PROCEEDINGS
OF
THE THIRTEENTH INTERNATIONAL SYMPOSIUM
ON
SPACE TECHNOLOGY
AND
SCIENCE

TOKYO

1982

The point C is the closest point to the point A on the line BD. Therefore we call the point C the opening time of the firing window. If there is an error in the δ component of the ABM firing attitude at firing time, the (i,Ω) of the drift orbit will be on the line EF. We defined the points E and F such that if the error in the δ component of the firing attitude is within 2σ, then the (i,Ω) of the drift orbit will be between the points E and F.

The closing time of the firing window D is defined such that if the ABM is fired in the firing window and the error in the δ component of the firing attitude is within 2σ, then the inclination of the drift orbit is less than $1.0°$.

6. Drift Orbit

Main tasks of the drift orbit phase are attitude control to make the spin axis normal to the orbit plane and orbit control to make orbit eccentricity be zero, semimajor axis geosynchronous, and subsatellite point be 130° East. Figure 6. shows the actural event sequence of the attitude and orbit maneuvers on ETS-II drift orbit. Abscisa is East longitude and ordinate is drift rate. Injected point into the drift orbit was 84.74° in east longitude and its drift rate was 11.77°/day.

Each orbit maneuver was executed by so called Hohman transfer method. Namely the time of maneuver was selected at the apogee or perigee crossing. Including the preliminary test maneuver (Man.#3 in Fig-6), 5 orbit maneuvers were carried out for geosynchronous station acquisition. The data of these maneuvers are shown in Fig-6.

7. Conclusion

This was the first atempt to launch and station a geosynchronous satellite by Japan. Everything was new to us although we had done an analysis for satellites of circular orbit with no active control, unlike the mission analysis for the geosynchronous or geostationary satellite. As a part of the mission analysis we developed our own methods using charts and we used them for ETS-II ABM firing and station acquisition, and we are pleased with the results. Now we would like to refine these charts and their usage for future geostationary satellites.

Reference

1. Kozai, Y. : Orbit of Artificial Satellite, Technical Memorandum, 1973.

Verification Testing of Japan's Experimental Broadcast Satellite

T. OHTAKE*, H. REICHERT**

ABSTRACT

The Medium Scale Broadcasting Satellite for Experimental Purpose (BSE) was begun in early 1974 by the industrial team of Toshiba and General Electric under contract to NASDA (National Space Development Agency of Japan). The design, fabrication, and test of the BSE spacecraft is now complete and the Flight Model spacecraft awaits the planned early 1978 launch. This paper briefly reviews the basic mission objectives of the BSE Program and describes its hardware elements. The primary features of the spacecraft design and its assembly process are described using photographs of the BSE flight hardware. The test program, which was applied to the critical spacecraft sub-assemblies, is described in detail including those for the multi horn K-band antenna, the K-band high power transponder, and the solar array deployment. Aspects of the spacecraft level environmental test program are presented with particular emphasis on the Thermal Balance, Thermal Vacuum, and Vibration tests. The authors wish to acknowledge the helpful discussions and contributions made by the MOPT, RRL . NASDA and NHK staffs.

1. Introduction

Development of the Medium-Scale Broadcasting Satellite for Experimental Purpose (BSE) was initiated by Japan in early 1974 and is now in the final stages of completion prior to the launch of the Flight Model spacecraft in February 1978. The basic mission objectives of the program and design and status of the space and ground systems elements will be briefly described. The major features of the spacecraft design assembly and test process will be reviewed.

2. BSE Program Objectives

The BSE objectives are to provide experimental color television signal and sound signal transmission at Ku band to remote and urban areas of the Japanese mainland and offshore islands. As shown in Figure 1, the experimental coverage area includes the Okinawa and Ogasawara Island groups as well as other outlying islands.

Receiving and transmitting stations located in selected key area will be used to measure the effects of atmospheric and geographic characteristics on television signal transmission. Other factors of interest such as modulation methods, channel separation, interference effects, and multiple access techniques will also be studied during the 3-year experiment phase.

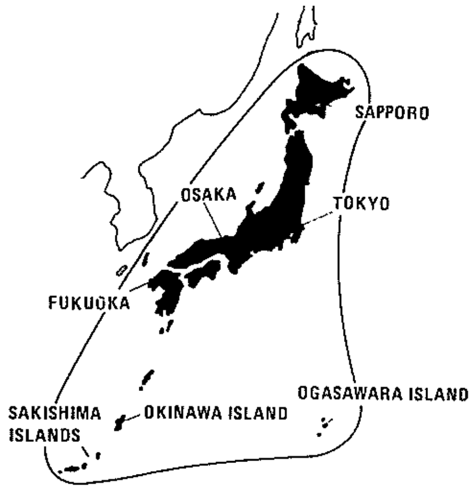

Figure 1. Coverage Area - Japanese Territory and Main Islands

3. System Definition

The BSE program elements are depicted in Figure 2. The BSE spacecraft will be launched into transfer orbit from the United States Eastern Test Range using the Delta 2914 launch vehicle. The launch vehicle and support services

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will be provided by the National Aeronautic and Space Administration (NASA) of the United States under contract to Japan (NASDA). Synchronous orbit injection and initial orbit stationing and checkout at 110° East Longitude are controlled from the NASDA Tracking and Control Center (TACC) at Tsukuba, Japan. Subsequent to spacecraft on-orbit checkout a coordinated Experiment Plan will be initiated through the Radio Research Laboratory's Main Transmitting and Receiving Station (MTRS) at Kashima. Also shown are the (TTRS) Transportable Transmitting and Receiving Stations and the (RS) Receiving Stations which are of several types.

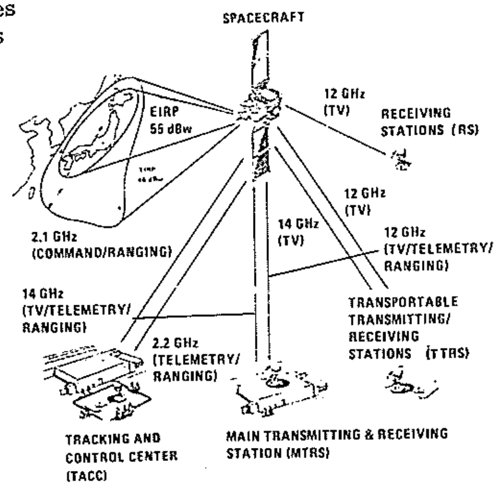

Figure 2. BSE Program

A. Spacecraft

The spacecraft shown in Figure 3 is a 3-axis body stabilized design with a fixed antenna and sun oriented arrays for efficient power generation. The Ku Transponder is mounted on the North Panel and is a two channel direct conversion system using three 100W TWT's

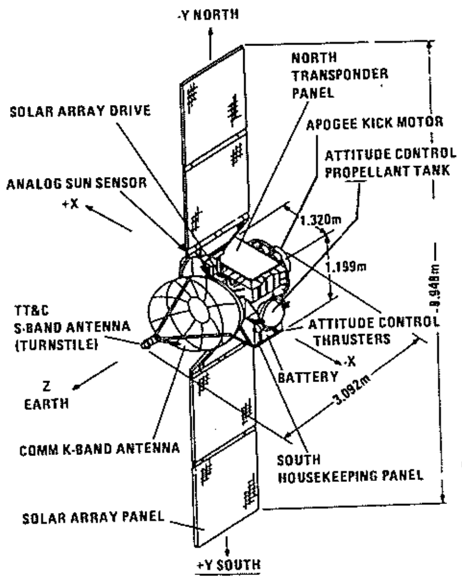

Figure 3. Spacecraft Orbit Configuration

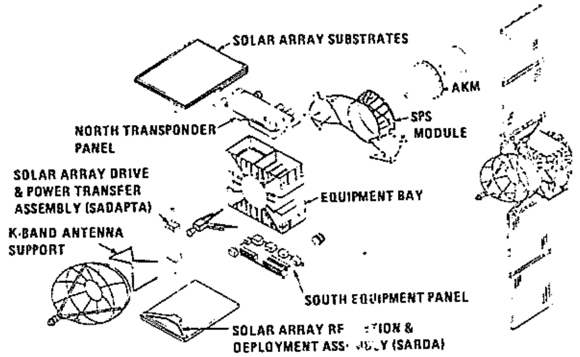

Figure 4. Spacecraft Exploded View

one of which serves as a backup for either primary channel. A modular spacecraft design providing good access and permitting parallel assembly and test activities was implemented as portrayed in Figure 4.

The basic BSE spacecraft communication parameters are defined in Table 1 with additional transponder characteristics detailed in Table 2. The frequency band assignment plan within the 50 MHz and 80 MHz channels is presented in Figure 5.

B. Ground Equipment

TACC/TACS

The TACC/TACS are vital systems which support launch and operation of the BSE spacecraft using the S-band TT&C System. The TACC (Tracking and Control Center) is located at the NASDA Tsukuba Space Center, 60 km North-East of Tokyo, and is equipped with major computer systems for spacecraft orbit determination, attitude determination and spacecraft housekeeping data edition. Spacecraft data will be received from TACS (Tracking and Control Stations) and from NASA STDN. TACS for BSE use are located at Katsuura, 70 km South-East of

Table 1. System Parameters

SATELLITE LOCATION	110° EAST LONGITUDE
EXPERIMENTAL COVERAGE	JAPANESE TERRITORY
FREQUENCY BANDS	14.25–14.43 GHz UPLINK 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/m^2) REMOTE TERRITORY (−117 dBw/m^2)
COMMAND AND CONTROL	S BAND AND K BAND FROM CONTROL STATIONS IN JAPAN

Table 2. Transponder Capability

PFD AT SPACECRAFT	−82 TO −96 dBw/m^2
LEVEL CONTROL	AUTOMATIC OVER 12 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

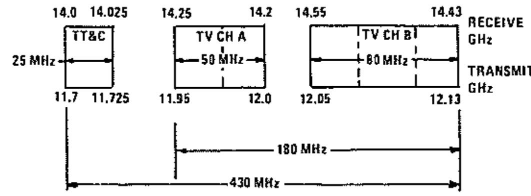

Figure 5. Frequency Plan

Figure 6. The Main Transmitting and Receiving Station

Tokyo and Okinawa, about 1500 km South-West of Tokyo. Both stations are equipped with S-band TT&C Transmitter/Receiver and minicomputers for quick-look analysis of spacecraft status and for command transmission.

MTRS

The MTRS (Main Transmit and Receive Station) is the key user experimental station for the BSE Program and is located at Kashima, 80 km East of Tokyo. The antenna is a 13 m dish with a gain of 61-62 dB in Ku-band. The MTRS is equipped with 2 TV transmitters having a maximum output power of 2 kW and 1 command/ranging transmitter with 200W output power (Figure 6). The MTRS receiver uses a 600°K low noise mixer.

TTRS

The TTRS (Transportable Transmit and Receive Stations) will be used for TV signal transmission and reception at many locations throughout Japan. These stations are equipped with 1 TV transmitter having 2 kW of maximum output power and 2 TV channel receivers with a 910°K system noise temperature including 1 dB for rainfall attentuation effects.

ROS

The ROS (Receive Only Stations) will be used for evaluation of community reception of satellite TV signals. The ROS have two configurations (Figure 7), one with a 2.5 m (smaller) dish antenna for mainland use and the other with a 4.5 m antenna is for remote island use. The system noise temperature of the receivers is less than 660°K.

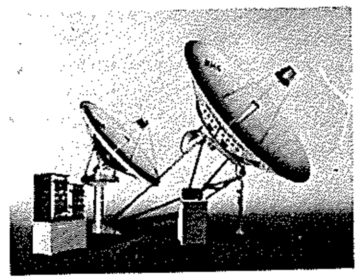

Figure 7. The Receive Only Stations (ROS)

SRE

The SRE (Simple Receive Equipment) has been developed by NHK Technical Research Laboratories for eventual application to future direct TV broadcast satellite systems in Ku-band. The receiver features 500°K system noise temperature over 180 MHz of bandwidth. The antenna is a 1.6 m dish for the mainland and a 4.5 m dish for remote island use.

4. BSE Program Status

The Ground Station Equipment and Spacecraft development and fabrication programs are now essentially completed. Two spacecraft have been built and tested. The command and control stations and BSE peculiar equipment has been designed and is in the final test phase. The fabrication and test of mission ground equipment has also progressed well and elements are already deployed and operating.

The Flight Model is shown in Figure 8 as it is completing the Mass Property Test. As shown, it is essentially in the launch configuration with stowed solar array panels and all external thermal insulation coatings and blankets installed. The BSE spacecraft assembly and test process is described in detail in subsequent sections.

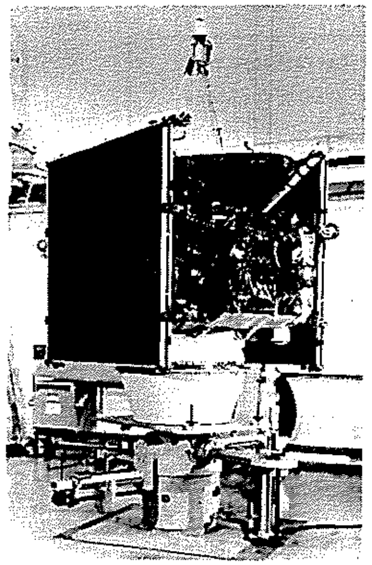

Figure 8. Completed Spacecraft

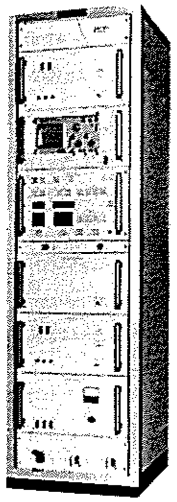

Figure 9. BSE Command Generator Rack

The TACC general purpose equipment is now being used to support the ETS-I and ETS-II operations. The unique BSE support equipment is being added and will be available for BSE training in September 1977. The BSE unique portions of the TACS at Katsuura and Okinawa are also being integrated. The hardware fabrication was completed in March 1977 and total system integration is to be completed by September 1977. Figure 9 shows BSE command generator rack at TACS.

All hardware fabrication and installation has been completed for both the MTRS and TTRS and compatibility tests with spacecraft hardware are scheduled for this summer.

The design and fabrication of the Receive Station equipment and is also well advanced. Figures 10 and 11 show the SHF to UHF converter portion and the FM-AM conversion portion of the SRE, respectively. Similar equipment has already been supplied by Japan for CTS experiments in the U.S. and Canada. All hardware is completed and available for testing.

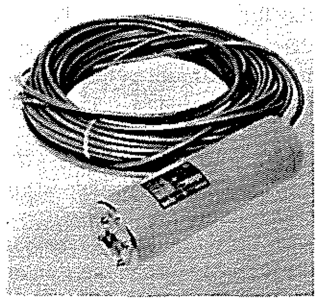

Figure 10. The SHF to UHF Converter Portion

5. Spacecraft Test Program

A comprehensive Test Program was planned and implemented for the BSE spacecraft to provide design confirmation in critical new design areas and to verify the performance characteristics of the Protoflight Model and Flight Model spacecraft. The modular design features of the spacecraft permitted extensive detailed verification at lower assembly levels prior to commitment of hardware to final assembly into the spacecraft. This modular assembly and test technique resulted in improved schedule flexibility in that parallel testing of major spacecraft elements could be carried out. Some of the major tests performed on the modular elements and on the spacecraft are subsequently described.

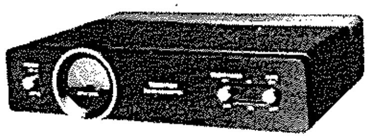

Figure 11. The FM-AM Conversion Portion of the SRE

A. Three Axis Motion Simulator Test

The Attitude Control Subsystem (ACS) of the BSE spacecraft provides attitude control and determination during the transfer orbit spin

mode and highly accurate $< 0.2^{o}$ pointing of the rigidly mounted communication antenna during the body stabilized synchronous orbit mode. The control system design employs Digital Sun Sensors and Horizon Crossing Indicators for the spin mode and an Earth Sensor, Sun Sensors, and a Monopulse Sensor when on-station. Hydrazine precession thrusters are used in attitude control when spinning and a set of orthogonal reaction wheels provide primary control in the de-spun on-station mode.

The control system design was verified via an extensive 3-axis computer simulation with correlative confirmation from a 3-axis Motion Simulator Test Facility (MSTF) using actual Engineering Models of the control subsystem components in a realistic, sensor stimulated closed loop environment.

The equipment used for the dynamic testing is shown in Figure 12. A hybrid computer facility interfaces with the MSTF and simulates the spacecraft inertial and dynamic response to the modelled hydrazine and reaction wheel characteristics.

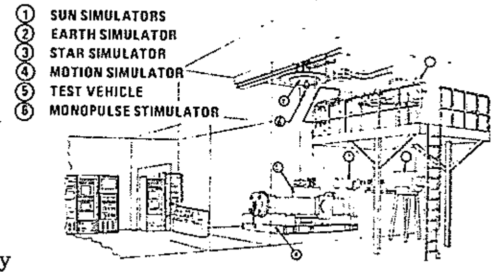

Figure 12. Motion Simulator Test Facility

The actual sensor components are mounted as shown in Figure 13 on the inner gimbal test platform and receive external simulated stimulation. Sensor outputs are processed by the actual ACS Control Electronics (ACE) which in turn operate the control torque simulation in the hybrid computer. The resulting spacecraft motions from the hybrid computer re-position the various gimbals of the MSTF thereby modifying the sensor response. Thus, complete closed loop control is effected.

Each of the spacecraft operating modes were evaluated during the test program including an uninterrupted on-orbit acquisition sequence starting from initial sun stabilization on thrusters and terminating with 3-axis orbit reaction wheel control.

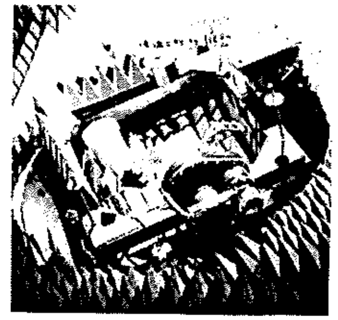

Figure 13. 3-Axis Motion Simulation Test

B. Structural Development Model (SDM) Test

The SDM Test is the qualification static and dynamic test of the primary structure and mechanisms of the spacecraft. It employs a structure built to flight hardware standards to which are mounted mass simulated components. The mass simulated components duplicate the mounting interfaces and the weight and center of gravity of the actual components. The SDM spacecraft is shown being readied for the longitudinal vibration test in Figure 14. The final test configuration includes both array assemblies.

The SDM tests demonstrate the structural integrity of the spacecraft including minimal modal frequencies, strength, alignment and dynamic envelope. The test results are used to confirm the spacecraft dynamic and static load analytical models by providing empirical data on system mode shapes, damping characteristics, and load distributions. Amplification factors throughout the SDM spacecraft are recorded in these tests so as to establish the interface environment for subsystems and components. These data determine the component level dynamic test specifications.

The environmental input to the SDM spacecraft are selected to be 1.5 times the maximum values expected in the actual flight environment. Thus the successful completion of the SDM test on BSE established that sufficient design margin existed in the structure design.

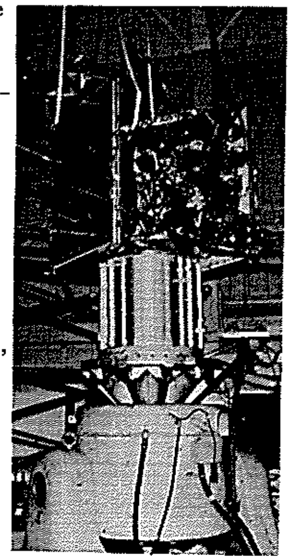

Figure 14. Z-Axis Vibration Test of SDM

C. North Panel Transponder Test

The north panel is subjected to a comprehensive evaluation of the operating characteristics of the communications transponder over temperatures ranging from 0^oC to $+40^oC$. Transponder tests include:

1. DC Power Consumption
2. Operation of K/S Converter
3. Measurement of RF Power Output
4. 100W TWT Drive Characteristics
5. System Gain (up to 100W TWT input
6. Frequency Selectivity
7. Frequency Selectivity
8. Group Delay
9. Noise Figure
10. Inter Modulation Characteristics
11. Susceptibility to High Power Electro Magnetic Interference
12. TV Transmission Characteristics

Two primary pieces of equipment are employed to test the transponder. The Control Unit simulates the spacecraft and provides power, transmits commands, and accepts and conditions telemetry signals. The Radio Frequency Test Unit (RFTU) is employed to inject uplink signals at desired levels into the transponder. The signals represent TV, TT&C, and Order Wire. They may be CW or modulated. The RFTU accepts the transponder output signals (downlink), measures their power level and analyzes their signal characteristics. The North Panel and Ku Transponder is shown in Figure 15 after completion of subsystem testing.

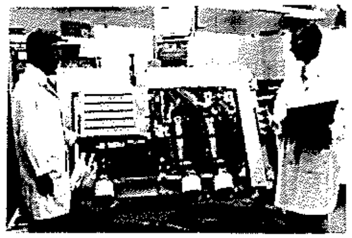

Figure 15. North Panel-K Band Transponder

D. <u>South Panel Integration Test</u>

The Attitude Control Subsystem (ACS), the Telemetry, Tracking & Command Subsystem (TT&C), and the Electrical Power Subsystem (EPS) are primarily located on the South Panel and are verified during South Panel Integration Testing. As shown in Figure 16, the test configuration consists of the South Panel and the Shunt Dissipators from the Solar Array Panels with the latter being required for EPS testing. The South Panel Integration Test sequentially verifies the EPS, TT&C and the ACS. These tests include harness verification, power regulation and ACS phasing. Subsystem functional and performance tests are also included so that any problems can be corrected before the South Panel is mated to the centerbody of the spacecraft.

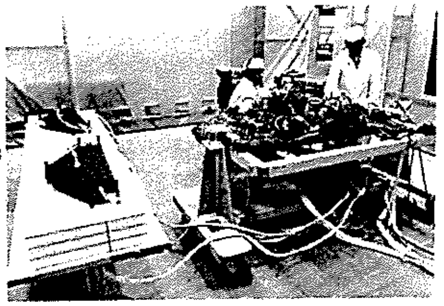

Figure 16. South Panel Integration Test

E. <u>Antenna Testing</u>

The stringent beam shaping requirements on the Ku-band antenna assembly are achieved through the shaped parabolic reflector and a power splitting three horn feed. The reflector and feed characteristics are verified prior to assembly by means of precision dimensional and RF power measurements. The feed phase center is located at the focal point of a best fit parabola for the reflector. After assembly of the reflector, feed, and connecting waveguide assemblies the Voltage Standing Wave Ratio (VSWR) of the antenna is measured on an Automatic Network Analyzer (ANA) in both the 12 and 14 GHz bands. The antenna assembly as shown in Figure 17 is then placed in a 1500 meter Outdoor Test Range to measure the gain characteristics. Gain patterns are obtained for a set of sidelobe suppressor fins and after detailed studies final feed adjustment and suppressor fin selection is accomplished.

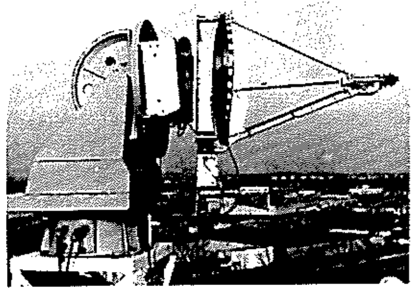

Figure 17. Antenna Range Test

F. <u>Solar Array Deployment</u>

The Solar Array Deployment test confirms that the Solar Array Retention and Deployment Assembly (SARDA) releases and deploys the arrays within the specified time limit. The array deployment also demonstrates that adequate clearance with the spacecraft exists throughout the deployment travel from the stowed position to the fully extended orbital position. The deployments also verify that the pyrotechnic devices operate satisfactorily and that the array elements and spacecraft successfully withstand the resulting shock loads. Deployments are conducted under thermal vacuum and ambient test conditions. For the thermal vacuum test a low-friction suspension support assembly was designed to simulate the zero-g in orbit condition. Air pad support fixtures were used during the ambient deployments which were performed on a smooth granite table. An ambient test is shown with a fully deployed array in Figure 18.

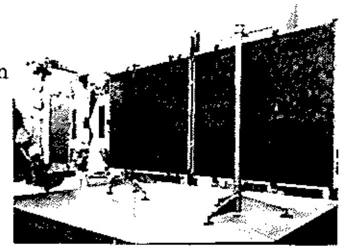

Figure 18. Solar Array Deployment Test

G. <u>RF Compatibility Test</u>

The RF Compatibility Test is conducted in an anechoic chamber and tests the assembled spacecraft for self-generated radiative interference and susceptibility to various configurations of spacecraft transmitters, receivers or due to internal spacecraft mode switching. The spacecraft test configuration employs "dummy" solar array panels of sufficient size to simulate the primary near field reflections which might affect spacecraft performance. The spacecraft is shown in Figure 19 in the anechoic chamber in the compatibility test configuration. In general, all possible operating modes of the spacecraft are tested with particular emphasis placed upon the numerous K-band transponder operating conditions. The tests are conducted under the worst case maximum power generation conditions. Video channels are continuously monitored through subjective and quantitative measurements.

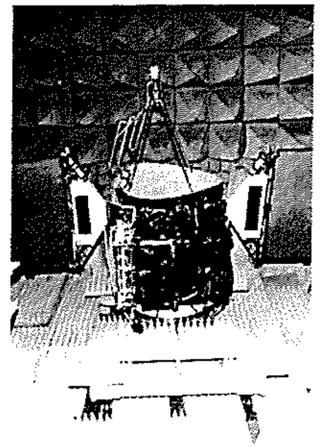

Figure 19. RF Performance Test in Anechoic Chamber

H. <u>Vibration Test</u>

The spacecraft vibration test verifies the structural integrity for launch vibration and pyrotechnic separation shock. The test also confirms that spacecraft stiffness exceeds the minimum longitudinal and lateral fixed free fundamental frequencies. Verification is also provided through these tests that those subsystems and components which must operate during the launch, ascent, and spacecraft separation phases of the mission, will do so satisfactorily. Both random and sinusoidal vibration inputs are provided for the tests on each spacecraft axis. For the tests the spacecraft is mounted on the 3740 Attach Fitting. The separation band used to join the spacecraft and the Attach Fitting is fired to induce the same shock loading which will occur at spacecraft separation. Complete "pre" and "post" vibration test performance data is taken on the spacecraft to determine whether the environment affected the spacecraft hardware.

In Figure 20 the spacecraft is shown on the MC-C220 Vibration Exciter prior to the longitu-

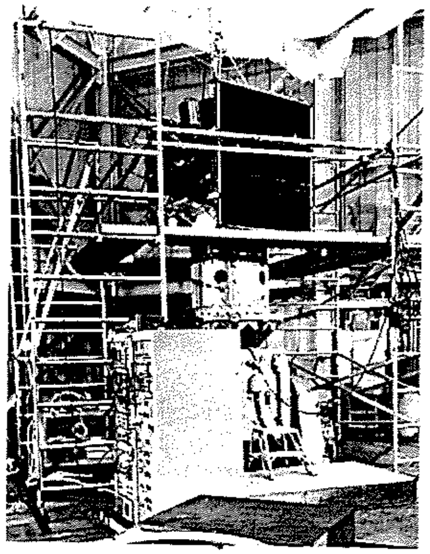

Figure 20. Spacecraft Z-Axis Vibration Test

dinal axis vibration test. The spacecraft is connected via an RF link to the System Test Ground Station where the telemetry performance data is recorded.

I. Thermal Balance and Thermal Vacuum Test

The spacecraft is shown in Figure 21 being lowered into the 17 meter height thermal vacuum solar simulator test chamber. This chamber is used for both the Thermal Balance and the Thermal Vacuum tests.

The Thermal Balance Test is performed only on the Protoflight Model spacecraft as part of the Qualification program. This test confirms the spacecraft thermal design by empirically demonstrating the equilibrium and transient temperatures which occur in the spacecraft for the various spacecraft-solar geometries expected during transfer orbit and synchronous orbit. These measured data are compared with the analytical thermal models which served as a basis for establishing design temperature limits for the various spacecraft components.

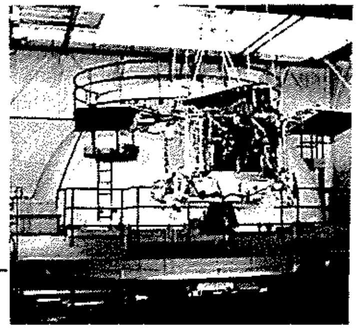

Figure 21. Protoflight Spacecraft Being Placed in Thermal Vacuum Facility

The Thermal Vacuum test consists of a cyclic temperature variation of the spacecraft under vacuum conditions to confirm the integrity of the design and manufacturing processes. Cyclic stressing of spacecraft components in a thermal vacuum environment has proven to be one of the most effective test screens. Both the Protoflight Model and the Flight Model BSE spacecraft were successfully subjected to these tests. Figure 22 shows a schematic representation of the spacecraft in the test facility with the various test orientations being presented in Figure 23.

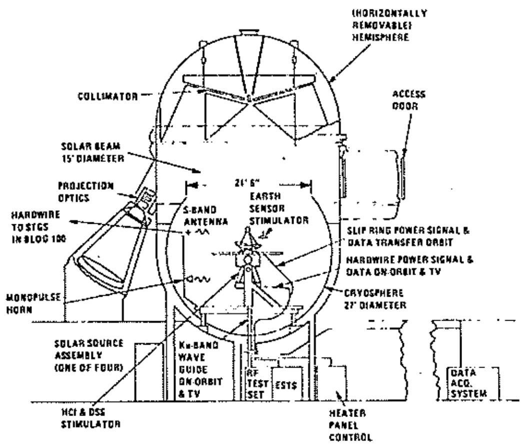

Figure 22. Spacecraft Thermal Balance and Thermal Vacuum Test Set Up

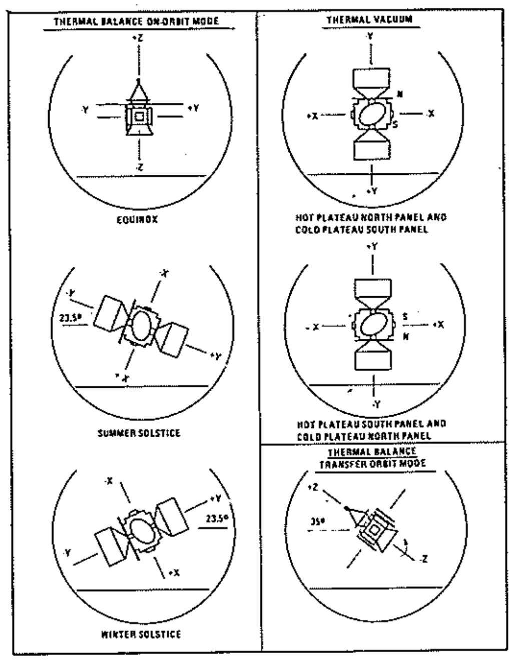

Figure 23. Spacecraft Positional Orientation Thermal Balance and Thermal Vacuum

6. Summary

The current status of the BSE program spacecraft, ground station equipment and software indicates that all will be in readiness for the February 1978 launch schedule. Further, the successful BSE test results which have been achieved during the development and fabrication program provide a good basis for the realization of a meaningful on-orbit experimental program.

The Training and Research Computer-Aided Vehicle Design System

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Abstract

The training and research computer-aided design (CAD) system (first release) in meant for training students and carrying out research in the field of computer-aided vehicle/assembly design. The principles of the CAD system development as well as some problems relating to its structural organization are being discussed in the paper.

1. Introduction

When developing the computer-aided aircraft design (CAD) system intended for conduction the teaching process and research at the higher school a variety of problems arises; the problems in question can be solved only experimentally at present. Moreover, as there is much complex work to do and the time-limits are rigid, flexible enough to provide fruitful research and development of the advanced CAD systems. The CAD system described below possesses the above-mentioned features.

2. The Design Principles of the System.

In the development of the CAD system, the following basic principles determined its configuration and specific implementation.

System flexibility is the ability of the CAD system to realize different design process schedules.

System modularity as the basis of the CAD system principally pertains to its hardware and software.

System readjustability implies the ability of the system to change its structure as well as the set of the modules comprising the system (its program modules, in the first place), or the set of data involved, etc.

The interactive mode is applied at the two phases of the system operation: in preparing a source job and in running an ad hoc object program.

The availability of the standard applied modules. Provisions have been made for the applied program modules corresponsing to the standard stages of vehicle preliminary design. In the CAD system there are also program modules implementing the required algorithms of numerical analysis and optimization. Besides, provision is made for the formation of temporary modules (effectual, as a rule, for the current job only) conforming to such parts of the aircraft design algorithm which have no appropriate program modules in the system library.

The automatic selection of an applied module from the system library implies the possibility of choosing such alternative modules which are indispensable for the execution of the job.

System stability with respect to possible failure of some components of the system is essential, as the interaction with the system involves the change of both the module set and the structure

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