be selected in the wheel to do slit-less spectroscopy with a resolution of ~ 100.

7.4.5.2 SXT (Soft X-ray imaging Telescope):

The SXT assembly employs focussing optics and a deep depletion CCD camera at the focal plane to perform X-ray imaging in 0.3-8.0 keV band. The optics consist of 41 concentric shells of gold-coated conical foil mirrors in an approximate Wolter-I configuration. The focal plane CCD camera is very similar to that flown on SWIFT XRT (X-Ray Telescope) of NASA. The CCD will be operated at a temperature of about -80ºC by thermoelectric cooling.

Telescope optics at the soft X-ray bands employ grazing incidence reflection from metal surfaces. The refractive index of metals in X-rays is slightly less than one so it is possible to get a total external reflection at a vacuum-metal interface if the X-rays are incident nearly parallel to the metal surface. The limiting angle of grazing incidence lies between a few degrees at ~0.1 keV to a few arcminutes at ~10 keV.

Table 7-14: Summary of the main SXT characteristics

Parameter	Value
Telescope length	2465 mm (including baffle, door and camera)
Telescope mirrors	Conical shells
Focal length	2000 mm
Telescope PSF (Point Spread Function)	1.5 - 2.5 arcmin (rms)
FOV (Field of View)	41.3 x 41.3 arcmin
Energy range	0.3-8.0 keV
Detector	E2V CCD-22 (Frame Store)
Detector format	600 x 600 pixels
Pixel scale	4.13 arcsec/pixel
CCD readout modes	Photon Counting, Imaging, Timing
Effective area	200 cm^2 @ 1.5 keV
Position accuracy	30 arcsec
Sensitivity expected	10 µCrab or better

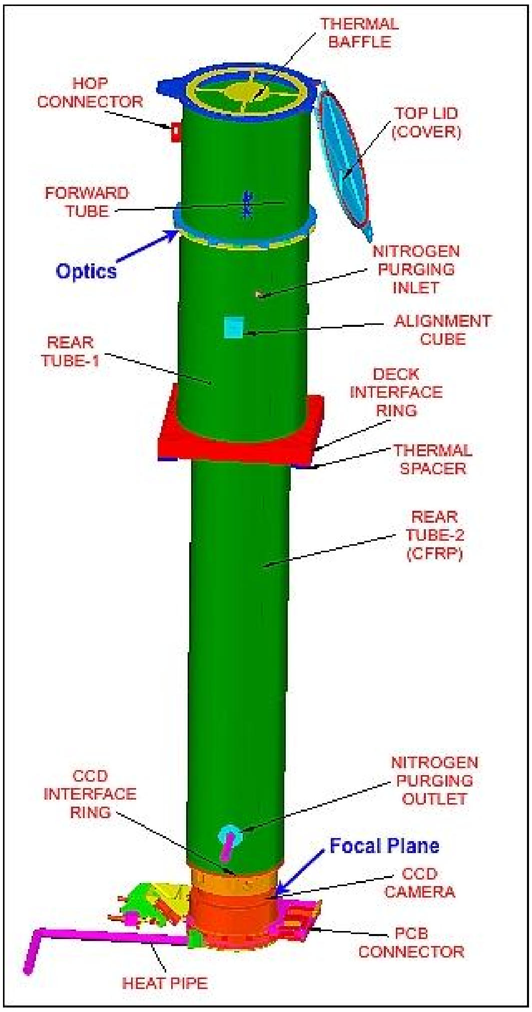


Figure 7-34: Illustration of the SXT structure

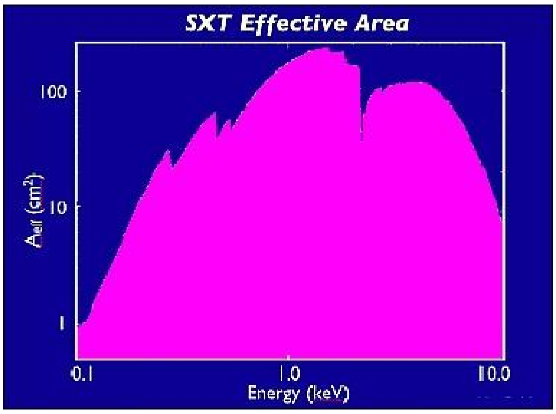


Figure 7-35: Effective area of the SXT as a function of photon energy

At its focal plane, the SXT carries a thermoelectrically cooled X-ray CCD camera, based on the e2V Technologies CCD-22 chip. The CCD has 600 x 600 pixels each of 40 micron square. It is a frame transfer device - an image transferred from image to store section can be read out while a new image is being acquired.

The CCD detector operated in single photon counting mode. Each X-ray photon, depending on its energy, will liberate about 100 to 1000 electron-hole pairs. Preserving this total charge information for each photon will lead to the measurement of its energy, thus enabling spectroscopic studies. The energy resolution is strongly degraded by system noise. To reduce thermal noise in the CCD it will be thermoelectrically cooled to an operating temperature of -80oC, which is expected to yield an energy resolution of about 2% at 6keV.

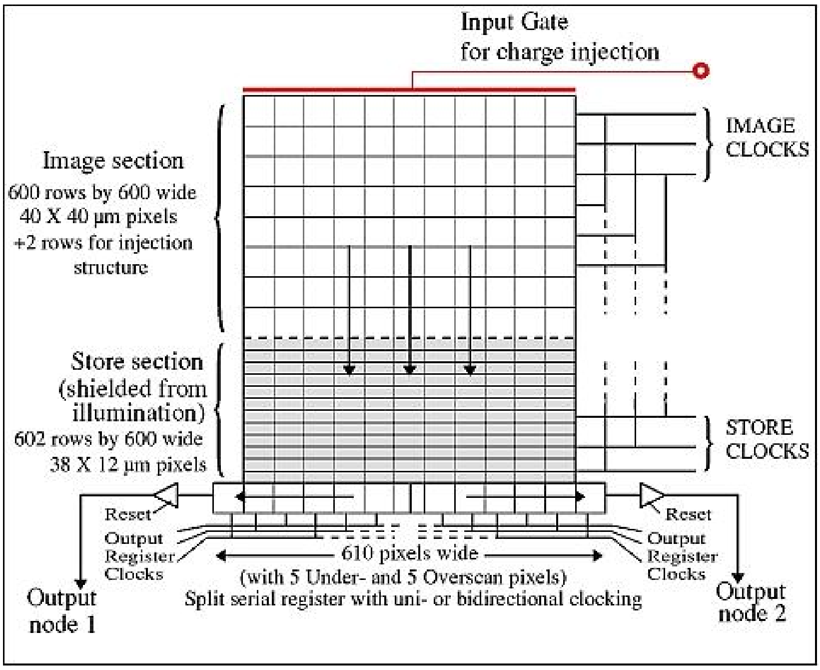


Figure 7-36: Schematic diagram of the CCD-22 detector (image credit: E2V)

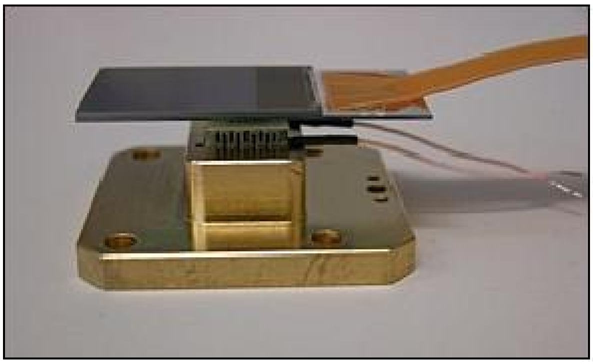


Figure 7-37: Photo of the thermo-electric cooler and CCD assembly

The focal plane camera assembly consists of the CCD and its cooling arrangement housed in a cryostat, which will also contain four Fe55 calibration sources, an optical blocking filter for the CCD and an aluminum proton shield to protect the CCD from proton damage while passing through the South Atlantic Anomaly region. The optical blocking filter is made of a single fixed polyamide film of thickness 184 nm, with a 48.8 nm thick aluminum coating on one side. This yields an optical transmission of about 0.25%, limiting the background light reaching the detector. The entire cryostat body is made of aluminum alloy, gold plated for thermal insulation.

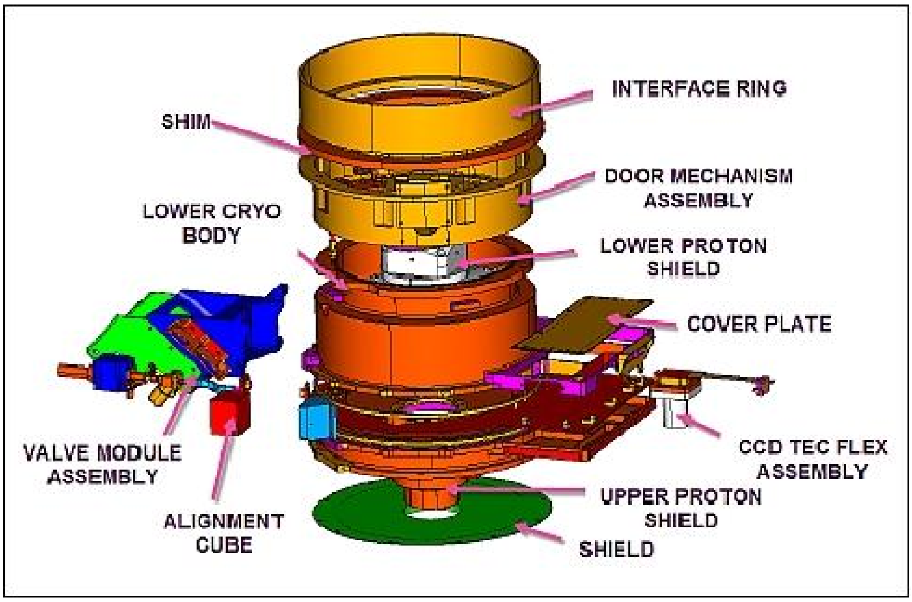


Figure 7-38 Focal plane camera assembly of SXT (image credit: AstroSat collaboration)

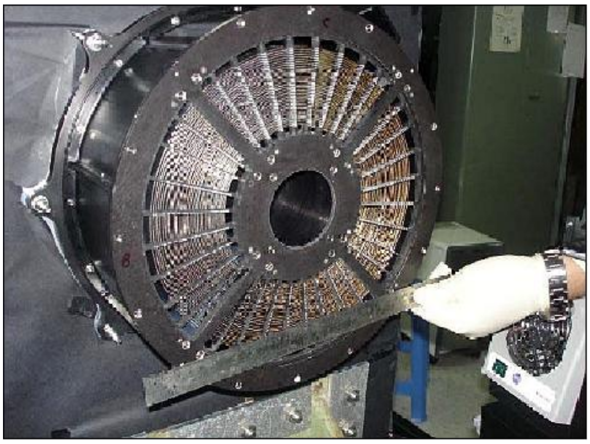


Figure 7-39 Photo of the SXT flight model optics entrance aperture

7.4.5.3 LAXPC (Large Area Xenon Proportional Counters):

The instrument is used for X-ray timing and low-resolution studies. The assembly consists of a cluster of three co-aligned identical Large Area X-ray Proportional Counters (LAXPCs), each with a multi-wire-multi-layer configuration and a FOV of 1º x 1º. These detectors are designed to achieve:

1) A wide energy band of 3-80 keV
2) High detection efficiency over the entire energy band
3) Narrow field of view to minimize source confusion
4) Moderate energy resolution
5) Small internal background and
6) Long life time in space.

A Xenon-based gas mixture at a pressure of two atmospheres will be filled in multilayer 15 cm deep detectors to achieve an average detection efficiency of close to 100% below 15 keV and about 50 % up to 80 keV. A thin (thickness of 25/50 μm) aluminized Mylar window for X-ray entrance ensures a low energy threshold of about 2-3 keV. The Mylar film is supported by a honeycomb shaped window support collimator with a 5º x 5º FOV. A FOV of 1º x 1º is provided by using mechanical collimators made of a sandwich of tin, copper and aluminum co-aligned with the window support collimator and sitting above it.

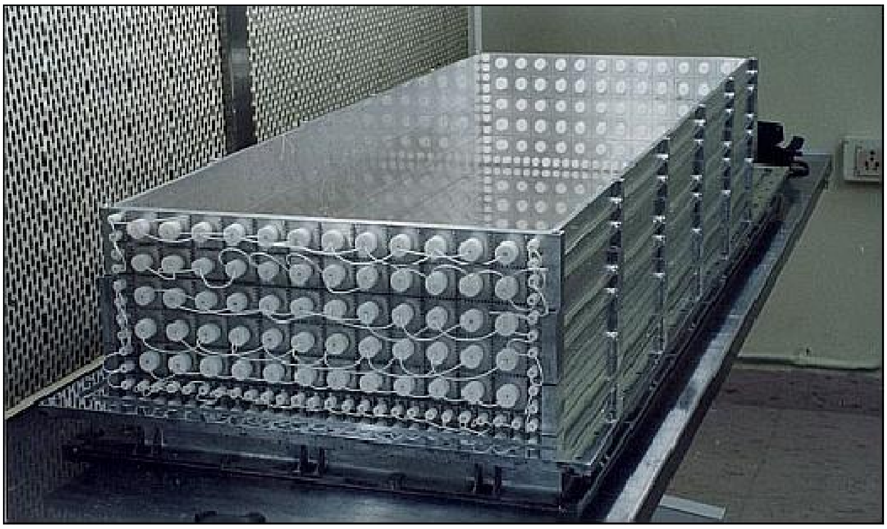


Figure 7-40 Photo of the LAXPC wired Anode Assembly

LAXPC X-ray detector anode assembly with veto layer on 3 sides mounted on the back plate. 60 anode cells are arranged in 5 layers to make the X-ray detection volume, 37 μm diameter Au-plated SS wires under tension used for anodes.

The total effective area of the 3 LAXPCs is ~ 6000 cm^2 at 5 keV. Due to its large depth and high gas pressure the LAXPC will have high detection efficiency right up to about 80 keV,

To achieve good energy resolution of the detectors, it is necessary to have a uniform gain over the entire area and the gas needs to be free from impurities like oxygen and water vapor. The former is achieved by precision placement of the anode wires at the center of the cells and by the use of anode wire of uniform diameter. An onboard purifier is being used to purify the gas from time to time; it will prevent degradation of energy resolution due to slow outgassing from detector walls.

The high sensitivity of the LAXPC instrument will allow the detection of a 0.1 mCrab source at the 5σ level in an exposure of about 10^4 seconds. This will enable the LAXPC to address a wide variety of science topics.

Four modes of operation:

- Broad band counting with variable integration time in many energy channels
- Onboard pulse height histograms with variable integration time
- Time tagging of each photon to 10 μs accuracy
- Fast counting mode to handle high counting rates from bursts.

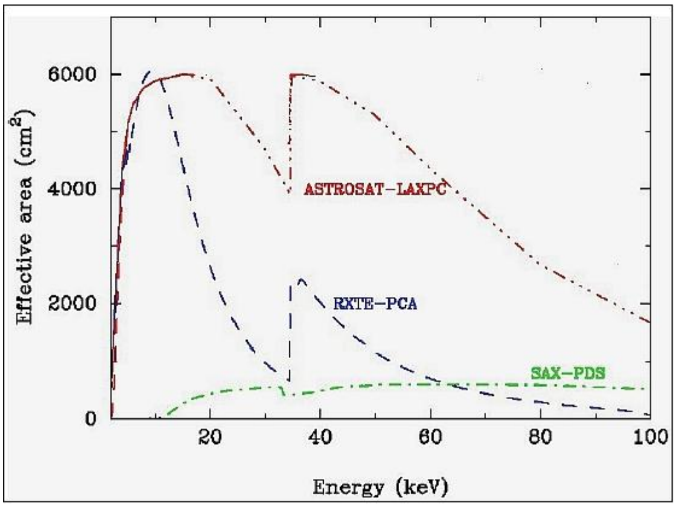


Figure 7-41 Effective area of the LAXPC instrument as a function of energy (image credit: AstroSat collaboration)

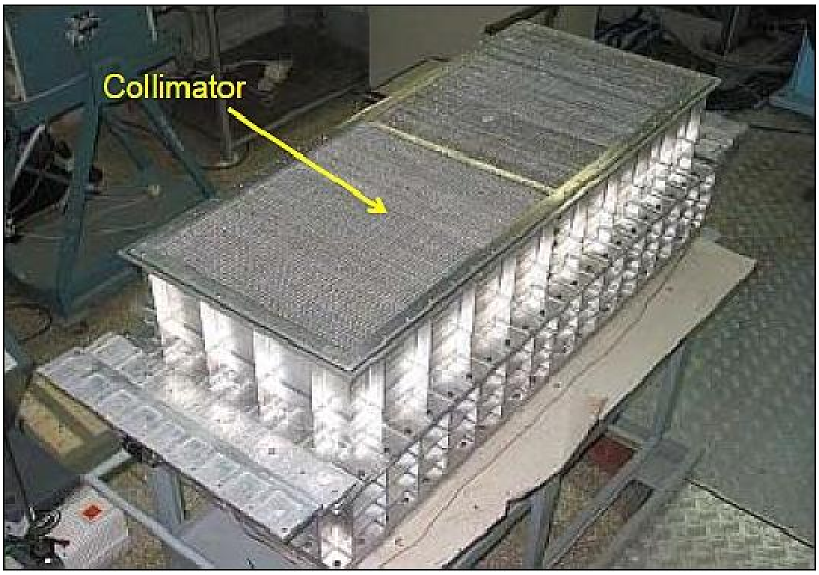


Figure 7-42 Photo of the LAXPC collimator (image credit: AstroSat collaboration)

The LAXPC instrument has a mass of ~ 390 kg.

7.4.5.4 CZTI (Cadmium-Zinc-Telluride coded-mask Imager):

The CZTI instrument consists of a pixelized CdZnTe (Cadmium-Zinc-Telluride) detector array of ~1000 cm^2 in geometric area. These detectors have very good detection efficiency, close to 100% up to 100 keV, and have a superior energy resolution (~2% at 60 keV) when compared to scintillation and proportional counters. Their small pixel size also facilitates medium resolution imaging in hard X-rays. The CZTI will be fitted with a two dimensional coded mask, for imaging purposes. The sky brightness distribution will be obtained by applying a deconvolution procedure to the shadow pattern of the coded mask recorded by the detector.

The coded mask imaging technique is one possible way of performing wide field imaging with photons of energy greater than a few keV. It comprises of utilizing the shadows of a multiple pinhole mask plate cast on the detector, with the shift in the shadows encoding the location of the source in the sky. The CZTI comprises of a two dimensional mask plate mounted on top of a pixelized CZT detector array.

Table 7-15: Key parameters of the CZTI instrument

Parameter	Value
Detector	CZT (Cadmium-Zinc-Telluride) detector array
Energy range	10 - 150 keV, up to 1 MeV (photometric)
Energy resolution	5% @ 100 keV
Pixel size, number of pixels	2.4 mm x 2.4 mm (5 mm thick)
Number of pixels	16384
Geometric area	1024 cm^2
FOV (Field of View)	6º x 6º (10-100 keV) (defined by collimator) 17º x 17º (> 100 keV) (defined by coded mask housing)
Angular resolution	8 arcmin (< 100 keV)
Veto layer	2 cm thick CsI crystal+PMT (Photo Multiplier Tube)
Read-out	ASIC based (128 chips of 128 channels)
Imaging method	CAM (Coded Aperture Mask)
Overall size	50 cm x 50 cm x 70 cm (height), without radiator plate
Instrument mass, power	50 kg, 50 W

The CTZI instrument is fabricated in four identical, independent quadrants which are joined together in the final configuration. Each quadrant has a 64 x 64 element coded mask and a detector array of the same number of pixels. The mask pattern of adjacent quadrants are rotated by 90º with respect to each other.

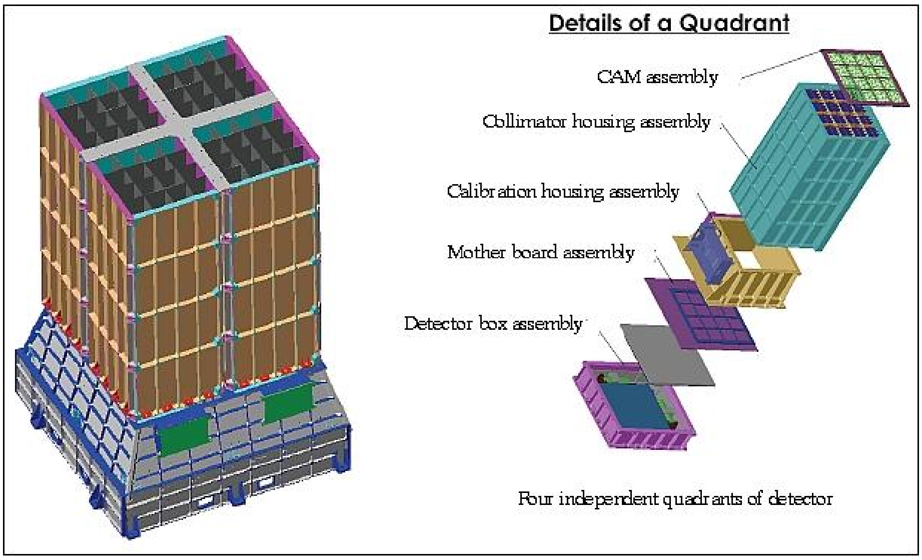


Figure 7-43: Schematic view of the CTZI instrument

7.4.5.5 SSM (Scanning Sky Monitor):

The SSM instrument consists of three position sensitive proportional counters, each with a one dimensional coded mask, very similar in design to the ASM (All Sky Monitor) on NASA's RXTE (Rossi X-ray Timing Explorer) satellite (launch Dec. 30, 1995). The gas-filled proportional counter features resistive wires as anodes. The ratio of the output charge on either ends of the wire provide the position of the X-ray interaction, providing an imaging plane at the detector. The coded mask, consisting of a series of slits, casts a shadow on the detector, from which the sky brightness distribution can be derived.

The objectives of SSM are:

- To detect, locate and monitor x-ray transients (nearly half of known x-ray binaries are transients)
- Monitor known bright sources (several samples/day; monitor for many months)
- Alert other instruments for detailed studies.

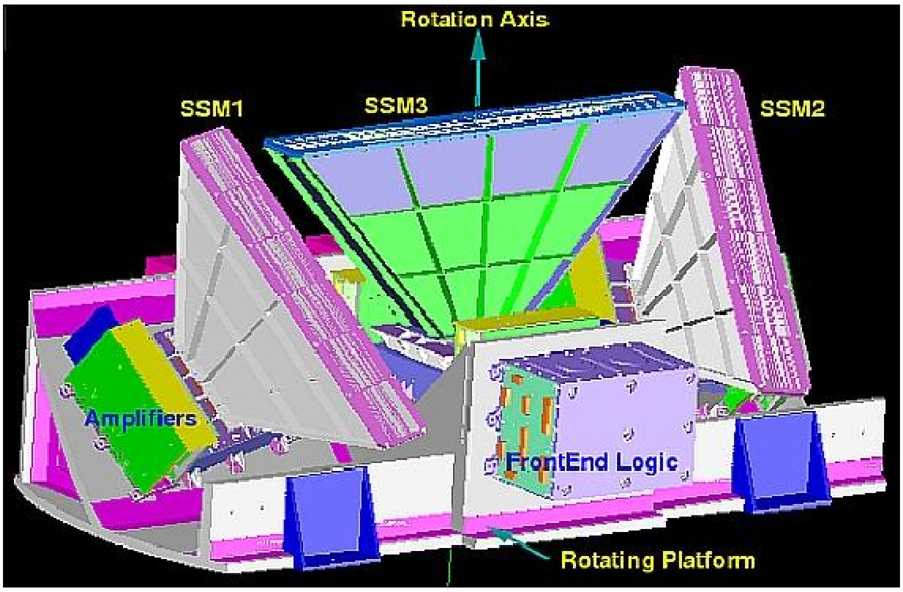


Figure 7-44 : Schematic view of the SSM instrument

Table 7-16: Table 5: Key parameters of the SSM instrument

Parameter	Value
Detector	Proportional counters with resistive anodes; ratio of signals on either ends of anode gives position
Energy range	2 - 10 keV
Position resolution	1.5 mm
Position determination	~0.5 mm
FOV	10º x 90º (FWHM)
Sensitivity	30 mCrab (5 minute integration)
Best time resolution	1 ms
Angular resolution	~ 10 arcmin
Instrument mass, power	48 kg, 30 W

The operation of the SSM is to scan the sky continuously irrespective of the functions of the other instruments on the spacecraft. A mounting arrangement is therefore necessary to enable these detectors to scan as much of the sky as possible, independent of the satellite pointing.

The three counters are mounted on a rotating platform providing a stepped rotation at discrete steps about one axis. One of the monitors (the boom camera: SSM3) is aligned with the rotation axis while the other two are mounted with their field of view forming an 'X' in the sky

Typical scan pointing will be ~10 º apart with ~10 minute integration at each location. This enables nearly half of the sky coverage, about 4 times per day (including nominal SAA exclusion orbits).

7.4.5.6 CPM (Charged Particle Monitor):

A CPM, an auxiliary instrument, is included in the sensor complement of AstroSat to control the operation of the LAXPC, SXT and SSM instruments. Even though the orbital inclination of the satellite is 8º, in about 2/3rd of the orbits, the satellite will spend a considerable time (15 - 20 minutes) in the SAA (South Atlantic Anomaly) region which has high fluxes of low energy protons and electrons. The high voltage will be lowered or put off using data from CPM when the satellite enters the SAA region to prevent damage to the detectors as well as to minimize the ageing effect in the Proportional Counters.

A Scintillator Photodiode Detector (SPD) with a Charge Sensitive Preamplifier will be used to detect the charged particles.

In the CPM, a cube of 10 mm side length of CsI (Tl) crystal (wavelength = 550 nm) with Teflon reflective material is coupled to the same area window of a Si-PIN diode. The incident charged particle energy is converted into light in CsI, and the light, seen by the photodiode, is converted into an electrical pulse with the help of a CSPA (Charge Sensitive Pre-Amplifier). The electrical signal is then passed through a LLD (Lower Level Discriminator) with a threshold level commandable from ground. The output is made available to all other instruments on board, and is also recorded as a part of the satellite housekeeping data.

Table 7-17: Key parameters of the CPM instrument

Parameter	Value
Scintillator	1cm x 1cm x 1cm CsI (Tl) crystal
Light collector	Photodiode with pre-amp (Hamamatsu s3590-08+eV5152)
Window	1 mil Kapton
Low energy threshold	1.2 MeV
Time resolution	5 s
Expected count rate	1 s^{-1} (in non-SAA region)
Maximum count rate	1000^{-1}
Instrument size, mass, power	18 cm x 15 cm x 5 cm, 2 kg, 2.3 W

8. Micro and Nano Satellite Series

8.1 IMS-1 (TWSAT)

8.1.1 Introduction

IMS-1 (earlier it was called as TWSAT) is the first micro satellite fabricated for remote sensing application purpose. This satellite can be launched mounted on the EB of PSLV with no additional cost for launch.

8.1.2 Mission Objective

- To build, launch and operate a 3 axis stabilized remote sensing micro satellite for launch, onboard PSLV, as auxiliary satellite, providing easy access of remote sensing data to the educational institutions, research organizations and government agencies in the developing countries. The spacecraft bus is developed as a versatile Micro Satellite bus in order to carry in future, a number of different payloads without significant changes in the bus.
- To develop low cost user terminals that can be used by users in Universities or Institutions of developing countries to receive the payload data.
- The Hyper-spectral imager being flown in Chandrayaan-1 is also being flown in IMS-1 to evaluate and validate the payload.

8.1.3 Orbit Details

Parameter	Value
Altitude	638 km
Semi Major Axis	7012.279 km
Inclination	97.94 º
Orbit	Polar Sun Synchronous
Eccentricity	0.001
Local time	09. 44 AM (descending node)
Orbits/day	14
Repeat cycle	369 orbits in 25 days
Period	97minutes
Path to Path separation	108.6 km

8.1.4 Salient Features of Spacecraft

Mechanical systems: IMS-1 is designed with Aluminum Honeycomb sandwich panels as a Cuboid structure with a bottom deck, top deck and four cross ribs in a staggered fashion connecting the top and bottom deck. This configuration generates a central core to house the fuel tank, thruster and plumbing while transferring the loads effectively to the interface ring. Payloads are mounted on the top deck.

The electronic packages are mounted on the four cross ribs / cover panels for mounting external appendages like antennas etc, forming the cuboid. The structure is a cuboid of size 552 x 600 x 600 mm. The overall size of spacecraft with stowed solar panel is 604 x 980 x 1129 (h) mm. IBL-298 interface ring interfaces with PSLV.

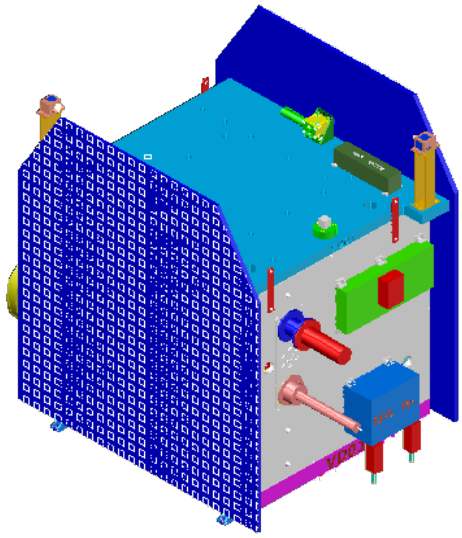


Figure 8-1 Stowed view of IMS-1

Thermal System: Thermal control is achieved by passive means using semi-active elements like paints, MLI, OSR and thermal tapes. However, provision is made for heaters wherever necessary. Thermistors, Platinum Resistance Temperature sensors and Thermocouples are used for temperature monitoring at required locations of IMS-1. The temperature sensor data is processed in BMU for heater control and telemetry to ground.

Data Handling System: The Payload consists of 4 band multi-spectral CCD camera (MxT) and 64 bands HYSI payload. Either MX data or HYSI data will be recorded and transmitted at a time. The BDH consists of P/L interface unit, Data Compression, RS coding and formatting unit. The Compression ratio is 3.401:1 for MXT Payload and no compression for HYSI P/L data. Compressed, RS encoded and formatted data is stored in SSR @ 10.66Mbps and simultaneously played back @ 8 Mbps in compression mode for MXT Payload. In compression bypass mode RS encoded, formatted data is stored in SSR @ 32 Mbps and simultaneously played back @ 8Mbps for MX. The transmission time required is 1.33 times (10.66Mbps / 8 Mbps) the imaging time in compression mode for MX. The transmission time required is 4 (32 Mbps / 8 Mbps) times the imaging time in compression bypass mode for MX. The playback data @ 8Mbps is differential encoded in BDH before transmission to RF Transmitter. Data transmission is S band BPSK.

A Solid State Recorder of 16 GB is configured to meet the mission requirement. The data from the payloads are formatted and a single stream is input to SSR. The basic operating modes of the SSR are Normal, Diagnostics, BER or Self-Test.

RF Systems: A standard S-band (RF) TTC system supported by global network of ISTRAC will be used for Telemetry, Telecommand of IMS-1. To minimize the transmitted power and bandwidth and thereby cost of user terminals, it is designed that Payload and Telemetry data transmission as well as Telecommand reception are in S-band. A single Telemetry / Data transmitter is used for TM (4Kbps) / Payload data (8 Mbps). The transmitter will have RF output power of 5W for payload data transmission and 100 mW for telemetry data transmission. The modulation scheme is PCM / BPSK for both Payload and Telemetry. Payload data will be transmitted through a separate data transmission antenna with higher gain (+3dBi) and telemetry data will be transmitted through TTC antenna (0dBi). The common TM / DATA transmitter output will be switched between the TTC antenna and DATA antenna using a coaxial switch. A filter is used in the data transmission path to restrict the out of band emission. IMS-1 also has on-board a miniature SPS (Main and Redundant), which is used for generating accurate position and velocity parameters used for onboard orbit determination.

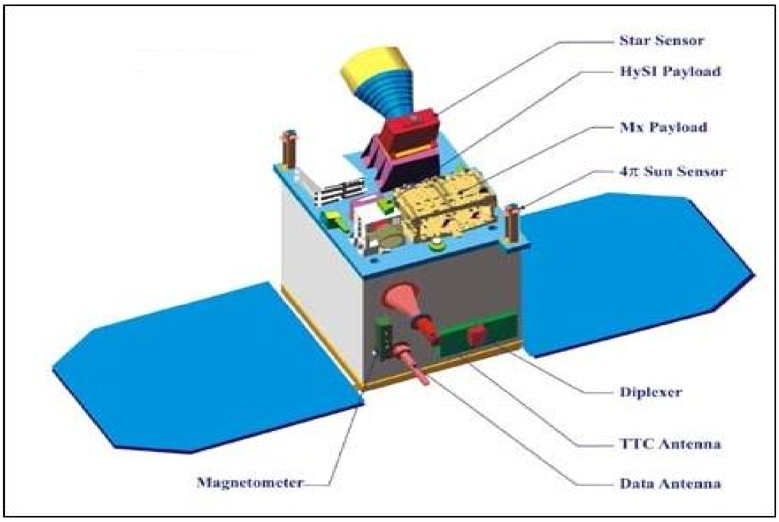


Figure 8-2 Deployed View of IMS-1

Power Systems: Power system is designed to meet the requirements of a micro satellite. IMS-1 Power system supports a nominal load of 72W, peak load of 132W during MxT Payload and peak load of 120 W during HySi-T Payload operation. Power system is based on a single bus of 28 – 33V. The solar array consists of two wings, each having one panel of 0.810m x 0.720m. In order to meet the higher specific power requirements, Triple Junction Solar cells are used for power generation. To match the envelope of the rocket, the solar panels are chamfered at two corners. IMS-1 spacecraft has one lithium ion battery of 10.5 Ah capacity. Power electronics of IMS-1 is similar to HAMSAT & SRE. To meet the powering requirements of new packaging concept, involving cards (instead of packages) for different subsystems, common DC / DC converters are used. Additionally, a new centralized power switching and distribution scheme is used for switching and distributing the outputs of DC/DCs & Raw Bus, to different user systems. This reduces the total number of DC/DC converter requirements.

BMU & Attitude and Orbit control system: The Bus Management Unit (BMU) executes the attitude and orbit control functions like TM, TC, and attitude orientation and orbit maintenance of the spacecraft to the required accuracies. Apart from this, the BMU does the 3-axis auto acquisition and control from the moment of injection into the orbit and puts the spacecraft in safe mode sun pointing orientation in the case of contingency.

The AOCS specifications are:

- Pointing accuracy : 0.1 deg (3σ)
- Drift Rate : 5.0 X 10^{-04} ° / sec.

Four heads of 4 π Sun sensor, a miniature tri-axial magnetometer, a single head Star Sensor and Inertial Reference unit (2 DTGs) are used for attitude sensing. In Magnetometer, for one axis, a MEMS sensor is used in place of conventional sensor. There are four Micro Reaction Wheels with 0.36Nms angular momentum arranged in tetrahedron configuration for attitude control and two magnetic torquers of dipole moment of 9 Am^2 along Roll and pitch axis used during detumbling. A monopropellant Reaction control System comprising a fuel tank with 3.5 kg fuel and one 1N thruster is planned for orbit correction.

Table 8-1 Specifications of IMS-1

Parameter		IMS-1 (TWSAT)
Mass		87 kg
Structure		Aluminum Honeycomb sandwich based Cuboid structure with a bottom deck, top deck and four cross ribs in a staggered fashion connecting the top and bottom deck.
Thermal	Components	Passive control using tapes , OSR, MLI Blankets and semi-active/active control using proportionate temperature controller and heaters
	Temp. Range	20±5 deg.C range for imaging sensors electro-optics 0 to 40 deg.C for electronic packages
Mechanism	Solar Panel	Paraffin based actuator for solar panel deployment mechanism.
Power	Solar Array	The solar array consists of two wings, each having one panel of 0.810m x 0.720m. Multi-junction cells, 206 watts @ the end of 2 years. Nominal load :72W, peak load of 132W
	Battery	Li-ion, 10.5 AH, 8S x 7P, one battery
	Electronics	Power system is based on a single bus of 28 – 33V.
Communication	Telemetry/data transmission	A single Telemetry / Data transmitter is used for TM (4Kbps) / Payload data (8 Mbps). The transmitter will have RF output power of 5W for payload data transmission and 100 mW for telemetry data transmission. The modulation scheme is PCM / BPSK for both Payload and Telemetry
	Telecommand	Modulation scheme is FM//FSK/PCM.
BMU(AOCE+ TM/ TC)	Attitude/Orbit sensors	Star sensor(1), 4 PI sun sensors(4), Dynamically Tuned Gyros (DTG)(2), Magnetometer, (Y) Mems (R&P) normal, SPS for orbit determination

	Attitude control	0.36 NMS, 0.018 Nm RW (4) mounted in tetrahedral configuration, Magnetic torquers(2),
	Orbit Control	Hydrazine thrusters (1 one Newton) 3.5 kg Fuel
Payloads		MX, HYSI

8.1.5 Payloads:

Payload system consists of two Payloads, namely Multi Spectral Camera and Hyper Spectral Camera

8.1.5.1 Multi Spectral Camera

The Mx-T is a four-band multi spectral camera with modular configuration having individual optics, detector assembly and Electronics separately for each band. The four bands selected for the instrument are identical to the previous IRS missions. The camera operates in a push broom scanning mode to image the earth. The Payload will be used for the purpose of natural resource management like Agriculture, Forest coverage and deforestation, urban infrastructure development, land use as well as disaster management. The major challenge in the design and development of camera has been to minimize size, weight and power and realization in shortest time.

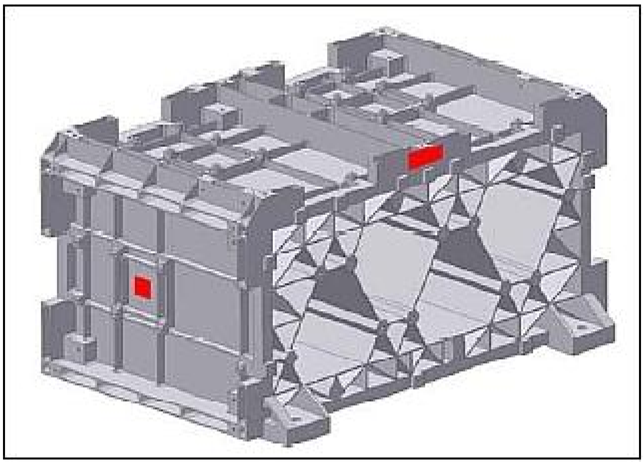


Figure 8-3 Mx Camera Structure

8.1.5.1.1 Payload Configuration

The TWSAT camera is configured to be a highly compact, low weight camera commensurate with the overall mission requirements of developing a low cost and lightweight micro-satellite. The camera, which operates in push broom mode, is multispectral with four bands in the visible and near infra-red (VNIR) spanning 0.45 microns to 0.86 micron. The spectral bands, viz., Band 1 (0.45 to 0.52 micron), Band 2 (0.52 to 0.59 micron), Band 3 (0.62 to 0.68 micron) and Band 4 (0.77 to 0.86 micron) are identical to the ones used in the previous IRS missions. The nominal ground resolution is 36.87 meters from an altitude of about 638 km.

All the four bands are nadir viewing with the linear detector array being used to image the scene in across track direction of the satellite motion. The width of the scene in across track direction (swath) depends on the array length and the focal length of optics used. Each time the array advances a distance equal to one resolution element, a new scan line is generated.

The TWSAT camera has a modular configuration with each of the four spectral bands having its individual optics, detector and associated electronics.

8.1.5.1.2 Multi Spectral Camera Specifications

Parameter	Value
Ground Resolution	36.87m
Altitude	636.18 km
Swath	151 km
Spectral Band	B1 (0.45 – 0.52 μm); B2 (0.52 – 0.59 μm); B3 (0.62 – 0.68 μm); B4 (0.77 – 0.86μm)
Integration time	5.23 ms
Camera SWR (at Nyquist frequency) (%) (TWSAT: 70lp/mm)	B1:≥ 20; B2:≥ 20; B3:≥ 20 ; B4:≥ 10
Saturation Radiance (mW/cm²/str/μm)	B1:55; B2:53; B3:47; B4:31.5
SNR (at Saturation radiance)	>400
Quantization(bits)	10
Data Rate	32 Mbps
No of ports	4
Detector	4 K Elements Linear CCD
Pixels per port Active	1024
Pixel size	7 x 7 Micron
Size(EOM) mm	300.2 x 151.7 x 227
Camera Weight (kg)	5.905 kg
Power (W)	10.4W (Four Bands); 160 mA @ 5.6V (Single Band); 90mA @ 18.7V (Single Band)

8.1.5.1.3 Description

The major subsystems of the Mx-T payload are

- Optics (Lens assemblies)
- Detector Head Assemblies
- Camera Electronics
- Mechanical System

Optics: The collecting optics for each of the spectral bands is an eight-element lens assembly with a thermal filter at the front and a band pass filter (to select the respective spectral range) at the rear. The optics is f/5 operating at a spatial frequency of 70 lp/mm over a field of view (fov) of ± 7 degrees. All four-lens assemblies (pertaining to the four bands) have identical designs for the ease of fabrication and assembly/alignment of lens elements. For details, please refer section 3 on optical configuration and design.

The optical configuration consists of a multi-element lens assembly with a thermal filter at the front and a band pass filter at the rear end. All the lens elements have spherical surface profiles. The last element is a plane parallel glass window with a band pass filter coating. The choice of two types of glass for elements ensures that the focus does not change appreciably within operating temperature of 20±10°C.

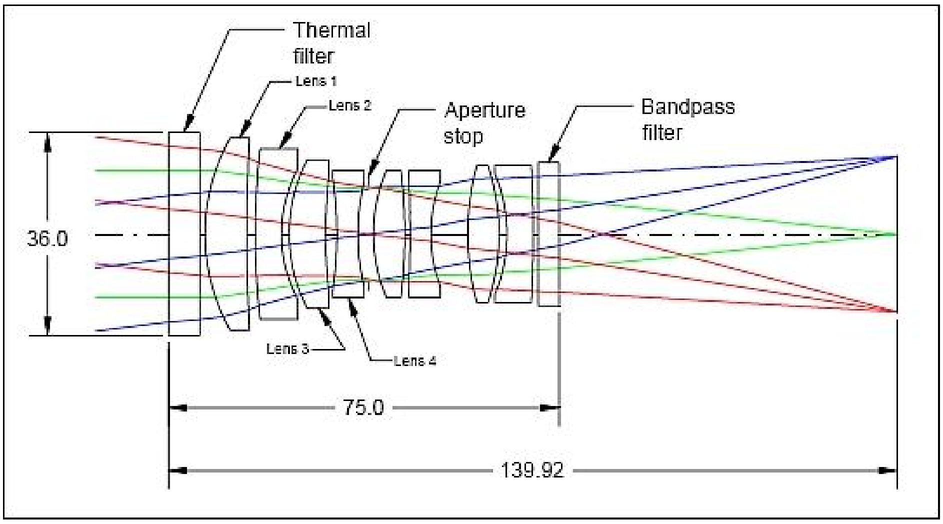


Figure 8-4 Optical Schematic of Mx

8.1.5.1.4 <u>Detector Head Assembly:</u>

Each of the four bands has a separate and identical detector head assembly (DHA), which essentially consists of a 4k linear array CCD (sc 3925a) with a pixel size of 7 micron x 7 micron, a PCB and mechanical housing. The detector is indigenously developed and qualified. SC 3925A, 4096 elements 7X7 micron linear CCD, manufactured indigenously by SCL, Chandigarh, is used. This is in line with the philosophy of using indigenous components for TWSAT wherever possible. Also, the performance of SCL made device is better than 4K device (manufactured by Thomson) used in IRS-1C/D. It has four buried channel CCD (charge coupled device) shift registers and four output amplifiers. The charge generated at the photosites is stored under storage gate (Vp\Vst) and is transferred to the shift registers by applying a transfer clock (Φx). The signals generated by pixels are shifted to register as follows. Pixels 1, 3, 5 ….2047 goes to shift register 1 and port. pixels 2, 4, 6…2048 goes to register 2 and port 2.

Pixels 4095, 4093, 4091 ….2049 goes to register 3 and port 3. Pixels 4096, 4094, 4092……..2050 goes to register 4 and port 4.

At each output channel, signal corresponding to 16 dummy pixels (6 isolation pixels + 4 dark pixel + 6 isolation pixels) arrives first, followed by the signal from sensitive 1024 pixels. After the 16 dummy pixel outputs, signal corresponding to end photosensitive pixels is delivered (pixel # 1 for $V_{os1,}$ pixel #2 for $V_{os2,}$ pixel # 4095 for Vos3 $_{and}$ pixel # 4096 for $V_{os4)}$. These output signals can be processed to reconstruct the image.

All four bands will have the same detector type. These devices have been source screened and qualified by the manufacturer, the SCL.

8.1.5.2 Hyper Spectral Camera

Hyper spectral imager (HySI-T) is the other payload in TWSAT. The Hyper-spectral imager being flown in Chandrayaan-1 is also being flown in TWSAT to evaluate and validate the payload. Inclusion of Hyper-spectral imager in TWSAT will enhance the mission capability. The data from this instrument will be useful for ocean and atmospheric studies.

Hyper-spectral Imager is already being developed for Chandrayaan-1 using Lens, wedge filter, active pixel detector, miniaturized camera electronics etc. Same configuration is used for TWSAT hyper-spectral imager. Necessary changes have been carried out in camera electronics FPGA logic design to match the TWSAT configuration and data rate. HySI-T will be an independent chain in TWSAT. Considering the transmittable data rate limits and power availability of spacecraft, it is planned to have either the multi-spectral or hyper-spectral payload operation at a time.

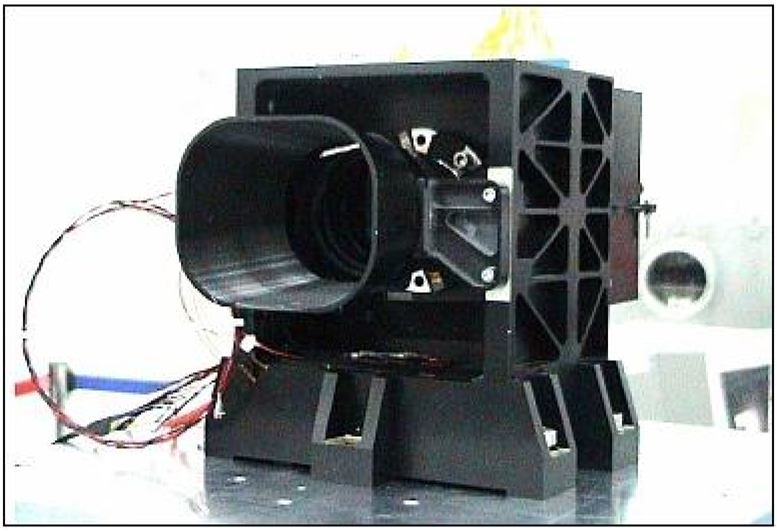


Figure 8-5 Hyper Spectral Camera

8.1.5.2.1 Specifications

Parameter	Value
Spectral range	400-950nm
No of bands	64 fixed
Spectral separation	8 nm
Ground track velocity	6.9302 km/s
Spatial resolution	505.6m
Along track sampling interval	543.6m
Swath	129.5 km
Bandwidth	<15nm
MTF	>0.2
SNR at saturation	>400 – 1500
No. of gains	1
No. of exposure settings	8
Clock input (BRC)	16MHz

WLS period	78.45ms
Digitization	16 bit
Data rate	4.0Mbps
Data type	16 bit serial
Power	0.8W (176mA @ 3.8V, 25mA @5.6V)
Weight	4 kg

8.1.5.2.2 Description

The major subsystems of the HySI-T payload are

- Optics (Lens assemblies)
- Detector Head Assemblies
- Camera Electronics
- Mechanical System

Optics: The collecting optics for HySI-T is a multi-element lens assembly with a thermal filter at the front. Effective focal length and F/No. are 62.5mm and 4 respectively.

Detector: A custom built area array with 512 rows and 256 columns based on active pixel technology with inbuilt 12 bit digitiser is used in HySI-T.

Spectral separation: Spectral separation is done using a wedge filter. The wedge filter is an interference filter with varying thickness along one dimension so that the spectral content transmitted through it varies in that direction. Thus when placed in front of the area array, all pixels in a given row will receive irradiance from same spectral (but different spatial) region. The pixels along a given column will receive irradiance from different spectral as well as different spatial regions. This arrangement of spectral dispersion results in spectral sampling at 1nm intervals and bandwidth of 8 nm for each row at system level.

Camera Electronics: Camera Electronics is designed around area array active pixel detector. It receives the 12 bit, 512 bands parallel data from detector. These 512 bands of over sampled data is processed to 64 bands and given to Base band data handling system. As the data is distributed in multiple integration times, the data from detector is stored. Real time data storage is incorporated in camera electronics.

Mechanical System: Camera structure is designed to hold various components like lens assembly, detector head assembly, hood etc. CE and power supply are in separate trays which are stacked together and mounted behind the camera.

Power System: HySI-T camera requires 3.8V and 5.6V power supplies and is similar to HySI Chandrayaan-1. Detector is powered though a filter which is part of power distribution system.

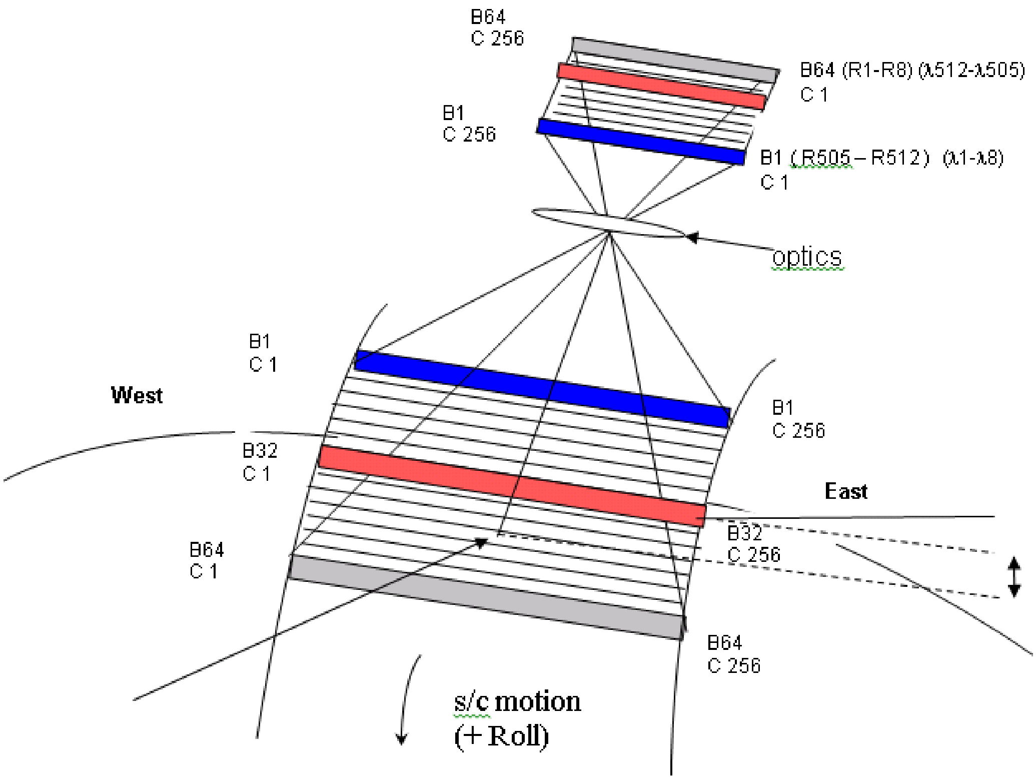


Figure 8-6 Spectral scanning and swath coverage of HySI

8.2 Youthsat

8.2.1 Introduction

The Youthsat is the second small satellite fabricated by ISAC. Youthsat carried three payloads namely SOLRAD, LiVHySI and RaBIT. The remote sensing data from this micro satellite is used for scientific studies like research of solar flare activity, mapping of Total Electron Content (TEC) of the ionosphere and measuring airglow of the earth's atmosphere.

8.2.2 Mission Objective

Mission Objectives of Youthsat are

- To build, launch and operate 3 axis stabilized Micro satellite for launch on-board PSLV as an auxiliary satellite with scientific payloads that are useful for observing solar flares and also for study of their impact on atmosphere.
- To involve the youth consisting of students, research scholars etc., for the development and use of payloads mentioned above, in order to inculcate interest and participation in space related activities and also to participate in the data analysis.

8.2.3 Orbital Parameters

Orbital parameters of Youthsat is as given below.

Paramater	Value
Local Time	10.30 AM
Altitude (km)	817

Semi Major Axis (km)	7195.11
Inclination (Deg)	98.69
Orbits/Cycle	341
Orbit/Day	14.22
Repetivity	24 Days
Period (Min)	101.35
Ground track velocity (km/s)	6.65

8.2.4 Salient features of Youthsat

The salient features of Youthsat are given in following table.

Table 8-2 Specifications of Youthsat

Parameter		Youthsat
Mass		87 kg
Structure		Aluminum Honeycomb sandwich based Cuboid structure with a bottom deck, top deck and four cross ribs in a staggered fashion connecting the top and bottom deck.
Thermal	Components	Passive control using tapes , OSR, MLI Blankets and semi-active/active control using temperature controller and heaters
	Temp. Range	20±5 deg.C range for imaging sensors electro-optics 0 to 40 deg.C for electronic packages
Mechanism	Solar Panel	Paraffin based solar panel deployment mechanism.
Power	Solar Array	Nominal load :72W, peak load of 132W The solar array consists of two wings, each having one panel of 0.810m x 0.720m. Multi-junction cells, 206 watts @ the end of 2 years
	Battery	Li-ion, 10.5 AH, 8S x 7P, one battery
	Electronics	Power system is based on a single bus of 28 – 33V.
Communication	Telemetry/data transmission	A single Telemetry / Data transmitter is used for TM (4Kbps) / Payload data (8 Mbps). The transmitter will have RF output power of 5W for payload data transmission and 100 mW for telemetry data transmission. The modulation scheme is PCM / BPSK for both Payload and Telemetry
	Telecommand	Modulation scheme is FM//FSK/PCM.
BMU (AOCE+TM/TC)	Attitude/Orbit sensors	Star sensor(1), 4 PI sun sensors(4), Dynamically Tuned Gyros (DTG)(2), mems Magnetometers(2), SPS for orbit determination

	Attitude control	0.36 NMS, 0.018 Nm RW (4) mounted in tetrahedral configuration, Magnetic torquers(2),
	Orbit Control	Hydrazine thrusters(1 one Newton) 3.5 kg Fuel
Payload		SOLRAD, LiVHySI, RaBIT

8.2.5 Youthsat Payloads

Youthsat is second in the Indian Mini Satellite-1 Series carrying three payloads namely SOLRAD, LiVHySI and RaBIT.

- SOLRAD by Moscow University (**Sol**ar **Rad**iation Experiment)
- RaBIT by SPL-VSSC (**Ra**dio **B**eacon for **I**onosphere **T**omography)
- LiVHySI by VSSC & SAC (**Li**mb **V**iewing **Hy**per **S**pectral **I**mager)

8.2.5.1 SOLRAD Payload

SOLRAD (**Sol**ar **Rad**iation Experiment) is a co-operative joint scientific mission between India and Russia with participation of youth from both the countries. The payload is developed with an aim to inculcate interest in the youth in space research and space technology.

Scientific goals: SOLRAD instrument is designed in SINP/MSU to study time variations of solar x-ray and gamma-ray flux and spectra as well as the variations of the flux of charged particles generated in the Sun or in the Earth vicinity. Astrophysical gamma-ray bursts and some variable sources can be also studied.

SOLRAD experiment will provide the measurements in the range:

- X-rays and gammas 0.02-10 MeV
- Electrons 0.3-3.0 MeV
- Protons 3-100 MeV, Alphas 5 - 24 MeV/nucl., nuclei of C, N, O group 6 - 15 MeV/nucl.

The goal is research of solar flare activity by measuring temporal and spectral parameters of solar flare X-rays and gamma rays as well as of charge particle (electron and protons) fluxes in the Earth Polar cap regions which are sensitive to solar flare activity. The scientific objectives are met with an X-ray and Gamma ray detector – spectrometer system using a NaI(TI) / Cs(TI) phoswich unit and a charged particle detector system using a silicon detector telescope unit. SOLRAD payload consists of two modules namely detector module and electronics module. The Detector module consists of two independent units: Detector Unit for Electrons (DUE) and Detector Unit for X-rays and Gamma (DUXG). Based on the scientific objective, SOLRAD payload has to be pointing towards Sun during Sun Pointing period of the orbit.

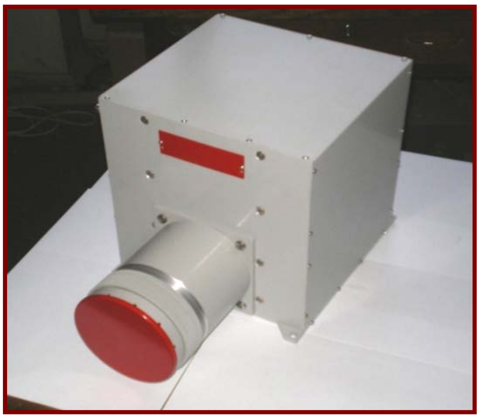


Figure 8-7 SOLRAD Detector Box

Figure 8-8 SOLRAD Information Box

The phenomena to be studied with SOLRAD particle detector are:

- SEP events and solar charged particle penetration boundaries in the Earth's magnetosphere during geomagnetic disturbances;
- Dynamics of the relativistic electron fluxes in the Earth's magnetosphere;
- Energetic particle precipitation under the Earth's radiation belts (at low and high latitudes).

The phenomena to be studied with SOLRAD x-ray and gamma detector are:

- Solar flares: fast x-ray and Gamma-ray flux variations
- Solar flares: thermal and non-thermal part of X-ray and gamma-ray spectra
- Solar flares: gamma-ray lines
- Astrophysical gamma-ray bursts (GRB)
- X-ray binaries, pulsars, SGR, etc

SOLRAD payload is always kept ON throughout the time in the orbit irrespective of the attitude geometry. SOLRAD payload data is stored in its internal memory. SOLRAD payload has a provision onboard to store the last 20 sessions payload data, which can be played back on requirement by issuing a SOLRAD multi data command appropriately. The estimated data volume is 100 Mbytes/day.

8.2.5.2 RaBIT Payload

Scientific Objectives: In the recent years it has become clear that the understanding of the ionosphere is central to the design of many modern communication, navigation and positioning systems. In the past, ionospheric studies have been confined to traditional areas of broadcast and radio communication. With the increasing use of satellites for navigation and positioning (GPS, GLONASS, etc.), characterizing and modeling of the ionosphere (its spatial and temporal variability) has become extremely important. This is because the position accuracy achievable from navigation satellites is largely affected by the intervening ionosphere. The range error is directly proportional to the total electron content (TEC) along the ray path. The equatorial ionosphere with its inter-related unique features like equatorial ionization anomaly (EIA), equatorial spread F (ESF), poses additional challenges due to their highly dynamical nature and large spatial and temporal variability which are not yet well quantified even for quiet conditions. The geomagnetic storms significantly alter the background ionospheric and thermospheric structure, energetic and dynamics and as a consequence modify the major equatorial ionospheric processes.

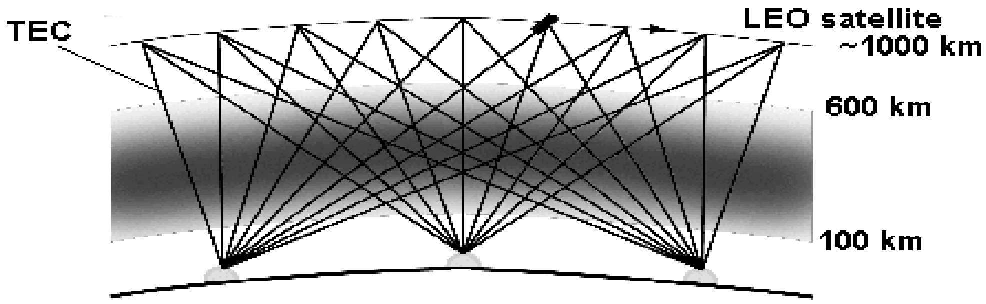
The morphological features of the equatorial ionosphere are well understood, but its day-to-day variability still remains enigmatic. These facts highlight the need for a comprehensive understanding of the complex processes of the ionosphere-thermosphere system including its response to the various external forcing so as to reach a level of predictive capability. One of the most important aspects still to be understood is the temporal and spatial variability in electron density distribution during space weather events. It has been established from the Indian Coherent Radio Beacon Experiment (CRABEX) program, using dual band coherent transmissions from Low Earth Orbiting Satellites (LEOS) that the tomographic techniques are very effective and useful in investigating the large-scale structures over low and equatorial latitudes, like equatorial ionization anomaly (EIA), equatorial spread F (ESF), their temporal and spatial variability, their inter relationship and response to space weather effects.

The main objective of RaBIT payload(**Ra**dio **B**eacon for **I**onosphere **T**omography) is to measure the Total electron content (TEC) of the Ionosphere. The position accuracy achievable from navigation satellites is largely affected by the intervening ionosphere. The range error is directly proportional to the total electron content along the ray path. It is understood that ionospheric TEC measurement simultaneously along a latitudinal chain of receivers could be used for tomographic imaging, i.e., for obtaining the latitude –altitude distribution of electron density of the ionosphere. RaBIT payload being an RF payload, it does not have any interface with Base Band Data handling system. RaBIT payload has to be on the earth-viewing side.

The scientific objectives of RaBIT payload are:

- To study the structure and dynamics of equatorial ionosphere over the Indian region using tomographic technique.
- To study the coupling between high and low latitudes during space weather events.
- To study the ionospheric effects of various solar and geophysical factors.
- Transcontinental studies of the ionosphere in Russia and India during different seasons and local time intervals

Ionospheric tomography: Ionospheric tomography is a powerful tool to address the spatial variability of the ionosphere. The advantage of tomography technique is that it can give a snapshot picture of the latitude-altitude variation of the ionosphere, using data from a chain of simple, inexpensive ground receivers, by recording coherent beacon signals from a low-earth orbiting satellite. The primary data for the tomographic inversion is the line of sight TECs estimated along a number of ray paths from a chain of ground receivers aligned along the same longitude. These TECs are then inverted to obtain the electron density distribution as a function of latitude and altitude over a given longitude. The schematic geometry of the CIT is shown in the figure below. In the simple case the ionosphere is replaced by pixels of appropriate size and electron density within each pixel is assumed to be a constant (piecewise constant).



Then mathematically, ionospheric tomography problem reduces to $Y = Ax + E$ Where *Y* is the observed TEC data, *x* is a the unknown electron densities, and *A* is the geometry matrix, which describes the relationship between the received TEC data and the electron densities on each ray path (the length of the ray in the corresponding pixel). Thus the electron density in each pixel is obtained as $x = A^{-1} Y$. In practice the inverse of the large geometry matrix is estimated by either truncated Singular Value Decomposition technique or Algebraic Reconstruction Technique.

Working Principle: The basic data for ionospheric tomography is the line of sight TEC (STEC). The STEC is obtained by Differential Doppler technique. Here the measured data, is the relative phase between 150 and 400 MHz, is proportional to the relative slant TEC (STEC) along the propagation path of the signal as

$$\phi = C_D \, x \, STEC \qquad (1)$$

Where, ϕ is measured in radians, STEC is in m^{-2} and $C_D = 1.6132 \times 10^{-15}$ for NNSS satellites (Leitinger, 1994). Since the phase measurements are accurate to $< 5°$when the receiver is at locked condition, and the data sampling is at 50 Hz, these observations yield accurate estimates of the relative TEC, with errors < 0.05%.

Estimation of TEC: The ground receiver measures the phase difference between the incoming signals, and the TEC is estimated by the method of Differential Doppler method. Here, based on the phase or frequency shift measurements which results from the changes in optical path length $P = \int n \, ds$: where n is the refractive index and is a measure of electron density, N_e. The relationship between N_e and n is given by Appleton Hartee equation as

$$n \sim 1 - 40.3 \, N_e / f^2$$

The reduction in phase path for a signal from satellite to ground receiver is

$$\Delta P = \int (n - 1) \, ds \sim - 40.3 \, N_T / f^2$$

Here N_T (= $\int N_E \, . \, ds$) is the TEC along the ray path.

Two coherent frequencies, f_1 (= p . f) and f_2 (= q . f) are transmitted and the received phases converted to a common frequency and compared. It can be shown that their phase difference ($\Delta\Phi$) is directly proportional to TEC (N_T).

i.e., $\Delta\Phi = [40.3\ N_T/(f\,.\,c)]\ \{q^{-2} - p^{-2}\}$

Generally in the case of the above satellites, the transmitted frequencies are 150 and 400 MHz (i.e., in the ratio 3: 8)

RaBIT Electronics: The purpose of RaBIT is to measure the Total Electron Content (TEC) of Ionosphere. RaBIT will generate two phase coherent frequencies, 150MHz and 400MHz. The relative phase of 150MHz with respect to 400MHz is proportional to the slant relative TEC along the line of sight. The basic source is a coaxial resonator oscillator (CRO) at 1200MHz. This is phase locked using an integer PLL. The reference to the PLL is a Temperature compensated Oven Controlled Crystal Oscillator (TC-OCXO). A clock distribution IC with programmable internal frequency divider generates two coherent frequencies viz, 400MHz, and 150MHz. An 8-bit microcontroller is used to program the PLL and to issue a synchronization command for synchronizing 400MHz and 150MHz signals. These outputs are filtered using in-house made band pass filters, which improve the signal quality. For improving return loss, attenuator pads are used at both the outputs, before and after amplification. The amplifiers are realized using Monolithic Microwave Integrated Circuits (MMIC). The final stage in each chain consists of a power amplifier, which enhances the power to 1.58 Watts. The entire circuitry works with a single 3.3V power supply. This power is derived onboard using a hybrid DC/DC converter with built in EMI filter. The 150MHz and 400MHz signals are combined using a lumped element frequency combiner to get nominal output power of 1W.

Antenna systems: RaBIT antenna is a deployable antenna. The deployable antenna system consists of (i) Boom assembly, (ii) Dipole sub assembly, (iii) UHF and VHF reflectors, (iv) Retention and release mechanism and (v) Deployment and locking mechanism. The major subassemblies of the system are detailed below:

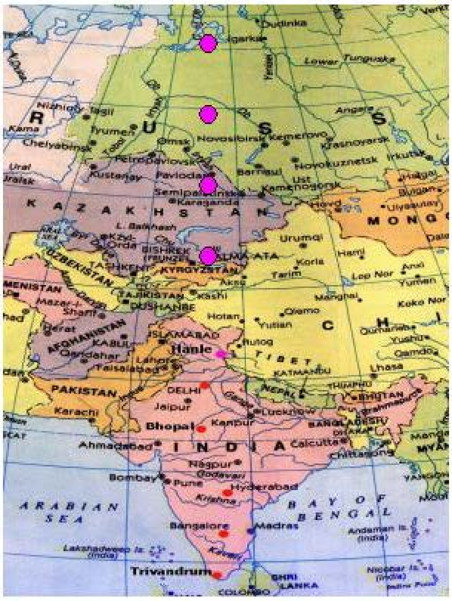
Boom Assembly: The deployable antenna system consists of a centrally positioned boom to which all subsystems are assembled. The boom assembly consists of two parts viz a conical lower part and a cylindrical upper part assembled using a lap joint at the center. The base of the boom is assembled to satellite deck. The top of the boom provides interface for mounting TTC antenna. Boom assembly is the central structural element of the antenna system. It provides the structural interface for various elements of the system. Dipole sub-assemblies and reflector are attached to the boom assembly through specially designed hinges.

Dipole sub Assembly: The dipole sub assembly consists of tubes (OD 12 mm, WT 1mm) made of brass. Two dipoles sub-assemblies are symmetrically attached to the top of the boom at diametrically opposite locations. Each dipole sub assembly is made of two brass tubes that is assembled using TRAP holder made of GFRP material to enable the system to work for dual frequency. The dipole sub assembly is electrically insulated from boom. The dipole tubes are bonded to TRAP holder using Hysol 9394. The trap holder houses the LC circuit which enables the antenna to work for dual frequencies.

UHF Reflector & VHF Reflector: The UHL reflector is made with rod made of Brass to meet the inertial constraint to avoid collision during deployment. The VHF reflector is made with tube made of Al alloy. The UHF reflector is a brass rod with OD 12 mm and the VHF reflector is an Al alloy tube with OD 12 mm. The UHF &VHF reflectors are positioned at a distance of 170mm and 425mm from dipole sub assembly respectively.

Retention and release Mechanism: In order to meet the envelope constraints, the dipole and reflectors are stowed during the ascent phase of the mission and deployed after injection of satellite into the orbit. The stowed dipole and reflectors are held in position using rope made of Nylon 6. The Nylon rope, either ends are attached to the strain gauged load links through bowline. The rope is finally assembled to the boom through a bracket at two locations. The antenna elements are preloaded against the boom by tightening the rope. The tension in the rope is controlled using preload bolts and monitored using the strain gauged load links. The rope is also touching the heating wire (SS 304 wire of dia 0.25 mm) routed through a block made of Machinable glass ceramic. The release of the antenna elements are achieved by fusing the rope using the heating wire. Two sets of heating wire are provided on the block to improve system reliability.

Deployment mechanism: The deployment mechanism moves the dipole and reflectors on release from the stowed configuration to the final position. Torsion springs are mounted at the hinge joint of dipole and reflectors to give necessary energy for deployment and also the necessary preload at the deployed condition.



Indian Block of Stations:		
Station	Lat. (°N)	Long. (°E)
Trivandrum	8.5	77.0
Bangalore	13	77.6
Hyderabad	17.8	78.0
Bhopal	23.2	77.2
Delhi	28.6	77.2

Russian Block of Stations:		
Station	Lat. (°N)	Long. (°E)
Norilsk	69.2	88.6
Turukhansk	65.46	87.56
Tomsk	56.3	84.55

8.2.5.3 LiVHySI Payload

In recent years the need for a comprehensive understanding of the complex processes of the ionosphere-thermosphere system, including its response to the various external forgings so as to reach a level of predictive capability, has been felt. As is known, the information regarding the thermosphere can be obtained through atmospheric emissions known as 'Airglow' while ionosphere variability's can be studied through radio wave propagation characteristics. As a consequence, simultaneous measurements of (i) airglow emission intensity from the menopause (height region around 90 km), ionosphere-thermosphere and, (ii) electron density distribution would provide important insight into the generation mechanisms and evolution of these processes. In this context, the combination of LiVHySI and RaBIT would provide excellent simultaneous measurements of neutral and plasma parameters respectively, complementing each other and also the solar radiation measurements through SOLRAD. Both these Indian experiments are the first of its kind indigenously built experiments onboard an Indian satellite.

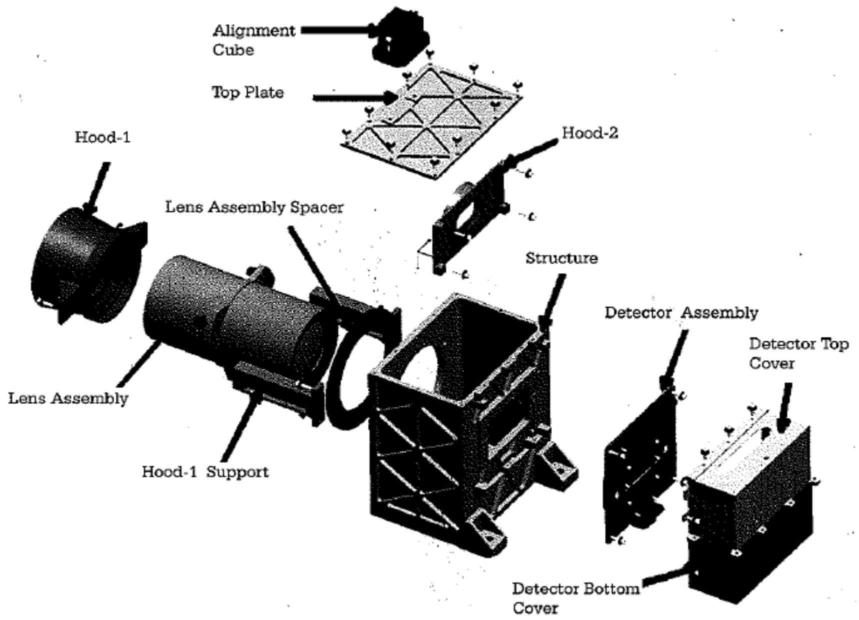
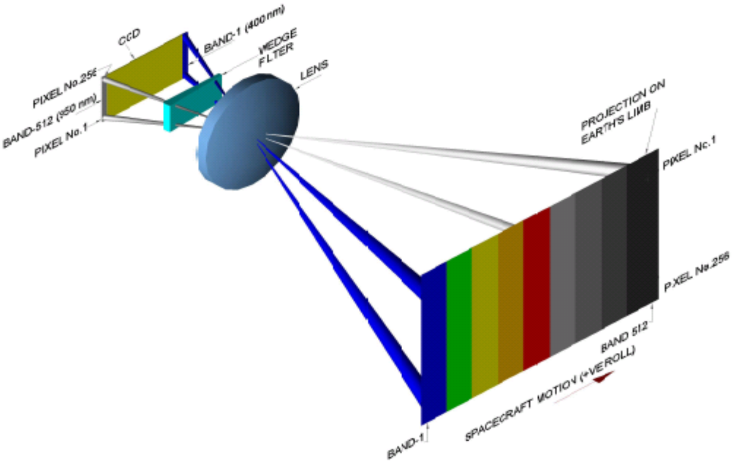


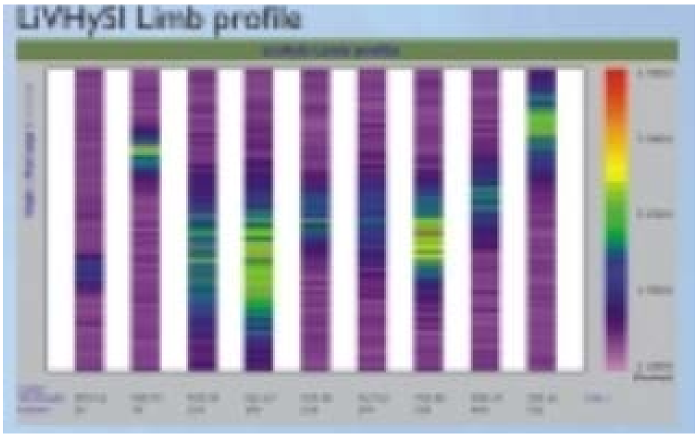
Figure 8-9: Exploded view of LiVHySI

The terrestrial upper atmosphere i.e. about 80 km to 1000 km is a closely coupled two component system where the neutrals (thermosphere) and plasma (ionosphere) coexist with linkages to magnetosphere higher above and the lower atmosphere below. This region consists of ionized matter (ionosphere) and neutral matter (thermosphere in the form of atoms and molecules). This part of the earth's atmosphere responds sensitively the solar radiation and wind reaching the earth through the interplanetary space. This region is controlled primarily by the solar EUV radiation through atmospheric heating, photo dissociation and photo ionization of the atmospheric species. The electrons and ions constitute the electrically conducting ionosphere with the neutral atmosphere (thermosphere) dominating the background.



SCHEMATIC OF SPECTRAL BAND PROJECTION ON EARTH'S LIMB

The state of thermosphere-ionosphere region at any given time and location is determined not only by chemistry but also by the transport through neutral winds, electric fields and field-aligned plasma diffusion. For instance, solar wind–magnetosphere interactions cause significant changes in the energies of this region over high latitudes. Over low and equatorial latitudes, the scenario is even more complicated as the energetic and dynamics is affected not only by the direct solar forcing but also by the non-local forcing from the high latitudes and the atmosphere lower below it.



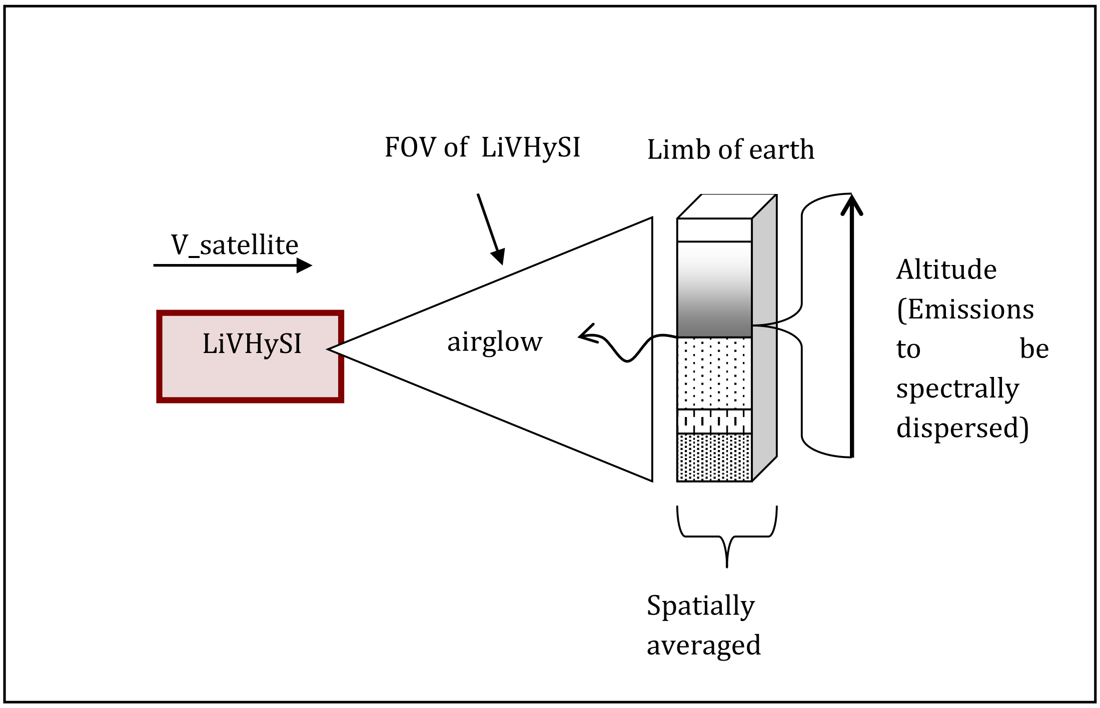
The individual constituents of the atmosphere whether they are atomic and molecular in nature play important role in the process of upper atmospheric energy balance. The lifetime of most of these species, to a large extent, are controlled by the photochemical processes involving them. A number of the atmospheric species get excited and undergo specific spectral transitions as a result of these processes. Consequently, atomic and molecular emissions occur depending on the lifetime of the meta stable state and the timescale of the ongoing quenching reactions. These atmospheric emissions are known as the `Airglow'. The broad classification of the airglow phenomenon is day glow, nightglow or twilight glow depending on the time of the day it is being observed. Twilight glow can also be termed as the day glow as seen from the night sky. The phenomenon of excitation and de-excitation over the polar-regions is known as Aurora (Aurora borealis for northern and Aurora austral is for southern polar region). Though, the variations in the airglow intensity would primarily be caused by the changing relative contribution of the various chemical channels causing the particular excitation, transport effects would also modulate the observed airglow intensity at any given time. As a consequence, these airglow emissions thus serve as a perfect tracer for the processes occurring in the altitude regions from which they emanate. LiVHySI is a wedge filter based instrument that is capable of making simultaneous measurements of the intensity of many airglow emissions at different wavelengths, emanating within 80-600 km altitude region within the limb of earth. The airglow emissions that are of interest to us are listed here giving details of the emitting species and corresponding wavelengths.

Wavelength(λ in nm)	Emission Altitude Range(km)	Type of emission
First negative band of N2427.8	120-250	Band
NI 520.0	100-220	Atomic
OI 557.7	90-120 & 150-200	Atomic
NaI 589.0	90-100	Atomic
OI 630.0	160-500	Atomic
OI 636.4	160-500	Atomic
OH 731.6	80-98	Band
OII 732.0	100-200	Atomic
OH 740.2	80-98	Band
O2 762.0	80-100	Band
OI 777.4	250-350	Atomic
OI 844.6	250-350	Atomic
O2 864.5	100-200	Band

8.2.5.3.1 Payload Operating Principle:

The 'Limb Viewing Hyper Spectral Imager' referenced to as LiVHySI would be continuously imaging the earth's limb along the meridian between ~ 80-600 km as it moves in the polar orbit. As mentioned earlier, the atmosphere in this altitude region is emanating a range of prominent airglow emissions at different wavelengths maximizing at different heights. The details of these emissions have been given in Table above. Further, the signals coming from the limb of earth would be spectrally dispersed only along the altitude while spatial variability that could exist horizontally would be time averaged depending upon the temporal resolution of the proposed measurements, since the satellite is moving with a velocity of ~8-10 km/s. In this context, the 256 pixels side of the detector would be aligned vertically (Yaw axis) along the altitude axis, while the 512 pixel side i.e. wavelength side of the detector would be employed horizontally (Band 1 is along +Roll side, Band 512 is along –Roll side). In this configuration, the imaged airglow emissions would not only provide the altitudinal distribution of the emitting species but also give us an insight into the physical, chemical and transport processes operating at different altitude regimes of upper atmosphere.

The main objective of the instrument is to perform airglow measurements in the Earth's upper atmosphere (80 to 600 km) in a spectral range of 450 nm to 950 nm. The observations would be carried out in the earth's limb viewing mode with a range of about 3172 km from a LEO sun-synchronous polar orbital platform (altitude of 817 km). Sensor Development Area (SEDA) at Space Applications Centre has developed this Hyper-spectral Imager as a part of scientific payload onboard Youthsat. This instrument has taken the advantage of the design and development of similar instrument hardware that was developed at SAC and used in Chandrayan-1 and IMS-1 missions.



Global coverage of air glow measurements is required to generate the required database to study and understand various aspects of the space weather. This is possible by satellite based observations. The Earth's Limb viewing geometry is chosen because it provides a number of advantages as compared to the nadir viewing geometry. The horizontal line of sight through the Earth's limb contains up to sixty times more emitting material than a corresponding nadir view, providing greater sensitivity for measurement of tenuous species. The combination of the spherical geometry of the Earth's atmosphere and the exponential decrease of gas density with altitude provides data heavily weighted around the tangent point altitude of viewing and also provides high vertical resolution. Further, the background viewed by the instrument is cold blackness of the space, which reduces the dark signal and noise and hence simplifies data interpretation.

This imaging spectrometer is based on a wedge filter as a dispersive element placed very close to an Active Pixel sensor (APS) area array which in turn is placed at the focal plane of F#2, f = 80 mm telecentric lens system. The principal advantages of wedge spectrometer approach are its relative simplicity, lack of complex aft optics, a compact and easily ruggedized instrument design, uncomplicated layout that results in minimal sensor integration and test time, reduced cost and delay time.

The estimated radiometric performance of the proposed instrument is < 50 Rayleighs at noise floor through the signal integration for 10 seconds. The pixel projection turns out to be 2 km/pixel at a range of about 3172 km with altitude coverage of 80 km to about 600 km and horizontal swath of 1024 km from a spacecraft altitude of 817 km. The spectral sampling distance is 1.1 nm. The observations could be carried out only during the eclipse due to the constraints imposed by the observational modes of other on-board SOLRAD payload.

The subsystems of the LiVHySI P/L are:

- Optics (f/2, f=80mm, telecentric lens)
- Wedge filter
- APS (Active Pixel Sensor) & its Temp. Controller
- Camera electronics
- Power supply
- EOM structure

Payload subsystems are detailed below:

Optics: The imaging system for LiVHySI consists of a collecting optics, a wedge filter, an APS area array and the associated electronics. The optical design consists of a single lens assembly. This optical design utilizes eight lenses, consisting of four types of Schott glasses. A telecentric design, in which the principal ray at all the field angles, is parallel to the optical axis, ensures that the angle of incidence on the band pass filter is nearly the same for all the wavelengths. Keeping the angle of incidence close to normal to the filter reduces the complexity of filter coating.

Detector head Assembly: LiVHySI DHA (Limb Viewing Detector Head Assembly) of Youthsat consists of512*256 elements Silicon based Area array Active Pixel Sensor (improved version of the sensor used in Chandrayaan-1 and IMS-1). Desired system sensitivity is achieved by fast optics and integrating the sensor for long duration. The sensor temperature is maintained with tight tolerance (21 ± 0.1°C) using heater and thermistor in close loop to minimize dark signal variation. The DHA responds to optical radiation covering spectral region from 450nm to 950 nm. Wedge filter is placed very close to the sensor array for obtaining spectral separation (512 spectral bands) along the row direction.

Thermal: The sensor temperature is maintained with tight tolerance (21 ± 0.1°C) using heater and thermistor in close loop. The temperature control system consists of two number of thermo foil heaters (type: MINCO HK5537R26.1L12E, 26.1ohm, 20W/inch, 12.7mm dia. Circular patch) and two numbers of thermistors (type: YSI44906/44907,one for the control loop and one for temperature monitoring). These components are mounted directly on back surface of the APS.

Camera Electronics - Limb viewing hyper-spectral imager (LiVHySI) consists of two trays stacked together.

1. Camera Electronics Main tray (PLE21)
2. Temperature Controller tray (PLE22)

CE configuration is similar to IMS-1-HySI with changes carried out in FPGA logic design to meet LiVHySI configuration requirements. In addition, a temperature controller is added as a part of CE to minimize dark signal accumulated due to large integration time.

Following are the main changes in Youthsat-LiVHySI CE w.r.to IMS-1 Camera Electronics.

- All 512 bands data is transmitted as compared to 64 bands in IMS-1
- WLS from BDH is with minimum 8 skips as compared to IMS-1
- No TC/TM interface
- All dark pixels included in the data format from CE to BDH without change in data rate
- 12 LSB's out of 16 bit output data represent valid data with 4 MSB's stuffed to logic "1"
- Inclusion of temperature controller to control detector temperature at 21°C with setting accuracy of ± 0.5 °C and control accuracy of ± 0.1 °C

8.3 Microsat

8.3.1 Introduction

Microsatellites (Microsats) are small satellites with small volume, low power requirements and weighing 100 kg. MICROSAT is an advanced optical remote sensing satellite for providing spot imageries with a high spatial resolution in the panchromatic band and night imaging using IR payloads. Payload requirements are met by choosing Sun-Synchronous near Earth orbit of 300 Km x350 Km with 10:30 AM descending node local time. The satellite is designed with heritage from IMS-1 and Youthsat.

The payload data will be transmitted in X-band after QPSK modulation and through SSPA. The total power generated is about 287W in sun-pointing mode during summer solstice. The payload will consist of one high-resolution PAN band and two IR bands for day and night imaging. Considering piggy-back launch, the satellite may be injected into 500km orbit. Over a period, with natural decay, the orbit will reach to 300Kmx350km. The mission life will be about 6 months to one year.

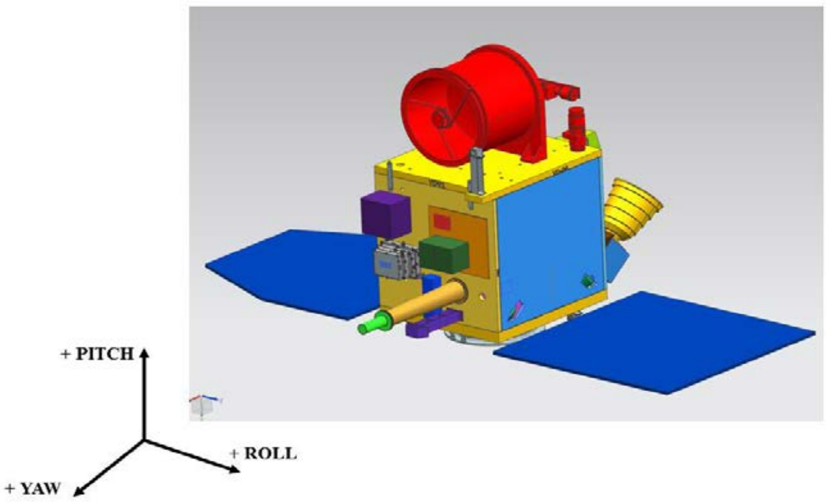


Figure 8-10 Microsat Deployed Configuration

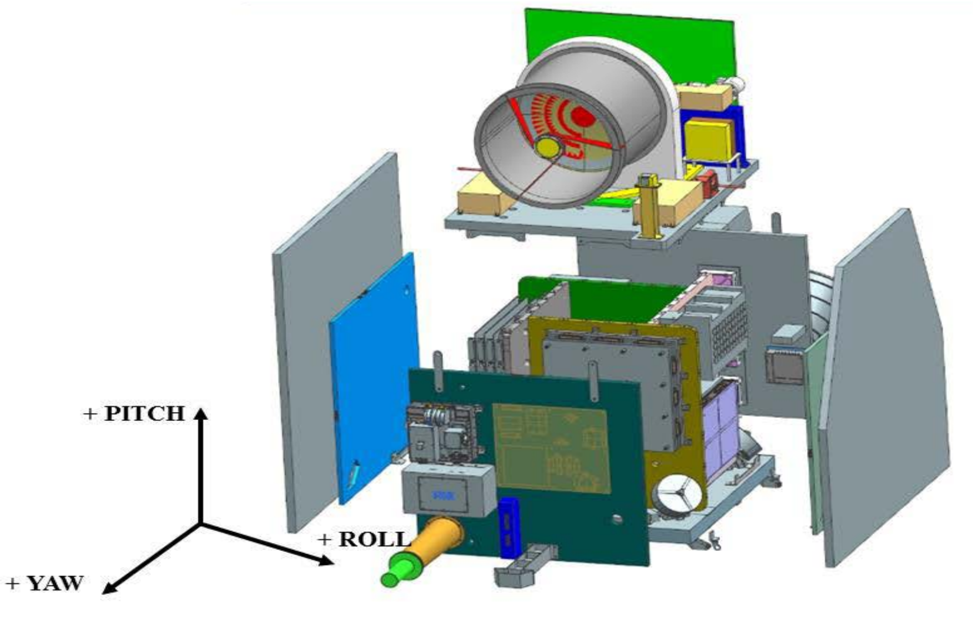


Figure 8-11 MICROSAT Exploded View

8.3.2 Mission objectives

The main objective of small micro satellite series:

- To design and develop a high agility advanced satellite with a high spatial resolution of around 0.8 m in the panchromatic band and provide better than 1 meter ground sample distance in panchromatic.
- To design and develop night imaging using IR payload.
- To meet the ever-increasing user demands for cartographic applications at cadastral level, urban and rural management, coastal land use and regulation, utilities mapping and development and various other LIS and GIS applications.

The major payload goals are defined as:

- Imaging capabilities of 0.78 m PAN band and 6 m MWIR and LWIR bands from an altitude of 300 Km
- Swath coverage of 3Km for PAN band and 2Km for IR bands
- Step and stare imaging for meeting high resolution and SNR improvement.
- Size, weight and power requirements based on modified IMS-1 satellite configuration

8.3.3 Orbit Details

Table 8-3 Orbit details of MICROSAT

Parameter	Value
Spacecraft Mass	120 Kg
Payload Mass	~40 Kg
Spacecraft Size	55.2 cm x 60 cm x 60 cm
Average Power Generated	287 W
Average Payload Power	6 W (PAN) 26 W (MIR) 91 W (LWIR)

Altitude	300 - 500 km
Orbit	LEO
No. of orbits/day	15 22/27
Inclination	96.756° (Nominal)
Local time	9 hrs 30 min (descending)
Orbital Period	91 min
Swath	3Km (PAN) 2 Km (IR)

8.3.4 Salient features of MICROSAT

Table 8-4 Features of MICROSAT System

Subsystem	Specification
Mass	120 Kgs
Payloads	PAN, MIR & LWIR Cameras
Structure	Aluminum Honeycomb sandwich panels as a Cuboid structure
Thermal Control	Flexible Optical Solar Reflectors (OSR), Multi-layer Insulation blankets (MLI) with Beta cloth
Mechanisms	Solar Panel – HDRM and Hinge Assembly
AOCS	
Pointing Accuracy	0.1° Drift Rate (Mean DC) for 0.1 Pixel (PAN) over 700 µs is 0.0213° /s Drift Rate (Mean DC) for 0.1 Pixel (MIR) over 2 ms is 0.0573° /s Drift Rate (Mean DC) for 0.1 Pixel (LWIR) over 0.4 ms is 0.2865° /s
Sensors	Micro Star Sensors 4-PI Sun Sensors Magnetometer IRU Gyros (2 Nos) with two channels in each Gyro
Actuators	1N Thruster (One No, Central Mounted) Reaction Wheels of 1.0NMS @ 6000 RPM and 0.02 Nm Magnetic Torquers (2Nos of 12.0 A-m^2)
Power System	
Solar Panels	2*1 | Azur cells | 286W (BOL)
Battery	15.6 Ah Li-ion | 10Sx6P
Electronics	Bus Voltage – 30-42 V Expected Load Current - 7A@35V
TT&C	
Telemetry	S-Band | 16 kbps | BPSK

Telecommand	S-Band | 4 kbps | FM/PCM/PSK
RF System	
Payload Data System	X-Band | 2*16 Mbps | QPSK
SPS	12 Channel | L1 (1575.42 MHz) | C/A Code
SSR	**32 Gb**

8.3.5 Payload

The payloads consist of a PAN camera operated in spectral band of 0.5 μm to 0.85 μm with a resolution of about 0.78m with a step-n-stare (SNS). It will have swath of 3.2km. This camera can be used during sun-lit portion of the orbit. It has two IR bands: MIR: 3.7μm to 4.8μm and LWIR: 7.7μm to 11μm. The resolution is about 6m. Satellite can provide a step-n-stare factor up to 1:6.19. This camera can be used during eclipse portion of the orbit and during day and night to cover the required area. Table 8-5 below gives the mission requirements.

Table 8-5 Mission Requirements/Goals

Parameters	PAN	IR
GSD (m)	0.78 @ 300 km	6 2@ 300 km
Swath (km)	3.2	~ 2
Spectral Range (μm)	0.5 - 0.85	MIR – 3.7 – 4.8; LWIR – 7.7 - 11
SnS Factor	Upto 1:6.19	
Data Rate (Mbps)	< 32 Mbps, X Band	
Power (W) (Unregulated)	7	MIR: 23 (42 during Cooling) LWIR: 78 (113 during Cooling)
Weight (Kg)	~ 40	

Table 8-6 Major Payload Parameters

Parameter	PAN	MIR	LWIR
Wavelength Range (μm)	0.5 – 0.85	3.7-4.8	7.7-11
Spatial Resolution(m) @ 300 km	0.78	6	6
Swath (km) @ 300 km	3.2	~2	~2
Pixel Size (μm)	7 x 7	30 x 30	30 x 30
Pixel elements	4096	320 x 32	320 x 32
Focal length (mm)	2692	1500	
Optics Aperture Diameter (mm)	300		
FOV	0.6°	0.37°	
Saturation radiance / temp (K)	100% albedo	375	375
Quantization (bits)	10	12	14
SNR/NEdT	> 100	250mK@300K scene	250mK@300K scene
MTF at Nyquist (%)	> 5	>10	>1
Step and Stare ratio	6.19 (ground track velocity of 7.38km/s)		

Integration time (µs)	654	<2000	<400
Number of Ports	4	1	1
Data rate (Mbps) with SNS	64	0.85	1

Microsat payload consists of telescope, focal plane, camera electronics and mechanical systems.

All the three payloads share a common telescope, which is based on RC system with two-mirror configuration. Each band has its own aft optics for focal length adjustment (PAN and IR bands requires different focal length) and field correction. Primary optics has clear aperture of 300mm. Effective focal length of 2692 mm is realized for PAN band and 1500 mm for MIR and LWIR bands based on the selected detectors. Focal plane uses flip mirror configuration or germanium beam splitter configuration for splitting PAN and IR chain and dichroic beam splitter for further splitting IR chain into MIR and LWIR bands. PAN focal plane is configured with a linear CCD while MIR & LWIR focal plane is configured with MCT area array detectors. PAN detector is available as 3D integrated package in which major electronic functionalities are implemented. Camera electronics for IR bands consists of front-end electronics, data handling electronics and cooler drive electronics (CDE). A honey-comb optical platform will act as the main structure. Figure 8-12 shows system level block diagram.

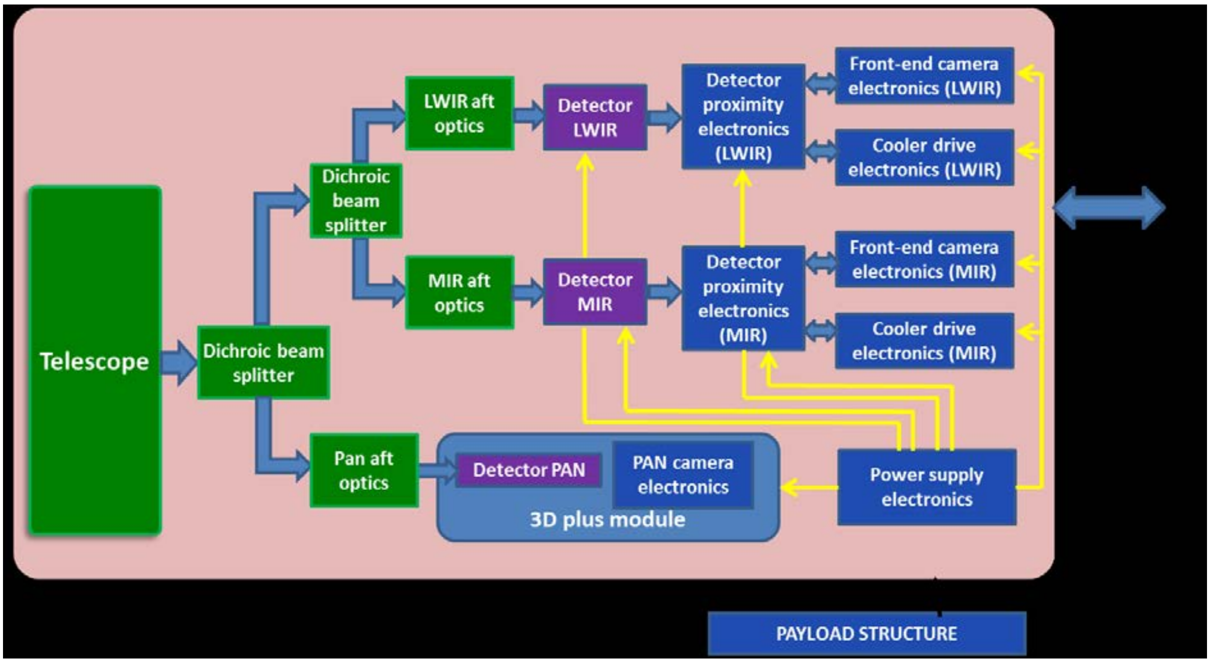


Figure 8-12 System Level Block Diagram of Payload

8.3.5.1 Optical System (Telescope)

Microsat payload consists of three different channels – one each in visible, MIR and TIR spectral range. The specifications of Microsat in three channels are:

Table 8-7 Microsat Detector Specification

Parameter	Visible	MIR	TIR
GIFOV (m)	0.78	6	6
Swath (km)	3.2	1.9	1.9

Spectral Range (µm)	0.5 to 0.85	3.7 to 4.8	7.7 to 11
Detector	4 k linear array	320 X 256 (32)*	320 X 256 (32)*
Pixel Size	7 µm	30 µm	30 µm
Spacecraft Altitude	300 km		
Aperture Diameter	300 mm		

* 32 rows out of 256 will be used in the along track direction

Table 8-8 Microsat Optical Specification

Parameter	Visible	MIR	TIR
EFL (mm)	2692	1500	1500
Fields of view	± 0.30	± 0.1850	± 0.1850
F/No.	F/8.97	F/5	F/5
Spatial Frequency	72 lp/mm	17 lp/mm	17 lp/mm
Spectral range (µm)	0.5 to 0.85	3.7 to 4.8	7.7 to 11

Two mirror systems with FCO is used for the design of PAN camera of Microsat due to its small size, relatively less complex fabrication and alignment and preference of fabricating these components. Beam splitter has been used to accommodate the IR channel together with PAN.

In IR optical design, IR beams will be refracting using germanium beam splitter in place of reflecting by flip mirror. There are six powered aft-optics elements in each IR channel. The first three optical components are common to MIR and TIR channels and are used to collimate the beam after the primary focus of the telescope. A fold mirror has been used to compress the physical envelope of EO module. A second beam splitter is kept in the collimated path to separate out the MIR and TIR channels. A set of 3 elements for each channel is used to focus with required magnification.

8.3.5.2 The Detector

Microsat payload consists of one panchromatic (PAN) camera operating in visible band and one infrared (IR) camera operating in two spectral bands: one in Mid Infrared (MIR) range and other in Long Wave Infrared (LWIR) range. The main factors that define the type and format of detector required for these bands are spectral region of interest, required radiometric performance (SNR/NEDT), required ground sampling distance and swath from given platform height, optics focal length, frame rate, operating temperature etc.

Table 8-9 Major Payload Parameters for Detector Selection

Parameter	Value		
	PAN	MIR	LWIR
Spectral band (µm)	0.50 – 0.85	3.7 – 4.8	7.7 – 11.0

GSD (m) (from 300km altitude)	0.78	6	6
Swath (km) (from 300km altitude)	3	~2	~2
Focal Length (mm)	2692	1500	1500
SNR/NEdT	>100 @ 100% albedo	<250mK @300K target temp.	<250mK @300K target temp.

8.3.5.3 PAN Camera

PAN camera operates in spectral region of 0.5µm - 0.85µm with a GSD of about 0.78m from platform height of 300km. For achieving 3km swath with the given 0.78m GSD, the required minimum number of detector pixels along swath direction is about 3847. Only one row of pixels is sufficient to generate an image in pushbroom mode. To meet the SNR requirements at 100% albedo imaging will be carried out in step-and-stare (SNS) mode. Required SNS for PAN camera is 1:6.19.

4K linear CCD (SC3925A) detector is a 4096 pixels visible linear imager with high speed operation and high dynamic range. Complete camera module for this detector has been developed using 3D packaging technology. The key advantage of 3D packaging of camera module is 80% reduction in volume and 65% reduction in weight compared to conventional discrete hardware used in IMS-1.

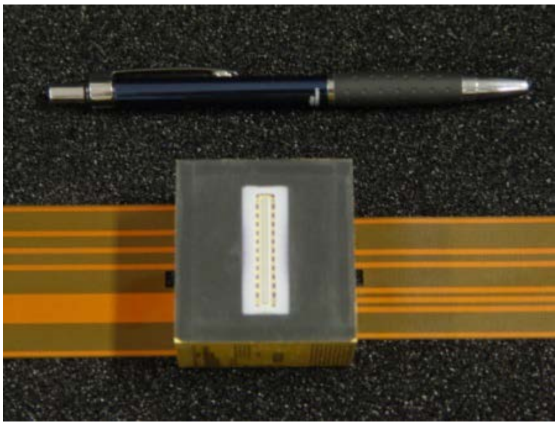


Figure 8-13 4K Linear CCD 3D plus Camera Module

Table 8-10 Major Specifications of 3D plus Camera Module

Parameter	Value
Detector	SCL 4K Linear CCD (SC3925A)
Pixel Material	Silicon
No. Of pixels	4096
Pixel Size	7 µm x 7 µm
Pixel Pitch	7 µm
Data Format	10bit serial
Data Rate	64 Mbps
Dark Noise	≤1 LSB

SNR	≥ 100
Weight	≤ 150 gm
Regulated Power (Typ)	3 W
Volume	52 x 49 x 28 mm^3

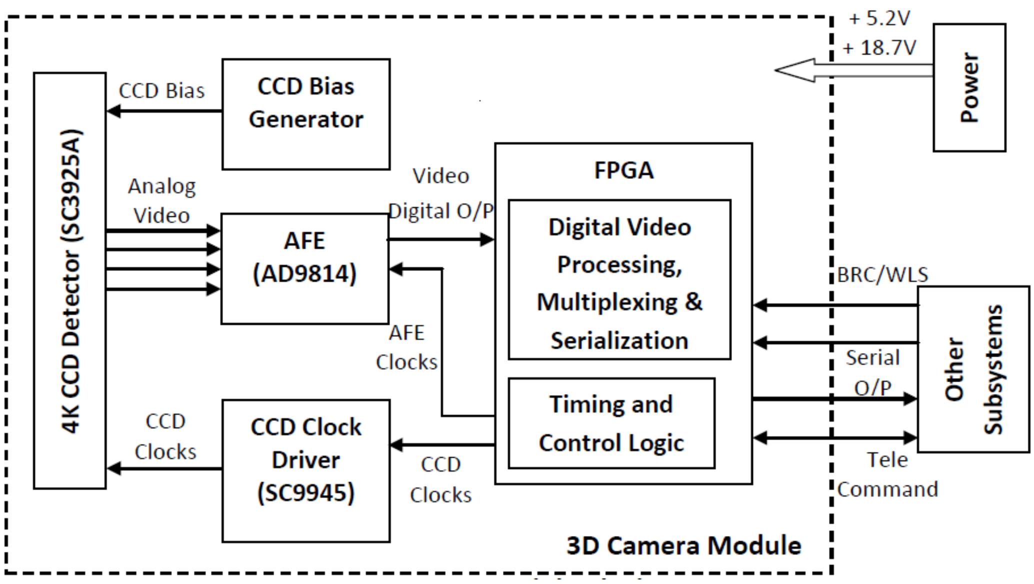


Figure 8-14 3D Camera Module Block Diagram

The camera module consists of:

(i) **SCL make 4K linear CCD (SC3925A):** It is an indigenously made high performance 4096 element linear array CCD which converts incident radiation into electrical charge. It has four buried channel CCD shift registers and four output amplifiers. The charge generated at the photodiodes is stored under storage gate and is transferred to the shift registers by applying a transfer clock. This CCD requires six different regulated biases ranging from 3V to 15V and four different clocks for its operation.

Typical electro-optical characteristics of the CCD are given in Table-8-11.

Table 8-11 Typical Electro-optical Characteristics of CCD (SC3925A)

Parameter	Symbol	Unit	Value
Saturation Output Voltage	Vsat	mV	800
Photo response non-uniformity	PRNU	%	≤±7
CTF @ 70 lp/mm	CTF	%	64
Responsivity	R	V/µJ/cm2	3.5 – 5
Conversion Gain	CG	µV/electron	2.2
Non-Linearity	NL	mV (rms)	≤ 0.7

(ii) **CCD Bias generator:** CCD requires six different biases ranging from 3V to 15V for photodiode, shift registers and output amplifiers operation. LM2914 bias generator inside the camera module generates all the regulated and low noise bias voltages required for CCD operation.

(iii) **SCL make CCD clock driver (SC9945):** It shifts the level of CCD clocks generated by FPGA to 13V (typical) MOS level required for CCD operation.

(iv) **Analog front end (AFE) (AD9814):** It digitizes the CCD analog video output. Two AFEs are used in camera module to process the CCD 4 port analog output.

(v) **FPGA based logic and data processing:** It generates clocks for CCD and AFE operation, serializes AFE output data and inserts camera status as telemetry in output data stream.

8.3.5.4 IR Camera

The infrared (IR) camera operates in two channels MIR: 3.7μm – 4.8μm and LWIR: 7.7μm – 11.0μm. Cooled IR detectors offer high sensitivity and high speed operation over their uncooled counterparts. Integrated Detector Dewar Cooler Assembly (IDDCA) is generally used for operation and characterization of such detectors. The focal plane array (FPA) is contained in a vacuum-sealed case or Dewar which is then cryogenically cooled using an active micro-cooler.

For both IR bands, the required GSD from 300km altitude is 6m. With effective focal length of 1500mm the pixel size shall be 30μm for imaging from 300km. Swath of 2km with 6m GSD requires an array of 334 pixels.

The IR Detector Head Assembly in general comprises of an IDDCA, proximity electronics, interface to camera electronics, cooler drive electronics, vibration damping assembly and heat sink.

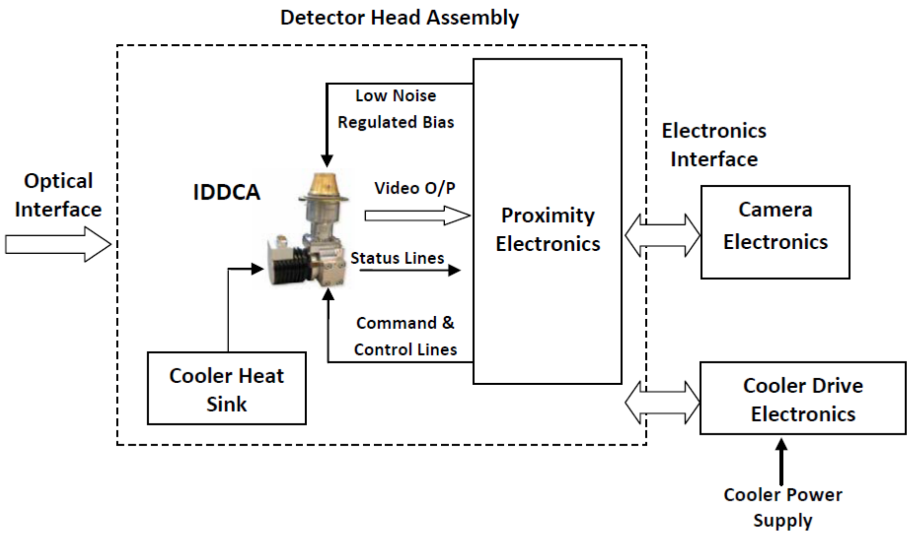


Figure 8-15 IR DHA Block Schematic

8.3.6 Payload electronics (PE)

Microsat Camera Electronics (CE) configurations are worked out based on system inputs and detector requirements. CE is planned to be realized considering following system and detector inputs. Table 8-12 provides system specifications and goals.

Table 8-12 System specification and goal for Camera Electronics

Parameter	PAN	MIR	LWIR
Detector	SCL 3925A	IR with IDDCA	IR with IDDCA
Number of Ports	4	1 or 4	1 or 4
Pixel Size (um)	7	30	30
Pixel elements	4096	320x256	320 x256
Pixel Readout	4096	320x32	320x32
Pixel Readout (MHz)	1.6	1	1
Integration time (ms)	0.654	2	0.4
Frame time (ms)	-	145	145
Step and Stare ratio	6.19		
Quantization (bits)	10	12	14
Electronics SNR (goal)	> 1000	> 4000	> 8000
Data Format	Serial		
Total Data rate (Mbps)	64	0.85 (16 in burst)	1 (16 in burst)
SNR /NEDT	> 100	250mK @ 300K Target Temperature	
Unregulated Payload Power (W)	7	23 (42 during cooling)	78 (113 during cooling)

Camera Electronics primarily consists of the following.

A. Detector Interface board (DIB): This card will be used for interfacing detector. A flexi cable will carry all signals from and towards FECE card.

B. Front End Camera Electronics (FECE): It consists of bias circuits for op-amplifiers, buffers, detector bias generator, digitizer and FPGA based logic and control electronics. FECE receives master clocks BRC and WLS from BDH and generates various clocks for detector and digitizer operations. It processes digitized data and transmits serial data to BDH.

C. Cooler Drive Electronics (CDE): It consists of control and drive electronics for cooler and controls the cooler temperature of IR detector to maintain to a desired temperature levels with feedbacks coming from IDDCA temperature sensors and position sensors.

D. Power Supply Electronics (PSE): PSE receives the raw bus supply and distributes the regulated power supply to different sections of CE.

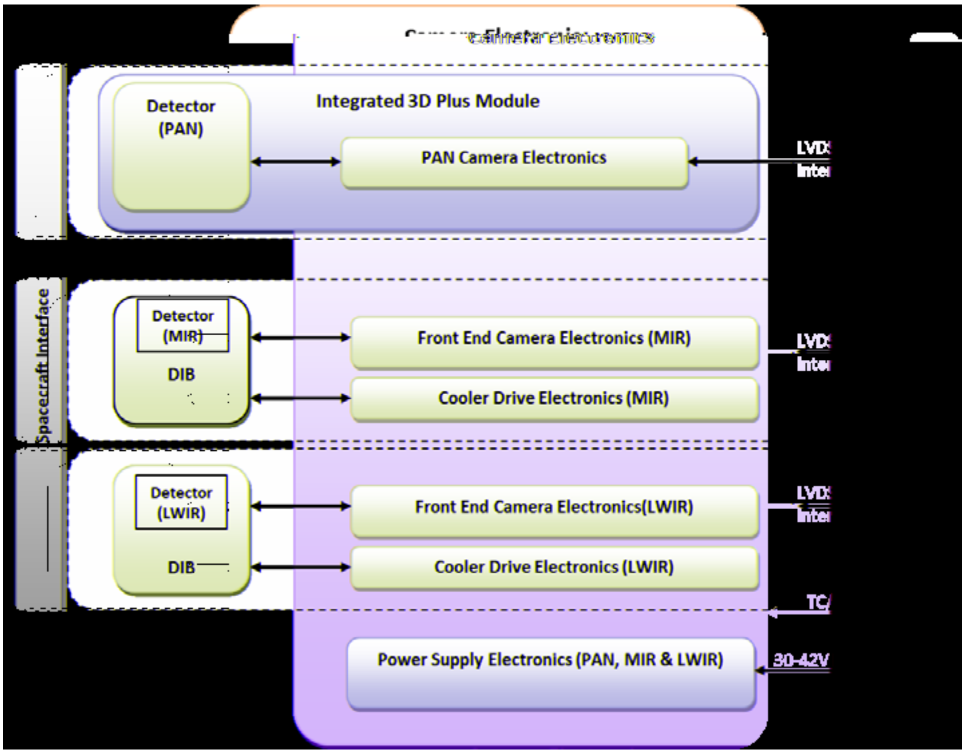


Figure 8-16 Microsat Camera Electronics

PAN electronics is realized using miniaturized 3D Plus module and PSE. IR camera electronics are realized using DIB, FECE, CDE and PSE.

8.3.6.1 PAN CE

PAN camera is developed using available 3D plus module. 3D plus module is miniaturized fully integrated detector and camera electronics (excluding power supply). It consists of indigenously developed SCL make CCD linear detector, SCL make clock driver, analog front-end, and FPGA based Timing Logic and data processing with standard BDH and BMU interface. Key advantage of using 3Dplus module is 80 % reduction in volume and 65 % reduction in weight as compared to conventional hardware. Figure 8-17 shows 3D plus module and Figure 8-18 shows functional blocks implemented in the integrated camera module.

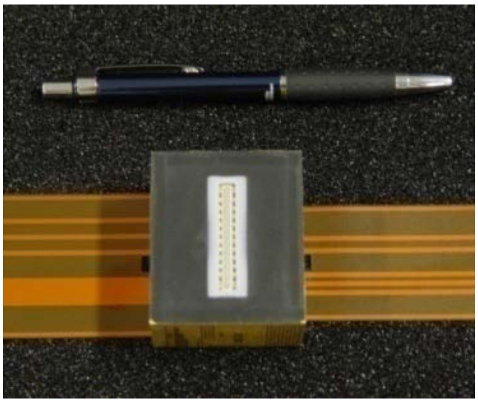


Figure 8-17 D Plus module for PAN camera electronics

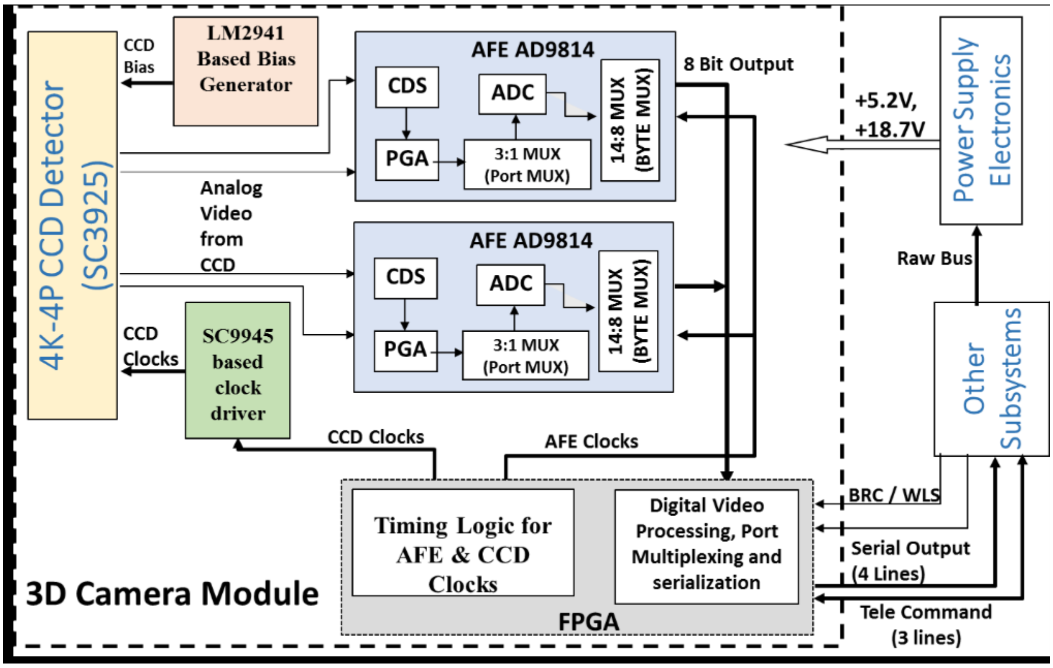


Figure 8-18 Block diagram of 3D plus Module

Table 8-13 Salient feature of 3D module

Parameter	Value
Programmable gain	6-bit
Quantization	10 bits
Data rate (Mbps)	1 to 64
Data Format	10 bit Serial
Dark noise	< 1 LSB (@10 bit)
Near Saturation SNR	> 500
Weight (gm)	<150
Power (typ, W)	3
Volume (mm3)	52 x 49 x 28

The integrated module consists of SCL make CCD detector, CCD bias generator, SCL make clock driver, AFE for digitization and FPGA based control and logic electronics. Programmable gain and offset features exists in camera electronics. If the default gain value is acceptable then no tele-command may be required. However provision exists in changing the gain with tele-command. The integration time is fixed at 20920 TPs (1BRC clock =1TP).

Unregulated power supply lines from spacecraft and PSE can be realized as a common PSE.

8.3.6.2 IR CE

Camera Electronics for IR bands is custom designed meeting the functional and performance requirements of the system for the identified detectors. MCT detectors are identified for MIR and LWIR band. MCT detectors are cooled detectors and require integrated cryogenic cooling mechanism for their operation. It has Integrated Detector Dewar Cooler Assembly (IDDCA). Although detector array has 320x 256, only 320x32 pixels of detector will be read out considering swath and resolution requirements. CE functional blocks are realized in DIB, FECE, PSE and CDE. Considering same ROIC for two IR bands, DIB and FECE are modular.

However in MIR, rotary BLDC (Brushless DC) motor based Stirling cooler is planned. In LWIR, following two coolers are under considerations.

- BLDC rotary motor based Stirling cooler, same as MIR (option-1)
- Linear motor based split-stirling cooler (option-2)

At present, CDE-LWIR design configurations are worked out for both options. Design for CDE-MIR and CDE-LWIR (option-1) will be same except thermal load.

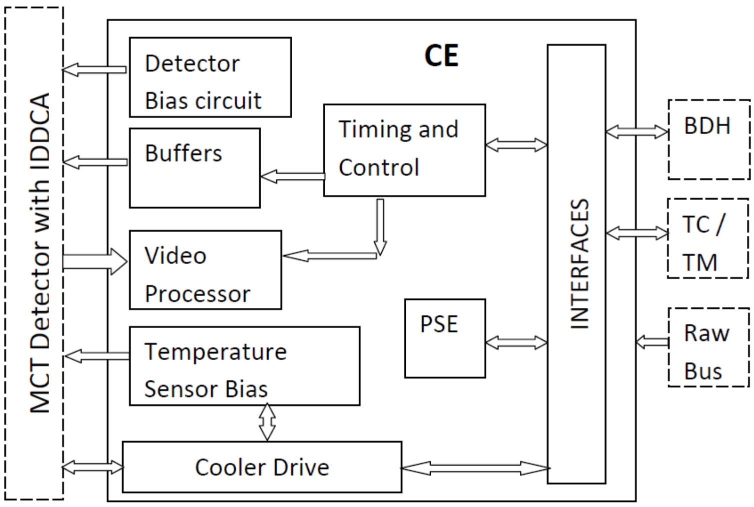


Figure 8-19 IR Camera Electronics Block Diagram

8.3.6.2.1 Detector Interface Board (DIB)

The basic function of DIB block is to interface detector pins. Detector along with IDDCA shall be available in its integrated form and only detector pins will be available. A PCB with cut-out is required to make connection with detector. Flexi-rigid based PCB is planned and shown in the figure 5-7. There will be minimum handling of DIB card after detector is mounted. Flexi-length will be kept as short as possible.

Mechanical: Estimated DIB card size is 60 x 60 mm^2 (detector housing rigid part) and estimated weight of card is 50 gm (excluding detector). Separate mounting mechanism for rigid PCB having micro-D connector, is envisaged.

8.3.6.2.2 <u>Front End Camera Electronics (FECE)</u>

The basic function of FECE block is detector control signal generation, detector bias generation, video processing electronics like amplification and digitization of detector output, timing-control signal generation, IDDCA temperature sensor biasing and its digitization. The proposed FECE configuration is worked out with following considerations

- Minimum power dissipation
- Use of available components
- No redundancy
- Miniaturize and low weight card
- Shortest possible turn-around time

The configuration and circuit design are carried out with the goal of realizing the detector noise limited system performance.

8.3.6.3 Cooler Drive Electronics (CDE)

IR detector is required to be cooled to low temperature (~90K for MIR, ~70K for LWIR) for its operation. The detector is encapsulated into a vacuum sealed Dewar and the focal plane assembly is cooled using active cooler, based on Stirling cooler. The whole assembly is called IDDCA (Integrated Detector Dewar Cooler Assembly). IDDCA temperature control can be divided into two stages i.e. initial cool-down phase and regulation phase. Imaging will be done during regulation phase. Hence, during cool-down phase detector and FECE will be OFF.

Temperature sensors are incorporated at FPA and their output is used in feedback for closed loop system for maintaining the FPA temperature. Biasing and digitization of temperature sensors is implemented in previous FECE block, for which supply lines and digital control interface will be provided by CDE. CDE also provides TC/TM interface with BMU.

In MIR, rotary BLDC (Brushless DC) motor based Stirling cooler is planned. In LWIR, following two coolers are under considerations.

- BLDC rotary motor based Stirling cooler, same as MIR (option-1)
- Linear motor based split-stirling cooler (option-2)

At present, CDE-LWIR design configurations are worked out for both options. Design for CDE-MIR and CDE-LWIR (option-1) will be same except thermal load.

8.3.6.4 Power supply Electronics (PSE)

Power supply Electronics (PSE) design caters to regulated power requirements of PAN, MIR, and LWIR camera electronics (CE), MIR and LWIR cooler drive electronics (CDE). The spacecraft raw bus is in the range of 30-42 V. The PSE design shall cater for the requirement of operating the following sub blocks independently with different tele commands.

- PAN
- MIR FECE
- MIR CDE

- LWIR FECE
- LWIR CDE

8.4 INS-1A/1B

8.4.1 Introduction

Nano Satellites offer a compelling alternative to large space projects, with a capability to support commercial, governmental, academic applications in a responsive and cost effective manner. Also, Nano Satellites have lot of potential in future and will give rise to lot of new developments in various technologies and are needed from strategic point of view in near future. In view of this, ISRO has developed and is launching the first two ISRO Nano Satellites (INS) namelyINS-1A and INS-1B. These two nano satellites will set a benchmark for development of nano satellites in future on a large scale.

8.4.2 Advantages of Nano Satellites

- Low cost
- Relatively short development period
- Launch on Demand
- Ideal for small scientific payloads
- Multiple satellites can be launched in one launch
- Short time to scientific results
- Devices have a comparable capability
- COTS products and consumer products
- Plug-and-play technology
- Versatile
- Revitalized scientific community
- International cooperation
- Low cost constellation missions possible

8.4.3 Mission objectives

The primary objectives of this mission are:

- Design and develop a low cost, modular Nano satellite Bus in the weight range of 5 kg capable of carrying payloads up to a weight of 5 kg and provide payloads volume of 185mm x 185mm x 160 mm.
- Provide up to 10 Watts of power for payload operation and up to 1 Mbps downlink for data collection
- Provide a 3 axis stabilized system for a control accuracy of ±0.5 degrees via orthogonal RW system
- Provide on-board data handling and storage of 8 Gb
- Provide an opportunity for ISRO technology demonstration payloads
- Provide a standard bus for launch on demand services
- Provide an opportunity to carry innovative payloads for Universities/R&D labs.

8.4.4 Orbit Details

The orbit details for INS-1A and 1B are given in the table 8-14 below:

Table 8-14 Orbit details of INS-1A/1B

Parameter	INS-1A	INS-1B
Spacecraft Mass	8.5 kg	9.7 kg
Payload Mass	3.5 kg	4.7 kg
Spacecraft Size	304 x 246 x 364.3 mm^3 (stowed) 304 x 670 x 364.3 mm^3 (deployed)	304 x 246 x 510 mm^3 (stowed) 304 x 670 x 510 mm^3 (deployed)
Average Power Generated	24W at EOL	24W at EOL
Altitude	505 km	505 km
Orbit	Polar sun synchronous Orbit	Polar sun synchronous Orbit
Inclination	97.4°	97.4°
Local time	9.30 am	9.30 am
Stabilization	3 Axes Stabilization	3 Axes Stabilization
Launch Date	26 September 2016	26 September 2016
Launch Site	SDSC SHAR, Sriharikota	SDSC SHAR, Sriharikota
Orbital Period	94.58 min	94.58 min

8.4.5 Configuration Block Diagram

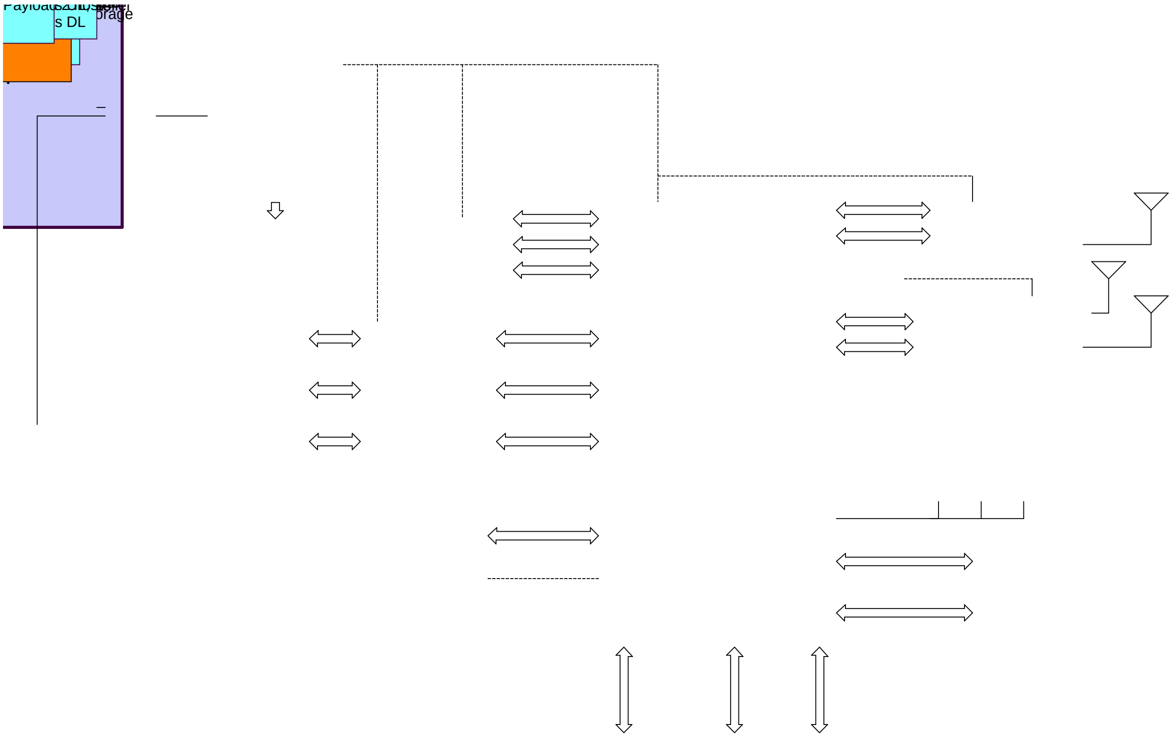


Figure 8-20 INS-1A & 1B Configuration

8.4.6 Salient features of INS Bus

Table 8-15 Features of INS Bus Systems

Subsystem	Specification	
Bus Mass	5 kg	
Payload	SBR & SEUM from SAC for INS 1A	EELA from LEOS MMX-TD from SAC for INS 1B
Structure	Milled aluminium cuboid (185 x 185 x 185 mm^3)	
Thermal Control	Passive (OSR, MLI, Paints, etc,.)	
Mechanisms	Solar panels & Antenna deployment	
AOCS		
Specifications	Pointing : ± 0.5° (3σ) about each axis	
Sensors	Star Sensor - 1 no. MEMS IMU – 1 no. Micro Digital Sun sensor – 1 no.	
Actuators	Reaction wheels – 3Nos. Magnetic Torquers – 3 Nos.	
Power system		

Solar Panels	ZTJ cells with Array power of ~24W at EOL Deployable panel - 250 x 235 x 8 mm^3 Body mounted panel - 163 x 160 mm^2
Battery	5.2 AH Lithium Ion battery
Electronics	Single Raw bus 12 to 16.8 V
TT&C	
Telemetry	Data rate :1.2 kbps Frequency Band : UHF
Telecommand	Data rate :1.2 kbps Frequency Band : VHF
RF System	
Payload Data System	Data Rate: 64 Mbps (After channel coding), QPSK, 4 W SSPA, Frequency: 8300 MHz
SPS	12 Channels, L1 & C/A
SSR	8Gb

8.4.7 Payloads

INS-1A carried a twin payload package from SAC. Science objectives of INS-1A:

1. SBR: BRDF (Bidirectional Reflectance Distribution Function) of the Earth surface. To take readings of the reflectance of different surface features due to sun albedo
2. SEUM to monitor single event upsets (SEU) occurring due to high energy radiation in the space environment in COTS components.

INS-1B carried EELA payload from LEOS and Miniature Multispectral Payload Technology Demonstration (MMX-TD) payload from SAC.

1. EELA: Registration of terrestrial exospheric line-of-sight neutral atomic hydrogen Lyman-alpha flux. Estimation of interplanetary hydrogen Lyman-alpha background flux by means of deep space observations.
2. MMX-TD: Remote Sensing Colour camera with a novel lens assembly for optical realization in a small package. Scope for future scalability and utilization on regular satellites.

8.4.7.1 Surface BRDF Radiometer (SBR) for INS-1A

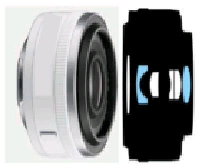
SBR payload is a frame camera operating in visible panchromatic band. It will generate BRDF (Bidirectional Reflectance Distribution Function) of the Earth surface. This type of measurement and study will be unique. It will provide valuable inputs for future payload development in visible range for various land, ocean and atmospheric applications. The measurements could also further help in optimizing the viewing geometry of future imaging sensors. This was a unique experiment from this perspective. In addition, the data could also be used in characterization of cloud-type / height and surface DEM using images with different views.

SBR payload had to observe the Earth from 505 km altitude with about 550 m spatial resolution (along) and 275 km swath. Imaging would be done during sunlit period only.

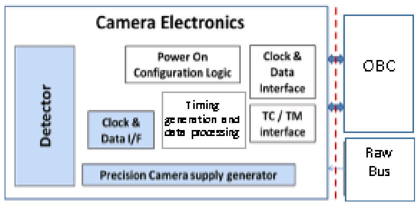
Along track 5 BRDF angles in steps of about 13 degrees to be measured. It was planned to acquire 18 frames in acquisition period of 6 minutes.

Table 8-16 Major Parameters for SBR

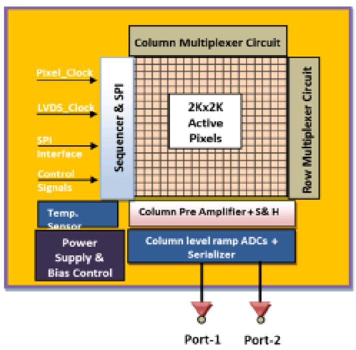
Parameter	Value
Spectral range	PAN (400-700 nm)
IFOV (urad)	550 (GSD = 346 m nadir from 630 km)
FOV (degree)	± 29 (along) x ± 14.5 (across) (Frame Size = 550 km (along) x 275 km (across)nadir from 505 km)
Pixel format	2K (along) x 1K (across)
Frame rate	1 frame every 20 seconds
Quantization (bits)	8
Data Interface	SPI



COTS OPTICS FOR SBR



BLOCK SCHEMATIC OF SBR ELECTRONICS



COTS 4 MEGA PIXEL CMOS DETECTOR FOR SBR

Figure 8-21 SBR – Optics, Electronics and Detector

8.4.7.2 SEU Monitor (SEUM) for INS-1A

SEUM had to monitor single event upsets (SEU) occurring due to high energy radiation in the space environment. This payload was supposed to be on for entire mission life recording single event upsets on the sensor.

Table 8-17 Major Parameters for SEUM

Parameters	Value
Radiation effect	SEU
Operation	Full orbit
Frame rate	1 sample every 2 secs

Quantization (bits)	24
Data interface	SPI

This payload consists of an electronics system with a component which is susceptible to SEU, while all other components are SEU immune. In this way, single event upsets due to heavy ions will be detected and number of such events in a given time interval will be recorded.

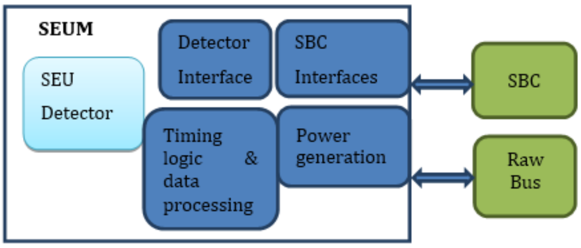


Figure 8-22 Block Schematic of SEUM Payload

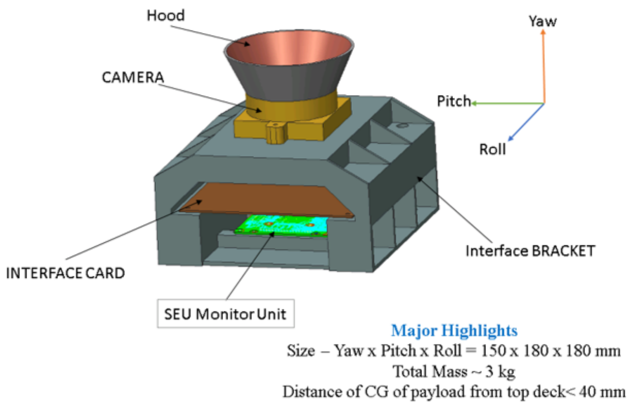


Figure 8-23 ISOMETRIC VIEW OF SBR AND SEUM PAYLOADS

8.4.7.2.1 <u>Overall payload parameters:</u>

Table 8-18 Overall Payload Parameters

Parameter	Value
Size (mm3) (R x P x Y)	180 x 180 x 150
Weight (kg)	3
Power (W)	SEUM: 3.8, SEUM+SBR : 6.7
Data volume per day (Mb)	SBR: 300; SEUM: 1

8.4.7.3 Earth Exosphere Lyman Alpha Analyzer (EELAA) from INS-1B

The most abundant neutral constituent in Earth's upper exosphere, atomic hydrogen (H), resonantly scatters solar Lyman-α (121.567 nm) radiation creating a phenomenon known as the geo-corona. Several space experiments have observed the geocorona under various conditions. The Lyman-alpha (Lα) radiation is of great importance of any planetary atmosphere, because the knowledge of the Lα radiation and its variation is important for many investigations of the middle and upper atmosphere. The short-listed payload i.e., 'Earth

Exosphere Lyman Alpha Analyze (EELAA)' is an FUV photometer that provides an opportunity to conduct experiments in various observational times (dawn to dusk) at different altitudes and understand nature of Lyman-alpha distribution around the earth. It further aids to understand the on-board calibration aspects of FUV instruments by looking at bright UV-stars. This instrument has the space heritage (LAP aboard MOM spacecraft) and similar kind of instruments had flown in a couple of interplanetary missions (HDAC for Cassini, LAD for Twinsat and UVS-P for Nozomi, ASLAF aboard Hotpay 1 payload) destined for Saturn, Earth Mars and Sun. Following are the primary science objectives of the instrument.

Science Objectives:

- Registration of terrestrial exospheric line-of-sight neutral atomic hydrogen Lyman-alpha flux.
- Estimation of interplanetary hydrogen Lyman-alpha background flux by means of deep space observations.
- Onboard calibration experiments by looking at UV-bright stars.

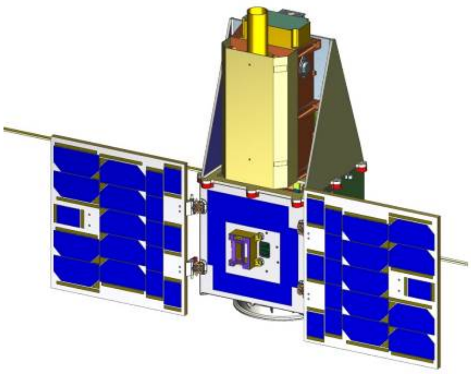


Figure 8-24 Schematic view of EELAA instrument aboard INS-1B

8.4.7.3.1 <u>System engineering aspects:</u>

EELAA instrument consists of an FUV detector equipped with a collecting lens, Lyman-alpha filter, light baffle and processing/read-out electronics unit.

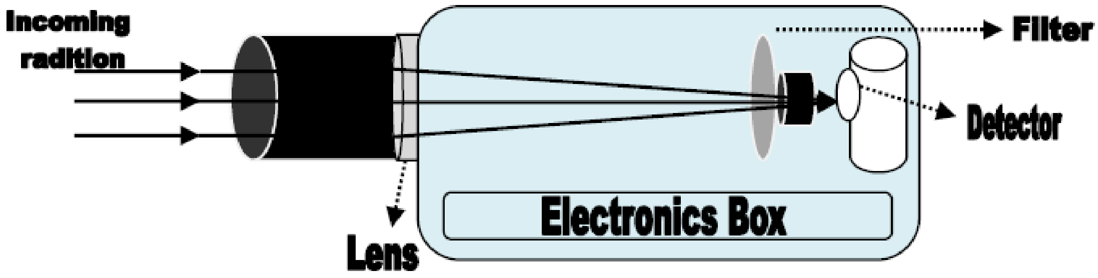


Figure 8-25 EELAA payload schematic view

8.4.7.3.2 Instrument configuration

It consists of:

- Primary baffle and Secondary baffle
- A collection lens (160 mm FL MgF_2 Plano-convex Lens)
- Bandpass filter (+/- 60 A^0)
- Detector (Solar-blind UV-PMT, 115 nm to 125 nm)

ELAA works on principle of resonant scattering of solar-Lyman alpha photons by earth's atmosphere molecules. The incoming hydrogen Lyman-alpha photons in the field-of-view of instrument will be collected by lens and is focused on to the solar-blind UV detector after passing through a band pass Lyman-alpha filter. Line-of-sight hydrogen Lyman-alpha intensity within the band pass of the filter will be registered as counts. The recorded average counts were supposed to be employed to estimate the flux using on ground-based/on-board executed calibration factors.

8.4.7.3.3 Salient features of EELAA payload*:*

Table 4.6 EELAA payload parameters

Parameter	Value
Range of Operation	600 km circular orbit
Mode of Operation	Photometer Mode
Principle of Operation	Resonant scattering
Pointing	Nadir, Limb, deep space and star view
Observation Time (hr)	1 hour – 1 ½-hour continuous operation
Integration Time(s)	1 -60 (selectable by ground command)
Data Rate (bps)	64
Size(mm3)	276x 138 x100.5
Mass(kg)	1.2
Power(W)	Steady-state: 4.6
Operating Temperature	0°C to +35°C

8.4.7.4 Miniature Multispectral Payload Technology Demonstration (MMX-TD)

MMX TD payload is a for technology demonstration of highly miniaturized multi-fold optics. Focal plane consists of RGB camera head. Provides RGB snaps of 40 km x 30 km area with 50 m GSD from 505 km orbit. Possible data utilization of this payload are topographical mapping, vegetation monitoring, aerosol scattering studies, cloud studies etc.

Parameter	Value
Altitude (km)	505
Spectral bands	RGB
IGFOV (m)	50
Pixel format	800 X 600
Swath (km)	40 x 30
Digitization (bit)	8
Size (mm^3) EO Module Electronics	 100 x 85 x 105 100 x 125 x 65
Weight (kg)	1.1
Power (W)	5
Frame rate	1 frame every 10s

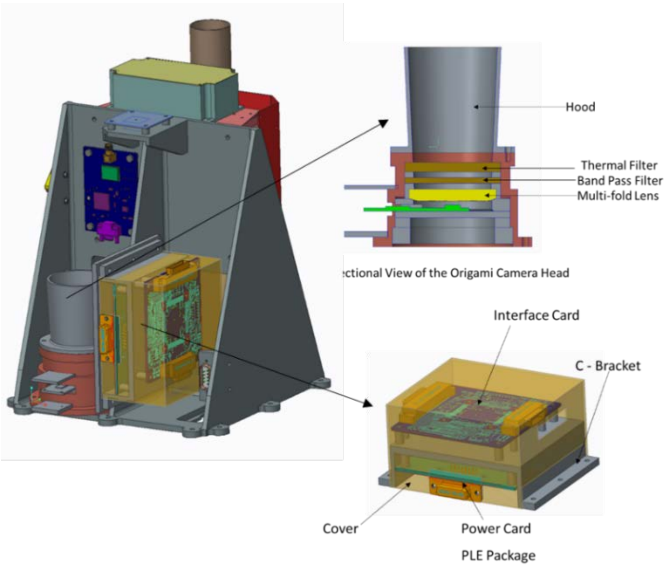


Figure 8-26 MMX-TD camera head & Electronics box

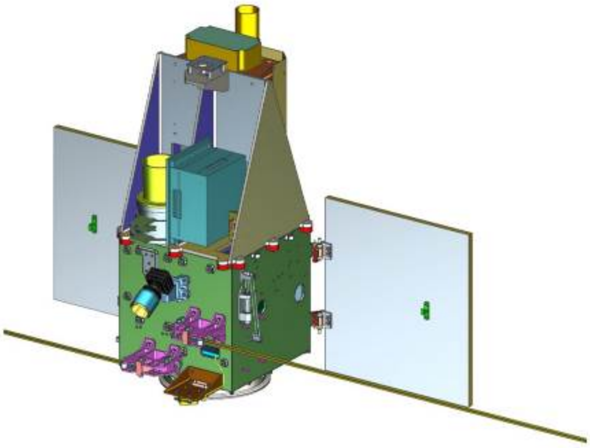
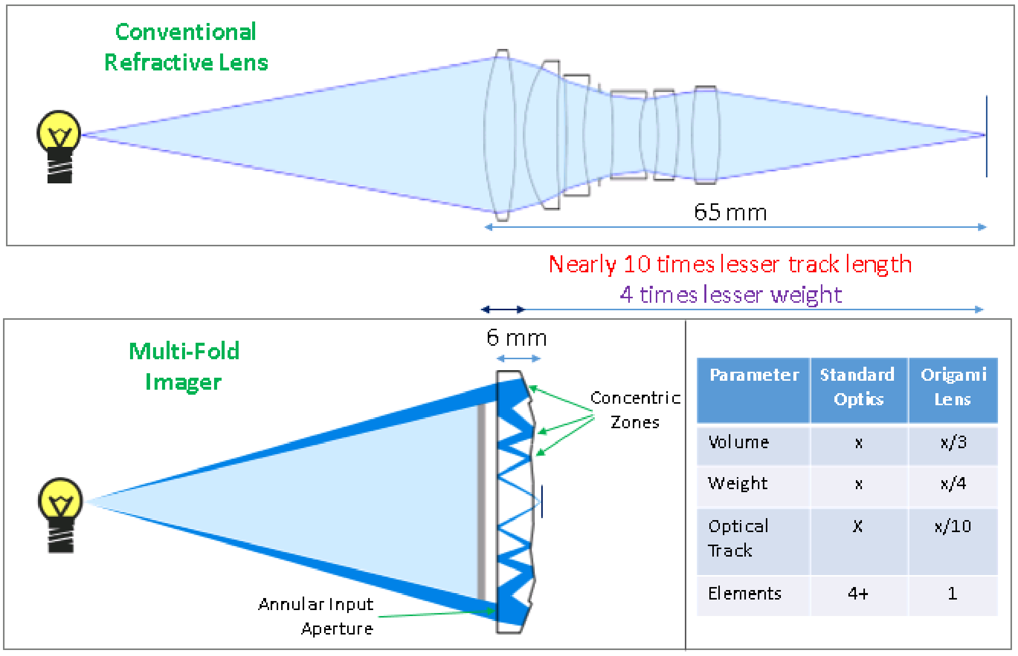


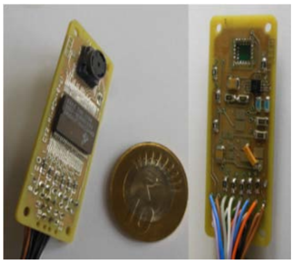
Figure 8-27 Placement of MMX-TD

8.4.7.4.1 Optics - Ultra-thin multi-fold lens

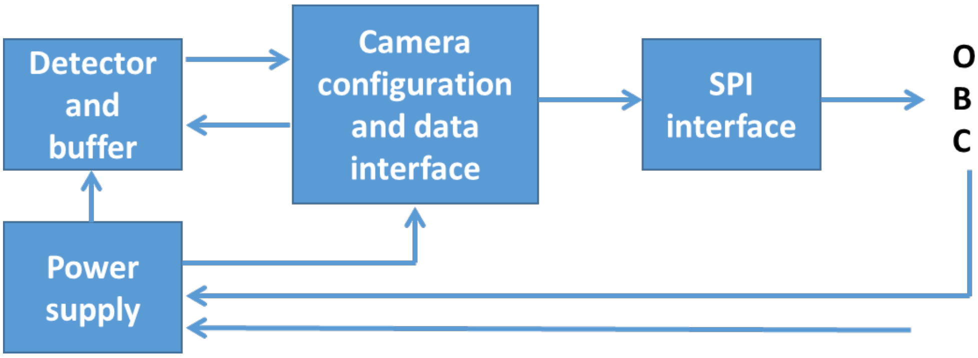


Parameter	Value
Optical Configuration	Multi-fold Optics
Focal Length (mm)	44
Effective F/number	2.41
Effective Aperture (mm)	18.23
FOV (Circular)	± 2.2°
Distortion	< ± 0.1%
Spatial Frequency of Operation	227 lp/mm
MTF @ Nyquist	> 5%

Origami Lens



8.4.7.4.2 Electronics:



Payload electronics consists of two parts

1. Camera Head: Consisting of detector and buffer
2. Camera Electronics: Consisting of interface electronics and power supply

The I2C protocol for camera configuration, SPI protocol for data transmission to OBC and On-board cache memory for data storage.

8.4.7.4.3 Mechanical Sub-System

Camera housing forms main supporting structure for mounting lens, filters, DHA. Aluminium material is chosen for the structure to minimize the mass. A hood fixed onto the structure in front of lens minimizes the stray light. Camera electronics is planned to be mounted separately on structure. Envelope dimension for camera head assembly is 100 x 85 x 105 mm3.

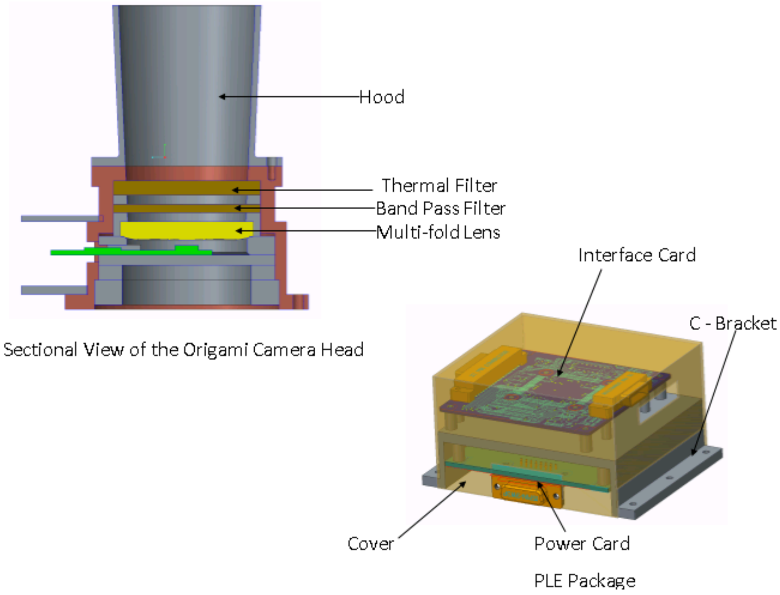


Figure 8-28 Mechanical Details of Payload

8.4.7.5 Development Philosophy

The payloads are developed with a short turnaround time without redundancy and with the use of COTS components wherever required and feasible. In-house qualification is carried out at component/sub-assembly level wherever required. Heritage designs are utilized to the extent possible. The development of the payloads is carried out within the existing resources.

8.5 INS-1C

8.5.1 Introduction

Nano Satellites offer a compelling alternative to large space projects, with a capability to support commercial, governmental, academic applications in a responsive and cost effective manner. Also, Nano Satellites have lot of potential in future and will give rise to lot of new developments in various technologies and are needed from strategic point of view in near future. In view of this, ISRO has developed and is launching the ISRO Nano Satellite INS-1C. This nano satellite will set a benchmark for development of nano satellites in future on a large scale. INS-1C will be carrying Miniature Multispectral Technology Demonstration (MMX-TD) Payload from SAC.

8.5.2 Mission Objectives

The primary objectives of this mission are:

- Design and develop a low cost, modular Nano satellite Bus in the weight range of 6-7 kg capable of carrying payloads up to a weight of 5 kg and provide payloads volume of 205x186x150 mm^3
- Provide up to 12 Watts of power for payload operation and up to 1 Mbps downlink for data collection
- Provide a 3-axes stabilized system for a control accuracy of ±0.5° via reaction wheels
- Provide on-board data handling and storage of 8 GB

- Provide an opportunity for ISRO technology demonstration payloads
- Provide a standard bus for launch on demand services
- Provide an opportunity to carry innovative payloads for Universities/R&D labs

8.5.3 Orbit Details

INS-1C satellite will be carrying payload MMX-TD from SAC. The satellite will be launched along with NOVASAR and the orbit parameters are as follows:

Table 8-19 Orbit details of INS-1C

Parameter	INS-1C
Altitude	583 km
Orbit	Polar sun synchronous Orbit
Inclination	97.7°
Local time at descending node	10:45PM
Stabilization	3 Axes Stabilization
Launch Site	SDSC SHAR, Sriharikota

8.5.4 Salient features of INS-1C

Table 8-20 Features of INS Bus Systems

Parameter	Specifications
Bus Mass	9.2 kg
Overall Size	245 x 227 x 217 mm^3
Structure	Milled aluminium decks
Thermal control	Passive (OSR, MLI, Paints etcs.)
Mechanisms	Solar panels & Antenna deployment
Solar Array Battery Power distribution	ZTJ cells with Array power of ~27W at EOL Deployable panel - 250 x 235 x 8 mm^3 Body mounted panel - 163 x 160 mm^2 11.2 Ah Lithium Ion battery Single Raw bus 12 to 16.8 V
Attitude and Orbit Control System (AOCS)	Attitude sensors : Star Sensor - 1 no.; MEMS IMU – 1 no. Micro Coarse Analog Sun sensor – 1 no. Sun Detection Sensor – 6 nos. Digital Magnetometer – 1no. Actuators: Reaction wheels – 4 nos. (15 mNm-s, 3 mNm) Magnetic Torquers – 3 nos. (1 A-m^2)
Control accuracy	<0.5° (3 σ) about each axis
TM and TC links	Telecommand- 1.2 kbps(VHF), 145.805 MHz Telemetry - 1.2 kbps(UHF), 435.080 MHz / 19.2 kbps (S - Band)
Payload data Handling	1 Mbps in S-band BPSK

	8 GB on-board Micro SD
Mission life	6 months to 1 year
Payload	MMX-TD

8.5.5 Payload

8.5.5.1 Introduction

Miniature Multispectral Technology Demonstration (MMX-TD) payload demonstrates the capability of highly compact multi-fold optics with optical performance at par with conventional optical assembly. The focal plane consists of RGB camera head. The payload provides RGB snaps of 32 km x 32 km area with 23 m GSD from 580 km orbit. Possible data utilization of this payload are topographical mapping, vegetation monitoring, aerosol scattering studies, cloud studies etc.

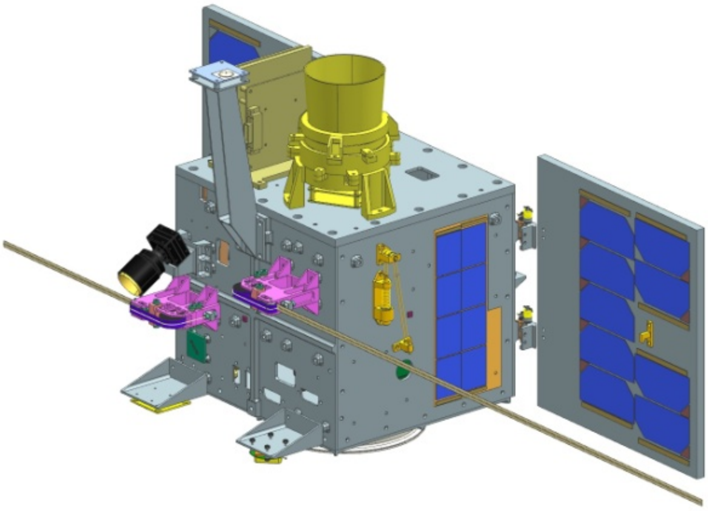


Figure 8-29 INS-1C Deployed configuration with MMX-TD Camera

8.5.5.2 Parameters of MMX-TD payload

Following table shows payload design parameters

Parameter	Value
Altitude (km)	583
Spectral bands	RGB
GSD (m)	23
Pixel format	1400 X 1400
Swath (km)	32 x 32
Digitization (bit)	8
EOM Size (mm^3)	150 X 120 X 190
Weight (kg)	2.0
Power (W)	4.5
Frame rate	1 frame every 32 s

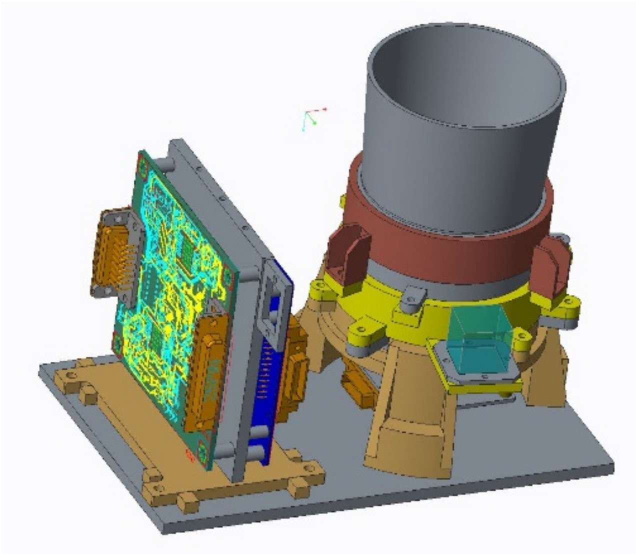


Figure 8-30 MMX-TD camera head & Electronics

8.5.5.3 Ultra-thin multi-fold Origami Optics

The optical system is based on two elements with four folds of optical rays so as to make the system compact along the optical axis. This optics provides similar imaging performance as compared to multi-element lens assembly with 1/5th the track length and 1/3rd the weight.

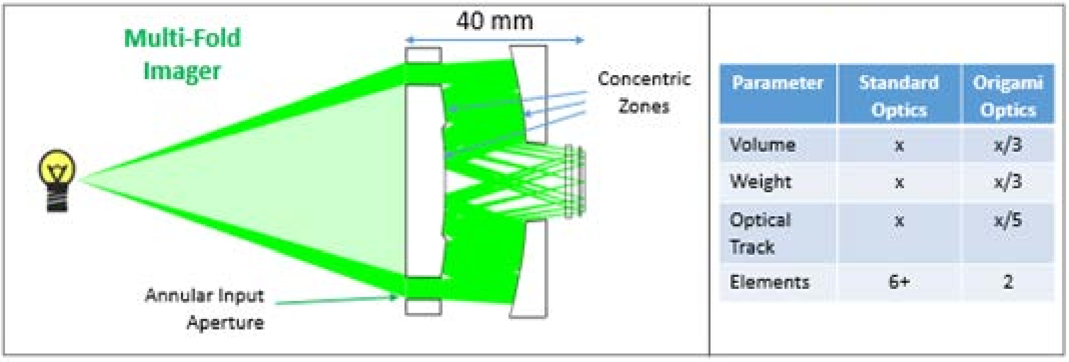
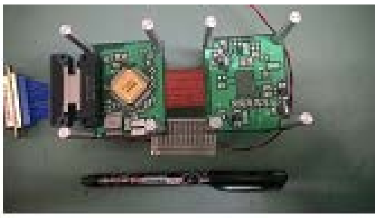


Figure 8-31 Multi-fold optics

Table 8-21 Optical parameters

Parameter	Value
Optical Configuration	Multi-fold Optics
Focal Length (mm)	138.9
Effective F/number	3.8
Effective Aperture (mm)	36.2
FOV (Diagonal)	±3.6
Spatial Frequency of Operation	91 lp/mm
MTF @ Nyquist/2	> 5%

Origami Mirrors



Camera Head

Figure 8-32 Origami mirrors and Camera head

8.5.5.4 Electronics

Payload electronics consists of two parts

1. **Camera Head:** Consisting of detector, bias generator and digital electronics
2. **Camera Electronics:** Consisting of interface electronics and power supply

Following figure shows the block diagram of electronics subsystems.

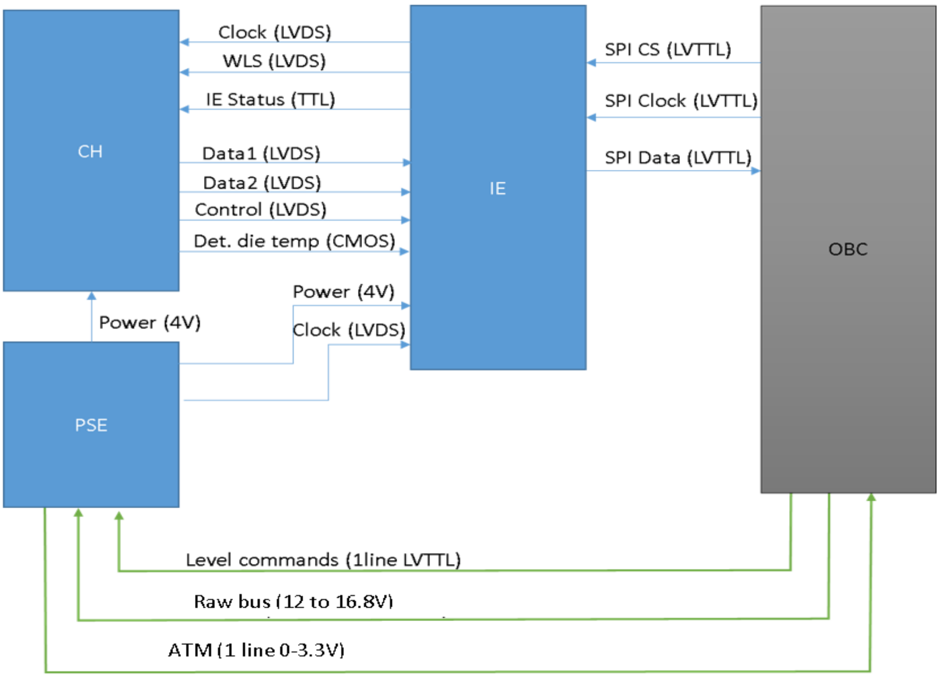


Figure 8-33 MMX-TD Electronics scheme

The SPI protocol is used for data transmission to OBC. SPI clock and chip select are sent by OBC and data from Camera Electronics is sent to OBC over the SPI clock. One analog and two thermister telemetry lines are also sent to OBC from power supply, EOM and Electronics packages respectively.

8.5.5.5 Mechanical Sub-System

Camera housing forms main supporting structure for mounting optics, filters and DHA. Aluminium material is chosen for the structure to minimize the mass. A hood fixed onto the structure in front of mirror minimizes the stray light. Camera electronics is mounted separately on the structure. Envelope dimension for camera head assembly is 150 X 120 X 190 mm^3.

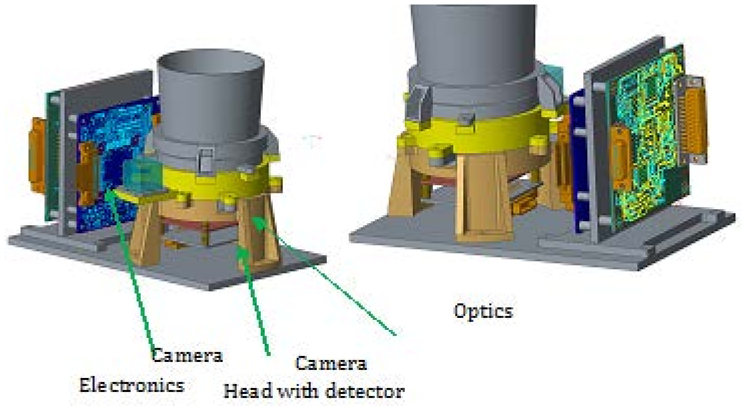


Figure 8-34 MMX-TD mechanical configuration

8.5.6 Major On-Orbit Observations in INS-1A and INS-1B

- Higher initial rates of more than 20 °/s observed after injection.
- Non-acquisition of telemetry during first Bangalore pass due to higher than anticipated UHF band noise floor.
- Inability to achieve 3-axes stabilization due to higher than designed rates on the satellite. Spacecraft was unable to acquire sun pointing and hence power generation was not adequate to power minimum required sub systems and control the spacecraft.

8.5.7 Major changes in INS-1C

Improvements in the power system

- Battery capacity increased by two times
- Body mounted solar cells are configured & solar cells mounted on both sides of deployable panels to support power requirement during de-tumbling phase
- Separate DC-DC converter for different sub-systems

Improvement in AOCS

- Separate Magnetometer
- Increased Torquer Capacity by three times to 1 A-m2 for faster detumbling
- 4RW Configuration as compared to 3RW configuration in INS-1A/1B
- Sun Detection sensors on all sides

Changes in Structure

- Individual panels are configured instead of milled cuboid integral structure and the size has been increased to accommodate additional packages.

Improvements in OBC

- Separate card to take care of cold sparing issues
- Implementation of Watch Dog Timer (WDT) for VHF Rx to facilitate Receiver reset. Facility to provide OBC reset through Receiver by direct command.

- Tele-command Watch-dog timer should automatically RESET the TC Receiver after a fixed period of time if no Tele-command is received and the logic should never get disable

Changes in RF

- In addition to UHF Tx for telemetry, S-Band Tx for telemetry has also been included based on the ground station support point of view. Both TM and Payload Data has been configured in single S-Band Card.

Improvements in Thermal Systems

Two heaters included for battery temperature control.

8.5.8 New Technologies in INS

- Lightweight Nano Reaction wheels for attitude control
- Micro magnetic torquers for attitude control
- Nano star sensor for attitude determination
- Micro Coarse Analog Sun Sensor offereing highest accuracy in its class
- Nano Satellite Positioning System for location determination
- 10 Degree of Freedom Inertial measurement unit consists of triaxial gyro, magnetometer, accelerometer & barometer
- Subsystem on a card philosophy for spacecraft electronics
- COTS electronics utilization to reduce costs and encourage hot swap ability
- Single card on-board computer including payload data handling & storage
- Single card S-band transmitter for payload data
- Single card TMTC transceiver for HK data
- Single card power control and distribution electronics
- Single card Nano SPS
- Single card multi-line DC-DC system
- SMA based micro mechanisms for Solar panel deployment

9. University Satellites

9.1 ANUSAT

ANUSAT (Anna University Satellite) is the first satellite built by an Indian University under the overall guidance of ISRO and will demonstrate the technologies related to message store and forward operations.

Parameter	Value
Altitude	550 km
Inclination	41 deg
Orbit Period	90 minutes
Mass	40 kg
Launch Vehicle	PSLV-C12

9.2 STUDSAT

Student Satellite (STUDSAT) is the first pico-satellite developed in the country by a consortium of seven engineering colleges from Karnataka and Andhra Pradesh. STUDSAT weighing less than 1 kg, has the primary objective of promoting space technology in educational institutions and encourage research and development in miniaturized satellites, establishing a communication link between the satellite and ground station, capturing the image of earth with a resolution of 90 meters and transmitting the payload and telemetry data to the earth station.

Parameter	Value
Mission	Experimental / Small Satellite
Weight	Less than 1 kg
Altitude	630 km
Orbit	Polar Sun Synchronous
Launch Vehicle	PSLV-C15

9.3 JUGNU

The nanosatellite JUGNU weighing 3 kg is designed and developed by Indian Institute of Technology, Kanpur under the guidance of ISRO.

The satellite is intended:

1. To prove the indigenously developed camera system for imaging the Earth in the near infrared region and test image processing algorithms.
2. Evaluate GPS receiver for its use in satellite navigation.
3. Test indigenously developed MEMS based Inertial Measurement Unit (IMU) in space.

Parameter	Value
Mission	Experimental / Small Satellite
Weight	3 kg
Altitude	630 km
Orbit	20deg Inclined orbit
Launch Vehicle	PSLV-C18

9.4 SRMSAT

The nanosatellite SRMSAT weighing 10.9 kg is developed by the students and faculty of SRM University attempts to address the problem of Global warming and pollution levels in the atmosphere by monitoring Carbon dioxide (CO_2) and water vapour (H_2O). The satellite uses a grating Spectrometer, which will observe absorption spectrum over a range of 900nm - 1700nm infrared range.

Parameter	Value
Mission	Experimental / Small Satellite
Weight	10.9 kg
Altitude	630 km
Orbit	20deg Inclined orbit
Launch Vehicle	PSLV-C18

9.5 SWAYAM

Mission objectives: To provide point to point messaging services to the HAM Community. Satellite from College of Engineering, Pune.

Parameter	Value
Mission	Experimental / Small Satellite
Weight	1 kg
Altitude	505 km
Orbit	Polar Orbit
Launch Vehicle	PSLV-C34

9.6 SATHYABAMASAT

Mission objectives: To collect data on greenhouse gases (Water vapor, Carbon monoxide, Carbon dioxide, Methane and Hydrogen fluoride). Satellite from Sathyabama University, Chennai.

Parameter	Value
Mission	Experimental / Small Satellite
Weight	1.5 kg
Altitude	505 km
Orbit	Polar Orbit
Launch Vehicle	PSLV-C34

9.7 PISAT

Mission Objectives: Design and develop a Nano satellite for remote sensing applications. Satellite from PES University, Bengaluru and its consortium.

Parameter	Value
Mission	Experimental / Small Satellite
Weight	5.25 kg
Altitude	670 km
Orbit	SSPO

Launch Vehicle	PSLV-C35

9.8 PRATHAM

Mission Objectives: To estimate the Total Electron Count (TEC) over India and Paris (France) with a resolution of 1 km x 1 km location grid. Satellite from IIT, Bombay

Parameter	Value
Mission	Experimental / Small Satellite
Weight	10 kg
Altitude	670 km
Orbit	SSPO
Launch Vehicle	PSLV-C35

9.9 NIUSAT

NIUSAT is an Indian University/Academic Institute satellite from Noorul Isalm University in Tamil Nadu State, launched by PSLV-C38. This 15 kg three axis stabilised satellite is built to provide multispectral imagery for agricultural crop monitoring and disaster management support applications.

A dedicated Mission Control Centre with UHF/VHF antenna for Telemetry/Tele-command operations and S-band antenna for Payload data reception has been established at the university.

Parameter	Value
Mission	Experimental / Small Satellite
Weight	15 kg
Altitude	505 km
Orbit	SSPO
Launch Vehicle	PSLV-C38

DOCUMENT CONTROL DATA SHEET

01. Security & Distribution Status	**R** — U: Unrestricted R: Restricted S: Secret
02. Projected Utility life	a) 6 Yrs. b) **6 -15 Yrs.** c) > 15 Yrs.
03. Report Status (Indicate replacement Of old document, if any)	**Updated Document**
04 Report No.: ISRO-ISAC-TR-1445	Part No. or Vol, No.:
06. Title & Subtitle: **Indian Remote Sensing Missions & Payloads- A Glance**	
07. Contract No.:	08. Collation (No. of pages): **325**
09. Personal Author(s): **Space Science and Ground Segment Section Team**	
10. Affiliation of Author(s) other than ISAC: None	
11. Corporate Author(s):	
12. Origination Unit: Division/Group: **ISSD/PMSG**	
13. Date of Submission: **January 2018**	14. No. of References:
Abstract: **This document gives the details of all payloads and spacecraft launched under Indian Remote Sensing (IRS) programme,**	
Keywords/Descriptors: 1. **Remote Sensing** 2. **Payload** 3. **Attitude Control**	B) Standardized by Bibliographical Control Agency: (To be given by Library) 1. Remote Sensing 2. Payload 3. Attitude Control
17. Supplementary Elements:	None

Approval by GD/PD/DD for 01: P. Murugan

पी मुरुगन / P Murugan
समूह प्रधान / Group Head
आई.आर.एस. कार्यक्रम प्रबंधन व प्रणाली समूह
IRS Programme Management & Systems Group

INDIAN REMOTE SENSING MISSIONS & PAYLOADS A GLANCE