# PROCEEDINGS
OF
# THE ELEVENTH INTERNATIONAL SYMPOSIUM
ON
# SPACE TECHNOLOGY
AND
# SCIENCE

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The primary structure, as shown in Fig. 5, consists of a horizontal equipment platform supported by a central thrust tube assembly. This assembly consists of a lower conical shell, a central cylindrical shell, and an upper cone.

At the thrust tube's upper end, a caging mechanism protects the drive motor assembly from launch loads, permits a lighter DMA design, and improves the spacecraft inertia ratio. In the caged condition, spinning and non-spinning caging rings are held together with a band clamp which is command released by fully redundant ordnance-actuated separation bolts.

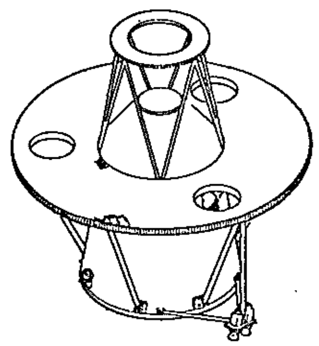

Fig. 5 Spacecraft Structure

8. Propulsion Subsystem

The propulsion subsystem consists of the apogee kick motor and the reaction control equipment. The AKM is a solid propellant SVM-6 rocket motor. A safe/arm (S/A) device initiates AKM ignition by ground command. The AKM characteristics are shown in Table 3.

The RCE is a monopropellant hydrazine system consisting of four thrusters (nominal 5 lb thrust each) supplied by three interconnecting tanks with a combined capacity of 38.6 kg. Each thruster, selected by ground command, is fired in 90 ms pulses. Two redundant radial thrusters, mounted side by side near the equipment platform edge, carry out east/west orbital maneuvers. Two redundant canted thrusters are used in a pulsed mode to change spacecraft attitude and in an extended firing mode to make orbit plane inclination corrections.

Table 3. AKM Characteristics, 13°C

| | | |
|---|---|---|
| Weight: | Pre-fired motor | 363 kg (max) |
| | Propellant | 314.2 kg |
| | Post-fired motor | 33.4 kg |
| Velocity Increment | | 1839 M/S |
| Thrust Level | | 45,434 N (max) |
| Specific Impulse (effective) | | 289.7 sec |
| Total Impulse | | 934,130 N-sec (max) |
| Burn Time | | 43 sec (max) |

9. Spacecraft Development Schedule

The basic design of the CS was completed in December 1974. Detail design for the manufacture of the protoflight model is currently in process. The Communications Satellite will be launched by NASA (U.S.A.) in the fourth quarter of 1977 using a Delta 2914 launch vehicle on the Eastern Test Range of the United States. It will be placed in synchronous orbit at a position of 135° E longitude.



# Development of a Medium-Scale Broadcasting Satellite for Experimental Purpose (BS)

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ABSTRACT

An Experimental Broadcast Satellite Program has been initiated by Japan to evaluate picture and sound transmission at $K_u$-band. Here the synchronous orbit spacecraft for this program is described. This spacecraft is a three-axis stabilized satellite having sun-oriented arrays for high power generation. Two wideband independent TV channels are provided by the direct conversion mission transponder. The modular features of the spacecraft, its performance capabilities and subsystem design concepts are described.

1. Mission Objectives

The BS Program has been initiated by Japan Ministry of Post and Telecommunications and developed by National Development Agency of Japan (NASDA). Manufacture is by the Toshiba/General Electric industry team.

The objectives of the BS Program are to conduct the experiment of transmission of TV test signal from the geostationary satellite to Japanese territory as shown in Figure 1 and to evaluate the performance of the experimental satellite broadcasting system and the rainfall effect on radio wave propagation at 12GHz frequency band in Japan.

The experiment will extend over three years to assess the seasonal characteristics of radio wave propagation and the degradation of satellite system.

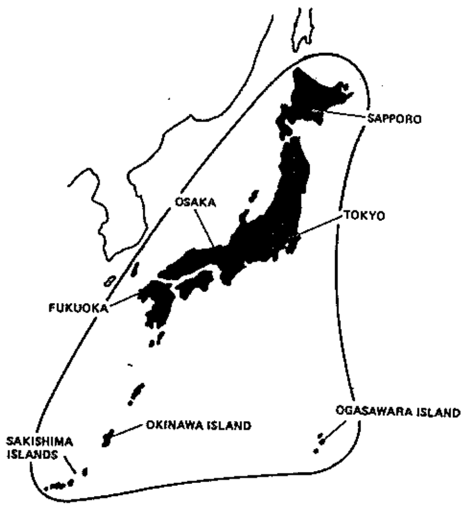


Figure 1. Coverage Area – Japanese Territory and Main Islands

2. System Definition

The BS program elements are depicted in Figure 2. The BS spacecraft will be launched into transfer orbit from the United States Eastern Test Range using the Thor-Delta 2914 launch vehicle. The launch vehicle and support services are provided by the National Aeronautic and Space Agency (NASA) of the United States under agreement to Japan. Synchronous orbit injection and initial orbit stationing and checkout at 110°East Longitude are controlled from the NASDA Ground Control Station (GCS). Subsequently the spacecraft control is transferred to the Main Transmitting and Receiving Station (MTRS) of Radio Research Laboratory.

The BS system parameters are defined in Table 1 and form the basis for Japan's experimental TV broadcasting satellite program.

For experimental color TV signal transmission a minimum bandwidth per channel of 25 MHz was selected. A separation of at least 50 MHz between channel groups is desirable to simplify multiplexer design. For BS a bandwidth per TV channel group of 50 MHz and 80 MHz has been allocated thereby providing additional experimental flexibility. Within these bands the proposed assignment of two TV channels are as shown in Figure 3. The dual channel capability will permit conduct of mutual interference and multiple access studies of simultaneous transmissions.

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Table 1. System Parameters

| | |
|---|---|
| Satellite Location | 110° East Longitude |
| Experimental Coverage | Japanese Territory |
| Frequency Bands | 14.25–14.43 GHz uplink<br>11.95–12.13 GHz downlink |
| Number of TV Channels | 2 |
| Picture Quality | S/N = 45 dB (TASO Grade 1) |
| Power Flux Density | Japan Mainland (-108 dBw/m²)<br>Remote Territory (-117 dBw/m²) |
| System Life | 3 Years |
| Booster | Thor-Delta 2914 |
| Command and Control | S Band and K Band from Control Stations in Japan |

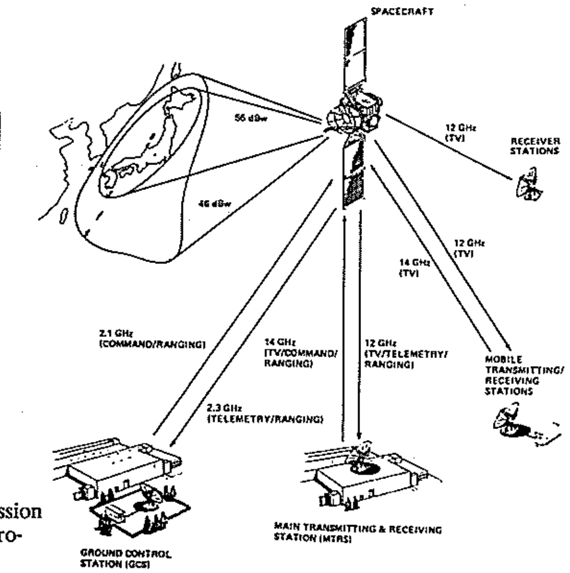


Figure 2. BS Program

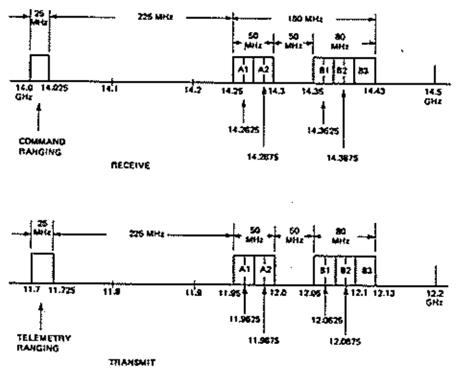


Figure 3. Channel Designation

3. Spacecraft Design Configuration

The spacecraft configuration maximizes the mission benefits of a three-axis controlled spacecraft by providing a fixed (non-gimballed or rotating) antenna platform, large North/South viewing equipment panels for passive heat rejection, and an oriented solar array for efficient generation of power. The spacecraft in the orbital configuration along with key spacecraft characteristics is shown in Figure 4.

The packaging flexibility of the three-axis design is utilized to obtain a favorable moment of inertia about the spin axis, assuring stable spin control with passive nutation damping during the transfer orbit and apogee motor firing.

For orbital operation, the earth viewing spacecraft surface provides required field of views for earth sensor, monopulse sensor and antennas. The K-band

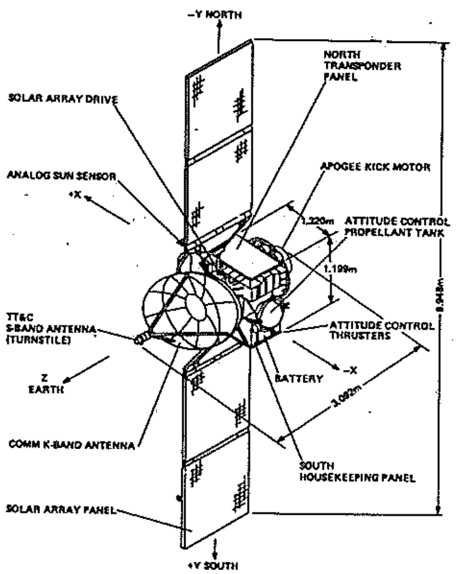


Figure 4. Spacecraft Orbital Configuration

mission antenna mounted on this platform is thermally isolated by an insulated truss to minimize temperature effects. An S-band telemetry antenna is mounted forward of the K-band feed horns to minimize pattern interference from any other spacecraft element.

Solar arrays which track the sun are positioned outboard of the antenna by stand-off yokes. This eliminates solar cell shadowing under the worst case solar inclination conditions and precludes solar reflections onto thermal radiating surfaces. Completely redundant solar array drives and power takeoff assemblies are connected via a throughshaft. Either drive can power the arrays through a passive wrap spring clutch, assuring redundancy as well as identical positioning of each solar array. The equipment module is the main structural element and supports the following subsystems:

a. Communications Subsystem
b. Tracking, Telemetry and Command Subsystem
c. Attitude Control Subsystem



d. Electrical Power Subsystem
e. Structure Subsystem
f. Thermal Control Subsystem
g. Secondary Propulsion Subsystem
h. Apogee Kick Motor

The primary heat generating components are mounted on the north and south viewing radiating areas. Housekeeping electronics are grouped together on the south equipment panel and the transponder subsystem electronics are mounted on the north panel. The reaction control propellant tanks are located on the east/west sides of the spacecraft at a maximum distance from the center line to obtain maximum moments of inertia about the transfer orbit spin axis. Propellant tanks are also located in the spacecraft orbital CG plane minimizing control torque variations as the propellant is depleted.

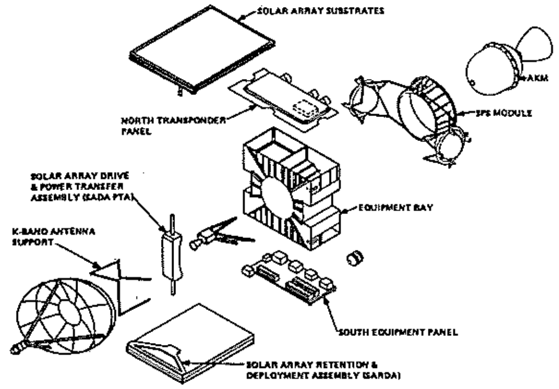


Figure 5. Spacecraft Exploded View

Redundancy of all propulsion functions are provided by thrusters which are located to prevent any significant plume impingement on the vehicle. The yaw and pitch thrusters on the east and west sides support dual functions of wheel unloading and stationkeeping.

Table 2. Spacecraft Performance Parameters

| | |
|---|---|
| Pointing Accuracy | ±0.15° (3σ) |
| Orbit Positioning Accuracy | ±0.1° |
| Solar Array Power | 970 watts (worst case BOL) |
| Orbit Life | 3 years (expendable limit) |
| Reliability | 0.65 (3 year mission) |
| Launch Weight | 669 Kg |

As shown in Figure 5, modularity and accessibility, permitting parallel subsystem assembly and test, are emphasized on the design. The transponder subsystem and the housekeeping equipment, on their separate panels, can be functionally checked out prior to final vehicle assembly. The antenna reflector feed support and feed are also a separate and complete assembly. The Secondary Propulsion Subsystem (SPS) is modularized and provided with a separate support structure, permitting the design of a completely welded assembly, including the alignment of thrusters. The key overall spacecraft performance capabilities are presented in Table 2.. Design features of the major subsystems are presented in the following paragraphs.

Communications Subsystem

The primary function of the communications subsystem is to receive experimental TV test signals from the Main and Transportable Transmitting Stations and retransmit these signals to Receive Terminals throughout Japan and its remote islands. The design of the communications system has considered the antenna size for all receiving stations and their type of service, spacecraft power, antenna size and transmitting station EIRP. The link calculations for this design are shown in Table 3 for both the uplink and downlink cases. A new 100-Watt transmitter power output tube based upon current technology was especially developed for the BS mission.

Figure 6 shows a functional block diagram of the Communications Transponder. Gain allocations requirements are distributed to minimize the number of hardware elements, and to maintain margin from the non-linear points of all active elements except for the 100-Watt TWT's. The transmit-receive diplexer consisting of directional cavity filters routes signals between the communications antenna and the TT&C subsystem. It also allows a receiver and two transmitters, one for each frequency band, to be connected to the single communications antenna port without mutual interference between transmit and receive functions. The 14 GHz switch routes received signals to either Receiver No. 1 or Receiver No. 2 which linearly preamplifies signals using Tunnel Diode Amplifiers in both Channel A and Channel B and down-converts their frequencies by 2.3 GHz using a local oscillator. The switch provides redundancy allowing signals from either receiver to be routed to the input multiplexer, which separates (by filtering) signals in Channel A from signals in Channel B. The input switching assembly routes the Channel A signal to Transmitter A and the Channel B signal to Transmitter B, and allows either the Channel A signal or the Channel B signal (but not both) to be routed through the redundant transmitter. Each transmitter consists of a driver stage employing a Low Level Traveling Wave Tube Amplifier (LLTWTA) with automatic leveling and remote power control and a 100-Watt output Traveling Wave Tube Amplifier (100 W TWTA). The Output Switching Assembly routes outputs from the two active transmitters to the output multiplexer section of the transmit-receiver diplexer/output multiplexer assembly. The transponder performance is summarized in Table 4.

Table 3. Link Calculations for Uplink and Downlink

| | | |
|---|---|---|
| **Downlink** | | |
| Transmitter power | 100 W | nominal |
| RF Losses | 1.5 dB | nominal |
| Spacecraft Antenna Gain | 37.0 dB | edge of area |
| Space Loss | 205.8 dB | 12.0 GHz |
| Rain Loss | 7.0 dB | 99.99% |
| Receiving Antenna Gain | 43.5 dB | 1.6mϕ 55% eff. |
| Receive Power | -113.8dBw | |
| Noise Power | -126.8dBw | 23MHz bw, 660°K |
| C/N | 13.0 dB | |
| Reqd C/N | 10.0 dB | for S/N = 45 dB |
| Margin | 3.0 dB | |
| **Uplink** | | |
| Ground Station EIRP | 81.5 dBw | 12mϕ |
| Space Loss | 207.2dB | 14.3 GHz |
| Spacecraft Antenna Gain | 41.5 dB | |
| Received Power, Excluding Rain Loss | -84.2dBw | |
| Receiver Noise Power | -123.4 dBw | |
| Received C/N, Excluding Rain Loss | 39.2 dB | |

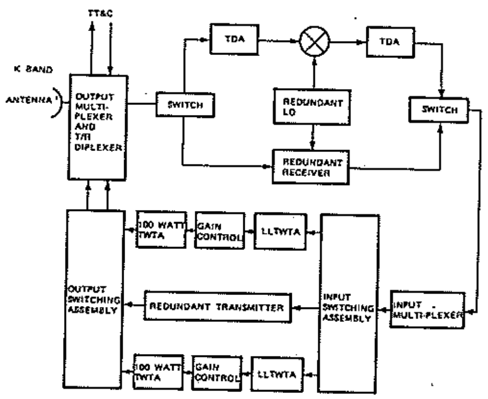


Figure 6. Transponder Block Diagram

Table 4. Transponder Capability

| | |
|---|---|
| PFD at Spacecraft | −82 to −96 dBw/m$^2$ |
| Level Control | Automatic over 16 dB range |
| TWT Drive Control | 64 levels by command |
| Noise Figure | Less than 8.5 dB |
| TWT Output Power | 100 watts minimum |
| Frequency Response | ±1.0 dB in band |
| Response Attenuation | −50 dB below peak at 50 MHz outside band |

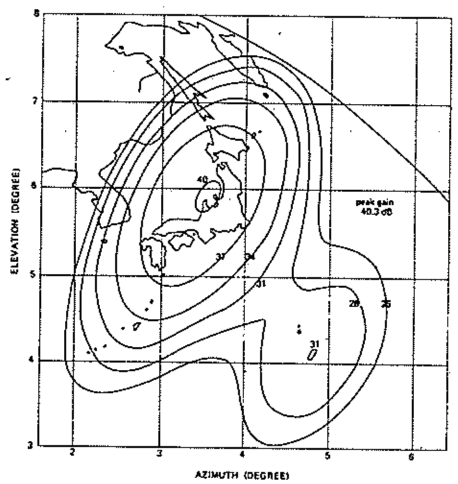


Figure 7. Ground Coverage Pattern

Spacecraft antenna pattern studies have developed a coverage pattern as shown in Figure 7. This multi-beam pattern provides for a rapid falloff to the westward of Japan and a wider beam to the eastward. This results in a maximum reduction of 10 dB for the remote islands of Japan. The gain reduction at the extremes of the main islands is less than 4.3 dB. This gain pattern is accomplished by means of an elliptical reflector using a power splitting three horn feeds.

Tracking/Telemetry and Command Subsystem (TT&C)

The TT&C Subsystem utilizes a redundant S-band receiver-transmitter combination and a redundant K/S-band Converter to accept uplink commands, send spacecraft telemetry to earth on the downlink carrier, and process a tone modulated signal for spacecraft ranging information. The ranging signal is received on the uplink carrier, demodulated, combined with the telemetry in the transmitter/baseband unit and then used to modulate the transmitter carrier and retransmit it back to earth. The demodulated outputs of the command receiver are processed by the command decoders.

Table 5. TT&C Characteristics

| Item | Telemetry | Command | Ranging |
|---|---|---|---|
| Carrier Frequency | S Band<br>Ku Band | S Band<br>Ku Band | S Band<br>Ku Band |
| Modulation | PCM/PSK/PM | PCM/PSK/FM/PM | Tone/PM |
| Bit Rate | 512 BPS | 1000 BPS | — |
| Capability | ≈300 Telemetry Points | ≈200 Commands | |

S-band antenna will be utilized for both receiving and transmitting during transfer, drift and synchronous orbit phases. In addition during the synchronous orbit phase K-band transmissions are received by the spacecraft via the Communications Subsystem, downconverted and processed. The amplified downlink signals are upconverted and transmitted at K-band. The local oscillator in the K/S-band Converter also provides a stable clock signal for monopulse sensor operation. Table 5 summarizes the TT&C Subsystem Characteristics.

Attitude Control Subsystem (ACS)

During the transfer orbit through completion of



the Apogee Kick Motor (AKM) burn, the spacecraft is spin stabilized using the nominal 60 rpm spin rate provided by the Thor-Delta third stage. The ACS develops data from the Horizon Crossing Indicator and Digital Sun Sensor which is transmitted to the ground along with ranging data for computation of spacecraft inertial attitude and subsequent generation of precession commands for transmission to the spacecraft to modify its attitude to that required for the AKM burn. A nominal precession of 135° is required and is accomplished by means of on-board controlled operation of the precession thrusters of the SPS. A passive fluid loop damper is employed to reduce any nutation induced throughout the transfer orbit operations. After injection into synchronous orbit, the ACS despins the spacecraft again employing the SPS and the Rate Gyro for low spin rate sensing.

When on-station in synchronous orbit, the ACS sensor complement includes an earth sensor, detecting spacecraft axis pitch and roll errors; a monopulse sensor, detecting roll and pitch errors referenced to the RF beam center; and analog sun sensors from which yaw error is extracted. Normal operation is the utilization of the attitude information from any two of these three sensors.

For on-orbit, control is effected by a zero-momentum system which nullifies the momentum effects of disturbance torques on each axis while maintaining spacecraft orbit rate rotation through use of three orthogonal reaction wheels aligned with each of the spacecraft axes. When the reaction wheel momentum storage capability is reached, the ACS automatically reduces the internal wheel momentum level through operation of the appropriate control thrusters of the SPS.

Electrical Power Subsystem (EPS)

The EPS utilizes an oriented solar array for power generation, and secondary batteries for energy storage. The EPS is a Direct Energy Transfer (DET) System in which the solar array primary power source is electrically connected to the user load, with no in-line power limiting device. The spacecraft bus voltage is regulated at 28 volts dc ± 1 percent at the user load terminals. Three 4-ampere-hour sealed nickel-cadmium 16-cell batteries provide energy during launch, ascent, and transfer orbit injection prior to illumination of the folded solar arrays and share the load as necessary with the array during transfer orbit and eclipse periods. Each battery has its own dedicated charge regulator and reconditioning circuitry which is initiated and terminated by ground commands. Battery discharge is accomplished at the regulated bus voltage through the use of a voltage boost converter. The EPS provides centralized power control and turns all power on and off by ground command, and provides all safing, arming, and actuation functions associated with all Electro-Explosive Device (EED) initiators on the spacecraft. The EPS performance characteristics are shown in Table 6.

4. Spacecraft Initialization and Orbit Stationing

Two nominal daily launch windows of 20 minutes each are provided while maintaining transfer orbit constraints for adequate power generation and sun, spacecraft, earth geometry for attitude determination. The third apogee is selected for nominal synchronous orbit injection but the spacecraft design for thermal control and power generation do not limit the length of time that may be spent in the transfer orbit, and similarly the time for accomplishing despin. Hydrazine is provided to remove worst case launch vehicle and AKM induced orbit errors (3σ) as well as providing orbit stationing at 110°E and zero degrees inclination within 30 days and maintaining this orbit station to within ± 0.1° in longitude and inclination for 3 years.

Table 6 Primary EPS Performance Characteristics

| | |
|---|---|
| Solar Array Area | 9.58 Meters$^2$ |
| Minimum Array Power — 3 Years | 780 Watts |
| Maximum User Load — 3 Years | 748 Watts |
| Regulation at EPS Terminals | 28 volts ± 1% |
| Maximum Depth of Discharge (including 2 battery condition) | 60% |