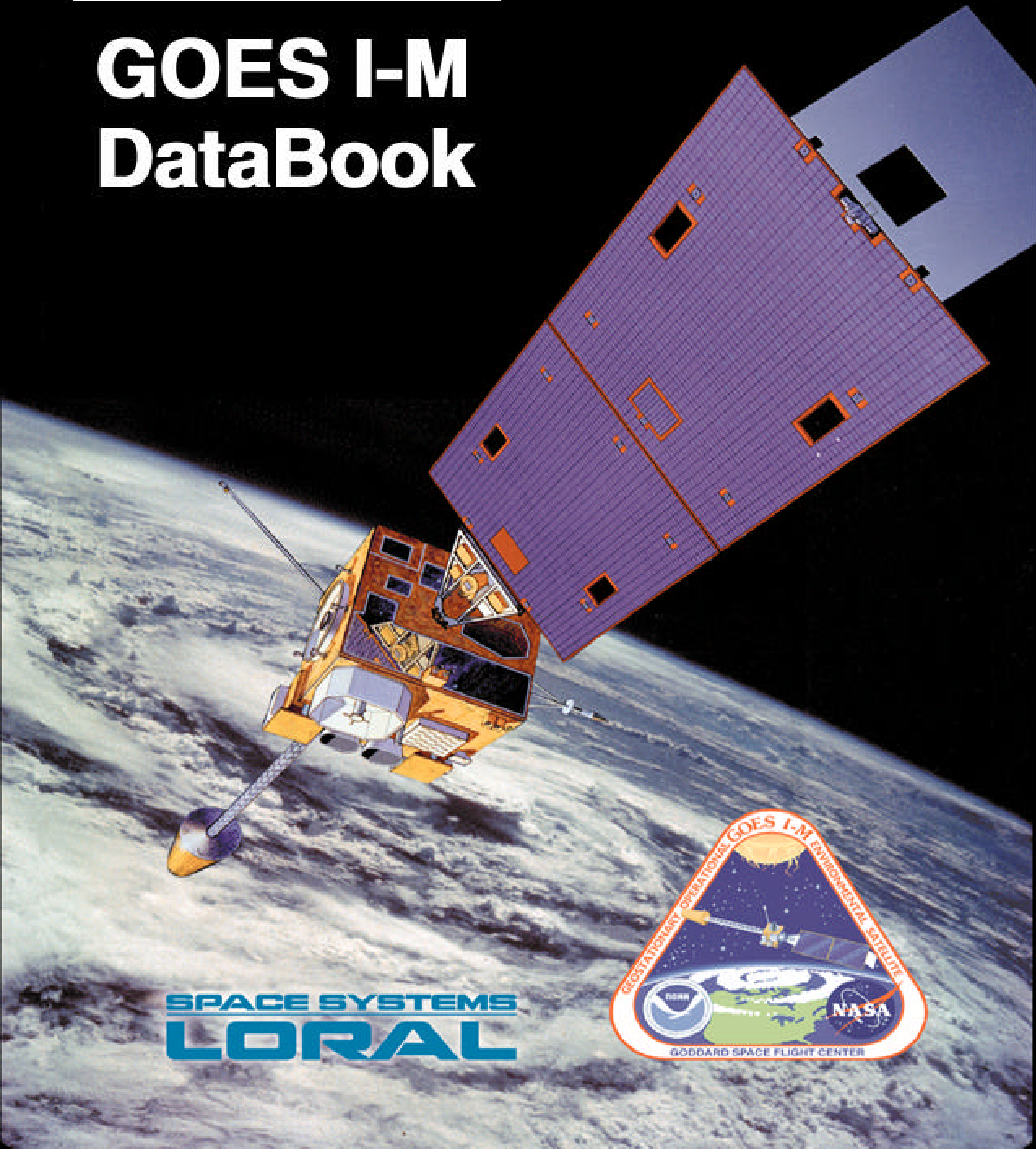
DRL 101-08
Revision 1
31 August 1996

GOES I-M DataBook



SPACE SYSTEMS LORAL

GOES I-M DataBook

Prepared for:
National Aeronautics and Space Administration
Goddard Space Flight Center
Greenbelt, Maryland 20771

Pursuant to:

DRL 101-08
GSFC Specification S-480-21A
Contract NAS5-29500

Reference #S-415-19

Revision 1 - 31 August 1996

Foreword

The Geostationary Operational Environmental Satellite (GOES) program is a key element in National Weather Service (NWS) operations. GOES weather imagery and quantitative sounding data are a continuous and reliable stream of environmental information used to support weather forecasting, severe storm tracking, and meteorological research. Evolutionary improvements in the geostationary satellite system since 1974 (i.e., since the first Synchronous Meteorological Satellite, SMS-1) have been responsible for making the current GOES system the basic element for U.S. weather monitoring and forecasting. Spacecraft and ground-based systems work together to accomplish the GOES mission.

Designed to operate in geosynchronous orbit, 35,790 km (22,240 statute miles) above the earth, thereby remaining stationary, the advanced GOES I-M spacecraft continuously view the continental United States, neighboring environs of the Pacific and Atlantic Oceans, and Central and South America. The three-axis, body-stabilized spacecraft design enables the sensors to “stare” at the earth and thus more frequently image clouds, monitor earth’s surface temperature and water vapor fields, and sound the atmosphere for its vertical thermal and vapor structures. Thus the evolution of atmospheric phenomena can be followed, ensuring real-time coverage of short-lived dynamic events, especially severe local storms and tropical cyclones — two meteorological events that directly affect public safety, protection of property, and ultimately, economic health and development. The importance of this capability has recently been exemplified during hurricanes Hugo (1989) and Andrew (1992).

The GOES I-M series of spacecraft are the principal observational platforms for covering such dynamic weather events and the near-earth space environment for the 1990s and into the 21st century. These advanced spacecraft enhance the capability of the GOES system to continuously observe and measure meteorological phenomena in real time, providing the meteorological community and the atmospheric scientist greatly improved observational and measurement data of the Western Hemisphere. In addition to short-term weather forecasting and space environmental monitoring, these enhanced operational services also improve support for atmospheric science research, numerical weather prediction models, and environmental sensor design and development.

The main mission is carried out by the primary payload instruments, the Imager and the Sounder. The Imager is a multichannel instrument that senses radiant energy and reflected solar energy from the earth’s surface and atmosphere. The Sounder provides data for vertical atmospheric temperature and moisture profiles, surface and cloud top temperature, and ozone distribution.

Other instruments on board the spacecraft are the search and rescue transponder, ground-based meteorological platform data collection and relay, and the space environment monitor. The latter consists of a magnetometer, an X-ray sensor, a high energy proton and alpha detector, and an energetic particles sensor, all used for in-situ surveying of the near-earth space environment.

End users, scientific and technical persons, program personnel, and others desiring mission data from or further information about the GOES system may contact the specific National Environmental Satellite Data and Information Service (NESDIS) point of contact.

Topical Area	**Point of Contact**	**Telephone**	**"E Mail Address"**
Product Manager	Jamison Hawkins	301-457-5125	–
NOAA PORT Manager	Carl P. Staton	301-457-5165	CStaton@NESDIS.NOAA.Gov
GOES TAP Manager	John A. Paquett	301-763-8051	Jnaquette@SSD.WWB.NOAA.Gov
Satellite Operations			
SOCC	Louis P. Barbieri	301-457-5130	–
Engineering	Michael A. Suranno	301-457-5240	–
Ground Systems	Richard Reynolds	301-457-5185	RReynolds@NEDIS.NOAA.Gov
Instrument Calibration	Michael Weinreb	301-457-5250	–
WEFAX / Data Collection	Wayne G. Winston	301-763-8325	Wayne_Winston@SSD.WWB.NOAA.Gov
Space Environment Monitor	Richard Grubb	303-497-3284	RGrubb@SEC.NOAA.Gov
Space Environment Center	Gary R. Heckman	303-497-5687	GHeckman@SEC.NOAA.Gov

Preface

To further enhance the utility of the GOES system, this DataBook presents a summary and technical overview of the GOES I-M system, its satellites, subsystems, sensor suite, and associated ground communication and data handling subsystems. The DataBook is intended to serve as a convenient and comprehensive, desktop technical reference for persons working on or associated with the GOES I-M missions. Sufficient technical information and performance data are presented to enable the reader to understand the importance of the GOES I-M mission, the system's capabilities, and how it meets the needs of end users.

Certain performance data presented herein, e.g., Imager and Sounder radiometric performance, were predicted from or measured on the GOES I satellite. As the satellites undergo on-orbit operations and actual data are obtained, such technical information in this book may not necessarily reflect current capabilities.

Space Systems/Loral (SS/L) is the prime contractor for the GOES I-M system under NASA Contract No. NAS5-29500. The GOES program is managed for the National Oceanic and Atmospheric Administration (NOAA), the principal user, by the National Aeronautics and Space Administration (NASA), Goddard Space Flight Center (GSFC). The Aerospace/Communications Division of ITT is the subcontractor to SS/L for the Imager and Sounder instruments. The space environment monitor sensors are all provided by Panametrics, except for the magnetometer, which is built by Schonstedt Instrument Company.

Contents

Overview of the GOES Mission

Goals of the Mission

The goals of the Geostationary Operational Environmental Satellite (GOES) system program are to:

- Maintain reliable operational, environmental, and storm warning systems to protect life and property
- Monitor the earth's surface and space environmental conditions
- Introduce improved atmospheric and oceanic observations and data dissemination capabilities
- Develop and provide new and improved applications and products for a wide range of federal agencies, state and local governments, and private users

To address these goals, the National Weather Service (NWS) and the National Environmental Satellite Data and Information Service (NESDIS) of the Department of Commerce established mission requirements for the 1990s that are the bases for design of the GOES I-M system and its capabilities. The GOES system thus functions to accomplish an environmental mission to service the needs of operational meteorological, space environmental, and research users.

The GOES System

To accomplish this mission, the GOES I-M series of spacecraft perform three major functions:

- ***Environmental Sensing***: Acquisition, processing, and dissemination of imaging and sounding data independent of imaging data processes and the *(in-situ)* space environment monitoring data, and measurement of the near-earth space "weather."
- ***Data Collection***: Interrogate and receive data from earth surface-based data collection platforms (DCPs) and relay to the National Oceanic and Atmospheric Administration (NOAA) command and data acquisition stations.
- ***Data Broadcast***: Continuous relay of weather facsimile and other meteorological data to small users, independent of all other system functions; relay of distress signals from aircraft or marine vessels to the search and rescue ground station of the search and rescue satellite-aided tracking system.

Each mission function is supported or performed by components of the GOES I-M payloads:

Environmental sensing:

- Five-channel Imager
- Nineteen-channel Sounder
- Space environment monitor (SEM)
  - Energetic particles sensor (EPS)
  - High energy proton and alpha particle detector (HEPAD)
  - X-ray sensor (XRS)
  - Magnetometers

Data collection:

- Data collection system (DCS)

Data broadcast:

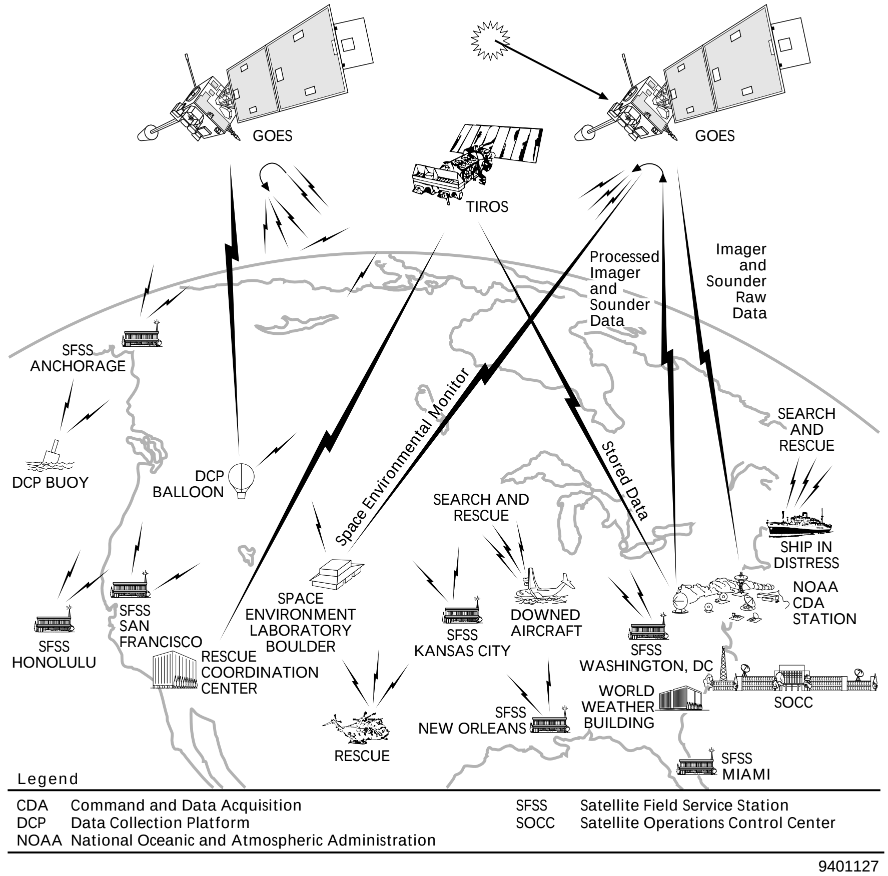
- Processed data relay (PDR) and weather facsimile (WEFAX) transponders
- Search and rescue (SAR)
- Sensor data and multiuse data link (MDL) transmitters

The remote sensing function is carried out by the 5-channel Imager and 19-channel Sounder, both of improved spatial and spectral resolution, and *in-situ* sensing by a SEM covering an extensive range of energy levels. The acquisition of sensed data and its handling, processing, and final distribution are performed in real-time to meet observation time and timeliness requirements, including revisit cycles. Remotely sensed data are obtained over a wide range of areas of the western hemisphere, encompassing the earth's disk, selected sectors, and small areas. Area coverage also includes the visibility needed to relay signals and data from ground transmitters and platforms to central stations and end users.

The Mission Capable Space Segment

The GOES I-M series of spacecraft are the prime observational platforms for covering dynamic weather events and the near-earth space environment for the 1990s and into the 21st century. These advanced spacecraft enhance the capability of the GOES system to continuously observe and measure meteorological phenomena in real-time, providing the meteorological community and atmospheric scientists of the western hemisphere with greatly improved observational and measurement data. These enhanced operational services improve support for short-term weather forecasting and space environment monitoring as well as atmospheric sciences research and development for numerical weather prediction models, meteorological phenomena, and environmental sensor design.

The Weather Watch System of the 1990s



The Observational Platform

The advanced GOES I-M spacecraft three-axis, body-stabilized design enables the sensors to “stare” at the earth and thus more frequently image clouds, monitor the earth’s surface temperature and water vapor fields, and sound the earth’s atmosphere for its vertical thermal and vapor structures. Thus the evolution of atmospheric phenomena can be followed, ensuring real-time coverage of short-lived, dynamic events, especially severe local storms and tropical cyclones, two meteorological events that directly affect public safety, protection of property, and, ultimately, economic health and development.

Innovative features incorporated in the GOES I-M spacecraft enable high volume, high quality data to be generated for the weather community. Two important capabilities are flexible scan control that allows small area coverage for improved short-term weather forecasts over local areas, and simultaneous, independent imaging and sounding. Precision on-orbit stationkeeping, coupled with three-axis stabilization, provides a steady observational platform for the mission sensors, greatly increasing earth-referenced data location and measurement accuracy. To maintain location accuracy, an innovative image navigation and registration (INR) methodology is employed that uses star sensing via the primary instruments. The INR subsystem provides daily imaging and sounding data on a precisely located, fixed earth coordinate grid without ground interpolation.

The Imager

The Imager is a multispectral, earth-scanning instrument capable of sweeping simultaneously one visible and four infrared channels in a north-to-south swath across an east-to-west path, and providing full earth imagery, sector imagery containing edges of earth's disk, and area scans of local regions. Besides simultaneous imaging, it features higher infrared spatial (4 kilometers, 2.5 statute miles) and spectral resolution in the surface and cloud detection channels, and increased sensitivity, all of which enhance quantitative estimates of surface temperature and low-level moisture and monitoring of convective intensity. Imaging over five channels significantly improves cloud and water vapor measurements and produces visual and infrared images of the earth's surface, oceans, cloud cover, and severe storm developments. Cloud imagery is available to users in mapped format (available for each channel) as well as the familiar GOES projection sectors. Two composite images, visible-infrared and infrared-water vapor, are also produced.

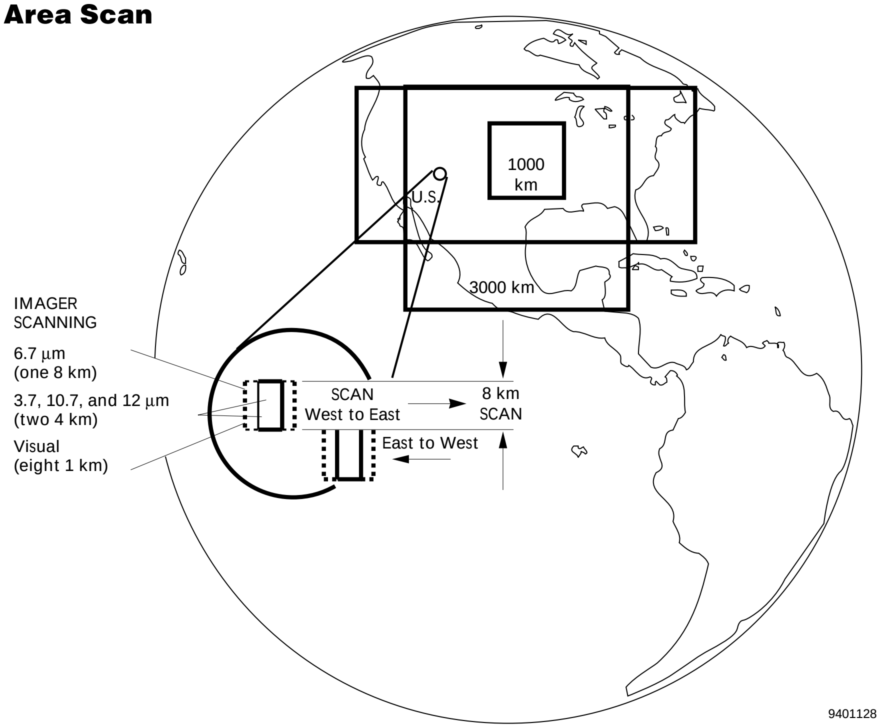
The Sounder

The GOES I-M Sounder features more spectral channels, higher spatial resolution (8 kilometers, 5 statute miles), and increased sensitivity for high quality soundings than are currently available. It is capable of stepping 1 visible and 18 infrared channels in a north-to-south swath across an east-to-west path. The Sounder and Imager both provide full earth imagery, sector imagery containing the edges of earth's disk, and area scans of local regions. Nineteen spectral bands (seven longwave, five midwave, six shortwave, and one visible) yield the prime sounding products of vertical temperature profiles, vertical moisture profiles, layer mean temperature, layer mean moisture, total precipitable water, and the lifted index (a measure of stability). These products are used to augment data from the Imager to provide information on atmospheric temperature and moisture profiles, surface and cloud top temperatures, and the distribution of atmospheric ozone.

Flexible Scan Control

Both Imager and Sounder employ a servo-driven, two-axis gimballed mirror system in conjunction with a 31.1-centimeter (12.2-inch) aperture Cassegrain telescope. As separate sensors, they allow simultaneous and independent surface imaging and atmospheric sounding. Each has flexible scan control, enabling coverage of small areas as well as hemispheric (North and South America) and global scenes (earth's full disk), and close-up, continuous observations of severe storms and of dynamic, short-lived weather phenomena.

Area Scan



A priority scan feature allows improved scheduling of small area and mesoscale scans for short range forecasts and storm warnings. Imager large area scans of 3000 by 3000 kilometers (1864 by 1864 statute miles) are accomplished in three minutes and small area scans of 1000 by 1000 kilometers (621 by 621 statute miles) can be made in 41 seconds. A 3000- by 3000-kilometer area can be sounded in 43 minutes, and full earth can be imaged in 26 minutes.

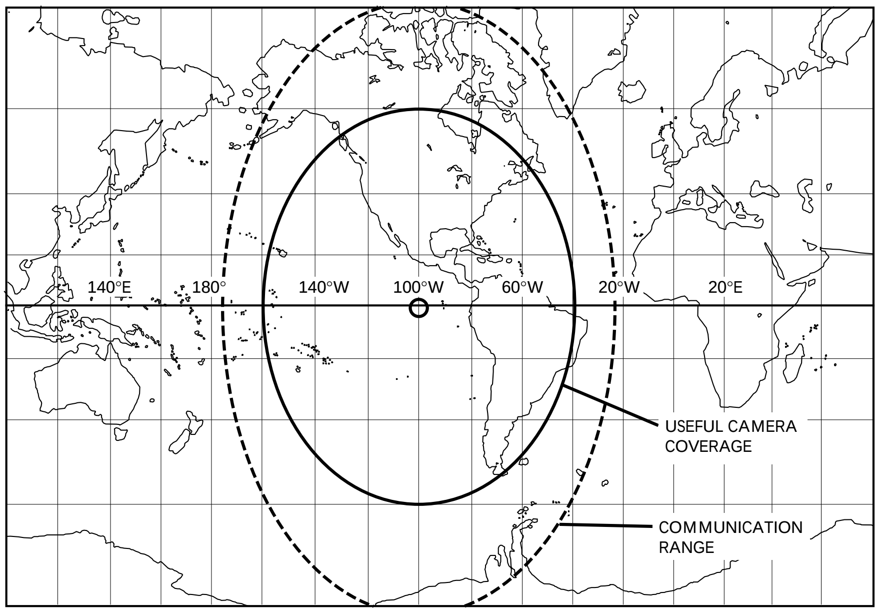
Space Environment Monitoring

The SEM instruments survey the sun, measuring *in situ* its effect on the near-earth solar-terrestrial electromagnetic environment. Changes in this “space weather” can affect operational reliability of ionospheric radio; over-the-horizon radar; electric power transmission; and most importantly, human crews of high altitude aircraft, the Space Shuttle, or a Space Station.

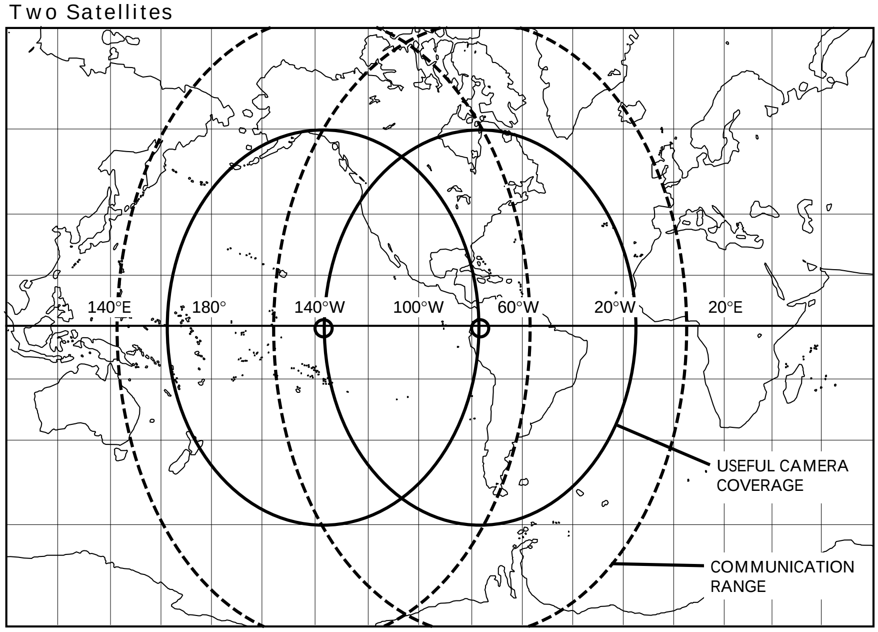
The XRS monitors the sun’s total X-ray activity. The EPS and HEPAD detect energetic electron and proton radiation trapped by the earth’s magnetic field as well as direct solar protons, alpha particles and cosmic rays. The magnetometer measures three components of earth’s magnetic field in the vicinity of the spacecraft and monitors variations caused by ionospheric and magnetospheric current flows.

GOES Geographic Coverage

One Satellite



Two Satellites



9401129

3Data Broadcast

GOES also enhances services for receiving meteorological data from earth-based data collection platforms and relaying the data to end-users. A continuous, dedicated search and rescue transponder on board provides for immediate detection of distress signals from downed aircraft or marine vessels and relays them to ground terminals to speed help to people in need. Increased communications capacity permits transmission of processed weather data and weather facsimile for small local user terminals in the western hemisphere.

Geographic Coverage

The GOES spacecraft, on-station 35,790 kilometers (22,240 statute miles) above the equator and stationary relative to the earth's surface, can view the contiguous 48 states and major portions of the central and eastern Pacific Ocean and the central and western Atlantic Ocean areas. Pacific coverage includes the Hawaiian Islands and Gulf of Alaska, the latter known to weather forecasters as "the birthplace of North American weather systems." Because the Atlantic and Pacific basins strongly influence the weather affecting the United States, coverage is provided by two GOES spacecraft, one at 75° west longitude, *GOES East,* and the other at 135° west longitude, *GOES West.*

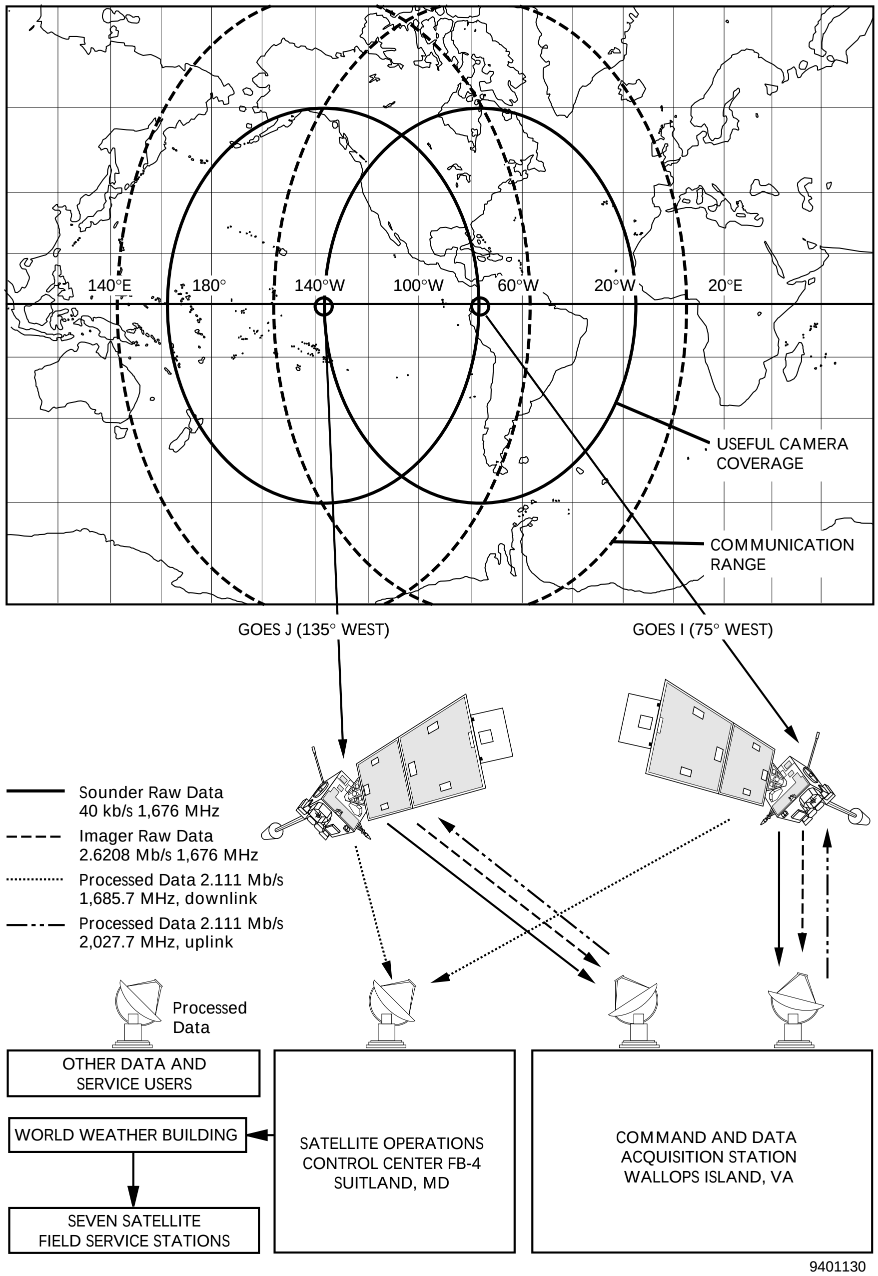
The combined footprint (radiometric coverage and communications range) of the two spacecraft encompasses earth's full disk about the meridian approximately in the center of the continental United States. Circles of observational limits centered at a spacecraft's suborbital point extend to about 60° north / south latitudes. The radiometric footprints are determined by the limit from the suborbital point, beyond which interpretation of cloud data becomes unreliable.

At least one GOES spacecraft is always within line-of-sight view of earth-based terminals and stations. The Command and Data Acquisition (CDA) Station is in line-of-sight to both spacecraft so that it can uplink commands and receive downlinked data from each simultaneously. Data collection platforms within the coverage area of a spacecraft can transmit their surface-based sensed data to the CDA Station via the onboard data collection subsystem. Similarly, ground terminals can receive processed environmental data and WEFAX transmissions.

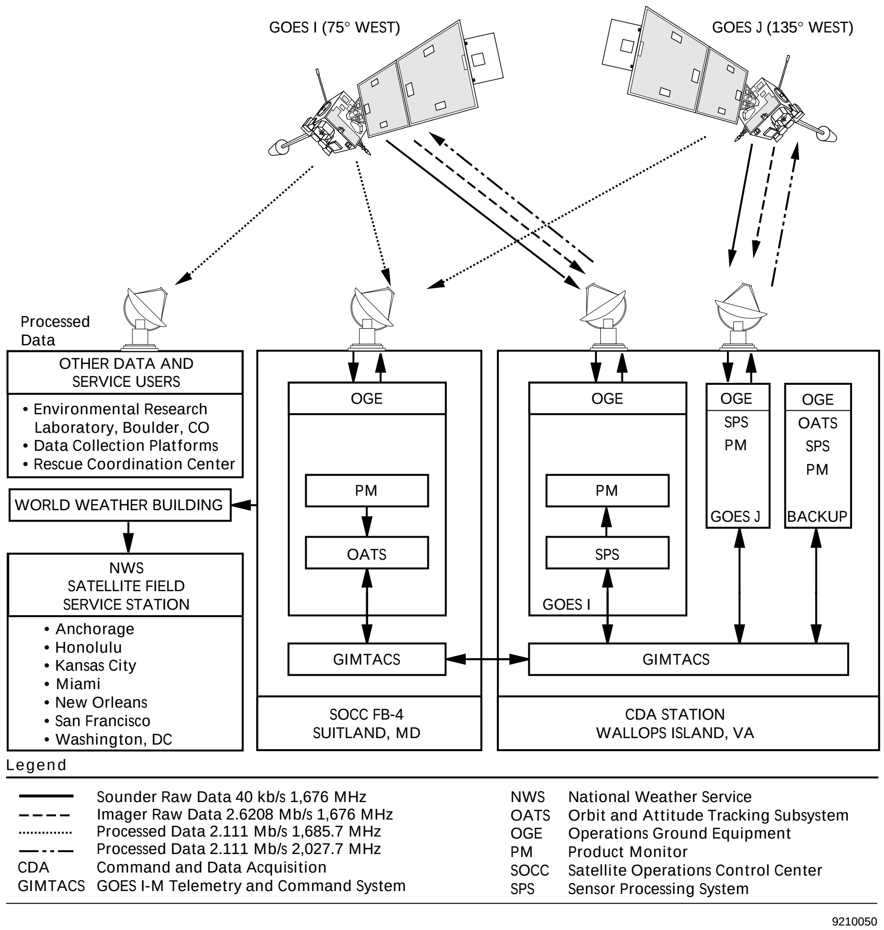
Ground Segment Support

Raw Imager and Sounder data received at the NOAA CDA Station are processed in the operations ground equipment (OGE) with other data to provide highly accurate, earth-located, calibrated imagery and sounding data in near real-time for retransmission via GOES spacecraft to primary end users, typically the seven NWS Field Service Stations located throughout the United States. Operational management and planning are performed at the Satellite Operations Control Center (SOCC), where all elements of the system are monitored, evaluated, scheduled, and commanded.

GOES System Functional Diagram



Data Transmissions



Network Architecture

The communications links, ground support equipment connectivity, and data transmission paths complete the interfaces among GOES I-M-specific and existing equipment. This network, transparent to current users, routes broadcast and mission data. The Imager and Sounder output serial bit streams are transmitted on the S-band carrier wave by the sensor data transmitter. The GOES spacecraft signal is received at the CDA Station where it is demodulated and processed by the OGE; the new uplink signal, containing calibrated, earth-located data, is uplinked from the CDA Station to the spacecraft, received by the S-band receiver, and converted to the appropriate transmit frequency. Before being multiplexed and retransmitted to user stations by the S-band transmit antenna, the signal is prefiltered to separate it from other uplinked signals.

GOES Variable Data Format

The GOES I-M variable (GVAR) data transmission format is primarily used to transmit Imager and Sounder meteorological data. It also includes telemetry, calibration data, text messages, spacecraft navigation data, and auxiliary products. The GVAR format originated in the operational visible infrared spin scan radiometer, atmospheric sounder (VAS) mode AAA of the early spin-stabilized GOES spacecraft. The AAA format consists of a repeating sequence of 12 fixed-length, equal size blocks whose transmission is synchronized with spacecraft spin rate (that is, one complete 12-block sequence for each rotation).

The range and flexibility of satellite operations have increased with the advent of three-axis stabilized GOES I-M spacecraft employing two independent instruments, each with a scanning mirror having 2 degrees of freedom. The use of a fixed-length transmission format would have imposed operational limits on the capabilities of the I-M spacecraft. To fully use these new capabilities, the GVAR format was developed, supporting variable length scan lines, while retaining as much commonality as possible with AAA reception equipment.

Operations Ground Equipment

The OGE consists of components located at the CDA Station, Wallops Island, Virginia, and the SOCC at Suitland, Maryland. The OGE receives input streams of raw Imager and Sounder data and MDL data from the spacecraft. Primary outputs are PDRs of those data streams in GVAR format. One GVAR-formatted output data stream is generated for each spacecraft downlink data stream. The GVAR data stream is transmitted to its corresponding GOES spacecraft for relay to primary system users, as well as back to the CDA Station and SOCC for other internal OGE functions. Communications among the several elements of the OGE are via the GOES I-M telemetry and command system (GIMTACS).

Internal OGE uses of GVAR data are primarily for monitoring the quality of processed instrument data (CDA Station and SOCC), determining spacecraft range and extracting landmark images as part of orbit and attitude determination, and monitoring on-board computation of north/south and east/west image motion compensation to provide continuous scan frame registration. Also data from the MDL are received at the SOCC as an independent data link that, for GOES I, contains angular displacement sensor and digital integrating rate assembly data. These data are ingested and processed by the OGE and used for diagnosing dynamic interactions among the instruments and the spacecraft.

Communications Links

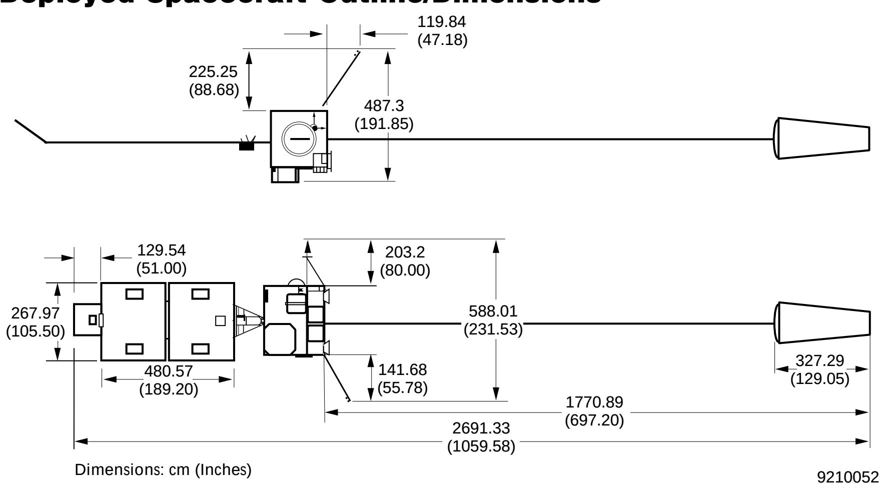
Description	Source	Uplink (MHz)	Downlink (MHz)	Destination
Command	CDA Station / DSN	2,034		Spacecraft
Telemetry (including SEM)	Spacecraft		1,694 / 2,209	CDA Station; DSN; Environmental Research Laboratory
WEFAX	CDA Station	2,033	1,691	Users; automatic picture transmission
Data Collection Platform Interrogate (DCPI)	CDA Station	2,034	468	DCP
Data Collection Platform Report (DCPR)	DCP	401	1,694	CDA Station; users
Search and Rescue (SAR)	Emergency Locator Transmitter (ELT)	406	1,544	Rescue coordination center
MDL (diagnostic data)	Spacecraft		1681.5	SOCC

The GOES Spacecraft Configuration

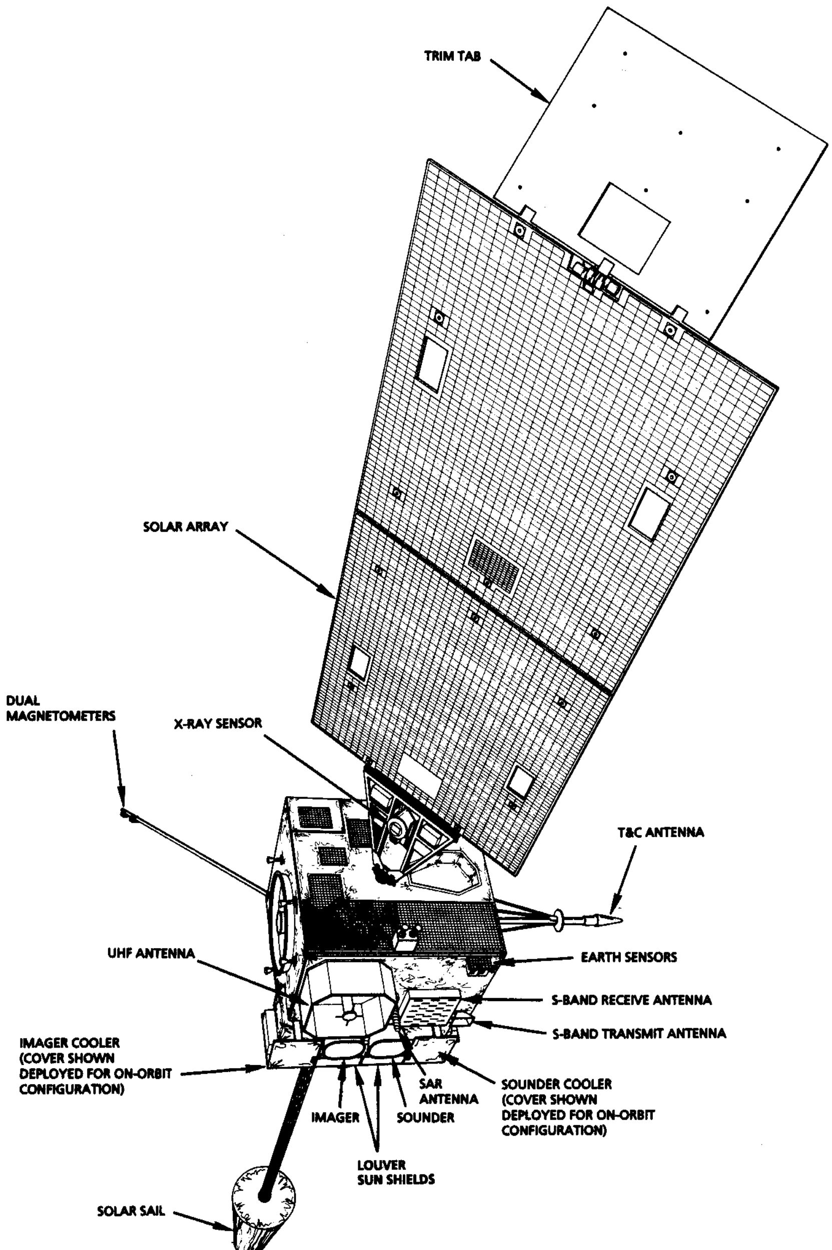
The GOES I-M spacecraft is a three-axis, body-stabilized design capable of continuously pointing the optical line of sight of the imaging and sounding radiometers to the earth. The spacecraft body contains all of the propulsion and electronic equipment and provides the stable platform on which the payload instruments are mounted. A single-wing, two-panel solar array on the south-facing side continuously rotates about the spacecraft pitch axis to track the sun during orbital motion. The use of a single-wing solar array mounted on the south-facing side of the spacecraft allows the passive north-facing radiation coolers of the Imager and Sounder to view cold space.

A conical-shaped solar sail mounted on a 17-meter (58-foot) boom on the north side balances the torque caused by solar radiation pressure. A trim tab panel at the end of the solar array provides the necessary fine balance control for the solar radiation pressure. All communications antennas, with the exception of telemetry and command, are hard mounted on the earth-facing panel for unobstructed earth coverage and maximum alignment stability. To provide near-omnidirectional coverage, the telemetry and command antenna is mounted on a fixed 2-meter (6.6-foot) boom on the east side of the spacecraft. Redundant three-axis magnetometers are mounted on a deployable 3-meter (9.8-foot) boom, attached to the anti-earth face, to minimize interference from the spacecraft.

Deployed Spacecraft Outline/Dimensions



Spacecraft On-Orbit Configuration

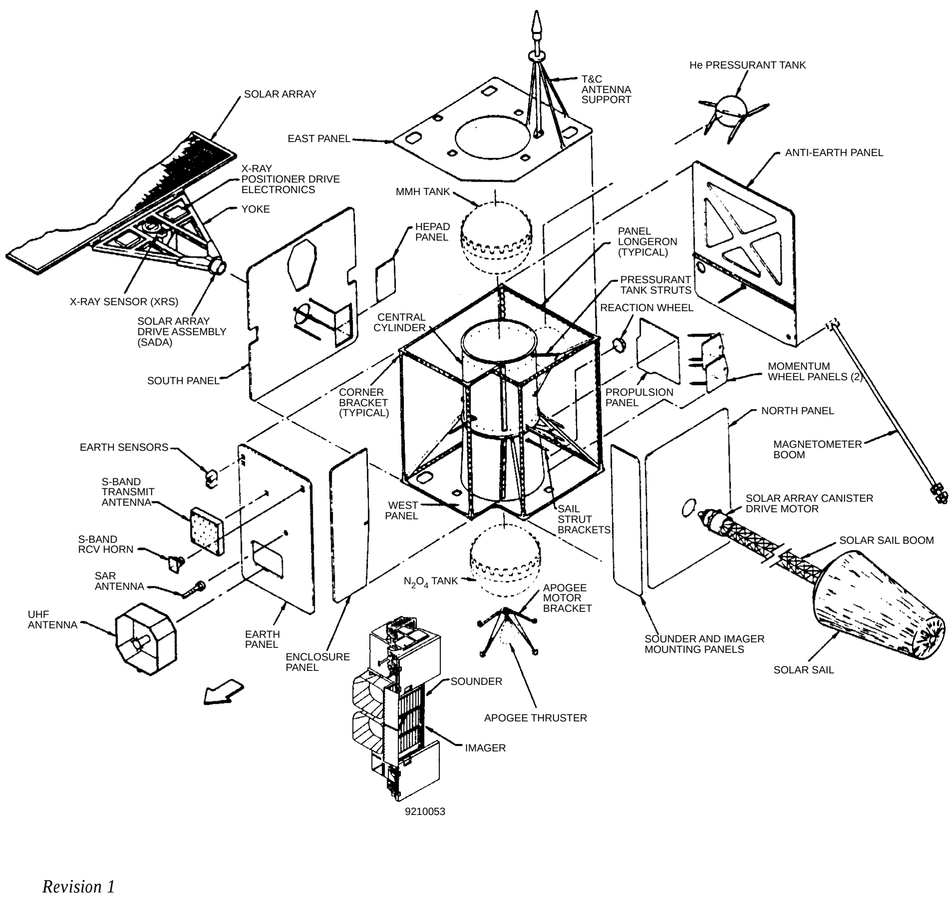


9210051

The Spacecraft is modular in design, allowing for assembly and test accessibility. It is made up of the propulsion module, electronics module, four major panels (earth, north, south, and anti-earth facing), the solar array and drive, the solar sail and boom, and Imager, Sounder, and space environment sensors. In its on-orbit operational configuration, the spacecraft is about 26.9 meters (88.3 feet) in overall length (solar sail to trim tab), about 5.9 meters (19.3 feet) in overall height (telemetry and command antenna to the dual magnetometers), and 4.9 meters (16.0 feet) in overall width (dual magnetometer to UHF antenna).

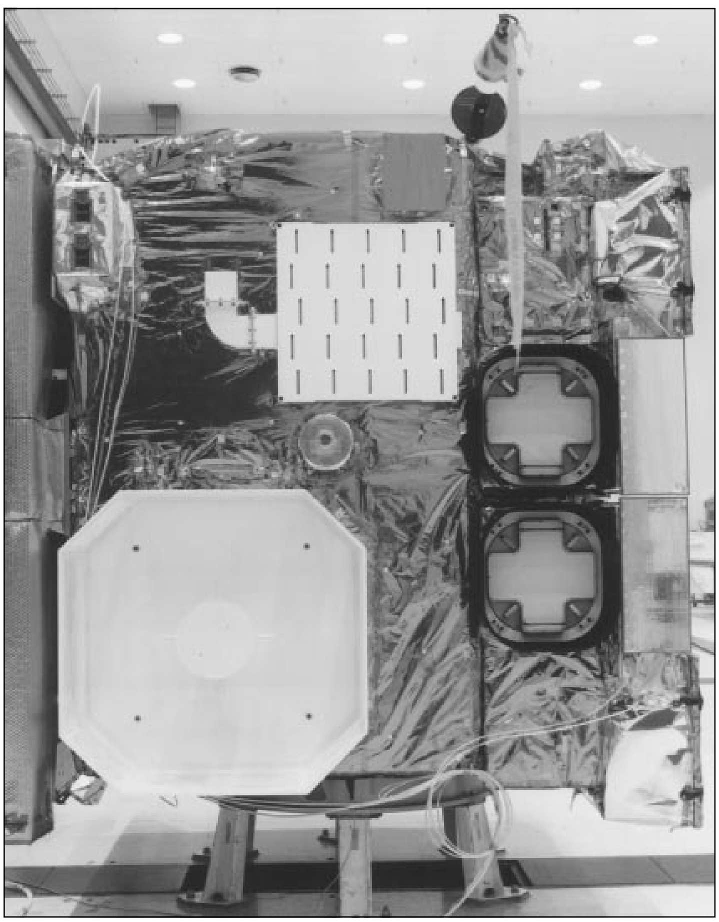
The spacecraft is designed for the Atlas I or Atlas II launch vehicle (adaptable to Space Transportation System). The on-orbit operational life is 5 years with the capability to maintain stationkeeping at ±0.5° in longitude and ±0.5° in latitude. The spacecraft provides for simultaneous and independent operation of the Imager and Sounder instruments over its lifetime and is capable of generating signals based on the image motion compensation (spacecraft orbit and attitude) and mirror motion compensation adjustments (spacecraft internal dynamics), with the capability to reprogram the on-board computer after launch.

Spacecraft Expanded Configuration



In contrast to the current GOES spacecraft, the GOES I-M design features three-axis stabilization, rather than spin stabilization, enabling the Imager and Sounder to continuously observe the earth, and thus monitor, track, and acquire extensive data on dynamic, short-lived weather events. The Imager and Sounder are now independent instruments that also can be operated simultaneously. The Imager has 5 imaging channels and the Sounder 19 sounding channels. In addition, the space environment monitor is provided with two more energetic particles sensor channels, broadening the range of particle energies that can be detected. The transmission of weather facsimile data, which was time-shared in the earlier GOES, is now independent of the Imager and Sounder instruments.

GOES with Imager and Sounder Installed



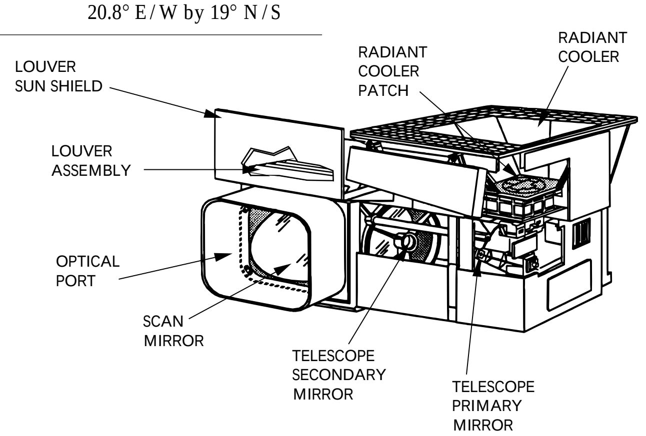
Imager

The GOES I-M Imager is a five-channel (one visible, four infrared) imaging radiometer designed to sense radiant and solar reflected energy from sampled areas of the earth. By means of a servo-driven, two-axis gimballed mirror scan system in conjunction with a Cassegrain telescope, the Imager's multispectral channels can simultaneously sweep an 8-kilometer (5-statute mile) north-to-south swath along an east-to-west / west-to-east path, at a rate of 20° (optical) east-west per second.

Imager Instrument Characteristics

Channel	Detector Type	Nominal square IGFOV at nadir
1 (Visible)	Silicon	1 km
2 (Shortwave)	InSb	4 km
3 (Moisture)	HgCdTe	8 km
4 (Longwave 1)	HgCdTe	4 km
5 (Longwave 2)	HgCdTe	4 km

Parameter	Performance
FOV defining element	Detector
Channel-to-channel alignment	28 µrad (1.0 km) at nadir
Radiometric calibration	300 K internal blackbody and space view
Signal quantizing	10 bits, all channels
Scan capability	Full earth, sector, area
Output data rate	2,620,800 b / s
Imaging areas	20.8° E / W by 19° N / S



Imaging Channels Allocation

Channel Number	Wavelength Range (µm)	Range of Measurement	Meteorological Objective and Maximum Temperature Range
1	0.55 to 0.75	1.6 to 100% albedo	Cloud cover
2 (GOES-I/J/K)	3.80 to 4.00	4 to 320 K	Nighttime clouds (space – 340 K)
2 (GOES-L/M)	3.80 to 4.00	4 to 335 K	Nighttime clouds (space – 340 K)
3 (GOES-I/J/K/L)	6.50 to 7.00	4 to 320 K	Water vapor (space – 290 K)
3 (GOES-M)	13.0 to 13.7	4 to 320 K	Cloud cover and height
4	10.20 to 11.20	4 to 320 K	Sea surface temperature and water vapor (space – 335 K)
5 (GOES-I/J/K/L)	11.50 to 12.50	4 to 320 K	Sea surface temperature and water vapor (space – 335 K)
5 (GOES-M)	5.8 to 7.3	4 to 320 K	Water vapor

Imager Performance Summary

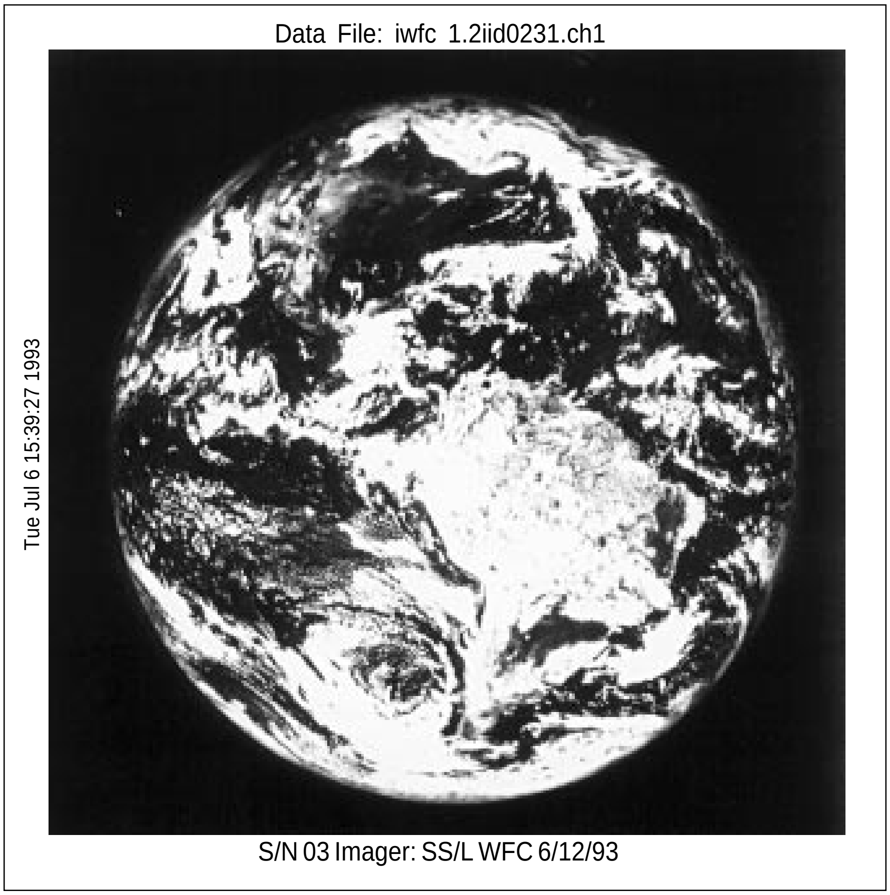
Parameter	Performance		
System absolute accuracy	Infrared channel ≤1 K Visible channel ±5% of maximum scene radiance		
System relative accuracy	Line to line	≤0.1 K	
	Detector to detector	≤0.2 K	
	Channel to channel	≤0.2 K	
	Blackbody calibration to calibration	≤0.35 K	
Star sense area	21° N/S by 23° E/W		
Imaging rate	Full earth ≤26 min		
Time delay	≤3 min		
Fixed Earth projection and grid duration	24 hours		
Data timeliness			
Spacecraft processing	≤30 s		
Data coincidence	≤5 s		
Imaging periods		*Noon ±8 Hours*	*Midnight ±4 Hours*
Image navigation accuracy at nadir		4 km	6 km
Registration within an image*	25 min	50 µrad	50 µrad
Registration between repeated images*	15 min	53 µrad	70 µrad
	90 min	84 µrad	105 µrad
	24 h	168 µrad	168 µrad
* For spec orbit	48 h	210 µrad	210 µrad
Channel-to-channel registration		28 µrad	28 µrad (IR only)

Wide Field Earth Target

Wide Field Collimator Test

Because the Earth subtends a very wide angle as seen from the operational GOES spacecraft, simulating on-orbit conditions in the laboratory for test purposes presents a special challenge to the test designers. The usual test source would be a standard collimator placed in front of the instrument being tested and capable of simulating only a small portion of an accurately sized image. Though much useful information can be obtained by this method, it has long been desired to simulate the entire scene that an instrument would observe from geostationary orbit.

Image of Earth Target from S/N 03 Imager



The wide field collimator provides such a “flight like” scene by utilizing a very wide field lens (~18 degrees) originally designed for use with an aerial reconnaissance camera. The collimator projects a high quality image of the Earth, obtained from an actual satellite photograph, through the wide field lens, producing the correct angular extent as viewed from geostationary orbit. The imager then forms an image of the Earth scene as it would while operating in space. A comparison of the resulting image with the original verifies the Imager’s performance. This method yields a simple end-to-end test that relies on the fewest number of assumptions.

The Subsystem

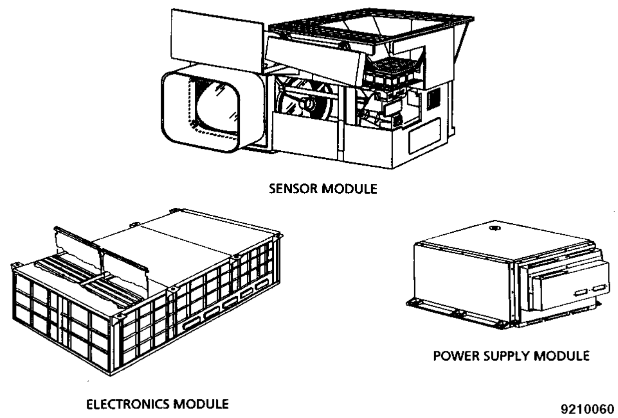
The Imager consists of electronics, power supply, and sensor modules. The sensor module, containing the telescope, scan assembly, and detectors, is mounted on a baseplate external to the spacecraft, together with the shields and louvers for thermal control. The electronics module provides redundant circuitry and performs command, control, and signal processing functions; it also serves as a structure for mounting and interconnecting the electronic boards for proper heat dissipation. The power supply module contains the converters, fuses, and power control for interfacing with the spacecraft electrical power subsystem. The electronics and power supply modules are mounted on the spacecraft internal equipment panel.

Signal flow through the Imager maintains the maximum capability of each part of the optical, detection, and electronic subsystems in order to preserve the quality and accuracy of the sensed information. The scene radiance, collected by the Imager's optical system, is separated into appropriate spectral channels by beam splitters that also route the signal to various infrared (IR) detector sets where they are imaged onto the respective detectors for each channel. Each detector converts the scene radiance into an electrical signal that is amplified, filtered, and digitized; the resulting digital signal is routed to a sensor data transmitter, then to an output multiplexer for downlinking to a ground station.

A user may request one or a set of images that start at a selected latitude and longitude (or lines and pixels) and end at another latitude and longitude (or lines and pixels). The Imager responds to scan locations that correspond to those command inputs. The image frame may include the entire earth's disk or any portion of it and the frame may begin at any time. Scan control is not limited in scan size or time; an entire viewing angle of 21° north / south (N / S) by 23° east / west (E / W) is available for star sensing. Imaging limits are 19° N / S by 20.8° E / W. Requests for up to 63 repeats of a given image can be made by ground command. A frame sequence can be interrupted for "priority" scans; the system will scan a priority frame set or star sense, then automatically return to the original set.

Infrared radiometric quality is maintained by frequent and timed interval views (2.2, 9.2, or 36.6 seconds, ground command selectable) of space for reference. Less frequent views of the full-aperture internal blackbody establishes a high-temperature baseline for calibration in orbit. Via ground command or automatically, repeat of this calibration every 10 minutes is more than adequate to maintain accuracy of the output data under the worst conditions of time and temperature. In addition to radiometric calibration, the amplifiers and data stream are checked regularly by an internal staircase signal to verify stability and linearity of the output data.

Imager Modules



Operation

The Imager is controlled via a defined set of command inputs. Position and size of an area scan are controlled by command, so the instrument is capable of full-earth imagery (19° N/S by 20.8° E/W), sector imagery that contains the edges of the earth's disk, and various area scan sizes totally enclosed within the earth scene. However, the maximum scan width processed by the operations ground equipment is 19.2°. Area scan selection permits continuous, rapid viewing of local areas for accurate wind determination and monitoring mesoscale phenomena. Area scan size and location are definable to less than one visible pixel, yielding complete flexibility.

The Imager's flexibility of operation also provides a star sensing capability (as dim as B0-class fourth magnitude). Once the time and location of a star is predicted, the Imager is pointed to that location within its 21° N/S by 23° E/W field of view (FOV) and the scan stopped. As the star image passes through the 1- by 8-kilometer visible array, it is sampled at a rate of 21,817 samples per second. The star sense sensitivity is enhanced by increasing the electronic gain and reducing the noise bandwidth of the visible preamplifiers, permitting sensing of a sufficient number of stars for image navigation and registration purposes.

By virtue of its digitally controlled scanner, the Imager provides operational imaging from full earth scan to mesoscale area scans. Accuracy of location is provided by the absolute position control system, in which position error is noncumulative. Within the instrument, each position is defined precisely and any chosen location can be reached and held to a high accuracy. This registration accuracy is maintained along a scan line, throughout an image and over time. Total system accuracies relating to spacecraft motion and attitude determination also include this allocated error.

Motion of the Imager and Sounder scan mirrors causes a small but well-defined disturbance of spacecraft attitude, which is gradually reduced by spacecraft control but at a rate too slow to be totally compensated. Since all physical factors of the scanners and spacecraft are known and scan positions are continuously provided by the Imager and Sounder, the disturbances caused by each scan motion on the spacecraft are easily calculated by the attitude and orbit control subsystem (AOCS). A compensating signal is developed and applied in the scan servo-control loop to bias scanning and offset the disturbance. This simple signal and control interface provides corrections that minimize any combination of effects. With this technique, the Imager and Sounder are totally independent, maintaining image location accuracy regardless of the other unit's operational status. If needed, this mirror motion compensation scheme can be disabled by command.

The AOCS also provides compensation signals that counteract spacecraft attitude, orbital effects, and predictable structural-thermal effects within the spacecraft-instrument combination. These effects are used to fit parameters for a 24 hour period during which they are used to predict disturbances. Ground-developed corrective algorithms are fed to the instruments via the AOCS as a total image motion compensation (IMC) signal that includes the mirror motion compensation described above.

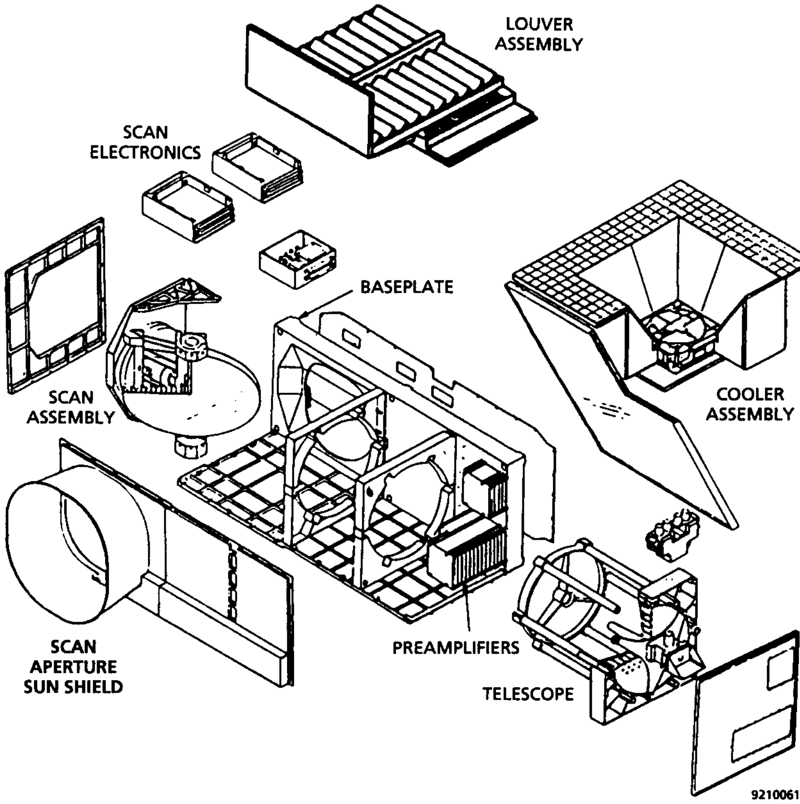
Sensor Module

The sensor module consists of a cooler assembly, telescope, aft optics, preamplifiers, scan aperture sunshield, scan assembly, baseplate, scan electronics, and louver assembly. The baseplate becomes the optical bench to which the scan assembly and telescope are mounted. A passive louver assembly and electrical heaters on the base aid thermal stability of the telescope and major components. A passive radiant cooler with a thermostatically controlled heater maintains the IR detectors at 94 K during the 6 months of winter solstice season and then at 101 K for the remainder of the year for efficient operation. A backup temperature of 104 K is also provided. The visible detectors are at instrument temperature of 13 to 30 °C. The preamplifiers convert the low-level signals to higher-level, low-impedance outputs for transmission by cable to the electronics module.

Imager Optics

To gather emitted or reflected energy, the scanner moves a flat mirror to produce a bidirectional raster scan. Thermal emissions and reflected sunlight from the scene pass through a scan aperture protected by a sun shield, then the precision flat mirror deflects them into a reflective telescope. The telescope, a Cassegrain type with a 31.1-centimeter (12.2-inch) diameter primary mirror, concentrates the energy onto a 5.3-centimeter (2.1-inch) diameter secondary mirror. The surface shape of this mirror forms a long focal length beam that passes the energy to the detectors via relay optics.

Expanded View of Sensor Module

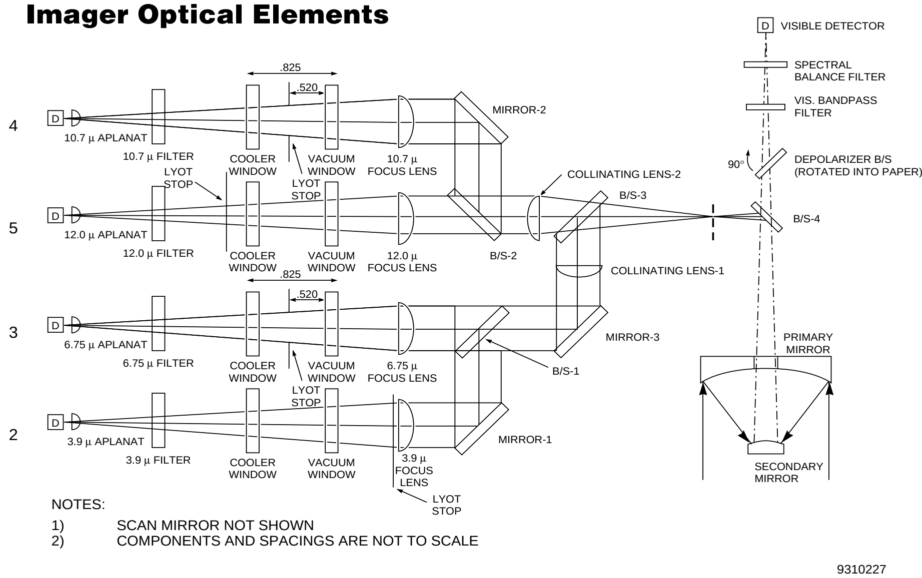


Dichroic beamsplitters (B/Ss) separate the scene radiance into the spectral bands of interest. The IR energy is deflected to the detectors within the radiative cooler, while the visible energy passes through the dichroic beamsplitters and is focused on the visible detector elements. The IR energy is separated into the 3.9, 6.75, 10.7, and 12 µm channels. These four beams are directed into the radiant cooler, where the spectral channels are defined by cold filters. Each of the four IR channels has a set of detectors defining the field size and shape.

Optical performance is maintained by restricting the sensor module total temperature range, and radiometric performance is maintained by limiting the temperature change between views of cold space (rate of change of temperature). Thermal control also contributes to channel registration and focus stability. Thermal control design includes:

- Maintaining the Imager as adiabatically (thermally isolated) as possible from the spacecraft structure.
- Controlling the temperature during the hot part of the synchronous orbit diurnal cycle (when direct solar heating enters the scanner aperture) with a north-facing radiator whose net energy rejection capability is controlled by a louver system.

Imager Optical Elements



- Providing makeup heaters within the instrument to replace the IR energy loss to space through the scan aperture during the cold portion of the diurnal cycle.
- Providing a sun shield around the scan aperture (outside the instrument FOV) to block incident solar radiation into the instrument, thus limiting the time the aperture can receive direct solar energy.

Detectors

The Imager instrument simultaneously acquires radiometric data in five distinct wavelengths or channels. Each of the five radiometric channels is characterized by a wavelength band denoting primary spectral sensitivity. The five channels are broadly split into two classes: visible (channel 1) and infrared (channels 2-5). For these five channels, the Imager contains a total of 22 detectors.

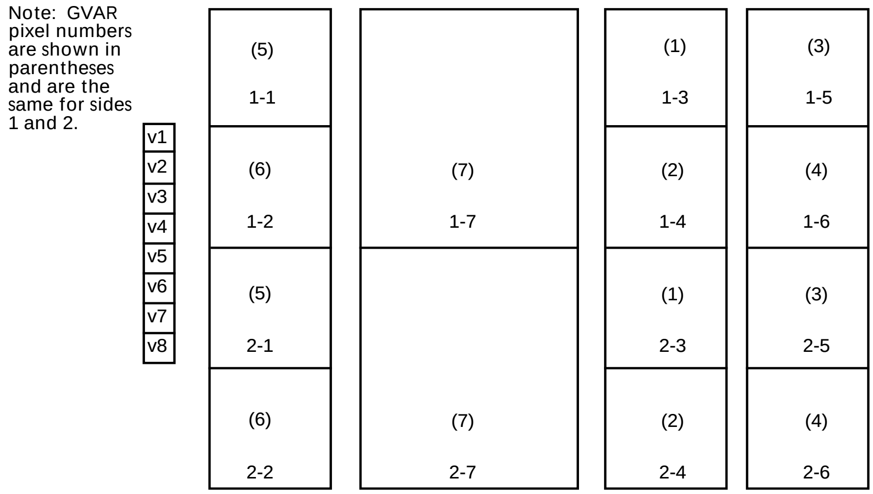
Visible Channel

The visible silicon detector array (channel 1) contains eight detectors (v1-v8). Each detector produces an instantaneous geometric field of view (IGFOV) that is nominally 28 microradians (µrad) on a side. At the spacecraft's suborbital point, on the surface of the earth, 28 µrad corresponds to a square pixel that is 1 kilometer (0.6 statute mile) on a side.

Infrared Channels

The IR channels employ four-element InSb (indium antimonide) detectors for channel 2 (3.9 µm), two-element HgCdTe (mercury cadmium telluride) detectors

Imager Detectors



CHANNEL	1	2	3	4	5
CENTRAL WAVELENGTH (μm)	0.65	3.9	6.75	10.7	12.0
DETECTOR IGFOV (nominal, μrad)	28	112 InSb	224 HgCdTe	112 HgCdTe	112 HgCdTe

9401131

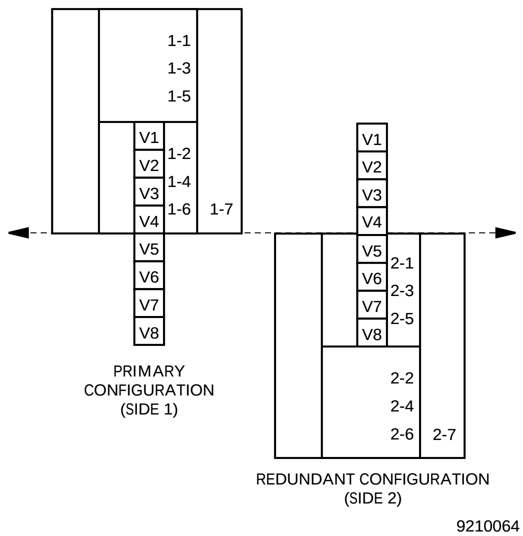
for channel 3 (6.75 μm), and four-element HgCdTe detectors for channels 4 (10.7 μm) and 5 (12 μm). A four-element set consists of two-line pairs providing redundancy along a line. Each detector in channels 2, 4, and 5 is square, with an IGFOV of 112 μrad, corresponding to a square pixel 4 kilometers per side at the suborbital point. Channel 3 contains two square detectors, each of which provides an IGFOV of 224 μrad, resulting in a suborbital pixel 8 kilometers on a side.

Configuration

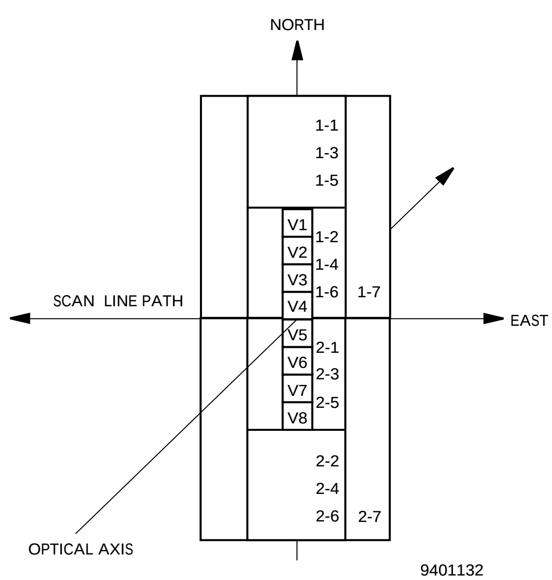
The five detector arrays are configured in either a side 1 or a side 2 mode, either of which can be the redundant set by choosing side 1 or side 2 electronics. The entire visible channel array (v1 to v8) is always enabled in both modes. In side 1 mode the IR channels have only their upper detectors (1-1 to 1-7) enabled and in side 2, only their lower detectors (2-1 to 2-7). The GVAR numbering of the pixels is shown in the diagram.

Though physically separated in the instrument, the detector arrays are optically registered. Small deviations in this optical registration are due to physical misalignments in constructing and assembling the instrument and to the size of the detector elements. These deviations consist of fixed offsets that are corrected at two levels: within the instrument sampling electronics and on the ground by the operations ground equipment (no corrections are applied during star sensing).

Operational Configurations



Optical Configuration



Because the combination of scan rate (20°/s) and detector sample rate (5460 samples/s for IR and 21840 samples/s for visible) exceeds the pixel E/W IGFOV, the Imager oversamples the viewed scene. Each visible sample is 16 µrad E/W and each IR sample is 64 µrad E/W.

Scan Control

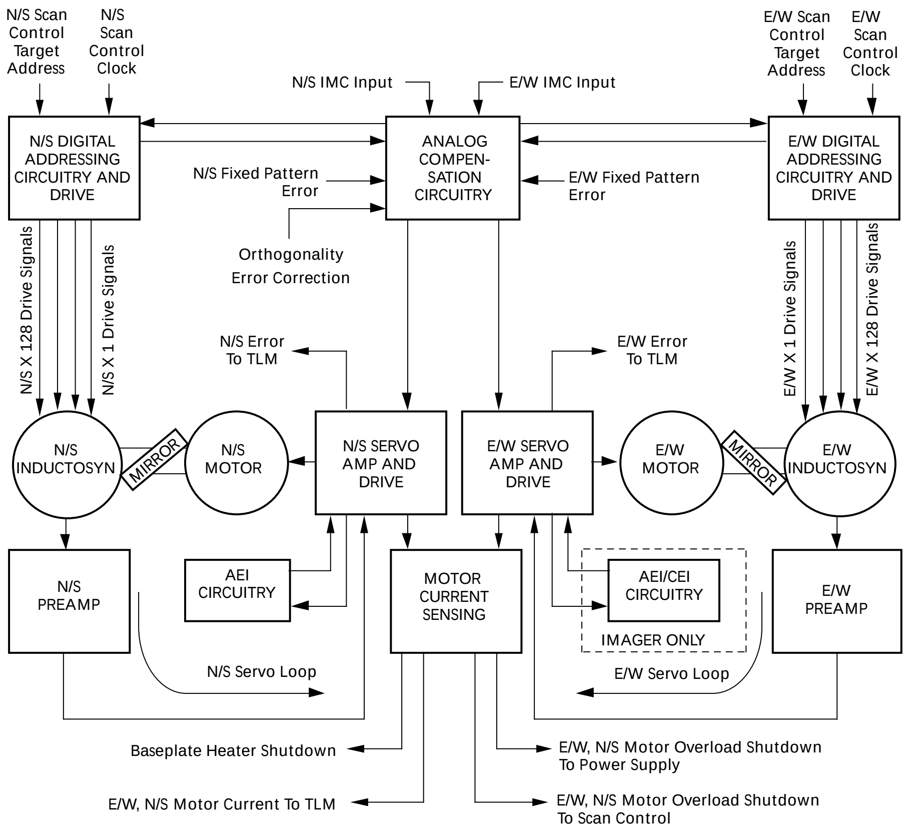
The scanning mirror position is controlled by two servo motors, one for the N/S gimbal angle, and one for the E/W scanning gimbal angle. Each servo motor has an associated inductosyn that measures the mechanical shaft rotation angle. The scanning mirror position and, hence, the coordinate system used for the Imager are measured in terms of the inductosyn outputs. Scan control for both axes is generated by establishing a desired angular position for the mirror. The desired angle is input to an angular position sensor (one inductosyn for each axis), which produces a displacement error signal. This signal is fed to a direct drive torque motor (one inductosyn for each axis) that moves the mirror and sensor to the null location.

For E/W deflection, the direct-drive torque motor is mounted to one side of the scan mirror and the position-sensing device (inductosyn position encoder) is mounted on the opposite side. All rotating parts are on a single shaft with a common set of bearings. Using components of intrinsically high resolution and reliability, coupling of the drive, motion, and sensing is therefore very tight and precise. North/South motion is provided by rotating the gimbal (holding the above components) about the optical axis of the telescope. This rotating shaft has the rotary parts of another torque motor and inductosyn mounted to it, again providing the tight control necessary.

Servo control is not absolutely accurate due to noise, drag, bearing imperfections, misalignment, and imperfections in the inductosyns. The principal servo pointing and registration errors are fixed pattern errors caused by the inductosyn position sensor and its electronic drive unit. Variations in individual inductosyn pole patterns, imbalance between the sine and cosine drives, cross-talk and feed-through in these circuits, and digital-to-analog (D/A) conversion errors contribute to the fixed-pattern errors. These errors are measured at ambient conditions and the correction values stored in programmable read-only memory. Corrections are applied in the scanner as a function of scan address. The measured values of fixed pattern errors vary between ±15 µrad (mechanical) with a frequency of up to four times the inductosyn cycle; after correction, the error is reduced to within ±4 µrad. Variations of the fixed pattern error over temperature, life, and radiation conditions are minimized by design, and residual errors are accounted for in the pointing budget.

Drive and error sensing components used for the two drive axes are essentially identical. The E/W drive system has a coherent error integrator (CEI) circuit that automatically corrects for slight changes in friction or other effects. Control components are optimized for their frequency and control characteristics, and logic

Scan Control Schematic



Legend

AEI	Average Error Integrator	IMC	Image Motion Compensation
CEI	Coherent Error Integrator	N/S	North/South
E/W	East/West	TLM	Telemetry

9210066

is developed for the precise control of position in response to a system-level control processor.

Scan Operation

Scan control is initiated by an input command that sets start and end locations of an image frame. A location is defined by an inductosyn cycle and increment number within the cycle, the increment number determining the value of sine and cosine for that location. Each E/W increment corresponds to 8 µrad of E/W mechanical rotation and 16 µrad of E/W optical rotation. Each N/S increment corresponds to 8 µrad of N/S mechanical and optical rotation. The distance between a present and start location is recognized, causing incremental steps (8 µrad) to be taken at a high rate (10°/s) to reach that location. After the E/W slew is completed, the N/S slew begins. From the scan start position, the same pulse rate and increments are used to generate the linear scan. The scan mirror inertia smoothes the small incremental steps to much less than the error budget.

At the scan line end location (where the commanded position is recognized) the control system enters a preset deceleration / acceleration. During this 0.2-second interval, the scan mirror velocity is changed in 32 steps by using a 32-increment cosine function of velocity control. This slows and reverses the mirror so that it is precisely located and moving at the exact rate to begin a linear scan in the opposite direction. During this interval, the N / S scan control moves the gimbal assembly 224 µrad (28 increments of 8 µrad) in the south direction. Linear scanning and N / S stepping continue until the southern limit is reached.

Scan to space for space clamp, or to star sensing, or to the IR blackbody uses the same position control and slew functions as for scan and retrace. Command inputs (for star sensing or priority frames) or internal subprograms (for space clamp and IR calibration) take place depending on the type of command, time factors, and location.

Image Generation

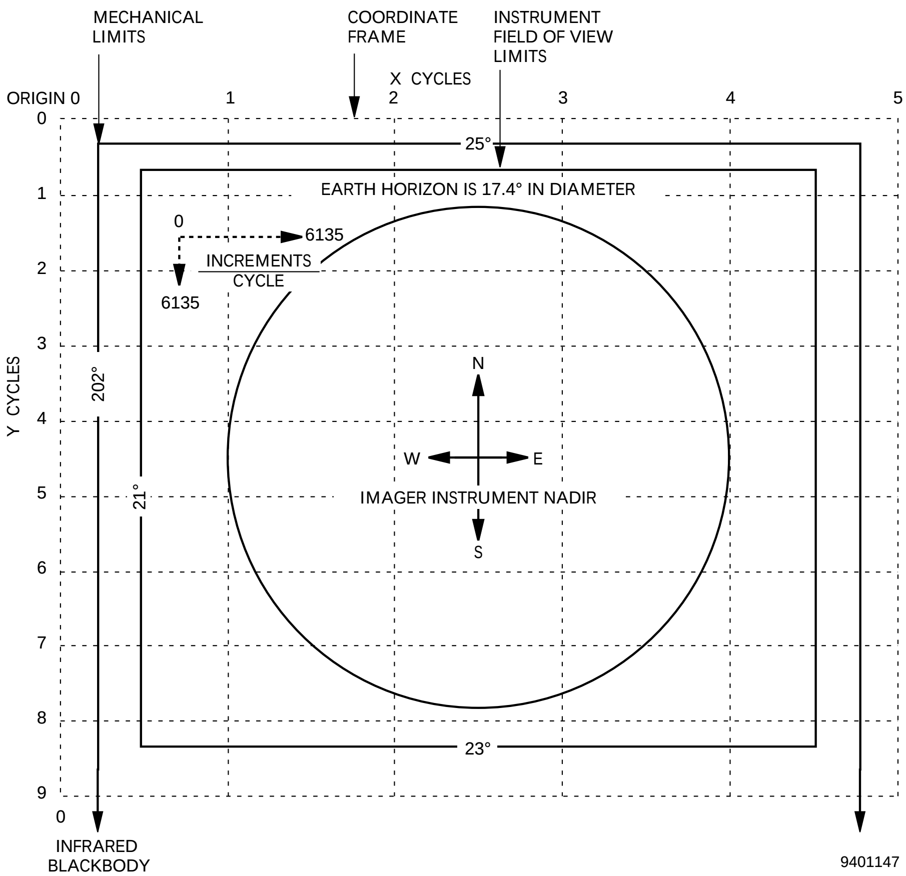
During imaging operations, a scan line is generated by rotating the scan mirror in the east-to-west direction (20° / s optically) while concurrently sampling each active imaging detector (5460 / s for IR and 21840 / s for visible). At the end of the line, the scan mirror elevation is changed by a stepped rotation in the north-to-south direction. The next scan line is then acquired by rotating the scan mirror in the (opposite) west-to-east direction, again with concurrent detector sampling. Detector sampling occurs within the context of a repeating data block format. In general, all visible channel detectors are sampled four times for each data block while each active IR detector is sampled once per data block.

The mapping between cycles and increments and the instrument FOV are referenced to a coordinate frame whose origin is zero cycles and zero increments (northwest corner of the frame). In geostationary orbit, the earth will be centered within the frame, at instrument nadir, which corresponds closely to the spacecraft suborbital point, also centered in the frame. The GVAR coordinate system is in line / pixel space and has its origin in the NW corner.

Three components making up the total misalignment in the sampled data are corrected by the instrument electronics and operations ground equipment:

- A fixed E / W offset caused by channel-to-channel variations in the signal processing filter delays.
- A fixed E / W and / or N / S offset caused by optical axis misalignments in the instrument assembly.
- A variable E / W and / or N / S offset caused by image rotation.

Imager Coordinate Frame



Electronics

The Imager electronics consist of a preamplifier and thermal control in the sensor assembly; command and control, telemetry, and sensor data processing contained in the electronics module; and the power supplies. The scan control electronics are contained in the electronics module. The servo preamplifiers are located at the scanner in the sensor module.

Signal Processing

Preamplification of the low-level visible and IR channel signals occurs within the sensor module. These analog signals are routed to the electronics module, which amplifies, filters, and converts the signals to digital code. All channels in the visible and IR bands are digitized to one part in 1024 (10 bits), the visible for high-quality visible imagery and to aid star sensing capability, and the IR for radiometric measurement. Data from all channels move in continuous streams throughout the system, thus each channel's output must enter a short-term memory for proper placement in the data stream. Each channel is composed of a detector, preamplifier, analog-to-digital (A/D) converter, and signal buffer. All signal chains are totally independent and isolated. Redundant chains of signal processing circuitry are provided with each circuit ending in a line driver designed to interface with the spacecraft transmitter (the video and formatter are redundant for the IR channels only).

Electronic Calibration

Electronic calibration signals are injected into the preamplifiers of channels 3, 4, and 5 while the Imager is looking at space. Electronic calibration is inserted after the preamplifiers of channels 1 and 2. Sixteen precise signal levels derived from a stepped D/A converter are inserted during the 0.2-second spacelook. The calibration signal, derived from a 10-bit converter of 0.5-bit accuracy, provides the accuracy and linearity for precise calibration.

Visible Channel

Each detector element of the visible channel has a separate amplifier/processor. These current-sensing preamplifiers convert the photon-generated current in the high-impedance silicon detector into an output voltage, with a gain of about 10^8 V/A. These preamplifiers are followed by postamplifiers that contain electrical filtering and space clamping circuits. The digitization of the data signals is also part of the space clamp circuitry. The visible information is converted to 10-bit digital form, providing a range from near 0.1% to over 100% albedo. Differences of approximately 0.1% are discernible, and the linear digitization provides for system linearity errors of 0.5 bit in the conversion process. The star sense channel uses the same visible channel detectors, but boosts the gain by approximately 4 times and reduces the bandwidth.

Infrared Channels

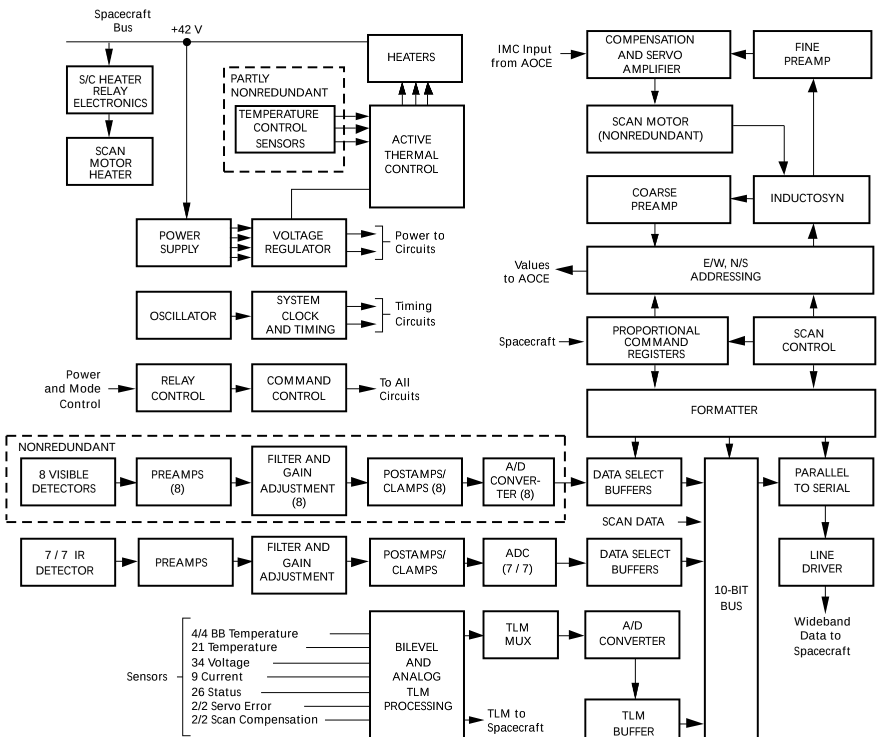
The IR channels have a separate amplifier/processor for each detector element. The 3.9 μm channel has a hybrid current sensing preamplifier for the high-impedance InSb detector. Individual preamplifiers for channels 3, 4, and 5 are mounted on the cooled patch in the sensor module.

The IR information is converted to 10-bit digital form, providing a range from near 0.1% to over 100% of the response range. Each channel has a gain established for a space-to-scene temperature of 320 K. The 10-bit digital form allows the lowest calculated noise level to be differentiated. The digital system is inherently linear with A/D converter linearity and accuracy to 0.5 bit. The binary-coded video is strobed onto the common data bus for data formatting by the system timing and control circuitry.

Formatting

The data format of Imager information is made up of blocks of data generated in a given sample time period. The Imager scans an 8-kilometer swath using combinations of 1-kilometer visible detectors, and 4- and 8-kilometer IR detectors. Oversampling causes the IR data to be collected each 64 μrad (2.28 kilometers or

Imager Block Diagram



Legend

A/D	Analog to Digital	MUX	Multiplexer
ADC	Analog-to-Digital Converter	S/C	Spacecraft
AOCE	Attitude and Orbit Control	TLM	Telemetry
IR	Electronics		

9210068

1.42 statute miles at nadir) using a data block format where the location of each bit within the data stream is completely identified, and all information can be separated and reformatted on the ground. The visible detectors are sampled four times during this 64 µrad period, yielding a collection rate of 16 µrad per sample. The four sets of visible data combine with one set from each IR detector in each data block.

The formats consist of data blocks, 480 bits in a block, each block being broken into 48 10-bit words. The format sequence during an active scan begins with a start-of-line command from the scan control system that synchronizes the data formatter with scan control and occurs when the Imager mirror is at the start of a scan line. The header format follows, containing block synchronization and data block identifiers, spacecraft and instrument identification, status flags, attitude and orbit control electronics data, coordinates of the current scan mirror position, and fill to complete the data block. After the header block, active scan data blocks follow; these contain synchronization and data block identifiers, motion compensation data, servo error, and radiometric data.

When the mirror reaches the end of the scan line, a scan reversal sequence begins with three active scan data blocks that permit full collection of radiometric data to the end of the scan line. A trailer format, similar to the header format, identifies the 39 blocks of telemetry format data to follow.

Digital signal processing starts where data from the IR and visible detectors and telemetry merge via multiplexing and processing; a parallel-to-serial conversion and data multiplexing take place to bring sensor data together. Other information, such as synchronization pulses, scan location, and telemetry data, is assembled in the data select circuitry. These data are then passed through a line driver where pulse amplitude and impedance levels are set for the transmitter interface. These data are transmitted at a rate of 2.6208 Mb/s or 5460 blocks per second.

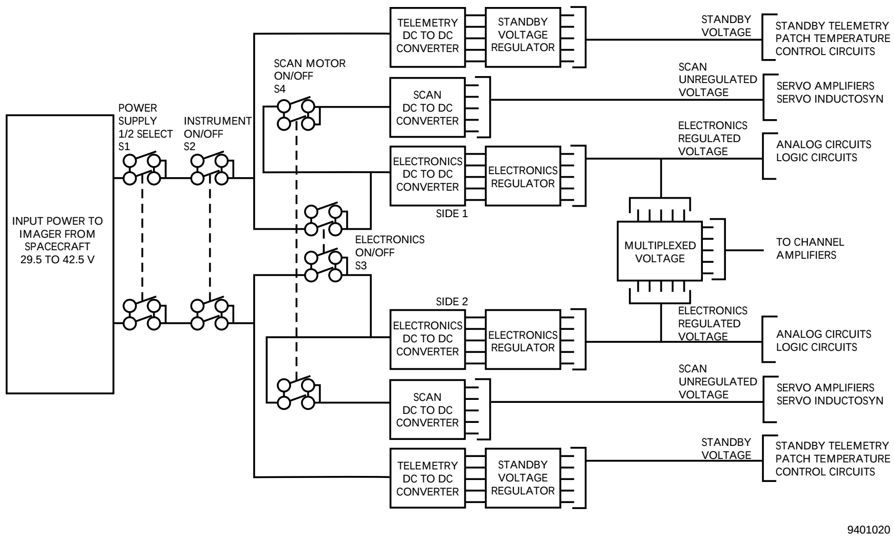
Power Supply

The power supply converts spacecraft main bus voltage (29.5 to 42.5 volts) to the required instrument voltages. There are two sides (1 and 2) to the unit, each totally independent and selected by command, although only one side operates at a time. A protective resistive filter permits operation of all nonredundant circuits (command input circuitry, inductosyn preamplifier, patch temperature control, detector preamplifiers, etc.) by either side. Redundant circuits are powered through separate fused links from the respective side that prevents system loss in the event of failure.

The power converters of both sides accept and convert the bus voltage to a steady 26.5 volts dc using switching regulators. The regulator output voltages power both the main and a standby dc/dc converter for the electronics during normal operation. The main converter consists of a power amplifier, transformer, rectifiers, and filters. It provides unregulated voltage to operate the servo power amplifiers and servo inductosyn drivers, and regulated voltage used principally to operate analog circuitry in the Imager and also power the logic circuitry.

The standby dc/dc converter consists of a synchronized oscillator, rectifiers, filters, and regulators and is used to operate the standby telemetry and patch temperature control circuits. It also provides a boost voltage used to improve the efficiency of the switching regulator and a 40-kHz signal that synchronizes the input to the main converter.

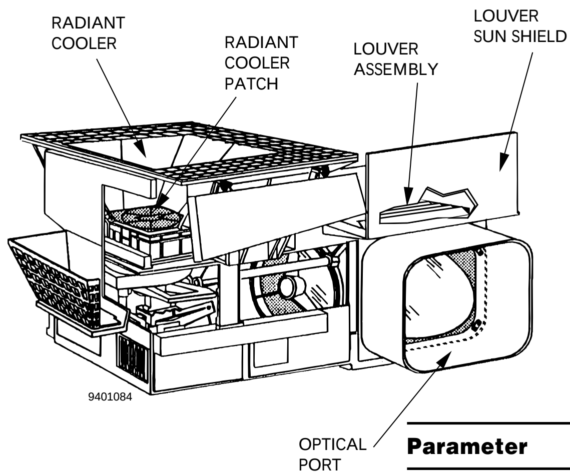
Power Supply Block Diagram



Sounder

The Sounder is a 19-channel discrete-filter radiometer that senses specific data parameters for atmospheric vertical temperature and moisture profiles, surface and cloud top temperature, and ozone distribution. As in the Imager, the Sounder is capable of providing full earth imagery, sector imagery (including earth's disk), and scans of local regions. The nineteen spectral bands (seven longwave (LW), five midwave (MW), six shortwave (SW), and one visible) produce the prime sounding products.

Sounder Instrument Characteristics



Parameter	Performance
FOV defining element	Field stop
Telescope aperture	31.1-cm (12.2-in) diameter
Channel definition	Interference filters
Radiometric calibration	Space and 300 K IR blackbody
Field sampling	Four areas N/S on 280 µrad centers
Scan step angle	280 µrad (10-km nadir) EW
Step and dwell time	0.1, 0.2, 0.4 s adjustable
Scan capability	Full earth and space
Sounding areas	10 km by 40 km to 60° N/S and 60° E/W
Signal quantizing	13 bits, all channels
Output data rate	40 kb/s
Channel-to-channel allignment	22 µrad

Channels	Detector Type	Nominal Circular IGFOV (µrad)
1 to 7 (LW IR)	HgCdTe	242
8 to 12 (MW IR)	HgCdTe	242
13 to 18 (SW IR)	InSb	242
19 (visible)	Silicon	242
Star sense	Silicon	28*

*square detectors

Sounder Detectors Channel Allocation

Detector	Channel Number	Wavelength (µm)	Wave No. (cm^{-1})	Meteorlogical Objective and Max. Temp. Range
Longwave	1	14.71	680	Temperature (space – 280 K)
	2	14.37	696	Sounding (space – 280 K)
	3	14.06	711	Sounding (space – 290 K)
	4	13.64	733	Sounding (space – 310 K)
	5	13.37	748	Sounding (space – 320 K)
	6	12.66	790	Sounding (space – 330 K)
	7	12.02	832	Surface temperature (space – 340 K)
Midwave	8	11.03	907	Surface temperature (space – 345 K)
	9	9.71	1030	Total ozone (space – 330 K)
	10	7.43	1345	Water vapor (space – 310 K)
	11	7.02	1425	Sounding (space – 295 K)
	12	6.51	1535	Sounding (space – 290 K)
Shortwave	13	4.57	2188	Temperature (space – 320 K)
	14	4.52	2210	Sounding (space – 310 K)
	15	4.45	2248	Sounding (space – 295 K)
	16	4.13	2420	Sounding (space – 340 K)
	17	3.98	2513	Surface temperature (space – 345 K)
	18	3.74	2671	Temperature (space – 345 K)
Visible	19	0.70	14367	Cloud

Sounder Performance Summary

Parameter	Performance		
System absolute accuracy	Infared channel ≤ 1 K Visible channel ± 5% of max. scene radiance		
System relative accuracy	Line to line	≤ 0.25 K	
	Detector to detector	≤ 0.40 K	
	Channel to channel	≤ 0.29 K	
	Blackbody calibration to calibration	≤ 0.60 K	
Star sense area	21° N / S by 23° E / W		
Sounding rate	3000 by 3000 km ≤42 min		
Time delay	≤3 min		
Visible channel data quantization	≤0.1% albedo		
Infrared channel data quantization	1 / 3 specified noise equivalent radiance difference (NEΔN)		
Data timeliness			
Spacecraft processing	≤30 s		
Sounding periods		*Noon ±8 Hours*	*Midnight ±4 Hours*
Image navigation accuracy at nadir		10 km	10 km
Registration within 120 minute sounding	120 min	84 µrad	112 µrad
Registration between repeated soundings	24 h	280 µrad	280 µrad
Channel-to-channel registration		28 µrad	28 µrad

The Subsystem

The Sounder consists of sensor, electronics, and power supply modules. The sensor module contains the telescope, scan assembly, and detectors, all mounted on a baseplate external to the spacecraft with shields and louvers for radiation and heat control. The electronics module provides redundant circuitry and performs command, control, and signal processing functions and also serves as a structure for mounting and interconnecting the electronic boards for proper heat dissipation. The power supply module contains the dc/dc converters, fuses, and power control for converting and distributing spacecraft bus power to the Sounder circuits. The electronics and power supply modules are mounted on the equipment panel of the spacecraft (internal north panel).

The Sounder's multi-detector array simultaneously samples four locations of the atmosphere in 0.1-second intervals (0.2- and 0.4-second dwells at the same FOV are also commandable). Each field of view (FOV) provides output from 19 spectral channels in each sample period. Infrared (IR) spectral definition is provided by a rotating wheel that inserts selected filters into the optical path of the detector assembly; the filters are arranged in three spectral bands on the wheel. Wheel rotation is synchronized with stepping motion of the scan mirror.

A user may request by command a set of soundings that start at a selected latitude and longitude and end at another latitude and longitude. The Sounder responds to scan locations that correspond to those command inputs. The sounding frame may include the whole or any portion of the earth and the frame may begin at any time. The Sounder scan control is not limited in scan size or time; thus an entire viewing angle of 21˚ north-to-south by 23˚ east-to-west is available for star location. Sounding limits are 19˚ north/south (N/S) by 19.2˚ east/west (E/W), limited by the scan aperture and end-of-scan-line conditions. Requests for up to 16 repeats of a given location can be made by ground command. Capability is provided for interrupting a frame sequence for "priority" scans. The system will scan a priority frame set or star sense, then automatically return to the original set.

Radiometric quality is maintained by frequent (every 2 minutes) views of space for reference. Less frequent views (20 minutes) of the full aperture internal blackbody establish a high temperature baseline for instrument calibration in orbit. Further, the amplifiers and data stream are checked for stability by an electronic staircase signal during each blackbody reference cycle.

Other aspects of the Sounder are the same as for the Imager.

Operation

The Sounder is controlled by a defined set of command inputs. The instrument is capable of full earth sounding (19° N/S by 19.2° E/W) and sector sounding, including various sounding area sizes totally enclosed within the earth scene. Area scan size can be as small as one sounding location. The sounding dwell at each step is selectable to be 0.2 or 0.4 second in place of 0.1 second. An optional capability is provided for skipping scan lines to increase the rate of area sounding at a dwell time of 0.2 second per sounding.

The Sounder's flexible operation includes a star sensing capability. Once the time and location of a star is predicted, the Sounder is pointed to that location within its 21˚ N/S by 23˚ E/W field of view and the scan stopped. A separate linear array of eight silicon detectors with a 240-µrad N/S coverage, similar to the Imager, is used. As the star image passes through the detectors, the signal is sampled, then encoded and included in each Sounder data block for extraction and use at the ground station. The star sense detectors are sampled at 40 times per second.

Duplication of the four-element array in each of the three bands (longwave, 12 µm to 14.7 µm; midwave, 6.5 µm to 11 µm; and shortwave, 3.7 µm to 4.6 µm) yields the spectral separation of the infrared bands; the filters are arranged on the wheel for efficient use of sample time and optimal channel coregistration. Each detector converts the atmospheric radiance into an electrical signal that is amplified, filtered, and digitized; the resulting digital signal is routed to a sensor data transmitter, then to an output multiplexer for downlinking to a ground station.

By synchronizing filter wheel rotation with the scan mirror's stepping motion, all sampling is accomplished with the mirror in a stopped condition. Upon ground command, the scan system can generate frames of any size or location using west-to-east stepping and east-to-west stepping of 280 µrad, with a north-to-south step of 1120 µrad, continuing the pattern until the desired frame is completed. The visible channel (0.7 µm), not part of the filter wheel, is a separate set of uncooled detectors with the same field size and spacing. These detectors are sampled at the same time as the infrared channels (3, 11, and 18), providing registration of all sounding data.

By virtue of its digitally controlled scanner, the Sounder provides operational sounding from full earth scan to mesoscale area scans. Accuracy of location is provided by the absolute position control system in which position error is noncumulative. Within the instrument, each position is defined precisely and any chosen location can be reached and held to a high accuracy. This registration accuracy is maintained along a scan line, throughout an image and over time. Total system accuracies relating to spacecraft motion and attitude determination also include this allocated error.

Motion of the Imager and Sounder scan mirrors causes a small but well-defined disturbance of spacecraft attitude, which is gradually reduced by spacecraft control but at a rate too slow to be totally compensated. Because all physical factors of the scanners and spacecraft are known and scan positions are continuously provided by the Imager and Sounder, the disturbances caused by each scan motion on the spacecraft are easily calculated by the attitude and orbit control subsystem (AOCS). A compensating signal is developed and applied in the scan servo-control loop to bias scanning and offset the disturbance. This simple signal and control interface provides corrections that minimize any combination of effects. With this technique, the Imager and Sounder are totally independent, maintaining image location accuracy regardless of the other unit's operational status. If needed, this mirror motion compensation scheme can be disabled by command.

The AOCS also provides compensation signals that counteract spacecraft attitude, orbital effects, and predictable structural-thermal effects within the spacecraft-instrument combination. These disturbances are detected from star sensing and land features. Ground-developed corrective algorithms are fed to the instruments via the AOCS as a total image motion compensation signal that includes the mirror motion compensation described above.

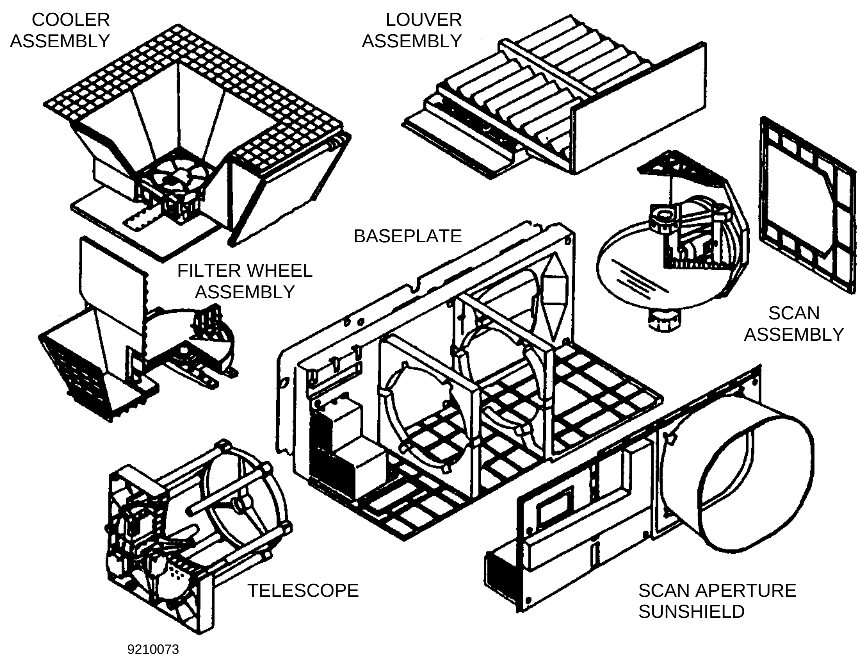
Sensor Module

The sensor module consists of a louver assembly, baseplate, scan assembly, scan aperture sun shield, preamplifiers, telescope, aft optics, filter wheel, and cooler assembly. The baseplate becomes the optical bench to which the scan assembly and telescope are mounted. A passive louver assembly and electrical heaters on the base aid thermal stability of the telescope and major components. A passive radiant cooler with a thermostatically controlled heater maintains the infrared detectors at 94 K during the winter solstice season and 101 K for the remaining portion of the year (with 104 K as backup). The visible and star sense detectors are at instrument temperature of 13 to 30 ˚C. Preamplifiers in the sensor module convert the low-level signals to higher level, low impedance outputs for transmission by cable to the electronics module.

Sounder Optics

The Sounder telescope is similar to that of the Imager. Dichroic beamsplitters separate the scene radiance into the spectral bands of interest. The IR energy is deflected toward the detectors located on the coldest stage of the radiative cooler, while the visible energy passes through a dichroic beamsplitter and is focused on the visible (sounding and star) detector elements. The SW and MW bands are reflected by another dichroic beamsplitter and the LW is transmitted through it. Optical separation of the 18 IR channels takes place at the filter wheel assembly.

Expanded View of Sensor Module

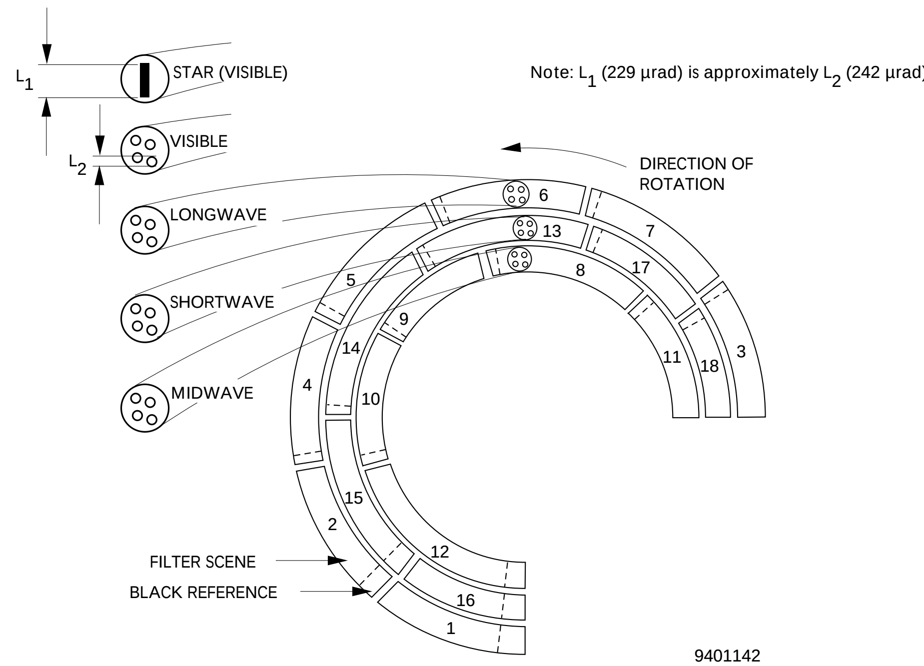


Filter Wheel

The filter wheel is a 28.2-centimeter (11.1-inch) diameter disk containing 18 filter windows divided into three concentric rings, one ring for each IR detector group. The outer ring contains seven LW channels, the middle ring contains six SW channels, and the inner ring contains five MW channels. Filter angular lengths are selected to provide nearly equal performance margin in each channel. The wheel has approximately one-fourth of its area clear of filters. By synchronizing the stepping of the scan mirror to occur in this “dead zone,” the wheel can continue rotating while the mirror steps to the next location and is stopped while the 18 channels are sampled. Stopping the mirror ensures that all channels sample the same column of the atmosphere; holding and sampling in 0.075 second provides virtually simultaneous sampling of the channels.

The first channels to be sampled are high altitude sensors that have little spatial definition and are less affected by the settling characteristics of the scan mirror. The earth surface viewing channels are grouped near the end of the sounding

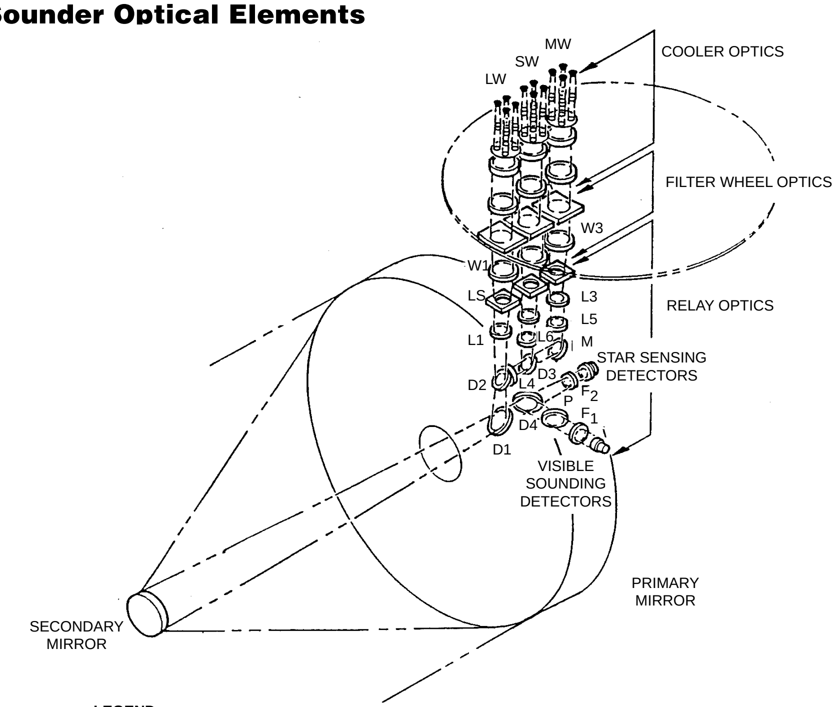
Filter Wheel and Channel Separation



period for maximum stability and coregistration. Though not viewed through the filter wheel, the visible detectors are gated so that they sample the same atmospheric column at the same time as the IR channels.

The filter wheel acts as the spectral defining element in the optics, though it also has a major effect on radiometric stability and signal quality. Each filter has a very narrow spectral bandpass, restricting the radiant input from the scene and contributions from optical parts in the path to the filter wheel. From filter wheel to detectors, there is no spectral limit other than a broadband limiting filter in the cooler. Any small deviation of radiance in this area may cause unwanted noise in the signal. To reduce emitted energy that might cause random noise and to provide very low background radiance input to the detectors, the filter wheel is cooled to 235 K. The temperature of the filter wheel housing is brought to about 238 K by thermal connection to a radiating surface. Heaters and a precision temperature control circuit maintains the housing within 1 ˚C of the set temperature.

Sounder Optical Elements



LEGEND

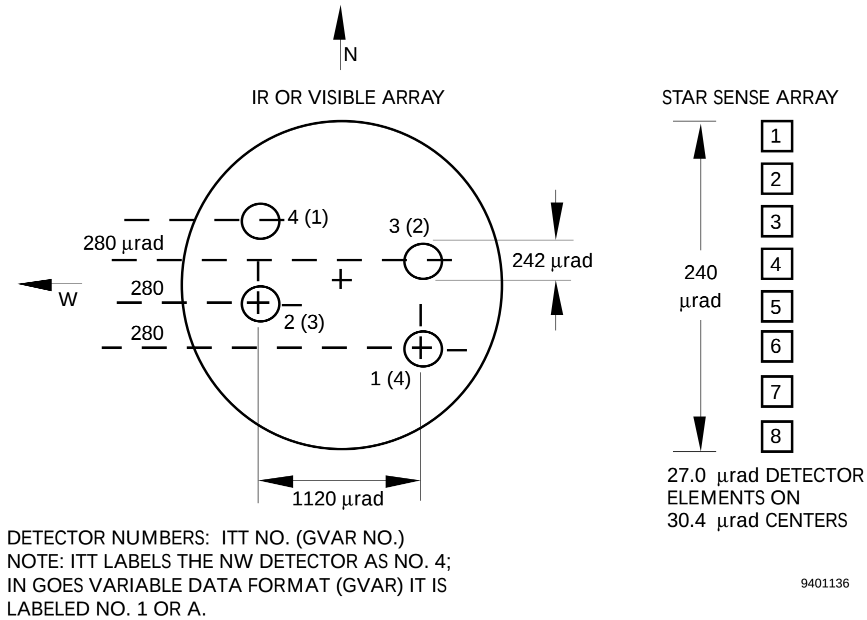
D1	Dichroic Beamsplitter, Visual/IR
D2	Dichroic Beamsplitter, LW/(SW+MW)
D3	Dichroic Beamsplitter, SW/MW
L1-L6	Intermediate Lenses
M	Folding Mirror (MW)
D4	STAR/Visual Trichroic Beamsplitter
W1	Filter Wheel, Window
W2	Filter Wheel, Window
W3	Filter Wheel, Window
F_1	Visible Filter
F_2	Star Sensing Filter
P	Depolarizing Window
LS	Lyot Stop

9608135

Detectors

The Sounder acquires radiometric data for 19 distinct wavelengths or channels through the use of four separate detector assemblies and a rotating filter wheel. This generates an 1120-µrad N/S swath that is moved latitudinally in 280-µrad (10-kilometer) steps. A fifth detector array provides the Sounder with star sense capabilities. Each of the radiometric channels is characterized by a central wavelength denoting primary spectral sensitivity. The 19 channels are broadly split into two classes: visible (channel 19) and infrared (channels 1-18).

IR, Visible, and Star Sense Detector Arrays



Visible Channel

The visible silicon detector array (channel 19) contains four detectors having an instantaneous geometric field of view (IGFOV) of 242 µrad in diameter set by the detectors, corresponding to an 8.7-kilometer (5.4-statute mile) diameter nominal pixel size at the spacecraft suborbital point. A star sensing array, consisting of a separate set of eight silicon detectors, is on the same mount and aligned to the center of the visible sounding detectors. It is identical to the Imager visible detector array but has 0.97-kilometer (0.60-statute mile) resolution and 8.5-kilometer (5.3-statute mile) array coverage.

Infrared Channels

The IR channels (1 through 18) are contained in three detector sets: LW, MW, and SW, each set consisting of four detectors. The fields of view are set by the field stops in a pattern the same as the visible channel.

Configuration

Each of the field stop or detector patterns is arranged in the same asymmetric fashion, with a nominal focal plane configuration. The star sensing array and visible radiometric array have a clear optical path in the instrument. The three arrays dedicated to IR wavelengths (LW, MW, and SW) are optically located behind the filter wheel assembly, each handling a different region of the infrared spectrum. Although physically separated in the instrument, the four radiometric arrays are coregistered optically, resulting in automatic coalignment of the pixels for all 19 channels.

Scan Control

As in the Imager, the Sounder scans the selected image area in alternate lines (that is, west-to-east followed by east-to-west or vice versa) and is capable of scanning both north-to-south and south-to-north. However, the OGE can only ingest north to south scans, which is an operational constraint. The Sounder's scanning mirror position is controlled by two servo motors, one for the N/S angle and one for the E/W scanning angle. The position of the scanning mirror, and hence the coordinate system employed for the instrument, is measured in terms of the inductosyn outputs. Scan control for both axes is generated by establishing a desired angular position for the mirror. The desired angle is input to an angular position sensor (one inductosyn for each axis), which produces a displacement error signal. This signal is fed to a direct drive torque motor (one for each axis) that moves the mirror and sensor to the null location.

For E/W deflection, the direct-drive torque motor is mounted to one side of the scan mirror and the position-sensing device (inductosyn position encoder) is mounted on the opposite side. All rotating parts are on a single shaft with a common set of bearings. Using components of intrinsically high resolution and reliability, coupling of the drive, motion, and sensing is therefore very tight and precise. North/South motion is provided by rotating the gimbal (holding the above components) about the optical axis of the telescope. This rotating shaft has the rotary parts of another torque motor and inductosyn mounted to it, again providing the tight control necessary.

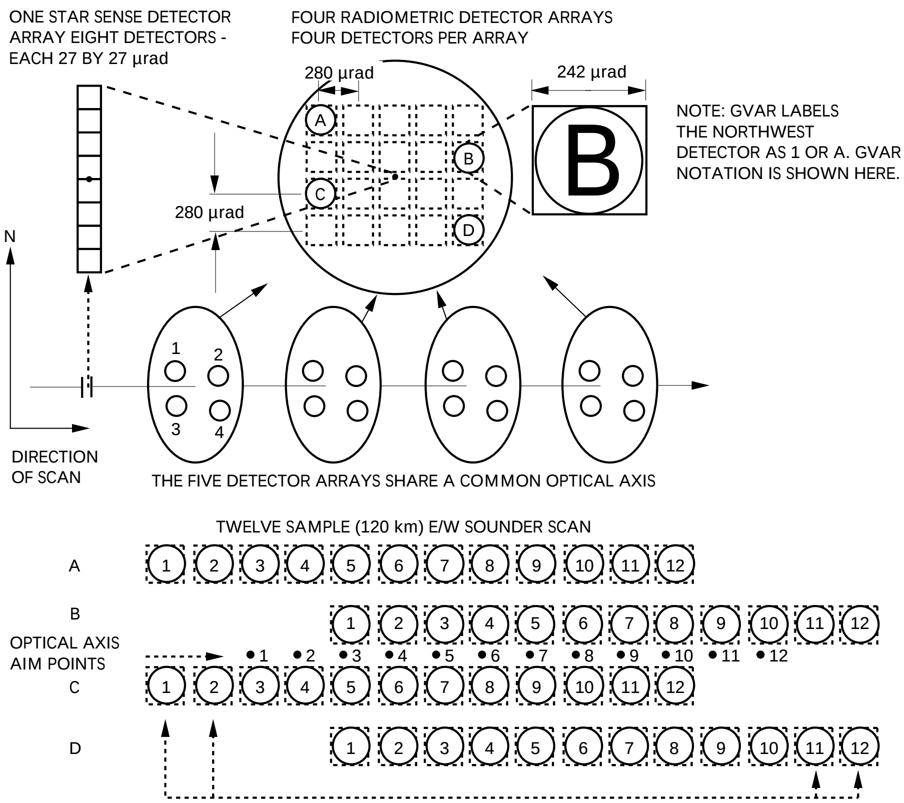
The servo system is not absolutely accurate because of noise, drag, bearing imperfections, misalignment, and imperfections in the inductosyn. Such inherent position-related errors cause pointing errors that preclude achieving the highest possible system accuracy. Slight variations of individual pole pairs cause a systematic pattern that is repeatable and measurable and can therefore be stored and subtracted to counteract the inductosyn's inherent error. This fixed error pattern and other systematic factors are measured, encoded, and stored in read-only memory. By injecting this stored error signal into the main control loop, the effect of inductosyn electromechanical errors and other systematic effects are reduced to less than one-fourth of their noncorrected values.

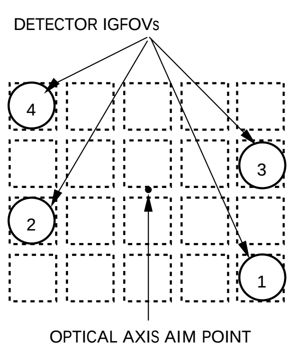
Drive and error sensing components used for the two drive axes are essentially identical. Control components are optimized for their frequency and control characteristics, and logic is developed for the precise control of position in response to a system-level control processor.

Scan Operation

Scan control is initiated by input commands that set start and end locations of a sounding frame. A location is identified by an inductosyn cycle and increment number within that cycle, the increment number determining the value of sine and cosine for that location. Each E/W increment corresponds to 17.5 µrad of E/W mechanical rotation or 35 µrad of E/W optical rotation. Each N/S increment corresponds to 17.5 µrad of N/S mechanical and optical rotation. The distance

Detector Separation and Scan Pattern





- Four Detectors Per Channel
- Each Detector Has 8.7 km (242.6 µrad) IGFOV (Maximum)
- Neighboring Detectors On 10 km (280 µrad) Centers In N/S Direction
- Neighboring Detectors On 10 km (280 µrad) Centers In E/W Direction
- Channels Aim Point For a Sample Is:
  - 20 km (560 µrad) East of 4 and 2 Detectors
  - 20 km (560 µrad) West of 3 and 1 Detectors
  - 15 km (420 µrad) South of 4 Detector
  - 5 km (140 µrad) South of 3 Detector
  - 5 km (140 µrad) North of 2 Detector
  - 15 km (420 µrad) North of 1 Detector
  - Detector Numbers As Identified In ITT Notation

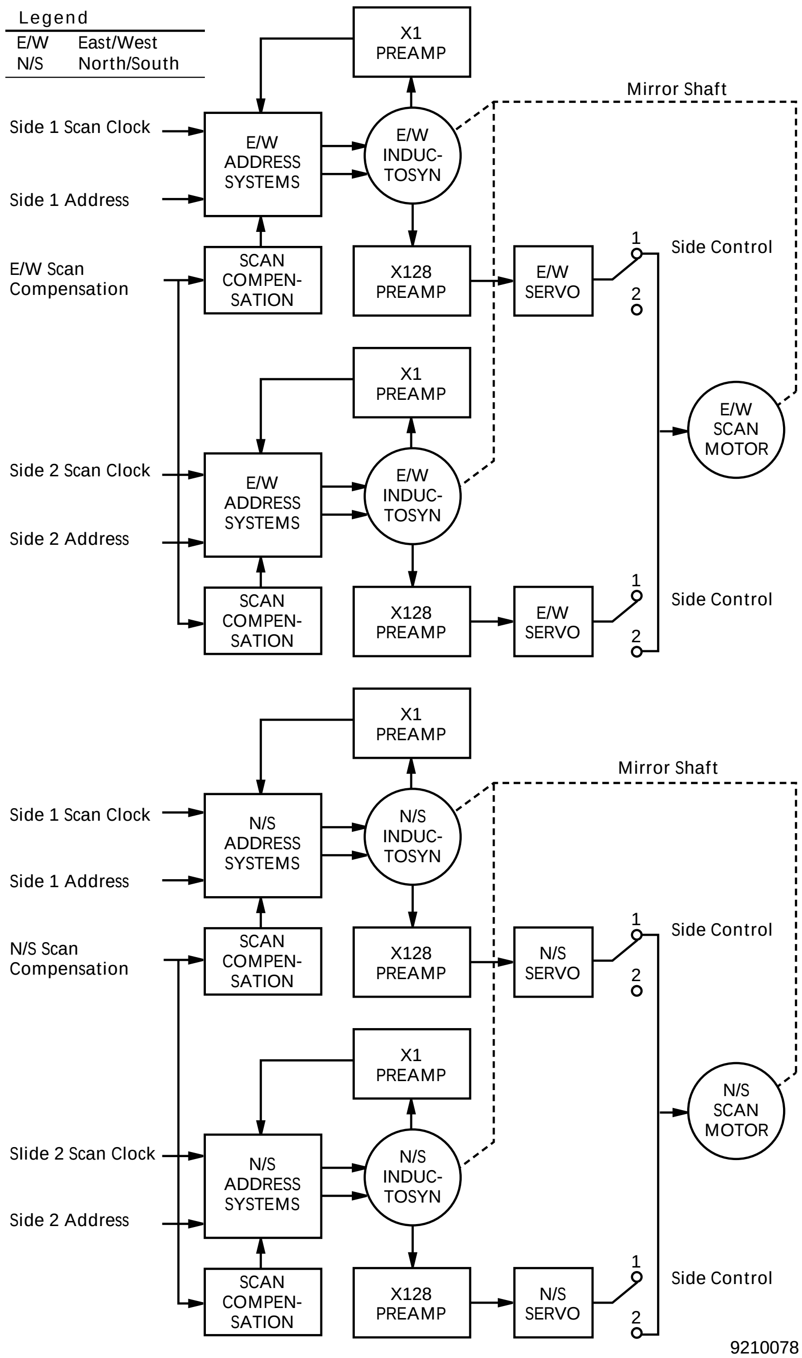
Legend

E/W	East/West
GVAR	Goes Variable Data Format
IGFOV	Instantaneous Geometric Field of View
N/S	North/South

NOTE: ITT LABELS THE NORTHWEST DETECTOR AS 4.
ITT NOTATION IS SHOWN HERE.

9401143

Scan Control Schematic



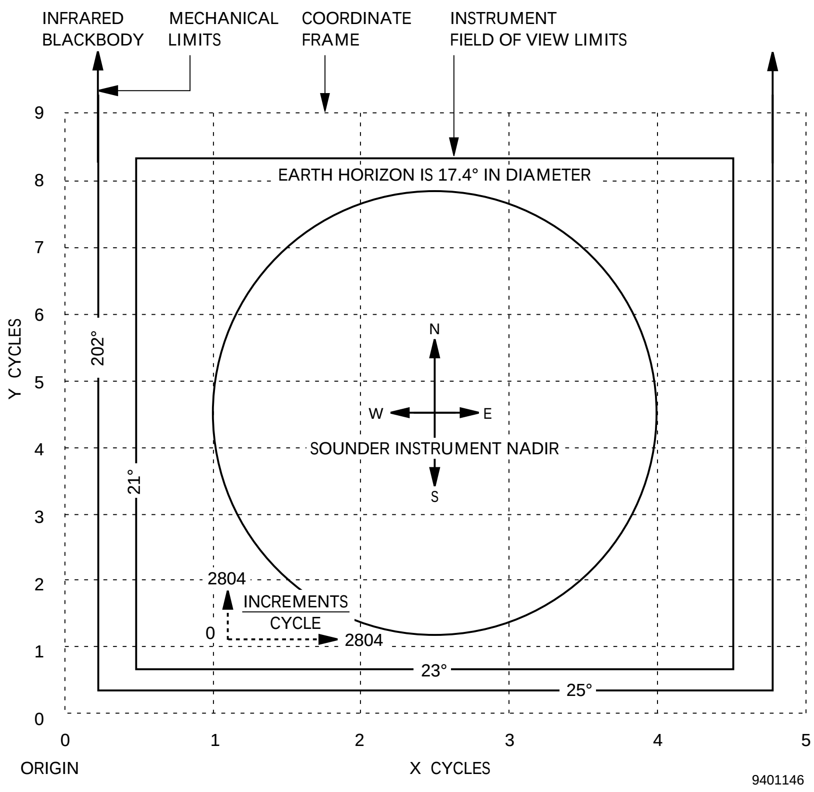
between a present and start location is recognized, causing incremental steps (17.5 µrad) at a high rate (10°/s) to reach that location. To minimize peak power demand the scan slews latitudinally, then longitudinally to a requested location.

Scan to space for space clamp or to star sensing, or to the IR blackbody uses the slew function. Command inputs (for star sensing or priority scan) or internal subprograms (for space clamp and IR calibration) take place at the proper time during a frame.

Sounding Generation

The E/W scan of the Sounder is acquired via a repeating sample-step-settle sequence that constitutes a 100 ms (single dwell), 200 ms (double dwell), or 400 ms (quadruple dwell) intervals. This is controlled by the filter wheel rotation. This step-settle sequence repeats until the end of the scan line is reached. At this point, a 100-ms interval is executed in which the mirror will be stepped 1120 µrad (40 kilometers at the spacecraft suborbital point) in the N/S direction, which is four times larger than the E/W scan step. At the conclusion of this interval, acquisition of the next scan line will be initiated in the opposite E/W direction

Sounder Coordinate Frame



using the sample-step-settle sequence. In the double (quadruple) dwell mode, two (four) detector samples are acquired at each step.

The mapping between cycles and increments and the instrument field of view are referenced to a coordinate frame whose origin is zero cycles and zero increments (southwest corner of the frame). In geostationary orbit, the earth will be centered within the frame, at instrument nadir, which corresponds closely to the spacecraft suborbital point, also centered in the frame. The GVAR coordinate system for both the Imager and Sounder is in line/pixel space and has its origin in the NW corner.

Electronics

The Sounder electronics module is similar to the Imager's, but with additional circuitry required for the filter wheel motor drive, synchronization, and channel registration. There is no coherent error integrator for the Sounder in the E/W direction, though an average error integrator (AEI) is active in the N/S and E/W directions to improve position accuracy. The AEI is a simple error correction circuit that acts upon the servo error signal to reduce that error to zero. The scan control electronics are contained in the electronics module. The servo preamplifiers are located at the scanner in the sensor module.

Signal Processing

Preamplification of the low-level IR and visible channel signals occurs within the sensor module. These analog signals are sent to the electronics module, which amplifies, filters, and converts the signals to digital code. All channels in the visible and IR bands are digitized to one part in 8192 (13 bits), the visible for high-quality visible sounding and to aid the star sensing capability, and the IR for radiometric measurement. Data from all channels move in continuous streams throughout the system; thus each channel's output must enter a short-term memory for proper formatting in the data stream. Each channel is composed of a detector, preamplifier, filter, postamplifier, analog-to-digital converter, and signal buffer. All signal chains are totally independent and isolated. Redundant chains of signal processing circuitry are provided with each circuit ending in a line driver designed to interface with the spacecraft sensor data transmitter.

Electronic Calibration

Electronic calibration signals are injected into the preamplifier of all channels while the Sounder is looking at space. Sixteen precise signal levels derived from a stepped digital-to-analog (D/A) converter are inserted during the 0.2-second spacelook. The electronic calibration signal is derived from a 10-bit converter having 0.5-bit accuracy, providing the accuracy and linearity for precise calibration. This is inserted into all preamplifiers of all channels, both visible and IR.

Visible Channels

The visible channel and star sensing detector arrays have a separate amplifier/processor for each detector element. These preamplifiers are current sensing types that convert the photon-generated current in the high impedance silicon detector into an output voltage, with a gain of about 10^8 V / A. The preamplifiers are followed by postamplifiers that contain electrical filtering and space clamping circuits. The digitization of the data signals is also part of the space clamp circuitry. The visible information is converted to 13-bit digital form, providing a range from near 0.1% to over 100% albedo for the visible channel. Differences of approximately 0.1% are discernible, and the linear digitization provides for system linearity errors of 0.5 bit in the conversion process.

Infrared Channels

The IR channels have a separate amplifier / processor for each detector element. Individual amplifiers, mounted on the cooled patch, are provided in the sensor module.

The IR information is converted to 13-bit digital form, providing a range from near 0.1% to over 100% of the response range. Each channel has a gain established for space-to-scene temperatures of 320 to 340 K. The 13-bit digital form allows the lowest calculated noise level to be differentiated. The digital system is inherently linear with analog-to-digital (A / D) converter linearity and accuracy to 0.5 bit. The binary coded video is strobed onto the common data bus for data formatting by the system timing and control circuitry.

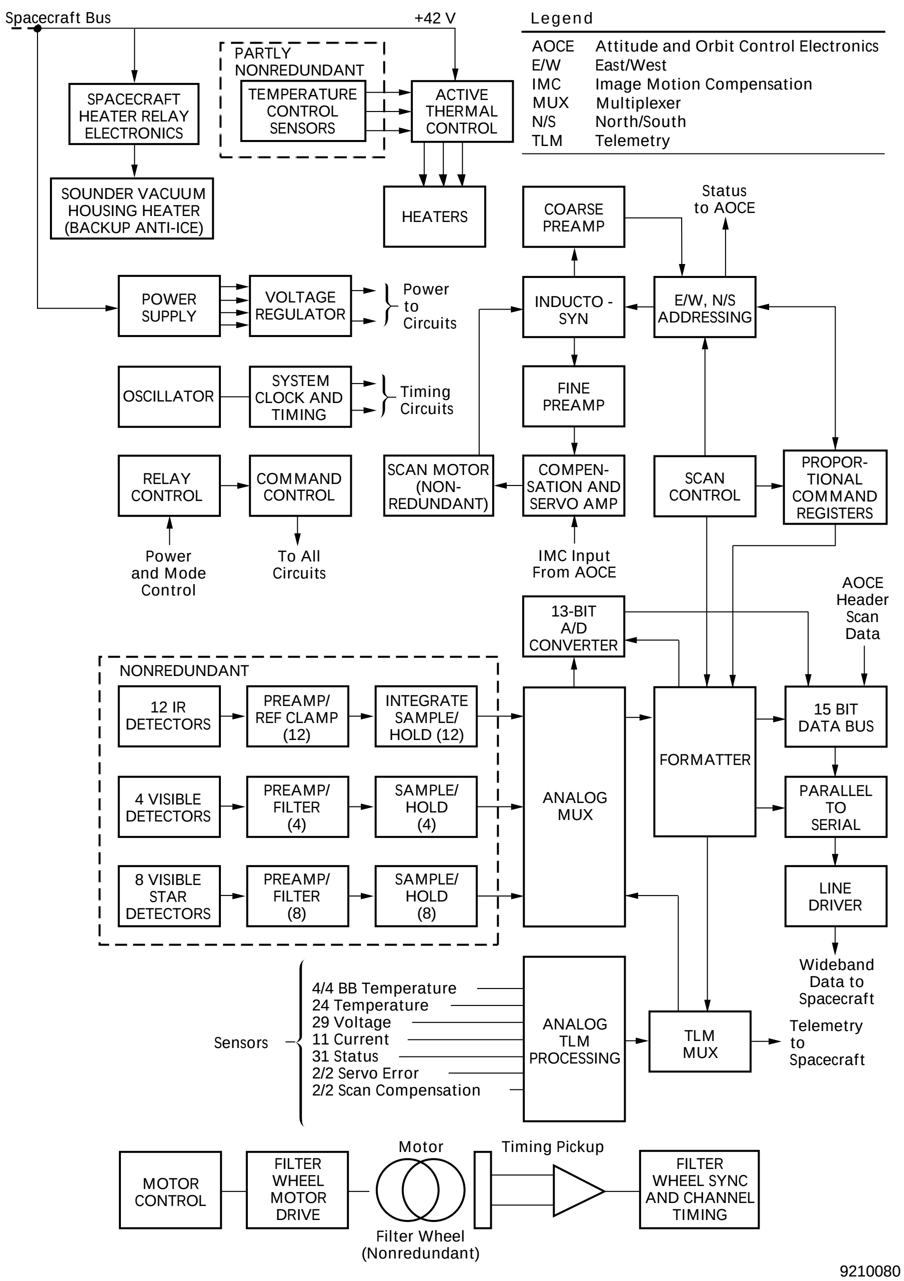
Formatting

Digital signal processing starts where data streams from the IR and visible detectors and telemetry merge via multiplexing (a parallel-to-serial conversion and data multiplexing take place to bring sensor data together). Other information, such as synchronization pulses, scan location, and telemetry data, are assembled in the data select circuitry. The data are then passed through a line driver where pulse amplitude and impedance levels are set for the transmitter interface.

A Sounder data block is transmitted during the time it takes for the filter wheel to complete one revolution (0.1 second). Unlike the Imager, there is no concept of multiple data block types that are formatted differently as a function of their data content. All of the data is contained in one Sounder data block stream, where each Sounder data block contains 250 16-bit words transmitted at a data rate of 40 kb / s or 10 blocks per second. A Sounder data block contains:

- Sounding data
- Star sense data
- Telemetry
- Header data
- Synchronization
- Attitude and orbit control electronics (AOCE) data
- Scan position
- Scan control data

Sounder Block Diagram



Power Supply

The Sounder power supply is very similar to that of the Imager. The Sounder power supply provides the additional power, control and regulation required by the filter wheel.

Space Environment Monitor Subsystem

The Space Environment Monitor (SEM) measures *in situ* the effect of the sun on the near-earth solar-terrestrial electromagnetic environment, providing real-time data to the Space Environment Services Center (SESC). The SESC, as the nation's "space weather" service, receives, monitors, and interprets a wide variety of solar-terrestrial data, and issues reports, alerts and forecasts for special events such as solar flares or geomagnetic storms. This information is important to the operation of military and civilian radio wave and satellite communication and navigation systems, as well as electric power networks, and to the mission of geophysical explorers, Shuttle and Space Station astronauts, high-altitude aviators, and scientific researchers.

The SEM subsystem consists of four instruments used for *in situ* measurements and monitoring of the near-earth (geostationary altitude) space environment and for observing the solar X-ray output. An energetic particles sensor (EPS) and high energy proton and alpha detector (HEPAD) monitor the incident flux density of protons, alpha particles, and electrons over an extensive range of energy levels. Solar output is monitored by an X-ray sensor (XRS) mounted on an X-ray positioning platform, fixed on the solar array yoke. Two redundant three-axis magnetometers, mounted on a deployed 3-meter boom, operate one at a time to monitor earth's geomagnetic field strength in the vicinity of the spacecraft. The SEM instruments are capable of ground command-selectable, in-flight calibration for monitoring on-orbit performance and ensuring proper operation.

Energetic Particles Sensor

The EPS performs three integral measurements (at geostationary orbit) of electrons from 0.6 to more than 4.0 megaelectronvolt (MeV), a seven-channel differential analysis of protons from 0.8 to 500 MeV, and a six-channel differential analysis of alpha particles from 4 to 500 MeV per nucleon. The EPS also provides all the support required by the HEPAD, which extends the EPS energy ranges to greater than 700 MeV for protons and to greater than 3400 MeV per nucleon for alphas. The EPS and HEPAD are housed within the spacecraft main body and view the space environment through apertures.

The EPS unit consists of a telescope subassembly, a dome subassembly and signal analyzer unit / data processing unit (SAU / DPU); the latter unit provides the final amplification of the telescope and dome output signals. These components are housed on a separate panel, mounted on the spacecraft's south equipment panel, providing a clear field of view towards the west.

The telescope uses two silicon surface barrier detectors that output charge pulses to charge sensitive preamplifiers within the telescope, converting them into voltage pulses; this preconditions the signals sent to the SAU/DPU. These detectors sense low energy protons in the range of 0.8 to 15 MeV and alpha particles in the range of 4 to 60 MeV. The two detectors, surrounded by tungsten shielding, are arranged in a telescope configuration: a 50-µm, 100-mm^2 front detector and a 500-µm, 200-mm^2 rear detector. Tungsten collimators define the field of view of 70° and eliminate detector edge effects. Sweeping magnets exclude electrons below about 100 kiloelectronvolts (keV), while a 0.145-mil aluminum foil excludes light. The outer surface of the front solid-state detector is covered with 130 µg/cm^2 of aluminum, rendering it light tight.

The dome employs three sets of two 1500-µm, 25-mm^2, silicon surface barrier detectors, each with different thickness moderators covering the respective pairs' independent fields of view, thus providing three different energy thresholds. As in the telescope, the solid state detector output charge pulses are passed through charge sensitive preamplifiers, converting them into voltage pulses before being routed to the SAU/DPU. After processing, the output of the detector pairs

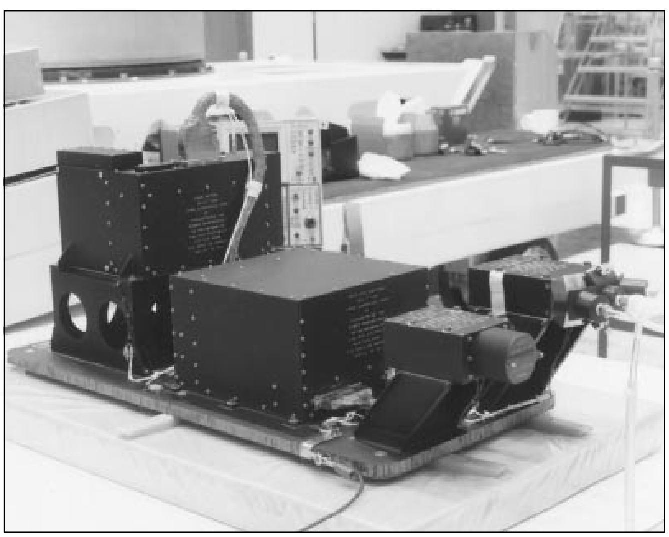
Energy Ranges for the Energetic Particles Sensor and High Energy Proton and Alpha Particle Detector

Particle Type	Channel Designation	Nominal Energy Range (MeV)	Detector Assembly
Proton	P1	≤ 0.8 to 4	Telescope
Proton	P2	4 to 9	Telescope
Proton	P3	9 to 15	Telescope
Proton	P4	15 to 40	Dome
Proton	P5	40 to 80	Dome
Proton	P6	80 to 165	Dome
Proton	P7	165 to 500	Dome
Proton	P8	350 to 420	HEPAD
Proton	P9	420 to 510	HEPAD
Proton	P10	510 to 700	HEPAD
Proton	P11	> 700	HEPAD
Alpha	A1	4 to 10	Telescope
Alpha	A2	10 to 21	Telescope
Alpha	A3	21 to 60	Telescope
Alpha	A4	60 to 150	Dome
Alpha	A5	150 to 250	Dome
Alpha	A6	300 to 500	Dome
Alpha	A7	2560 to 3400	HEPAD
Alpha	A8	>3400	HEPAD
Electron	E1	≥ 0.6	Dome
Electron	E2	≥ 2.0	Dome
Electron	E3	≥ 4.0	Dome
"Singles"	S1 to S5	—	HEPAD

Energetic Particles Sensor Performance Summary

EPS Parameter	Performance
Proton bands (P1 to P7)	7 bands 0.8 to 500 MeV, logarithmically spaced
Alpha particle bands	6 bands 3.2 to 500 MeV, logarithmically spaced
Electron bands (E1 to E3)	Integral bands with thresholds of E1: 0.55 MeV E2: 2.0 MeV E3: 4.0 MeV
Dynamic range	Cosmic ray background to largest solar particle event
Accumulation efficiency P1, E1, E2, and E3 bands P2 to P7 bands	 25% 50%
Sampling rate	Once every 10.2 or 20.5 s
Band edge stability	≤ ± 3%
Resolution	No worse than pseudolog compression of 19 to 8 bits, using 4 bits of mantissa and 4 bits of exponents
Geometric factor Telescope Dome	 > 0.06 cm^2 steradian > 0.25 cm^2 steradian
Field of view Telescope Dome	 1.1 steradian 2.0 steradian

EPS/HEPAD



provides data for four proton, three alpha, and three electron energy bands, ranging from 15 to 500 MeV for protons, 60 to 500 MeV for alphas, and less than 0.6 to more than 4.0 MeV for electrons.

High Energy Proton and Alpha Detector

The HEPAD senses incident flux of high energy protons (350 to greater than 700 MeV) and alphas (640 to greater than 850 MeV / nucleon). The unit consists of a telescope subassembly with two silicon surface barrier detectors, a Cerenkov radiator, and a photomultiplier tube (PMT), all arranged in a telescope configuration, and a signal analyzer subassembly. The Cerenkov radiator and PMT provide directional (front / rear incidence) discrimination and energy selection. The solid-state detectors differentiate between minimum ionizing protons and alpha particles and are shielded from protons below 70 MeV and electrons below 15 MeV by aluminum and tungsten barriers.

High Energy Proton and Alpha Detector Performance Summary

HEPAD Parameter	Performance
Spectral bands	
Proton	3 bands from 350 to >700 MeV
Alpha particle	2 bands from 2560 to 3400 MeV
Field of view	Conical, ~34° half angle
Geometric factor	0.9 cm^2-sr
Dynamic range	0 to 10^4 counts / s
Accumulation efficiency	100%
Stability and accuracy	≤ ± 15%
Data rate	
Primary data channel	Once every 10.2 s
Single channels	Once every 41 s
Count resolution	No worse than pseudolog compression of 19 to 8 bits, using 4 bits of mantissa and 4 bits of exponent
Contaminants	
Proton contamination in alpha channels	≤ 0.1%
Characterize response to penetrating electron in 2-13 MeV range	As specified
Lifetime	Ground commands to compensate for performance degradation during 5-year lifetime

The signal analyzer subassembly contains all the amplification and processing electronics required to sort the accepted telescope events into particular particle types and energy levels and to transmit data to the EPS SAU/DPU on 11 data lines (4 proton, 2 alpha, and 5 single channels). The five single channels are required to correct proton and alpha channel data. PMT operating voltage is changeable upon ground command to compensate for aging effects; turn-on detection circuitry ensures that the PMT voltage always initializes at the lowest value. In-flight calibration capability is also included.

The SAU/DPU multiplexes all of the EPS/HEPAD particle data (27 channels) and accumulates the results in compression counters. Outputs of these compression counters are transferred serially to spacecraft telemetry in an 80-channel submultiplexed data stream. The SAU/DPU also processes timing signals from the spacecraft telemetry unit, housekeeping telemetry, in-flight calibration, and control commands to the EPS/HEPAD subsystem. Upon ground command, the SAU/DPU in-flight calibration circuitry generates a sequence of amplitude modulated test pulses that are applied to the charge-sensitive preamplifier inputs to determine the stability of amplification chains and threshold discriminators.

X-Ray Sensor

The XRS is an X-ray telescope that measures in real-time solar X-ray flux in the spectral range of 0.5 to 3 angstroms (short sun channel) and 1 to 8 angstroms (long sun channel). The XRS assembly consists of a telescope collimator and sweeper magnet subassembly, dual ion chamber and preamplifier subassemblies, a DPU subassembly, and a bucking magnet subassembly.

X-rays are detected by two ion chambers, one for each spectral range. The detector output signals are processed by separate electronic channels that provide automatic range selection. Nominal flux levels expected are on the order of 2×10^{-8} to 2×10^{-3} W/m^2 for the long channel and 5×10^{-9} to 5×10^{-4} W/m^2 for the short channel. The capability is provided to calibrate each channel via ground command. Data transmitted through the spacecraft telemetry permit real-time ground determination of the solar X-ray emission in the two spectral bands.

The aperture of the XRS is provided with a pair of sweeper magnets to deflect incoming electrons away from the ion chambers so that only X-rays are admitted. An external bucking magnet is mounted to the XRS to minimize the magnetic signature induced at the spacecraft magnetometers.

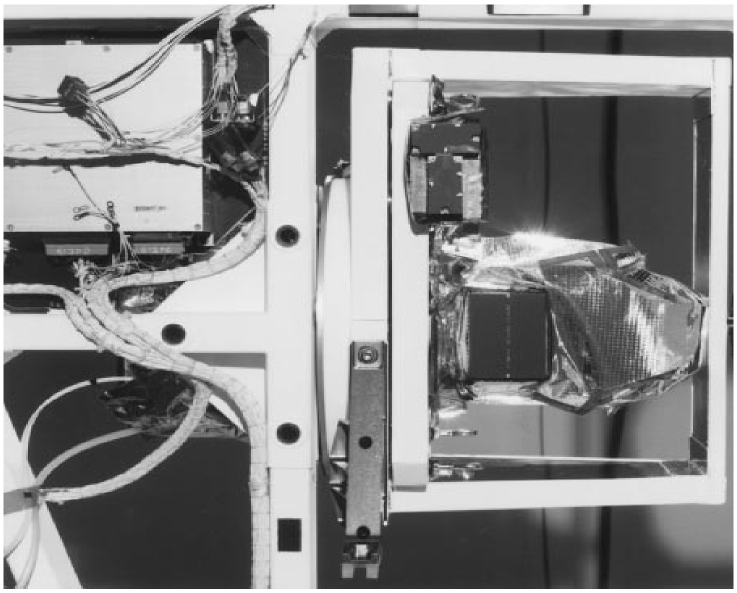
The XRS as a single unit is housed on the XRS drive assembly; this combination is mounted on the solar array yoke on the south side of the spacecraft body in a position that provides the XRS and drive assembly sun sensors a clear field of view to the sun at all times. The drive assembly moves the XRS along the north/south axis to track the sun's declination of ±23.5˚. The east/west motion of the

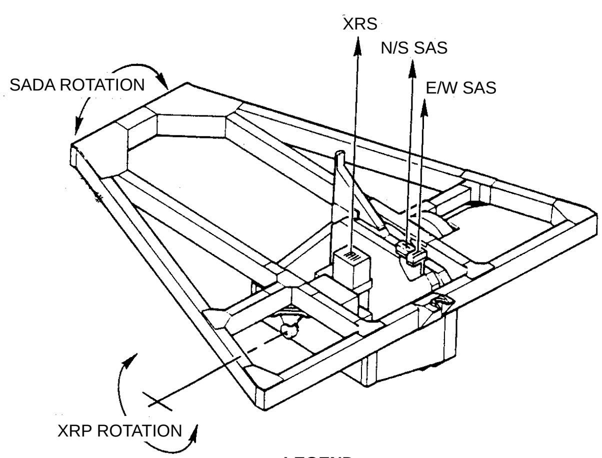
XRS is provided by the solar array drive assembly. During north / south stationkeeping maneuvers, the XRS is stowed; in this position the telescope is in-plane with and pointing to the edge of the solar array.

X-Ray Sensor Performance Summary

XRS Parameter	Performance
Spectral bands	
Channel A (short)	0.5 to 3.0 Å
Channel B (long)	1.0 to 8.0 Å
Threshold sensitivity	
Channel A	5×10^{-9} W / m^2
Channel B	2×10^{-8} W / m^2
Dynamic range	
Channel A	5×10^{-9} to 5×10^{-4}
Channel B	2×10^{-8} to 5×10^{-3}
Resolution	
Fluxes >20 times threshold	≤ 2% of reading
Sampling rate	Once every 0.512 s
Response time	
Time for output to reach 90% of final value after step change	2 s
Position determination	
Resolution	0.25°
Accuracy	0.50°

XRS Mounting on Yoke Assembly





LEGEND

E/W	East/West
N/S	North/South
SADA	Solar Array Drive Assembly
SAS	Sun Analog Sensor
XRP	X-Ray Positioner
XRS	X-Ray Sensor

9608143

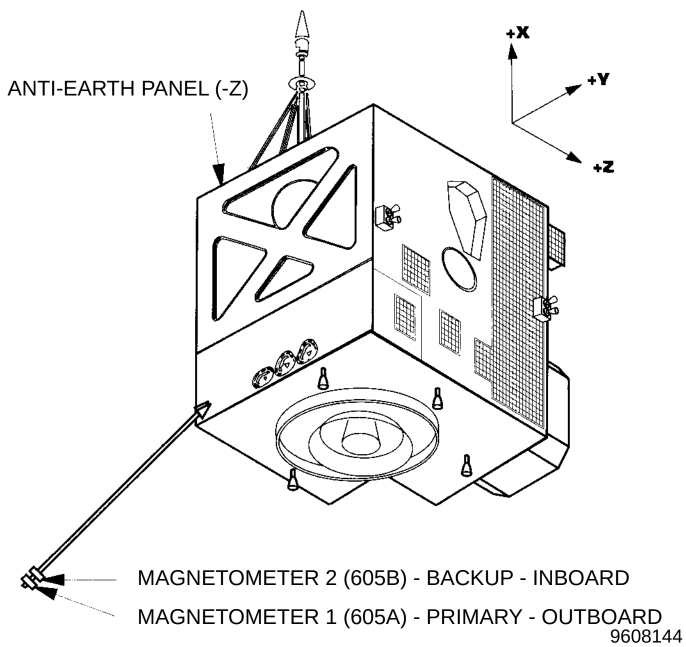
Magnetometers

The redundant magnetometers use two sensor heads, each containing three orthogonal flux-gate magnetometer elements, to measure three orthogonal vector components of the dc magnetic field applied to three flux-gate sensor elements. The applied field consists of the naturally occurring earth's field and the interfering field from the spacecraft components (the latter could be greater than the former). The determination of the ambient magnetic field in the vicinity of the spacecraft is continuous and simultaneous. The flux gates are located in a sensor assembly, attached to the end of a boom that places the sensor 3 meters away from the spacecraft body. Each flux gate magnetometer is aligned to within 0.5° of the spacecraft X, Y, and Z axes and has a linear range of ±1000 nanoTesla (nT).

The excitation and feedback signals from the sensors are routed to a dual magnetometer electronics unit located within the spacecraft main body where the signals are processed and formatted for spacecraft telemetry. An analog signal processor demodulates the flux-gate signals to produce an analog voltage proportional to the field magnitude with a polarity related to the direction of the field vector component being measured. Three analog signals representing the X, Y, and Z components of the surrounding magnetic field are digitized by a 16-bit analog-to-digital converter, producing as output a serial bit stream in which three groups of 16 bits are allocated to the polarity and magnitude of each of the three axes (a total of 48 bits).

The two three-axis magnetometers provide redundancy for measuring the geomagnetic field. One magnetometer is mounted on the boom 3 meters (9.8 feet) away (outboard) from the spacecraft, and the second, 30 centimeters (12 inches) inboard from the first on the same boom. Both magnetometers share the same telemetry channel, though only one magnetometer, with its associated three flux-gate sensors, can be powered at any time.

Magnetometers' Location



Magnetometer Performance Summary

Magnetometer Parameter	Performance
Null setting error	±2 nT
Dynamic range	±1000 nT, ambient field in any orientation
Sensitivity	10 mV/nT
Linearity	0.02% of full scale
Resolution	0.03 nT
Accuracy	< ±4 nT without temperature correction < ±1 nT with temperature correction
Noise	≤0.3 nT or 0.5% of reading whichever is greater
Data rate	1.95 Hz
Bandwidth	Dc to 0.5 ±0.032 Hz, presampling filter at –3 dB
Sensor axes orthogonality	Within ±0.5°
Sensor orientation	≤ ±1.0°, in spacecraft coordinates
Spacecraft field contamination Maximum permanent field Residual contamination after ground correction	 <200 nT < ±0.5 nT

Solar X-ray Imager

The Solar X-ray Imager (SXI) is used to determine when to issue forecasts and alerts of “space weather” conditions that may interfere with ground and space systems. These conditions include ionospheric changes that affect radio communication (both ground-to-ground and satellite-to-ground) and magnetospheric variations that induce currents in electric power grids and long distance pipelines and can cause navigational errors in magnetic guidance systems, introduce changes in spacecraft charging, produce high energy particles that can cause single event upsets in satellite circuitry, and expose astronauts to increased radiation. SXI will observe solar flares, solar active regions, coronal holes, and coronal mass ejections. Images from SXI will be used by NOAA and U.S. Air Force forecasters to monitor solar conditions that affect space weather conditions that are used to describe the dynamic environment of energetic particles, solar wind streams, and coronal mass ejections emanating from the Sun.

The SXI, performing as a part of the Space Environment Monitor (SEM) instruments, provides the means for obtaining the solar data required to:

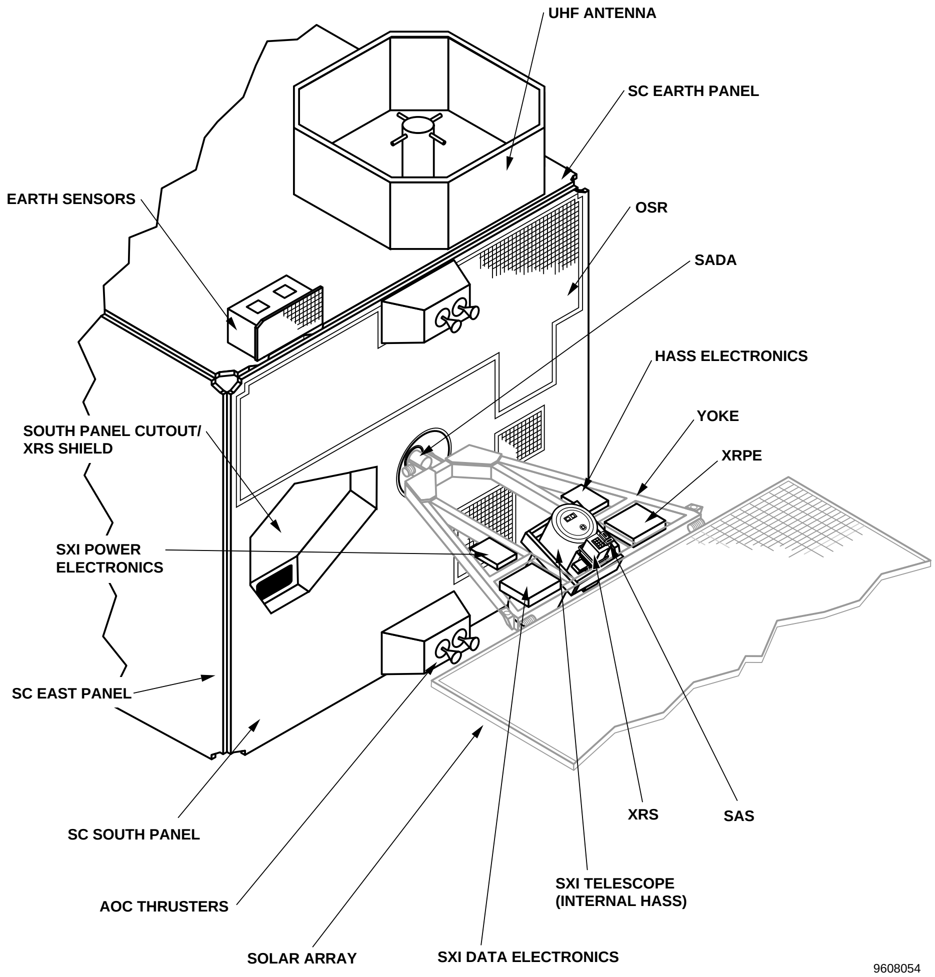
- Locate coronal holes for predicting high speed solar wind streams causing recurrent geomagnetic storms, and also locate transient coronal holes as a source of ejecta.
- Locate flares on the disk and beyond the west limb for proton event warnings.
- Monitor for changes indicating coronal mass ejections (CME) that may impact Earth and cause geomagnetic storms. Large-scale, long duration, possible weakly emitting events, and brightening of coronal filament arcades are used as evidence of CMEs.
- Observe active region size morphology and complexity, and temperature and emissions measure, for flare forecasts. Monitor for active regions beyond east limb that will be rotating onto the solar disk.

Other solar feature observations include flare properties, newly emerging active regions, X-ray bright points, and following CME ejecta at 1000 km / sec. To meet these objectives, the SXI images the solar corona in the soft X-ray to extreme ultraviolet (XUV) region of the electromagnetic spectrum. Full-disk solar images are provided with a 512 X 512 array with 5 arcsec pixels in several wavelength bands from 6 to 60 Å (0.6 to 6 nm). A regular sequence of exposures that are downlinked at one-minute intervals is used to cover the full dynamic range needed to monitor solar activity. The SXI telescope is mounted on the X-ray Positioner (XRP), and its associated electronics boxes are on the solar array yoke of the GOES-M spacecraft. The SXI is Government-furnished equipment.

THE SUBSYSTEM

The SXI consists of a telescope assembly, and three electronic boxes: the data electronics box for the instrument command and data management system (C&DMS), the power electronics box, and the High Accuracy Sun Sensor (HASS) electronics box. The electronics boxes are mounted on the solar array yoke.

SOLAR X-RAY IMAGER DESIGN



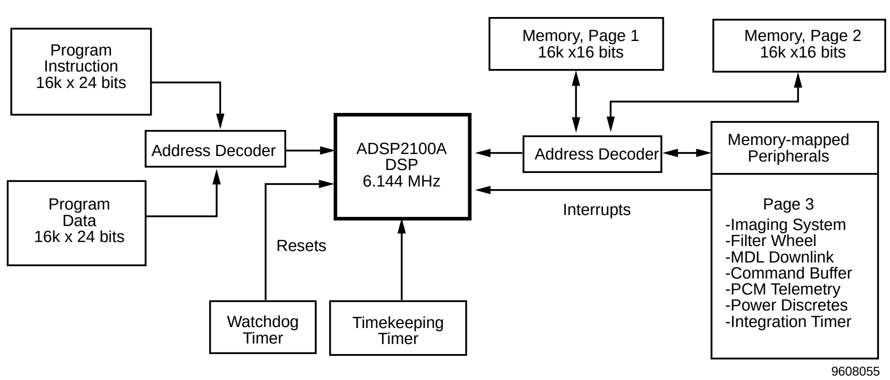
The SXI telescope is mounted to the X-ray positioner (XRP), which is a single-axis gimbal, aligned in the north-south direction. The SXI is coaligned with the X-ray Sensor (XRS) and Sun Analog Sensor (SAS), which are also mounted on the XRP. The XRP is attached to the solar array yoke. The solar array drive assembly (SADA) controls the east-west pointing of the yoke, adjusting the yoke position at a constant rate. The XRP N-S pointing is controlled during routine observations by a closed-loop control system. The solar image will drift within the field of view because the XRP and SADA do not remove all of the orbital effects and spacecraft attitude errors that affect solar pointing. Pointing adjustments to the XRP and SADA are possible through ground command.

The total mass of the SXI (telescope and electronics) is 22.7 kg, of which 14.8 kg is the telescope assembly. Electrical connections to the GOES spacecraft cross the SADA interface through slip rings and nine lines are assigned to the SXI. The SXI image and housekeeping data are interfaced to a dedicated multiuse data link (MDL) in the GOES spacecraft, capable of handling a data rate of 100 kbps, allowing a single image to be transmitted to the ground receiving station in about 36 sec. A limited amount of SXI health and safety data is provided in the pulse code modulated (PCM) data stream.

OPERATION

Operation of the SXI is controlled through the Data Electronics Box (DEB) via a microprocessor operating at 2.0 MHz. The microprocessor receives and interprets uplinked commands, controls image sequencing, processes image data, controls interface peripherals, downlinks image and housekeeping data, and keeps the internal time to a resolution of better than one ms.

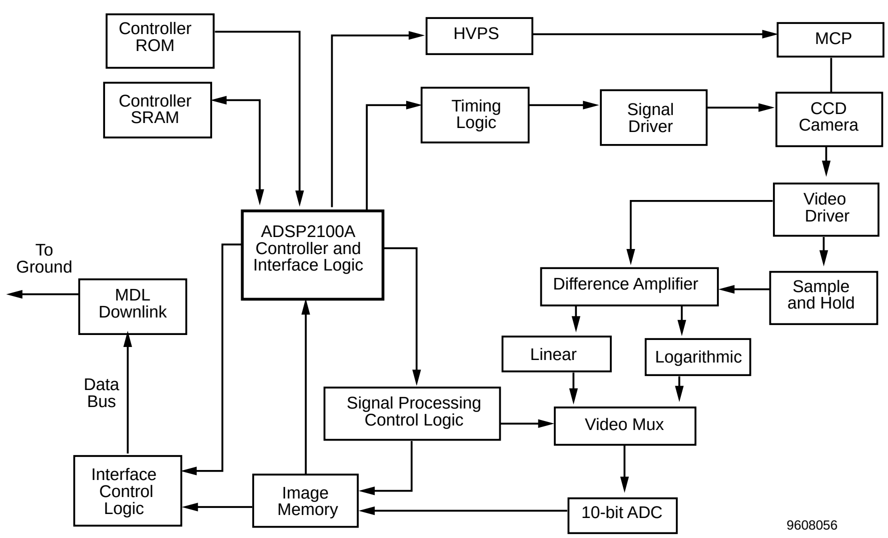
Functional Flow Diagram of SXI Microprocessor



A watchdog timer provides closed-loop recovery from single event upset (SEU)-induced errors or other anomalous conditions that may cause the instrument to enter an undesirable state. Flight software periodically strobes the watchdog timer, resetting the count, as a method of indicating continued health and functionality.

The microprocessor controls the acquisition of image data, as well as processing data and transmitting it to the ground. Based on tables stored in read only memory (ROM) or uplinked into Random Access Memory (RAM), the processor generates commands to the high voltage power supply (HVPS) and the charge coupled device (CCD) timing logic to begin an image-taking sequence. At the start of imaging, a clock in the camera is reset. The clock automatically advances the CCD one column if the exposure exceeds 333 ms, or multiples thereof, to compensate for orbital motion of the spacecraft. Once the exposure is complete, the CCD is read out at a 500 kHz pixel rate and transferred to either of two channels which provide a linear or a logarithmic transfer function. The latter improves sampling at the low end of the dynamic range. Either channel output can be selected and then passed to the input of the 10-bit analog/digital converter (ADC) in the Data Electronics Box.

SXI Control System Logical Flow Diagram



Solar X-ray Imager Performance Summary

SXI Parameter		Performance
Imaging exposure times		
Solar flare sites		<10 ms
Active regions		<100 ms
Coronal loops		<1 s
Coronal hole boundaries		<10 s
Spacecraft SXI boresight pointing (to center of solar disk)		Within 3 arc-min elevation, within 3.5 arc-min azimuth
Field of view		42 by 42 arc-min, minimum
Pixel size		5 by 5 arc-sec, squared pixels, maximum
Spectral sensitivity (integration time ≤100 ms)		
Spectral band	Source	Minimum detectable photon radiance incident on the telescope entrance (photon cm^{-2} $arcsec^{-2}$ sec^{-1})
6 to 20 Å	Al (8.3 Å)	7
6 to 60 Å	C (44.7 Å)	132
Dynamic range		1000 when measured with mono-chromatic illumination at 44.7 Å
Telemetry amplitude digitization		10 bits (linear or logarithmic channels)
Point response (image on pixel array) Percentage of total energy incident on detector falling inside		
1 by 1 pixel		25%
2 by 2 pixel		40%
HASS resolution		±5 arc-sec or better
SXI on-orbit useful life		3 years with a goal of 5 years (after 5 years ground storage)

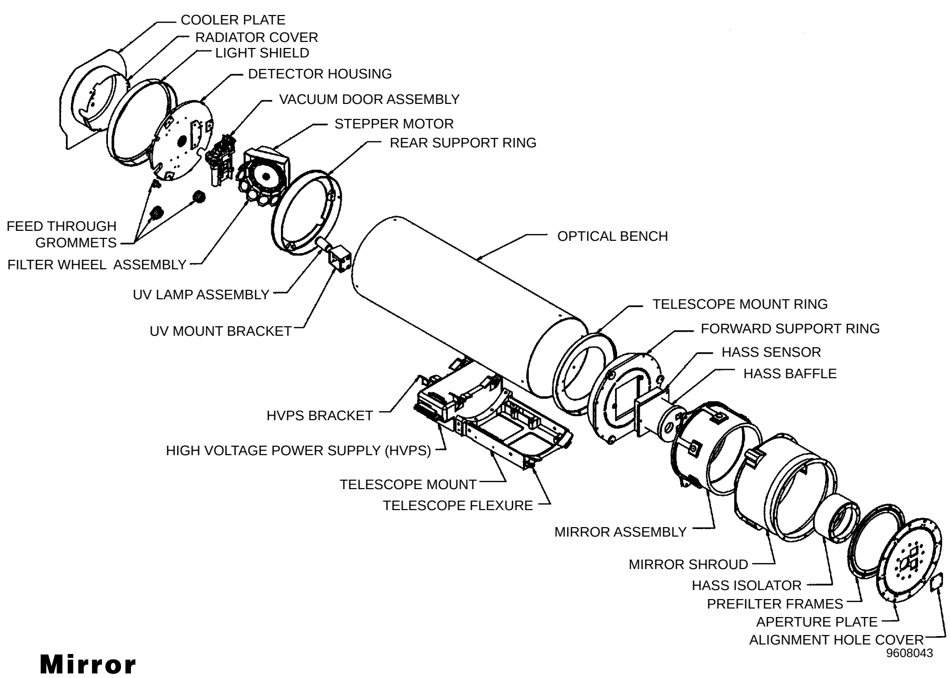
At this point, the microprocessor can select one of two modes: pixel conversion or pixel readback. In pixel conversion mode, the pixels from the CCD camera are read, passed through the ADC, and stored in image memory (static RAM). This memory has the capacity to store a complete 512 by 512 image. In readback mode, the microprocessor reads 16 bits at a time from the X-ray Pixel Processor (XPP). The lower 10 bits of each word are pixel intensity and the next two bits are parity.

One of the two parity bits is generated over the 10-bit pixel value at the time of digitization and is stored in the RAMs. The other is not and is generated at the time the data are read from the RAMs, and is generated over the 11-bit field which includes both the 10-bit pixel value and the 1-bit parity. This scheme allows any parity error to be identified as originating from a single event upset (SEU) in the memories, or an RF bit-flip during downlink. Housekeeping and ancillary data have no parity checks but are repeated several times per frame.

The data are transmitted to the ground through the MDL interface with a 100 kb/s capacity using a split phase data coding. The data are downlinked as transfer frames that are in close compliance with the Consultative Committee for Space Data Systems (CCSDS) format. These data are received directly at the Space Environment Center (SEC) in Boulder, CO. In general, image data are downlinked as rapidly as possible after they are acquired.

TELESCOPE ASSEMBLY

The basic telescope design consists of a Wolter I design grazing incidence x-ray mirror, a twelve position broadband filter wheel, and a focal plane subassembly containing an intensified MCP/CCD camera with 5 arcsec pixels.



Mirror

The mirror design consists of a Wolter Type I grazing incidence mirror in a parabola-hyperbola configuration. Both optical surfaces are fabricated from a single Zerodur element. The mirror is supported by six equally spaced titanium fingers bonded to super invar pads which are in turn bonded to the mirror. The fingers mate to a mounting ring attached to the optical bench.

Nominal Mirror Parameters

Optical design	Paraboloid-hyperboloid(Wolter I)
Radius at principal plane	8.0 cm
Axial mirror element	4.75 cm
Separation of principal and focal planes	65.0 cm
Microroughness	10Å rms (goal 5 Å rms)
Mirror material	Zerodur
Surface coating	Nickel
Geometrical area	7.3 cm^2

Optical Bench

The optical bench is both a metering structure for the optical system and a structural support member for the telescope assembly. As a metering structure, the optical bench maintains the separation between the mirror and the focal plane subassembly to ±0.001 cm over an expected temperature range of ±40˚C. The bench is hand laid up using sheets of carbon fibers impregnated with a cyanate ester resin. The bench material is highly hydrophobic and thermally stable, varying from 2.5% to less than 0.2% by weight, thereby avoiding water vapor outgassing and possible condensation on the detector array.

Focal Plane

The focal plane design incorporates three components: a microchannel plate (MCP) as the detecting element, and a phosphor-coated fiber optic taper (FOT), and a CCD. The MCP is a high output technology device of greatly improved dynamic range. The plate has 8 micron pores on 10 micron centers and is capable of a resolution of 42 line pairs/mm.

The electron avalanche from the MCP strikes the phosphor deposited on the face of the FOT. The phosphor is yttrium oxysulphide which matches the peak sensitivity of the CCD. The optical coupler provides a magnification of 1.2 which matches the plate scale of the focal plane to the CCD; i.e., the fiber pitch increases from 5 to 6 microns, changing the plate scale from 15.8 to 19 microns so that one CCD pixel corresponds to 5 arcsec. At 42 line pairs/mm, the modulation transfer function (MTF) of the taper is about 80%.

The CCD, a 512 x 512 device, has full well depths of 450,000 e^-. The practical dynamic range is of order 1000. This is achieved by adjusting the gain of the MCP so that the average signal per photon is 150 e^- (dynamic range of 300 photons). The average signal per photon of 150 e^- is well above the CCD's thermal noise level.

This configuration allows the detector to be electronically shuttered by controlling the accelerating voltage across the MCP. This voltage can be as high as 1200V and must be turned on and off with rise times on the order of 100 µs in order to accommodate exposure times as short as one ms. The voltages are generated using voltage doublers whose capacitance is much larger than the external load. The capacitance of the voltage doublers determines the rise time.

A shutdown circuit is connected across the output load and is controlled by a pulse width modulator. The signal which inhibits the modulator also activates the shutdown circuit, effectively shorting the output.

The detector assembly is contained within an evacuated titanium housing, allowing the MCP to be operated and tested on the ground. The electrical penetrations are through glass-to-metal seals, and a vacuum port is provided that can be closed with a valve. The base of the chamber is formed by a metal disk. The optical taper is bonded to a ring which mates to the detector housing with an O-ring between them.

In this configuration, the CCD and its front-end electronics lie outside the chamber, providing easier access. The forward end of the vacuum chamber contains a door that can be opened automatically but must be closed manually. The operation of the door is controlled by a paraffin actuator. The actuator is a cylinder filled with paraffin. When heated electrically, the paraffin changes state and provides the driving force for a piston. The piston lifts the door away from its seal and releases a locking pin, allowing a torsion spring to open the door.

The door contains a sapphire window which allows the MCP to be illuminated with ultraviolet (UV) light when the door is closed. The MCP is sensitive to the UV light which can be used to demonstrate functional operation during ground testing and storage. A small UV lamp is incorporated into the SXI instrument to enable functional testing and limited flat field tests to be performed before launch and in flight.

Pre-filters

Pre-filters are used to block solar ultraviolet, visible, and infrared radiation from the interior of the telescope. The pre-filters are composed of a sandwich of titanium, polyimide, and aluminum. The outermost layer is of aluminum about 1000 Å thick, which is the primary rejection element. Aluminum has a transmission window between 200-800 Å, whereas titanium is strongly absorbing in this region. A titanium layer is therefore used to suppress a very strong chromospheric line, He 304 Å, which falls in this band. The polyimide provides the strength needed to survive the acoustic launch loads.

Spectral Filtering

A filter wheel with 12 positions is provided for broadband spectral filters, with one position reserved for radiation shield. There are nine broadband filters, four made from beryllium (0.05 mm, 0.025 mm, 0.0127 mm) and five made from aluminum/polyimide/titanium (4800Å, 7300Å, 11,800Å). Of the remaining positions, one is open and the other contains a UV diffuser for use with the UV lamp. The filter system is designed to minimize the effect on the image of any non-uniformity in the filter materials. The position of the filter wheel is commandable from the ground. The filters are sized so as to not obstruct or vignette the field of view. The various wavebands are selectable by the filters in conjunction with the pre-filter.

Spectral Filtering

Position	Wavelength (Å)	Filter	Filter Material (mm)	Poly Layers (mm)
1	6 to 65	Poly thin	Al/Poly/Ti	0.003/0.0010/0.0008
2	6 to 60	Poly med	Al/Poly/Ti	0.005/0.0015/0.0008
3	6 to 50	Poly thick	Al/Poly/Ti	0.008/0.0030/0.0008
4	6 to 20	Be thin	0.0127 Be	
5	6 to 80	Open	Blank	
6	0	Radn	Radiation shield	
7	6 to 12	Be thick	0.05 Be	
8	6 to 16	Be med	0.025 Be	
9	6 to 20	Be thin	0.0127 Be	
10	6 to 60	Poly med	Al/Poly/Ti	0.005/0.0015/0.0008
11	6 to 65	Poly thin	Al/Poly/Ti	0.003/0.0010/0.0008
12	UV	UV	UV diffuser	

HIGH ACCURACY SUN SENSOR

The pointing knowledge required of the SXI is ±10 arcsec, which is met by the HASS. The HASS consists of a Sun sensor head and a Sun sensor electronics box. Various reticles and associated solar cells form the optics of the sensor head. The electronics package provides multiplexed processors for the coarse and fine Sun data, and for detecting Sun presence. The Sun angle output signals are passed through shift registers. The HASS output to the data electronics box is digital. Serial data interface circuits and control circuits provide command capability. The HASS provides pointing knowledge, but no pointing control, during image integration.

HASS Features

Field of view	±2° square
Resolution	5 arcsec (each axis)
Sampling rate	32 Hz
Stray light rejection	>10.5° from optical axis
Power	35 mA from 5 to 20 V dc bus
Mass: - Sun sensor - Electronics	 0.6 kg 1.4 kg

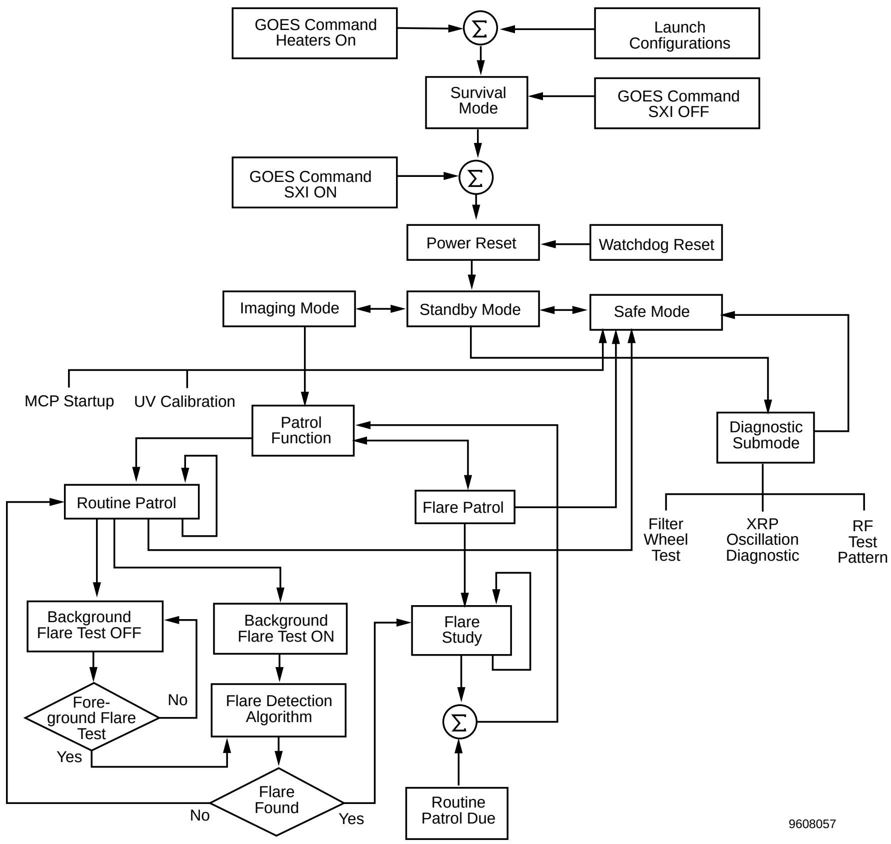
OPERATIONAL MODES

The SXI status and configuration are controlled or monitored during five operational modes.

Survival Mode

The SXI data system is unpowered and the outputs from the SXI dc/dc converters are inhibited. Survival power is provided for thermostatically controlled heaters. The temperatures are monitored by the GOES PCM telemetry system during this mode. The GOES systems provide the conditioning circuitry, with SXI providing calibrated thermistors. Temperatures monitored in this mode are mirror assembly, CCD assembly, data electronics box, and power electronics box.

SXI Control System Logical Flow Diagram



Standby Mode

The SXI defaults to this mode at power-up. The HVPS is inactive and no imaging is performed. Commands to perform diagnostics and housekeeping are allowed in this mode. Modifications to imaging tables are also allowed, and housekeeping is downlinked.

Imaging Mode

This is the operational mode for the SXI. This mode may be entered only by ground command. In this mode the SXI can:

- Image the sun
- Take UV bulb image for diagnostic purposes
- Background flux images (radiation shield in place)

Windowing

The windowing option allows selected arrays of data to be transmitted to the ground at a higher cadence than is available with the full disk downlink normally utilized. When commanded into windowing, the SXI downlinks the pixel values from up to six ground-defined windows. The size of the windows can range from 16 x 16 pixels, up to the entire CCD. (Windowing is not a mode, but an option.)

Diagnostic Mode

Diagnostics mode is entered from Standby Mode when executing the Filter Wheel Diagnostics, MCP startup, XRP Oscillation Diagnostics, and RF Test Pattern.

Safe Mode

The Safe Mode is entered either by ground command or, if a serious hardware or commanding error is detected, by software. When transitioning to Safe Mode, the HVPS is deactivated and the filter wheel is rotated to the radiation shield and disabled. The Safe Mode is used to prepare the instrument to lose power and effects a graceful shutdown of the SXI subsystems.

POWER

Electrical power is provided to the SXI via the power electronics module from the spacecraft electrical power subsystem. The electrical interfaces to the spacecraft are connections across the yoke gimbal via slip rings. The power required by the SXI is 57 watts. The spacecraft provides power to the SXI instrument from the primary bus that is regulated at 42.0 ± 0.5 V dc during sunlight operation. During eclipse, this primary power bus is controlled by battery voltage, which may vary between 29.0 and 43.0 V dc. All of the SXI power needs come from this supply.

The power source for the SXI primary input power and for its heaters is the spacecraft primary power bus. The primary bus and heater power provided to the SXI are protected with fuses within the spacecraft.

The input power consumption by the SXI is:

Sunlight: 57 W maximum from the main bus and heater bus, combined.

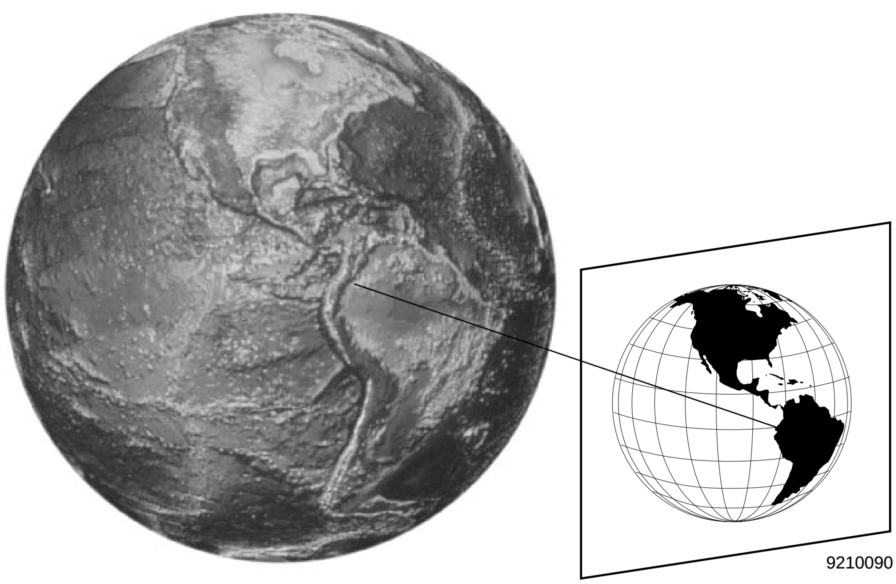
Eclipse: 15 W; this power represents the heater power allocated to the SXI during eclipse, to maintain SXI temperatures within allowable limits.

Image Navigation and Registration

In addition to the acquisition of imaging and sounding data, the GOES I-M system is capable of registering to a high degree of accuracy the earth's latitude and longitude locations of each picture element (pixel) within an image. This accurate determination of pixel location is accomplished by a ground computer in the operations ground equipment (OGE), which processes star and landmark data obtained by the Imager and Sounder. Working in conjunction with the spacecraft attitude and orbit control electronics (AOCE), the process maintains pointing of the pixels.

The accurate location and pointing of each pixel is a two-part process. The first part, image navigation, determines the location of a pixel within an image relative to an earth-referenced latitude and longitude. The second part, registration, entails maintaining the location of the pixels within an image and between repeated images to their earth-referenced latitude and longitude. This unique process, image navigation and registration (INR), yields daily imaging and sounding data on a precisely located, fixed-earth coordinate grid without ground interpolation.

Image Navigation/Registration with Fixed Earth Projection



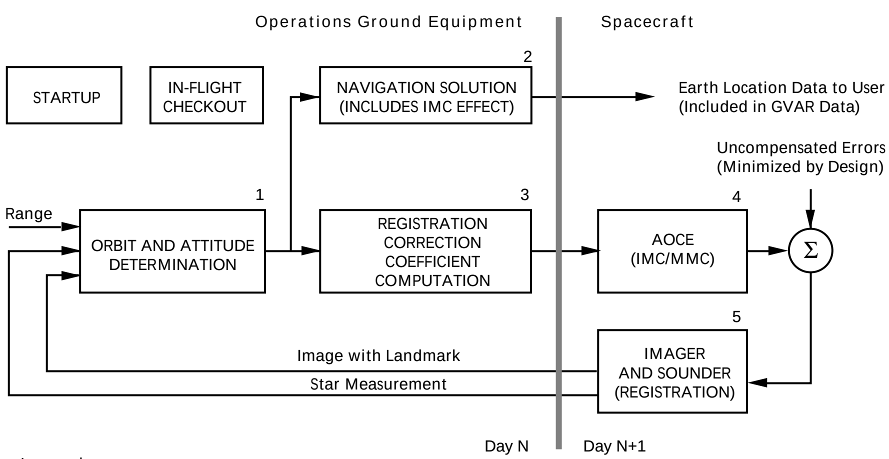
Fundamental Basis for INR Approach

Ideally, if the spacecraft were truly geostationary and the optical axes of the imaging instrument were to be held fixed relative to a geocentric coordinate frame, corresponding pixels of successive images would have the same earth longitude and latitude. In reality, however, orbital motion of the spacecraft and perturbations in the attitude of the optical axes, cause varying image motions over time. The purpose of the INR system is to correct for such motions at the source (the scan mirror) so that the apparent pixel shift due to these effects is compensated for in real-time, and the resulting earth projection remains fixed in the scan coordinate frame.

To accomplish this purpose the design of the INR system is based on three basic approaches:

- Spacecraft range data (altitude) together with star and landmark observation data (angles) are used in the spacecraft orbit and attitude determination process (block 1) in the OGE.
- The star and landmark data are obtained through the Imager (Sounder) itself (block 5); therefore all long-term effects are "zeroed out," that is, the long-term effects are the same for each pixel as they are for stars and landmarks.
- Once long-term effects are modeled to compute registration correction coefficients (block 3), the mirrors of the Imager and Sounder are biased in east / west (E / W) and north / south (N / S) directions by the AOCE to compensate for those effects (block 4). The remaining error in navigation

Closed-Loop Image Navigation and Registration System



Legend

AOCE	Attitude and Orbit Control Electronics	IMC	Image Motion Compensation
GVAR	GOES Variable	MMC	Mirror Motion Compensation

9210091

(block 2) and registration of each pixel (block 5) is primarily due to errors in attitude modeling, spacecraft stability, and scanner repeatability errors, which are minimized by design.

The real-time image motion compensation (IMC) and mirror motion compensation (MMC) of the INR system on board the spacecraft compensate for the orbit and attitude variations of the instruments to obtain images and soundings with a fixed-earth projection.

Image Navigation

Pixel navigation for the Imager and Sounder is independently established by orbit and attitude determination of each unit's optical line of sight. This computation, performed every 24 hours, is based on ranging data, landmark observations, and star observations made by each instrument. Orbit determination is accomplished primarily from range data and landmark observations. The ranging measurements are obtained by using the processed data relay (PDR) link and computing the round trip propagation time of the GOES variable (GVAR) format data bit stream. Attitude of the line of sight is based on star sensing through the Imager and Sounder optics and landmark observations; the star sense data are based on at least three different stars from both instruments and obtained about every 30 minutes. Hourly landmark data points are taken during the daylight intervals.

The spacecraft attitude is maintained and controlled with respect to the earth by means of an infrared earth sensor (ES), which provides the reference, and a momentum bias system with two skewed momentum wheels (MWs) for pitch and roll control. The spacecraft attitude control provides the short-term stability.

After accumulating the above line-of-sight information over a 24-hour period, the OGE models pixel drift experienced as a result of the long-term spacecraft motions due to both orbital effects and structural thermal distortion. The OGE then derives a set of compensation coefficients to be applied to the servo drives of the Imager and Sounder mirrors. These image motion compensation coefficients are uplinked daily to the spacecraft via the Command and Data Acquisition (CDA) Station, processed by the AOCE, and applied in the registration portion of the INR process.

Image Registration

The approach to registration is based on neutralizing the deterministic errors by applying a compensating signal to the scan mirror servo drive to correct the apparent pixel shifting. This shifting is caused by spacecraft motions, including long-term orbital and thermal effects and short-term effects due to the mechanical interaction of one instrument's mirror upon the other. The scan mirror is driven in such a way that the resulting image remains fixed in the scan coordinate system, thus achieving the required registration accuracy and fixed grids over a 24-hour period. The onboard correction compensates for mirror and image motions:

- IMC corrects for the long-term thermal distortions of the spacecraft, Imager and Sounder instruments, and Earth Sensor, as well as from spacecraft yaw and orbital deviations.
- MMC corrects for spacecraft short-term rigid body disturbances caused by mirror motion such as normal scan, retrace, blackbody and star sense.

Compensation for the mirror interaction effects is computed by the spacecraft AOCE and an analog mirror motion compensation signal is superimposed on the IMC signal prior to its application to the Imager and Sounder drives. The IMC coefficients are derived by the OGE.

INR Functional Configuration

Near real-time image navigation and registration are accomplished by implementing the various INR aspects on board the spacecraft and in the OGE at the SOCC and the CDA Station, all guided by the basic approach described above. The major functions of the INR system are:

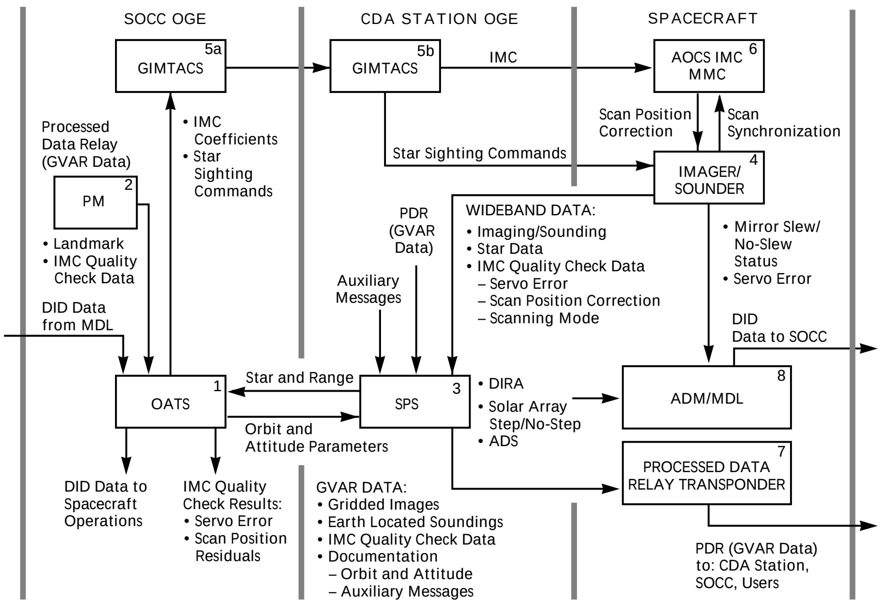
- Determination of spacecraft orbit and Imager and Sounder attitudes
- Generation of star sighting command data
- Landmark observation and ranging
- IMC and MMC
- Provision of real-time earth location information for users
- IMC quality check
- Dynamic interaction diagnosis

Orbit and Attitude

Spacecraft orbit and Imager/Sounder attitudes are determined in the orbit and attitude tracking system (OATS, block 1 of the following diagram) of the OGE at the Satellite Operations Control Center (SOCC). The product monitor (PM, block 2) sends to OATS data consisting of landmark longitude and latitude and corresponding instrument coordinates for the various landmarks while the sensor processing system (SPS, block 3) sends star data observed by the Imager/ Sounder (block 4) and spacecraft range measured at the SPS. The off-line OATS determines spacecraft orbit and Imager/Sounder attitude parameters from spacecraft range, star, and landmark data and transmits the parameters to the SPS for gridding and inclusion in the GVAR data stream to users.

The OATS also generates star sighting command data issued directly to the Imager/Sounder via the GOES I-M telemetry and command system (GIMTACS, block 5), and periodically updates IMC coefficients for transmission to the onboard AOCE (block 6) through GIMTACS. The SPS provides the necessary

Functional Configuration - Image Navigation and Registration System



Legend

ADM	Attitude Data Multiplexer	IMC	Image Motion Compensation
ADS	Angular Displacement Sensor	MDL	Multiuse Data Link
AOCS	Attitude and Orbit Control Subsystem	MMC	Mirror Motion Compensation
DID	Dynamic Interaction Diagnostic	OATS	Orbit and Attitude Tracking System
DIRA	Digital Integrating Rate Assembly	PDR	Processed Data Relay
GIMTACS	GOES I-M Telemetry and Command System	PM	Product Monitor
GVAR	GOES Variable	SOCC	Satellite Operations Control Center

9210092

data averaging, threshold detection, signal peak or midpoint determination, and time tagging of star sense data required as input to OATS.

Spacecraft range is determined at SPS from the time difference between uplinking the GVAR data stream from the SPS to the spacecraft and receiving the retransmitted GVAR downlink. The determined range and associated statistics are transmitted to OATS via GIMTACS. The range residual is calculated in OATS and used to estimate the spacecraft orbit and attitude. The ranging error due to noise is about 10 meters (33 feet).

Earth landmark data are essential to separate the orbit and attitude effects; landmarking is a manual task using the PM. Landmarks can be obtained from the visible detectors in the Imager (1-kilometer or 0.6 statute mile resolution) that are primary data, visible detectors in the Sounder (10-kilometer resolution) that are backup, and IR detectors in the Imager (4-kilometer resolution) that are also backup.

Image and Mirror Motion Compensation

In the IMC/MMC system, the spacecraft AOCE processor receives updated IMC coefficients from OATS and scan synchronization signals from the Imager and Sounder. With these data, the AOCE generates and sends to the Imager and Sounder servos mirror scan compensation signals, which are the sum of the IMC and MMC adjustments. These compensation signals are then converted to the proper scan adjustments in the Imager and Sounder, producing images and soundings with a fixed-earth projection.

The SPS receives wideband data from the spacecraft and generates in real-time GVAR formatted data consisting of gridded images, earth-located soundings, IMC quality check data, and documentation data (orbit and attitude parameters, auxiliary messages). The GVAR data stream is transmitted in real-time from the SPS to the spacecraft (block 7) and retransmitted in real-time to users, the CDA Station, and the SOCC. The SPS uses the received GVAR data for range measurements.

INR Quality

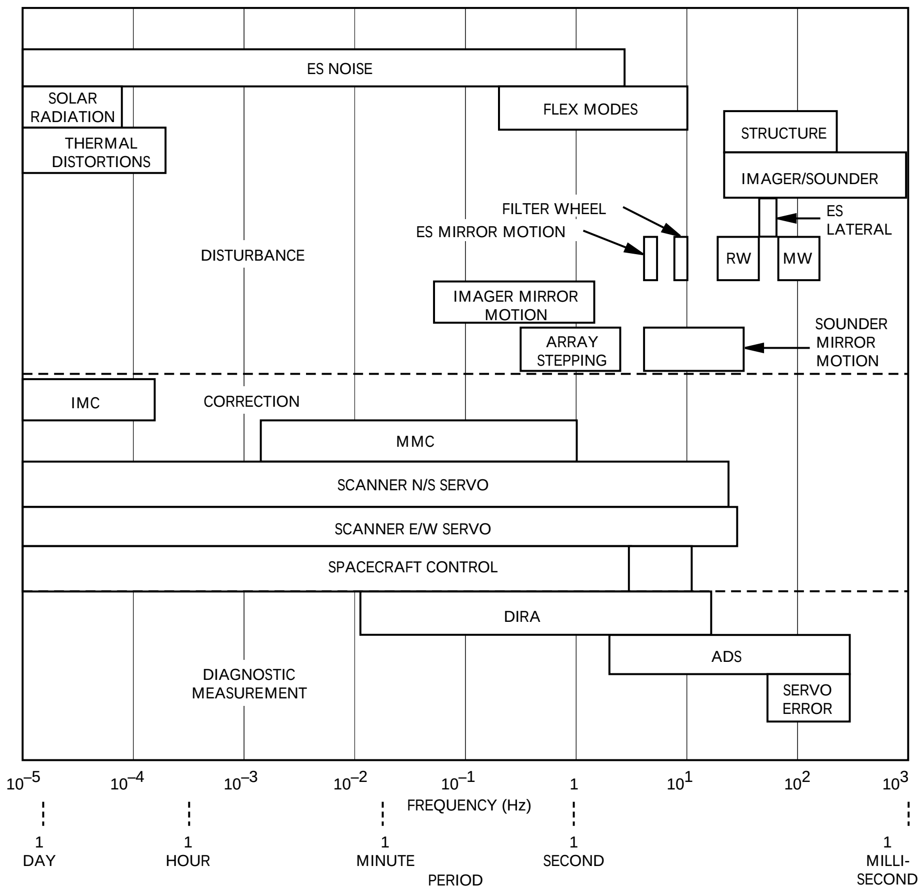
IMC quality check data consists of mirror position, compensation signal in effect, and status of the Imager and Sounder mirrors. Quality data are removed from the wideband data by the SPS, translated into the GVAR format, extracted from the GVAR downlink by the PM, and routed to OATS. The OATS then determines the IMC quality check results that are servo errors and mirror scan position residuals for both the Imager and Sounder. These results should be within prescribed limits for proper IMC operation; such limits are in the OATS data base. In addition to the above, monitoring of star, landmark, and range residuals by the OATS operator provides another source of data for verifying INR system operation.

An independent diagnostic attitude measurement system monitors any unforeseen disturbances affecting INR, including spacecraft dynamic interactions. From the multiuse data link (MDL, block 8), the OATS receives and processes dynamic interaction diagnostic (DID) data consisting of attitude data from the digital integrating rate assembly (DIRA), angular displacement sensor (ADS) data, and servo error data from the Imager and Sounder. The processed DID data, provided to SOCC, assist spacecraft operators and analysts to initialize and monitor the INR system.

INR Dynamic Environment

The complete dynamic environment of the spacecraft for image navigation and registration includes the dynamic interaction disturbance sources that affect INR, approaches for correcting the effects of such disturbances, and the dynamic interaction diagnostic measurement system installed on the spacecraft. It also includes the frequencies associated with the minor disturbances, as well as the natural vibration frequencies of the various spacecraft elements.

Spacecraft Dynamic Environment for INR



Legend

ADS	Angular Displacement Sensor	IMC	Image Motion Compensation
DIRA	Digital Integrating Rate Assembly	MMC	Mirror Motion Compensation
ES	Earth Sensor	MW	Momentum Wheel
E/W	East/West	N/S	North/South
		RW	Reaction Wheel

9401141

Disturbance Sources

Minor disturbances to the spacecraft on orbit are caused by moving elements and flexible appendages that could lead to jitter in the line of sight of the Imager, Sounder, and ES. Such disturbances are caused by residual static and dynamic imbalance of MWs / reaction wheel (RW), scanning mirror motions of the ES, Imager and Sounder, solar array stepping, and oscillations of flexible appendages (solar array, solar sail and boom, and magnetometer boom). Imager and Sounder line-of-sight pointing errors are also affected by variations in thermal distortions of and solar radiation pressure on the spacecraft.

Correction Approaches

The spacecraft design incorporates means for compensating to a large extent the effects of the dynamic environment on INR. The IMC system on board the spacecraft mainly compensates for orbit and attitude errors due to thermal distortion and solar radiation effects. The MMC system compensates for the effects on the rigid body portion of the Imager and Sounder mirror motions. These correction approaches are also supported from the ground through available commands which can select stepping profiles that prevent solar array stepping motions from sympathetically interacting with the flexible appendages and from causing resonance in the solar array.

The AOCS attenuates the ES noise effect on spacecraft motion by a factor of 29 (from 375 to about 13 μrad). The Imager and Sounder scanner servo errors (E / W and N / S) are minimized by design (for example, using a coherent error integrator for Imager E / W motion and an average error integrator for Imager and Sounder N / S stepping motions).

Diagnostic Measurement (GOES-I & K)

As noted above, the diagnostic attitude measurement system monitors unforeseen disturbances affecting INR. The attitude motions in frequencies from near zero to 300 Hz can be determined from DID data. The DIRA provides attitude data in the range from zero to 15 Hz, the ADS from 2 to 200 Hz, and the Imager and Sounder servo error telemetry scan mirror pointing data from 80 to 150 Hz. The star and landmark measurements provide information on the effects of thermal distortion and solar radiation on attitude.

Communications Subsystem

The spacecraft communications subsystem provides the conditioning, transmission, reception, and routing of attitude telemetry and mission data signals for the GOES space segment. It consists of six major component groups that provide a variety of functions:

- Sensor data (SD) and multiuse data link (MDL) transmitters
- Data collection platform interrogate (DCPI) transponder
- Data collection platform report (DCPR) transponder
- Processed data relay (PDR) and transponders
- Weather facsimile (WEFAX) transmission
- Search and rescue (SAR) transponder

Four antennas, each with full earth coverage beamwidth, are used to provide communications with the ground segment:

- S-band receive horn: receives uplink DCPI, PDR, and WEFAX signals
- S-band slotted array: transmits SD, MDL, DCPR, and PDR signals
- UHF cavity backed dipole: receives DCPR and SAR signals; transmits DCPI signal
- L-band helix: transmits SAR downlink signal

Multiplexers are used to interface multiple signals with low RF loss into and/or out of an antenna. The S-band output multiplexer filters and combines the SD, MDL, DCPR, PDR, and WEFAX signals and a UHF diplexer multiplexes the downlink DCPI, uplink DCPR, and SAR signals. Stand-alone filters reject unwanted signals and band limit desired signals; filters after the S-band receivers separate the PDR, DCPI, and WEFAX signals for later hard-limiting amplification in their respective sections. Transponders and receivers are used for low-noise amplification and frequency conversion, while transmitters provide carrier modulation and power amplification.

Space-Ground Communications Interfaces

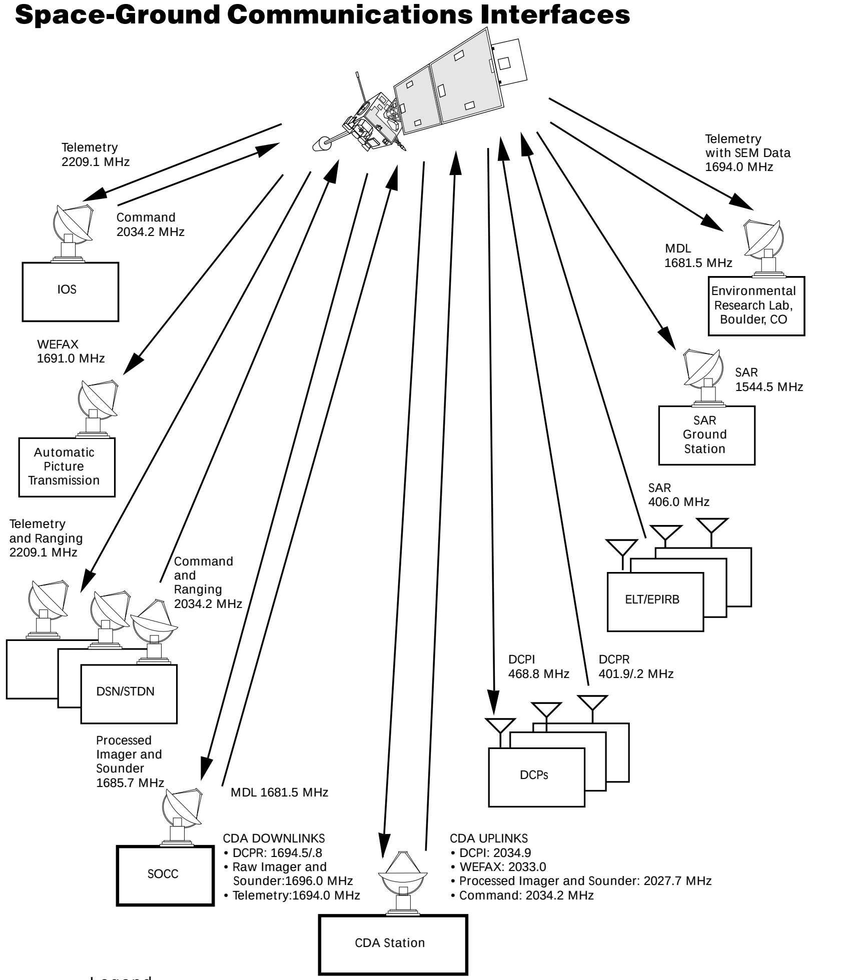
The flexibility and multitude of services provided by the GOES spacecraft are illustrated by the communication interfaces between the spacecraft and ground. The major system interfaces are those linking the GOES I-M to the command and control ground stations, end user equipment, and communications service terminals. The principal interfaces are those between the spacecraft and the Command and Data Acquisition (CDA) Station:

GOES Communications Performance Summary

Communications Parameter	Performance
Bit rate stability	±0.03%
Bit error rate	1 x 10^{-5} with 3.9 dB margin
Command receiver G/T	–39.2 dB/K
Command link margin	13.9 dB CDA Station 22.2 dB Deep Space Network (DSN)
Simultaneous ranging margin	12.5 dB DSN
Telemetry EIRP	28.9 dBm CDA Station Telemetry link margin 18.5 dB 17.0 dBm DSN Telemetry link margin 5.1 dB
Sensor data EIRP	47.3 dBm
Imager link margin	4.4 dB
Sounder link margin	16.6 dB
Processed data relay	
EIRP	54.9 dBm; link margin 3.9 dB
G/T	–15.3 dB/K
MDL EIRP	45.2 dBm; link margin 11.9 dB
Data collection platform	
Interrogate EIRP	46.2 dBm; link margin 11.0 dB
Interrogate G/T	–15.3 dB/K
Report EIRP	34.9 dBm; link margin 9.2 dB
Report G/T	–18.6 dB/K
WEFAX G/T	–15.3 dB/K
SAR	
EIRP	46.0 dBm; link margin 3.4 dB
G/T	–16.3 dB/K

- CDA downlinks:
  - Data collection platform report
  - Raw Imager and Sounder data
  - Spacecraft telemetry
- CDA uplinks:
  - Data collection platform interrogation
  - Weather facsimile transmission
  - Processed Imager and Sounder data
  - Spacecraft commands

Space-Ground Communications Interfaces



Legend

CDA	Command and Data Acquisition	IOS	Indian Ocean Station
DCP	Data Collection Platform	MDL	Multiuse Data Link
DCPI	Data Collection Platform Interrogate	SAR	Search and Rescue
DCPR	Data Collection Platform Report	SEM	Space Environment Monitor
DSN	Deep Space Network	SOCC	Satellite Operations Control Center
ELT	Emergency Locator Transmitter	STDN	Satellite Tracking Data Network
EPIRB	Emergency Position Indicator Response Beacon	WEFAX	Weather Facsimile

9210041

Data Links

Telemetry containing space environment data is downlinked to the end user at the Environmental Research Laboratory in Boulder, Colorado. Also, processed Imager and Sounder (calibrated, earth-located) data are downlinked to the Satellite Operations Control Center (SOCC) at Suitland, Maryland, then to the World Weather Building for subsequent distribution to end users, which are typically satellite field service stations located throughout the United States. The SOCC also receives diagnostic data from the MDL for further analysis.

Center Frequency Assignments

Downlink Center Frequency	MHz
UHF	
Data collection platform interrogation	
Frequency 1	468.8250
Frequency 2	468.8375
S-Band	
SAR	1544.500
WEFAX	1691.000
DCP report	
Pilot	1694.450
Frequency band 1	1694.500
Frequency band 2	1694.800
Telemetry	
CDA Station	1694.000
DSN*	2209.085
Tracking	2209.085
Raw imaging and sounding data	1676.000
Processed imaging and sounding data	1685.700
Multiuse data	1681.480

* DSN telemetry data is modulated on a 1.024 MHz subcarrier, then used to phase modulate the DSN telemetry carrier frequency.

Uplink Center Frequency	MHz
UHF	
Data collection platform report	
Pilot	401.850
Frequency band 1	401.900
Frequency band 2	402.200
SAR	
Normal mode	406.050
Narrowband mode	406.025
S-Band	
WEFAX	2033.000
DCP interrogation	
Frequency 1	2034.9000
Frequency 2	2034.9125
DSN ranging	2034.200
CDA Station and DSN spacecraft command frequency	2034.200
Processed imaging and sounding data	2027.700

Telemetry, Command and Ranging

Telemetry, command, and ranging (TC&R) data are downlinked and uplinked among a network of stations, including the Indian Ocean Remote Tracking Station, the NASA Deep Space Network (DSN), and Satellite Tracking and Data Network (STDN), with the CDA Station as the center for the origin of commands and reception of spacecraft telemetry (refer to Telemetry and Command Subsystem).

Search and Rescue

The Search and Rescue (SAR) subsystem onboard each GOES satellite is a dedicated transponder that detects the presence of distress signals broadcast by Emergency Locator Transmitters (ELTs) carried on general aviation aircraft and by Emergency Position Indicating Radio Beacons (EPIRBs) aboard some classes of marine vessels. The SAR mission is performed by relaying the distress signals emitted from the ELT/EPIRBs via the GOES satellite to a Search and Rescue Satellite-Aided Tracking (SARSAT) ground station located within the field-of-view of the spacecraft. Through a Rescue Coordination Center, help is dispatched to the downed aircraft or ship in distress.

Data Collection System

The GOES Data Collection System (DCS) collects near real-time environmental data from data collection platforms (DCPs) located in remote areas where normal monitoring is not practical. The DCS receives data from DCPs on aircraft, ships, balloons, and fixed sites in a region from Antartica to Greenland and from the west coast of Africa to just east of the Hawaiian Islands, an area covered by the GOES satellites. The system encompasses almost every level of the atmosphere, land, and ocean. It is used to monitor seismic events, volcanoes, tsunami, snow conditions, rivers, lakes, reservoirs, ice cover, ocean data, forest fire control, meteorological and upper air parameters, and to provide ground truth information.

Weather Facsimile Transmission

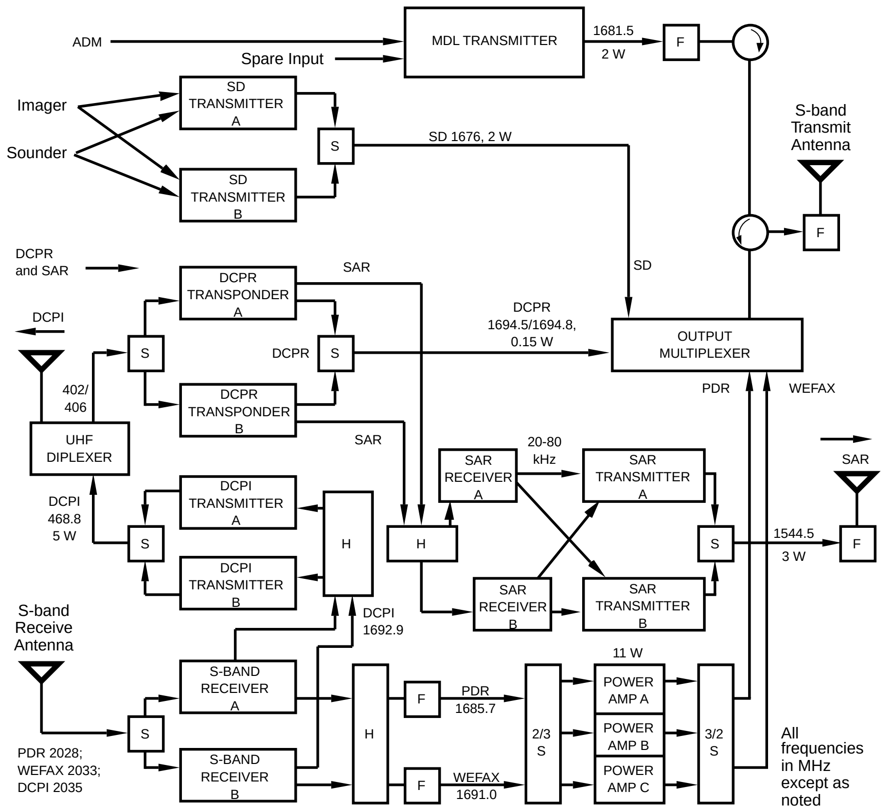
The Weather Facsimile (WEFAX) transmission is a communication service provided through a transponder on board the GOES satellite. Low resolution satellite imagery and meteorological charts from GOES and polar orbiting satellites are uplinked by the CDA ground station. The GOES transponder provides for continuous WEFAX transmissions, except during eclipse periods, relaying the data to small, local user ground terminals in the Western Hemisphere. Information distributed by this transmission service provides the only satellite imagery for many countries as well as standard meteorological charts from the World Meteorological Center, Washington, D.C. The WEFAX transmission frequency is common with that used by the European Space Agancy’s METEOSAT and Japan’s GMS spacecraft, enabling near-global access to WEFAX service. This global availability is especially important to commercial shipping and U.S. military operations.

Sensor Data and Multiuse Data Link Transmitters (GOES-I & K)

Both the SD transmitter and MDL transmitter can accept two asynchronous baseband data streams and simultaneously modulate them onto the downlink carrier in quadrature with 6-dB amplitude unbalance. The two-for-one redundant SD transmitters each receive a 2.62-Mb/s signal from the Imager and a 40-kb/s signal from the Sounder. Two ground-commandable switches allow either of the Imager’s or Sounder’s two output signals to be routed to either SD transmitter. The Imager signal is provided to the modulator at a 6-dB higher level than the Sounder, producing an unbalanced, asynchronous, quadraphase shift-keyed signal amplified to 2 watts.

For GOES I, and one other spacecraft to be identified later in the program, the single MDL transmitter receives only the 32-kb/s signal from the attitude data multiplexer (ADM). In this configuration the modulator produces a biphase shift-keyed signal amplified to 2 watts. Both the SD and MDL signals are band-limited by filters and multiplexed into the S-band transmit antenna. The 3-dB bandwidths of SD and MDL filters are 6 and 4 MHz, respectively.

Communications Subsystem Configuration



Legend

ADM	Attitude Data Multiplexer	PDR	Processed Data Relay
DCPI	Data Collection Platform Interrogate	S	Switch
DCPR	Data Collection Platform Report	SAR	Search and Rescue
F	Filter	SD	Sensor Data
H	Hybrid	UHF	Ultra High Frequency
MDL	Multiuse Data Link (GOES I & K)	WEFAX	Weather Facsimile

9210043

Data Collection Platform Interrogate Transponder

The two DCPI uplink carriers are digitally phase modulated at 100 b/s at center frequencies of 2034.9 and 2034.9125 MHz; both signals may be transmitted simultaneously. The signals share the same receive antenna and receiver with the PDR and WEFAX signals, although the desired DCPI signal exits the selected receiver from a coupled port. The coupled output from each receiver is combined in a hybrid and provided to both DCPI transmitters; in the selected transmitter the signals are downconverted to UHF and bandpass filtered. The 3-dB bandwidth of the filter is about 200 kHz. The signal is then hard-limited and amplified to 5 watts. The operating transmitter output is provided to the UHF diplexer and UHF antenna for transmission to the ground platforms.

Data Collection Platform Report Transponder

Up to 233 DCPR signals may be accommodated in one of the two frequency bands: 401.9 and 402.2 MHz. These signals, digitally phase modulated at 100 b/s, are received by the UHF antenna and routed via the diplexer to one of two DCPR transponders, where they are amplified (using low-noise amplifiers) and provided 400-kHz, 3-dB bandwidth filtering in one of two selectable bands. The signals are then upconverted in frequency and amplified with automatic gain control to 150 milliwatts. The output of the selected transponder is filtered (2-MHz, 3-dB bandwidth), combined with the other S-band downlink signals by the output multiplexer, then transmitted to the ground via the S-band transmit antenna.

Processed Data Relay and Weather Facsimile Transponders

The PDR and WEFAX signals are received by the S-band receive antenna and amplified (using low-noise amplifiers) and downconverted in frequency in one of two S-band receivers. The receiver outputs are combined in a hybrid and provided to independent PDR and WEFAX channel filters. The PDR signal is 2.1-Mb/s biphase shift keyed and is band limited by the PDR filter, which has a 3-dB bandwidth of about 5 MHz. The WEFAX signal is FM with an approximate 30-kHz bandwidth and a 1-MHz wide filter. The filters reject the adjacent channels (including the DCPI and command signals) and send their outputs to a three-to-two switch. The signals are sent to two of the three S-band power amplifiers. Power amplifier A is dedicated to the PDR channel, power amplifier C is dedicated to the WEFAX channel, and power amplifier B can be switched to either PDR or WEFAX.

The power amplifiers hard-limit the signals, amplify them to 12 watts, and provide them to an output three-to-two switch. The input and output three-to-two switches are controlled by the same commands. The signals are then bandpass filtered and multiplexed for transmission to the ground via the S-band transmit antenna. The 3-dB bandwidths of the PDR and WEFAX are 5 MHz.

Search and Rescue Transponder

The uplink SAR signals, digitally phase modulated at 400 b/s in the frequency band of 406.01 to 406.9 MHz, are received by the UHF antenna and amplified, using low-noise amplifiers, by one of two DCPR transponders. The signal is then coupled out to a hybrid that combines both DCPR preamplifiers and drives both SAR receivers. The operating SAR receiver filters out the DCPR band, downconverts the signal frequency, performs bandpass filtering in one of two bands (20 or 80 kHz, 3-dB bandwidth), then further downconverts the signal to near baseband. The SAR receiver can be commanded into a fixed gain or automatic level control mode.

The output of each SAR receiver is routed to both SAR transmitters. The selected SAR transmitter phase modulates a UHF carrier with the signals, multiplies the frequency to L-band, then filters and amplifies the modulated carrier to 3 watts. The nominal modulation index is one radian; a baseband limiter in the transmitter ensures the index does not exceed two radians. The output of the selected transmitter is passed through a harmonic filter (100-MHz, 3-dB bandwidth) and finally transmitted by the helical antenna.

Attitude Data Multiplexer (GOES-I & K)

The attitude data multiplexer (ADM) provides diagnostic data used to determine the dynamic interaction among spacecraft components with mechanical motion. The data are provided to the ADM at 32 kb/s where they are multiplexed and the output routed to the MDL transmitter for downlinking to the SOCC. The sources and types of data provided to the ADM are:

- Angular displacement sensor (ADS): roll, pitch, yaw and ADS temperature
- Digital integrating rate assembly (DIRA): roll, pitch, and yaw
- Solar array: step and slew events
- Imager and Sounder: slew events, north/south and east/west servo error

Emergency Locator Transmitter and Emergency Position Indicating Radio Beacon Performance Characteristics

ELT/EPIRB Parameter	Performance
RF Signal	
Frequency	406.025 ±0.001 MHz
Stability	1 in 10^9 in 100 ms 2 kHz in 5 years
Phase jitter	10° RMS measured in 50 Hz bandwidth
Power output	3 W +1 dB; –2 dB into 50 Ω
Spurious	50 dB below carrier; harmonics, 30 dB below carrier
Data encoding	Biphase L
Modulation	Phase modulation 1.1 ±0.1 radians peak referenced to unmodulated carrier
Modulation rise time	Rise and fall times of modulation wave forms must be less than 0.25 ms
Digital message	
Repetition rate	50 s ±5%
Transmission time	440 ms ±1%, short message 520 ms ±1%, long message
CW preamble	160 ±1%
Digital message	280 ms ±1%, short message 360 ms ±1%, long message
Bit rate	400 ±5 b/s
Bit synchronization	All ones (fifteen ones)
Frame synchronization	000101111
Continuous emission	Not to exceed 45 seconds

Telemetry and Command Subsystem

The telemetry and command (T&C) subsystem provides the functional interface between the spacecraft and ground command and control. Telemetry parameters describing the status, configuration, and health of the spacecraft payload and subsystems are downlinked to the Command and Data Acquisition (CDA) Station and sent to the Satellite Operations Control Center (SOCC). Commands are received on board the spacecraft for controlling mission operations and managing expendable resources. To perform these functions, the T&C subsystem is composed of a single bicone antenna mounted on the spacecraft's east panel, two radio frequency (RF) receivers, four transmitters, redundant digital telemetry and command units, and a triplexer that allows simultaneous operation of the transmitters and receivers into the bicone antenna without interference.

The T&C subsystem primarily interfaces with the NOAA Wallops CDA Station during on-orbit operations, with the NASA Deep Space Network (DSN) available as backup. The ground interfaces during orbit raising are with DSN, Air Force Indian Ocean, and NASA Wallops CDA Stations; these stations are compatible with the interface to the spacecraft T&C subsystem.

T&C Subsystem Design Features

Telemetry unit

- Fully redundant unit; one unit on at a time
- 2000 b/s data rate
- 128-word minor frame every 0.512 second
- Serial digital, bilevel, analog, and temperature parameters
- Remote multiplexer on solar array yoke (housed in XRP electronics)
- Simultaneous dwell and normal mode capability
- Simultaneous ranging and telemetry capability

Command unit

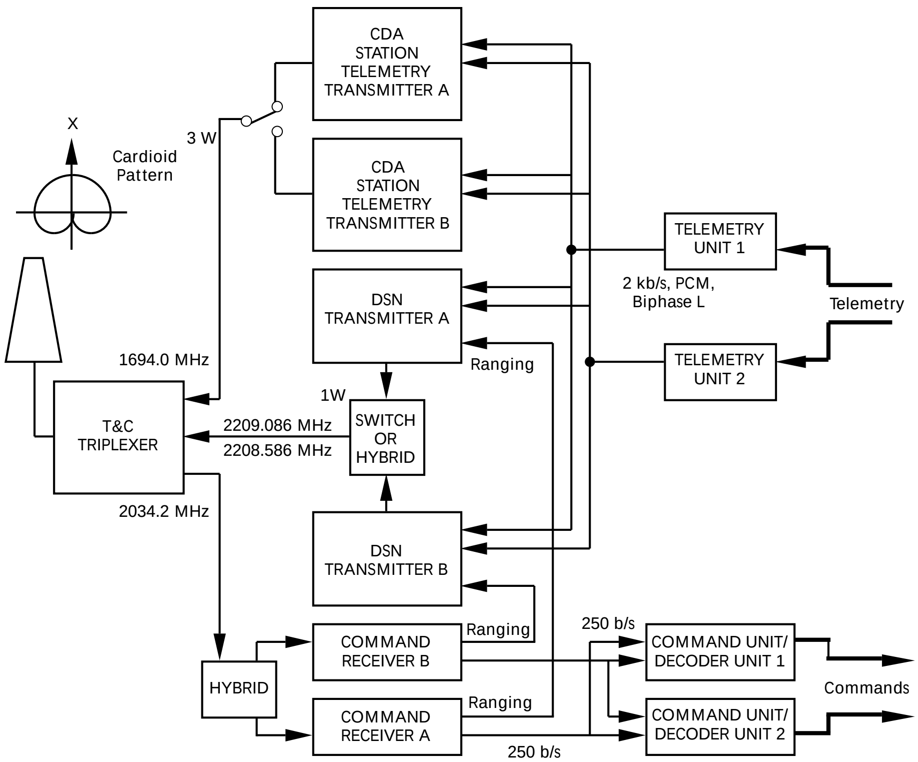
- Fully redundant unit
- 250 b/s data rate
- Added encryption system
- Relay, pulse, and proportional commands
- Three commanding modes:
  - Store only
  - Execute and store
  - Store and execute
- Simultaneous commanding and ranging
- Secure or plain text mode operation

Telemetry

Information from the spacecraft provided via telemetry is:

- Configuration status and housekeeping data for payload instruments
- Environmental sensing data from space environment monitor (SEM) instruments
- Automatic gain control setting for each receiver

Telemetry and Command Subsystem Configuration



Legend

CDA	Command and Data Acquisition	PCM	Pulse Code Modulation
DSN	Deep Space Network	T&C	Telemetry and Command

9210045

- RF power output for each transmitter
- Power system parameters and voltages of critical electronic modules
- On/off status of all commandable equipment and heaters
- Temperatures of all major subassemblies
- Spacecraft attitude determination and control parameters
- Parameters of frame synchronization, spacecraft identification, command counter, secure mode operation, 16-bit error detection code, etc.

The telemetry units are “standby redundant,” meaning only one unit is on and operating at a time. The operating telemetry unit collects the data, encodes, multiplexes, and formats it into two (normal and dwell) serial pulse code modulated (PCM) bit streams. Both normal and dwell PCM data are generated simultaneously by the selected telemetry unit; either normal or dwell PCM data are provided to the four telemetry transmitters. Only one of the two CDA transmitters can be on at a time, but the DSN transmitters can be configured for single or dual operation. For DSN dual transmitter operation, offset frequencies are used to avoid having the two transmitters operating at the same frequency and interfering with telemetry reception. Each operating transmitter can be independently selected for either normal or dwell PCM data.

The telemetry data processed are either analog or digital. Analog data are received differentially by the telemetry unit to minimize susceptibility to noise. Each analog signal is converted into a digital signal, typically with a resolution of 8 bits. Exceptions to this resolution are measurements of the currents in the magnetic torquers of the attitude and orbit control subsystem, which are quantized into 10 bits, and the three axes of magnetic field strength as measured by the SEM magnetometers, which are digitized to 16 bits each axis. Digital inputs are both serial and parallel depending on the source. The telemetry subsystem can dwell on (that is, measure over a longer period of time) any mainframe or subframe channel, except for serial digital data sources and some fixed format channels (for example, subframe identification number, frame synchronization). During each word period (8 ms), a PCM word and a dwell word are generated, making both regular and dwell PCM available for simultaneous transmission over two RF links.

Command

The command unit features:

- Command format defined by Goddard Space Flight Center
  – 7-bit error detection code
  – Uniquely defined command decoder address
- Three command codes
- 250-b/s, phase shift keyed modulation on 16-kHz subcarrier
- Simultaneous DSN commanding and ranging
- Command override of every automatic function

The uplink signal, which can contain command and ranging data simultaneously on the same carrier, is routed to both onboard command receivers. The receivers cannot be commanded off (that is, they are “active redundant”), and once phase locked onto the carrier, they provide command data to each digital command unit/decoder, each of which is also active redundant. The command units independently switch their inputs between receivers every 320 ms until a signal is detected. Upon detection, switching is terminated and phase locking, demodulation, bit synchronization, and decoding are performed. The command units can be operated in either a clear text or encrypted/secure mode, the latter precluding unauthorized access to the spacecraft. The command decrypter can be reset to the clear text mode by an automatic timer, a power-on reset, or an automatic function provided in the event of loss of earth lock. Every command is

Barker-code checked and decoder-address checked for accuracy and, if acceptable, it is loaded and / or executed by the single command unit decoder that was selected in the command.

There are two modes of command execution: real-time and ground-command word-verify. In the real-time mode, verification of the uplinked command is performed within the command unit. If no errors are detected, the command is executed. If an error is detected, processing of the command is halted and a flag bit is telemetered to the ground station indicating that the command must be retransmitted. In the ground-command word-verify mode, the bits of the decoded command are telemetered back to the ground station for verification, and a subsequent execution message must be uplinked. The command unit outputs are completely redundant.

Ranging

Ranging is performed to determine the spacecraft orbital elements during transfer and geosynchronous orbits. Channelized to the DSN transmitter only for downlink, ranging is accomplished by ground-commanding one of the DSN transponder ranging channels on and into the coherent mode. The ground station uplinks ranging tones to the command receivers where they are routed to the selected DSN transmitter and downlinked to the ground station with the output (downlink) carrier frequency. In the coherent mode, the downlink carrier frequency is maintained at a ratio of 240 to 221 relative to the uplink carrier. With the ranging channel on, ranging and telemetry are provided simultaneously on the same downlink carrier. As with commanding, ranging can be performed only with a phase-locked receiver.

Electrical Power Subsystem

The electrical power subsystem provides conditioned power to all spacecraft and payload subsystems and components via a single power bus regulated at 42.0 ±0.5 volts dc in sunlight and 30.0 volts dc minimum during battery / eclipse operations. The power subsystem consists of a solar array, a wing-mounted sequential shunt assembly, two 12-ampere hour (Ah) nickel-cadmium batteries, and a power control unit (PCU).

Electrical Power Subsystem Design Features

Solar Array

- Single wing sun tracking
- 5-year minimum lifetime with margin

Battery

- Two 12-Ah nickel cadmium batteries
- 28 cells per battery
- 60% maximum depth of discharge

Regulation

- 30 to 42.5 V dc bus voltage
- Sequential shunt limits sunlight voltage to 42 ±0.5 V
- 32.4 V minimum bus voltage at end of eclipse

Functioning

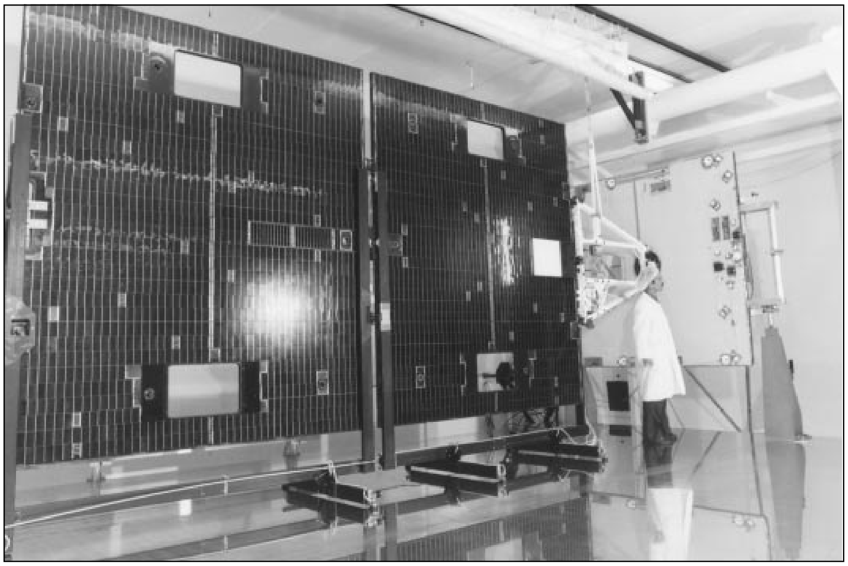
- Automatic disconnect / reconnect of sunlight loads
- Override of automatic load switching on function-by-function basis
- All loads commandable except command functions
- Redundancy for all critical functions:
  - Multiple battery charge circuits
  - Single battery cell failure tolerant
  - Fully redundant solar array drive electronics
  - Backup solar array drive motor winding

Solar Array

Primary power is supplied by a lightweight, two-panel solar array and distributed to spacecraft loads through the power harness. The two panels are attached to the single-axis, sun-tracking, continuously rotating solar array drive assembly motor by a graphite yoke. The outboard panel is initially deployed 90° to provide power during the transfer orbit phase. When the spacecraft is in geostationary orbit, both panels are deployed to their final operational position. The array is capable of generating an end-of-life, summer solstice power of 1057 watts.

The generated power is provided to the primary bus via the solar array drive assembly slip rings and the main enable plug. The array consists of 22 strings containing 121 series cells, 6 battery-charge cell groups composed of 2 sets of 12 series by 3 parallel cells, and 4 groups of 12 series cells. Array output is controlled under changing spacecraft load conditions by shunting the lower two-thirds of 20 of the main bus strings to transistors in the sequential shunt assembly (SSA). Wiring of the harness and shunting is designed to reduce the

Solar Array



current loop area, thereby minimizing the induced magnetic dipole moment of the array.

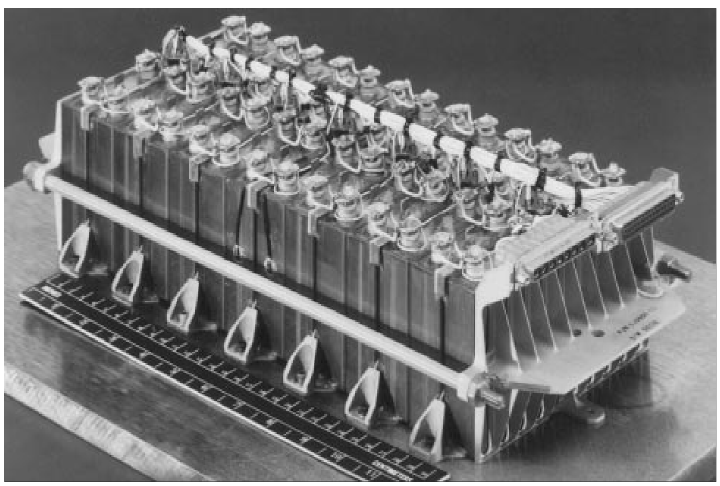
The six battery-charge cell groups are nominally arranged to allow selection of charge current levels to each of the two batteries. Application of the charge current is selected by ground command-actuated relays in the PCU. By means of a charge sequencing circuit within the PCU, selected charge rates may be continuously applied to each battery or sequenced alternately between batteries on 5-minute intervals. Ground-commandable cross-strap relays within the PCU provide redundancy and allow selection of eight different charge current levels.

Batteries

The two nickel-cadmium batteries provide the power required during launch and ascent phases of the flight (prior to outer solar panel deployment), when in eclipses, and under peak load demands. The battery power is supplied to the primary bus via parallel, redundant battery isolation diodes, redundant battery relays, and the main enable plug. An automatic eclipse-load disconnect/reconnect control capability and battery undervoltage disconnect capability are provided; both automatic features can be overridden by manual command. The spacecraft load condition in sunlight is such that some equipment must be powered off during eclipse. An automatic load shedding capability is provided which may be enabled and/or overridden upon ground command.

Each of the two batteries consists of 28 series-connected 12-Ah nickel-cadmium cells. The cell interconnects have been optimized to reduce induced magnetic dipole effects during charge and discharge current conditions. Individual cell voltages in each battery are monitored via the spacecraft telemetry through the PCU.

Heaters (resistors) are mounted in parallel on the battery intercell separators. These heaters, in conjunction with temperature sensors on each battery and the automatic temperature control circuitry contained in the PCU, supply a thermostatically controlled 19 watts of heating to maintain battery temperatures between +1 and +5 ˚C during periods of cold exposure. This thermostatically controlled interface is redundantly protected by a manual command override capability.

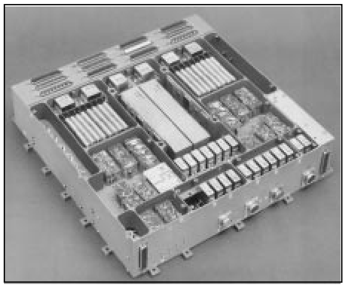


12 Ampere-hour Nickel-Cadmium Battery

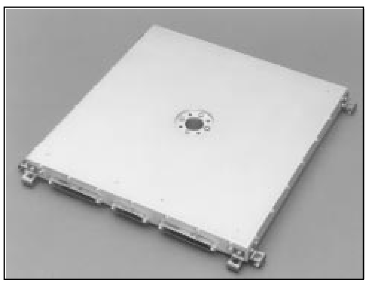
Power Control, Electroexplosive Devices, and Sequential Shunt Assembly Units

The PCU, as the functional center for all spacecraft power generation and control, regulates the solar array output through the sequential shunt assembly, maintaining 42.0 volts in the power bus during sunlight operation despite changing load conditions. This is performed by a redundant, majority-voting, error amplifier in the PCU that generates control signals for sequentially turning on shunt transistors in the SSA. The SSA is mounted on a heat sink on the dark back side of the inboard solar array panel to allow thermal dissipation.

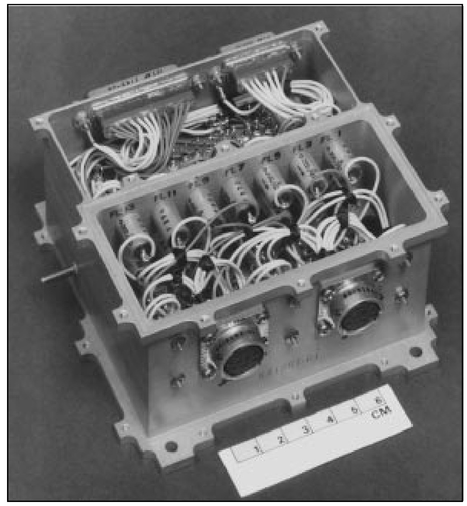
The PCU contains circuitry and relays for battery charging, sequence charging, undervoltage protection, reconditioning, and temperature control. It also



Power Control Unit



Sequential Shunt Assembly



EED Extension Unit

contains a set of radio frequency interference (RFI) -tight cavities that house relays used to select and fire electroexplosive devices (EEDs); these initiate equipment deployment early in the flight. The number of pyrotechnic events requires additional EED bridgewire actuators beyond those available in the PCU. Three EED extension units provide these actuators, each housing the EED relays in an RFI-tight cavity as with the PCU. Relay drivers and individual current limiting resistors are housed in the remaining volume of each extension unit. Primary bus voltage for the actuators is derived from the PCU.

The PCU also provides interfaces to the spacecraft telemetry and command subsystem, allowing the above PCU functions to be monitored and controlled.

Spacecraft Load Control

Each primary power bus load (except for command receivers and command units) is connected to and disconnected from the primary bus by command. To conserve battery power during eclipse, power is automatically removed from sunlight loads upon entering and automatically restored upon exiting the eclipse period. Command override of this automatic function is provided.

Commanded Load Control

Application of power to individual loads is provided by on/off control input to each load dc/dc converter and by direct power bus switching of nonelectronic loads (heaters). Command unit latching relays control application of primary bus voltage to the dc/dc converter control input and the nonelectronic loads. Command receivers and command units are permanently connected to the primary bus and cannot be disconnected by command.

Automatic Load Control

Automatic control of sunlight loads, including override, is accomplished on a function-by-function basis. The PCU performs automatic load shedding for eclipse operations via redundant load controllers that monitor the solar array and control bus currents. When the array current drops below 5 amperes, the load controller generates output signals that turn off selected loads. Upon emerging from eclipse, the controller delays until the SSA current exceeds 8 amperes, then sequentially issues signals that reapply the loads. The time duration between individual load turn-on output signals is 20 seconds. Output signals are relay contact closures sent to the associated command unit and incorporated into the turn-on command structure. Override of each load control output is provided in the associated command unit and controls turn-on/turn-off of spacecraft load groups during eclipse transitions.

Sequential Load Restoration

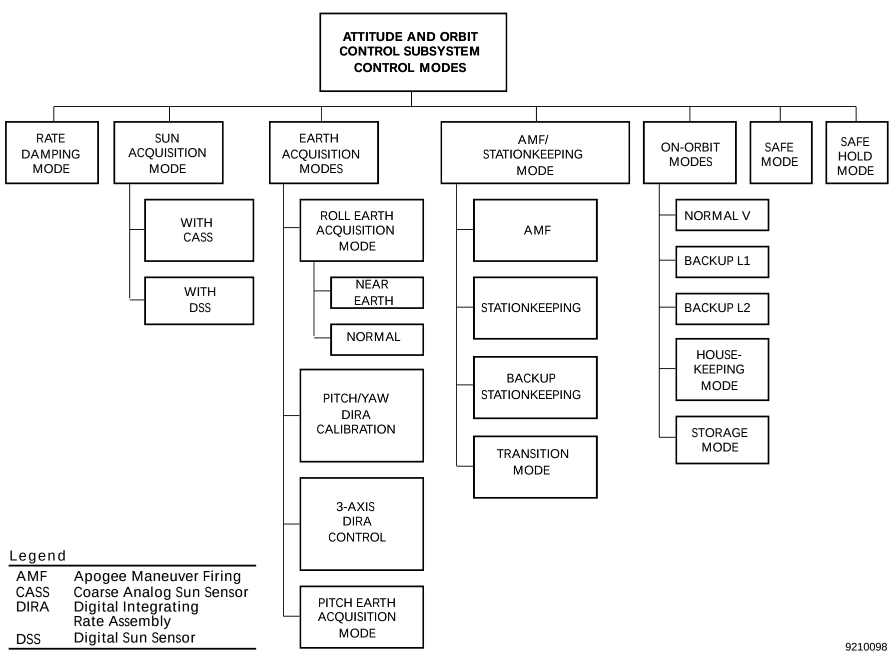
Power bus transients are minimized by sequential load restoration; precharge of input capacitors is not required.

Attitude and Orbit Control Subsystem

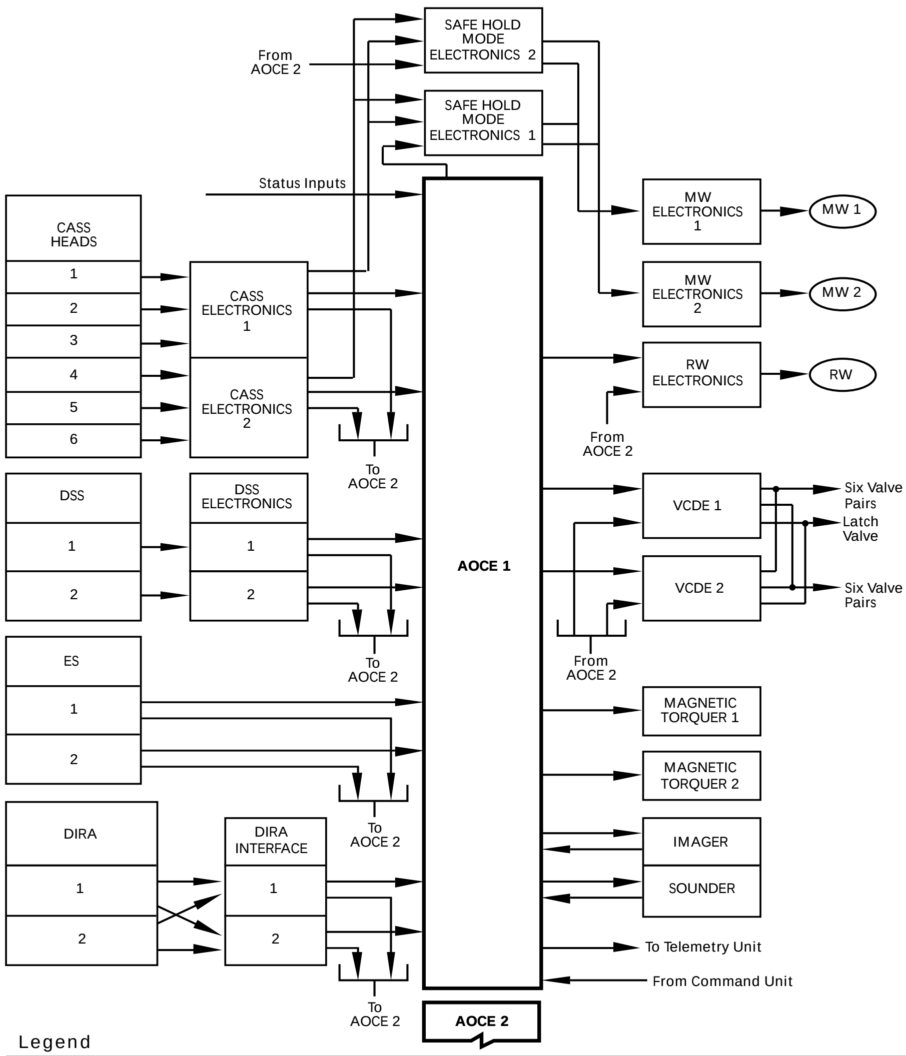
The attitude and orbit control subsystem (AOCS) provides attitude information and maintains the required spacecraft attitude during all phases of the mission, starting at spacecraft separation from the launch vehicle and throughout its operational lifetime. The subsystem consists of redundant microprocessor-based control electronics, sun and earth sensors, gyros, momentum wheels (MWs), a reaction wheel (RW), magnetic torquers, thrusters, and solar array and trim tab positioners.

Normal on-orbit attitude control operations are based on a momentum bias concept that provides precise pointing of the Imager and Sounder, communications service equipment, and scientific instruments. Control is accomplished by applying torque to internal MWs and the RW or by modulating the current applied to roll and/or yaw magnetic torquing coils. Attitude control during orbit maneuvers is provided by twelve 22-N bipropellant thrusters. Control during transfer orbit uses thrusters only, without momentum bias.

AOCS Control Modes



AOCS Functional Block Diagram



Legend

AOCE	Attitude and Orbit Control Electronics	DIRA	Digital Integrating Rate Assembly	MW	Momentum Wheel
CASS	Coarse Analog Sun Sensor	DSS	Digital Sun Sensor	RW	Reaction Wheel
		ES	Earth Sensor	VCDE	Valve Coil Drive Electronics

9210097

Attitude and Orbit Control Electronics

The attitude and orbit control electronics (AOCE) contain electronic circuitry and software to collect attitude data from a variety of sensors and to control the attitude of the spacecraft. The AOCE includes a microprocessor that performs attitude data processing and control algorithm calculations to close the loop between the sensors and the actuators. With the exception of the solar array drive electronics, the AOCE receives all commands to the subsystem as derived from the command unit, processes them, and coordinates related hardware functions. The various AOCE control modes are selected by ground command. Most telemetry signals from the AOCS are formatted in the AOCE.

The interface electronics portion of the AOCE provides appropriate gates and clocks, analog-to-digital and digital-to-analog conversion, magnetic torquer control, thruster control, and MW speed regulation. Sensors interfaced to the AOCS include coarse analog sun sensors (CASSs), digital sun sensors (DSSs), digital integrating rate assemblies (DIRAs), and infrared earth sensors (ESs). The sensors provide attitude data in the form of absolute attitude, attitude error, and rates for processing by the AOCE. These data are also formatted and telemetered to the ground to be used for mission operation checks of attitude determination.

Momentum and Reaction Wheels

Three wheels are associated with the AOCE: two 51-newton-meter-second (N·m·s) MWs and one 2.1-N·m·s yaw-axis RW. Nominal on-orbit operation uses the two MWs to provide gyroscopic stiffness and pointing control about the spacecraft pitch and roll axis. The yaw RW can be used as a redundant backup in conjunction with either MW. The two MWs are mounted with their spin axes skewed ±1.66° off the spacecraft pitch axis in the spacecraft pitch/yaw plane. In addition to pitch control, this configuration (V-mode) allows the wheels to be used to control spacecraft yaw momentum by differentially modulating the wheel speeds. Similarly, in the backup, or L-mode, modulation of the yaw RW speed is used to control yaw momentum.

Solar Array and Trim Tab Positioning

The positioning mechanisms used on orbit, as a part of the AOCS and included within its functional responsibilities but not involved in attitude determination, are the solar array drive assembly (SADA) and the solar array trim tab drive electronics (SATTDE). The SADA structurally supports the solar array wing;

rotates the wing about the spacecraft pitch axis to maintain sun pointing; and transfers power, control, and telemetry signals across slip rings at the rotary interface. Two SATTDE units are provided for redundancy.

The SATTDE: (1) processes array and trim tab encoder analog signals to derive position telemetry; (2) provides power to drive the redundantly wound stepper motor that rotates the array; and (3) drives the redundant motors that position the trim tab assembly. The trim tab compensates for the large, seasonally varying, inertial roll, solar pressure torque unbalance between the solar array and solar sail. The tab position is varied in a slow sinusoidal sequence over a year's span to also compensate for small changes in torque due to the solar flux and sun declination. A higher slew rate can be selected, by ground command, to help set the desired initial trim tab position. Magnetic torquer coils, in conjunction with the momentum bias system, provide desaturation of the angular momentum caused by slowly varying roll/yaw torques due to higher order solar radiation pressure.

Image and Mirror Motion Compensation Support

The AOCE also provides mirror motion compensation (MMC) and image motion compensation (IMC) signals to the Imager and Sounder. It accepts ground command-selected coefficients that the AOCE processor uses to determine the magnitude and timing of the servo signals for the image navigation and registration (INR) function.

Space Environment Monitor Support

The AOCS also supports the space environment monitor (SEM) payload. The X-ray positioner (XRP), a single-axis gimballed platform located on the solar array yoke, supports SEM equipment dedicated to solar studies. To maintain sun pointing throughout the seasons, the XRP contains a yoke-mounted electronics unit and a sun analog sensor that generate closed-loop drive signals to a stepper motor which in turn moves the XRP in a north/south (declination) direction to track the sun. By ground command, small east/west (azimuth) adjustments can be performed, with no effect on output power, by slewing the solar array slightly east or west of its nominal position.

Upon ground command, the XRP electronics can slew the platform to any declination position ±25° of the sun to facilitate instrument calibrations and/or background measurements. For solar X-ray imaging or other SEM tasks requiring higher azimuthal pointing accuracy, the electronics design incorporates the ability to accept control signals derived from a high-accuracy analog sun sensor, providing closed-loop sun pointing in both axes. The XRP electronics is also able to multiplex telemetry parameters of the yoke-mounted SEM equipment, housekeeping, and position data, routing it to the spacecraft telemetry and command subsystem via slip rings in the SADA.

Safe Hold Mode

A major operational safety feature of the GOES I-M spacecraft is a backup control mode that is manually commanded should loss-of-earth lock occur during normal on-orbit operations or if the interrupt safety system (ISS) is tripped. This "safe hold mode" (SHM) is designed to allow flight controllers to place the spacecraft in a safe state without the use of thrusters. Implemented via analog electronics, SHM electronics (SHME) are independent of the attitude and orbit control electronics (AOCE). SHM ensures spacecraft health and safety for long periods of time (more than 24 hours under nominal conditions, if needed) by providing sufficient solar array power and a stable thermal environment. Such extended periods allow time for technical experts to resolve spacecraft anomalies. Further, by not using thrusters to control attitude, the stored bias momentum is preserved, contributing to an enhanced level of safety and a more efficient return to normal operations.

Safe Hold may be invoked when the spacecraft loses attitude control in any wheel control mode with pitch rates within specified limits. Commanded by a ground operator (it is not automatic), SHM is capable of controlling the spacecraft to attain and hold a sun-pointing attitude starting from any orientation in pitch. Using the output of the coarse analog sun sensor electronics (CASSE) in conjunction with control wheels, SHM orients the spacecraft -X axis (or +X axis) toward the sun. After the spacecraft is brought to a stable sun-pointing mode, the solar array is slewed to face the sun and the SHME is then commanded for long-term stability. In steady-state operation, the SHM provides thermal, power and attitude safety, as well as continuous telemetry and command access except for a predictable 4-hour null per day.

As a backup mode, SHM safes the spacecraft differently than the *safe mode* which does not provide continuous solar array power or a stable thermal environment for the spacecraft. In contrast, the purpose of Safe Mode is to ensure that a continuous telemetry and command null does not occur by disabling all actuator torques and changing the commanded wheel speed bias to induce a spacecraft pitch rate of -1°/s. And because of insufficient power and intermittent telemetry, the spacecraft cannot be held in Safe Mode for long time periods. Whereas SHM is always manually commanded, the Safe Mode can be either manually commanded or automatically activated by the AOCE when certain anomalies are detected (e.g., earth sensor out of limits, processor cycle time too long). Moreover, because Safe Mode control is internal to the AOCE there is no protection against any AOCE failure.

Propulsion Subsystem

The propulsion subsystem provides the means for attitude and orbit control (AOC) and the incremental velocities at apogee required for orbit raising and final injection into geostationary orbit. The subsystem consists of one 490-N (110-lb) apogee thruster and twelve 22-N (5-lb) attitude and orbit control thrusters, using liquid bipropellants — about 694 kilograms (1530 pounds) of nitrogen tetroxide (N_2O_4) and 431 kilograms (950 pounds) of monomethyl hydrazine (MMH). These propellants are contained in two spherical titanium tanks pressurized by helium (He) supplied from a single tank.

The 490-N apogee thruster is a restartable unit designed for a minimum of three apogee maneuver firings (AMFs). During AMF, six AOC thrusters provide attitude control. The apogee thuster consists of two high-response, integral solenoid-operated valves; a thrust chamber with a 164:1 expansion area ratio nozzle; and an injector assembly.

Six of the 12 AOC thrusters are primary with the remaining six acting as backup for complete thruster redundancy. The six thrusters supply the control torques necessary to maintain proper spacecraft orientation during apogee motor firing and subsequent sun and earth acquisition maneuvers. They also support stationkeeping and on-orbit control throughout the mission.

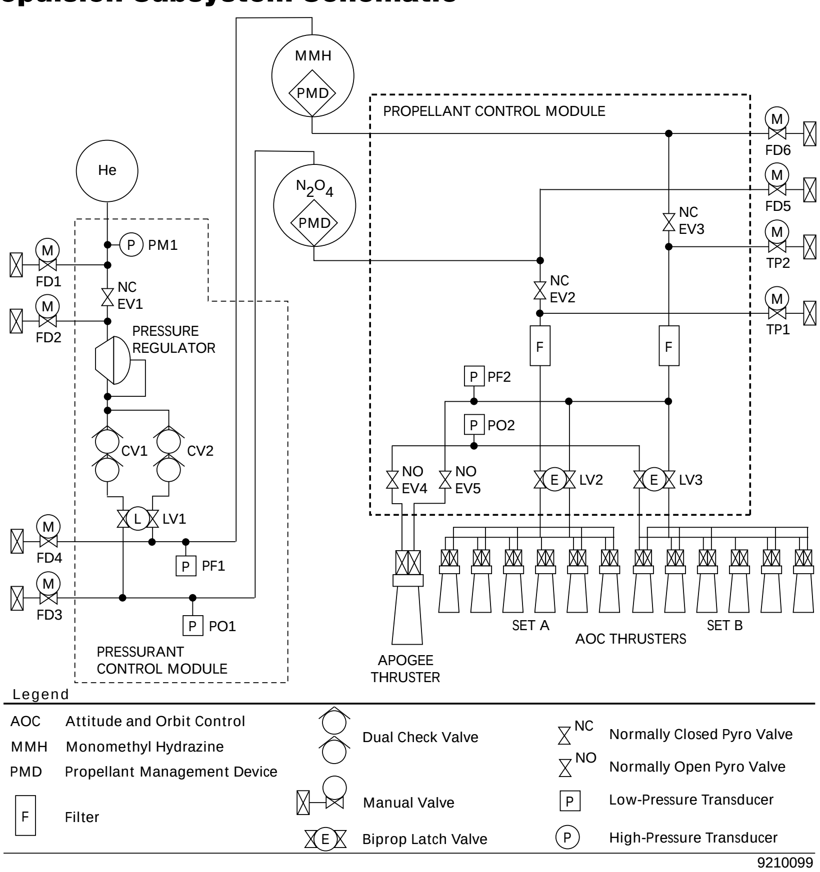
After apogee maneuvers are completed, the propulsion subsystem is operated in a blow-down mode for the remainder of the mission. A surface tension propellant management device (PMD) in each propellant tank controls the location of propellant in the zero-gravity space environment. This device enables bubble-free propellant to be supplied to the tank outlet for all thruster firings throughout the spacecraft's operational life. The stainless steel lines distributing propellant from each tank to each set of six AOC thrusters use latching isolation valves that can isolate either redundant branch of the thrusters in the event of a thruster failure, thruster valve leakage, or failed on signal. The valves are selectable by ground command for either manual closure or automatic closure by the attitude and orbit control subsystem should it detect a failure (for example, thruster remaining activated beyond a predetermined time period).

Most of the subsystem's components are mounted on the propulsion module, which is built around the central structural thrust tube. The 103-centimeter (40.6-inch) diameter spherical propellant tanks are mounted in the tube while the high pressure helium tank is externally mounted on the central thrust tube. The 490-N apogee thruster is mounted to the west end of the structure as opposed to the twelve 22-N AOC thrusters, which are hard-mounted to the central structure by brackets and panels for alignment stability. Four pitch and yaw thrusters are mounted on the east end and four others on the west end. Four roll thrusters are mounted on the south panel.

The operations required in geostationary orbit to control the spacecraft over its mission life are performed by the 22-N AOC thrusters. These operations include:

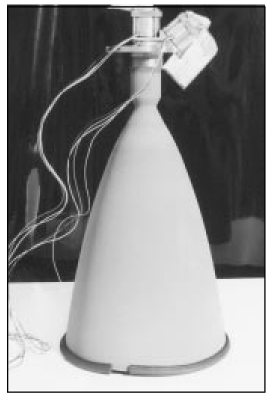
- Sun acquisition and attitude maintenance
- Earth acquisition
- Attitude control during apogee thruster firing
- Momentum wheel spinup control
- Apogee dispersion correction
- North / south stationkeeping
- East / west stationkeeping
- On-orbit attitude control operations
- Station change (relocation in geostationary orbit)
- Boost from geostationary orbit at end of life

Propulsion Subsystem Schematic



Propulsion Subsystem Components

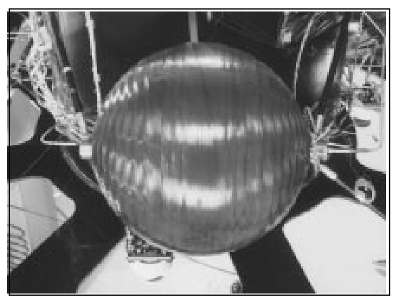
Propellant Tank



Apogee Thruster

AOC Thruster

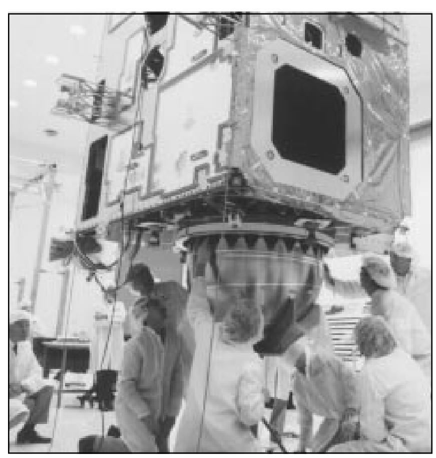
Fill and Drain Valve



Helium Pressurant Tank

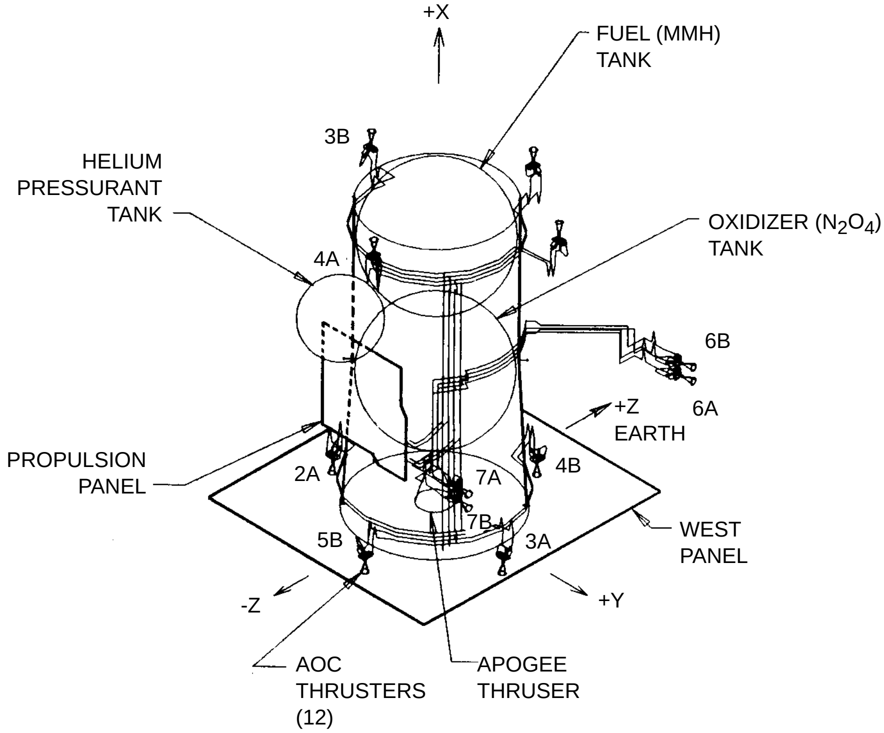
Propellant Tank Installation

Fuel (Monomethyl Hydrazine) Tank



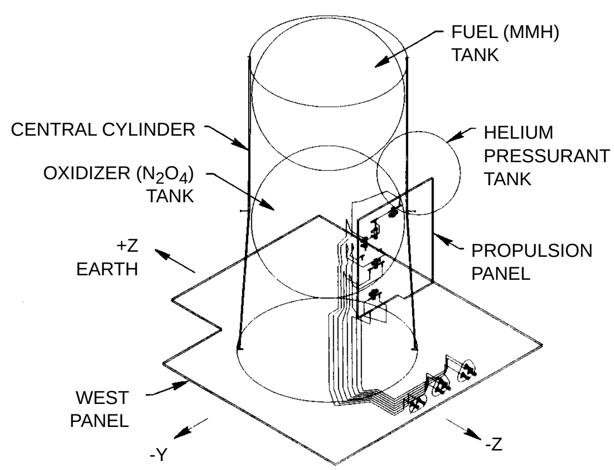
Oxidizer (Nitrogen Tetroxide) Tank

Propulsion Subsystem Thruster Line Routing



9608153

Propulsion Subsystem Fill/Drain Valve Line Routing



9608154

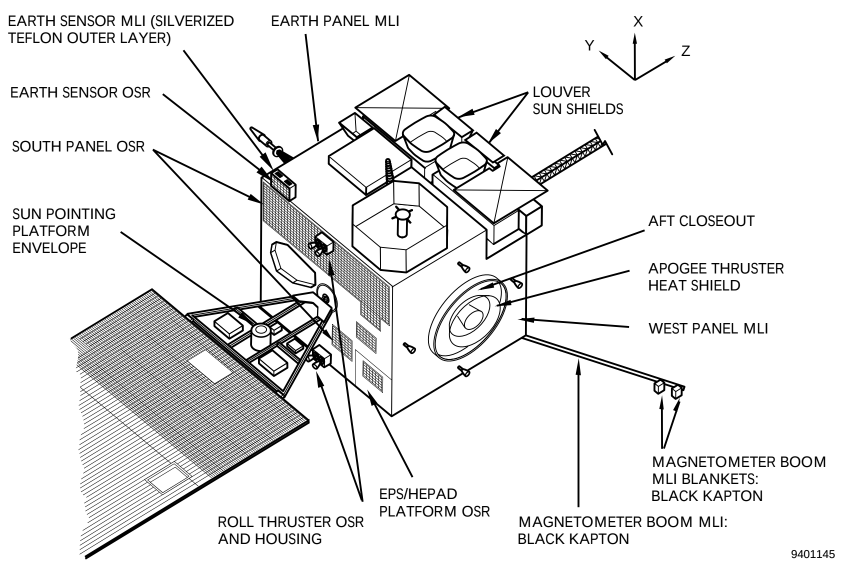
Thermal Control Subsystem

The thermal control subsystem design is configured for simple and reliable temperature control that provides the flexibility to accommodate variations in the spacecraft heat load. The approach to thermal control of the spacecraft uses conventional passive techniques such as selective placement of power dissipating components, application of surface finishes, and regulation of conductive heat paths. The passive design is augmented with heaters for certain components (particularly those with relatively narrow allowable temperature limits) and with louvers for the Imager and Sounder.

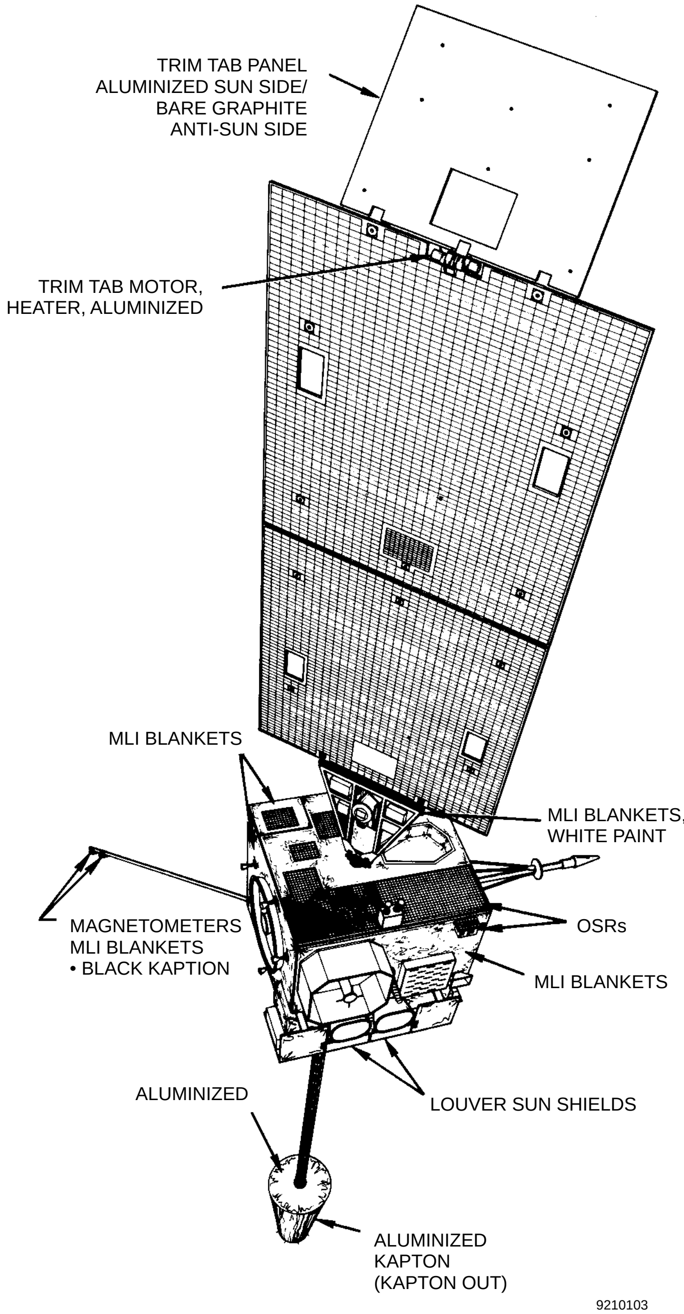
Thermal control is achieved with minimum heat transfer among the major parts of the spacecraft: the main body including energetic particles sensor / high energy particle and alpha detector; antenna; solar array including shunt; trim tab; X-ray sensor and positioner; magnetometers; Imager and Sounder; apogee thruster; and attitude and orbit control electronics. Thermal control of the main body is essentially independent of the Imager, Sounder, and appendages (antennas, magnetometers, solar array). Overall temperature control of the main body is achieved by:

- Thermal energy dissipation of components and compensation heaters in the main body
- Absorption of solar energy, particularly by the optical solar reflectors (OSRs) on the north and south panels
- Emission of infrared energy into space by the OSRs

Thermal Design - South/West/Earth Features



Thermal Design - Synchronous Orbit Configuration



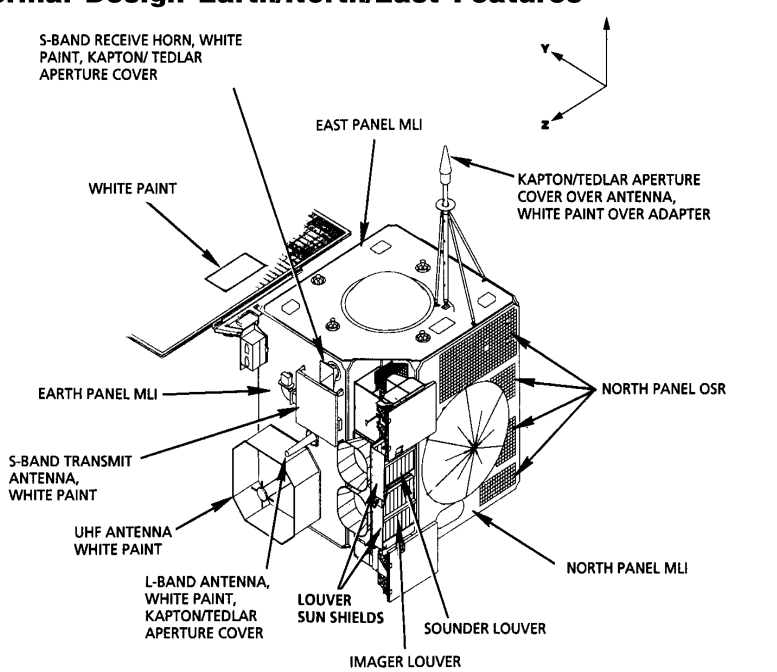
High thermal dissipators, such as S-band power amplifiers and digital integrating rate assemblies, are located on the north and south panels so that they may efficiently radiate their energy into space via heat sinks and OSRs.

Heaters are the basic means for temperature control of the spacecraft during all phases of transfer and synchronous orbit operations. The heater types and components to which they are applied are:

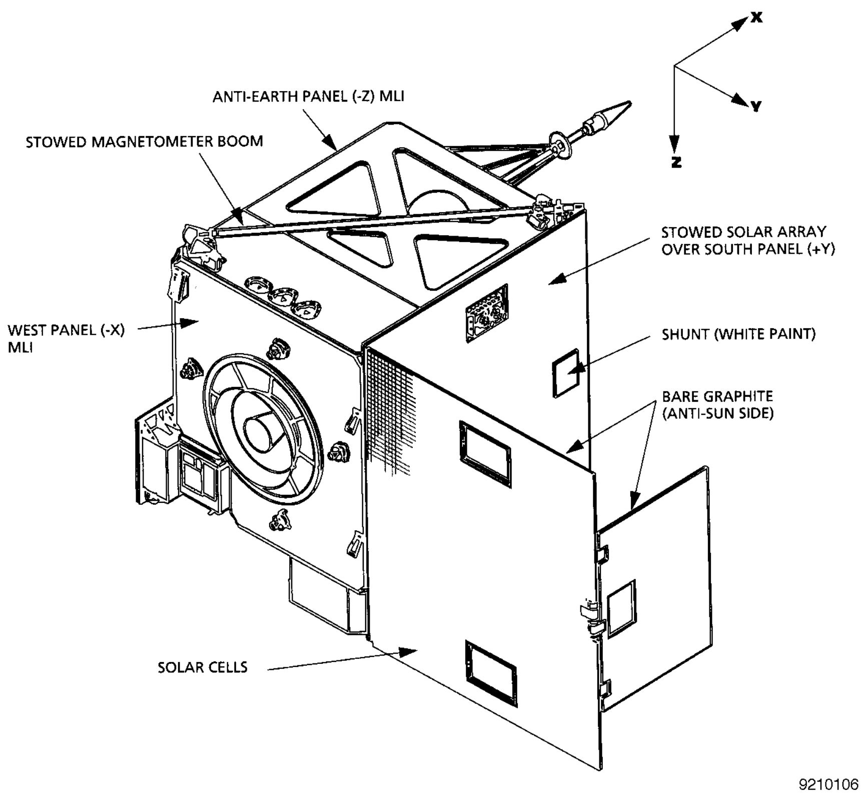
- Clayborn (adhesive-backed heater tape): fuel tank, oxidizer tank
- Tayco (patch heater): panel heaters, X-ray sensor
- Resistance wire: propulsion lines
- Dale resistors: thruster assemblies, batteries
- Proportional heaters: earth sensors, Imager, Sounder

Two types of temperature sensors are used: thermistors and platinum resistors. Thermistors are calibrated over two ranges: –40 to +70 °C with a resolution of ±1 °C, or –18 to +175 °C with a ±2 °C resolution. Thermistors are generally applied to electronic units whose typical operating range is –10 to +45 °C, with a non-operating range of –25 to +51 °C. They are also used extensively on propulsion components whose typical operating range is –3 to +165 °C. The platinum resistors have a range of –200 to +125 °C with a resolution of ±2 °C. These are used on components with wide temperature ranges such as the solar array, which has a range of –165 to +70 °C.

Thermal Design Earth/North/East Features



Thermal Design - Transfer Orbit Configuration



Imager and Sounder Thermal Control

Optical and radiometric performance of the Imager and Sounder are maintained throughout the 24-hour orbit by a combination of louver cooling and electrical heating. Thermal control is divided into two primary areas. First is thermal control for the sensor module as defined by the scan mirror and telescope assembly along with the optical bench or telescope baseplate and all structural sidewalls. Second is thermal design of the detector radiant cooler assembly; this is treated separately from the first inasmuch as these two components are intended to be adiabatic (thermally isolated) from each other; the thermal performance of one has little or no effect on the other.

Optical performance is maintained by restricting the total temperature range. Radiometric performance is maintained by limiting the temperature change between views of cold space (rate of change in temperature). Thermal control also contributes to channel registration and focus stability.

The basic thermal design concepts include:

- Maintaining the instruments as adiabatic as possible from the rest of the spacecraft structure.
- Controlling the temperature during the hot part of the synchronous orbit diurnal cycle (when direct solar heating is received into the scanner aperture) with a north-facing radiator whose net energy rejection capability is controlled by a louver system.
- Providing makeup heaters within the instruments to replace the infrared energy loss to space through the scanner aperture during the cold portion of the diurnal cycle.

Additionally, a sun shield is provided around the scan aperture (just outside the instrument field of view) to block incident solar radiation into the instruments, thus limiting the time in a synchronous orbit day when the scanner can receive direct solar energy. Uncontrolled temperature variations are reduced by the sun shield around the scan cavity opening, a passive automatic louver-controlled cooling surface, and electrical heating. Electrical heat decreases temperature excursions during the cold part of the daily cycle, but increases the average temperature. To obtain lower temperature ranges, louver-controlled cooling is provided during the direct sunlight portion of the orbit. A sun shield is installed on the earth end of the louver system to reduce incident radiation.

Multilayer insulation (MLI) blankets are applied on the outside of all but the north side of the instruments. The cover over the radiation cooler is designed to provide thermal protection of the radiation cooler patch during transfer orbit. This cover has MLI blankets on both sides and is deployed onto the earth face after reaching synchronous orbit.

Deployment Mechanisms and Structures

Several spacecraft appendages are stowed during launch and later deployed at various mission phases, starting soon after separation of the spacecraft from the launch vehicle. The deployable appendages are:

- Solar array
- Magnetometer boom
- Solar sail and boom
- Imager and Sounder radiant cooler covers

These deployments are initiated by ground commands and occur at three different time periods:

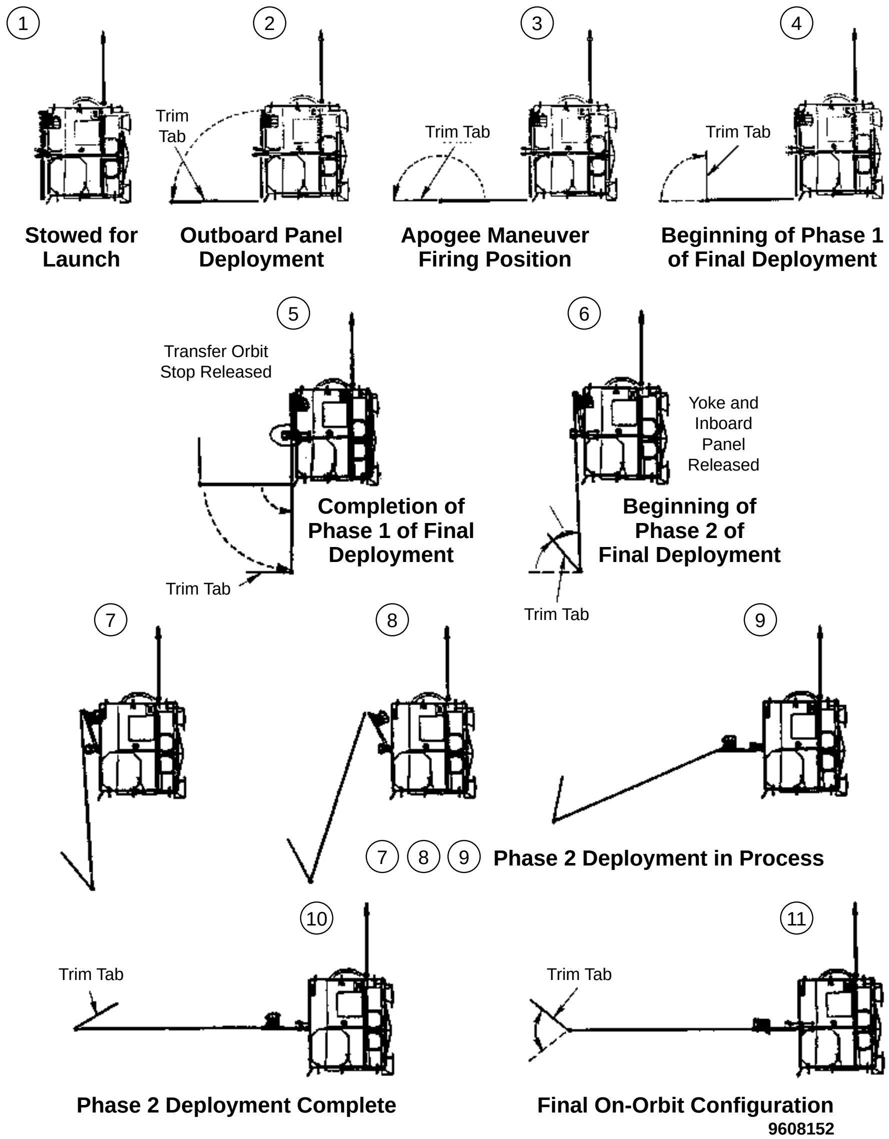
- First, early in the transfer orbit, about 90 minutes after launch, the outer solar panel is partially deployed to about 90° from its launch position, exposing its solar cells to the sun and providing power for the spacecraft during the transfer-orbit phase.
- Later, shortly after completion of all apogee maneuvers required for orbit raising and early in the near-geosynchronous drift orbit:
  – The magnetometer boom is deployed.
  – The final phases of solar array deployment are completed.
  – After the spacecraft is in wheel control mode, the solar sail and boom are extended.
- Finally, after 14 to 21 days of outgassing in space, the Imager and Sounder radiant cooler covers are deployed.

All of the deployable appendages are released by pyrotechnically driven cutters (electroexplosive devices, EEDs) that cut a tensioned cable or rod holding the appendage in its stowed, launch position. The cutters are fired by ground command. All cutters are fully redundant with independent knives, firing circuits and commands. If the first cutter does not release the appendage, the redundant cutter may be used later.

Solar Array

The solar array consists of two panels covered with solar cells on one side and a yoke that holds the panels away from the spacecraft to avoid shadows on the cells. The yoke is mounted to the shaft of the solar array drive assembly (SADA) on the body of the spacecraft. When fully deployed, the solar array extends from the south side of the spacecraft where the SADA continuously rotates to keep the solar cells oriented towards the sun. The panels and yoke are hinged together so they can be folded against the south side of the spacecraft during launch; they are held in place by latches with tensioned cables.

Solar Array Deployment Sequence



Upon ground command, the panels are released by cutting the cables with an EED. The panels are deployed to their operational positions in three discrete steps: transfer orbit, phase 1 of synchronous orbit, and phase 2 of synchronous orbit. Deployment is driven by the unwinding of redundant prewound torsion springs at each hinge. Switches on each hinge line indicate, by telemetry, when the panels are near their deployed position.

Transfer Orbit

The initial, transfer-orbit deployment occurs soon after launch with the spacecraft in the off-axis, sun acquisition control mode. In this mode, the sun warms the hinges prior to deployment. The outer panel is released by firing a cutter that severs the primary holddown cable. With the cable slack, six spring-driven latches rotate clear of holddown rods freeing the outer panel to deploy. The inner panel and yoke are retained in their stowed position by a secondary holddown system which is similar to the primary system. After rotating about 90°, the outer panel is stopped at its transfer-orbit position by removable mechanical stops that engage the hinges. In this position the cell side of the outer panel is exposed to the sun, thus generating power to support the spacecraft electrical load. The trim tab, which is stowed behind the outer panel, is slewed 180° so that it is inplane with the deployed, outer panel.

Phase 1 Synchronous Orbit

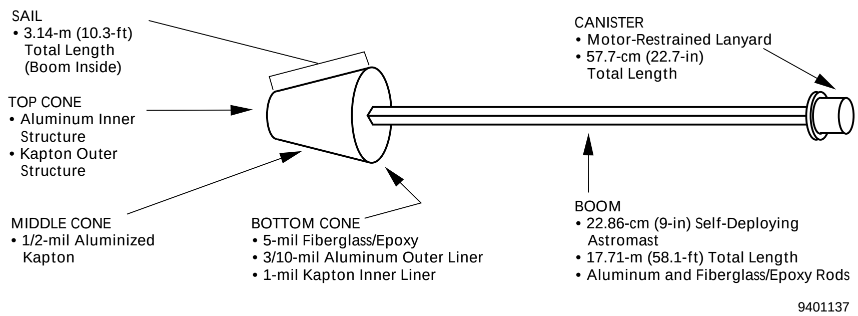
After orbit raising with the spacecraft near its geosynchronous orbit position, solar array deployment is completed. The spacecraft is in the pitch earth acquisition control mode with the sun warming the outer panel and hinges. The cable retaining the transfer-orbit stops is cut. This allows the stops to move out of the way thus freeing the outer panel to complete its 180° deployment and to latch. The trim tab must be slewed to its 90° position at the start of this phase to reduce moment loads on the trim tab motor at latch-up.

Phase 2 Synchronous Orbit

In DIRA attitude reference control mode, the spacecraft is rotated about the yaw axis to allow the sun to warm the inner panel and the yoke-to-inner-panel hinges. After warming for about 40 minutes, the secondary holddown/release retaining the yoke and inner panel is fired, cutting a cable and freeing the entire wing to deploy. The yoke rotates 90°, while the inner and outer panels rotate together 180° (with respect to the yoke). The SADA-to-yoke and yoke-to-inner panel hinges latch, completing the solar array deployment.

Solar Sail and Boom

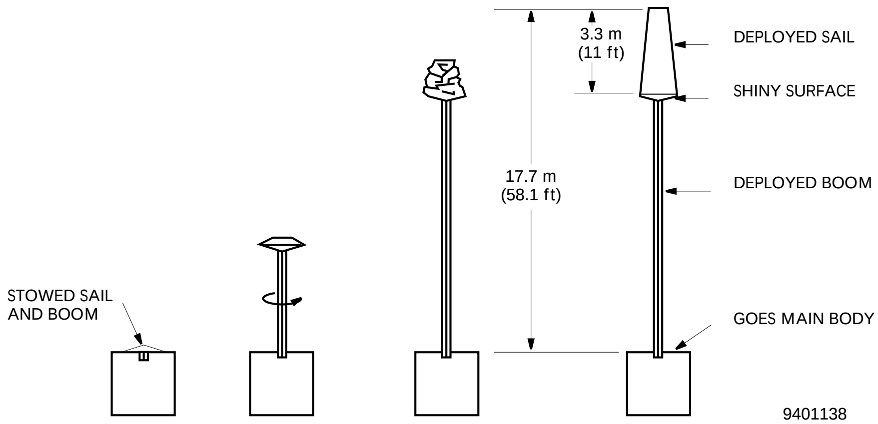
Design Description, Solar Sail and Boom



To balance the torque caused by solar pressure on the solar array, a solar sail is deployed from the north side of the spacecraft, opposite the solar array. The sail is mounted on an extendable boom (Astromast). The sail and boom are lightweight collapsible structures, mounted on the north face of the spacecraft. During launch, the boom is stowed in a canister and the sail is folded against the north face. The sail and boom are held in their stowed positions by a metal tie rod that extends through the stowed boom assembly and into a redundant pyrotechnic cutter. Severing the rod with the cutter releases the sail to deploy and the boom to extend to a total length of 17.7 meters (58.1 feet).

When released the coiled boom jumps a few centimeters (about 1 inch) from its canister and is restrained by a lanyard fastened to the outer end of the boom. To obtain a slow, safe boom extension, the lanyard is payed out from a reel controlled by a redundant dc motor. At the end of its extension, the boom latches into a stiff configuration and the sail is fully deployed. Deployment switches indicate the initial jump-out and the start of motor-controlled extension. During extension, up until the boom latches into its stiff configuration, the motors may be stopped to halt deployment, though it is not possible to reverse the motors and retract the boom.

Solar Sail and Boom



Two redundant, miniature, dc-torque motors control the extension rate of the boom. When the boom is completely deployed, a redundant pair of switches on each motor actuate, disconnecting motor power which stops the motors. If any pair of switches fails to operate, the motor continues to run as long as the execute signal is being sent. If this happens, the end of the lanyard will disconnect from its reel to prevent damage.

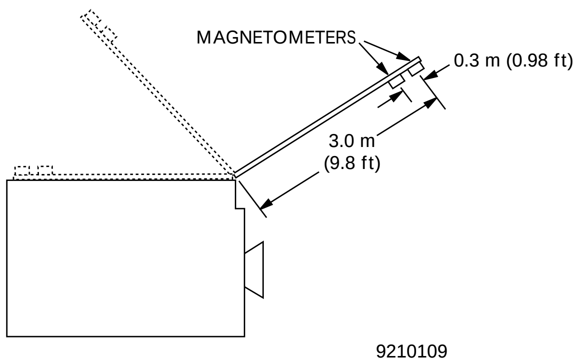
Magnetometer Boom

The magnetometer boom moves two redundant magnetometers away from the spacecraft main body to reduce interaction with the spacecraft's magnetic field. The boom is a thin-walled graphite tube, 3 meters (9.8 feet) long and 5 centimeters (2 inches) in diameter. One end of the boom is hinged to the northwest corner of the anti-earth panel. The other end of the boom holds the two magnetometers. One magnetometer is 0.3 meter (1 foot) inboard of the other. The boom and the magnetometer support brackets are made of graphite epoxy for low mass and small thermal distortions.

During launch, the magnetometer boom is stowed diagonally across the anti-earth panel. It is held by a single holddown/release device located between the two magnetometers. A tensioned cable preloads the boom against the holddown bracket. Upon ground command, a pyrotechnically actuated cutter severs the cable releasing the boom for deployment. Deployment is driven by two redundant torsion springs located on either side of the hinge. The boom rotates 135° from its stowed to its deployed position. If needed, a redundant cutter is available.

In the deployed position, the boom is locked in position by a roller/latch-arm that follows a cam during deployment and enters a close-tolerance slot at the deployed position. The roller and latch arm are held against the cam and pushed into the slot by multiple leaf springs. Once so engaged, any further motion or vibrations will not cause the roller to leave the slot. Deployment switches indicate release from the holddown device and latching in the deployed position.

Stowage and Deployment of Magnetometer Boom



Imager and Sounder Radiant Cooler Covers

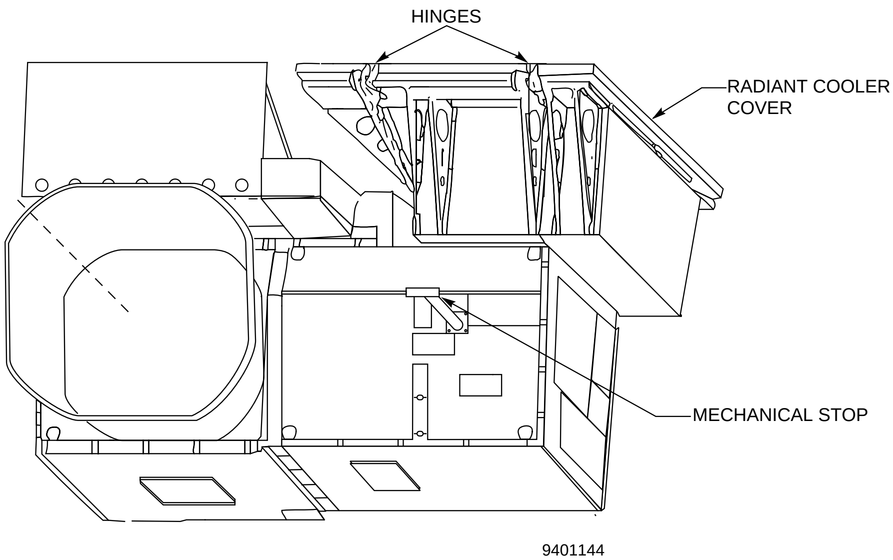
The two cooler covers, one each for the Imager and Sounder, serve to:

- Protect the infrared detectors from direct sunlight during orbit insertion
- Protect the radiant cooler and emitters from contamination during launch and orbit insertion
- Reduce outgas heater requirements at acquisition of geosynchronous orbit

Each cooler cover consists of an aluminum/honeycomb sandwich panel, two spring-driven deployment hinges, a holddown/release device (pyrotechnically actuated), and thermal blankets on the inside and outside. Each cooler cover attaches to the cooler assembly. The hinges attach to the external shroud and are isolated thermally by fiberglass standoffs. The holddown/release device attaches to the emitter panel. Pretensioning devices for the holddown cable are mounted on the external shroud.

During launch, each cooler cover is held in the stowed position by a cable that preloads the cover to the emitter panel. Upon ground command, pyrotechnically actuated cutters (one at each cooler cover) sever the cables, simultaneously releasing both covers for deployment. Deployment is driven by redundant torsion springs located on the hinges. Each cover rotates 270° from its stowed to its deployed position. At the deployed position, each cover hits a mechanical stop and is held against the stop by the residual deployment spring torque and by a Velcro fastener. If needed, a redundant cutter is available.

Instrument with Cover in the Closed Position



Microswitches on the deployment hinges indicate stowed and fully deployed positions. Deployment is also indicated by changes in temperature of the lower cooler housing, the radiator, and the infrared detector.

Spacecraft Structure

The GOES spacecraft main body consists of a graphite-fiber-reinforced plastic (GFRP) central cylinder that houses the propellant and oxidizer tanks of the propulsion subsystem, a propulsion panel, momentum wheel panels, external panels mounted to longerons, and struts supported by the central cylinder. The panels are of sandwich construction with aluminum honeycomb core and either GFRP or aluminum faceskins. The struts are made of GFRP tubes and the longerons are aluminum.

The solar array structure consists of the yoke, two solar panel substrates, and the trim tab panel. The yoke structure is made of GFRP beams. The solar panel substrates are of sandwich construction with lightweight aluminum honeycomb core and thin GFRP faceskins. A thin Kapton film (3-mil thick) is bonded to the solar cell side of the substrates to electrically insulate the solar cells from the graphite faceskin. The solar cells are bonded to the Kapton film. The trim tab panel is of sandwich construction with a lightweight aluminum honeycomb core and thin GFRP faceskins.

Operations Ground Equipment

The GOES ground segment consists of three parts: ground station radio frequency (RF) equipment, telemetry and command system, and the payload processing and spacecraft operations system. The ground station RF equipment is located at the SOCC and CDA facilities. Telemetry and command are provided by the GOES I-M telemetry and command system (GIMTACS), also located at the SOCC and CDA. Payload processing and spacecraft operations support is provided by the operations ground equipment (OGE).

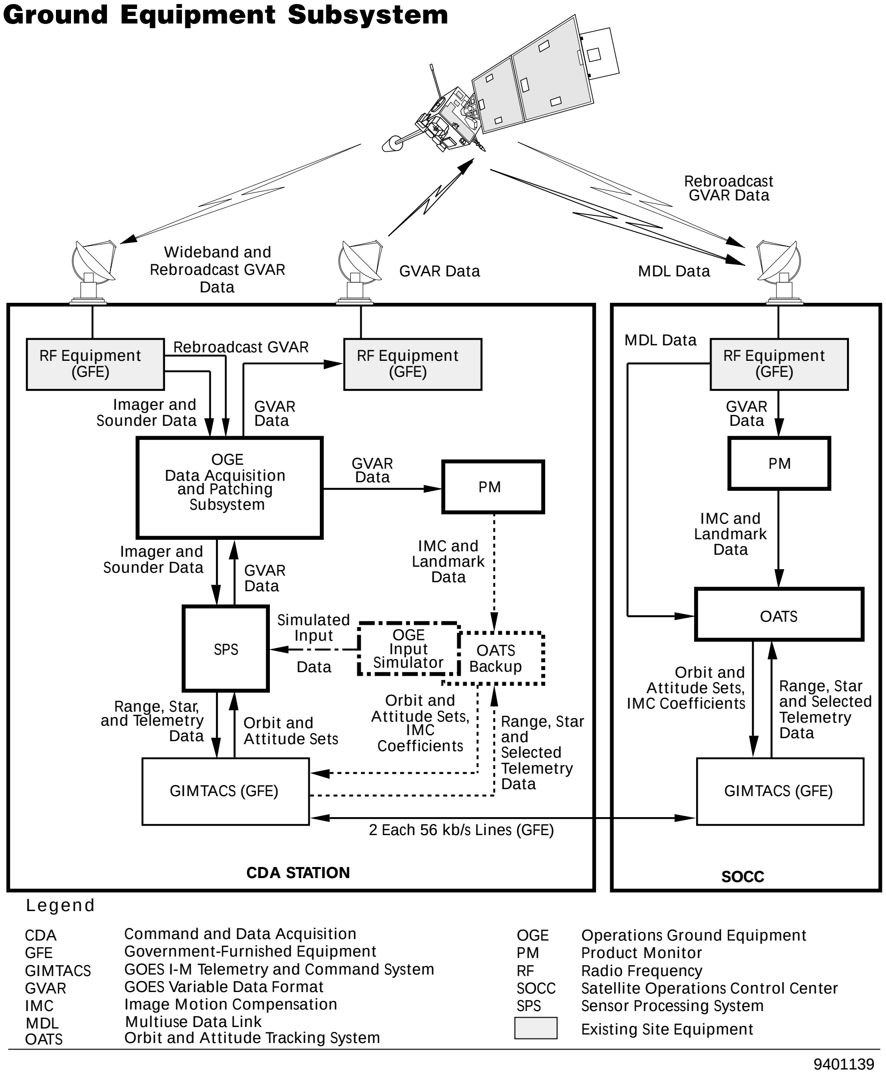
The OGE performs several important functions that support on-orbit operations of the GOES spacecraft:

- Radiometric calibration, visible data normalization
- Earth location and gridding
- Range, star location, and landmark measurements
- GVAR data product monitoring
- Calibration database management
- Orbit and attitude determination
- Image motion compensation (IMC) coefficient generation
- Star command generation, scan frame command generation
- Raw instrument data simulation
- Diagnostic telemetry data processing
- Stationkeeping maneuver planning and command generation
- On-board propellant remaining estimation
- Daily trim tab angle prediction

Five major functional elements perform the above activities: OGE data acquisition and patching subsystem (ODAPS), sensor processing system (SPS), product monitor (PM), orbit and attitude tracking system (OATS), and OGE input simulator (OIS). These elements are located at the Command and Data Acquisition (CDA) Station, Wallops, Virginia, and the Satellite Operations Control Center (SOCC) at Suitland, Maryland. Communications among several of these functional elements are provided by the GOES I-M telemetry and command system (GIMTACS), which is part of the OGE network and provided as Government-furnished equipment (GFE).

Primary inputs to the OGE are the Imager and Sounder sensor data streams and the multiuse data link (MDL). Primary outputs are the processed data relays for the Imager and Sounder data streams in the GOES variable data format (GVAR), one GVAR-formatted output data stream being generated for each instrument downlink data stream. The GVAR data stream is transmitted to its corresponding spacecraft for relay to principal users, as well as to the CDA Station and SOCC for OGE internal purposes. Internal OGE uses of GVAR data are primarily for monitoring the quality of the processed data (CDA Station and SOCC), determining spacecraft range for use in orbit and attitude determination (CDA),

Ground Equipment Subsystem

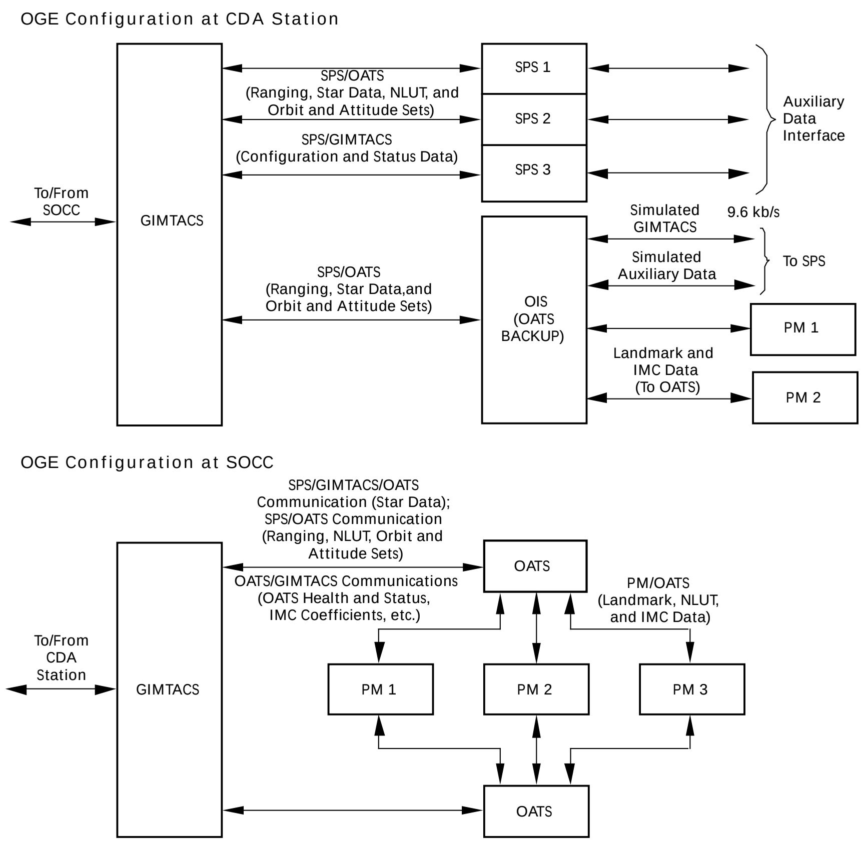


and extracting landmark images as part of the orbit and attitude determination (CDA and SOCC).

Data from the MDL are received at the SOCC and processed by the OGE. The MDL is received as an independent data link, containing angular displacement sensor (ADS) and digital integrating rate assembly (DIRA) data from the spacecraft attitude control subsystem. These data are ingested and processed by the OGE to diagnose dynamic interactions between the Imager and Sounder instruments and the spacecraft.

The OGE at the CDA Station consists of three SPSs, two PMs, and an OIS which may be switched to the backup OATS. Communications among the three SPSs and the two SOCC OATS computers or the CDA OATS backup (OIS) are via GIMTACS. Message traffic between the SPSs and the OATS computers consists primarily of ranging data, star measurement transmissions to the OATS, and orbit and attitude data from the OATS. GIMTACS receives spacecraft orbit and attitude related data, such as IMC coefficients, from the OATS. The OATS receives selected spacecraft telemetry (for example, DIRA data) to support spacecraft stationkeeping functions by GIMTACS. In addition, each SPS receives configuration messages from the GIMTACS, and both the SPS and OATS transmit status to the GIMTACS.

Operations Ground Equipment Subsystem Configuration



Legend

CDA	Command and Data Acquisition	OGE	Operations Ground Equipment
GIMTACS	GOES I-M Telemetry and Command System	OIS	OGE Input Simulator
IMC	Image Motion Compensation	PM	Product Monitor
NLUT	Normalization Look-Up Table	SOCC	Satellite Operations Control Center
OATS	Orbit and Attitude Tracking System	SPS	Sensor Processing System

9210115

Communications from the PM to OATS consist of landmark measurements, IMC and servo error messages, and normalization look-up table (NLUT) data sent to SPS via OATS and GIMTACS. In the SOCC configuration, each PM provides synchronous data communication interface to each OATS. In the CDA Station configuration, synchronous communication interfaces are provided from each PM to the OIS to support the OATS backup function.

Operations Ground Equipment

GVAR Transmission Format

The GVAR data transmission format was developed to allow full use of the capabilities of the advanced, three-axis-stabilized spacecraft while retaining as much commonality as possible with receiving equipment presently in use from earlier spin-stabilized GOES spacecraft. The GVAR format is based on the operational visible and infrared spin scan radiometer atmospheric sounder (VAS) mode AAA format, which consists of a repeating sequence of 12 fixed-length equal size blocks. The transmission of these blocks is synchronized with the spin rate of the earlier GOES spacecraft; that is, one complete 12-block sequence per rotation of the satellite.

The GVAR transmission sequence consists of 12 distinct blocks numbered 0 through 11. Blocks 0 through 10 are transmitted when an Imager scan line is completed. Block 10 is followed by a variable number of block 11s, according to what data are available for transmission.

GOES I-M Variable (GVAR) Data Transmission Format

GVAR Data Block Type

	Document	IR 1	IR 2	Visible 1	Visible 2	Visible 3	Visible 4	Visible 5	Visible 6	Visible 7	Visible 8	Sounder and Auxiliary Data
GVAR Block Number	0	1	2	3	4	5	6	7	8	9	10	11
Word Size, Bits	8	10	10	10	10	10	10	10	10	10	10	6, 8, 10
Field Length, Words	8,040	68 – 21,008	51 – 15,756	20 – 20,960	20 – 20,960	20 – 20,960	20 – 20,960	20 – 20,960	20 – 20,960	20 – 20,960	20 – 20,960	10,720/ 8,040/ 6,432
Number of Records	—	4/block	3/block	1/block	1/block	1/block	1/block	1/block	1/block	1block	1/block	1 – 8
IR Detector Data, Words	—	5,236	1 – 5,236	4 – 20,944	4 – 20,944	4 – 20,944	4 – 20,944	4 – 20,944	4 – 20,944	4 – 20,944	4 – 20,944	–

Each GVAR block has
- 10,032-bit synchronization code
- 720-bit header
- N-bit information field
- 16-bit cyclic redundancy check

Block Characteristics

• Period	15.25 – 104.6 ms
• Synch Length	10,032 bits
• Header Word Length	8 bits/word
• Header Length	90 words (720 bits) (Triple Redundant)

Scan Characteristics

• Period	Variable
• Block/Imager scan	11
• Bit Rate	2,111,360 b/s

- Blocks 0 and 11 have fixed length information field of 64,320 bits
- Blocks 1 through 10 have variable length information fields directly dependent on width of scan, with minimum information field of 21,440 bits
- A single Imager scan generates blocks 0 through 10 in sequence
- Blocks 0 through 10 may be followed by any number of block 11s (0–N) depending on data available; in priority order, the next block(s) transmitted will be:

1. Next Imager scan	Blocks 0 through 10
2. Imager compensation and servo errors	One block 11
3. Sounder compensation and servo errors	One block 11
4. Imager telemetry statistics	One block 11
5. Imager spacelook statistics and data	Six block 11s
6. Imager calibration coefficients and limits	One block 11
7. Imager electronic calibration statistics and data	Two block 11s
8. Imager blackbody statistics and data	Two block 11s
9. Imager visible NLUT	Two block 11s
10. Imager star sense data	Nine block 11s
11. Sounder scan data	2 to 523 block 11s
12. Sounder telemetry statistics	One block 11
13. Sounder spacelook statistics and data	Five block 11s
14. Sounder calibration coefficients and limits	Two block 11s
15. Sounder electronic calibration statistics and data	Three block 11s
16. Sounder blackbody statistics and data	Five block 11s
17. Sounder visible NLUT	Nine block 11s
18. Sounder star sense data	Nine block 11s
19. GIMTACS text messages	One to two block 11s
20. SPS text messages	One block 11
21. Auxiliary data	One to N block 11s
22. Fill data	One block 11

Block 0 and all varieties of block 11s are fixed, equal-length structures. Blocks 1 through 10 vary in length according to the length of the Imager scan line. The smallest possible block size (blocks 1 through 10 for scan widths less than 1.9°) has a total length of 32,208 bits, while the largest block size (block 1 for a 23°-wide scan) is 262,288 bits. The maximum values for blocks 1 through 10 correspond to the specified maximum scan width of 19.2°. Scan widths up to 23° are possible with either instrument, although radiometric and pointing accuracies degrade at widths above 19.2°. The GVAR format handles scans wider than 19.2° in order to support special tests that may be desired following spacecraft launch. During normal operations, the 19.2° specified limit represents the upper boundary.

OGE Data Acquisition and Patching Subsystem

The ODAPS provides intermediate frequency (IF) level processing and routing/patching of data streams to/from the CDA Station equipment for two operational spacecraft. The OGE also utilizes the VAS interface electronics (VIE), currently in use at the CDA Station, to provide reception of one GVAR data stream used as a backup. The VIE also provides redundant switching necessary to select GVAR data streams from the three available for routing to the PMs. ODAPS also provides IF demodulation and bit synchronization functions.

Operations Ground Equipment Data Acquisition and Patching Subsystem

Data Handling Summary

ODAPS Data	Sensor Data	GOES Variable Data Uplink	GOES Variable Data Downlink
Source	CDA Station equipment; SPS intermediate frequency switch	SPS/uplink interface output	CDA Station receive equipment/intermediate frequency splitter
Source format	Unbalanced asynchronous quadraphase shift keying modulated, 64.4 MHz intermediate frequency; Q Ch Imager data at 2.6208 Mb/s; I Ch Sounder data at 40 kb/s	Biphase shift keying modulated; 67.7 MHz intermediate frequency; 2.11136 Mb/s	Biphase shift keying modulated; 65.7 MHz intermediate frequency 2.11136 Mb/s
ODAPS processing	Demodulate; bit synchronize	None	Demodulate; bit synchronize; frame synchronize into GOES BUS format
Destination	SPS/SD interface input	CDA Station transmit equipment/intermediate frequency switch	VAS interface electronics selector unit input, SPS/uplink interface ranging input

Sensor Processing System

The SPS, the functional element responsible for real-time processing of Imager and Sounder data, receives auxiliary data products from the GFE, combining them with processed data. The combined data in GVAR format are transmitted to users via the corresponding GOES spacecraft. GVAR data, organized in user friendly segments, are calibrated, earth-located, and gridded (Imager only). OIS to SPS serial communications interfaces that allow simulation of GIMTACS communications and auxiliary data products, are provided for OGE system testing.

To accomplish its tasks the SPS performs:

- Data ingest, including frame synchronization, decommutation by channel, detector scan alignment, and alternate scan line reversal.
- Infrared calibration for conversion of raw data to engineering units.
- Computation of Imager and Sounder space look, blackbody, electronic calibration and instrument telemetry statisitcs for inclusion in GVAR data stream.
- Visible sensor normalization for stripping the visible data.
- Coregistration function which applies correction factors (computed by PM) to Imager visible sensor data to ensure visible and IR data alignment.
- Imager gridding to convert geopolitical latitude and longitude grid points to scan line and pixel coordinates as a function of spacecraft orbit and Imager attitude.
- Earth location to convert Imager and Sounder instrument coordinates to latitude and longitude as a function of spacecraft orbit and instrument attitudes.
- Data formatting to create the GVAR data stream.

To support the orbit and attitude determination function of the OATS subsystem, the SPS:

- Performs spacecraft range measurements using the GVAR data stream round trip propagation time; these measurements are sent via GIMTACS to OATS for orbit determination.
- Performs star crossing event measurements by processing Imager and Sounder star view data; these are also sent to OATS via GIMTACS for attitude determination.
- Extracts periodic IMC and servo error data from the sensor data for use in the IMC quality check function performed by OATS; formatted messages of these data are periodically sent to OATS via the GVAR through the PM subsystem.

Further, SPS sends wideband telemetry data, including command register echo information extracted from the Imager and Sounder data streams and scan position to GIMTACS every 10 seconds, as long as valid telemetry is being processed in the SPS. The telemetry message data consists of the latest values received for the telemetry words extracted from the telemetry blocks of the Imager turn-around sequence and telemetry words extracted from the Sounder block.

Product Monitor

PMs are located at the CDA and the SOCC. Under normal operational circumstances, the PMs at the CDA Station perform only the monitoring function while the PMs at the SOCC perform the OATS support functions as well as the monitoring functions. In the backup operational configuration, with the OATS resident at the CDA Station, the PM roles at the CDA Station and SOCC are reversed. The PM also supports processed data quality monitoring and system troubleshooting, and the orbit and attitude determination function performed by the OATS.

In support of the orbit and attitude determination function, the PM provides landmark identification by storing, displaying, and registering small areas of visible Imager data (visible Sounder and infrared Imager as backup) defined as landmark sectors. Landmark registration is performed by a semiautomatic correlation of selected landmark sectors to previously stored landmark sectors referred to in landmark correlation chips. Once correlated, landmark measurement data in the form of earth location coordinates are sent to the OATS. The PM also captures the IMC and servo error data included in the GVAR data by the SPS and passes it to OATS, which provides quality checks of the image navigation and registration function performed on board the spacecraft.

Product Monitor

Orbit and Attitude Tracking System

The OATS performs three major functions in support of mission operations. The primary function is to provide daily computational support for implementing the INR process. This support consists of a closed-loop sequence that:

- Ingests star, range, and landmark observations
- Determines spacecraft orbit and Imager and Sounder attitudes
- Determines station and sensor intrusions; predicts eclipses
- Computes image motion compensation
- Determines star observation coordinates

The second major function of OATS is to provide the capability to plan, generate command data, and evaluate stationkeeping and repositioning maneuvers. The evaluation utilizes processes to estimate the onboard propellant remaining and to calibrate the propulsion system.

Finally, OATS requests, accepts, and processes telemetry data to support other functions such as determination of command data for daily operations of the trim tab, evaluation of AOCE data to verify and calibrate IMC, verification and calibration of MMC, DIRA calibration for stationkeeping and reacquisition support, and evaluation of thruster firing data.

The outputs generated by OATS are:

- Orbit and Imager/Sounder attitude coefficients for the SPS
- IMC coefficients uplinked to the spacecraft via GIMTACS
- Star view command data to support Imager and Sounder star sense and sequence operation
- Maneuver planning information and spacecraft stationkeeping command data
- Estimates of onboard propellant remaining
- Command data for daily operation of the trim tab
- Orbit and station events prediction
- Sensor intrusion predictions
- Scan frame coordinates conversion
- IMC calibration factors
- MMC calibration factors

Not a part of OATS, though resident in its computer, is the dynamic interaction diagnostic (DID) function. This function processes and displays telemetry data contained in the MDL for analyzing possible dynamic interactions between the Imager and Sounder instruments and the spacecraft. Telemetry ingested for diagnostic purposes includes ADS and DIRA angular data (roll, pitch, and yaw), instrument servo error, and solar array and instrument events. These spacecraft data are available on GOES I and one other spacecraft to be identified later in the program.

OGE Input Simulator

The OIS provides simulated Imager and Sounder data streams and outputs OGE message communications and auxiliary products in support of OGE integration and test. In addition, the OIS serves as a diagnostic tool and a backup computer for the OATS at the SOCC during the operational phase. The OIS provides the ODAPS with Imager and Sounder data, which may be manually patched into any one of three SPS inputs.

The GVAR simulator outputs a simulated GVAR data stream in support of OGE and GOES user system integration and testing. The outputs are in both NRZ-S serial baseband and GOES bus frame synchronized serial data. The latter output allows a direct interface to the PMs and does not require ODAPS to provide GVAR frame synchronization.

Spacecraft Mission Profile

To reach the required on-station location in geostationary orbit (station acquisition), the GOES spacecraft undergoes five distinct mission phases:

- Launch Phase: From Atlas I lift-off to spacecraft separation.
- Transfer Orbit Phase: Spacecraft separation to end of apogee maneuver firing (AMF) 1.
- Phasing Orbit Phase: End of AMF 1 to end of AMF 2.
- Trim Orbit Phase: End of AMF 2 to station acquisition.
- Synchronous Orbit Phase: Station acquisition to end of operational life.

The nominal chronological sequence of orbit raising for a nominal transfer orbit consists of AMF 1, AMF 2, AMF 3, an apogee adjust maneuver (AAM) at perigee, and one or more trim maneuver firings (TMFs), as required. During AMFs the spacecraft is maintained in a nonrotating configuration by the attitude and orbit control subsystem (AOCS) using the earth as a reference. Throughout each drift period, between firings, the spacecraft is returned to and maintained in the sun-acquisition mode, with rotation about the X-axis, to supply solar array power.

Ground Stations

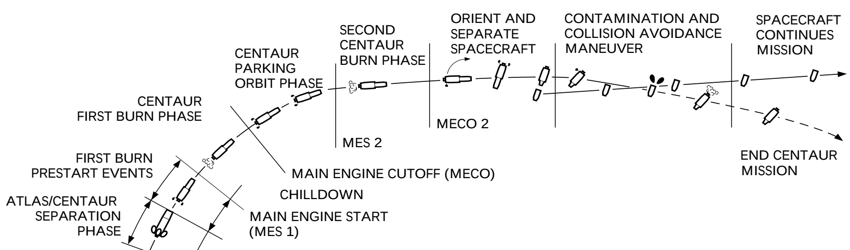
Various ground centers and tracking stations are involved throughout the mission phases:

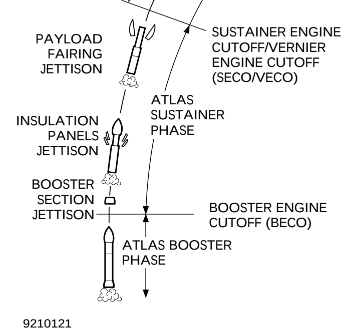
- Deep Space Network (DSN) stations at Canberra, Australia; Madrid, Spain; and Goldstone, California, support orbit raising maneuvers with Goldstone acting as backup to the command and data acquisition (CDA) station when the spacecraft is in synchronous orbit. The NASA antenna at Wallops, Virginia, provides telemetry and command (T&C) support during orbit raising.
- Indian Ocean Station (IOS), an Air Force remote tracking station, is used for initial spacecraft T&C functions and to support the transfer orbit whenever this station is available.
- Command and Data Acquisition (CDA) station located at Wallops, Virginia, houses the telemetry and command transmission system (TACTS), portions of the operations ground equipment (OGE), and GOES I-M telemetry and command system (GIMTACS). The CDA performs spacecraft telemetry acquisition, formatting, and command transmission.
- Satellite Operations Control Center (SOCC) houses the orbit and attitude tracking system (OATS), part of the OGE, and GIMTACS. The SOCC is the prime control center for all mission phases. This station is also capable of receiving processed instrument data in GOES variable (GVAR) data format and multiuse data link (MDL) diagnostic data.

Launch and Injection

The GOES I-M spacecraft are launched from Cape Canaveral Air Force Station Launch Complex 36B by a General Dynamics Atlas I. During the launch phase the spacecraft batteries supply power to support thermal control and T&C functions, yielding a battery depth of discharge of about 8.3% at separation. The spacecraft is separated from the Centaur stage with a spin about its major moment of inertia axis (Z axis) of about 1.2 revolutions per minute (7° / s) at the time of ground station acquisition-of-signal. The +Z axis of the spacecraft is pointed generally toward earth, providing an effective T&C signal for first acquisition-of-signal. Atlas I injects the spacecraft into a highly elliptical supersynchronous orbit (48,789 kilometers (30,316 miles) apogee radius, 6545 kilometers (4067 miles) perigee) to begin the transfer orbit phase. The maximum time from lift-off to separation is about 35 minutes, nominal being about 28 minutes.

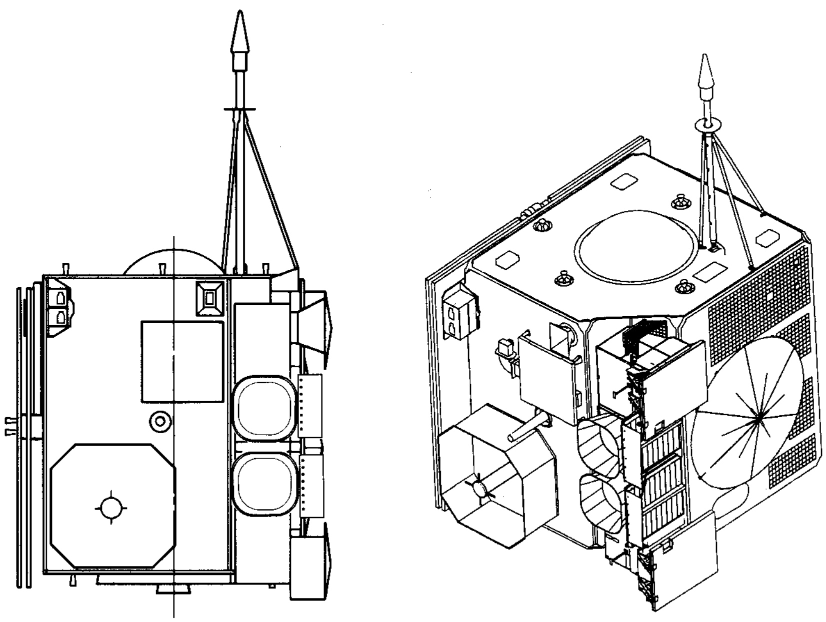
Atlas I Major Mission Events





Event	Basis	Approx Time from Lift-off (seconds)
Lift-off (L / O)	In motion	0
Roll program	L / O + 2 to L / O + 15 seconds	2 to 15
BECO	5.5 g	156
Booster section jettison	BECO + 3 seconds	159
Insulation panel jettison		181
Payload fairing jettison	3 σ qv <1135 W / m²	217
SECO		265
Atlas / Centaur separation	SECO + 2 seconds	267
MES 1		289
MECO 1		605
MES 2	Guidance	1454
MECO 2	Hohmann apogee	1549
Spinup Option	Spacecraft separation	1551
Separation		1741

Launch Configuration

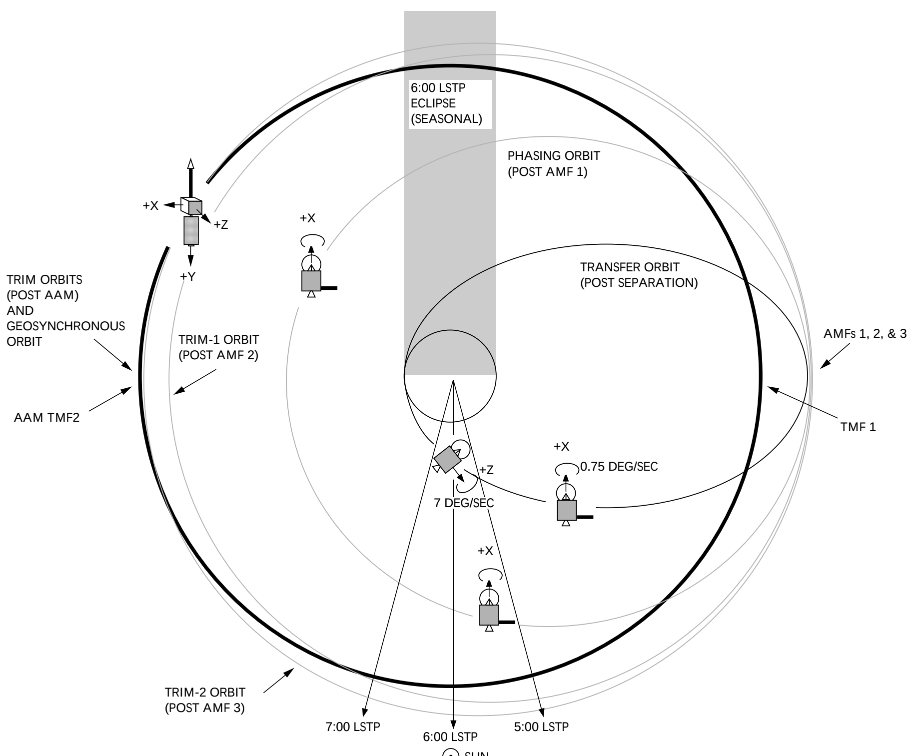


Transfer and Phasing Orbits

The spacecraft attitude at injection (onto the transfer orbit) is oriented so as to maximize continuous T&C coverage by both primary and backup ground stations. The effective spacecraft T&C antenna pattern is a cardioid with a maximum ±130° look angle from the +X axis. T&C visibility from the ground is obtained when the spacecraft's elevation angle with respect to the ground station local horizon is greater than 5° and the ground station is within the spacecraft T&C antenna pattern. During transfer and phasing orbits, redundant and near-continuous coverage is provided by the five T&C stations (Goldstone, Wallops, Madrid, Canberra, and Indian Ocean), except for unavoidable 1- to 2-hour outages at perigee due to the spacecraft's low altitude.

After acquisition-of-signal by the Indian Ocean Station is established and a command link verified at the beginning of the transfer orbit phase, the spacecraft is configured by ground command to prepare for initial off-axis sun-acquisition mode. Using the attitude and orbit control (AOC) thrusters, the AOCS orients the spacecraft's west face toward the sun. The outboard panel of the solar array wing is then pyrotechnically released to face the sun and provide solar power to operate the spacecraft subsystems and recharge the batteries. For a nominal timeline of 90 minutes from launch to panel deployment, the battery reaches a depth of discharge of about 20%. This level allows sufficient time for possible

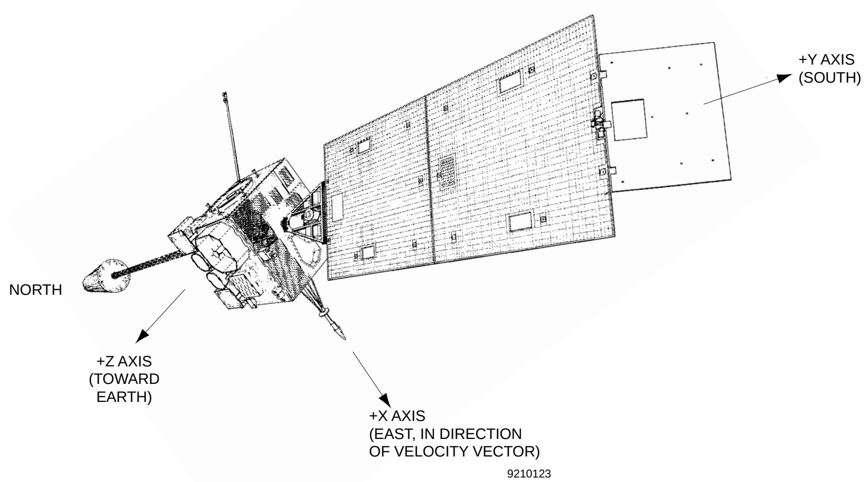
Spacecraft Orbit Geometry



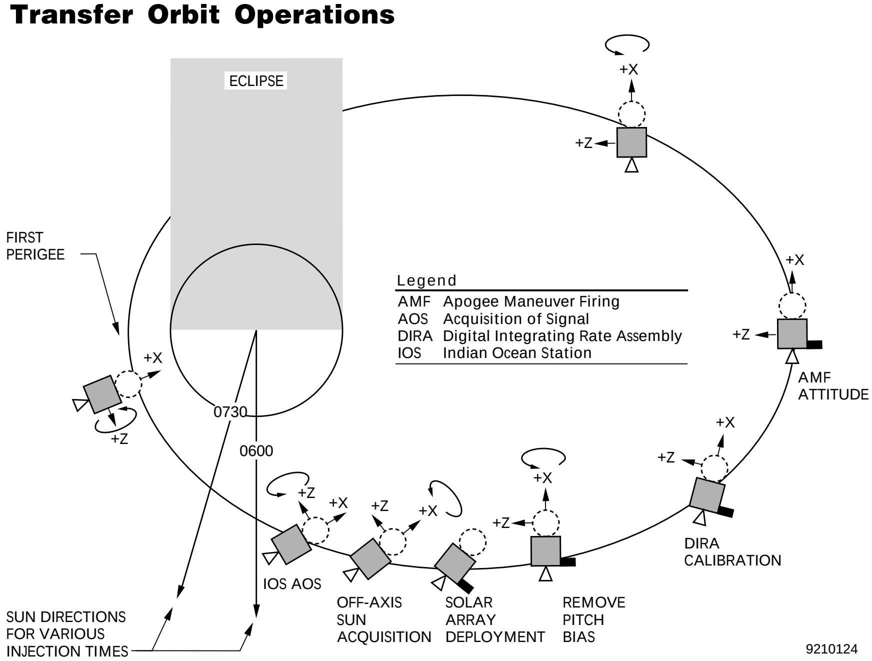
Legend

AAM	Apogee Adjust Maneuver
AMF	Apogee Maneuver Firing
LSTP	Local Solar Time at Perigee
TMF	Trim Maneuver Firing

Body-centered Axes of the



Transfer Orbit Operations



delays and contingencies, while remaining within the safe limit of 60% depth of discharge. If there is any delay in deploying the outer solar array panel, adequate station coverage for continuous T&C operations is provided since the spacecraft remains in view of the Indian Ocean and Canberra stations throughout this period.

The 490-N apogee thruster is used to target the spacecraft into the proper trim orbit, that is, correct apogee radius, inclination, and ascending node. The apogee motor firings minimize north / south stationkeeping (NSSK) propellant expenditures by targeting the apogee thruster for the correct ascending node and inclination that places the spacecraft on the proper NSSK schedule. Propellant consumption is minimized by using the higher specific impulse apogee thruster rather than the 22-N AOC thrusters.

Optimum targeting also involves splitting AMF into three parts. The first places the spacecraft into a high drift rate orbit, phasing the spacecraft to the on-station longitude. The second AMF corrects orbit dispersions induced by the first, placing the spacecraft into a near-synchronous trim orbit. Each AMF is retargeted before the burn to compensate for any apogee thruster dispersion or pointing errors incurred during the previous AMF. The AAM uses the apogee thruster to reduce the apogee altitude to the geosynchronous altitude. Any remaining orbit dispersions after the three AMFs and the AAM are corrected in the trim orbit using the 22-N AOC thrusters. The spacecraft arrives on station after several revolutions in the trim orbit, about 15 days after separation from the Atlas I.

The transfer orbit parameters cited here are for a nominal orbit with AMF 1 occurring at fourth apogee. The time to arrive on station is 15.2 days for the nominal transfer orbit apogee radius of 48,789 kilometers (supersynchronous). At apogee four, the spacecraft is visible to the Goldstone and Wallops stations. At the completion of AMF 1 the spacecraft is in the phasing orbit, which continues through AMF 2. Nominal AMF 2 occurs at the sixth apogee, about 38 hours after AMF 1. AMF 2 occurs two orbit revolutions after AMF 1, ensuring visibility from Madrid with Indian Ocean for backup.

Transfer orbit eclipse durations depend primarily on the launch date and launch window. Battery management, pre-eclipse and post-eclipse, is to be determined based on a firm launch date, and command sequences are to be included in the final sequence of events. During eclipse, the sun reference is lost though the spacecraft continues in the sun-oriented roll mode in the absence of the solar radiation pressure and aerodynamic perturbations. After eclipse, the sun acquisition sequence is automatically reinitiated to reacquire the sun.

Trim Orbit

The trim orbit phase begins after AMF 2 and continues until the spacecraft is on station, defined as the time when the spacecraft is in synchronous orbit positioned at 90° west longitude ready for on-orbit testing, and all launch vehicle and apogee thruster dispersions have been corrected. This phase nominally lasts for about 10.7 days. After AMF 2, the spacecraft is nominally at 13° east longitude, drifting west at 20° per day toward station location. With the spacecraft at 27° west two days later, the apogee thruster is fired (AMF 3), raising the perigee radius, increasing the drift, and also correcting any previous AMF pointing errors. One and a half revolutions later, the apogee thruster is fired at perigee to change the apogee radius (apogee adjust maneuver), and reduce drift rate. One and a half revolutions after that, near apogee six, trim maneuvers by the AOC thrusters arrest spacecraft drift. Depending upon the actual orbit parameters at this phase of the mission, additional TMFs may be required. The major operations performed during the trim orbit phase, following AAM are:

- Deploy magnetometer boom; calibrate spacecraft residual magnetic field
- Complete deployment of solar array
- Spin up momentum and reaction wheels
- Deploy solar sail and boom

On-Station/Synchronous Orbit

At this point the synchronous orbit phase begins and the spacecraft is checked for proper performance before entering service at either of two assigned locations. At the checkout station the orbit apogee and perigee radii will respectively be 155 kilometers (96.3 statute miles) above and below the geosynchronous radius of 42,164 kilometers (26,199 statute miles). By international agreement for the GOES system, two spacecraft orbital positions have been assigned: 75° and 135° west longitudes. From these two vantage points, roughly over Ecuador and the Marquesas Islands respectively, the GOES Imager and Sounder instruments are able to provide coverage of both Atlantic and Pacific Oceans. The major operations performed upon station acquisition are:

- Deploy Imager and Sounder cooler covers
- Activate space environment monitor equipment
- Start spacecraft checkout

Normal on-orbit operations entail periodic stationkeeping maneuvers that keep the spacecraft within a 0.5° inclination about the equator and within ±0.5° of the on-station longitude. These maneuvers are needed because of several forces that produce small changes over a short period of time: interactive effects of the sun's and moon's gravity, solar pressure variations, and the fact that the earth's mass is distributed and does not act as a single point. Stationkeeping is performed by a particular set of the 22-N thrusters selected by ground command for manually controlled firing. The OATS software determines the needed AOC thruster combination and thruster firing durations. The AOCE maintains spacecraft attitude throughout the EWSK firing using the earth sensor and by actuating other control thrusters. (The DIRA is used for NSSK.)

Nominal Orbit Parameters

Apogee Maneuver Firing 1 at Fourth Apogee

Parameter	Orbit Type						
	Transfer	Phasing	Trim				
			Post-AMF 2	Post-AMF 3	Post-AAM	Post-Trim	Geo-Synch
Perigee, radius (km)	6,544.5	23,408	38,641	41,909	41,909	42,009	42,164
Apogee, radius (km)	48,789	48,789	48,789	48,789	42,419	42,319	42,164
Inclination (degrees)	27.0	4.4	0.25	0.5	0.5	0.5	0.5
Period (hours)	12.7	19.0	25.3	26.7	23.4	24.0	24.0
Drift (degrees/revolution east)	168.7	74.8	-20.0	-41.6	0.01	0.0	*

*Spacecraft drift induced by solar, earth and lunar gravitational effects and solar pressure corrected by stationkeeping maneuvers

On-Orbit Mission Operations

As a platform for the payload and in support of the mission's objectives, the spacecraft undergoes or executes numerous operational events: configuring subsystems, orienting attitude, bus maintenance, compensating for environmental effects or changes, adjusting operational parameters, performing maneuvers. The operational events occurring on orbit are grouped into two categories: daily operations and periodic operations. Daily operations include events scheduled to occur, as determined in advance, based on normal spacecraft operations and adjusted to account for the seasons. Periodic operations entail infrequent but recurring events such as stationkeeping, space environment monitor calibration, attitude and orbit control subsystem adjustments, sun / moon intrusion, station relocation, etc., which occur over the spacecraft's life.

Typical Daily Operations

Daily operations for the Imager and the Sounder are structured to satisfy the meteorological needs of the National Weather Service. These operational scenarios for the GOES spacecraft Imager and Sounder also must comply with spacecraft state-of-health requirements and operational constraints. The initial, "Day-l", operational scenarios for GOES spacecraft feature one of three modes for the Imager: full disk, routine, and rapid scan. The Sounder has three modes which operate concurrently with the three Imager modes. The mode being used at any given time is related to the severity of the meteorological activity being observed.

Full Disk Mode

The Imager full disk mode consists of a full disk scan of the earth followed by star looks and a blackbody calibration. This sequence is repeated every half hour. The full disk scan is changed to an extended northern hemisphere scan once every 6 hours. This allows sufficient time to perform the 10-minute spacecraft housekeeping activities.

The corresponding Sounder operations follow a summer mode (June to November) or winter mode (December to May) schedule. This is a 6-hour repeated schedule. The schedule starts with a full regional northern hemisphere sounding repeated three times on one-hour centers; then a full regional southern hemisphere sounding (winter mode) or a limited regional sounding and a mesoscale sounding (summer mode), followed by a limited regional sounding. Spacecraft housekeeping activities are then performed to complete the 6-hour schedule. The soundings are interrupted for star looks each half hour and for blackbody calibrations.

Routine Mode

The Imager routine mode is a repeated 3-hour sequence. The 3 hours start with a full disk scan, followed by the half-hour sequence of an extended northern hemisphere scan, a continental U.S. (CONUS) scan, and a southern hemisphere-south scan which is repeated five times. The last southern hemisphere-south scan is omitted every 6 hours to allow for spacecraft housekeeping . Star looks and blackbody calibrations are performed every half hour.

The Sounder performs the same summer mode or winter mode schedule as during the Imager full disk mode.

Rapid Scan Mode

The Imager rapid scan mode is a repeated 3-hour sequence. The 3 hours start with a full disk scan. Then the half-hour sequence of a northern hemisphere scan is followed by a CONUS scan, a small southern hemisphere scan, a second CONUS scan, star looks and a blackbody calibration, and five more CONUS scans. The last two CONUS scans are omitted every 6 hours to allow for spacecraft housekeeping activities.

The Sounder warning mode is performed in conjunction with the Imager rapid scan mode. This is a 6-hour repeated schedule. The schedule starts with a limited regional sounding, then nine repeated mesoscale soundings. Then another limited regional sounding is performed, followed by eight mesoscale soundings, and then the spacecraft housekeeping activities. The soundings are interrupted for star looks each half hour and for blackbody calibrations.

Scan Sector Boundaries and Durations

The typical Imager and Sounder scan sector boundaries and scan durations are for the operational scenarios described above. The boundaries assume that the GOES-East satellite subpoint will be located at 75 degrees west longitude.

GOES-East Imager Scan Sectors – Boundaries and Duration for Day-1 Scenarios

(Subsatellite Longitude: 75° west)

					Scan Duration (minutes)					
Frame Name	Boundaries (Lat/Long)				Scan Clamp		9.2 second Space Clamp		36.6 second Space Clamp	
	North	South	West	East	West	East	West	East	West	East
Full Disk	Earth Edge				26.26	26.26	47.00	47.00	27.66	27.66
Northern Hemisphere	60˚N	0˚N	112˚W	30˚W	11.21	10.73	15.03	14.76	9.86	9.84
Northern Hemisphere-Extended	65˚N	20˚N	112˚W	30˚W	16.33	15.64	21.81	21.48	14.32	14.28
Southern Hemisphere-South	20˚N	55˚N	116˚W	23˚W	6.12	5.83	8.60	8.44	5.61	5.59
Continental U.S. (COUNUS)	60˚N	14˚N	112˚W	64˚W	5.99	7.40	6.23	6.65	4.70	4.75
Southern Hemisphere-Small Sector	0˚	20˚S	100˚W	80˚W	3.12	4.25	2.23	2.40	1.85	1.88

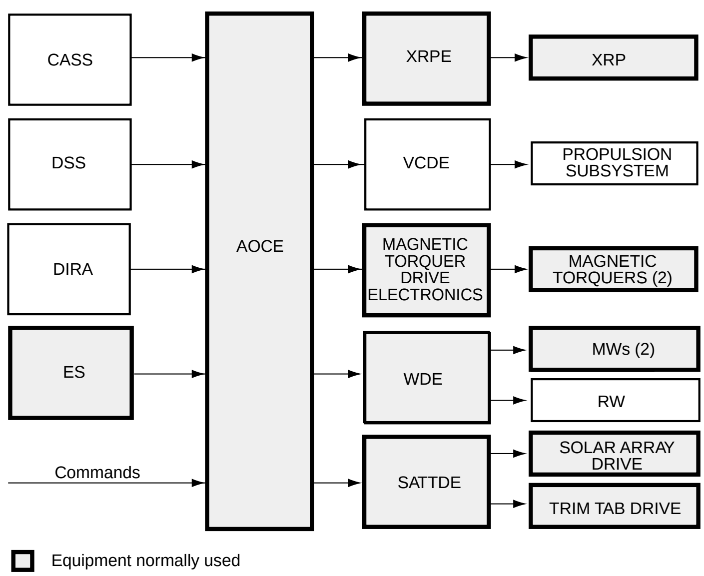
GOES-East Sounder Scan Sectors – Boundaries and Duration for Day-1 Scenarios

Frame Name	Boundaries				Scan Duration (minutes)
	North	South	West	East	
Full Regional-NH	51.5°N	23.3°N	120°W	63.6°W	52.1
Full Regional-SH	20°S	50°S	130°W	75°W	52.0
Limited Regional	50°N	26°N	118°W	66°W	39.8
Mesoscale-CONUS	43.6°N	26.8°N	106.2°W	87.9°W	12.2
Mesoscale-Tropics	24.7°N	15.0°N	70.2°W	44.5°W	12.1

(Subsatellite Longitude: 75° west)

Note: The mesoscale sectors included in this table are representative of a number which could be defined.

AOCS On-Orbit Control During Imager and Sounder Operations



Legend

AOCE	Attitude and Orbit Control Electronics	SATTDE	Solar Array Trim Tab Drive Electronics
CASS	Coarse Analog Sun Sensor	VCDE	Valve Coil Drive Electronics
DIRA	Digital Integrating Rate Assembly	WDE	Wheel Drive Electronics
DSS	Digital Sun Sensor	XRP	X-Ray Positioner
ES	Earth Sensor	XRPE	X-Ray Positioner Electronics
MW	Momentum Wheel		
RW	Reaction Wheel		

9210127

Imager/Sounder

Imaging and sounding are performed at predefined scan coordinates. If scan frame coordinates are required, the GOES I-M tracking and command system (GIMTACS) requests the orbit and attitude tracking system (OATS) to provide scan frame conversion from scan lines and pixel number (or longitude and latitude) to cycles and increments for use by the Imager and Sounder. GIMTACS also specifies the stepping mode of the Sounder as part of the request. In response, OATS converts scan coordinates to cycles and increments for the Imager and Sounder and sends scan start and stop coordinates and scan start and stop times. These data are then used by GIMTACS in the command message to the Imager and Sounder.

For its daily schedule, GIMTACS requests from OATS star view command parameters of a specified duration. OATS responds with star view coordinates in cycles and increments, dwell times, and look start time for each instrument. Start time is the time at which the pulse command, star-sense start is received at the instrument.

Spacecraft

The daily operational procedure for spacecraft subsystems (for example, telemetry and command, electrical power, attitude and orbit control, propulsion, thermal, and communications) generally involves monitoring and control of operational parameters (such as bus voltages, battery reconditioning, roll/yaw controller gain, etc.). Housekeeping operations, such as trim tab slew and solar array adjustments, that generally allow controlled changes in spacecraft attitude are performed in the manual mode during intervals of 10 minutes each, at 05:50, 11:50, 17:50 and 23:50 hours, spacecraft local time. During each eclipse season the 05:50 universal time coordinated (UTC) housekeeping period will be replaced by periods at 03:50 and 07:50 UTC to accommodate earth sensor single chord operations. Bus maintenance, such as heater configuration changes, redundancy switching, etc., are performed as required.

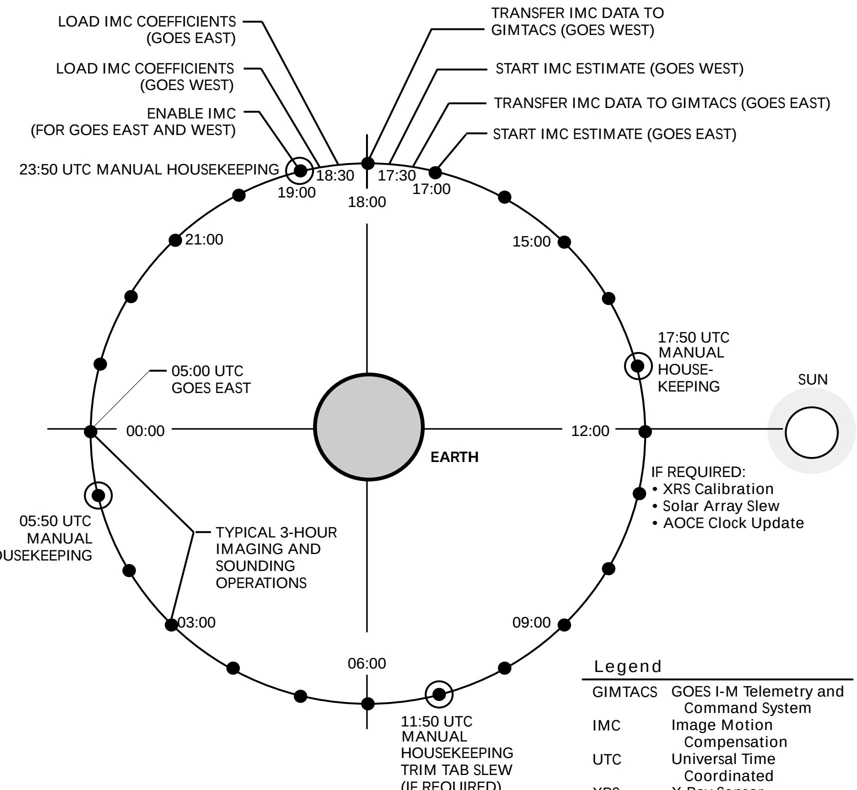
Space Environment Monitor (SEM)

The SEM is on and operational once the spacecraft reaches station. Calibration of the X-ray sensor (XRS) is performed periodically during the housekeeping period as required. Magnetometer and energetic particles sensor (EPS)/high energy proton and alpha detector (HEPAD) calibrations are performed periodically and can be done at any time without affecting Imager and Sounder operations.

Image Navigation and Registration

To support image navigation and registration (INR), a parent image motion compensation (IMC) coefficient set is generated every day and uploaded to the spacecraft. This is scheduled during the final hour of each day so the data set can be enabled at 00:00 hours UTC for the next operational day. Normally this set is effective throughout a coregistration period of 24 hours. OATS provides GIMTACS with the new IMC coefficient data set to be uploaded, the message normally being sent once a day and enabled at a time specified by OATS. IMC

Daily Operation Schedule (Typical)



Legend

GIMTACS	GOES I-M Telemetry and Command System
IMC	Image Motion Compensation
UTC	Universal Time Coordinated
XRS	X-Ray Sensor

Notes:

1. Typical 3-hour imaging and sounding operations are repeated for every 3-hour interval in normal, watch, or warning mode.
2. Timings are GOES East spacecraft local time unless otherwise specified.
3. GOES West (135° W) 00:00 spacecraft local time = 09:00 UTC.
4. GOES West operations are similar to GOES East but occur 4 hours later, except for IMC, which is performed as shown above.
5. IMC epoch of GOES West is 4 hours later than GOES East in terms of spacecraft tme.
6. Houskeeping times are typical.

9210126

9210126

operation for GOES East and GOES West is performed together and keyed to GOES East spacecraft time, as it is closer to satellite operations control center local time.

Equinox Operations

The equinox seasons occur from about 28 February to 12 April for the vernal equinox and 31 August to 13 October for autumnal equinox. OATS determines the actual start times and duration of the solar eclipses that occur once a day during these seasons. For the duration of the eclipse (at most 72 minutes) the spacecraft depends solely on batteries for electrical power. Because battery capacity is limited, the spacecraft electrical loads are reduced before the eclipse starts. This is automatically accomplished by the load control function except for the Imager and Sounder scan motors, which must be turned off by ground command. Load reduction ensures that the maximum battery depth of discharge is not exceeded for the rated battery capacity and longest eclipse duration. The spacecraft loads are automatically reconfigured after the eclipse when the solar array resumes generating electrical power.

To conserve battery power and minimize ground commanding, noncritical thermal control heaters are turned off automatically upon entry into eclipse and turned on after emergence. The Imager and Sounder are in standby mode, with the scan motors and electronics turned off until the eclipse is over. Because the Imager and Sounder electronics are automatically turned off prior to the eclipse and turned on after the eclipse, the instruments can be quickly reinitialized after eclipse to return to normal operations.

Solstice Operations

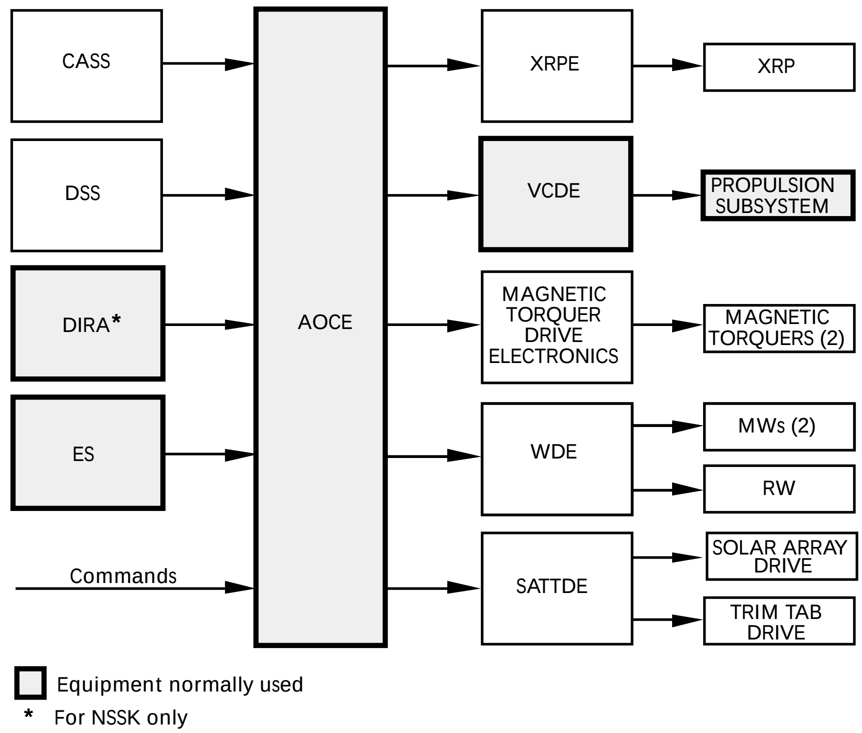
The solstice seasons occur approximately from 13 April to 30 August for summer solstice and about 14 October to 27 February for winter. Except for any periodic operations that need to be scheduled, the spacecraft has very few configuration changes during the solstice periods; it is the same as in the initial on-orbit configuration.

Periodic Operations

Periodic operations are infrequent but recurring events that are scheduled periodically through the spacecraft's on-orbit lifetime. These operations can be inserted into the daily operations schedule as needed and performed during one of the housekeeping intervals, the other scheduled functions being altered accordingly. Major periodic operations are:

- East / west stationkeeping (EWSK)
- North / south stationkeeping (NSSK)
- SEM calibration
- AOCS adjustments

AOCS On-Orbit Control During Stationkeeping



Legend

AOCE	Attitude and Orbit Control Electronics	RW	Reaction Wheel
CASS	Coarse Analog Sun Sensor	SATTDE	Solar Array Trim Tab Drive Electronics
DIRA	Digital Integrating Rate Assembly		
DSS	Digital Sun Sensor	VCDE	Valve Coil Drive Electronics
ES	Earth Sensor	WDE	Wheel Drive Electronics
MW	Momentum Wheel	XRP	X-Ray Positioner
NSSK	North/South Stationkeeping	XRPE	X-Ray Positioner Electronics

9210128

- Solar array slew
- Sun / moon intrusion
- Image interaction dynamics
- Station relocation
- Propellant remaining estimation

East/West Stationkeeping

Frequent EWSK is required to counteract the effects of earth's triaxiality on spacecraft drift and solar radiation pressure on orbital eccentricity. The maneuver strategy is to start the stationkeeping (SK) cycle with the spacecraft at one edge of the longitude deadband (±0.5°), drifting across the deadband with negative perturbing acceleration. With an initial drift rate of just the right magnitude, the spacecraft drifts to the desired longitude, where an EWSK maneuver is applied to reverse the spacecraft drift and keep it within the deadband at the assigned station longitude (either 75° or 135° west). OATS software performs the necessary calculation for determining when to perform the maneuver and the corresponding command data for thruster selection and required duration.

North/South Stationkeeping

About once a year (for 0.5° inclination) NSSK maneuvers are required to counteract the gravitational forces exerted by the sun and moon on the spacecraft. The maneuver strategy is to start the SK cycle with the spacecraft at one edge of the inclination deadband (±0.5° of the equator) at the optimum node, allowing it to drift to zero inclination and then back to 0.5°. The maneuver is again performed to bring the spacecraft back to the beginning of the deadband (optimum node). This minimizes the velocity increment required and, hence, propellant used. OATS software performs the calculation for determining when to perform the maneuver and the corresponding command data.

For the 0.5° inclination deadband, the NSSK maneuver is only required once a year. Because the spacecraft south panel thrusters provide the needed orbit change thrust, the maneuver must be done at the orbit's descending node. When the maneuver is completed, the right ascension of the ascending node is at 270° from the vernal equinox. The required orbital configuration, together with constraints on the solar array orientation with respect to the south panel thrusters, suggests an optimum time for annual NSSK of noon time at winter solstice or midnight during summer solstice.

SEM Calibration

The operation of the SEM sensors involves periodic in-flight calibrations (IFCs) of the magnetometer, XRS, and EPS / HEPAD. An IFC of the magnetometer is performed to verify proper operation. The IFC mode lasts about 82 seconds and can be initiated at any time by ground command. Once initiated, a calibration signal is generated by the magnetometer and superimposed on the ambient magnetic field being measured. All three channels (X, Y, Z) receive the calibration signal simultaneously.

An IFC of the XRS is initiated by ground command to verify basic sensor operation and determine the electronic processing gain of the data processing unit to an accuracy of ±2%. IFC requires an XRS pointing offset from the sun of at least 13° to ensure that all solar X-ray emissions are out of the XRS field of view (FOV), thus obtaining a low background noise environment. The nominal calibration frequency is weekly. IFC is typically performed during housekeeping periods (when trim tab adjustment is not performed), thereby avoiding interference with Imager and Sounder operations. Conditions permitting, calibration can also be performed immediately after NSSK maneuver when the XRS is still in the stowed position.

An IFC of the EPS / HEPAD is initiated by ground command to verify proper operation of the instrument and to adjust the photomultiplier tube high voltage for optimum performance. Once calibration is initiated, the IFC circuitry provides a series of calibration signals to the dome, telescope, and HEPAD amplification channels. The calibration sequence lasts approximately 11 minutes and is self-terminating.

AOCS Adjustments

Adjustments in the AOCS are infrequent and involve wheel unload pulse widths, gain settings for the roll / yaw and pitch loops, pitch and yaw momentum desaturation levels, magnetic torquer parameters, and attitude and orbit control electronics (AOCE) clock update.

Normally, for pitch / yaw wheel unload a 5-ms pulse is required. A range of pulse widths is selectable by ground command and the present pulse width may be updated if required after a few days of on-orbit operation.

There are two sets of gains for the roll / yaw on-orbit mode operation: short-term roll / yaw control and long-term magnetic torquer controller gains. The short-term roll / yaw control gain sets are derived to give optimum performance in the V- and L-modes for imaging and housekeeping operations. High gains are used for housekeeping in V- and L-modes and low gains for imaging in V- and L-modes. Medium gains are used if roll / yaw performance shows a need for modification and are provided for contingencies. The selected on-board gain is available via telemetry.

Magnetic torquers are used to remove residual inertial roll / yaw torque and momentum, thus controlling the yaw error and the amount of momentum stored in the wheels. The magnetic torquer controller has four gain sets used for V- and L-mode operation. One set of two is used for V- and L-mode operation while both magnetic torquers are operational, the other set for the same modes when one of the torquers fails. These gains are used in conjunction with current limit off for the failed torquer. During normal imaging operations, the magnetic torquers' peak limit is constrained to improve yaw repeatability from day to day. This limit setting is based on the operational configuration of the roll / yaw control system during magnetic torquer startup operation.

INR functions are synchronized to ground station-maintained UTC via the 24-hour AOCE internal clock. Periodic resynchronization to the ground-based clock compensates for drift in the AOCE clock's crystal oscillator (16.384 MHz). Due primarily to temperature fluctuations, drift can yield a worst-case error of 20 parts per million (1.73 s / day), based on the nominal on-orbit temperature range of 0 to 40 ˚C. The INR process can accept a maximum clock error of 10 seconds, so that an uncorrected worst-case error would reach the acceptance limit in less than 6 days. Under nominal conditions, oscillator drift is lower than this worst case so a clock update would occur about once a week. The update can be performed at any time when imaging and sounding are not in progress, preferably at the 12:00 noon housekeeping period.

Solar Array Slew

The solar array is slewed to correct errors in pointing the array toward the sun and to meet NSSK maneuver constraints, if required. Errors will occur because the solar array cannot step during the period that trim tab slews are performed. These errors are measured by the east/west sun analog sensor (SAS) telemetry output, though SAS data cannot be used if the error exceeds ±2°. In this case the initial reset is performed using known spacecraft orbital time and readouts from solar array position telemetry. If the detected error exceeds 0.1° (XRS performance constraint), the solar array drive is put into slew mode, the slew direction selected, and the slew time commanded. After slewing, the solar drive is returned to run mode and normal stepping resumed. The SAS telemetry output is again monitored to verify correct solar array pointing.

Sun/Moon Intrusion

The sun, and to a lesser extent the moon, periodically interferes with spacecraft operations, affecting Imager and Sounder radiometric reference and spacecraft attitude control, which in turn degrades INR accuracy. The sun also affects telecommunications between the spacecraft and ground. The sun and moon at low declination can interfere with the Imager and Sounder. For the sun, this occurs during the eclipse season centered on the equinoxes (22 March and 23 September). Moon intrusion can occur at any time during the year, the height occurring in eclipse season when the moon is full or nearly full as it reaches low declinations. The sun is also expected to cross the command and data acquisition (CDA) antenna beam and degrade spacecraft communications during eclipse season.

Upon request from GIMTACS, OATS computes the orbital events and sensor intrusions given a future time span and provides the intrusion start and end times, the sensor(s) being impaired (Imager, Sounder, earth sensor (ES)), and the edge (east or west for the Imager and Sounder) or scan (north or south for the ES) being affected. With these data the GIMTACS scheduler determines when operations of a particular sensor are impaired and formulates commands to switch operating modes to account for the interference.

When the sun or moon enters the ES's FOV, one of the scans is affected. This requires that one of the ES scans be inhibited before the sun or moon enters the scan FOV in order to reduce spacecraft attitude error to an acceptable level. The intrusion occurs approximately 2.5 hours during each eclipse season and the moon intrusion occurs about 2 hours on any day when the moon's phase is near full.

The sun, when at a relatively low declination, is expected to pass through the CDA antenna radio frequency beam, degrading the telemetry and command (T&C) link between the spacecraft and ground station. Interference is also expected for other communication antennas, degrading the signal-to-noise ratio, as high as 30 dB. OATS predicts these events and provides the data to GIMTACS. As no direct work-around is feasible, the spacecraft controller simply adjusts the operating schedule to avoid uplinking commands during the time interval the communications link is impaired.

Image Interaction Diagnostics

Several periodic operations relate to the image navigation and registration (INR) process: dynamic interaction diagnostics, ES single chord operation, mirror motion compensation tuning, and image motion compensation calibration.

- **Dynamic Interaction Diagnostics.** A dynamic interaction diagnostic telemetry system is provided (in GOES-I & K) to measure excessive interaction between mechanical motion events (such as momentum/reaction wheel, solar array drive assembly, and Imager/Sounder mirror motion). Measurement data consist of three-axis angular displacement from the angular displacement sensor, three-axis angular rate and motion from the digital integrating rate assembly (DIRA), Imager and Sounder servo errors, and discrete event information related to solar array drive assembly stepping and Imager/Sounder mirror motion. The diagnostic data are used during initial on-orbit checkout to identify dominant effects that produce excessive interaction and to develop operational scenarios that avoid or minimize such interaction. Diagnostic telemetry can be used at any time during subsequent spacecraft orbital operations for the same purpose.
- **Single Chord Operation.** When the sun or moon intrudes into the ES's FOV, one of the scans is inhibited, yielding ES single chord operation. Inhibiting a scan before intrusion occurs is necessary in order to reduce the spacecraft attitude error to an acceptable level and meet INR requirements. From OATS predictions, the chord in which intrusion occurs and the intrusion start/end times are known to GIMTACS ahead of time. OATS also provides the star command data for the intrusion windows provided by GIMTACS to perform star sightings after inhibition of the affected scan.
- **Mirror Motion Compensation.** A set of mirror motion compensation (MMC) coefficients is established during initial on-orbit startup operations and updated yearly to account for spacecraft mass changes due to propellant consumption. MMC tuning involves a series of Imager and Sounder mirror motion command sequences performed by GIMTACS according to a predefined procedure. MMC tuning is typically performed at the same time of the day each time (fixed solar array position) throughout the spacecraft life to monitor trends in MMC scale factor. Tuning is performed during the post-NSSK stabilization period to minimize interference with normal payload operations, requiring at most 1 hour 15 minutes. OATS uses the resulting Imager/Sounder wideband data with spacecraft telemetry (wheel speed, DIRA data, etc.) to update the coefficients; these are then uplinked to the spacecraft via GIMTACS.
- **Image Motion Compensation Calibration.** A set of baseline in-flight IMC scale factors is established during initial on-orbit startup operations. This baseline, updated yearly, compensates for errors introduced by the digital/analog converters in the AOCE. The updated baseline provides the best agreement between commanded IMC offsets and actual instrument line-of-sight offsets. IMC calibration can be performed at any time if imaging and sounding operations are suspended during the IMC calibration period (at least one hour) and the instruments are specifically configured to support the

calibration process. IMC calibration involves performing a series of star sightings with different IMC offsets as defined by OATS. The resulting Imager and Sounder wideband data are used by OATS to update the east/west and north/south compensation scale factors which are then transmitted to the spacecraft via GIMTACS.

Station Relocation

Three station relocations are anticipated during the spacecraft's lifetime. On-station longitude is changed by applying an incremental velocity, typically at an apse, to maintain eccentricity within acceptable limits and change the radius of the opposite apse. If the velocity increment is applied in the direction of motion, the orbit radius is increased and the spacecraft drifts westerly with respect to earth; if applied opposite to the direction of motion, orbit radius is decreased and the spacecraft drifts easterly. When the desired on-station longitude is reached, an incremental velocity of equal magnitude but opposite direction is applied at the same apse as the first to arrest the drift. The total maneuver is essentially a pair of EWSK maneuvers separated by a period for the spacecraft to drift to its new station. OATS software provides the support needed to compute the maneuver sequences that place the spacecraft on a specified longitude at a specified time.

Propellant Remaining Estimation

The estimate of propellant remaining on board, and hence the remaining spacecraft lifetime, is divided into three phases: prelaunch phase, orbit raising phase, and on-orbit phase. In the prelaunch phase, the initial loading of propellant (fuel, oxidizer, and helium gas) is provided after propellant loading is completed.

The estimate of propellant used during orbit raising operations, about 86.2% of the amount loaded, is provided by Goddard Space Flight Center. The actual amount consumed is tracked through various telemetered parameters and subtracted from the initial loading. The estimate of propellant remaining based on available telemetry is compared with the resultant orbital incremental velocity data to calculate propellant usage and, thus, propellant remaining.

The on-orbit phase, consisting of the operational and storage/standby modes, uses the remaining 13.8% of the propellant loaded. Because maneuvers during the storage/standby mode are performed by OATS, the process of estimating propellant remaining is the same in all on-orbit modes. OGE/GIMTACS records and monitors remaining propellant during the on-orbit phase.

Deorbit

At the end of its operational life, the spacecraft is raised 350 kilometers (217 statute miles) above synchronous altitude to allow other spacecraft to use the vacated orbital slot. In this deorbit maneuver, the AOCS thrusters impart an incremental velocity to the spacecraft, typically at an apse, in the direction of motion, producing an elliptical transfer orbit to the higher orbit radius and a westerly drift with respect to earth. The new orbit is circularized by a second velocity increment applied at the opposite apse in the direction of motion. If the propellants are depleted before deorbit, the remaining pressurized helium can be used to achieve a 120-kilometer (74.6 statute miles) altitude above synchronous. The spacecraft payloads are then shut down to eliminate unwanted transmissions.

Major Features of the GOES Spacecraft and Subsystems

GENERAL SPACECRAFT DATA

Configuration	Body stabilized
Design Life	7-yr (5-yr mission)
Launch Vehicle	Atlas I or Atlas IIA
Maneuver Lifetime	7 to 11 years

SPACECRAFT DIMENSIONS

Launch Configuration Envelope:	
Width Earth Face	2.5 m (97 in)
Height (T&C antenna deployed)	4.6 m (180 in)
Depth	2.9 m (113 in)
On-orbit Configuration:	
Array to Body	6.1 m (242 in)
Solar Sail to Body	17.7 m (697 in)
Magnetometer to Body (true length)	3.0 m (118 in)
Overall Length (sail to array)	26.9 m (1060 in)
Overall Height (T&C antenna to Magnetometer boom)	5.9 m (232 in)
Overall Depth (Earth face to Magnetometer boom)	4.9 m (192 in)

SPACECRAFT MASS	GOES-I/J/K/L	GOES-M
Deployment Mass	2105 kg (4641 lb)	2270 kg (5005 lb)
Dry Mass	977 kg (2154 lb)	1042 kg (2297 lb)
Propellant and Pressurant	1128 kg (2487 lb)	1128 KG (2487 lb)

COMMAND

Receive:	
Frequency	2034.2 MHz
Minimum EOC Antenna Gain (on-orbit)	-3.7 dBi
Minimum G/T	-40 dB/K
Dynamic Range:	
Command only	-113 to -50 dBm
Command and Ranging	-95 to -50 dBm
Transmission Bandwidth:	
Signal Bandwidth	
Without Ranging	<40 kHz
With Ranging	<1 MHz
Uplink Bit Rate	250 lbs

Major Features of the GOES Spacecraft and Subsystems

ATTITUDE AND ORBIT CONTROL SUBSYSTEM (AOCS)

Transfer Orbit	3-axis stabilized w/thrusters
On-orbit Stabilization	3-axis stabilized momentum bias
Pointing Accuracy:	
Antenna Pointing (3σ)	
Roll	±0.25 deg
Pitch	±0.25 deg
Yaw	±0.25 deg
Payload Operations	
Roll	±9.1 μrad
Pitch	±9.4 μrad
Yaw	±73.3 μrad in 90 minutes
Imaging Stability (15 min imaging interval)	42 μrad E-W, N-S Noon ±8 hr 70 μrad E-W, N-S, Midnight ± 4 hr
Stationkeeping Window:	
North-South (N-S), latitude	±0.5 deg about equator
East-West (E-W), on-station	±0.5 deg in longitude

PROPULSION SUBSYSTEM

Propellant	Bipropellant
Tank Volumes/Capacity:	
Fuel - Monomethyl Hydrazine (MMH)	570.0 L (20.13 ft^3)/473 kg (1043 lb)
Oxidizer - Nitrogent Tetroxide (N_2O_4)	570.0 L (20.13 ft^3)/776 kg (1711 lb)
Pressurant - Helium	75.7 L (2.67 ft^3)
Total Propellant Mass Required:	
Fuel	431.1 kg (950.4 lb)
Oxidizer	694.1 kg (1530.2 lb)
Helium	2.72 kg (6.00 lb)
Thrusters:	
AOC (12)	22 N (5 lb)
Apogee (1)	490 N (110 lb)

Major Features of the GOES Spacecraft and Subsystems

ELECTRICAL POWER SUBSYSTEM

Solar Array:	Single axis, Sun tracking	
Number of Panels	2	
Panel Size (each)	236.2 cm x 268.0 cm (93 in x 105.5 in)	
Power Output (watts):	Output	Load
BOL Summer Solstice	1167	1026
BOL Autumnal Equinox	1304	1126
EOL Summer Solstice	1057	1026
EOL Autumnal Equinox	1164	1126
Transfer Orbit	638	596
Batteries:	2 Nickel-Cadmium	
Number of Cells	28 each	
Capacity	12 ampere-hour each	
Depth of Discharge	60% maximum with eclipse	
Eclipse Load Supported	400 watts, 72-minute eclipse	
Bus:	Single Bus System	
Voltage (Sunlight)	42.0 ±0.5 volts dc	
Voltage (Eclipse)	32.4 volts minimum	
Eclipse Load Control	Automatic load disconnect; sequentially reconnected in 6 selectable groups	

Major Features of the GOES Spacecraft and Subsystems

SEARCH AND RESCUE (SAR)

Receive:	
Frequency, Wideband Mode	406.050 MHz
Narrowband Mode	406.025 MHz
Minimum EOC Antenna Gain	9.9 dBi
Minimum G/T	-17.6 dB/K
Dynamic Range	Noise to -125 dBm
Transmission Signal Bandwidth:	
Wideband/Narrowband Mode	80/20 kHz
Transmit:	
Frequency	1544.5 MHz
Power	3 watts
Antenna	
Gain	12.3 dBi
Coverage	Earth
Polarizaiton	RHC
EIRP	45.4 dBm

WEATHER FACSIMILE (WEFAX)

Receive:	
Frequency	2033.00 MHz
Minimum EOC Antenna Gain	11.0 dBi
Minimum G/T	-18.4 dB/K
Transmission Bandwidth:	
Available Transmission Bandwidth	1 MHz
Occupied Signal Bandwidth	30 kHz
Offset	342 MHz
Transmit:	
Frequency	1691.00 MHz
Power	11 watts
Antenna	
Gain	16.5 dBi
Coverage	Earth
Polarization	Linear
EIRP	54.4 dBm

Major Features of the GOES Spacecraft and Subsystems

COMMAND & DATA ACQUISITION (CDA) STATION TELEMETRY

Transmission Signal Bandwidth	<10kHz
Data Rate	2 kbits/sec
Transmit:	
Frequency	1694.0 MHz
Power	3 watts or 35 dBm
Antenna	
Gain (90˚)	-3.5 dBi
Pattern	Cardioid
Polarization	RHC
EIRP	28.9 dBm

DEEP SPACE NETWORK TELEMETRY

Transmission Signal Bandwidth	2.1 MHz
Transmit:	
Frequency	2209.086 MHz
Power	1 watt
Antenna	
Gain	-2.7 dBi
Pattern	Cardioid
Polarization	RHC
EIRP	24.5 dBm

DEEP SPACE NETWORK RANGING

Transmission Signal Bandwidth	<1.0 MHz
Transmit:	
Frequency	2209.086 MHz
Power	<1 watt
Antenna	
Gain	-10 dBi
Pattern	Cardioid
Polarization	RHC
EIRP	17.2 dBm

Major Features of the GOES Spacecraft and Subsystems

DATA COLLECTION PLATFORM REPORT (DCPR) TRANSPONDER

Receive:	
Frequency, Band 1	401.900 MHz
Band 2	402.200 MHz
Minimum EOC Antenna Gain	10.2 dBi
Minimum G/T	-18.7 dB/K
Dynamic Range	Noise to -100 dBm
Transmission Signal Bandwidth	
Transmit:	700 kHz
Frequency, Band 1	1694.500 MHz
Band 2	1694.800 MHz
Power	0.15 watt
Antenna	
Gain	16.5 dBi
Coverage	Earth
Polarization	Linear
EIRP	33.7 dBm

DATA COLLECTION PLATFORM INTERROGATE (DCPR) TRANSPONDER

Receive:	
Frequency 1	2034.9000 MHz
Frequency 2	2034.9125 MHz
Minimum EOC Antenna Gain	11.0 dBi
Minimum G/T	-18.4 dB/K
Dynamic Range	-110 to -90 dBm
Transmission Signal Bandwidth	200 kHz
Transmit:	
Frequency 1	468.8250 MHz
Frequency 2	468.8375 MHz
Power	4.5 watts
Antenna	
Gain	10.7 dBi
Coverage	Earth
Polarization	RHC
EIRP	45.4 dBm

Major Features of the GOES Spacecraft and Subsystems

PROCESSED DATA RELAY (PDR)

Receive:	
Frequency	2027.7 MHz
Minimum EOC Antenna Gain	11.0 dBi
Minimum G/T	-18.4 dB/K
Dynamic Range	-92 to -86 dBm
Transmission Signal Bandwidth	5.2 MHz
Transmit:	
Frequency	1685.7 MHz
Power	11 watts
Antenna	
Gain	16.5 dBi
Coverage	Earth
Polarization	Linear
EIRP	54.9 dBm

MULTI-USE DATALINK (MDL) (GOES-I & K)

Frequency	1681.48 MHz
Power	2 watts
Antenna:	
Gain	16.5 dBi
Coverage	Earth
Polarization	Linear
EIRP	44.0 dBm

SENSOR DATA

Transmission Signal Bandwidth	<4 MHz
Transmit:	
Frequency	1676.00 MHz
Power	1.6 watts
Antenna	
Gain	16.5 dBi
Coverage	Earth
Polarization	Linear
EIRP	45.4 dBm

Important Features of the GOES Sensor Suite

IMAGER INSTRUMENT

Field of View Defining Element	Detector
Optical Field of View	Square
5-Channel Imaging	Simultaneously
Scan Capability	Full earth/sector/area
Channel/Detector:	Instantaneous FOV:
Visible/Silicon	1 km
Shortwave/InSb	4 km
Moisture/HgCdTe	8 km
Longwave 1/HgCdTe	4 km
Longwave 2/HgCdTe	4 km
Radiometric Calibration	Space and 290 k IR internal blackbody
Signal Quantizing	10 bits all channels
NEΔT or S/n	Minimum 3X better than spec
Frequency of Calibration:	
Space	2.2 sec for full disk; 9.2 or 36.6 sec for sector/area
Infrared	30 minutes typical
System Absolute Accuracy	IR channel $\leq$1K Visible Channel 5% of maximum scene irradiance
System Relative Accuracy	IR channel $\leq$0.1 K

Important Features of the GOES Sensor Suite

IMAGER IMAGE NAVIGATION AND REGISTRATION (INR)

Imaging Rate		60˚N to 60˚S ≤26 min 12 sec	
Time Delay		≤3 min	
Fixed Earth Projection and Grid Duration		24 hours	
Data Timeliness:			
Spacecraft Processing		≤30 sec	
Data Coincidence		≤5 sec	
Imaging Periods		Noon±8 hr	Midnight±4 hr
Image Navigation Accuracy at Nadir		4 km	6 km
Registration within an Image*	25 min	50 µrad	50 µrad
Registration between Repeated Images*	15 min	50 µrad	70 µrad
	90 min	84 µrad	105 µrad
	24 hr	168 µrad	168 µrad
	48 hr	210 µrad	210 µrad
Channel-to-channel Registration		28 µrad	28 µrad (IR only)

*For Spec Orbit

Important Features of the GOES Sensor Suite

SOUNDER INSTRUMENT

Field of View Defining Element	Field Stop
Channel Definition	Interference filters
19 Channels:	
Longwave IR	7: 14.7-12.02 µm
Midwave IR	5: 11.03-6.51 µm
Shortwave IR	6: 4.57-3.74 µm
Visible	1 at 0.67 µm
Scan Capability	Full Earth & space
Frequency of Calibration:	
Space	2 minutes
Infrared	30 minutes
Nominal IGFOV	242 µrad, all channels
Sounding Areas	10x10 km 60˚ N-S and 60˚ E-W
Sounding of 19 Channels (every 100 ms)	75 ms
North-South Step Size	1120 µrad
East-West Step Size	280 µrad
Signal Quantizing	13 bits all channels
NE∆N	Minimum 4X better than spec
System Absolute Accuracy	IR channel ≤ 1 K
System Relative Accuracy	IR channel ≤ 0.1 K

Important Features of the GOES Sensor Suite

SOUNDER IMAGE NAVIGATION AND REGISTRATION (INR)

Sounding Rate		3000x3000 km ≤ 42 min	
Time Delay		≤3 min	
Fixed Earth Projection and Grid Duration		24 hours	
Visible Channel Quantization		≤0.1% to 100% albedo	
Infared Channel Data		1/3 of specified NEΔN	
Data Timeliness:			
Spacecraft Processing		≤30 sec	
Sounding Periods		Noon±8 hr	Midnight±4 hr
Image Navigation Accuracy at Nadir		10 km	10 km
Registration within a 120-minute Sounding	120 min	84 µrad	112 µrad
Registration between repeated soundings	24 min	280 µrad	280 µrad
Channel-to-channel Registration		28 µrad (IR/Vis)	28 µrad (IR only)

Important Features of the GOES Sensor Suite

SPACE ENVIRONMENTAL MONITOR (SEM)

MAGNETOMETER	
Function	Measure ambient magnetic field to ± nT, w/ corrections
Sensor Element	Forster fluxgate probe
Sensor Assembly	Redundant magnetometers, 3 orthognonal fluxgate probes each mounted on 3.1 m boom
Dynamic Range	±1000 nT, any orientation
Resolution	0.03 nT
SOLAR X-RAY SENSOR (XRS)	
Function	Measure solar x-ray in 2 bands
Spectral Bands	0.5-3.0 and 1.0-8.0 Å
Resolution: Fluxes	
>20 times threshold	≤2% of reading
Sampling Rate	Once every 0.512 sec
Sun Tracking:	
Diurnal	Mounted on solar array Sun tracking yoke
Seasonal	Single axis positioner with Sun sensor, closed loop, north-south tracking
ENERGETIC PARTICLES SENSOR (EPS)	
Function	Measure flux of proton, alpha particles and electrons in 16 energy bands from 0.55 to 500 MeV
Sensor Elements	Solid state nuclear detectors
Sensor Assemblies	One 2-detector telescope, one dome assembly of three detectors
Sampling Rate	Once every 10.2 or 20.5 sec
Dynamic Range	Cosmic ray background to largest solar particle event

Important Features of the GOES Sensor Suite

HIGH ENERGY PROTON & ALPHA PARTICLE DETECTOR (HEPAD)

Function	Measure flux of protons and alpha particles from 350 to 3400 MeV
Spectal Bands:	
Protons	3 from 370 to 970 MeV
Alpha Particles	1 at ≥970 MeV
Sensor Element	Cerenkov Scintillation sensor
Sensor Assembly	Telescopic arrangement of Cerenkov crystal and 2 solid state detectors
Field of View	Conical, ~34° half angle
Dynamic Range	0 to 10^4 counts/second

Abbreviations and Acronyms

A

A	ampere
A/D	analog to digital
AAM	apogee adjust maneuver
ADC	analog to digital converter
ADM	attitude data multiplexer
ADS	angular displacement sensor
AEI	average error integrator
Ah	ampere-hour
AMF	apogee maneuver firing
AOC	attitude and orbit control
AOCE	attitude and orbit control electronics
AOCS	attitude and orbit control subsystem
AOS	acquisition of signal

B

b/s	bits/second
BB	blackbody
B/S	beam splitter
BECO	booster engine cutoff

C

C	Celsius
CASS	coarse analog sun sensor
CASSE	CASS electronics
CCD	charge coupled device
CCSDS	Consultative Committee for Space Data Systems
CDA	command and data acquisition
CEI	coherent error integrator
Ch	channel
CME	Coronal Mass Ejection
cm	centimeter
CONUS	Continental United States
C&DMS	Command & Data Management System

D

D/A	digital to analog
dB	decibel
dc	direct current
DCP	data collection platform
DCPI	data collection platform interrogate
DCPR	data collection platform report
DCS	data collection system
DID	dynamic interaction diagnostic
DIRA	digital integrating rate assembly
DPU	data processing unit
DSN	deep space network
DSS	digital sun sensor

E

EED	electroexplosive device
EIRP	effective isotropic radiated power
ELT	emergency locator transmitter
EOC	edge of coverage
EPIRB	emergency position indicating radio beacon
EPS	energetic particles sensor
ES	earth sensor
EST	Exposure Setting Table
E/W	east/west
EWSK	east/west stationkeeping

F

FB	Federal Building
FET	Field Effect Transistor
FOT	Fiber Optics Taper
FOV	field of view

G

g	gram
g	gravity
GFE	Government-furnished equipment
GFRP	graphite-fiber-reinforced plastic
GIMTACS	GOES I-M telemetry and command system
GMS	Geostationary Meteorological Satellite (Japan)
GOES	Geostationary Operational Environmental Satellite
G/T	gain-to-noise temperature ratio, dB/K
GVAR	GOES variable data format

H

HASS	High Accuracy Sun Sensor
He	helium
HEPAD	high energy proton and alpha particle detector

HgCdTe mercury cadium telluride (mercadtelluride)
hr hour
HVPS High Voltage Power Supply
Hz hertz

I

IF intermediate frequency
ICT Imaging Control Table
IFC in-flight calibration
IGFOV instantaneous geometric field of view
IMC image motion compensation
INR image navigation and registration
InSb indium antimonide
IOS Indian Ocean Station
IR infrared
ISS interrupt safety system

K

K degrees kelvin
kb/s kilobits per second
keV kiloelectronvolt
kHz kilohertz
km kilometer

L

L liter
L/O lift-off
LSTP local solar time at perigee
LW longwave

M

m meter
Mb/s megabits per second
MCP Micro Channel Plate
MDL multiuse data link
MECO main engine cutoff
MES main engine start
MeV megaelectronvolt
MHz megahertz
min minute
MLI multilayer insulation
mm millimeter
MMC mirror motion compensation
MMH monomethyl hydrazine
ms multi second
MTF Modulation Transfer Function
MUX multiplexer
MW midwave
MW momentum wheel

N

N newton, north
NASA National Aeronautics and Space Administration
NEΔT noise equivalent change in temperature
NESDIS National Environmental Satellite Data and Information Service
NEΔN noise equivalent radiance difference
NH Northern Hemisphere
NLUT normalization look-up table
NOAA National Oceanic and Atmospheric Administration
NRZ Non-return to zero
N/S north/south
NSSK north/south stationkeeping
nT nanoTesla
NWS National Weather Service
N·m·s newton-meter-second
N_2O_4 nitrogen tetroxide

O

OATS orbit and attitude tracking system
ODAPS OGE data acquisition and patching subsystem
OGE operations ground equipment
OIS OGE input simulator
OMUX output multiplexer
OSR optical solar reflector

P

PCM pulse code modulation
PCU power control unit
PDR processed data relay
PM product monitor
PMD propellant management device
PMT photomultiplier tube
PT Parameter Table

Q

q dynamic pressure

R

rad radians
RAM Random Access Memory
RF radio frequency
RFI radio frequency interference
RHC right hand circular

RMS	root mean square
ROM	Read Only Memory
RW	reaction wheel

S

s	second
SADA	solar array drive assembly
SADE	solar array drive electronics
SAR	search and rescue
SARSAT	search and rescue satellite-aided tracking
SAS	sun analog sensor
SATTDE	solar array trim tab drive electronics
SAU	signal analyzer unit
S/C	spacecraft
SD	sensor data
SECO	sustainer engine cutoff
SEC	Space Environment Center
SEM	space environment monitor
SEU	Single Event Upset
SESC	Space Environment Services Center
SFSS	satellite field service station
SH	Southern Hemisphere
SHM	safe hold mode
SHME	safe hold mode electronics
SK	stationkeeping
SMS	Synchronous Meteorological Satellite
S/N	signal-to-noise ratio
SOCC	satellite operations control center
SPS	sensor processing system
sr	steradian
SSA	sequential shunt assembly
STDN	satellite tracking data network
SW	shortwave
SXI	Solar X-Ray Imager

T

T&C	telemetry and command
TACTS	telemetry and command transmission system
TC&R	telemetry, command and ranging
TLM	telemetry
TMF	trim maneuver firing

U

UHF	ultra high frequency
UTC	universal time coordinated
UV	Ultraviolet

V

v	velocity
V	volt
VAS	visible and infrared spin scan radiometer atmospheric sounder
VCDE	valve coil drive electronics
VECO	vernier engine cutoff
VIE	VAS interface electronics

W

W	west, watts
WDE	wheel drive electronics
WEFAX	weather facsimile

X

XRP	X-ray positioner
XRPE	X-ray positioner electronics
XRS	X-ray sensor

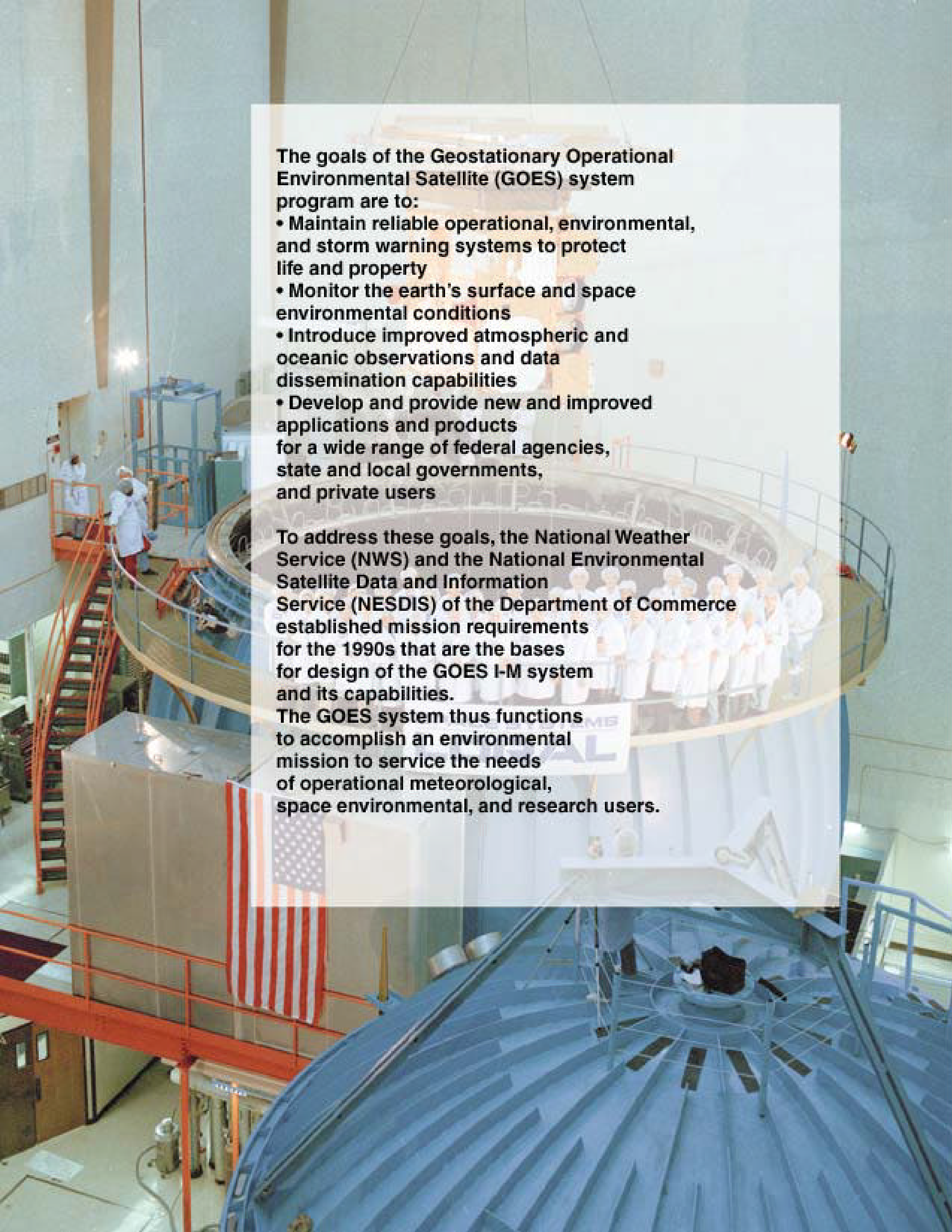
Symbols

Å	angstrom
Δ	difference
μ	micro (10^{-6})
Ω	ohm

Notes

Notes

Notes



The goals of the Geostationary Operational Environmental Satellite (GOES) system program are to:

- Maintain reliable operational, environmental, and storm warning systems to protect life and property
- Monitor the earth's surface and space environmental conditions
- Introduce improved atmospheric and oceanic observations and data dissemination capabilities
- Develop and provide new and improved applications and products for a wide range of federal agencies, state and local governments, and private users

To address these goals, the National Weather Service (NWS) and the National Environmental Satellite Data and Information Service (NESDIS) of the Department of Commerce established mission requirements for the 1990s that are the bases for design of the GOES I-M system and its capabilities. The GOES system thus functions to accomplish an environmental mission to service the needs of operational meteorological, space environmental, and research users.