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GSFC PHASE A ANALYTICAL REPORT FOR ATS-F&G

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HARRY L. GERWIN

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(Second Printing)

Compiled by

Harry L. Gerwin

Approved by:

R. A. Stampfl

R. A. Stampfl

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FOREWORD

This Phase A Analytical Report which evolved into a preferred concept for the ATS-F&G spacecraft required the group efforts of many GSFC personnel. This report is a result of the interchange of ideas, discussions, and critiques among a number of individual authors and groups. Consequently, assigning specific credit is very difficult.

The first five sections, page I-1 to page V-19 (colored pages), provide the essence of this report. These sections have been intentionally provided for the convenience of readers whose interest is broad in scope. For readers who are interested in further detail, Section VI, VII and the Appendices are provided.

The complete document was reviewed and coordinated by the ATS-F&G Study Project Office consisting of Harry L. Gerwin, the Study Manager; Joseph V. Fedor, the Project Technologist; Marius B. Weinreb, the Spacecraft Manager; and Aldo A. Merollini, the Project Coordinator.

Sections I, II, and III, consisting of the Introduction, Project Objectives and Project Feasibility, were written by Harry L. Gerwin.

The Potential Experiments, Section IV, was written by Joseph V. Fedor of the Mechanical Systems Branch, Spacecraft Integration and Sounding Rocket Division, and Harry L. Gerwin.

Section V, covering the Functional Description, was written by George Keller of the Advanced Plans Staff.

The Launch Vehicle Selection, Section VI A, and the Apogee Motor, Section VI B, were written by Kenneth Duck of the Auxiliary Propulsion Branch, Systems Division. Mr. Duck is also the co-author of Appendix B concerning the ATS-F&G Attitude Control Transfer Orbit, Apogee Motor Burn, and Earth/Polaris Acquisition.

Section VI D, Computational Support was written by David Stewart and Anthony Durham of the Tracking and Data Systems Directorate.

Reliability, Section VI E, was written by Aldo Merollini in cooperation with Samuel Keene of the Quality Assurance Branch.

Section VII A, Structure, was written by Marius Weinreb and Edwin Stengard of the Systems Engineering Branch, Systems Division.

The Antenna, Section VII B, was written primarily by William Korvin of the Communications Research Branch, Systems Division. Hossein Bahiman of the Communications Research Branch wrote the Thermal Analysis portion of this section. The Feeds and Experiment Package Truss System was written by Hossein Bahiman and John Gates. The Reflector Concepts Section was written by John Gates. Harry L. Gerwin is the author of Appendix A, Antenna Feed Support Mast Concept Selection.

The Thermal Approach, Section VII C, was written by Marius Weinreb in cooperation with Joseph Skladany of the Thermal Systems Branch, Spacecraft Technology Division.

Section VII D, Transponders, was written by Paul Heffernan, of the Communications Research Branch. He is also the author of Appendix E, Communications Experiments.

Section VII E, describing the Attitude Controls, was written by James Gatlin of the Stabilization and Controls Branch, Systems Division. He is also the co-author of Appendix B concerning the ATS-F&G Attitude Control Transfer Orbit, Apogee Motor Burn, and Earth/Polaris Acquisition and of Appendix C, Closed Loop Interferometer and of Appendix I, Gimbaled Gravity Gradient Boom.

Thomas Cygnarowicz of the Auxiliary Propulsion Branch, Systems Division, wrote Section VII F, Auxiliary Propulsion.

Section VII G, Electric Power, was authored by Marius Weinreb of the Systems Engineering Branch, Edwin Moses of the Space Power Technology Branch, Spacecraft Technology Division.

Telemetry and Command Logic System, Section VII H was written by Joseph Silverman of the Systems Engineering Branch.

Section VII I, Spacecraft Environment, was written by G. L. Coble of Test and Evaluation Division.

Ground Support, Section VII J, was written under the direction of Anthony Durham of the Tracking and Data Systems Directorate (T&DS) in cooperation with Howard W. Shaffer and Thomas Grenchik of T&DS. This group also wrote Appendix D, Link Calculations for Tracking, Telemetry and Command.

The Electric Propulsion Experiment, Appendix F, was written by Robert Hunter, Head of the Auxiliary Propulsion Branch.

The Interferometer Experiment, Appendix G, was authored by David Nace of the Communication Research Branch.

The SCADS Experiment, Apendix H, was written by Irving Lowen of the Systems Engineering Branch.

The Phased Array Experiment, Appendix J, was written by Thomas S. Golden, Head of the Antenna Systems Branch of T&DS.

Appendix K, Reflector Antenna Beam Scanning Experiment, was written by Milton K. Mills of the Communications Research Branch.

The LOCAST Experiment, Appendix L, was authored by Charles R. Laughlin of the Systems Engineering Branch.

Appreciation is expressed to Karl Plitt of Materials Research and Development Branch, Systems Division, for his contributions and suggestions in the materials areas of the spacecraft subsystems for the preferred approach.

In addition, appreciation is expressed to R.A. Stampfl for his technical consultation, critique, and review in overall systems engineering.

SECTION I

INTRODUCTION

This report discusses the feasibility and practicability of a 30-foot-diameter, deployable spacecraft antenna, of providing spacecraft fine pointing (0.1 degree) and slewing (17.5 degrees), and of providing an oriented spacecraft in synchronous equatorial orbit for advanced technology experiments. The report is based on the work conducted by three mission study contractors and supplementary studies made by the Goddard Space Flight Center (GSFC). The three mission study contracts were awarded by NASA Headquarters, and technical direction was provided by GSFC. The three study contractors were General Electric Company, Valley Forge, Pennsylvania; Fairchild-Hiller Corporation, Germantown, Maryland; and Lockheed Missile and Space Company, Sunnyvale, California. The GSFC study was undertaken to supplement the effort of the mission study contractors, and to establish the nucleus of an in-house team technically well updated and knowledgeable in the problems related to the ATS-F&G program.

The four studies all produced similar technical conclusions in many areas associated with the program, but different conclusions for others. The four reports agree that the validity, feasibility, and practicability of mission execution have been proven. These aspects are discussed in the two sections entitled Project Objectives and Project Feasibility. On the basis of the material contained in these sections and more detailed treatment in other sections of the report, it is concluded that the project is considered feasible and worthy of further definition.

The progress made in Phase A (Advance Studies) has carried the program into Phase B (Project Definition). Extensive trade-off studies have been conducted in all the areas where specific technical approaches must be selected. As a direct result of the contractor and the in-house Phase A studies, trade-off conclusions yielding single approaches were established for many technical areas. Those problems not resolved directly by such studies were assessed in the period following the completion of the contractor work. These assessments of the remaining technical problems have yielded a system approach.

The approach which is preferred at this time, called the preferred system, is described in Section V. The trade-off analysis for the approach selected for each area in the preferred system is described in Sections VI and VII. Launch vehicle trade-off is given in Section VI A. This preferred system will provide a specific reference for continuing system design effort at GSFC during the administrative lead time period between the submission of the Phase A analytical report and the date that contracts are awarded and during the first few months of the contractor effort. Design and analysis during the stated period of time offers the opportunity to investigate further alternate subsystem approaches and to optimize others. It is thus anticipated that a firm spacecraft design will evolve rapidly. While most optimization can be arrived at by detailed design, some subsystems or parts thereof will have to be breadboarded or modeled. By necessity, these latter ones will advance at a different pace than those designed on paper. It has been recognized that there is a need for interlacing the design of subsystems in this fashion and advancing design at an optimum pace. Therefore, it is recommended that Phases B and C be combined into one phase for this program.

SECTION II

PROJECT OBJECTIVES

Stated briefly, the ATS-F&G project objectives are to

1. Demonstrate the feasibility of a 30-foot-diameter, deployable spacecraft antenna with good RF performance up to 10 GHz

2. Provide spacecraft fine pointing (0.1 degree) and slewing (17.5 degrees in 30 minutes)

3. Provide an oriented, stable spacecraft at synchronous altitude for advanced technology experiments.

The spacecraft design is dictated largely by the first two objectives. Provisions are being made for additional experiments which require an oriented spacecraft in synchronous orbit. Experiments which are being considered (subject to Headquarters approval) are discussed in Section IV.

For the demonstration of a 30-foot-diameter, deployable spacecraft antenna, the antenna reflector will be folded to fit inside a launch shroud. When deployed, it will form a reflecting parabolic surface of sufficient accuracy to provide proper performance for frequencies up to 10 GHz (X-band). The feed will be capable of performing at several frequencies. To provide error control signals in roll and pitch, monopulse feeds at X-band and at UHF will be provided. At X-band, the error signal accuracy will be 0.01 degree.

Since antenna pattern beamwidths in the order of 0.3 degree are feasible, a necessary and complementary technology is that of precise spacecraft attitude control. Attitude control is necessary for quick pointing toward predetermined RF energy sources or receivers located on the earth surface or in space. This control technology involves the use of angle-error detection devices such as earth sensors, RF angle-measuring sensors, and star trackers. Furthermore, control is concerned with the technology of forcing systems such as microthrust impulse thrusters, high performance inertia wheels, and gimbal gravity gradient booms. It also involves the design of spacecraft which have structural dynamics response characteristics compatible with three-axis closed loop precise attitude control. The magnitude and effect of the perturbing forces which the control system must correct can be predicted. The noise and error sources in the attitude sensors can be estimated. These factors cannot be completely measured on earth since all the environments cannot be duplicated. Therefore, a significant part of the control system operational program is to determine, by operational measurements, what aspect of the control problem limits precision pointing.

SECTION III

PROJECT FEASIBILITY

Goddard Space Flight Center considers that, based on the absence of any known scientific limitations which require research, it is feasible to achieve the objectives set forth in Section II of this report. There are, however, many elements of the problem which pose engineering challenges requiring the highest quality engineering talent. The feasibility of the project can be illustrated best by examining three specific elements of the problem.

1. Launch vehicle capability

2. Antenna and feed feasibility

3. Control system feasibility

To determine the total spacecraft weight that could be put into synchronous orbit, it was necessary to establish the total weight that the selected launch vehicle, SLV-3C Atlas-Centaur, could put into the transfer orbit. Total weight was established at 4000 pounds when the existing Surveyor shroud is used. Since lengthening of the Surveyor shroud would be necessary to provide adequate volume for the ATS-F&G spacecrafts, it was also established that each 1-foot extension of the shroud would reduce the 4000 pounds reference weight by 5 pounds. Based on these constraints, the payload weight into synchronous orbit was calculated to be 1797 pounds for the GSFC Concept Design Study and varied from 1767 to 1675 for the three mission contractor studies.

Following these studies, a number of new variables and changes have been identified. The shroud trade-off weight, which was 5 pounds, has changed to 7.6 pounds. It has been determined that a 16-pound destruct system must be added. Some inconsistencies existed in previous GSFC and contractor studies relating to the weight assignment of the Centaur payload adapter. These inconsistencies have now been resolved and it has been determined that 1805 pounds could be injected into synchronous orbit with the system described in this report. This results in an on-station spacecraft weight capability of 1389 pounds after jettisoning the adapter. The total weight of the spacecraft as now configured, including the antenna and feed, is 1152 pounds. This provides for 237 pounds of weight growth or additional experiments. It is anticipated that studies by Lewis Research

Center will show that the 4000 pounds of weight can be increased 200 to 400 pounds, yielding a 35 to 65-pound increase in spacecraft weight. In addition, an optimized kickstage could provide an additional spacecraft growth weight of about 200 pounds. Based on these data, it is considered completely feasible to design a spacecraft with the weight capability of the SLV-3C Atlas-Centaur. It should be noted that the allowance for experiment weight and weight uncertainty in the present state of design is small.

The feasibility of manufacturing and deploying a 30-foot parabolic reflector can best be examined by considering manufacturing accuracy, deployment reliability, and reflector accuracy when subjected to the environments of space. Parabolic reflectors can and are being manufactured to a surface accuracy of 0.050 inch. Thus, there should be no reason why these accuracies cannot be maintained in the manufacturing of a deployable antenna. The hinged petal configuration, deployed in a one-operation driving sequence, has been selected because of its inherent reliability. The flight model antenna deployment system will be cycled through its deployment sequence several times to prove its operability before it is flown.

The surface accuracy achieved with such structures when subjected to the environment of space can only be measured in space. Particular emphasis was placed on this problem by a review in depth of the work carried out by Goodyear and General Electric in their ATS-F&G Phase A mission study. Both companies conducted relatively complete computer analyses, starting with heat flow as the input condition and RF gain and pattern plots as the outputs. Several computer programs were required to conduct the analyses. A program was used to compute the thermal gradient and temperature contour on the reflector due to heat flow. This included shadowing effects as the spacecraft traveled through its 24-hour orbit. From these gradients and temperature contour, a second program computed the surface distortions of the reflector. A third program computed antenna gain and pattern using the reflector shape previously calculated. These computer programs made it possible to construct and test various design concepts to arrive at an optimized paper redesign. Computer evaluation of the redesign indicated that, under the worst conditions, the loss in gain resulting from thermally induced deformation was 0.2 db at 8 GHz.

While the antenna and feed system are considered feasible, they will pose the most challenging engineering problem in the ATS-F&G project. The principal obstacle is the inability to simulate zero g, solar radiation, and vacuum simultaneously. Since tests are limited to those which can be conducted with available facilities, it will be necessary to extrapolate from the test results to the expected performance in space. This approach applies to the correlation of results from structural vibration tests conducted in a vacuum to the dynamic performance of the spacecraft as it will affect spacecraft control performance in space. It also applies to correlation between structural distortion of the antenna as measured in a thermal test chamber under a 1-g field and antenna performance under flight conditions.

An orbital control system capable of meeting the 0.1-degree pointing requirement is considered within the state-of-the-art. Goddard has considerable experience programming such control systems on analog and digital computers. This technique, along with analytical analysis, was used to examine a variety of proposed control systems. A limiting element in such a control system is the attitude measuring transducer. It appears that an earth sensor will satisfy the 0.1-degree pointing requirement. However, either an interferometer sensor or a monopulse sensor utilizing the large dish will permit control system performance better than a 0.1-degree pointing requirement. In no case has the slewing requirement been considered a serious problem. Based on these factors, it is considered that the orientation control system required for the ATS-F&G is completely feasible.

The foregoing discussion pertains to the three major specific elements of program feasibility. Table III-1 completes the feasibility assessment by showing the concepts considered in selecting a preferred approach. This preferred approach is shown in the second column of the table. The three mission study contractor concepts are shown in columns 3, 4, and 5. Column 6 shows the concept selected in the Goddard Space Flight Center Concept Design Study. All the individual concepts selected by the three contractors and by Goddard Space Flight Center were arrived at independently by conducting trade-off studies of a broad selection of concepts. Sections VI and VII of this report discuss these trade-offs in detail. Therefore, the comments which follow on Table III-1 are limited to a brief discussion of logic for selecting the approach indicated in the preferred system or the logic for rejection of a specific concept.

The preferred antenna reflector approach which has now been selected as the final approach is the side-hinged petaloid concept. The petaloid approach was selected over flex-rib and other concepts because it provides a more accurate reflecting surface. The side-hinged petaloid

Table III-1

Analysis of Proposed Design Concepts					
Parameter	Preferred	GE	F-H	LK	GSFC
Antenna concept	Petaloid side-hinge	Petaloid side-hinge	Petaloid	Flex-rib	Petaloid side-hinge
F/D	0.44	0.4	0.32	0.4	0.5
Deploy. mech.	Motor drive	Motor drive	Springs	Motor drive	Motor drive
Feed support	Fixed-truss	Fixed-truss	Fixed-truss	Boom deployable	Fixed-truss
Antenna thermal design	Coatings	Coatings	Thermal blankets	Coatings	Coatings
Control concept	Wheel-jet hybrid pulsed jet backup	Wheel-jet hybrid	Wheel-jet hybrid	Pulsed jet	Wheel-jet hybrid pulsed jet backup
Sensors	Earth, Polaris, interferometer monopulse	Earth, Polaris, interferometer	Earth, Polaris, interferometer monopulse	Coarse: earth inertial gyro Fine: interfer.	Earth, Polaris, interferometer monopulse
Torquers	Reaction wheels NH_3 resistojet	Reaction wheels NH_3 resistojets	Reaction wheels N_2H_4 monopropellant	Coarse: NH_3 resistojet Fine: ion engine	Reaction wheels NH_3 resistojet
Station-keeping	NH_3 resistojet	NH_3 resistojet	N_2H_4 monopropellant	Ion engine	NH_3 resistojet
Transfer orbit control	Spinning	Spinning	Spinning	3-axis	Spinning
Number of transfer orbits	2nd apogee	2nd apogee	2nd apogee	1st apogee	2nd apogee
Sensors	RF polar. and sun SCADS	Sun and RF polar.	No spin axis control	Inertial gyro with earth sensor	RF polar. and sun SCADS

Table III-1 (continued)

Parameter	Preferred	GE	F-H	LK	GSFC
Torquers	Hydrazine	Hydrazine	Hydrazine	Hydrazine	Hydrazine
Spinup	Solid rockets	Solid rockets	Solid rockets	N/A	Solid rockets
Despin	Yo-yo	Hydrazine	Yo-yo	N/A	Yo-yo
Power System	Solar conversion	Solar conversion	Solar conversion	Solar conversion	Solar conversion
Solar array	Fixed	Fixed	Fixed	Oriented	Fixed
Batteries	NiCd	NiCd	NiCd	NiCd	NiCd
Structural concept					
Launch vehicle	SLV-3C Atlas-Centaur	SLV-3C Atlas-Centaur	SLV-3C Atlas-Centaur	SLV-3C Atlas-Centaur	- - - - -
Apogee kickmotor	TE-364-3	TE-364-3	TE-364-3	TE-364-3	TE-364-3
Shroud	Surveyor extended 15 feet	Surveyor extended 15 feet	Surveyor extended 12 feet	Surveyor extended 2 feet	Surveyor extended 15 feet
Station location	Initial 53° W Operational 100° W	Initial 53° W Operational 110° W	Initial 53° W Operational 110° W	Initial 100° E Operational 147° W	Initial 53° W Operational 100° W

approach was selected over the bottom-hinged because of its inherent higher reliability. Also, the fixed-truss feed support was selected over the deployable boom because of its higher reliability. Analysis to date indicates that thermal coatings will keep the reflector temperature profile within tolerable limits. Thermal blankets, although they have an inherently better performance potential, have limitations such as performance degradation due to insulation crushing when subject to deployment flexing, additional weight, cost of installation, and additional handling problems.

Ion engines for spacecraft attitude control and station-keeping were rejected. These engines represent no particular weight advantage over reaction jets, and within the present state-of-the-art reaction jets have a much higher proven reliability. It then follows that the highly successful reaction wheel jet hybrid was selected to provide attitude fine pointing control. The interferometer was rejected as the roll, pitch, and yaw attitude sensor because it is an experiment and the primary operating mode of the spacecraft should not depend on an experiment for attitude sensing. The earth sensor and the Polaris tracker to measure spacecraft roll, pitch and yaw angles for attitude control were selected because of their proven reliability. The interferometer and monopulse experiments will serve as backup attitude sensors.

Spinning control was selected over three-axis control while the spacecraft is in the transfer orbit because the spinning control mode has fewer in-line operating functions than the three-axis control. Therefore, the spinning control mode is more reliable. The weight difference between the two approaches, while the spinning mode is favored slightly, is not considered significant. Both General Electric and Goddard Space Flight Center selected the flight-proven RF polarization and sun sensor angle measuring technique. Goddard Space Flight Center proposed SCADS, a star mapper, as a supplementary measuring technique since it offers higher accuracy than the other approaches. This sensor will be flight tested on ATS-C. Because of higher accuracy and simplicity this sensor is included in the preferred approach. Injection into synchronous orbit in the second apogee is preferred because injection occurs near the desired station location. Injection in the first apogee is required with the three-axis control mode because attitude data stored in the gyroscopes will have degraded to an unacceptable value in the time required to go to the second apogee.

As noted before, the fixed-truss structural concept proposed by General Electric, Fairchild-Hiller, and Goddard Space Flight Center was selected because of its higher reliability. The General Electric and

Goddard Space Flight Center configuration was selected in preference to that of Fairchild-Hiller because failure of the adapter to jettison would abort the entire mission. The General Electric approach retained the adapter. The GSFC jettisoned the adapter but failure of jettison to occur would only have a minor effect on spacecraft performance. In addition, the General Electric and Goddard Space Flight Center structural concepts allow for more versatility in spacecraft growth.

In conclusion, based on the three specific elements which were discussed at the beginning of this report and data in Table III-1, it is considered that the feasibility of achieving the objectives set forth in this program have been proven.

SECTION IV

POTENTIAL EXPERIMENTS

The intent of the Application Technology Satellite Program, of which ATS-F&G is a part, is to develop progressively the technology required for various spacecraft applications of the future. The technology for spinning satellites is being developed by ATS-B and ATS-C; the technology for passive gravity gradient systems is being developed by ATS-A, ATS-D, and ATS-E. It is the intent of the ATS-F&G project to develop the technology for the deployment, three-axis control, and utilization of large, high-gain antenna systems, and to provide an oriented spacecraft at synchronous altitude for advanced technology and scientific experiments.

In addition to the primary antenna and the three-axis fine control technology experiments, which were covered earlier, two more experiments were specified in the scope of work issued to the mission study contractors by Headquarters. They were the interferometer and self-steered phased array communication repeater. These two experiments are briefly described along with others in which there is high interest. A more detailed description of the experiments is provided in Appendixes E through L. Technology proposals for these experiments are being prepared for submission to Headquarters. No implication is intended as to the approval or exclusive desirability of these experiments since it is anticipated that a significant number of additional candidate experiments will be proposed during the next phase of this program.

INTERFEROMETER

The interferometer is a radio-frequency sensor that measures spacecraft attitude (three-axis) during the mission phase. Interferometer technology is well established and essentially all interferometer parameters can be tested on the ground. However, to evaluate a closed-loop servo system in which the interferometer is the source of the attitude error used to control a spacecraft in orbit, the interferometer must be flown. Flight test is required to qualify the interferometer as an operational component. The basic sensor consists of two antennas with phase centers separated by a fixed distance, and electrical phase-comparing elements. A measurable phase shift occurs between the RF signals received by each antenna. The amount of phase shift depends upon the direction from which the RF signal is received. When orthogonal pairs of antennas are appropriately located on the spacecraft, the phase shift indicates spacecraft attitude. The interferometer can also be used to generate error signals for attitude control of the spacecraft.

SELF-STEERING PHASED ARRAY COMMUNICATION REPEATER

This is a microwave repeater consisting of a self-phasing antenna system that automatically forms high-gain beams to receive and retransmit microwave communications signals between two or more earth stations. At least one version of the phased array concept is relatively new, but like the interferometer its performance parameters can be tested on the ground. It would be flown to demonstrate communication to small terminals in conjunction with beam steering, and to qualify it as an operational component in a communications system. In operation, the stations acquire the beams by sending a cw pilot signal to the satellite (repeater) which has an RF pattern encompassing the earth. The antenna electronics then direct the beam toward that station and the beam remains fixed to the station as long as the cw pilot is in operation. A wideband signal can be received, amplified, and converted to another frequency for re-transmission. Another ground station can receive the signal by also transmitting a cw pilot signal to the repeater. This pilot signal causes the wideband signal sent by the first station to be re-transmitted along a narrow beam to the second station. Another satellite can also transmit data to the earth via the phased array.

The phased array experiment and 30-foot dish beam-scanning, mentioned later in this section, have performance characteristics that are overlapping in some areas and different in others. For instance, with a 10-watt spacecraft transmitter tube, a phased array similar to the X-band engineering model, developed with SRT funds, will provide an effective radiated power (erp) of 36-40 dbw. With the same transmitter power the 30-foot diameter dish will provide an erp of 60 dbw. In the receiver role, the 30-foot dish provides approximately 14-18 db better performance than the phased array engineering mode. The phased array is capable of communicating with numerous ground transmitting and receiving stations. The 30-foot dish in the present state-of-the-art is limited. Beam-scanning of the dish is limited to approximately ±3 degrees at X band and ±9.5 degrees at S-band. The engineering model phased array can operate through an angle of ±15 degrees. It appears unlikely than an operational system would employ both the phased array and the large dish because they compete directly for the same RG aperture area.

SCANNING CELESTIAL ATTITUDE DETERMINATION SYSTEM (SCADS)

This is a star-mapping system that provides a simple means of determining spacecraft attitude information for ATS-F&G during the transfer ellipse mode (spin-stabilized) and during the mission mode (three-axis stabilized). The principle used has been demonstrated at night on the ground. Because of light scattering dust particles in the atmosphere, a daylight demonstration cannot be made. An actual test flight in space is required to test this item and to qualify it for operating systems. SCADS consists of a single on-board device and a ground-based data-reduction system. The sensor device, by observing the generally-known star field, provides signals from which the viewed star field can be positively identified and three-axis vehicle attitude information derived.

INERTIA WHEEL-GRAVITY GRADIENT BOOM HYBRID

This is a gain-active control system with a gimbaled two-degree-of-freedom gravity gradient boom. This system can be tested only in a very weak gravity field; thus, space flight is necessary to carry out the experiment. It can serve as a source of reaction torque for attitude maneuvers, and as a source of external torque for preventing wheel saturation and for minimizing momentum storage requirements. The control system provides its own damping, so that an additional passive damper is not required.

COLLOID MICRO-THRUSTER

This is an electrostatic thruster in the same general category as ion thrusters: that is, exhaust beam kinetic energy is obtained by accelerating charged particles by an electrostatic field. Tests in space are required to determine the effect of a zero gravity field on the fuel feed system, to determine the degree of electrical charge build-up around the spacecraft caused by thruster operation, and to accurately measure the thrust under space flight conditions. The colloid thruster differs from the ion thruster in that the charged particles are multi-molecular rather than atomic. The propellane, which is a moderately-conductive liquie with a low vapor pressure is sprayed from the tips of metallic capillary tubes by a high electrical potential. The need for exhaust beam neutralization is eliminated by having the capillary tubes produce both positive and negative charged particles simultaneously from adjacent tubes. The thruster would have a nominal thrust level of 200 micro-pounds, specific impulse of 800 to 1000 seconds, and would require less than 20 watts of power. The thruster could serve an operational function such as north-south stationkeeping.

COMMUNICATIONS EXPERIMENTS AND DEMONSTRATIONS

A large number of communications technology experiments and demonstrations have been suggested for ATS-F&G. They include such experiments as: TV and FM broadcast, "man-pack" satellite communication, weather forecast/facsimile broadcast, satellite-to-satellite data, relay, etc. Some of the suggestions have potential as candidate experiments, others do not. Table IV-1 is a summary of suggested ATS-F&G experiments involving small terminals. Although some of the experiments are considered impractical, they are also included, at the end of the table.

No single spacecraft can be designed to handle the total frequency spectrum required by the experiments. Good management practice then indicates that limited objectives should be set for ATS-F to suit the alloted time and resources. Therefore, the following frequencies have been selected for ATS-F: UHF, S-band, and X-band. Since ATS-G is scheduled later, different frequencies reflecting different experiments can and should be used with this spacecraft.

LOCATION OF AND COMMUNICATIONS WITH AIRCRAFT BY SATELLITE TRANSPONDER (LOCAST)

This experiment uses a specially-designed transponder in the satellite, a single ground control facility, and an aircraft equipped with an L-band antenna. The LOCAST experiment would define the requirements and parameters of an operational air traffic control system capable of worldwide application. This system would provide two-way voice and digital data communications between all cooperating aircraft and their associated ground control facilities by means of a transponder on a satellite at synchronous altitude. Real-time surveillance over all aircraft would be provided through continuous position-tracking and automatic reporting from telemetry sensors on board the aircraft.

S-BAND BEAM SCANNING

This experiment will investigate the potential of an S-band antenna system with a 40-db gain and a 1.3-degree 3-db beamwidth. The first part of the experiment determines if the deployable 30-foot diameter parabolic reflector can efficiently produce a secondary beam. The second part of the experiment determines the antenna's capability to scan beyond the limits of the earth's disk (that is, without moving the reflector). Communication experiments will also be performed to determine the antenna system's capability to rapidly switch beams between one or more widely-spaced ground station or other spacecraft. Design of the S-band scanning system is described in Section VII B.

Table IV-1

Summary of ATS-F&G Experiments Involving Small Terminals			
ATS-F&G Application	Spacecraft RF Power in Watts (if applicable)	Design Objective Source or Reference System	Margin with respect to Design Objectives or Reference System; General Comments
FM-TV relay to small central receivers (system 3-466MHz)*	40	CCIR standards for TV relay systems	+4.1 db; frequency allocation problems for this service would be very severe**
FM-TV Relay to Small Central Receivers (system 4-7.3 GHz)*	24	CCIR standards for TV relay systems	+4.9 db; a promising ATS-F&G application**
FM-TV Relay to Small Central Receivers (system 5-860 MHz)	100	CCIR standards for TV relay systems	+0.4 db; coordination with existing services seems feasible
UHF FM voice broadcast to specialized home receivers	40	Applicable military standards for 4 KHz UHF FM voice broadcast	+10.0 db; an ATS-F&G application worth serious consideration
X-band FM communications (system 2-7.3 GHz)	24 (for ten duplex channels)	Applicable military standards for voice communications	+1.7 db; a promising ATS-F&G application; area coverage limited
UHF air traffic communications and control	40 (per channel)	ATS-B links with aircraft at VHF	+8.0 db; a poor frequency for the proposed service
S-band air traffic communications and control	1.0	ATS-B links with aircraft at VHF	+11.3 db; experiments performed at 1.7 GHz could be used to justify use of the 1.5 GHz band for this service

* Analysis given in Reference A and results only quoted here.

** ATS-F&G antenna efficiency of 50% assumed in these applications.

Table IV-1 (continued)

ATS-F&G Application	Spacecradt RF Power in Watts (if applicable)	Design Objective Source or Reference System	Margin with respect to Design Objectives or Reference System; General Comments
Satellite-to-satellite IRLS data relay	---	Platform-Nimbus link	Nimbus erp of 12.3 dbw at 401 MHz would be required
Satellite-to-satellite video data relay	10	Nimbus-Rosman S-band link	4 x 10^5 bps feasible
Satellite-to-satellite Apollo launch phase data relay	12 (assumed)	----------------	4 x 10^5 bps feasible **
Position-location systems	2.6	Proposed OPLE system	+12.2 db; platform power could be reduced an order of magnitude
UHF communications (System 1 - SSB with compandors at 466 MHz)	40 (for ten duplex channels)	Applicable military standards for voice communications	-0.3 db; a generally attractive ATS-F&G application
Direct broadcast TV (System 1-466 MHz) *	40	Downgraded CCIR standards for TV relay systems	-31.5 db; completly unfeasible with present ATS-F&G RF power levels **
UHF TV Relay to Small Central Receivers (System 2-466 MHz) *	40	CCIR standards for TV relay systems	-19.3 db; at best, a marginal operation; frequency a problem
FM voice Broadcast to Home Receivers with UHF tuners (466/860 MHz)	40	FCC field strength requirements for home receivers	At least 20 db more spacecraft power required for quality service.
UHF FM Voice Broadcast to Home TV Sets, Audio Sections only	---	50 db output ratio	Spacecraft RF levels in the kilowatt range required for quality service.

* Analysis given in Reference A and results only quoted here.

**ATS-F&G antenna efficiency of 50% assumed in these applications.

SECTION V

FUNCTIONAL DESCRIPTION

INTRODUCTION

The purpose of this section is to provide an overall view of the ATS-F&G systems in brief form for readers whose interest and responsibilities are broad in nature. A description of the spacecraft is presented, followed by a description of the mission from launch through spacecraft operation and thence through ground operations. Readers requiring detailed information on subsystems, components, and trade-off selections will find such information and data in Sections VI and VII.

SPACECRAFT CONFIGURATION

The preferred approach to the spacecraft is illustrated in Figures V-1 and V-2. Figure V-1 shows the spacecraft in the launch configuration. The parabolic antenna and solar array are folded to fit into a modified Surveyor shroud. In the folded position, the solar array and antenna are supported to tolerate the launch environment and to limit deflections in order to prevent contact with the shroud. The weight of the spacecraft (Atlas-Centaur capability) in this configuration, including the TE-364 and adapter complex, is 3254 pounds. This is the configuration which will be spin-stabilized prior to antenna deployment. Hydrazine thrusters and fuel used during orbit acquisition are located in the adapter section.

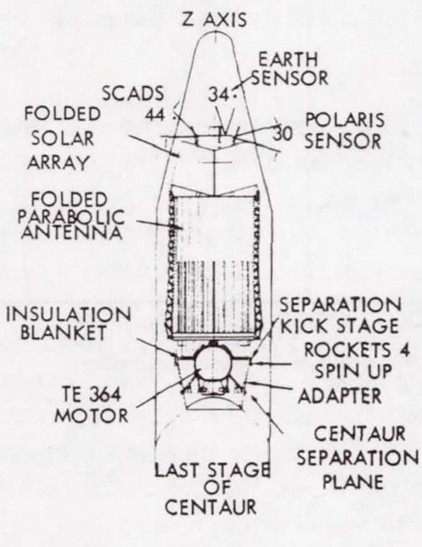

Figure V-1. ATS-F&G launch configuration.

Figure V-2 shows the spacecraft fully deployed. This is the operational configuration. In this mode the spacecraft weighs 1392 pounds and the overall dimensions are 42 feet across the solar array tips and 20.5 feet along the antenna feed axis.

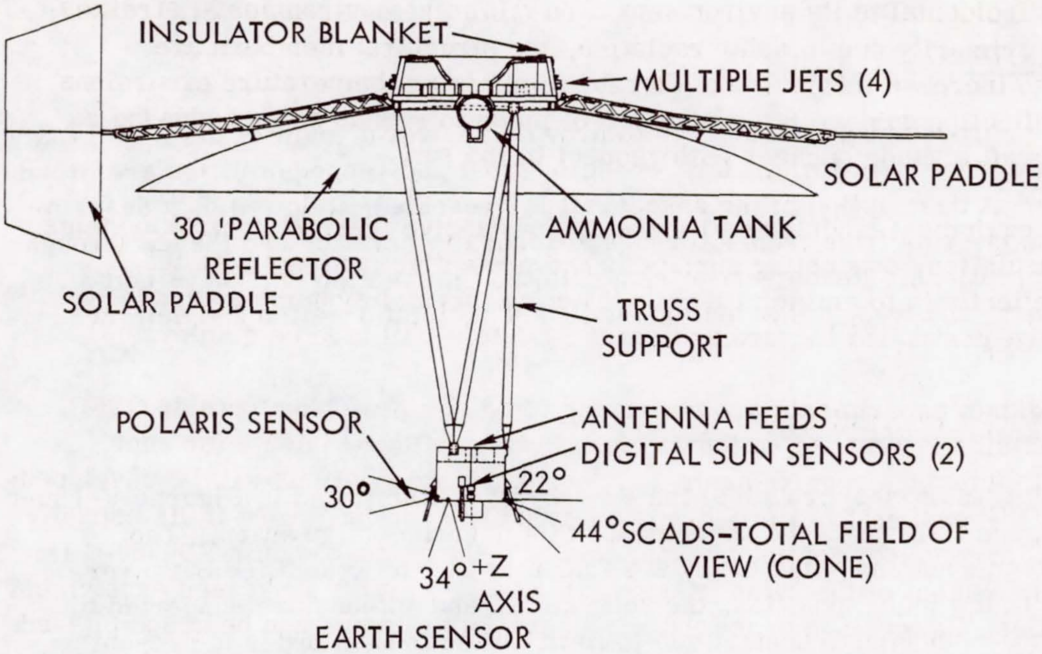

Figure V-2. ATS-F&G orbital configuration.

The spacecraft consists of an aft-equipment module, connected by a tubular truss mast to an earth viewing equipment module. The 30-foot diameter parabolic antenna and the solar array are attached to the aft-equipment module. Equipment and experiments will be mounted in both the earth viewing and aft modules.

The solar array location minimizes shadowing by the antenna. Both sides of the solar paddles are covered with solar cells. The paddles are fixed in the optimum position for solar energy collection.

The batteries, power conditioning, momentum wheels, and electronic components are contained in the aft module. The aft-equipment module also houses the vernier and attitude propulsion systems and propellant tanks. Attached to the aft-equipment module is the adapter section, which contains the apogee motor and the hydrazine propulsion system. The adapter section is jettisoned during spacecraft deployment prior to the yo-yo despin maneuver. The earth viewing equipment module contains the feeds for the main reflector; the earth, sun, and star sensors; and the earth viewing experiments.

The parabolic antenna is composed of petals hinged at the hub and to each other. The relaxed shape of the petals is that of a true section of a parabola. This concept permits the antenna to maintain its shape without stresses other than those incidental to its environment. To minimize environmental stresses, which are primarily due to solar radiation, the structural members are insulated to increase thermal lag, thereby minimizing temperature excursions. The RF reflecting surface is constructed of mesh to reduce self-shadowing as the spacecraft attitude changes with respect to the sun.

The equipment modules will make use of passive thermal control systems such as insulation, conducting surfaces, and protective coatings with the desired reflectivity to emissivity ratio. Active thermal systems such as temperature-controlled louvers and shunt resistors will also be employed.

Candidate experiments are continuing to be solicited from outside GSFC. When a decision is reached on the full complement of experiments for each spacecraft, interface requirements and design accommodations will be developed. The ATS-F&G system has limited growth capability and is limited in its capacity to carry a large variety of heavy experiments if the Atlas-Centaur TE-364-3 is used. Employment of the Titan booster offers considerably increased experiment opportunities; however, a practical and meaningful mission can be accomplished with the Atlas-Centaur booster. This being the more critical system, it will be described in more detail throughout the report.

LAUNCH AND TRAJECTORY

The ATS-F&G can be launched from ETR (Cape Kennedy) by an Atlas-Centaur TE-364-3, at a launch azimuth of 90 degrees. Figure V-3 shows the launch vehicle sequence and maneuvers through Centaur second burn. Figure V-3 also illustrates the Centaur maneuver after second burn, Centaur separation and apogee motor (TE-364-3) burn. Figure V-4 is a schematic of the transfer orbit; the numbers provide the sequence.

The Atlas burnout and Centaur first burn place the spacecraft, still attached to the Centaur, in a circular, nominal 90-nautical mile parking orbit, inclined 28.5 degrees to the equatorial plane. The Centaur is then oriented so that the second burn will result in an inclination reduction of 8.45 degrees. At the first equator crossing, the second Centuar burn is initiated, resulting in a plane change (8.45 degrees) and injection into transfer orbit (apogee 36,000 km, inclination 20.05 degrees). Figure V-5 illustrates the transfer orbit and deployment sequence.

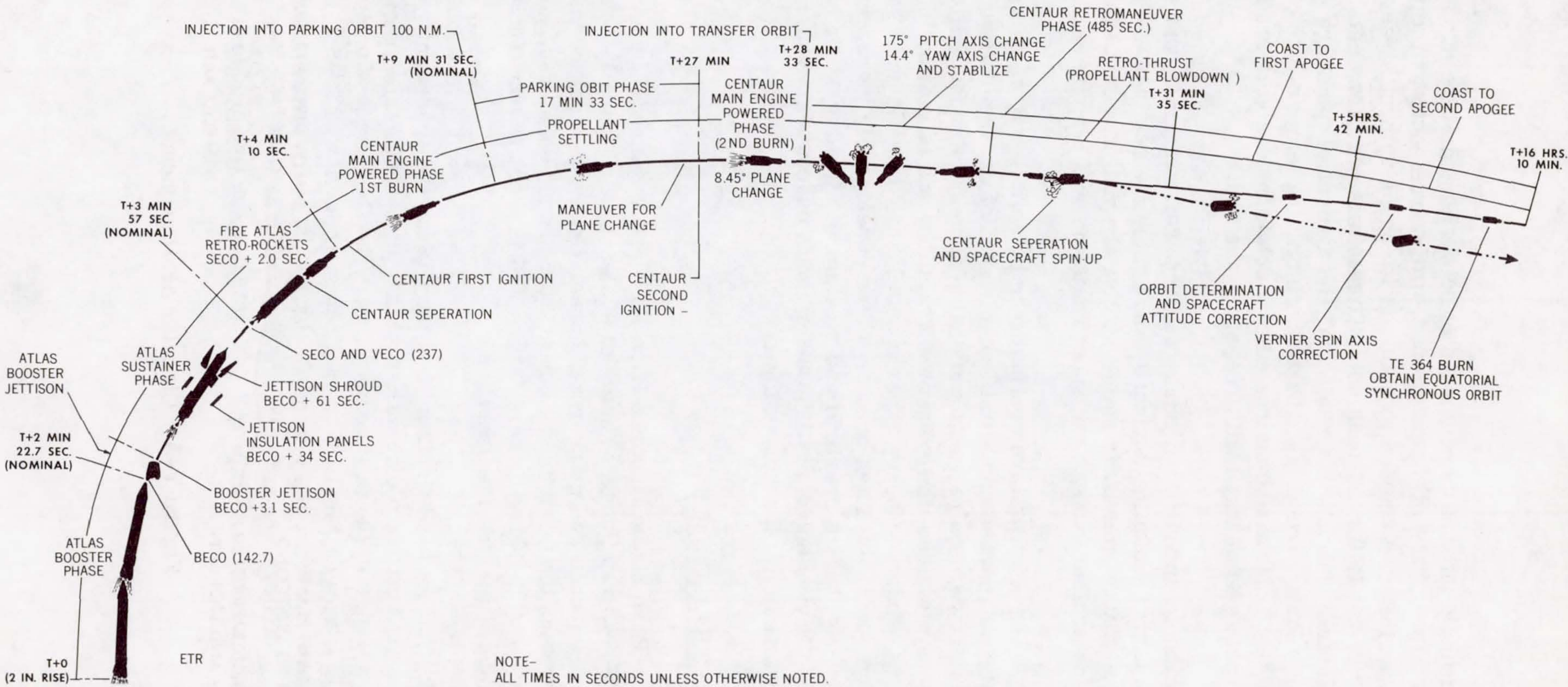

Figure V-3. ATS-F&G Atlas-Centaur launch.

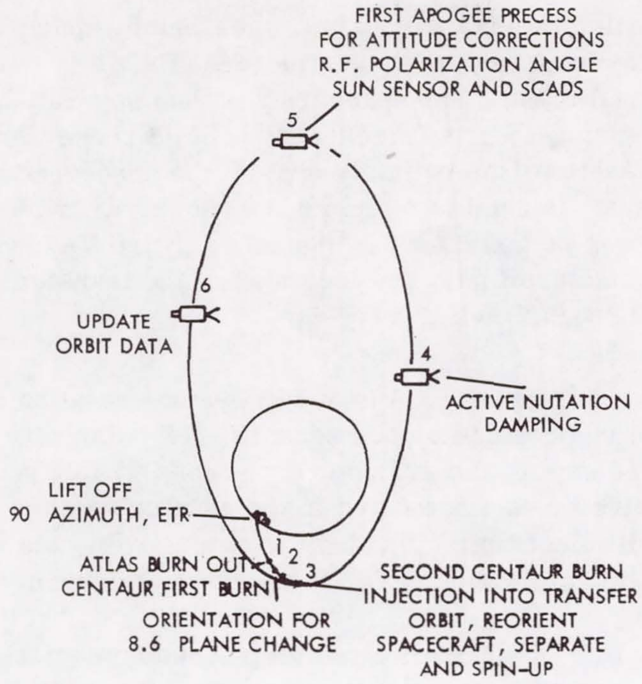

Figure V-4. Transfer orbit sequence.

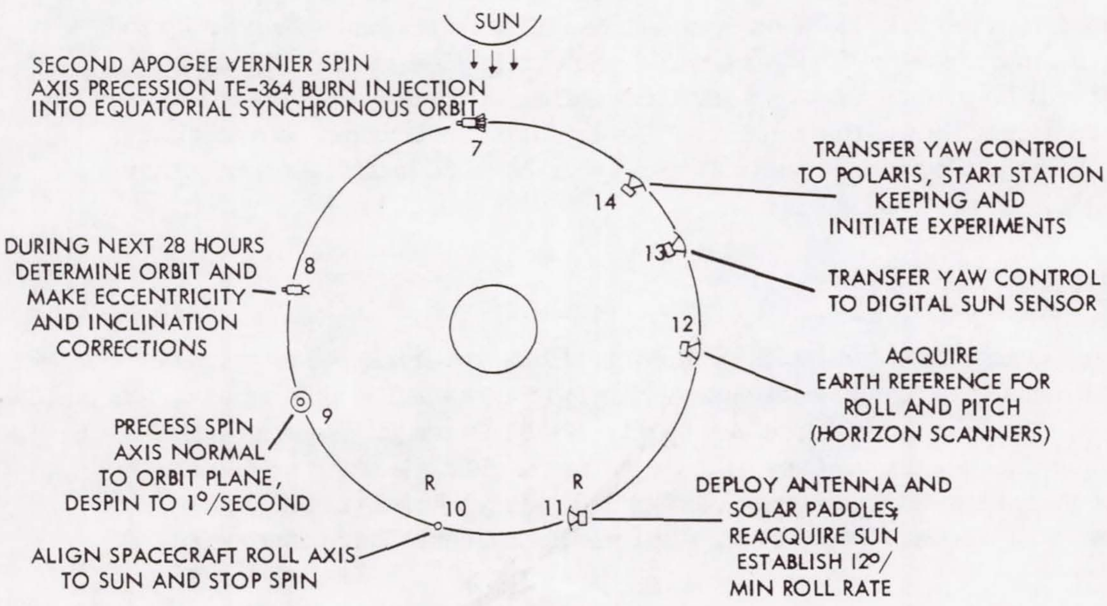

Figure V-5. Circular orbit sequence.

Shortly after injection into transfer orbit, the Centaur spacecraft is oriented to provide the attitude for the apogee motor (TE-364-3) burn, which will result in an equatorial orbit. The spacecraft is then separated from the Centaur by means of a spring-ejecting mechanism. Solid rocket motors, attached to the adapter section, immediately spin up the spacecraft to 60 rpm. The spin stabilization mode is used to maintain the spacecraft attitude during the coast-to-second apogee period (about 16 hours). Figure V-6 is a mercator chart showing the projected orbit path and locations of the transfer orbit sequences, noted as 1 through 7 in Figure V-5.

During the coast phase, accelerometers will provide nutation sensing. Attitude information will be provided by sun sensors, RF polarization, and SCADS. Active nutation damping and attitude corrections will be provided by two redundant hydrazine thrusters located in the adapter section. This coast period provides sufficient time (16 hours) to obtain orbit data and make necessary vernier attitude corrections prior to apogee motor burn.

At second apogee, the TE-364-3 is fired (40 seconds duration) to provide the necessary plane change and to circularize the orbit. Injection into synchronous orbit occurs over the western Atlantic (53 degrees W./Long.)

After injection into synchronous orbit, the spacecraft will be tracked for 16 hours to develop a sufficient body of information for accurate orbit determination. During this period, nutation damping and attitude control will be effected by means of the hydrazine thrusters. Upon determination of orbit the spinning spacecraft will be precessed to achieve the desired attitude; then both thrusters are fired continuously for the required time to adjust inclination, eccentricity, or both. These orbit corrections will take from 28 to 76 hours, depending upon the number of reiterations required.

ACQUISITION

After eccentricity and inclination corrections are made, the hydrazine thrusters are used to precess the spacecraft spin axis until it is normal to the orbit plane. The adapter section and spent TE-364-3 are jettisoned. Spacecraft despin is accomplished by means of a yo-yo mechanism, leaving a residual spin about the yaw axis of 1 degree per second. Figure V-5, positions 9 through 14, will assist in visualizing the remaining acquisition maneuvers.

The 1 degree per second spin rate causes the sun sensor, mounted on the roll axis, to scan the sun. The spacecraft spin is stopped when the roll axis

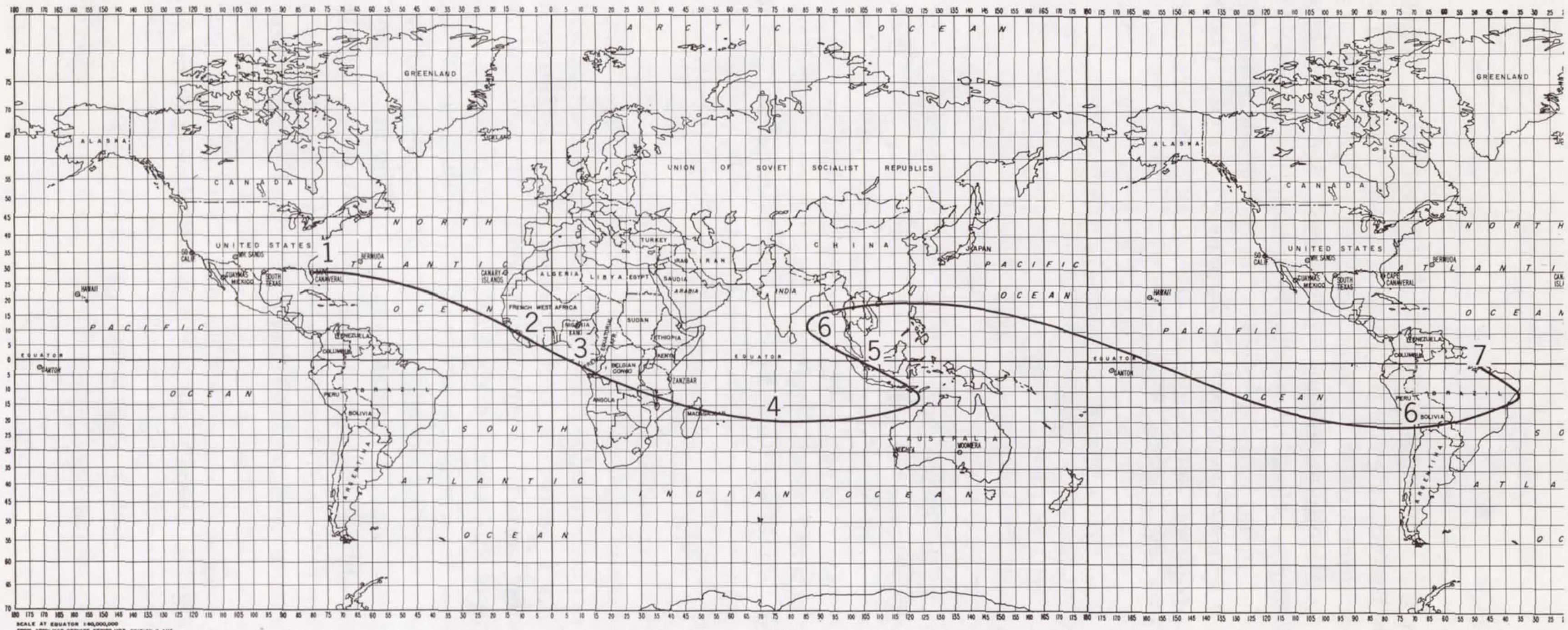

Figure V-6. Projected orbit path and locations of transfer orbit sequences.

acquires the sun. The resistojet thrusters are used to stop the spin, and for all subsequent maneuvers.

After sun acquisition, the antenna is deployed, and a roll rate of 12 degrees per minute is established. Rotation about the roll axis will result in the earth sensors scanning the earth. The roll maneuver is stopped when the earth is acquired. The earth sensors now control the roll and pitch. The sun sensor still controls the yaw. Yaw control is now transferred to the digital sun sensor to provide an increased accuracy over the required Polaris acceptance angle. A yaw bias is established for the acquisition of Polaris. Yaw control is transferred to Polaris. The three momentum wheels are then turned on and the spacecraft is in the operational mode.

OPERATIONAL MODE

During operation the spacecraft will be in a synchronous equatorial orbit and the 30-foot parabolic antenna will be earth-oriented. At synchronous orbit, the earth subtends an angle of 17 degrees. The spacecraft will be capable of directing the antenna beam to any point on the earth disc by using spacecraft attitude control for antenna beam pointing. In addition, feed displacement, by means of electronic switching in one axis and mechanical scan in the other, will permit the S-band transmitter and receiver beams to be pointed ± 9 degrees from the spacecraft axis. Also monopulse capability will be provided at X-band and VHF.

Desired spacecraft attitude changes during spacecraft operation are effected by means of a reaction-jet inertia wheel hybrid. The orthogonal inertia wheels provide the high accuracy attitude control capability necessary to the primary mode of spacecraft operation. The inertia wheels also provide a means for storing angular momentum to offset cyclical perturbations, thereby economizing on power and fuel consumption. The reaction jets are resistojet thrusters which use electrically heated ammonia as the propellant. These jets can provide an acceptable degree of attitude control for certain modes of spacecraft operation. The gravity-gradient boom experiment will make available an alternate to the resistojets for torqueing the spacecraft about the roll and pitch axis.

The attitude sensors used in the normal operational mode are the IR earth sensors and the Polaris star sensor. The earth sensors determine roll and pitch. The Polaris sensor determines yaw.

In addition to the sensors used in the normal mode, the complement of experiments, acquisition sensors, and communication links provide a number of additional attitude references which may be brought into use. Some of them are described in the following paragraphs.

The sun sensors which were used during the acquisition maneuvers are available for repeating those maneuvers or for providing sun orientation. The sun sensor and the Polaris sensor can be used for earth orientation by continuously updating the sun sensor angular bias. The spacecraft can also be controlled, but in a more limited sense, by means of sun sensor and earth scanners. During this mode of control, attitude about the yaw axis will be degraded as the noon position is approached. During that interval of time, and while passing through the earth umbra, the body rate gyros, which were used during the acquisition maneuver, can provide the stabilizing signals.

As an extreme, the sun alone can be used to provide a three-axis control reference. The sun can provide pitch reference except while the spacecraft is passing through the earth umbra (1 hour, 12 minutes). Yaw and roll axis reference alternately degrade 90 degrees apart. The resulting three-axis control system would make use of the body-rate gyros to hold the attitude of the axis which is not under sensor control at any particular moment. Errors equivalent to 6 hours of rate gyro drift can be expected (1 degree).

The SCADS experiment is available as an attitude sensor both during spin stabilization and during the operational mode. This type of sensor will be qualified on ATS-C for use on a spinning spacecraft. The flight on ATS-F&G will evaluate a modified configuration capable of being used on spinning or stabilized spacecraft. The system scans a section of the celestial sphere and transmits the star pattern above a preset intensity threshold. This star pattern is then relayed to the ground station where it is used to determine pointing orientation. Indexing may be performed by recognizing star intervals or by comparative intensities. The SCADS can provide three-axis, open loop attitude determination.

The RF interferometer experiment provides an additional means of attitude determination. This experiment functions in X-band (8 GHz) and measures phase relationships between two coplanar antennas. The use of two ground stations with this experiment will make it capable of determining three-axis spacecraft orientation to better than 0.1 degree. This sensor may be used for open loop or closed loop attitude control.

Antenna pointing direction will be controllable to 0.1 degree by utilizing the parabolic antenna monopulse system. In this closed loop mode, pointing direction is controlled by the communicating ground station.

Spacecraft attitude may be controlled by any of the foregoing systems or by combinations of them. The repeatability of the acquisition maneuver and the availability of the body rate gyro system will permit an unhurried attitude determination and the selection of the most appropriate operational mode.

COMMAND, TELEMETRY, AND GROUND SUPPORT

During the lifetime of the spacecraft, command and telemetry requirements will vary over a wide range for the different situations of transfer orbit, apogee motor firing, antenna deployment, station-keeping, and normal operation. The command and telemetry system has been designed to provide an optimum flexibility for meeting these diverse mission requirements while maintaining minimal system complexity, size, weight, and power.

COMMAND

The present ATS-F&G command requirements are given below. A detailed discussion of the command system is contained in Section VII H.

1. ≈ 160 latching-type relay commands
2. ≈ 171 ten-bit digital-command-type words
3. ≈ 8 timed tone-execute commands for real time control of spacecraft functions

To provide this command capability, the following system is proposed. Expansion capability is available.

1. 256 commands in an X-Y matrix to provide current drivers for latching-type relays (16 by 16 X-Y Matrix)
2. 10-bit command word (4-bit address, 6-bit command) -- total of 12 address and 64 commands/address (2 bits of the address will specify relay driver or command word)
3. "Tone-execute" command capability

4. Command rate of 2 commands per second

5. Internal-spacecraft command verification and execute indication in telemetry bit stream

6. No stored commands

7. No frequency generation

A basic block diagram of a system which will afford this capability is shown in Figure V-7.

TELEMETRY

The telemetry requirements of ATS-F&G are such that only certain parameters require continuous read out, with the intermittent requirements varying from time periods of a few minutes to several hours. To best satisfy these conditions, a spacecraft data processing system is proposed which can vary the sampling rates and sampling points through the employment of various stored programs. The system for ATS-F&G is based upon the adaptable or programmable concept, which allows wide variations in sampling rates, points, and formats, thereby removing many of the restrictions imposed by a non-selective system. The data processing system capabilities are as follows:

1. Adaptable (programmable) concept to be employed, which will allow wide variations in sampling rates, points, and formats

2. Two basic transmission rates:

   40 samples per second (prior to satellite on station)
   400 samples per second (after satellite on station)

3. Sample words to be 10 bits in length

4. 8-bit analog-to-digital conversion capability

5. Capability of accepting analog, 10-bit serial digital information or 10-single bit (i.e., on-off)digital information

6. Distributed commutation to minimize hardness

7. Optional ground programming mode

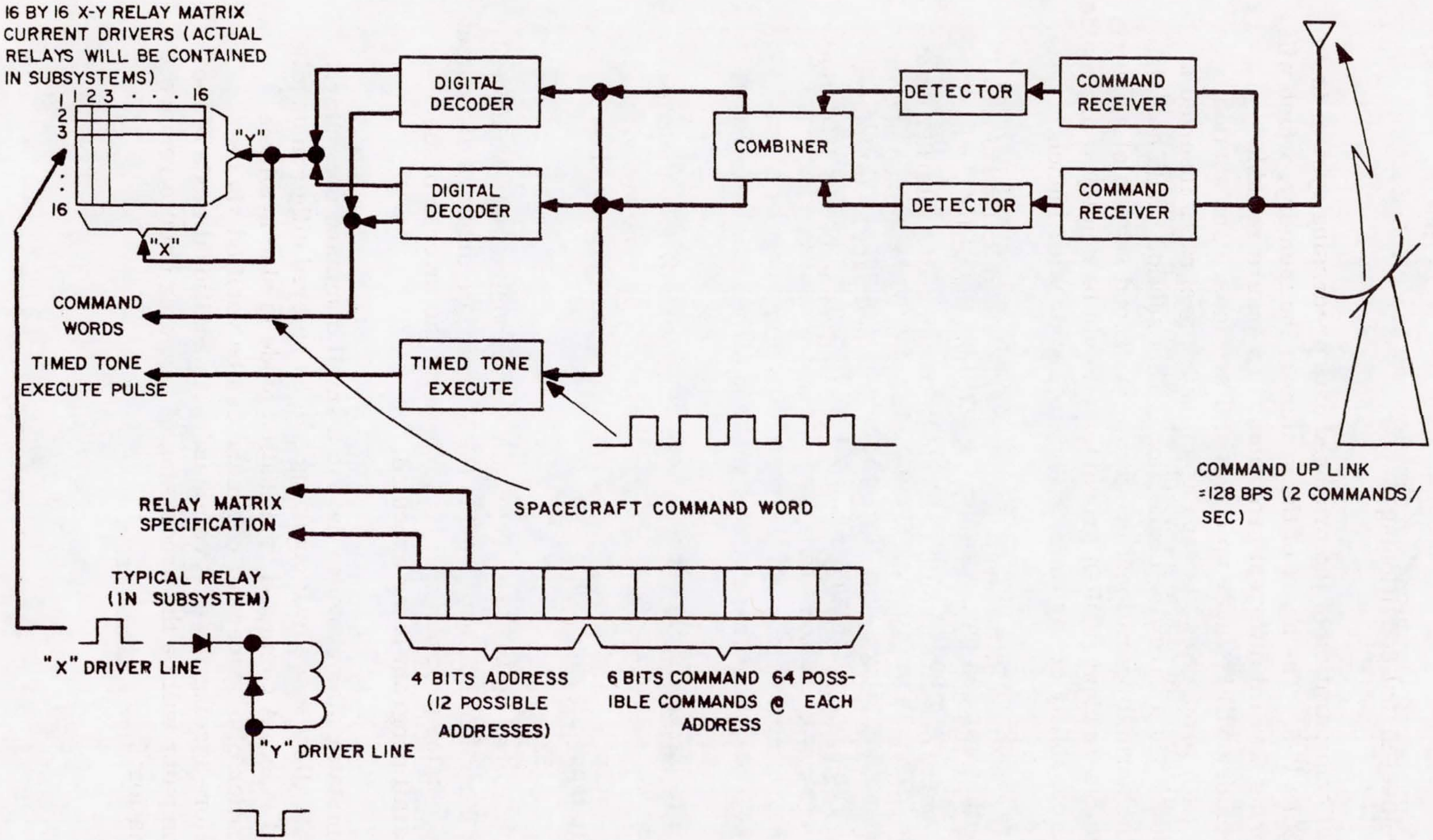

Figure V-7. Command System.

8. Optional auxiliary telemetry points for failure analysis

9. No low-level signal conditioning.

The internal configuration of the proposed data processing system is illustrated in Figure V-8. The heart of the system is the memory, which will be capable of storing several different programs. To ensure reliable operation, the memory will be made up of several sections. One section, having read-only (RO) properties, is capable of having programs inserted at any time up to launch via a hardwire connection. This section would contain, as a minimum, the normal operational program. A second section of memory, having nondestructive-readout (NDRO) properties, would be employed to provide a reprogramming capability at any time, either before or after launch.

In the actual operation of the system, programs would be loaded into the memory via either the command link or a hardwire connection. The specific program being executed would be cycled through under internal logic control. The bit patterns retrieved from memory would be examined and utilized to open a gate from one of the approximately 1000 data or telemetry points the system can handle. Present ATS-F&G requirements are discussed in greater detail in Section VII H.

GROUND STATIONS

The primary ground stations for ATS-F&G support will take the configuration shown in Figure V-9. Telemetry signals will be received via a standard receiver system, and synchronization will be accomplished on either the 400- or 4000-bps bit stream.

The signal will then be fed to a programmable decommutator which will decommutate the incoming data and provide the appropriate displays and output data. The decommutator program will be essentially the inverse of the particular spacecraft program being executed.

The decommutator also provides data to a small command computer. This computer will allow spacecraft commands to be entered either manually or via a command tape. A command verification mode is also available whereby the execution of a spacecraft command can be verified via the telemetry link before issuing the next command. An additional mode of the decommutator computer will be the processing of incoming data to remove redundant information.

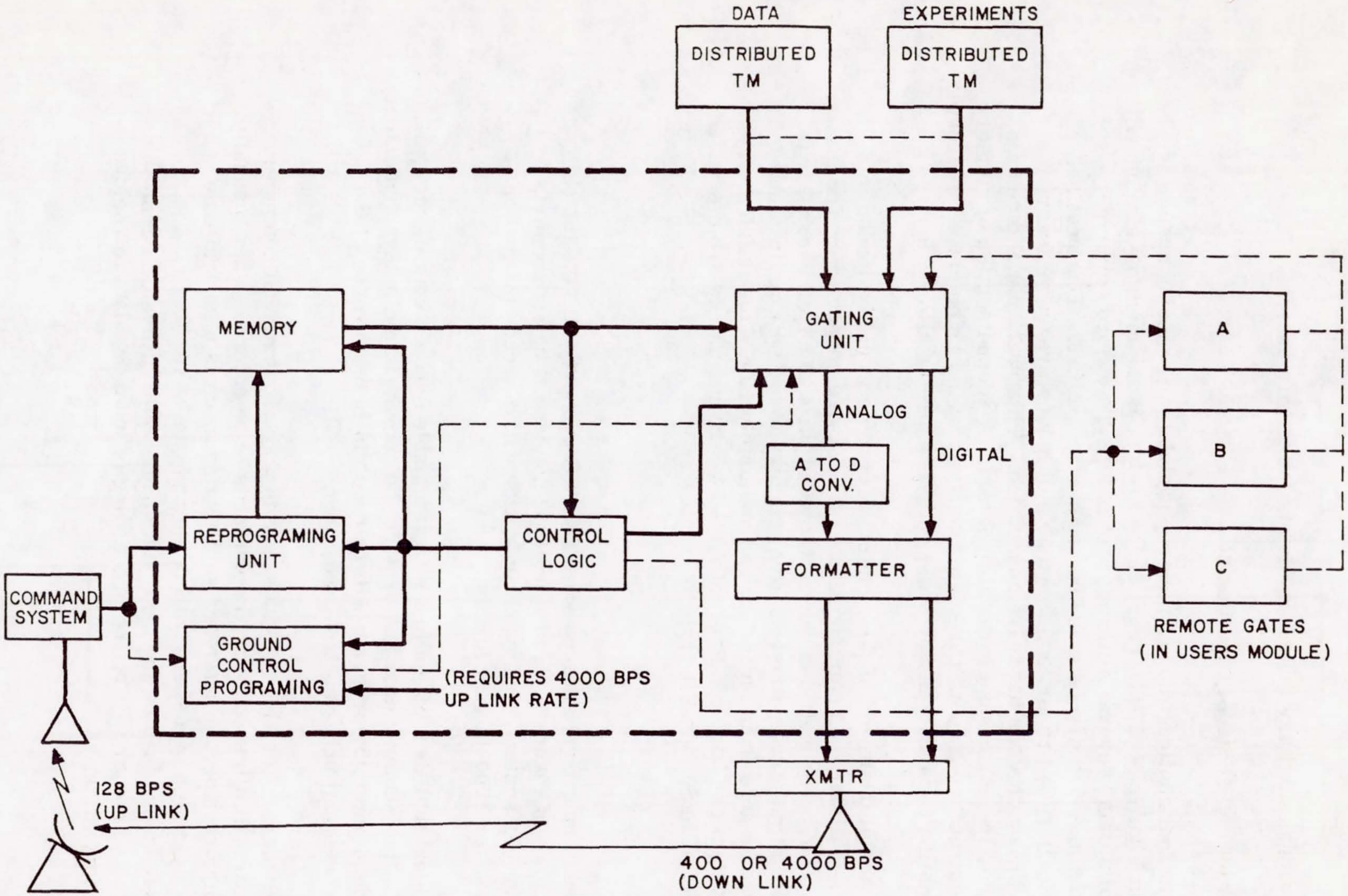

Figure V-8. Telemetry system.

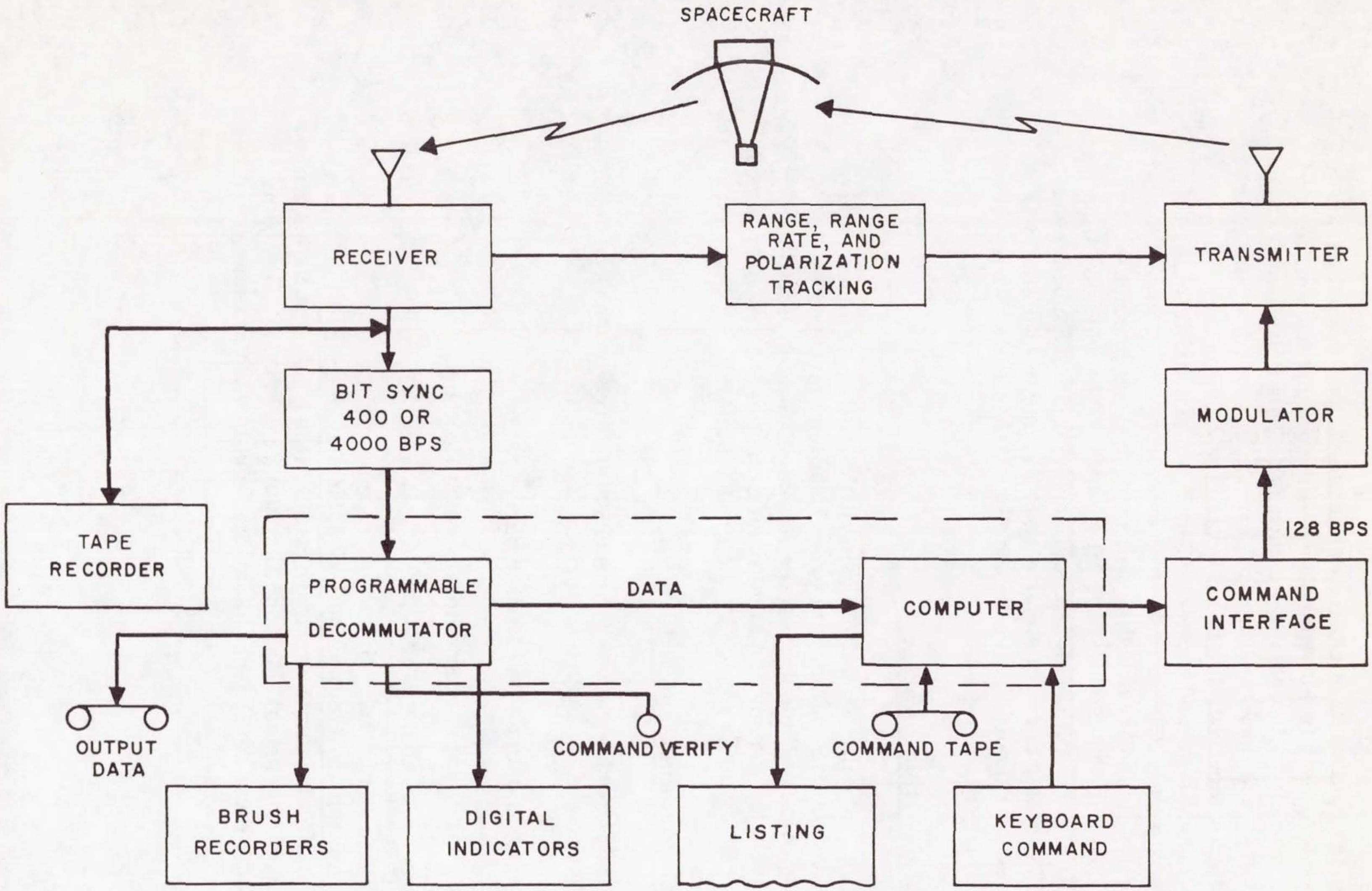

Figure V-9. Ground station.

The range and range rate measurement system, and the polarization angle measurement system is incorporated into the ground transmitter and receiver system used for telemetry and command. This unification of tracking, telemetry, and command into one system reduces the number of separate subsystems required for ground support and maximizes the probability of successful ground support. The command and tracking uplink will occupy approximately 250 Hz in the frequency range of 1760 to 1850 MHz, and the tracking and telemetry downlink will occupy at the maximum approximately 7 MHz in the 2200 to 2300 MHz range. The wide downlink bandwidth results from the RARR tones being modulated onto subcarriers at 1.4 or 3.2 MHz, which permits simultaneous ranging by two stations. Section VII J, Ground Support, gives the details of tracking, telemetry, and command ground operation.

Ground Station Orbit Coverage

The tracking and data-acquisition capabilities of Cape Kennedy and of down-range stations in conjunction with ground stations managed by GSFC are required to provide sufficient coverage for the ATS-F&G satellites. Figure V-10 is included to show the significant events of the proposed ascent trajectory and to illustrate coverage by the ground stations.

Data Acquisition and Command

Data acquisition and command is discussed in three phases, covering three distinct sections of the launch trajectory and orbit.

1. Launch-to-injection into circular parking orbit

2. Second burn through reorientation of the Centaur

3. After reorientation

Phase 1. Launch-to-Injection into Circular Parking Orbit - This phase includes liftoff and point 1 in Figure V-10. Point 1 marks the location of Atlas burnout, shroud separation, and Centaur first ignition. Tracking data will be available via the C-band Radar Systems located at Cape Kennedy and downrange.

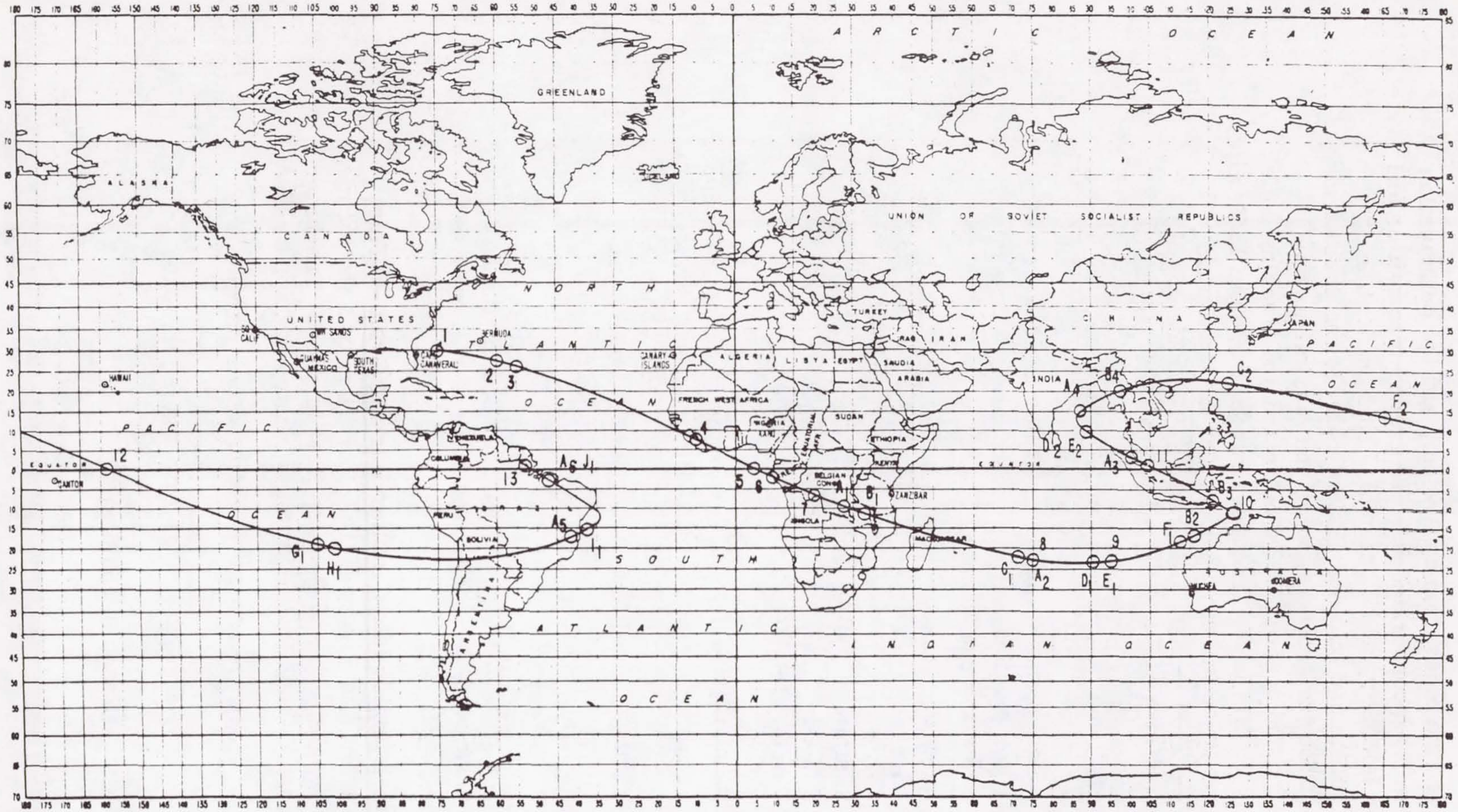

Figure V-10. Proposed ATS-F&G ascent trajectory.

Phase 2. Second Burn Through Reorientation of the Centaur - This phase covers point 2 (termination of first Centaur burn, injection into circular parking orbit, initiation of coast phase), points 3 and 4 (coast phase), point 5 (initiation of Centaur second burn and plane change), and point 6 (completion of Centaur second burn, transfer orbit established and reorientation of vehicle through 175 degrees of the pitch axis).

Phase 3. After Reorientation - The S-band frequencies of 1750 to 1850 MHz for ground-to-spacecraft and of 2200 to 2300 MHz for spacecraft-to-ground transmission will be used for this program in conjunction with the transponder developed for unified operation with Goddard Range and Range Rate System (GR&RR) for command reception, telemetry transmission, and tracking data (GR&RR technique) retransmission. A new transponder antenna will be designed providing linear polarization and a wider beamwidth than is available from the present transponder antenna design.

Figure V-10 shows that the GR&RR equipped stations at Madagascar, Carnarvon, Santiago, and Rosman, have more total orbital coverage than the present ATS-equipped stations. (See Section VII J for details). Quito would be available for telemetry reception dependent on spacecraft data-acquisition schedules and priorities by 1970.

It is also anticipated that the remaining dish sites would have S-band capability during this time frame. A lightweight Minitrack transmitter will be added as a backup system to provide Minitrack data from Johannesburg, Madagascar, Orroral, Santiago, and Lima during the transfer orbit.

Orbit Determination

The Cowell Orbit Determination System will be used for the ATS-F&G program. During the transfer orbit the following stations will provide the orbit tracking data as indicated:

Rosman	(Range and range rate and polarization angle)
Santiago	(Range and range rate and polarization angle)
Carnarvon	(Range and range rate and polarization angle)
Madagascar	(Range and range rate and polarization angle)
Minitrack network	(Direction cosines)

STADAN 40 foot and 85 foot paraboloids	(X, Y angle data)

During the synchronous orbits, primary tracking will be by range and range rate from Rosman and Santiago. Approximately 3 hours after liftoff, sufficient tracking data will have accumulated to determine the orbit. The output of the orbit determination program will be position and velocity of the satellite as functions of time. Approximately 3 hours after each maneuver, sufficient tracking data will have been accumulated to redetermine the orbit.

SECTION VI

PREFERRED SYSTEM DESCRIPTION AND TRADE-OFFS

A. LAUNCH VEHICLE SELECTION

For the ATS-F&G mission studies, the three contractors were required to analyze the applicability of three launch vehicles to the ATS-F&G mission. The three vehicles under consideration were SLV-3A Atlas-Agena D, SLV-3C Atlas-Centaur, and Titan IIIC. These vehicles were studied and compared on the basis of usable payload injected into synchronous orbit, cost, and vehicle limitations (e.g., nose shrouds, coast limits, etc.). For synchronous missions the Atlas-Agena and Atlas-Centaur require a kickstage to inject the spacecraft into a final synchronous orbit while the Titan IIIC does not.

Table VI-1 presents a comparison of payload capabilities and cost for all three vehicles. In Table VI-1 it was assumed for purposes of comparing maximum payload capability that optimized kick motors be used with the Atlas-Agena and Atlas-Centaur vehicles. This table indicates that the payload delivered by the Atlas-Centaur and Titan IIIC is comparable but that the payload delivered by the Atlas-Agena is much less (~50 percent) than that delivered by the other two vehicles. Studies performed by General Electric, Lockheed, and Fairchild-Hiller indicate that an Atlas-Agena launched spacecraft, designed to meet all the ATS-F&G objectives, would be a marginal system with no growth capability and would be crude compared to the tvpe spacecraft injected by the Atlas-Centaur or Titan IIIC. The three contractors, therefore, recommended that the Atlas-Agena vehicle not be considered as a launch vehicle for the ATS-F&G mission because of its payload and shroud limitations.

Table VI-1 also shows a comparison between individual cost and cost per pound of payload into synchronous orbit for all three vehicles. It indicates the cost per pound of payload for the Titan IIIC is 21 percent more than the Atlas-Centaur. Thus, on the basis of cost per pound of useful payload, the Atlas Centaur is a more desirable vehicle. However, it must be realized that some uncertainties exist in the cost figures. Spacecraft weights have a tendency to grow during the course of a program. Therefore, with a small weight margin, experiments and weight design must be rigidly controlled. Cost savings and reliability advantages might be realized by using the Titan with its greater weight lifting capability.

Table VI-1

Comparison of Payload Capabilities and Vehicle Costs

Vehicle	Atlas-Agena	Atlas-Centaur	Titan IIIC
Payload Capability			
1. In synchronous orbit	1080*	1940*	2100
2. In transfer orbit	2270	4000	----
Cost (millions of dollars) **	7.9	13.4	17.6
Cost/pound of payload (thousands of dollars)	7.31	6.91	8.38

* Based on use of an optimized kickstage

** These cost estimates are those given by NASA Headquarters, Code SV, and are considered preliminary. Further refinement on cost estimates is required.

Since one of the prime purposes of the ATS-F&G mission is to place into synchronous orbit a 30-foot diameter parabolic antenna, packaging of the spacecraft is a critical problem. The results from all the study contracts and the studies performed in house indicate that a shroud extension or new shroud is required for all three launch vehicles in order to launch the ATS-F&G. Studies indicate that the development of a new shroud would be more costly than modification of existing shrouds. Detailed descriptions of the fairing modification for the various vehicles are to be given in the following paragraphs.

General Electric recommends using the OAO shroud with the Titan IIIC vehicle. This shroud allows a spacecraft diameter of up to approximately 110 inches. General Electric also indicates that the Titan IIIC can be flown with up to a 25-foot shroud cylindrical length without a significant reduction in launch availability and payload capacity since the nominal vehicle performance must be reduced to meet range safety requirements (i. e., sub-range Africa). This vehicle has a lower payload acceleration and vibration level than the Centaur during launch. Thus, a payload designed for the Centaur can be flown on the Titan with a minimum of modification.

Based on information supplied by the Centaur Project Office, the preferred shroud for use with the Atlas-Centaur vehicle is the Surveyor type shroud or some extended version of this shroud. The Surveyor type shroud is composed of a 15-foot, 30-degree conical section attached to a 5-foot cylindrical section. Modification of this shroud is achieved by extending the cylindrical section. This extension weighs 76 pounds per foot of extension. A payload (into transfer orbit) weight loss of 0.1 pound per pound of shroud extension results from the modification. The maximum feasible shroud extension is 15 feet, which results in a 114-pound penalty in the Hohmann transfer orbit and a maximum payload penalty of 35 pounds in the final synchronous orbit.

There are no serious launch constraints for the Atlas-Agena vehicle.

The launch constraint on the Centaur is that the maximum time between burns must be 20 to 25 minutes because of limited battery power capability and boil-off of the cryogenically stored fuel and oxidizer. To launch a spacecraft from a transfer orbit to the synchronous orbit using the Atlas-Centaur vehicle requires that the Centaur be ignited twice. The first ignition injects the Centaur/spacecraft into a low altitude circular parking orbit, and the second ignition initiates the Hohmann transfer orbit to the synchronous altitude. This constraint makes it necessary to inject the spacecraft into the transfer orbit at the first equator crossing. However, if the Centaur coast period is extended so that the second burn can occur at the ascending node, the time in transfer orbit will be drastically reduced. Lewis is presently conducting a study with the aim of removing this constraint.

The launch constraint on the Titan IIIC vehicle is the limited life of the transtage (2nd stage). The transtage guidance and control functions degrade to an unacceptable level 6 1/2 hours after launch. This constraint makes it necessary to inject the spacecraft into synchronous orbit at the first apogee of the Hohmann transfer ellipse. Note, however, that in the low altitude parking orbit the transtage can be allowed to coast to the second equator crossing before initiating the transfer orbit.

In conclusion, all three study contractors agreed that a payload launch by SLV-3C Atlas-Centaur/TE-364-3 is feasible. This launch vehicle combination is the most cost effective under the assumptions made. It should be noted, however, that if the ATS-F&G spacecraft weight requirement were to outgrow this launch vehicle combination, additional capability can be provided by upgrading the TE-364-3 or by using the Titan IIIC. Both are feasible with the present spacecraft configuration.

B. APOGEE MOTOR DESCRIPTION AND PERFORMANCE TRADE-OFF

The propulsion system recommended for apogee kick by all three study contractors (General Electric, Lockheed, and Fairchild-Hiller) and the GSFC study group was a version of the Thiokol Chemical Corporation TE-M-364 solid propellant motor. This is the motor which was successfully used on the Surveyor I launch and the two Boeing Burner II launches. The TE-M-364 motor is currently being qualified for use as a third stage propulsion system on the Improved Delta launch vehicle. The specific motor version recommended by all three study contractors and the GSFC study group was the TE-M-364-3 which is being qualified for use in the Delta program. It is committed to be flown in the Radio Astronomy Explorer program by the third quarter of 1967. Figure VI-1 is a drawing of the TE-M-364-3 motor. Its characteristics are outlined in Table VI-2.

Table VI-2

TE-M-364-3 Motor Characteristics

Characteristic	Value
Specific impulse	classified
Average thrust	9890 lb_f
Burn time	41.3 sec
Total impulse	417,600 lb-sec
Propellant weight	1440 lb
Loaded weight	1579 lb
Motor length	52.3 in
Motor diameter	37.5 in
Nozzle length	17 in
Average chamber pressure	600 psia
Chamber material	steel

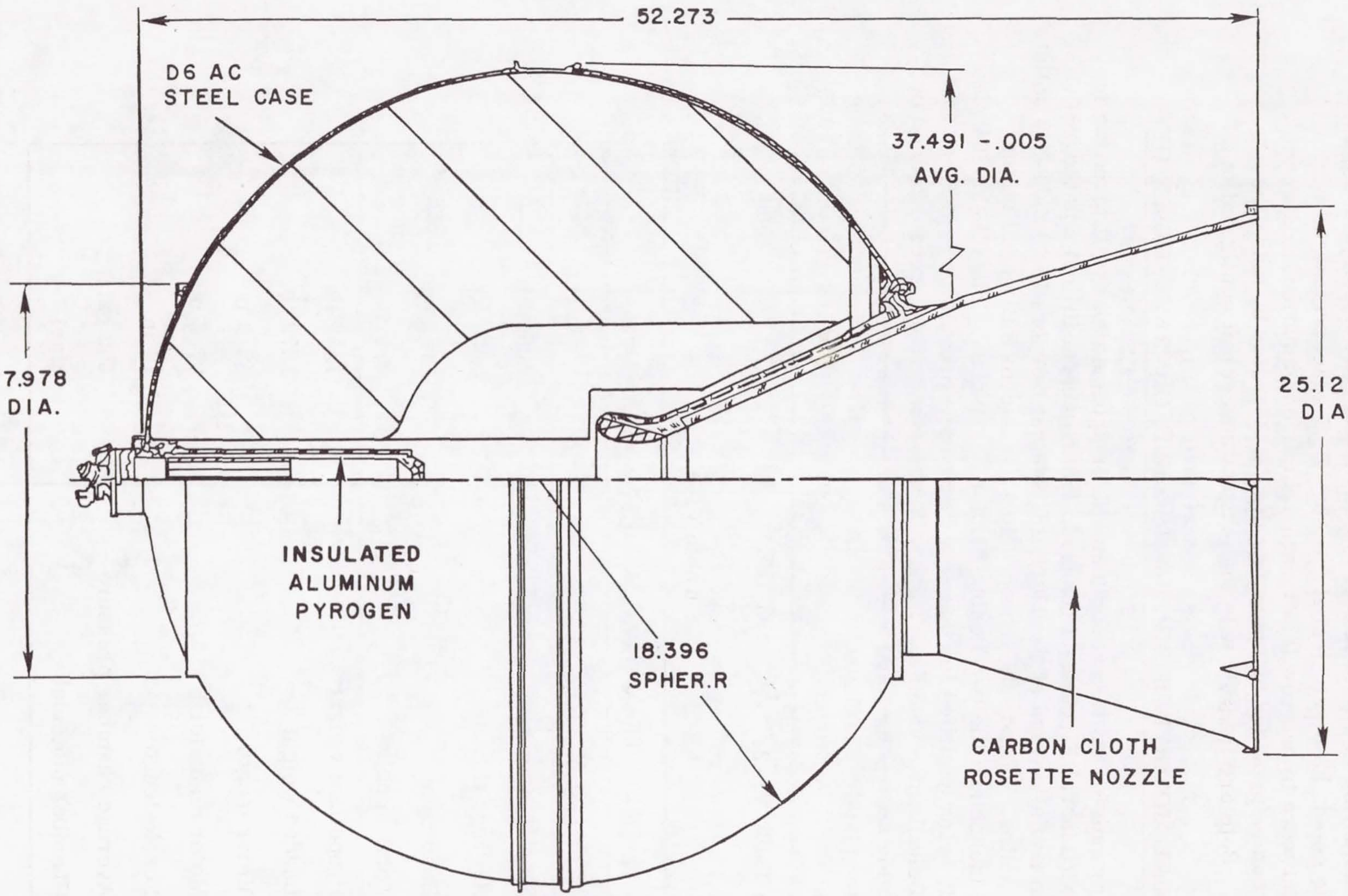

Figure VI-1. DSV-3E-15 Delta 3rd stage rocket motor (TE-M-364-3).

The payload optimization for ATS-F&G using the Atlas-Centaur vehicle and the TE-M-364-3 solid propellant motor is shown in Figure VI-2. Figure VI-2 was generated on the basis of the following assumptions: (1) a 15-foot shroud extension, (2) 9 pounds of expendables used in the transfer orbit (based on the GSFC study), and (3) a 16-pound payload destruct system which is carried for all spacecraft having an apogee motor. Figure VI-2 indicates that the optimum perigee plane change performed by the Centaur vehicle is 8.45 degrees and the maximum weight injected into the final near synchronous orbit is 1805 pounds. This weight includes the 139-pound empty TE-M-364-3 motor case. Table VI-3 presents a total weight breakdown for the ATS-F&G mission based on the results of Figure VI-2.

Table VI-3

ATS-F&G Weight Breakdown

Item	Weight (lb)
Atlas-Centaur base line capability	4000
15-foot shroud extension penalty	114
8.45 perigee plane change	616
Payload destruct system (attached to Centaur)	16
Weight into transfer orbit	3254
Expendable usage from spinup and nutation damping	9
Weight at apogee	3245
TE-M-364-3 propellant weight	1440
Weight injected into synchronous orbit	1805

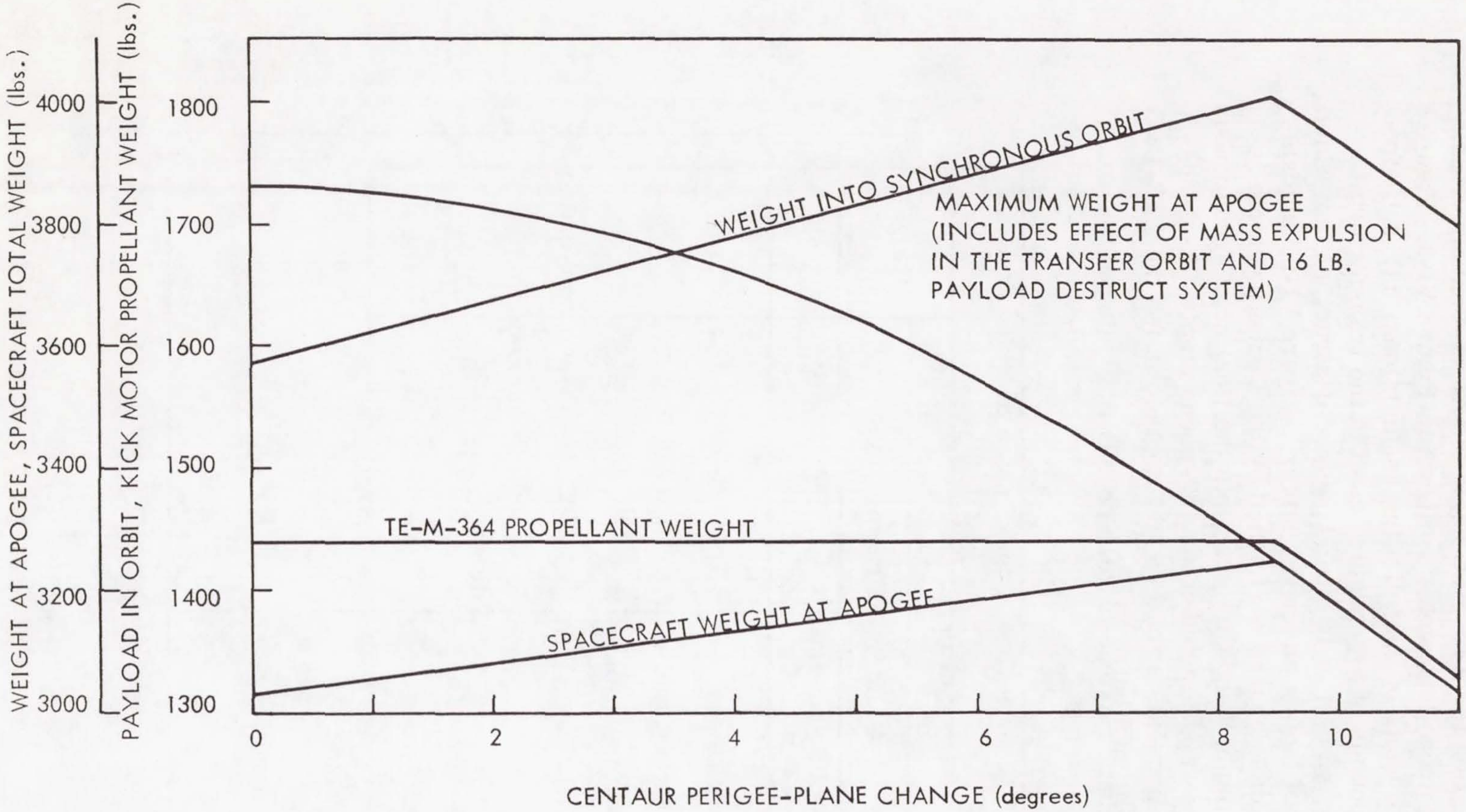

Figure VI-2. Atlas-Centaur/TE-M-364 payload optimization.

The TE-M-364-4 motor, an optimized version of the TE-M-364-3 motor, was examined to determine its applicability for the ATS-F&G mission. The same assumptions were used as those used for the optimization of the capability of the TE-M-364-3/Atlas-Centaur combination. This motor consists of the nominal TE-M-364 motor with a 10.8-inch cylindrical section inserted between the hemispheres of the motor case. The motor case is fabricated from titanium instead of steel. This case weighs 144 pounds and allows a propellant loading of 2100 pounds. Figure VI-3 is a drawing of the TE-M-364-4 motor. Its characteristics are outlined in Table VI-4. It was found that in order to fall within the Centaur capability that the motor would have to be off-loaded by 249 pounds of propellant and that a payload of 2010 pounds (including motor case) would be injected into the synchronous orbit. A weight brekdown is shown in Table VI-5. It was learned, however, that the TE-M-364-4 motor has not been developed to date and is not a flight proven piece of hardware.

Table VI-4

TE-M-364-4 Motor Characteristics

Characteristic	Value
Specific impulse	classified
Average thrust	11,700 lb
Burn time	49.8 sec
Total impulse	602,700 lb-sec (maximum)
Propellant weight	2100 lb (maximum)
Loaded weight	2244 lb
Motor length	68.1 in
Motor diameter	36.9 in
Nozzle length	17 in
Average chamber pressure	600 psi
Chamber material	titanium

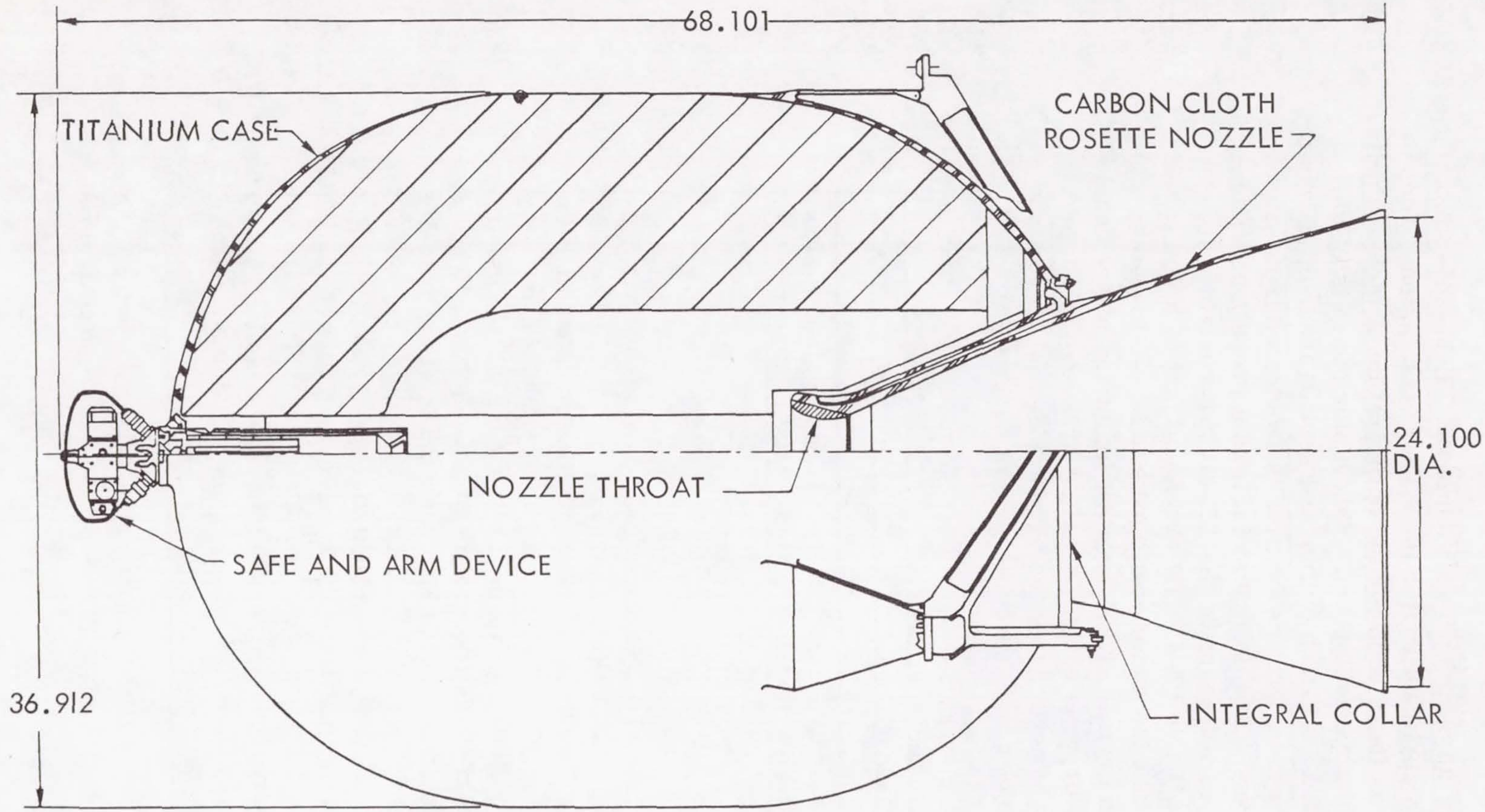

Figure VI-3. TE-M-364-4 rocket motor assembly.

Table VI-5

Weight Breakdown Using TE-M-364-4 Motor	
Item	Weight (lb)
Atlas-Centaur baseline capability	4000
15-foot shroud extension penalty	114
Payload destruct system	16
Expendable usage from spinup and nutation damping	9
Weight at apogee	3861
TE-M-364-4 propellant weight	1851
Weight injected into synchronous orbit	2010

In summary it is concluded in all of the ATS-F&G mission studies that the TE-M-364-3 can be used in conjunction with a Centaur perigee plane change in order to achieve minimum ATS-F&G objectives.

C. LAUNCH AND ASCENT SEQUENCE

The launch profile for the ATS-F&G mission is similar to that of all the existing synchronous operational satellites (e.g. Syncom, Early Bird, and ATS-B); i.e., the spacecraft/launch vehicle combination injects into a low (90 to 100 n.m.) circular inclined orbit and then at some equator crossing the spacecraft is injected into a Hohmann transfer ellipse having a perigee altitude equal to that of the low circular orbit and an apogee altitude equal to the synchronous altitude (19,323 n.m.). At some apogee passage of the transfer orbit a propulsion system attached to the spacecraft is ignited, injecting the spacecraft into a near circular equatorial orbit; i.e., the thruster simultaneously removes the eccentricity and inclination of the transfer orbit leaving slight residuals resulting from non-perfect systems performance These residuals are then removed by a vernier propulsion system carried on-board the spacecraft. Figure VI-4 illustrates the total launch profile. This mission sequence will be discussed in greater detail in the subsequent paragraphs.

In examining the possible launch trajectories all the contractors and the GSFC study group chose a 90 degree launch azimuth (measured from south).

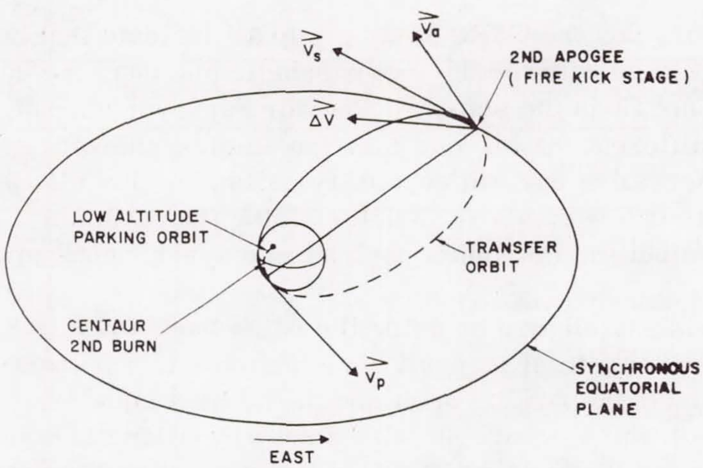

Figure VI-4. ATS-F&G ascent and injection geometry.

It should be noted, however, that if the launch azimuth were decreased, the longitude of the second apogee of the transfer orbit would lie closer to the operating longitude of 100 degrees west proposed by the study contractors (see Table VI-6) Figure VI-5 presents the ΔV penalty imposed on the apogee kick motor as a function of launch azimuth. It is seen that if the launch azimuth deviates from 90 degrees in either direction, the ΔV penalty increases with the deviation. Figure VI-6 presents the payload capability into the transfer ellipse for the Atlas-Centaur vehicle with a 15-foot shroud extension as a function of launch azimuth and orbit inclination. Figure VI-6 indicates that a payload penalty results from any deviation of the launch azimuth from the nominal 90 degrees.

Table VI-6

Launch Trajectories		
Launch Azimuth (measured from south)	First Perigee Longitude	Second Longitude
90	2.8°W	59.05°W
85	12°W	68.25°W
80	21°W	77.25°W
75	29.5°W	85.75°W

The study contractors and the GSFC study group all indicate that due to packaging problems associated with the 30-foot parabolic antenna, the undeployed spacecraft will not fit in the standard Centaur Surveyor shroud. The shroud extensions were different for all four mission studies; therefore, in order to arrive at a conservative payload capability estimate, the full 15-foot extension was assumed as indicated above in determining the optimum perigee plane change to inject the maximum payload into synchronous orbit.

The transfer orbit is established by using the Atlas burn and a first Centaur burn to inject the Centaur/spacecraft combination into a 90 nautical mile nominally circular inclined parking orbit and at the first equator crossing (descending node) which occurs at 2.8 degrees west longitude the Centaur is reignited to inject the spacecraft into a Hohmann transfer orbit having a perigee altitude of 90 nautical miles and an apogee altitude of

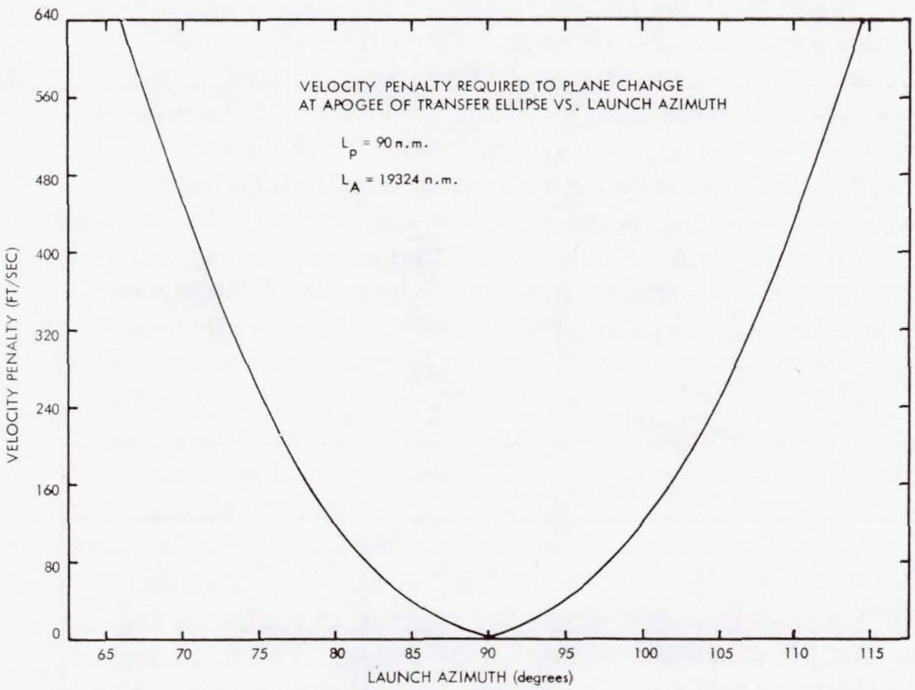

Figure VI-5. Velocity penalty on apogee kick motor as a function of launch azimuth.

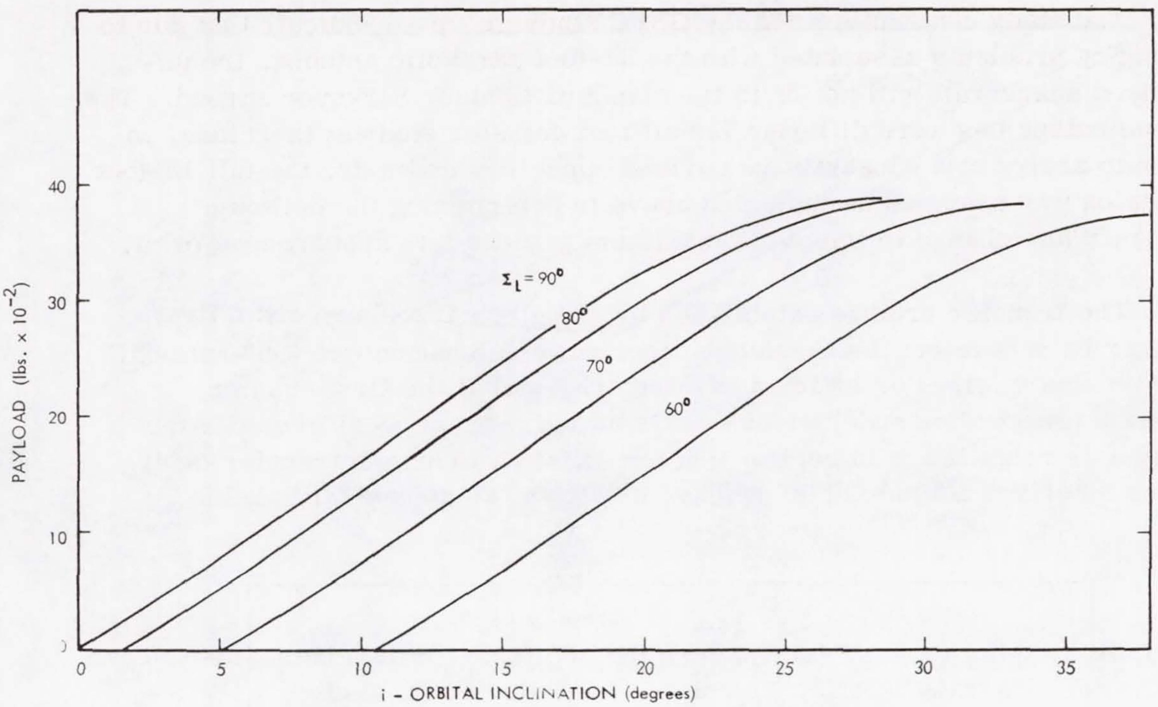

Figure VI-6. Payload versus orbital inclination.

19,323 nautical miles. Between the two Centaur burns the vehicle is yawed (rotated about the local vertical) 32.7 degrees above the orbit plane in order that an 8 45 degrees orbit plane change be performed with the second burn. The reason for the plane change is that the TE-M-364-3 motor is not capable of injecting into synchronous orbit as large a spacecraft as the Atlas-Centaur can inject into the transfer orbit. Thus, the total payload capability of the Atlas-Centaur vehicle is reduced by executing the plane change, but the resulting smaller orbit inclination reduces the ΔV requirement on the TE-M-364-3. The net result is more payload injected into the final orbit. The optimum plane change is one where the payload at apogee of the transfer orbit matches the ΔV capability of the TE-M-364-3. The 8.45 degree plane change is the optimum as explained in Section VI-B of this report. Figure VI-7 presents the geometry of the Centaur second burn and Figure VI-8a illustrates the vehicle yaw before the second burn.

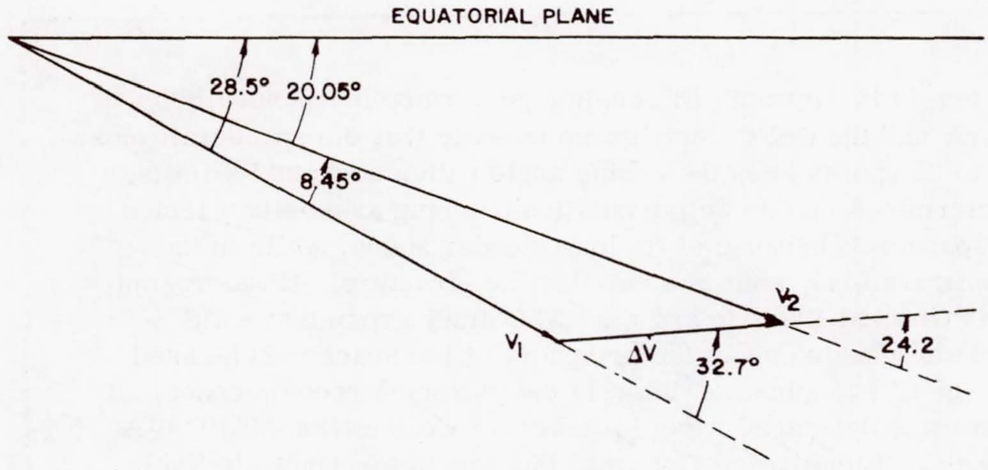

Figure VI-7. Centaur perigee plane change geometry.

Following the Centaur second burn the approach to the mission chosen is that of the General Electric, Fairchild-Hiller and GSFC studies. Following the Centaur second burn the Centaur is yawed down in to the orbit plane (negative rotation) through an angle of 165.1 degrees (see Figure VI-8b). This is the direction that the spacecraft thrust axis must point for apogee injection. The spacecraft is then separated from the Centaur and spun-up. The Centaur Project Office estimates the angular separation rates will not exceed 0.75 degrees per second. The justification for selection of the spin-

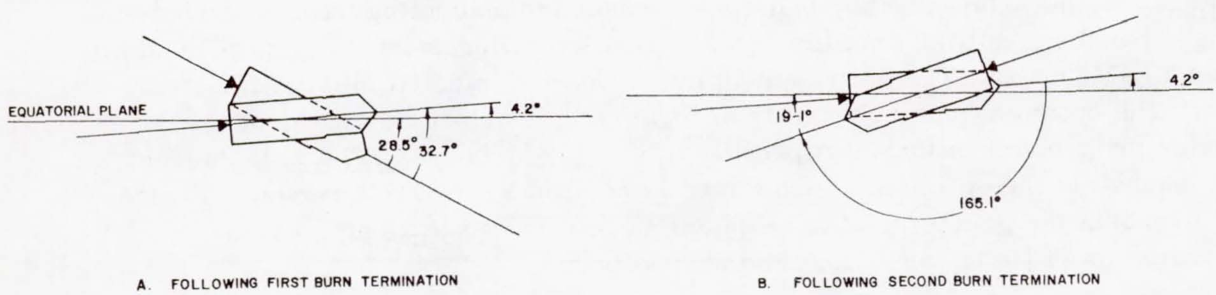

Figure VI-8. Centaur yaw maneuvers.

stabilized system is in Appendix B. Studies performed independently by General Electric and the GSFC study group indicate that the spacecraft must be spun at 60 to 80 rpm to keep the coning angle within acceptable limits. Also it was determined that an active nutation damping system be carried on-board the spacecraft because of its long slender shape, while in the undeployed configurations, and its semi-flexible structure. It was recommended by both General Electric and the GSFC study group that solid propellant rockets mounted about the periphery of the spacecraft be used for spin-up. The GSFC spin-up system is the one being recommended. It consists of four equally spaced Atlantic Research Corporation MARC 6-A 1-KS-210 rockets. The characteristics of this motor are shown in Table VI-7 and Figure VI-9. The reason for this choice over the General Electric approach is that since the General Electric system uses only two rockets, the failure of one motor would give the spacecraft only 50 percent of its desired spin rate and would place a large translational acceleration on the spacecraft. After separation and spin-up the spacecraft coasts to the first apogee (104 degrees east longitude) where a spin axis measurement is taken and any necessary correction is performed. Just prior to the second apogee (59 degrees west) the TE-M-364-3 motor is ignited and the spacecraft injected into a near synchronous equatorial orbit. The apogee burn geometry is presented in Figure VI-10.

Figure VI-11 is a mercator plot of the sub-satellite point for the entire mission. Table VI-8 lists the major events of the mission and the item number corresponds to the same number on Figure VI-11.

With the selection of the spin-stabilized launch concept and the specification that an active nutation damping system should be carried on-board

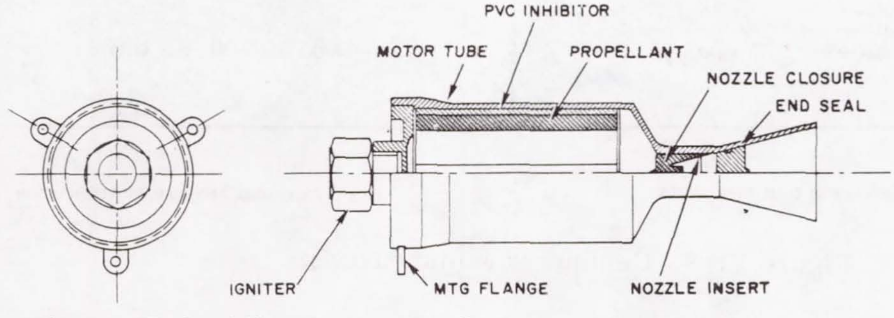

Figure VI-9. Atlantic Research Corporation MARC 6-A 1-KS-210 rocket motor.

Table VI-7

MARC 6 A1 1-KS-210 Characteristics (vacuum)	
Characteristic	Value
Burn time (70°F)	1.077 sec
Action time	1.501 sec
Average thrust	406 lb
Average pressure	1046 psia
Total impulse	222 lb-sec
Specific impulse	218 sec
Burning rate	0.367 in/sec
Loaded weight	3.38 lb
Expended weight	2.32 lb
Length	10.2 inches
Diameter (maximum)	3.062 inches
Igniter	two parallel squibs
Firing current	3.0 amps
Maximum no-fire current	0.2 amp
Resistance	0.35 - 0.65 ohm

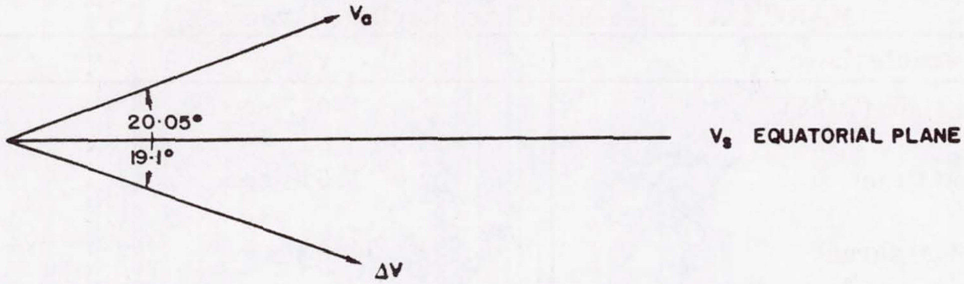

Figure VI-10. ATS-F&G apogee burn geometry

the spacecraft, the injection error removal and despin system selection narrows down to General Electric and GSFC (Lockheed is three-axis stabilized and Fairchild has no active nutation damping capability prior to antenna deployment).

The GSFC station acquisition propulsion system uses two 5-pound hydrazine thrusters for all spacecraft maneuvers. Therefore, a number of precessions are required to remove the eccentricity and inclination. General Electric uses a 10-pound radial jet together with 1-pound axial jets to avoid the necessity of spacecraft precession. From the standpoint of orbit correction either system is acceptable. Their relative merits are discussed in Section VII F (Auxiliary Propulsion).

General Electric uses 1-pound thrusters to despin the spacecraft while GSFC uses a yo-yo system which requires resistance jet vernier propulsion to take out the residual rates. The yo-yo system presents no problem if it is wrapped around the CG. If it is not, it may be critical because of the unstable inertia ratio for the undeployed configuration. General Electric, on the other hand, takes 5 to 6 minutes to despin the spacecraft. This is

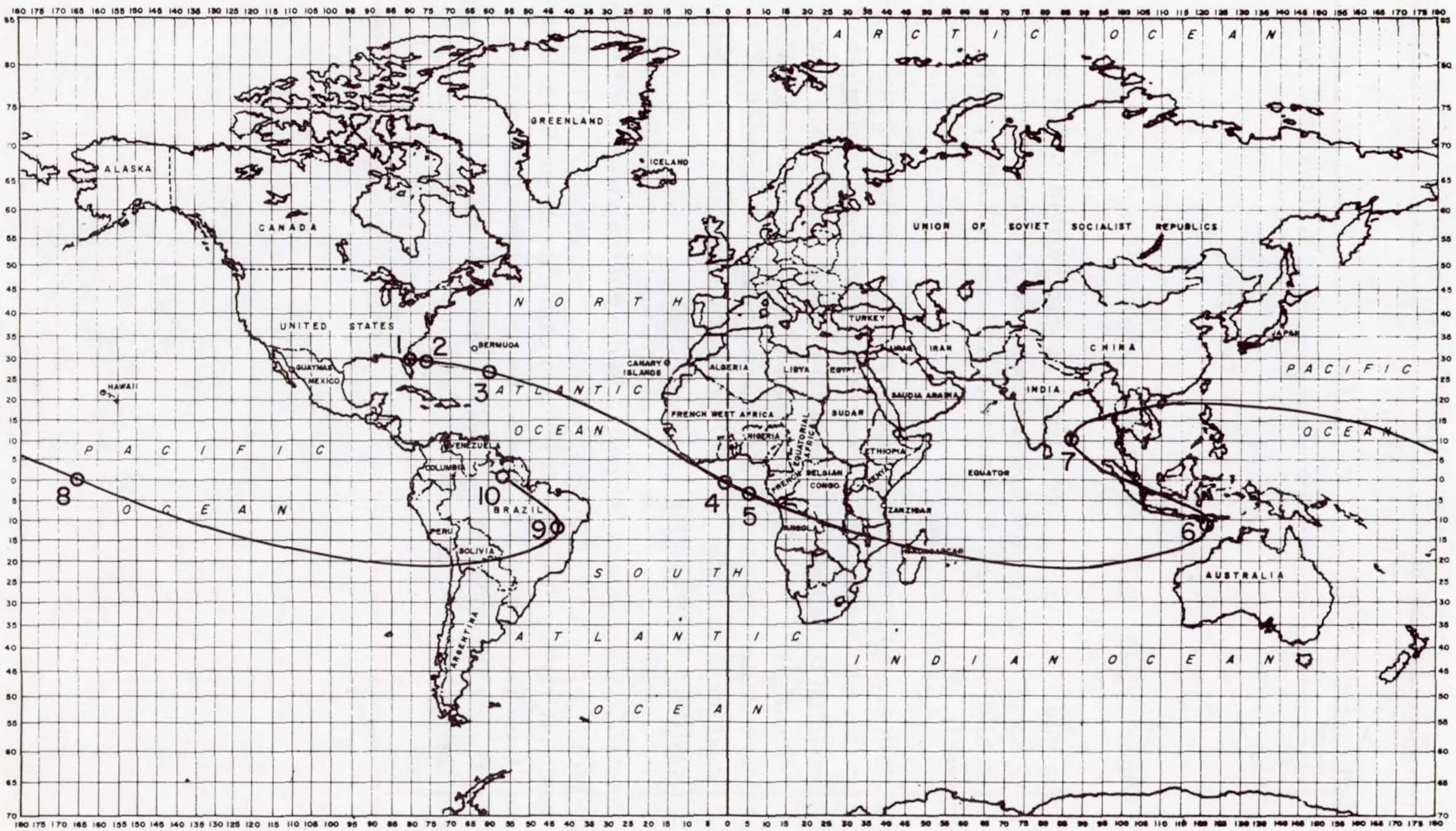

Figure VI-11. ATS-F&G trajectory.

Table VI-8

Mission Major Events

Item	Time (hours)	Event
1	0	Lift-off
2	0.070	Shroud drop
		Centaur first ignition
3	0.165	Centaur first cut-off 90 n.m. parking orbit established Centaur 32.7^{o} yaw maneuver
4	0.427	Centaur 2nd ignition
5	0.454	Centaur 2nd cut-off 165.1^{o} yaw maneuver Separate and spin-up
6	2.517	Initiate spin axis measurement
7	5.677	Complete spin axis correction
8	10.927	Second perigee
9	13.017	Initiate spin axis measurement and correction
10	16.177	Ignite apogee motor

an undesirable characteristic. Both General Electric and GSFC set a minimum spin rate for spacecraft stability. General Electric proposes to despin slowly through the unstable region presenting the danger of tumbling. The relative merits of a yo-yo and a hydrazine despin system will have to be determined by a more detailed analysis when the design details are further refined.

It can be concluded that the sequence presented in this section will reliably place the spacecraft at the desired synchronous station with errors reduced to operational specifications.

D. COMPUTATIONAL SUPPORT

The ATS-F&G program will require substantial computational support in the following five areas:

1. General Mission Support
2. Orbit Determination
3. Attitude Determination
4. Apogee Motor Firing
5. Spacecraft Reorientation and Orbit Adjustment

Details of the appropriate computer programs are available (Reference VI-1) and will not be repeated here.

GENERAL MISSION SUPPORT

Appendix I (Reference VI-2) describes a near nominal support sequence similar to that which will be required by ATS-F&G. The spin axis will be reoriented during the transfer orbit, and orbit synchronization will take place at second apogee. Following synchronization the spin axis is oriented in the orbit plane, and the axial thrusters are used to remove residual orbit eccentricity. The spin axis is then oriented normal to the orbit plane, and the same axial thrusters are used to remove residual orbit inclination. Finally, minor adjustments are made in the orbit period so that the spacecraft drifts to the desired location at 100 degrees west where a final period adjustment places it in synchronous orbit. Figure VI-12 is a general flow diagram of ATS-F&G computational mission support.

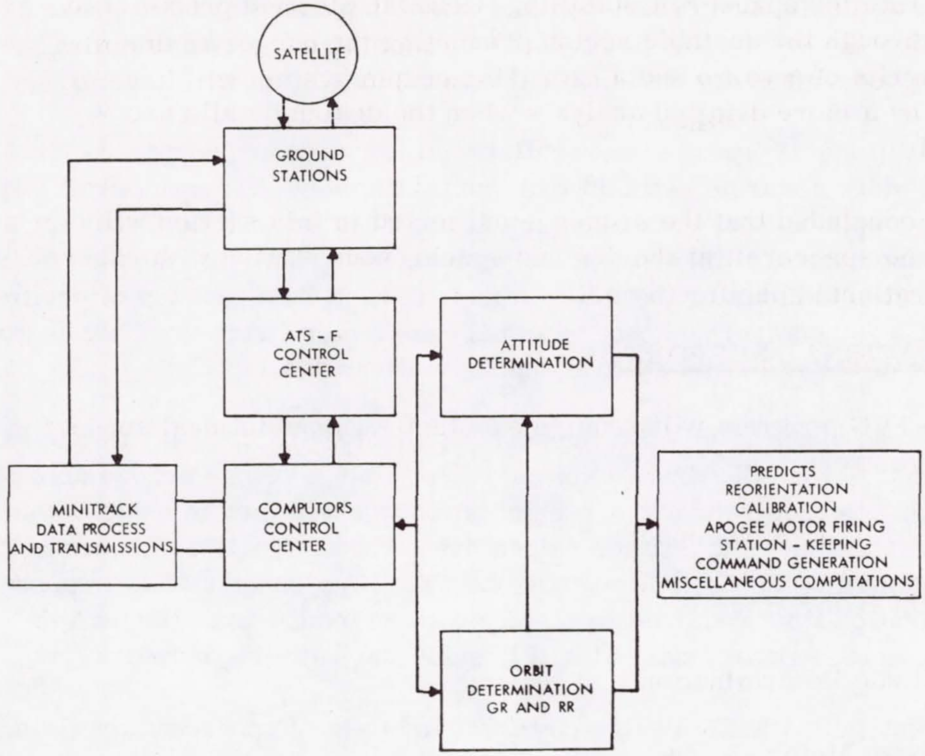

Figure VI-12. ATS-F&G computational mission support.

The initial support phase runs from liftoff, t(0), to approximately t(0) + 11.0 hours. During this time computational support provides:

1. Updated acquisition ephemerides for all participating stations based on the successively more accurate transfer orbit approximations obtained from the range and range rate and Minitrack data

2. Determination of spacecraft attitude using the nominal orbit and initial attitude data. When the orbit estimates attain an acceptable confidence level, the latest orbital elements will be used in place of the nominal set

3. Determination of time and desired attitude or apogee motor firing

4. Specification of reorientation maneuvers to be performed

5. Rapid definition of non-nominal mission parameters and presentation of contingency plans that will maximize the usefulness of the spacecraft under non-nominal conditions.

The initial support phase is essentially an iterative sequence, utilizing successively more accurate estimates of orbital elements and spacecraft attitude. The two spin-axis orientations (actual and desired) will be linked through definition of the control maneuvers to be performed during later support phases. It should be noted that during the initial mission phase the accuracy of preliminary estimates for control maneuvers and apogee motor firing are only as good as the quality of the orbit and attitude approximations available.

The second phase of mission support begins at t(0) + 11.0 hours and extends somewhat beyond the time of apogee motor firing. Since second apogee synchronization is required, and a reorientation is necessary to achieve this goal; the orbit estimate and attitude values available at t(0) + 11.0 hours will be used in computing the control maneuver. This time constraint is imposed by the preparation time required to specify a given maneuver. During this second phase computational support will provide the following outputs:

1. Determination of desired attitude and time of apogee motor firing to a high degree of accuracy

2. Specification of all maneuver parameters

3. Definition of an attitude correction maneuver, if required

4. Determination of an initial post correction attitude estimate

5. Determination of initial post synchronization orbit and attitude estimates

6. Location of acquisition ephemerides for the synchronous orbit.

The third phase of mission support begins at injection into the synchronous orbit and ends when the spacecraft arrives on station. During this interval computational support will provide the following outputs.

1. Definition of maneuvers to adjust the orbit period

2. Definition of maneuvers to circularize the orbit and remove the residual orbit inclination

3. Definition of maneuvers to reduce the drift rate to zero at the on-station longitude (100 degrees west).

Computational support of the ATS-F&G missions will require the use of three computers, one for orbit determination, one for attitude determination, and the third for apogee motor firing and related computations.

Orbit Determination

Determination of the orbital elements will be accomplished by a differential correction program based on a variable-order Cowell numerical integration method. The effects of solar radiation, luni-solar gravitation, zonal and tesseral harmonics of the earth potential as well as biases in station position will be included.

Inputs to the differential correction program will be the nominal orbit parameters and tracking data from the Ground Range and Range Rate tracking stations, from the Minitrack stations, and from STADAN's X-Y tracking antennas.

Outputs from the differential corrections program will be the parameters defining the orbit. These parameters will be semi-major axis, eccentricity, inclination, argument of perigee, mean anomaly, right ascension of ascending node, drift rate, period, height of apogee, and height of perigee.

The outputs of the differential program will be sent to the computer control center to be distributed to ATS-F&G control center and the attitude determination group. Further outputs are tracking and telemetry predictions for the tracking stations. Figure VI-13 is a flow diagram for the orbit determination program.

Attitude Determination

The purpose of the ATS-F&G attitude determination program is to estimate and update knowledge of the satellite spin-axis orientation during the transfer ellipse up to the time of the despin maneuver in synchronous orbit. Sensor data for this purpose is available when the vehicle is within the coverage zone of a designated tracking station and can include one or more of the following data types:

1. Polarization angle (POLANG) of the received electric field radiated from the satellite, defined as the dihedral angle between two planes intersecting along the line-of-sight vector to be the satellite. One plane contains the spacecraft spin axis and the other plane contains the local station zenith vector.

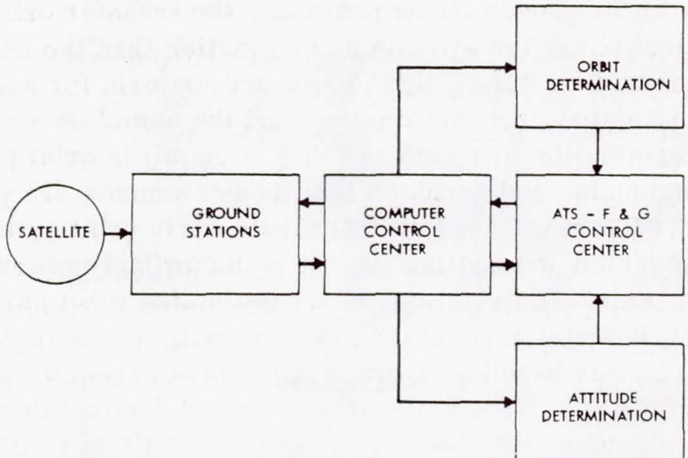

Figure VI-13. Orbit determination program.

2. The time between consecutive pulses from the two sun sensor slits is transformed in a ground computer program to the sun angle between spin axis and sun line.

3. SCADS (Scanning Celestial Attitude Determination System) is a star mapping system, which through a ground computer comparison with a known star map provides spin axis orientation directly. Although presently planned as an experiment, proof of feasibility on earlier flights could result in SCADS becoming the primary attitude determination method with the polarization angle-sun sensor technique serving as a secondary or back-up system.

Polarization angle data is distinct for each station, being dependent on zenith and line of sight directions. Also POLANG measurements may include a fixed bias for each station and the attitude program is required to estimate these biases together with the spin axis attitude, the latter to be given in terms of right ascension and declination. Appropriate error estimates or confidence levels for these determined parameters are also to be provided.

The above information will be supplied primarily by the Attitude Determination program (ATTDET) which is an iterative weighted least squares differential correction routine requiring initial estimates of right ascension and declination. Options are available for including POLANG biases as additional variables. Standard deviations of measurement noise for each data type can be either preassigned or estimated from the data residuals, for later use in the least squares weighting matrix. Error estimates are expressed in terms of the error ellipse parameters.

Because of the relatively short time interval available and the need for accurate attitude inputs to the apogee firing program, the transfer orbit is a considerably more critical phase for attitude determination than the subsequent mode. Of particular importance during this period are criteria for assessing the quality of both the estimated orbit parameters and the actual observational data. The Attitude Determination program (ATTDET) requires orbit parameters to generate observation residuals, and assumes that these elements are perfectly known. Premature ATTDET runs based on inaccurate early orbit estimates will yield incorrect right ascension and declination; it is accordingly necessary to defer such runs until the standard deviations of the estimated orbit parameters are reduced. The quality of the orbit estimate can be checked, for example, by making an attitude computation based on a minimum data set from a station (e.g., one sun angle and one POLANG). The right ascension and declination of the spacecraft spin axis thus computed is then compared with ATTDET results based on the orbit parameters. A large discrepancy would indicate the presence of orbit errors. It is intended to exploit this and similar types of tests during the early portions of the transfer orbit.

A different evaluation problem arises when the orbit is accurately known and the quality of the observations is to be assessed. Obvious "wild" points can usually be rejected by simple inspection. Significant errors in the remaining data may be discovered in one of the following ways:

1. Acquisition Table Program run based on previous updated attitude estimates, and comparison of computed and actual observational data

2. Comparison of RMS of residuals after an ATTDET run with prior knowledge of noise for each data type.

Tests of this nature will be performed on a continuing basis as new data is received and attitude updates computed.

When the orbit is well known and data acceptability has been established, several additional considerations relating to data type and quantity will affect the operational implementation of ATTDET differential corrections. Included are type of data, number of time spans of data points, observation noise level for each data type, and the effect on convergence of the initial attitude estimate. For example, sun angle data alone is insufficient for an accurate attitude update; some POLANG information is also required. However, POLANGS over a short time span are not as useful as perhaps fewer measurements over a longer interval where the overall change is greater. Considerations such as these will influence when to effect an up date, when to defer ATTDET runs until additional data is received, etc. In order to obtain a prior working knowledge of system performance under such conditions as cited above, a number of tests will be completed prior to launch. Results obtained will aid in scheduling program runs under actual operating conditions and in anticipating accuracy levels for the corrected attitude parameters.

Apogee Motor Firing

The objective of apogee motor firing is to remove the transfer orbit inclination and circularize the orbit. Since the single burn apogee motor is of fixed characteristics, a supplemental controllable thrust source is required to remove the residual orbital errors and when the spacecraft arrives on station (100 degrees west) to reduce the drift rate to zero.

Determination of apogee motor ignition requirements involves specifying several spacecraft/maneuver characteristics; during launch, these specifications must be made available well in advance of actual firing time to permit real-time command and control of the ATS mission.

To facilitate real-time decision making, preliminary estimates of these quantities will be formed early in the transfer orbit. As spacecraft data are refined, the estimates will be correspondingly improved and firm recommendations made. For each of the above apogee-fire procedures, the Hughes Aircraft Company developed FUSIT class of computer programs will be used to generate the required orbital and spacecraft maneuver data. All recommendations will be based upon this data and upon system/mission tolerances specified by the ATS-F&G project office. For further details, including apogee motor firing programs, refer to Reference VI-1 and VI-2.

Spacecraft Reorientation and Orbit Adjustments - To define reorientation maneuvers the following input data are required:

1. Latest orbital parameters

2. Present attitude of the spacecraft axis

3. The desired attitude of the spacecraft axis

4. Spacecraft status data

    a. Data which remain constant or which can be specified for a range of spacecraft states

b. Data which change between maneuvers and must be determined for each maneuver

(1) Fuel system pressures for each system

(2) Spacecraft weight, pounds

(3) Spacecraft spin axis MOI, slug ft^2

(4) Fuel weight available, each system.

Figure VI-14 shows the reorientation flow diagram.

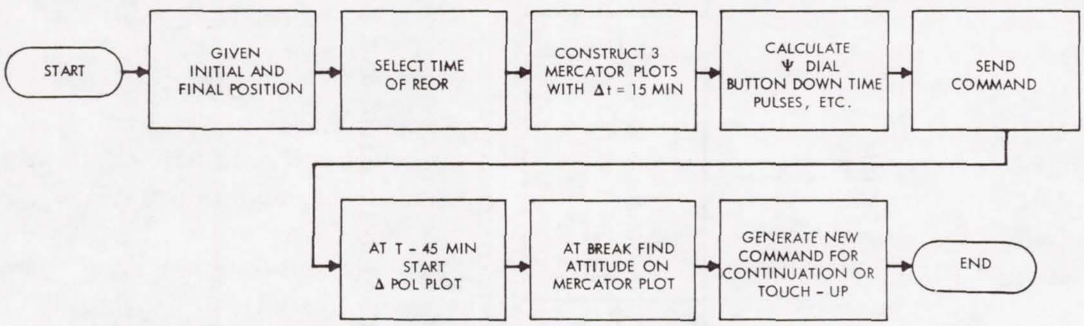

Figure VI-14. Spacecraft reorientation flow diagram.

A direct communication link between ATS-F&G Operations Control center and the commanding station is necessary. POLANG data will be received on a real-time basis and used as a direct check to assess the progress of the maneuver. In planning the maneuvers, the sun angle will be kept within the range of 90 degrees $\pm$ 25 degrees.

The data which must be supplied for orbit adjustment command is identical with the reorientation requirements. If special orbit adjustment requirements are needed, these will be supplied by the ATS Computer Control Center or the ATS Project. Figure VI-15, is a flow diagram for the Orbit Control Data program. The following orbit parameter objectives are defined:

1. On-station location at 100 degrees west longitude

2. Eccentricity approaching zero

3. Inclination approaching -1 degree (2-year spacecraft life results in 0-degree inclination at 1 year and +1 degree at 2 years).

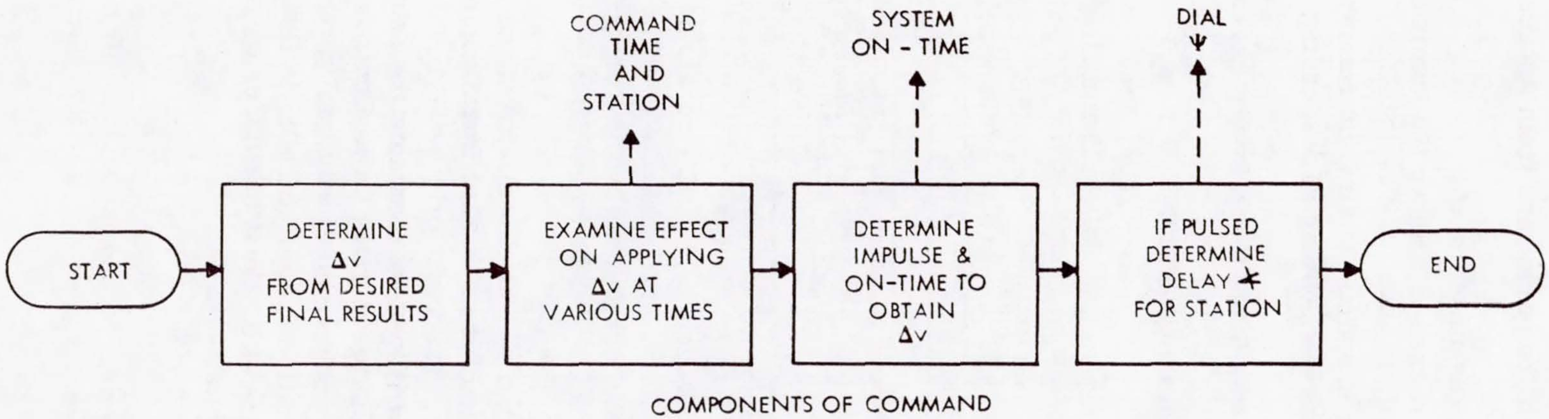

Figure VI-15. Flow diagram for orbital control data program.

The spacecraft axis will be reoriented during the drift to station at a time to aid in achieving the desired orbital parameters if no special time constraint is applied. The following maneuvers will be performed from apogee motor firing to on-station synchronism:

1. Drift rate increased or decreased to achieve the nominal drift

2. At ATS-F&G control center's option, inclination removal if needed using thrusters in continuous mode firing at descending node

3. Entry into a transfer orbit which has the following characteristics:

   a. Perigee at synchronous radius (22752.3 n.m.)

   b. Line of apsides rotated so that the perigee and on-station position are co-incident when the satellite arrives at perigee

   c. Semi-major axis such that the period of the satellite is commensurate with condition 2

   The time and velocity increment to enter into this special transfer orbit are calculated automatically in a specially designed program.

4. Removal of remaining transfer orbit energy at perigee of transfer orbit when exactly on station to achieve zero eccentricity and synchronous on station condition.

In general, all orbit adjustment maneuver times will be selected to remove eccentricity and inclination as well as to achieve the desired adjustment. A basic constraint will be minimal fuel usage.

The station to make the command will be selected as noted in Section 5.5.1 (Reference VI-1) except that no continual data readout facility is necessary.

Other then updating programs and system constants the only area where additional work by the computational support group is required is in the attitude determination portion of the programming system. If the GSFC-developed Scanning Celestial Attitude Determination System (SCADS) is flown as a primary attitude determination system on ATS-F&G, the ATTDET program will be modified to accept this new type of data.

E. RELIABILITY

Reliability was emphasized throughout the study period in choosing the spacecraft design. Flight-proven hardware was selected where possible, the number of components performing operational functions on the spacecraft was minimized, and redundant components were specified where the highly proven reliability of the component could not be established. Since the Phase A studies consider feasibility only, spacecraft systems, subsystems and components are not selected until late in the study, or not at all. Even if they are selected, they are subject to major changes during Phase B; consequently, the problem of reliability can be approached only in a general way.

Since the ATS-F&G spacecrafts are considerably more complex in design and objectives than the present ATS spacecraft series, a design lifetime of two years will be very difficult to achieve, and perhaps not possible.

During Phases B & C of the program, a reliability contractor will review the system design and recommend any improvements needed. The reliability assessments will consider design, quality assurance procedures, high reliability parts, and reliability specifications such as NPC-250-1. Upon completion of the reliability contractor's studies and the studies of the GSFC Quality Assurance Group, the design lifetime can be established and, if required, be part of the project's objectives.

Some specific reliability considerations are described in the following paragraphs.

ANTENNA REFLECTOR

A side-hinged petaline antenna was selected over flex-rib and other designs because it provides a more accurate reflecting surface. The side-hinged antenna was selected over the bottom-hinged, and the fixed-truss feed support was selected over the deployable boom, both because of higher reliability.

SPACECRAFT ATTITUDE CONTROL AND STATION-KEEPING

Reaction jets have a much higher proven reliability than ion engines. The highly successful reaction wheel jet hybrid was selected to provide fine pointing attitude control. The earth sensor and star tracker were selected to measure spacecraft roll, pitch and yaw angles because of their proven reliability. The interferometer and monopulse experiments can also provide these angles and will be used as back-up attitude sensors if they are selected.

SPIN-STABILIZATION DURING TRANSFER ORBIT

This approach was selected over three-axis control because of higher reliability. Spin-stabilization requires one 5-pound hydrazine thruster; two are provided for greater reliability. The three-axis control method would require four 60-pound and four 1-pound hydrazine thrusters and no redundancy is provided. The accelerometer used for nutation sensing is a simple device and no redundancy is planned.

DESPIN SYSTEM

Spacecraft spin is to be stopped by a yo-yo mechanism; however, the gas thruster system will be large enough to stop spacecraft spin by itself if the yo-yo devices fail.

THREE-AXIS STABILIZATION DURING OPERATIONAL MODE

No redundancy is planned for the sun sensor and gyro rate sensors, used during the sun acquisition mode, because of the extreme simplicity and flight-proven reliability of these devices. The gas jets are arranged so that the failure of one jet would be compensated for by the others. For the earth acquisition mode, the monopulse error detector serves as a back-up in case of failure of the earth horizon detectors, and the gyrocompass can be used in place of the Polaris tracker.

If the gimbaled gravity gradient boom experiment is chosen, it can provide a complete back-up for the control gas system after the spacecraft has reached the operational mode.

CONTINUED EMPHASIS ON RELIABILITY

After acceptance of a spacecraft design, reliability will be considered throughout all phases of the program: in conducting system and component tests; in selecting components, and in system design. GSFC policies and the appropriate NPC documents will establish the detailed reliability requirements.

SECTION VII

SPACECRAFT

A. SPACECRAFT STRUCTURE

The studies of the three contractors and GSFC have evolved four different structural concepts. Their similarities and differences will be discussed. Figures VII-1 through VII-4 are illustrations of the four concepts.

When viewed in the deployed condition, the four overall configurations look quite similar, the major difference being the number and size of the solar paddles. The preferred solar paddles configuration is discussed in Section VII G (Power). When viewed in the stowed configuration, it can be seen that two of the concepts, General Electric and GSFC, have two equipment modules with the antenna mounted to the module nearest the launch vehicle. The reflector which utilizes the petaloid concept, folds up from the launch vehicle around the upper module. The main differences between the GSFC and General Electric concepts are the f/d ratio of the antennas and the equipment arrangement in the aft-equipment module. GSFC chose an f/d ratio of 0.5 while General Electric chose a ratio of 0.4. The determination of the preferred f/d ratio of 0.44 is discussed in Section VII-B. General Electric arranged the electronic equipment in two large bays located outside the load carrying structure while GSFC arranged the electronic equipment throughout the inside of the load carrying structure. The advantage of the GSFC concept is that more efficient use of available space is possible. With the General Electric arrangement, future growth would require extensive structural modification and addition. Both concepts mount their solar paddles to extended reflector-deployment trusses. However, General Electric has four paddles while GSFC has two. General Electric's concept of four solar panels will require more solar cell area for the same power output due to the two panels in the orbit plane, shaded by the spacecraft and its reflector. The stowed arrangement of the solar paddles also differs between the two concepts. General Electric's paddles are folded back along side the reflector. This arrangement requires that some paddle launch loads be transmitted through the reflector. The GSFC concept allows support of the solar paddles directly by the feed support truss.

Fairchild-Hiller chose to configure their spacecraft with a single module mounted on the adapter which contains the apogee motor. The inverted reflector, also of the petaloid type, is folded down toward the launch vehicle. The primary advantages claimed for this concept are (1) a weight saving for the reflector mounting base support (this is partially offset by increased reflector weight), (2) an accessible (barely) CG for more efficient stationkeeping, and (3) simpler electrical harnessing since all the electronics are in one module. The disadvantages of this approach are:

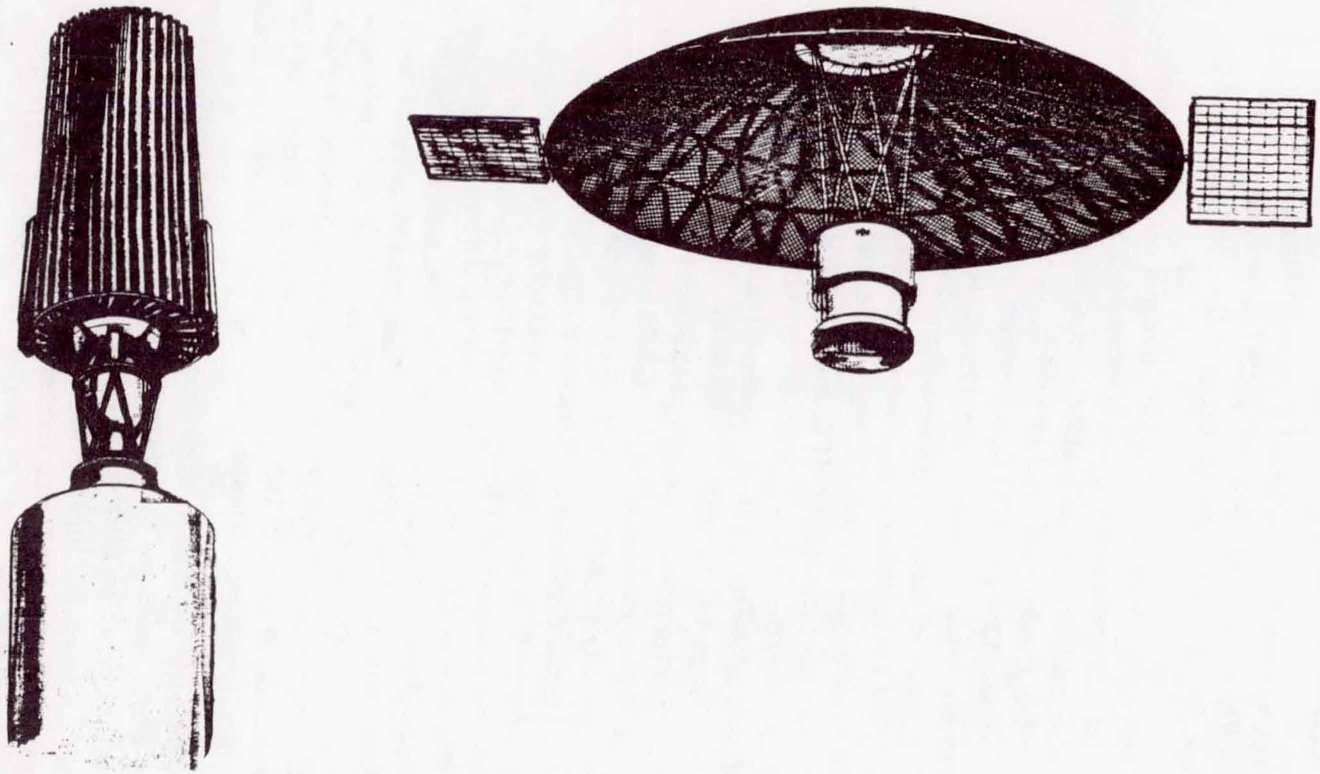

Figure VII-1. Fairchild-Hiller spacecraft concept.

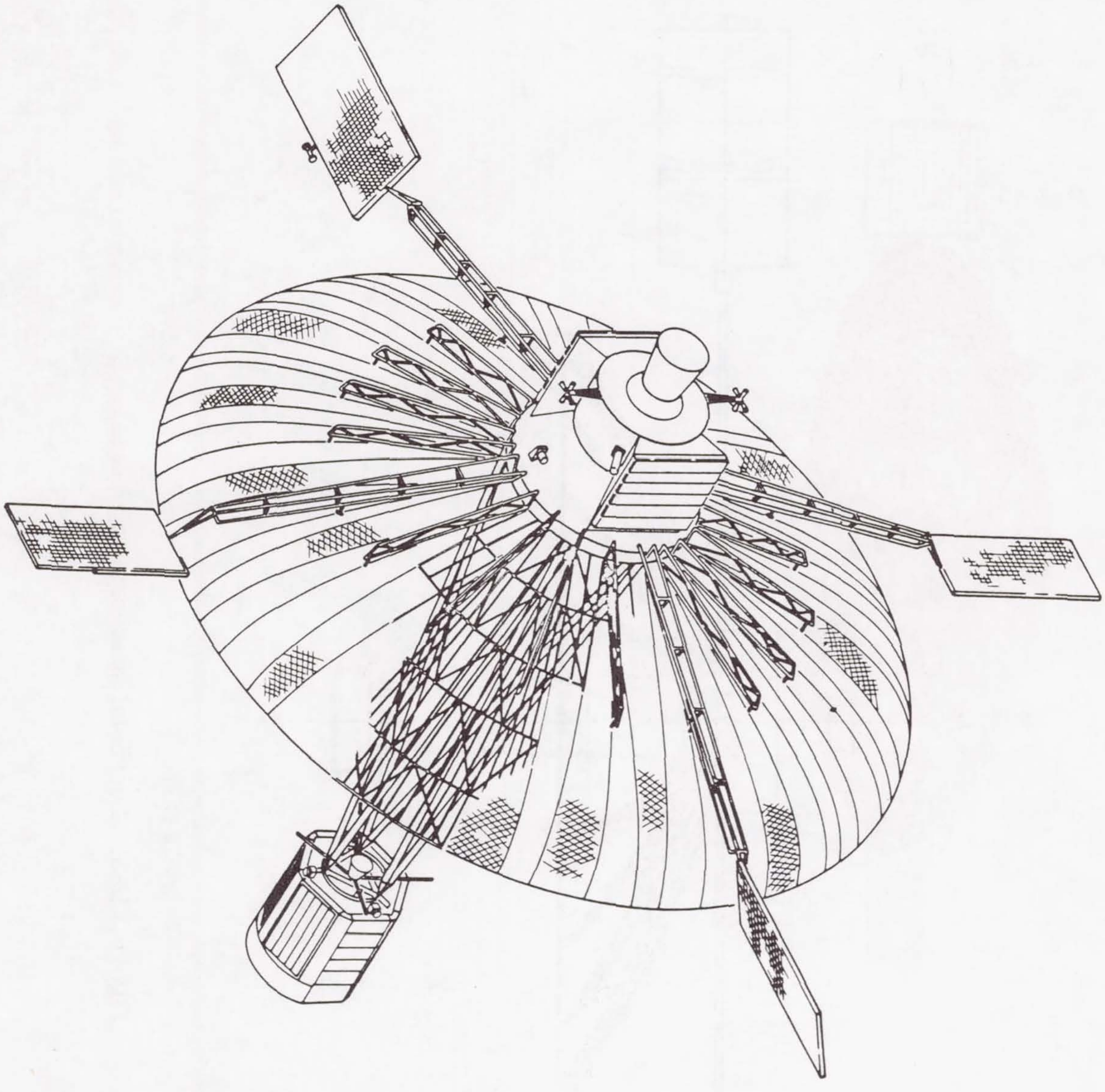

Figure VII-2. General Electric spacecraft concept.

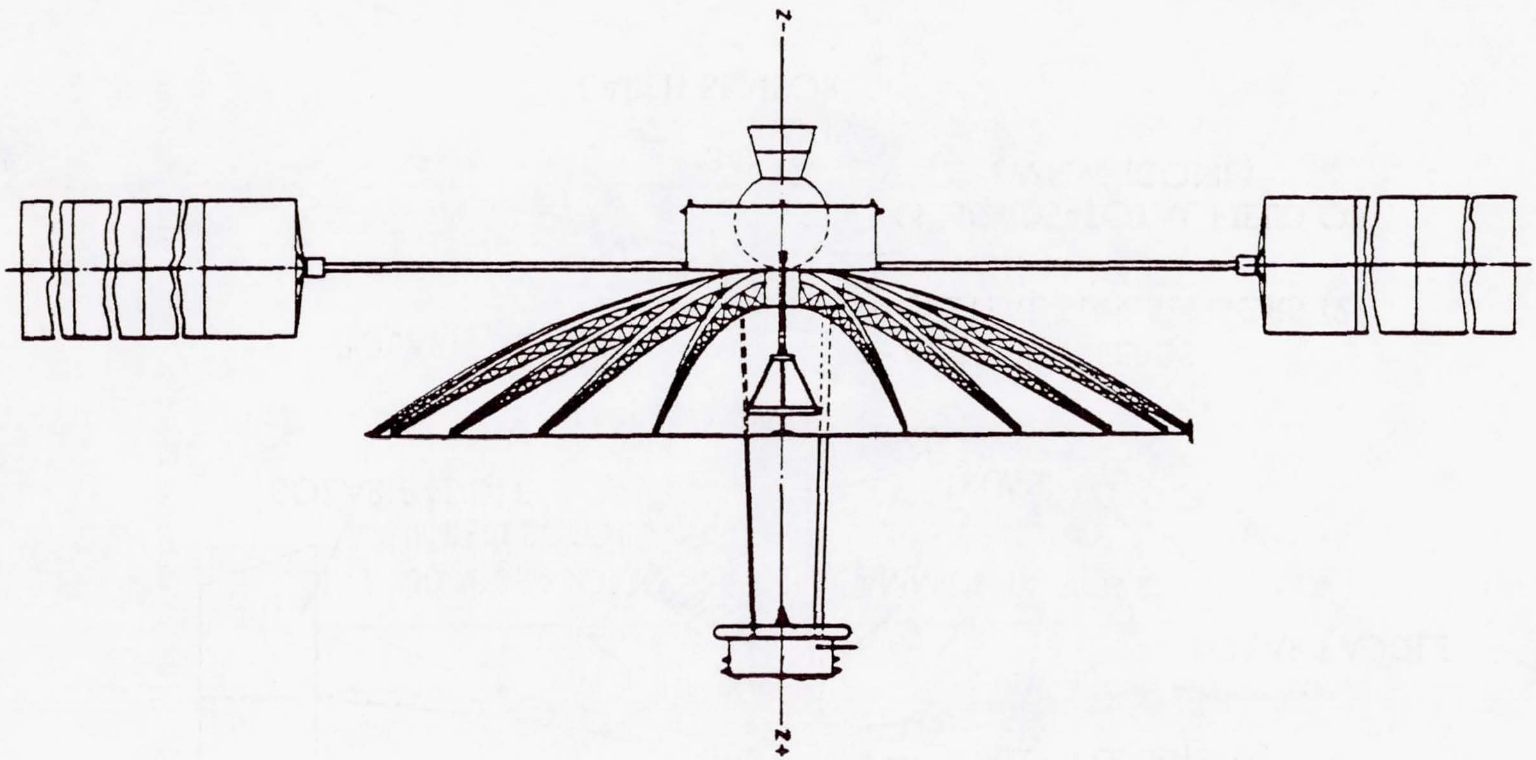

Figure VII-3. Lockheed spacecraft concept.

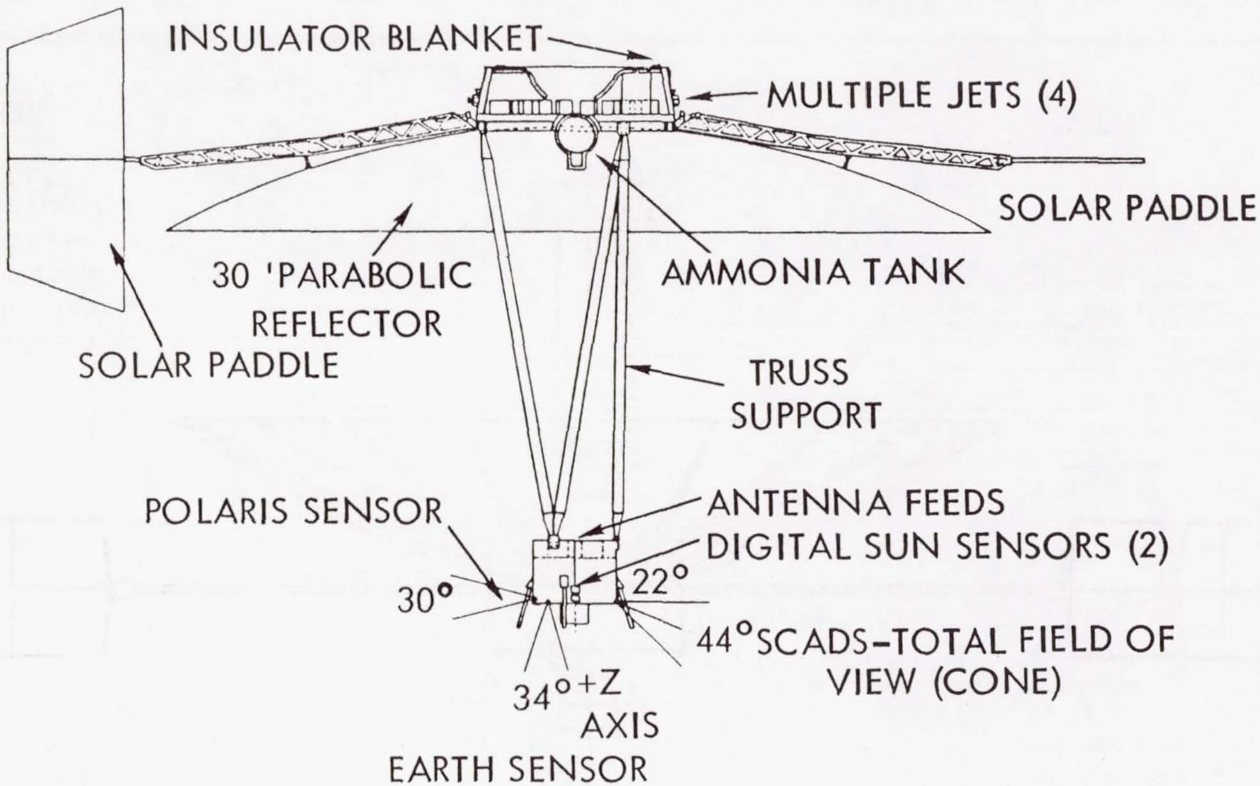

Figure VII-4. GSFC spacecraft concept.

Table VII-1

Weight Comparisons

Item	Weight (lb)			
	GSFC	GE	F-H	Lockheed
Attitude control	108	117	139	93
Station-keeping and attitude control	78	127	126	87
Power system	241	276	180	325
Telemetry and command	60	61	40	51
Parabolic reflector	200	178	274	200
Feeds	10	10	15	7
Spacecraft structure	160	168	150	199
Feed support	75	51	46	30
Thermal control	50	65	--	10
Transponders	60	73	55	46
Harness	75	72	30	65
Antenna instrumentation	35	13	**35	22
	1152	1211	1090	1135
Adapter	125	136	150	168
Bal. weights	30	*30	*30	*20
Vernier propulsion	110	107	106	134
Spin up despin	20	* 7	9	0
Apogee motor fuel	1440	1440	1440	1440
Apogee motor dry weight	140	139	145	130
TOTAL	3017	3073	2970	3027
Capability	(1) 3254	3254	3254	3254
Excess for experiments and growth	247	194	294	237

(1) Launch vehicle capability as determined by NASA/Lewis and GSFC.

* Omitted by the three contractors, but required.

** Assumed by GSFC from their equipment description.

1. Failure to jettison the apogee motor is catastrophic to mission success because it would prevent antenna deployment.

2. The growth potential and versatility of this concept are very limited.

3. If the payload capability were significantly increased by using Titan IIIC, this additional weight would have to be accommodated in the same module.

4. The size of the module is limited by the shroud's conical shape and acceptable RF degradation, hence full advantage of the increased capability may not be realized without adding a second module behind the antenna.

5. Spacecraft structure concept that will have the necessary flexibility to conduct various experiments, including space looking ones, some undefined at this time, requires versatility which is not provided by this concept.

These disadvantages are significant enough to cause rejection of the Fairchild-Hiller structural concept.

The Lockheed concept utilizes the flex-rib antenna. This makes the launch configuration quite different because the antenna wraps on a drum to make a compact package. In addition, Lockheed chose a deployable feed package and deployable solar paddle supports. In the launch configuration this further reduces the size of the overall spacecraft. The feed package and solar paddles are deployed by a Lockheed Missile and Space Company-developed tri-beam extendible boom. Offsetting the advantage of the compact spacecraft is the reduced reliability associated with deployable structures. Since GSFC has selected the hinged petaloid reflector concept, the Lockheed spacecraft concept would lose its compact advantage and would not be compatible with the selected reflector.

For ATS-F&G the on-station weight of the spacecraft is a primary area of concern. The Centaur capability allows a spacecraft weight of 3259 pounds to be launched, however, when deductions are made for the apogee motor, transfer orbit propulsion, and adapter; the on station weight of the spacecraft

In the area of attitude control and station-keeping propulsion, the GSFC weight is lower than that of General Electric and Fairchild-Hiller because the Goddard study assumed minimum north/south station-keeping. The Lockheed weight is lowest because their control system does not utilize reaction wheels, and their thrusting system utilizes primarily ion engines. In the area of solar paddles, Fairchild-Hiller weight is low because they attach the paddles to the tip of the reflector, hence no support weight is included. This effect results in a higher weight for the reflector. Lockheed's weight for solar power is much greater than that of the others because of their large power requirements (900 watts average). In the area of feed support the Goddard weight is high because of the assumed f/d ratios of 0.5. General Electric and Fairchild-Hiller assumed f/d ratios of 0.4 and 0.32 respectively, hence their lower weights. Lockheed's weight of 30 pounds is derived from the fact that they have a deployable feed. In the area of thermal control, Fairchild Hiller and Lockheed assumed passive control. It is felt that this assumption is not realistic and consequently, these weights will increase. Fairchild-Hiller harness weight is low because of their single module approach; however, it is felt that their estimate is somewhat low. Because Lockheed uses a three-axis control during the transfer orbit, they require no weight for spin up and despin. The table shows about 200 to 250 pounds of available weight for growth and additional experiments.

In this spacecraft 285 pounds for the antenna which is inherently a part of the spacecraft structure can be considered experimental work. This yields about 485 to 530 pounds of total experiments for ATS-F&G. As noted in Section VI B of this report, an optimized apogee motor can be developed which will provide an additional 205 pounds capability. Also, growth in the Centaur launch vehicle could provide an additional 35 to 65 pounds increase in capability. The 200 to 250 pounds of additional experiment and spacecraft weight growth is an uncomfortably small margin at this point in the program. Thus, this problem must be assessed in more detail.

Portions of the spacecraft structure are considered difficult problem areas, therefore, a related SRT task has been submitted to conduct analyses of these problems at an early time. The SRT task, "ATS-F&G Spacecraft Structures", will deal mainly with the feed support and the solar paddle support structures.

To summarize, the structural approach that promises to meet the requirements of ATS-F&G most reliably is two equipment modules separated by a fixed truss with the antenna mounted to the aft-equipment module. With this configuration, the weight available for growth and additional experiments is about 200 to 250 pounds.

B. ANTENNA

INTRODUCTION

This section discusses the trade-off considerations in establishing the design of a deployable 30-foot diameter, X-band surface-tolerance parabolic reflector. The discussion will include the RF feed system and feed support structure which, when combined with the reflector, form the composite ATS-F&G large aperture antenna system.

DEPLOYABLE REFLECTOR CONCEPTS

The GSFC Concept Design Study Report, Section 11, contains a description of the various reflector stowage and deployment techniques that have been developed over the past four or five years. Therefore, this section will be limited to a comparison of the contractor's proposed concepts and a description of the preferred concept.

Lockheed Study

Lockheed selected the flex-rib concept. The flexible rib concept uses flexible-truss radial ribs to which a flexible reflective mesh is attached. This concept is shown in Figure VII-3, Section VII- A For stowage, the ribs and mesh are spirally wrapped about the central hub, which consists of concentric inner and outer rings. The ribs are attached to the inner ring and pass out through openings in the outer ring. For packaging, the rings are rotated opposite to each other about a common central axis so that the ribs and mesh are flexed sideways (tangentially) and drawn in against the inner ring. For deployment, a motor rotates the inner and outer hub rings in opposite directions and the ribs, with attached mesh, are forced out and into the deployed state. In the final operating condition the mesh is stretched taut between the ribs.

The flexible rib concept possesses an excellent launch configuration (i.e. low weight and compact); and deployment reliability appears to be good. Obtaining the required tolerances on the reflective surface, which is made up of single-curved fabric surfaces (inherent in this concept), is expected to be very difficult. Environmental cycling of the reflective surface, which will be under tension, adds to the problem of tolerance control and may easily degrade performance to an intolerable extent. Depending solely on tension to provide stability in the circumferential direction is believed questionable. This concept is an excellent possibility for a lower frequency antenna, but for this mission it is not considered adequate. An alternate petaloid concept proposed by a subcontractor, Electro-Optical Systems, Inc., was considered by Lockheed but was rejected in favor of their own design. The detailed description of both these systems can be found in the Lockheed final report.

Fairchild Mission Study

The Fairchild-Hiller Mission Study selected a petaloid concept for the reflector as their preferred configuration. This concept is shown in Figure VII-1, Section VII A. The petaloid type is composed of curved, rigid-truss, petal frames covered with RF reflective mesh (on standoffs) and hinged at their base. For stowage, the petals are rotated about their base line and are stacked front to back in a circular manner. (The petals and base hinge lines are skewed slightly from the parabolic radial lines). Deployment is actuated by torsion springs located at the base hinge line. Each petal is tiplocked to the adjacent petal and at the same time is hinged to the hub along its base in the operational configuration.

The concept approach of fabricating individual hard petals is believed to be a good method of obtaining the desired tolerance. However, a possible reduction in reliability exists because of the requirement for deployment of single petals, which depend mainly on base hinge line control and alignment; and successful latching of 32 individual tip locks.

General Electric Study

General Electric chose Goodyear Aerospace Corporation (GAC) as a subcontractor in the area of reflector design. This concept is shown in Figure VII-2, Section VII A. It should be noted that GAC was independently selected by GSFC by open competition in December 1966, for design and development of this concept. The description of the original Goodyear reflector is given in Section 11 of the "GSFC Concept Design Study Report."

Preferred System

Since the initiation of the reflector contract with Goodyear, several configuration changes have been incorporated in their basic design as follows:

1. 72 petals reduced to 36
2. 24 deployment trusses and deployment mechanism reduced to 12
3. Petal deployment mechanism changed to ball screw drives

4. Titanium face sheets over aluminum honeycomb core reflector material
5. $f/d = 0.44$
6. RF feed and earth viewing equipment module support truss changed to K truss.

The advantages inherent in these changes are discussed under their related sections which follow.

Thermal

Two major thermal problems must be solved in the design of the ATS-F&G antenna. Temperature gradients must be minimized to keep thermal distortions in the antenna within acceptable limits, and thermal excursions during an orbit must be minimized so as not to induce thermal stresses which may cause material failure.

The study conducted at Goddard indicates that the solution to both problems seems to maintain a low absorptivity to emissivity ratio (α/ε) and also a low ε. The low α/ε insures a long thermal time constant in sunlight while the low ϵ gives a long thermal time constant during earth shadow (a maximum of approximately one hour). Since a low α/ϵ and a low ϵ are somewhat incompatible, a compromise is required.

Fairchild's solution involves wrapping the entire antenna with superinsulation. The solution appears to be good from a thermal standpoint. However, the problems associated with applying and handling the insulation on such a complicated structure may outweigh its advantages.

General Electric's approach was to coat the convex and concave faces with evaporated aluminum ($\alpha = 0.12$, $\epsilon = 0.04$). This meets the requirements of low α and low ϵ but not low α/ϵ. The α/ϵ is reduced by painting the edges of the honeycomb black. This also has the effect of raising the temperature of the antenna when the sun in 90 degrees to the yaw axis which, except for possibly the period in the earth's shadow, is the coldest position for the antenna. Although the temperature gradients are more severe than for an insulated antenna, General Electric maintains that the deflections are still within tolerance. Assuming the deflection analysis is correct, General Electric's approach would seem to be the preferred design.

The flex-rib antenna was not analyzed by Lockheed to the same degree that the petaloid approach was analyzed by the other contractors. As noted earlier, the Lockheed concept was rejected in the competitive procurement and thus its thermal characteristics are not discussed further.

Thermal analyses conducted by the mission study contractors are discussed in their respective final reports. In general, the key to a realistic evaluation of antenna performance degradation is a sophisticated analytical evaluation of the antenna structure. Such an analysis can be carried out using a discrete member simulation of the antenna. Figure VII-5 shows the discrete member structure used by General Electric and Goodyear Aerospace Corporation. Using this analytical approach the following cases were studied:

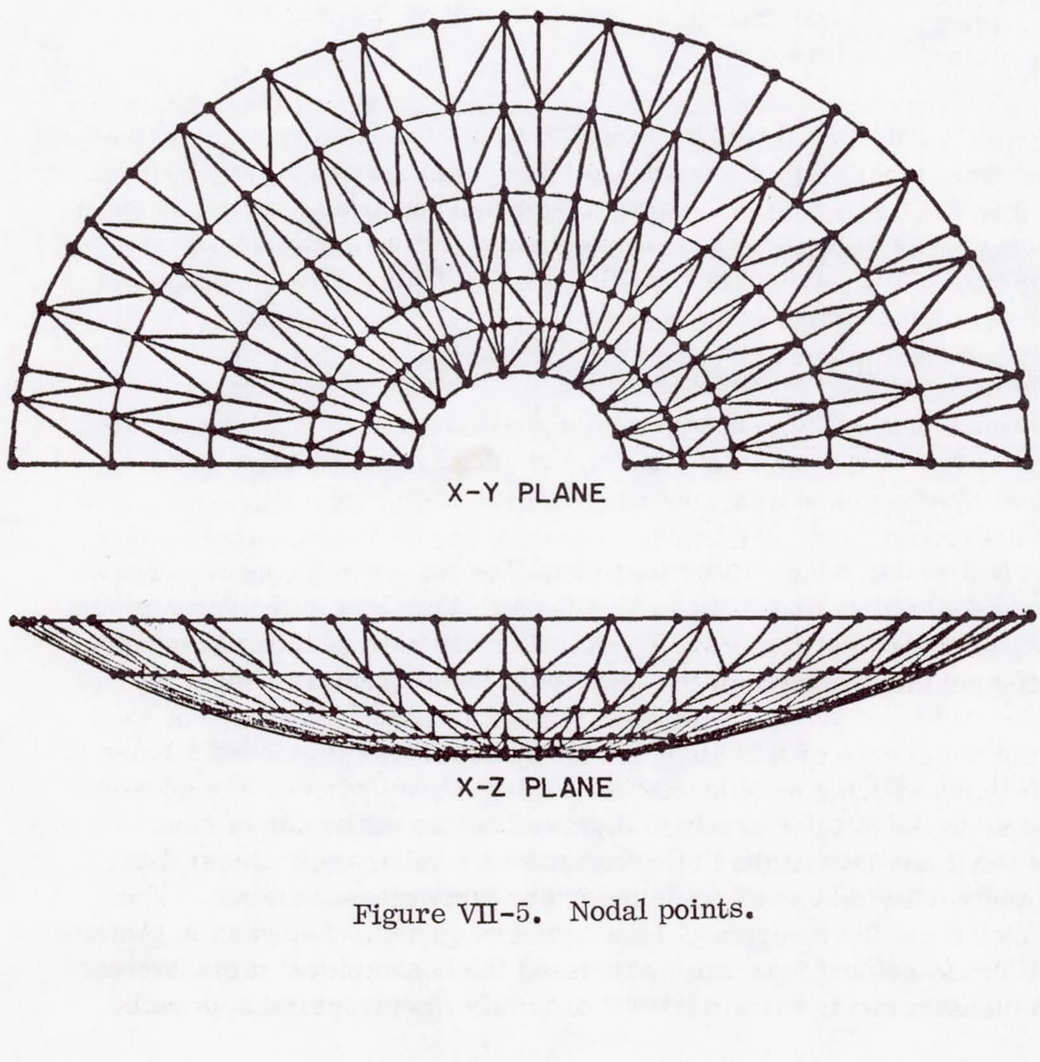

Figure VII-5. Nodal points.

Material

1. Titanium Sandwich
2. Beryllium Bars
3. Aluminum Bars

Hub Condition

1. Free
2. Fixed

Orbital Position

1. 90 degrees
2. 116 degrees

It was observed that aluminum honeycomb with titanium face sheets is a good selection due to its lightweight, high strength, and reasonable low thermal deflection. (See Figure VII-6.) The free hub is thermally better (small deflections) than fixed hub, but the resulting difference in RF gain is only about 0.3 db. Because of simpler structural attachment, the fixed hub is preferred. Preliminary studies show that an orbital position of 116 degrees which corresponds to sun being in the plane tangent to the reflector surface at an edge point is the most critical.

RF Feed and Earth Viewing Module Support Structure

A variety of support structures were suggested by the contractors. Both Fairchild-Hiller and General Electric proposed a fixed-truss. Lockheed proposed a deployable boom. The deployable boom approach was rejected because it introduced an unnecessary in-line function, thus a reduction in reliability. Calculations also showed that under the normal thermal environment an off-axis displacement of the feed could be as great as 1 inch.

During the course of this study all the contractors considered a truss to support the earth viewing module. At the mid-contract review, the question was raised as to the RF performance degradation caused by use of such structures. All contractors had theoretical loss numbers but no test data. Following the review they each made quick and dirty measurements. The results varied from the theoretical loss numbers quoted. Appendix A (Antenna Feed Mast Trade-off and Selection) discussed the contractors' measurements along with measurements made at GSFC to justify the preferred approach selected.

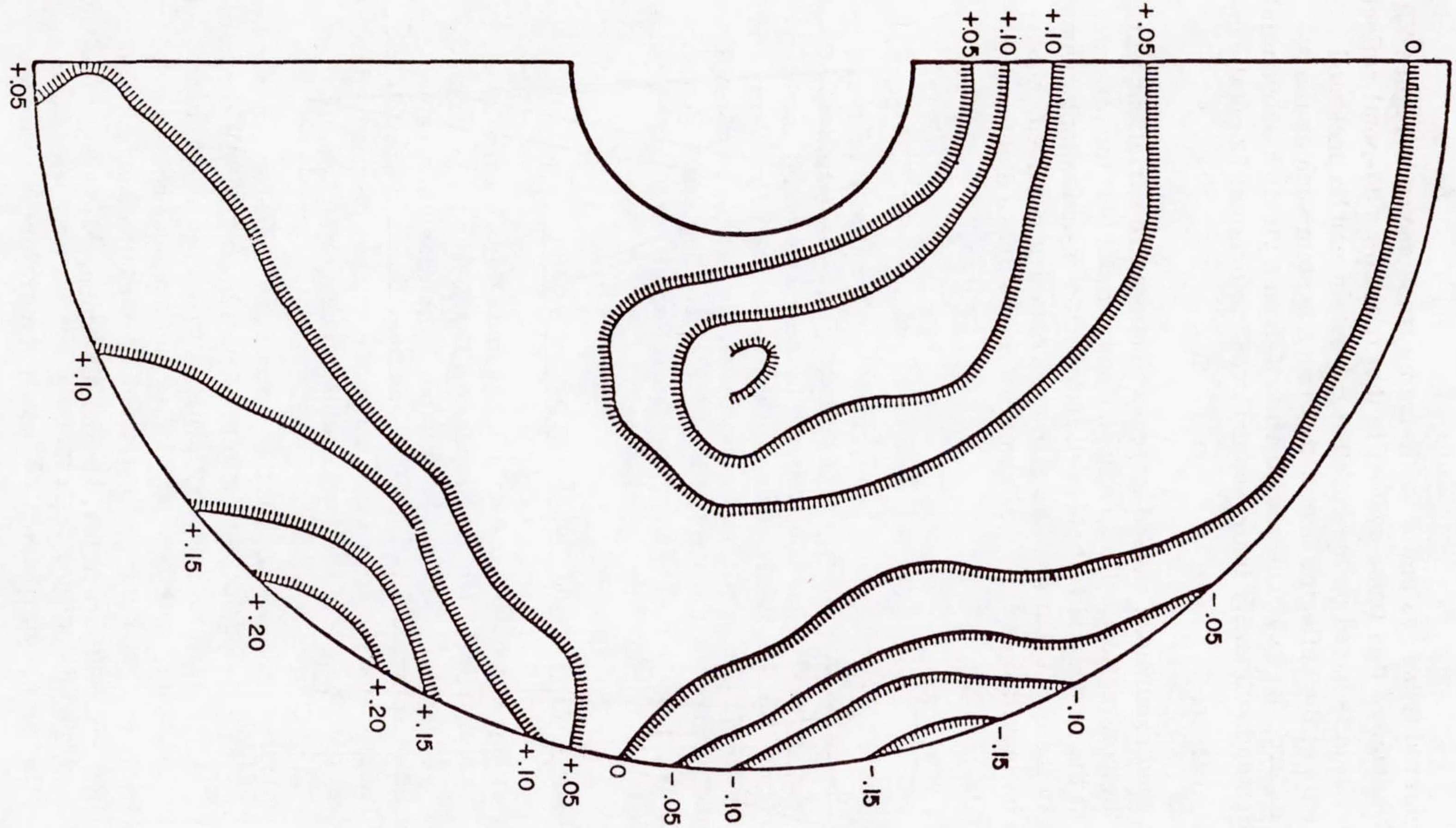

Figure VII-6. Contour deflection for fixed hub.

The preferred truss system is the K-frame truss shown in Figure VII-7a. The major advantage of this truss system is that it allows a 36-petal reflector configuration. This is based on the fact that the petals can be packaged closer to the axis of the reflector than with a truss system such as the A-frame. (See Figure VII-7b.) The combination K-frame yields a substantial weight reduction and reliability improvement over the original 72-petal configuration.

Antenna feed excursions caused by truss temperature variations were calculated by the contractors. These calculations showed that the off-axis displacement of the feed could be held to 0.10 inch and on-axis displacement to 0.15 inch. In the event that smaller displacements are required, superinsulation can be installed on the truss struts to reduce these displacements.

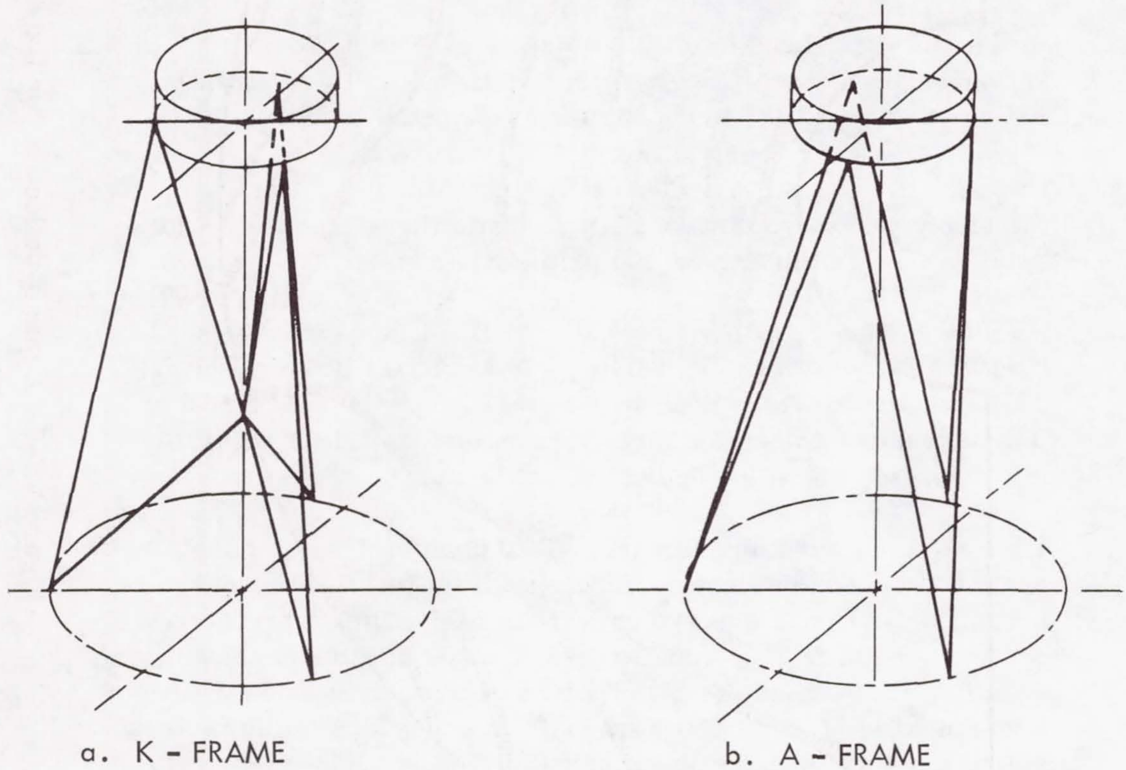

Figure VII-7. K and A frames.

As noted earlier, the petaloid side-hinged reflector concept has been chosen. Therfore, only the deployment mechanism under development at Goodyear will be discussed. In this system (Figure VII-8), deployment will be performed through translation of ball nuts due to rotation of ball screws which are in turn activated through the attached chain (or belt sprockets), the continuous chain (or belt) and the motor. There are 12 such systems attached to the 12 supporting trusses. Each of the deployment mechanisms weighs less than 1 pound. Further investigation and optimization will be carried out in the next phase of the program.

REFLECTOR RF FEED

The results of the mission study contractors antenna feed configuration studies are summarized as follows:

1. Reflector Focal Length to Diameter Ratio: The various f/d ratios were: General Electric preferred 0.4 to 0.5; Fairchild Hiller preferred 0.25 to 0.325; and Lockheed considered 0.5.

2. Cassegrain versus Prime Focus Optics: Although the three mission contractors discussed cassegrain (in some cases in great detail), all three preferred the prime focus optics.

3. RF Beam Scanning: In the scanning area all three discussed briefly their approach to scanning the secondary beam of the 30-foot aperture reflector. In general, it may be said that there was agreement among the three contractors that the scan limit off axis is governed by the data from the curves of Figure VII-9.

4. RF Feed Configuration: Figure VII-10 depicts the feed horn system proposed by General Electric. Figure VII-11 shows the proposed Lockheed feed system. Fairchild-Hiller's preferred system consisted of composite feed of a 100 MHz turnstile antenna, an 800 MHz broadside array of 4 turnstile antennas and a multiple arm spiral of M arms operated in M-1 modes. This multi-arm spiral would allow monopulse operation. The 100 MHz and 800 MHz are electrically driven against gound planes.

PREFERRED CONCEPT

To arrive at a preferred RF feed system for the large aperture antenna, it is necessary to evaluate the several parameters which influence the configu-

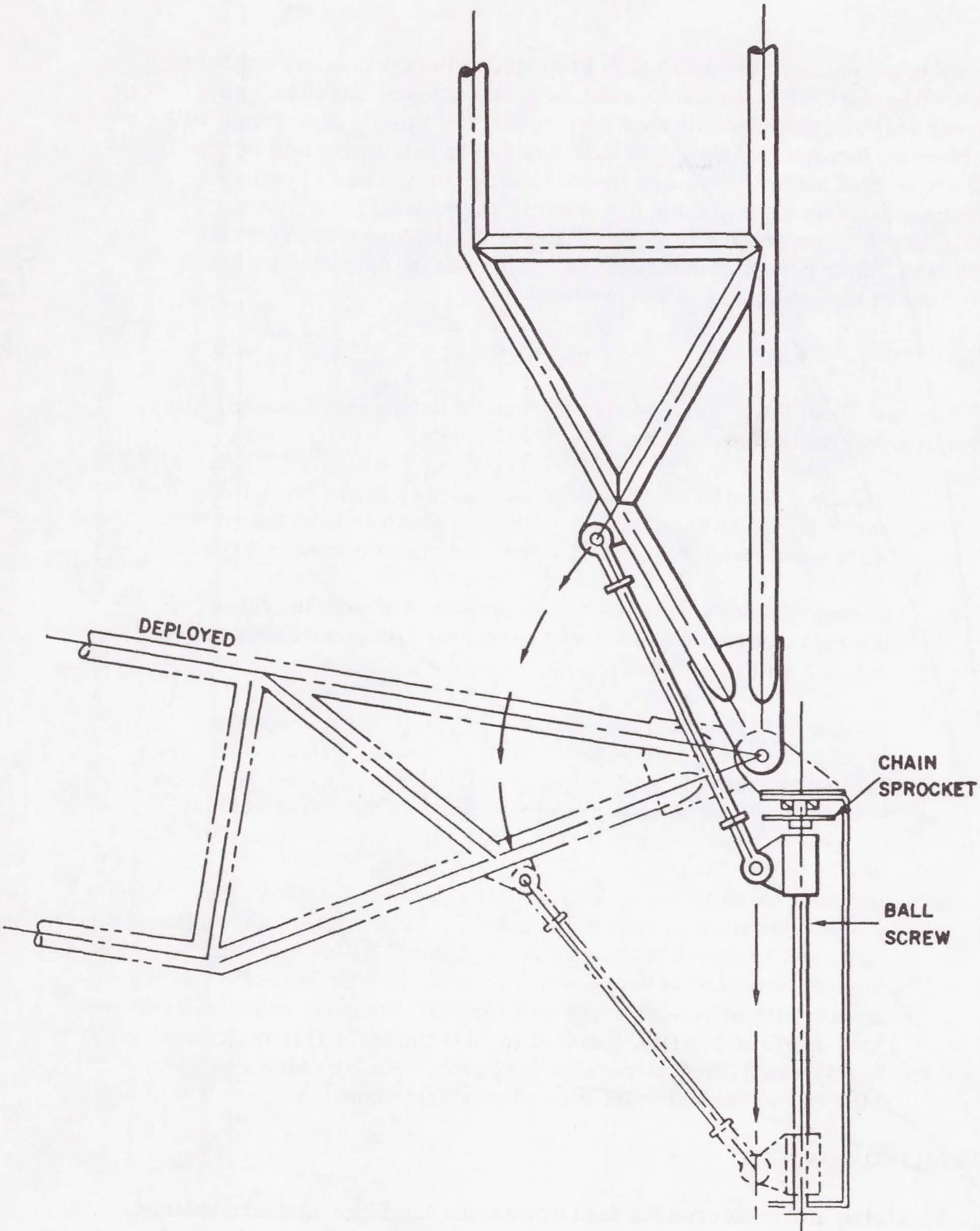

Figure VII-8. Deployment system.

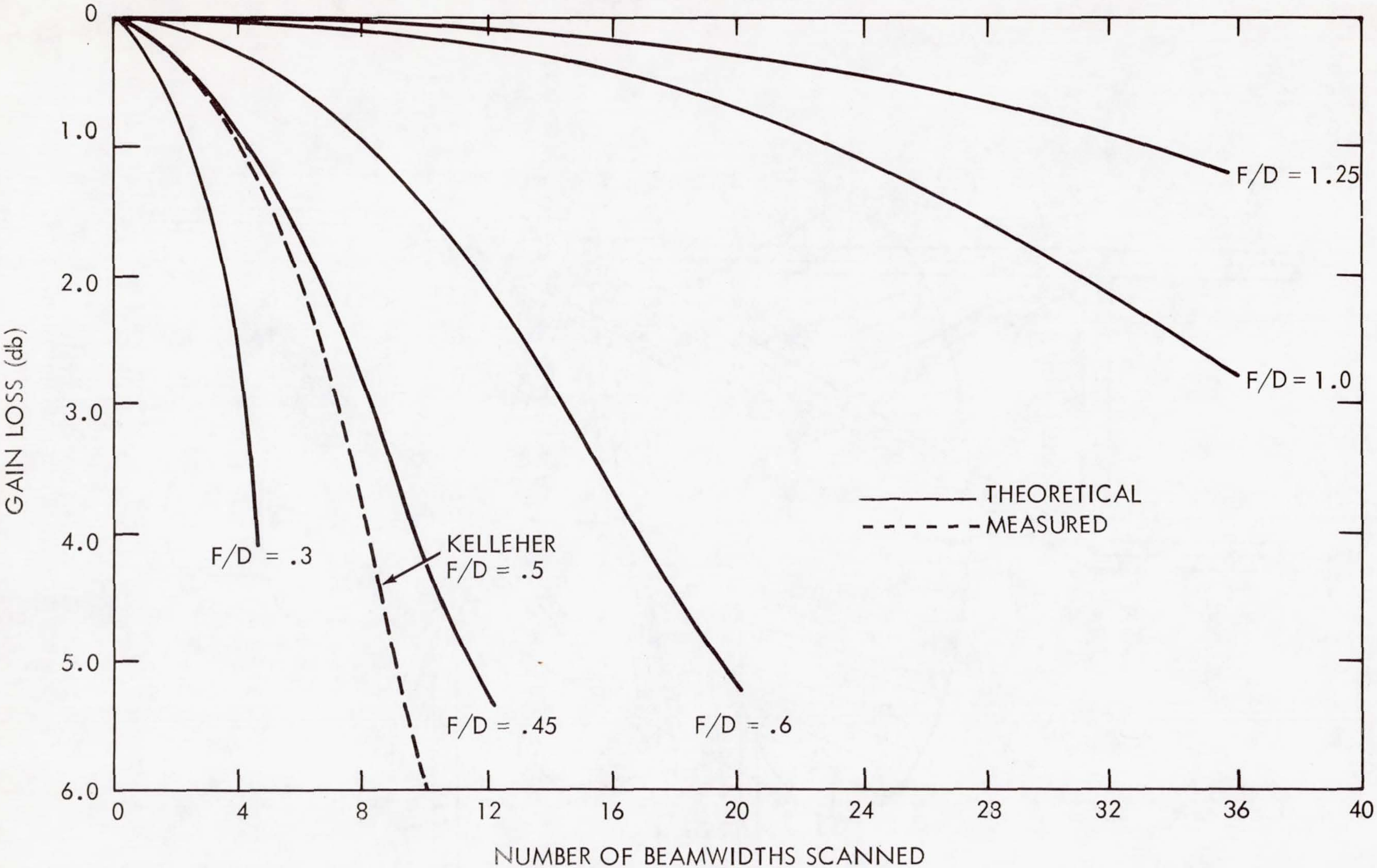

Figure VII-9. Paraboloid gain loss as a function of beamwidths scanned.

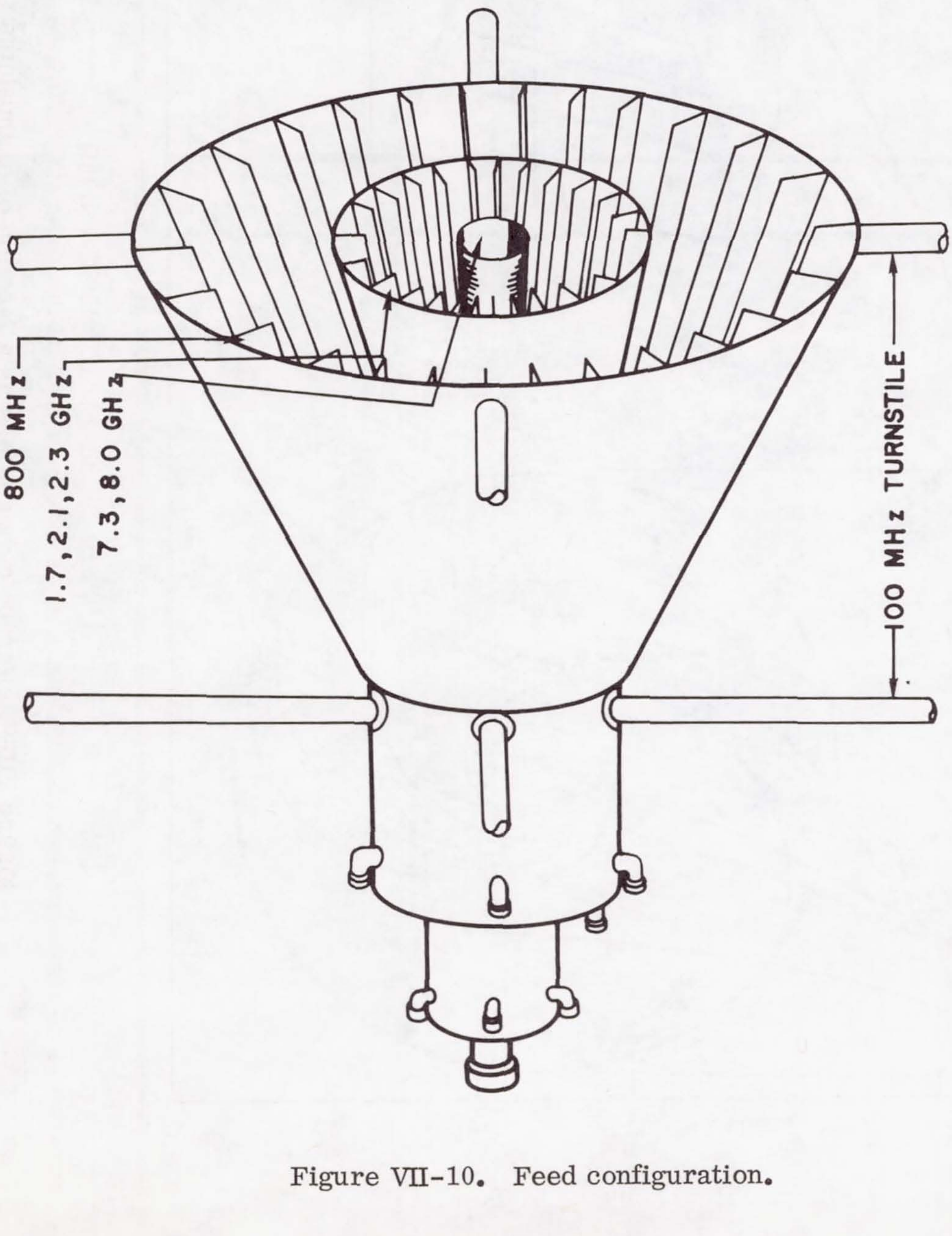

Figure VII-10. Feed configuration.

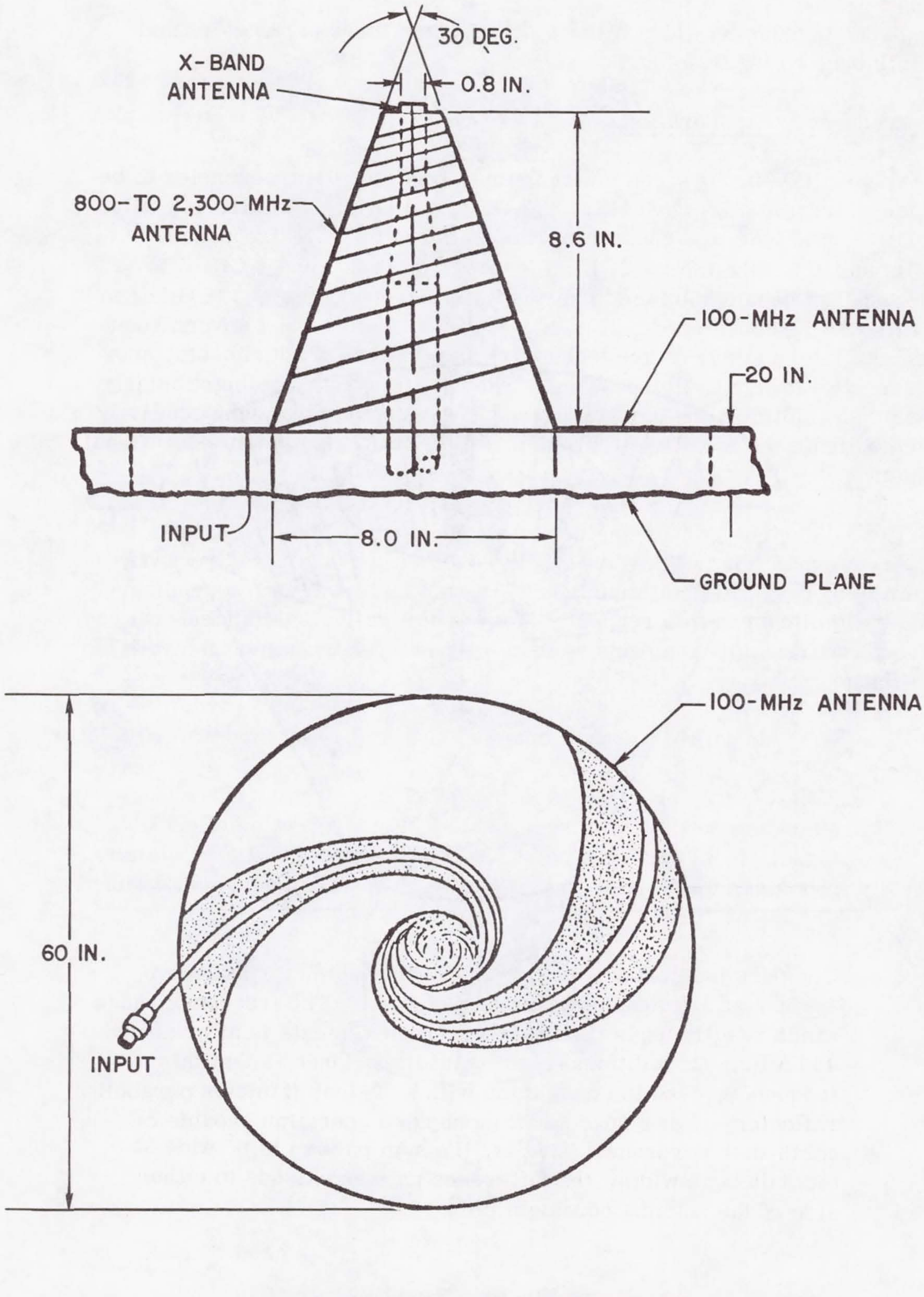

Figure VII-11. 30-foot parabola spiral feed antennas.

ration. The discussion which follows will present these parameters and their influence on the design.

Frequency Considerations

As specified in the three mission study reports, the frequencies to be considered are 100 MHz, 800 MHz, 1.7 GHz, 2.1 GHz, 2.3 GHz, 7.3 GHz, and 8 GHz. The NASA in-house report considered the UHF frequencies 400 MHz and 460 MHz and the X-band frequencies of 7.3 and 8 GHz. In addition, it has been established that the S-band frequencies of 1.8 GHz and 2.25 GHz will be considered. The choice of frequencies most advantageous in terms of information, future technology applications, experiments, and most versatile in terms of usefulness. Also it was considered particularly important to minimize the number of frequencies rather than unnecessarily over-complicate the feed design problem with a resulting system operational degradation.

The development of the engineering technology for deploying large aperture antennas (diameter = 30 feet) in space to provide high directivity and gain is of first order of importance. Hence, microwave frequencies are most significant in this respect. More specifically, 8 GHz (receive) and 7.3 GHz (transmit) are proposed since associated equipment is available in this range.

It was considered important and necessary to choose an additional frequency to:

1. Allow the capability of commanding, upon reception of coded signals from earth, to point the entire spacecraft in the vicinity of a given or desired ground station in terms of gross directional alignment.

2. Compare the performance of the 30-foot diameter reflector surface at a frequency other than X-band. The frequency range which readily lends itself to these requirements is at or about 400 MHz. Of additional significant importance is that this frequency, used in conjunction with a 30-foot diameter parabolic reflector, allows spacecraft monopulse operation capable of earth-disk coverage. That is, the sum pattern is 6° wide at the 3 db beamwidth; the difference pattern extends to either side of the antenna boresight.

The sum-difference pattern produced allows earth-station tracking within a cone of approximately 20^{o} (note: the angle subtended by the earth's disk at synchronous altitude is 17^{o}).

A third frequency range considered important is in the S-band. Specifically, 2.25 GHz for spacecraft transmission and 1.8 GHz for receiving. These frequencies are compatible with existing ground station equipment for telemetry and command functions. In addition, the S-band frequency range will allow the reflector surface to be evaluated (RF performance) at an intermediate frequency between X-band and UHF.

Reflector Focal Length to Diameter Radio (f/d)

In choosing a f/d ratio, a trade-off must be made to best accommodate the overall feed system performance. The effect on system performance and reflector configuration as a function of f/d is summarized in the general comments below.

1. The larger the f/d ratio, the flatter the surface curvature of the reflector.

2. Small f/d ratios allow a more compact configuration because the focal point is closer to the reflector.

3. A small f/d ratio results in a more curved reflector surface and hence greater loss of energy for polarization losses.

4. For beam scanning purposes, the smaller the f/d ratio, the smaller the required off-axis primary feed displacement for a given angular beam scan, but the amount of beam scan is much more limited because the secondary beam deteriorates (gain, beamwidth, secondary lobe level) at a greater rate than for a larger f/d ratio configuration.

With consideration given to all the parameters mentioned above, it has been decided to fix the f/d ratio at 0.44 as the best compromise. Degradation of the secondary beam as a function of off-axis scan angle from f/d = 0.5 is given in Figure VII-12. The edge illumination is -20 db for the prime focus feed in this case.

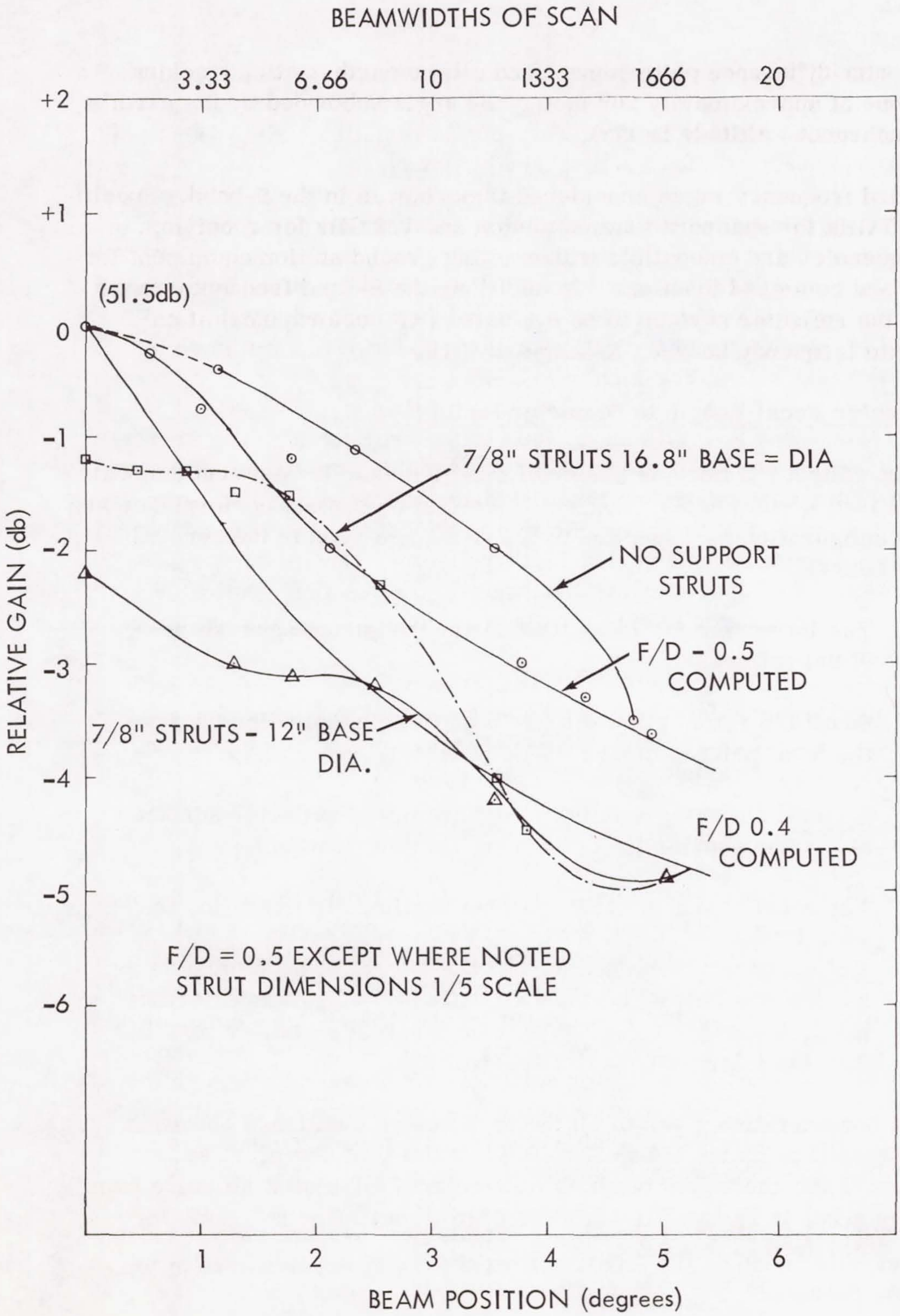

Figure VII-12. Elevation plane pattern.

Cassegrain versus Prime Focus Feed

The decision to design the antenna system around the prime focus feed rather than the cassegrain feed is discussed below.

The cassegrain optics technique allows the overall antenna package to be smaller in the direction of the parabolic reflector axis. In addition to being physically smaller, it also allows the C. G. of the system to be moved closer to the reflector vertex where the controls are located. With the RF feed located in the vicinity of the reflector vertex, there is a minimum of primary power and RF cabling lengths to consider.

The prime focus feed eliminates the requirement for a sub-reflector. This is an important consideration since the allowable surface tolerance for the total reflector antenna system must be shared between 2 reflector surfaces (hyperbolic sub-reflector and parabolic main reflector). Also, an additional alignment problem exists when using a sub-reflector. The prime focus feed presents less of a physical aperture blockage. Finally, a sub-reflector will produce more shadowing on the main reflector, hence, complicating the reflector thermal problem.

After careful consideration it was concluded that the prime focus feed presented less of a problem for the ATS-F&G mission (including scanning the secondary beam) than the cassegrain feed system.

RF Feed Support Structure

The interaction between radiation energy from the RF feed and the feed support structure is an area which required study, measurements and evaluation. Normally, the RF feed supports used in conjunction with a reflector are fixed so that a comparatively large angle is formed between the RF feed area and the intersection of the supports with the reflector. Under these conditions, the high level energy from the RF feed is not obstructed by the feed support when propagating toward the reflector surface. When the energy is reflected from the parabolic dish, it is intercepted by the feed supports resulting in a small area of energy blockage. Both analytical and measured data are available which describe energy losses under these conditions.

Unfortunately, this information for system performance did not exist for a configuration such as the ATS-F&G reflector and RF feed system for the following reasons:

1. The reflector must be folded for launching, allowing only a 6 to 8-foot diameter circumference to attach the feed support into the 30-foot diameter reflector. Since the earth viewing module in the area of the RF feed is approximately 4-feet in diameter, the resulting feed support structure approached a column of cylindrical configuration. Thus, the highest RF energy levels from the primary feed are intercepted by the feed supports in traveling toward the reflector and also on being reflected from the reflector.

2. To allow relatively large scan angles, it is planned to use a higher than normal taper, (-20 db energy level on the periphery of the reflector with the RF feed on-axis.) Therefore, a higher than normal concentration of energy will be directed through the feed support.

3. The X-band RF feed the ATS-F&G system is planned to be a scanning feed and therefore represents a variable interaction problem.

4. Three widely spaced frequencies (X-band, S-band, and UHF) are being considered, thus preventing the possibility of designing the feed support to accommodate one frequency, i.e. tuning.

5. Monopulse performance degradation must be considered, particularly with respect to the difference mode slope and antenna boresight.

Because of the small feed support angle, multiple frequency operation, monopulse operation, higher primary feed energy tapers, and the scanning feed, measurements were made to determine the extent of performance degradation. Configurations investigated and gain performances are shown in Table VII-2.

Systems Operation

In the following paragraphs the system operations and performances for the three frequency bands (X-band, S-band, UHF) will be described. It should be noted that the gain and beamwidth relations for the ATS-F&G antenna system as indicated in Figure VII-13 are less than optimum for the 30-foot reflector capability. These figures are, nevertheless, realistic and quite good, particularly at X-band when consideration is given to the broad, multiple frequency operation, non-optimized prime feed energy taper, adverse RF feed support structure and anticipated reflector surface deviations from solar energy.

Table VII-2

Feed Support Structure - RF Gain Interaction						
Strut Configuration	No. of Members	Member dim. (in)	Base dia (in)	Top dia (in)	Δ Go (db)	ΔG_{12} (db)
A Frame	8	4.37	60	42	2.2	2.2
A Frame	8	4.37	84	42	1.2	2.0
A Frame	8	6.25	60	42	3.4	3.4
A Frame	8	6.25	84	42	2.0	2.5
A Frame	8	5.63	60	42	2.8	2.8
K Frame	3	3.75	76	50	0.3	0.25
	6	1.87				
K Frame	3	6.25	80	51.25	1.1	1.2
	6	3.14				

NOTES: 1. All base and top diameters are for circles passing through the center of the struts

2. ΔGo = on axis no strut gain minus on axis strut gain

3. $\Delta G_{12} = 12^{th}$ beam width no strut gain minus 12^{th} beamwidth strut gain

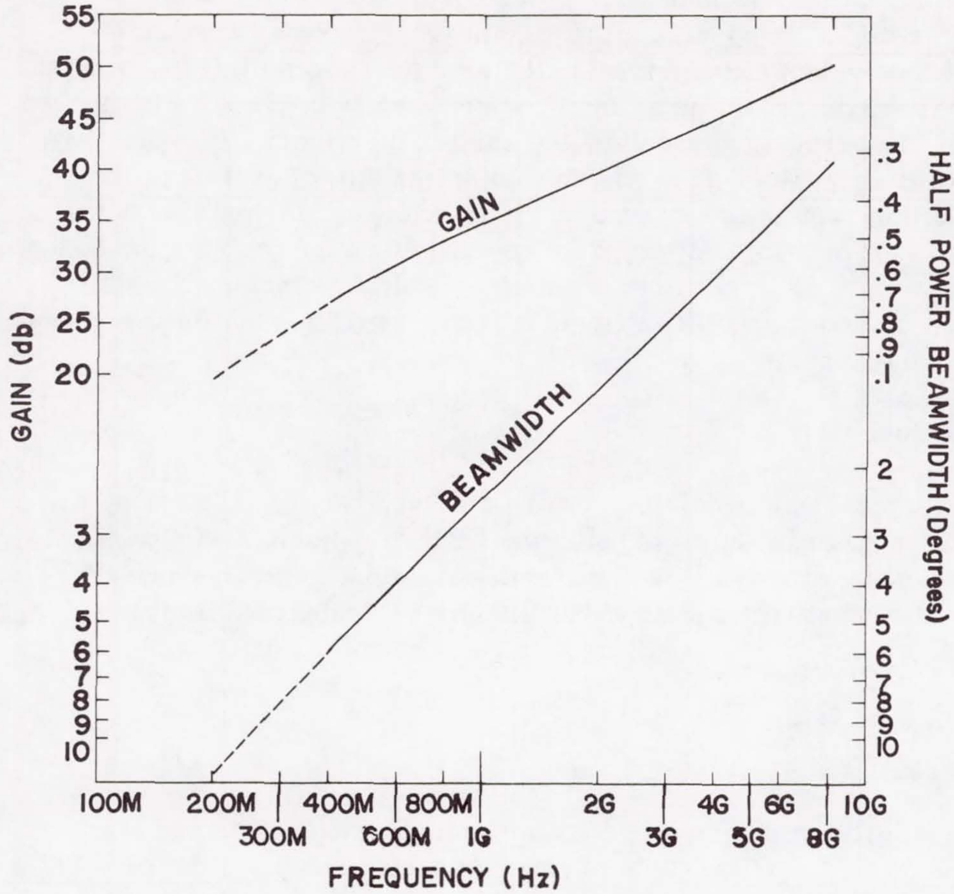

Figure VII-13. Frequency, gain, and beamwidth relation for ATS-F&G 30-foot diameter antenna.

MONOPULSE

Two of the frequency bands (i.e., X-band and UHF) will operate in a monopulse mode. Therefore, it is worthwhile to express the monopulse pointing error capability.

<u>UHF Monopulse</u>

The UHF monopulse system will provide a pointing error signal as a result of a transmission from a ground station to the spacecraft. The pointing error signal can be used to command the spacecraft to a zero error condition, or it can be telemetered to the ground. For the zero error condition, the spacecraft axis will be aligned to pass through the ground station. Of prime importance in this system is that UHF (more specifically in the 400 MHz region) monopulse can reliably provide an error signal from a ground station $\pm$ 9.5 degrees off of the 30-foot diameter reflector axis with a resulting pointing accuracy of 0.1 degree. This is an included angle of 19 degrees as compared to the angle of 17 degrees subtended by the earth's disk from the spacecraft a synchronous altitude. Thus, it is possible to orient the spacecraft to the correction position even in the event that the initial programmed spacecraft orientation is in error. In addition, this system allows a reorientation in the event that a new pointing is required within the 19-degree included angle. A switch is incorporated in the UHF system to allow a communications transponder to be placed in the system.

X-Band Monopulse

Once the spacecraft is pointing toward the specific ground station, an accuracy of 0.1 degree is achieved using the UHF monopulse system. Then the 8 GHz monopulse system is switched into operation. This X-band monopulse is effective in providing error signals $\pm$ 1/2 degree off of the antenna boresight axis with a resulting pointing accuracy of approximately 0.01 degree.

S-Band System

The transmit S-band frequency is approximately 2.25 GHz and the receive frequency is 1.8 GHz. It is planned that the S-band system be capable of scanning the feed of the parabolic axis. The desirable scan angle off-axis by definition is $\pm$ 9.5 degrees which approximates pointing the axis of the beams (1.0 degree 8 db beamwidth) 400 miles to either side of the earth's disk from synchronous altitude.

In order to scan a beam off axis to 9 1/2 degrees requires the center of phase of a feed horn to be displaced from the focal point (in the focal plane) in the order of 5 1/2 feet. Preliminary calculations and tentative data indicate that a usable radiation pattern scan range of $\pm$ 9.5 degrees is feasible. This area will be investigated in detail.

It is interesting to note that investigation by measurements has shown that the secondary beam of a parabolic reflector can be scanned at much greater angles off axis than literature previously available indicates. Figure VII-12 contains realistic curves which have been measured for a 213 diameter reflector. This should be compared with Figure VII-9 which shows what industry proposals specify as realistic scanning range capability.

The versatility of S-band scan can be increased by using a mechanical scan in a direction normal to the electronic scan thus forming a hybrid scan system. Another hybrid scan which will be investigated will be one in which the electronic scan array is rotated about its center allowing communication between stations at angles other than on a given spacecraft roll and pitch axis.

In summary, the electron scan would allow transmitting and receiving on the same beam or any two beams within the total electronic scan.

C. THERMAL APPROACH

Thermal design of the large antenna is identified as a major problem area for ATS-F&G. The antenna thermal problem was discussed in Section VII B, Antennas. This section of the report concerns itself with a discussion of the thermal problem associated with the spacecraft earth viewing and aft-equipment modules.

From the study conducted at Goddard, it appears that a completely passive thermal design of spacecraft, i.e., earth viewing module and aft-equipment module, would be marginal. A semipassive system using louvers seems to be the best solution to the thermal problems associated with the spacecraft. General Electric recommends such a semipassive system but in addition they state that heaters are required. Because of their power requirement, heaters should be avoided. Fairchild's system, which is completely passive, contains only an earth viewing module. Their thermal analysis concerned itself with computing the radiating area necessary to maintain the spacecraft within tolerable limits. Although this analysis indicated that passive control is possible, a more detailed analysis would probably show the need for some active control of components with high power dissipation or with erratic duty cycles. Lockheed's recommended system is completely passive, but lack of any detail makes it impossible to discuss Lockheed's thermal design. It would suffice to repeat that a completely passive thermal design appears to be marginal at best.

A louver system has the ability to control temperatures over a much greater variation of power dissipation than a passive system since the effective emittance can be varied by a factor of four or more. It is interesting to note the "squaring off" or flattening of the north and south faces of the earth viewing module and the aft-equipment module in the G.E. design. Since a louver system is at its best when it does not face the sun, the "squaring off" enables the north and south faces to be used to best advantage as the primary surfaces for heat rejection. The louver system, described as the preferred approach, is very similar to that successfully used on Nimbus where accurate temperature control has been achieved in orbit.

D. COMMUNICATIONS SUBSYSTEMS

INTRODUCTION

A relatively large number of potential ATS-F&G communications operations at some ten different frequencies have been discussed in Appendix E of this Analytic Report, Section 20-A of Reference 1, and in the three contractor reports (References 2, 3, and 4). Table VII-3 summarizes the frequencies considered, and notes briefly the ATS-F&G communications operation(s) involved.

The question of suitable communications subsystem configurations for accomplishing the various operations described in Table VII-3 has been treated in considerable depth in each of the four references. These documents contain a wealth of sound technical material which may be drawn upon for actual spacecraft design during Phases B, C, and D of the ATS-F&G program. In no case was communications subsystem design and/or component availability considered a significant pacing item in establishing overall mission feasibility.

While it is clear that no single ATS-F&G type spacecraft could be designed to accomplish all the functions listed in Table VII-3, it is also the case that most of the operations at the various frequencies could be accomplished by assigning some to the ATS-F spacecraft and others to ATS-G. The philosophy of planning different communications subsystems for the two spacecraft has been adopted in this report. Certain of the frequencies in Table VII-3 have been assigned to ATS-F and others reserved for ATS-G. This approach is reflected in the following discussions.

ATS-F COMMUNICATIONS SUBSYSTEMS

Independent UHF, X-Band, and S-Band communications subsystems are assigned to ATS-F. Table VII-4 summarizes the operating frequencies, RF bandwidths, and CW power ratings for the three subsystems.

In each case, the subsystem would be a frequency translating repeater of the same basic type that has been used in previous satellites for space communications. The UHF and X-Band subsystem designs presented below are identical (with the exception of bandwidth in the UHF case) to systems discussed in greater detail in Reference 1.

Table VII-3

Summary of Proposed ATS-F&G Frequency Assignments

Frequency	Function	ATS-F&G Operation	Where Considered
100 MHz	Transmit	FM direct broadcast	References 2, 3 & 4
401 MHz	Receive	Data collection, multiple-access/communications, etc.	Reference 1 & Appendix E
466 MHz	Transmit	Data collection, multiple-access communications, etc.	Reference 1 & Appendix E
800 MHz	Transmit	TV relay and FM broadcast	References 2, 3, 4 & Appendix E
860 MHz	Transmit	TV relay	Appendix E
1.7 GHz	Receive & transmit	Data relay, aircraft communications, etc.	All
2.1 GHz	Receive	Data relay and telemetry	References 2, 3 & 4
2.3 GHz	Receive & transmit	Data relay and telemetry	All
7.3 GHz	Transmit	TV relay, multiple-access communications, etc.	All
8.0 GHz	Receive	Up-link relay, etc.	All

Table VII-4

Summary of ATS-F Communications Subsystems				
Communications Sybsystem	Receive Frequency	Transmit Frequency	Overall RF Bandwidth	CW Power Output
UHF	401 MHz	466 MHz	1 MHz	40 watts
X-Band	8.0 GHz	7.3 GHz	40 MHz	24 watts
S-Band*	1.8 GHz	2.25 GHz	10 MHz	10 watts

* Note that the ATS-F S-Band frequencies indicated here differ slightly from those listed in Table VII-3

UHF Subsystem

A functional block diagram of the proposed ATS-F UHF communications subsystem is given in Figure VII-14, and important characteristics are summarized in Table VII-5. The entire subsystem is redundant with the exception of a common preamplifier employed to avoid the 3 db penalty in receiving system sensitivity which would be incurred if a hybrid divider were used immediately after the diplexer. After up-conversion from IF to 466 MHz, a hybrid combining network prior to power amplification allows both transponders drive either output stage, ensuring maximum flexibility and dependability.

Both linear and hard-limiting modes of transponder operation are included to accommodate the large number and variety of modulated signal types which will be encountered in proposed applications and demonstrations of the ATS-F spacecraft. For example, signals such as single-sideband modulated frequency-division-multiplexed voice channels and amplitude-modulated video signals require a linear transponder, whereas wideband frequency modulated or spread-spectrum signals would be handled most efficiently by a hard-limiting transponder. The choice of transponder, mode, and output stage would be controlled through the spacecraft command subsystem.

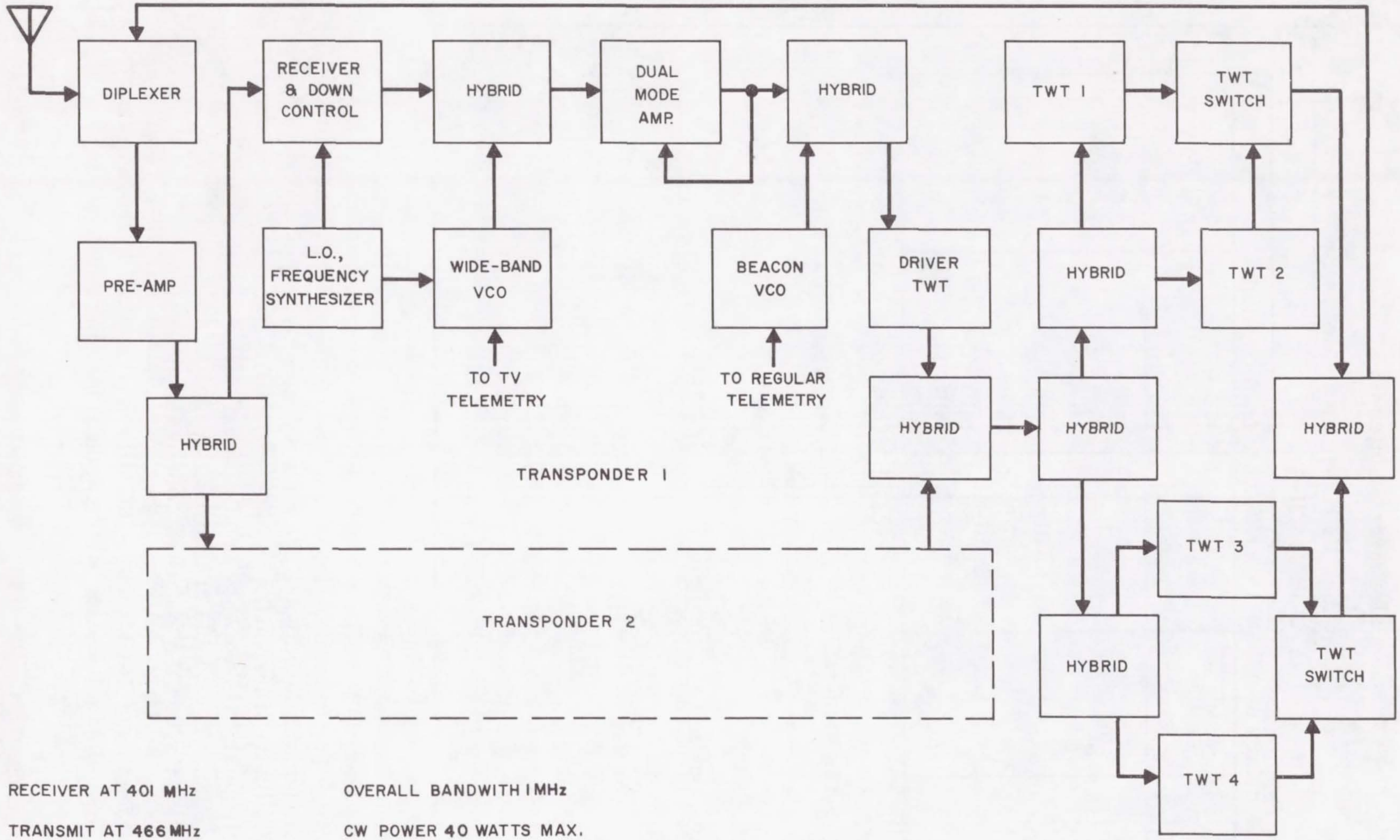

Figure VII-14. ATS-F UHF communications subsystems.

Table VII-5

Essential Characteristics of ATS-F UHF Communications Subsystem	
Receive center frequency	401.0 MHz
Transmit center frequency	466.0 MHz
3 db bandwidth (overall)	1.0 MHz
Receiver noise figure	3 db (Max)
IF frequency (tentative)	60 MHz
Modes of operation	(a) Linear (reduced power) (b) Hard-limiting
RF output stage	Redundant solid state
RF power output	40 watts (saturated)
Estimated weight	20 lb
Estimated prime power requirements	72 watts
Estimated command requirements	12 latching type relay commands
Estimated telemetry requirements	(a) 6 analog channels; 2 at 20 samples/sec min; 4 at 1 sample/sec max (b) 12 1-bit digital channels; all at 1 sample/sec max

The proposed UHF communications subsystem is a rather straightforward extrapolation of state-of-the-art communications satellite technology to meet the envisioned requirements of the ATS-F mission. The basic frequency-translation transponder design is, with the exception of bandwidth and dual-mode capability, very similar in concept to the VHF transponder built for the ATS-B spacecraft under NASA-5-9593. Also, the ATS-F UHF receive and transmit frequencies of 401 MHz and 466 MHz are identical to those of the IRLS Program, for which solid-stage RF equipment is being developed by Motorola under NAS-5-10195.

X-Band Subsystem

A functional block diagram of the proposed ATS-F X-Band communications subsystem is given in Figure VII-15. Important characteristics are summarized in Table VII-6. As with the UHF subsystem described above, the X-Band subsystem is redundant, with the exception of a common high performance front end. A further similarity is the capability of dual-mode operation. Important features peculiar to the X-Band communications subsystems are:

Provision for transmission to earth of frequency-modulated video data via a wideband VCO operable through the transponder.

Provision for transmission of conventional narrowband spacecraft telemetry by angle-modulating a low-power beacon working into the transponder power stage.

A highly versatile TWT power amplification arrangement, in which either transponder can drive any one, any combination, or all of the four TWT's in parallel to develop output RF levels into the large aperture antenna of 6, 12, 18 or 24 watts. This arrangement makes for a highly flexible and reliable output stage, and guarantees graceful degradation in the event of one or more TWT failures.

The subsystem proposed here is of essentially the same design as that used in Comsat's "Bluebird" HS303A spacecraft. Apart from operating frequencies and bandwidth, the only critical differences are the dual-mode capability and inclusion of the wideband VCO for transmission of local video.

S-Band Subsystem

Subsystem design for the ATS-F S-Band communications frequencies listed in Table VII-4 is a somewhat more complex matter than it is at UHF or X-Band because of requirements to interface, both operationally and physically, with the ATS-F unified S-Band TT&C (telemetry, tracking, and command) subsystem (See Section 18 of Reference 1 or Sections VII H&J of this Report). Generally, what might otherwise be normal hardware interface problems are greatly complicated by considerations for using the spacecraft (with its 30' reflector) in satellite-to-satellite data relay links with, for example, the Apollo capsule during the latter's launch phase.

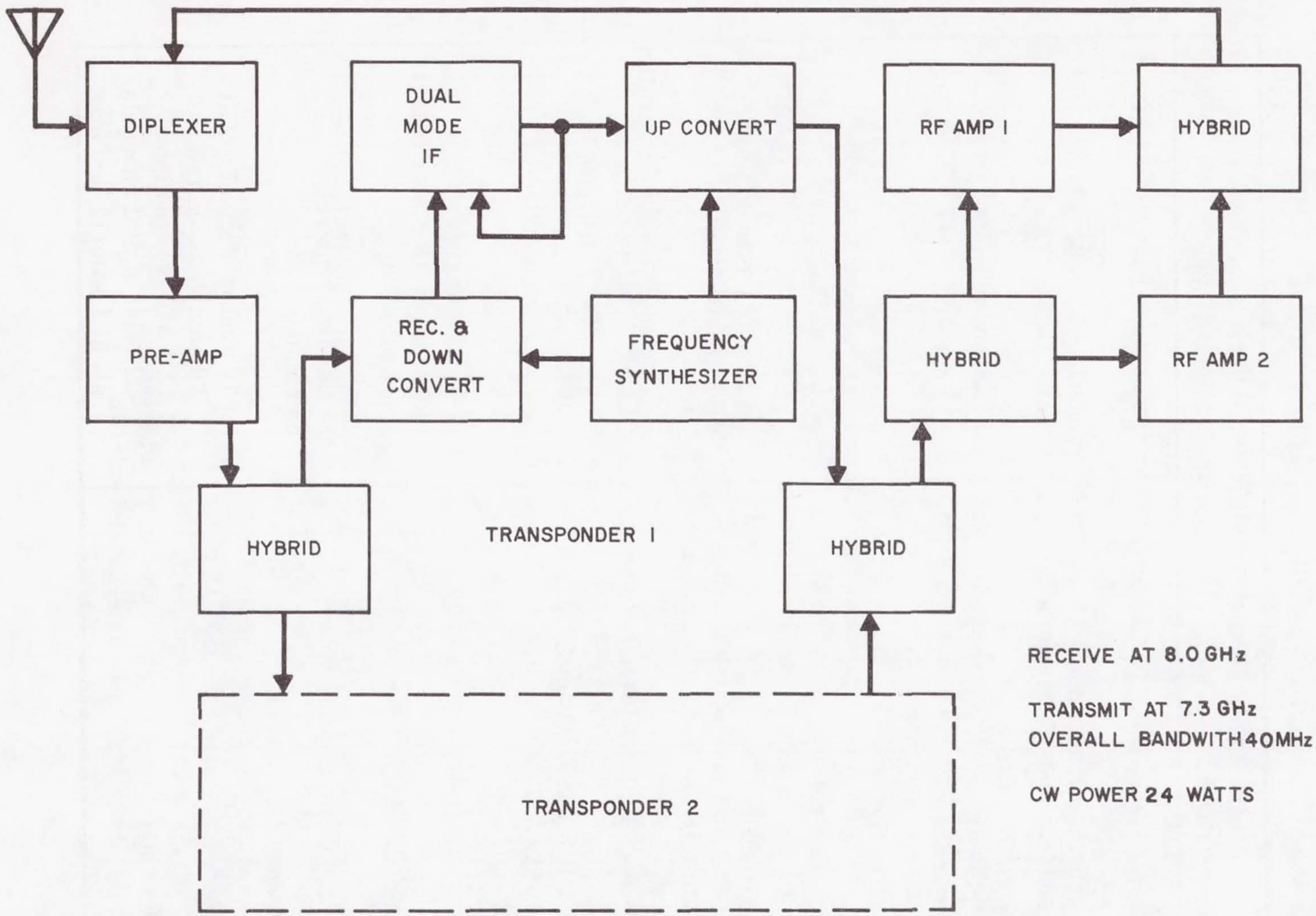

Figure VII-15. ATS-F X-band communications subsystem.

Table VII-6

Essential Characteristics of ATS-F X-Band Communications Subsystem	
Receive center frequency	8.0 GHz
Transmit center frequency	7.3 GHz
3 db bandwidth (overall)	40 MHz
Beacon frequency	7.2 GHz
Receiver noise figure	7 db (Max)
Type repeater	Single conversion
Modes of operation	(a) Linear (reduced power) (b) Hard-limiting
RF output type	4 switchable TWT's in parallel
RF output power	6, 12, 18, or 24 watts
Estimated weight	40 lb
Estimated prime power requirements	80 watts (Max) (depends on RF power output level selected)
Estimated command requirements	22 latching type relay commands
Estimated telemetry requirements	(a) 12 analog channels; 2 at 20 samples/sec min; 10 at 1 sample/sec max (b) 22 1-bit digital channels; all at 1 sample/sec max

Because of these considerations, subsystem design for ATS-F S-Band communications is at present in a preliminary stage and no block diagrams, estimated weight and power requirements, etc. are given in this report. It is not anticipated that the eventual design will differ radically from the types described above for ATS-F UHF and X-Band communications, i.e., a wideband frequency translating repeater with dual-mode capability and redundant components for maximum flexibility and dependability. Also, it is fairly certain that the power stage will be one or more TWT's rather than solid-state. The present state of the art in S-band solid-state transmitters (NASA Contract NAS5-10258) does not suggest the availability of efficient 10 watt devices within the ATS-F time frame, whereas efficient S-Band TWT systems have been used with general success since the early days of space communications.

Beyond these general remarks, it is not possible to define the ATS-F S-band communications subsystem further in this report.

ATS-G COMMUNICATIONS SUBSYSTEMS

ATS-G frequency assignments will in part duplicate those of ATS-F, and to this extent it is probable that the same communications subsystems (UHF, X-band, and/or S-band) will be used in both spacecraft. It is also planned, however, that ATS-G will have a UHF (800 or 860 MHz) transmit capability and possibly (although less likely) a VHF (100 MHz) transmit capability. Respectively, these two capabilities would be for FM-TV relay and FM direct broadcast experiments of the type discussed in Appendix E of this report and in References 2, 3, and 4.

VHF Subsystem

No problem areas are seen in furnishing the ATS-G spacecraft with an efficient solid-state 100 MHz transmit capability. References 2, 3, and 4 all proposed essentially equivalent solid-state power output stages, with CW levels of the order of 10 to 20 watts. Although ATS-F&G applications of VHF have been given no emphasis in either Reference 1 or this report, it is noted briefly in passing that any such operations (in particular, FM broadcasting in the commercial band to home receivers) would probably use X-band (8.0 GHz) for the ground-to-satellite up-link.

800/860 MHz UHF Subsystems

Considerable emphasis has been placed on using UHF frequencies in the upper end of the TV band for ATS-F&G FM broadcasting and FM-TV relay experiments, both in this report (Appendix E) and in References 2, 3, and 4. There do not appear to be any serious problems in designing spacecraft subsystems for these applications; and as in the VHF case, it is likely that X-band frequencies would be used for the ground-to-satellite up-link.

One question which at this writing must still be resolved is that of solid-state vs. tubes for the RF power stage. For CW levels of the order of ten watts, it seems entirely reasonable to assume the availability of efficient solid-state devices within the ATS-G time frame. For power levels greater than this (power levels near 100 watts would be required for quality FM-TV relay to truly small central receivers), it would appear to be a questionable extrapolation of current technology to propose an all solid-state ATS-G communications subsystem at these frequencies.

CONCLUSIONS

In summary, it is concluded that communications subsystem design and component availability are not critical areas in overall ATS-F&G mission feasibility. Several problems have not been resolved at this writing, particularly in the area of ATS-F S-band communications, and many details of subsystem design remain to be clarified as the ATS-F&G mission evolves. In any event, it is not foreseen that communications subsystem design or component availability will become pacing items in the program.

REFERENCES

1. "ATS-4 GSFC Concept Design Study," Document X-730-67-10, GSFC, January 1967.

2. "ATS-4 Study Program Final Report," Contract NASW-1411, Fairchild-Hiller Space Systems Division, December 1966.

3. "Final Study Report," Contract NASW-1410, General Electric Spacecraft Department, November 1966.

4. "Advance Study of an Applications Technology Satellite (ATS-4) Mission, Final Report," Contract NASW-1412, Lockheed Missiles and Space Company, November 1966

E. <u>CONTROLS</u>

INTRODUCTION

This section describes the spin-control system used during the transfer orbit and the three-axis stabilization system used during the synchronous orbit. The performance requirements are stated, followed by the complete control systems sequence of operation, beginning with separation from the Centaur launch vehicle to final operational stabilization in orbit.

An alternate technique for transfer orbit control is a three-axis controlled trajectory. A detailed comparison of spin stabilization versus three-axis control during the transfer orbit, apogee motor burn, and vernier orbit corrections is given in Appendix B. A more detailed description of the calculations and the limiting performance criteria can be found in <u>ATS-4-GSFC Concept Design Study</u>, pages 13-1 through 13-49.

The control systems' sequence of operation covers three distinct phases:

1. Spinning body control
2. Acquisition control
3. Operational control.

For operational control, it is necessary to stabilize the spacecraft in all three axes to 0.1 degree and to slew the entire spacecraft to any desired point within the earth disk at slew rates of up to 1 degree per minute. It is also necessary to maintain stabilization to these axes during station-keeping maneuvers.

The difficulties in detailed design are readily recognized when the dynamics of the spacecraft structure are taken into account. Although no doubt exists as to the feasibility of designing such a control system in the first design approach, considerable design ingenuity will be required to achieve spacecraft dynamics compatible with control operation.

The recommended control system is based on the concepts recommended in the GSFC, General Electric Company, and Fairchild-Hiller studies. The Lockheed concept is rejected on a basis of weight, power, and reliability considerations.

The preferred control concepts and the control system design guidelines are given below, followed by a trade-off discussion. The key control system parameters are presented in outline form in the ATS-4 GSFC Concept Design Study, page 13-45.

Preferred Control Concepts

1. Synchronized axial jet pulses, 12-ft-lb, 0.2-second pulses: used to control the spinning body

2. Pulse-modulated jets: used in the acquisition control to provide the torque for sun acquisition and operation control

3. Wheel-jet hybrid: used with the digital operational control system to provide momentum control for the operational mission requirements.

Control System Design Guidelines

1. Reaction jet thrusters: use one set of thrusters for both attitude control and station-keeping; use the same thrust level for sun acquisition, local vertical acquisition, and backup operational control.

2. Three-axis, time-shared, digital controller: used to obtain high-accuracy processing of command and error signals and to generate low-frequency compensating networks.

3. Control system bandwidth: minimize to prevent control system/structure resonance.

SPINNING BODY CONTROL

The General Electric, Fairchild-Hiller, and Goddard studies conclude that spin stabilization during transfer orbit is the preferred approach. Appendix B gives the trade-off and justification for this approach.

Spin stabilization will be used in the transfer orbit during apogee burn and vernier orbit corrections. The attitude control functions during these phases of the mission will involve active nutation damping (since the moment of inertia ratio is less than one), spin-axis precession, and spin-axis orientation measurement. Spin-axis precession will use the synchronized axial jet techniques pioneered on Syncom and used on ATS-A through E. Active nutation damping will be accomplished using a nutation sensing accelerometer and synchronized axial jet pulses as on ATS-D&E. The spin-axis orientation will be measured using the sun sensor/RF polarization techniques as on ATS-A through E. A star field mapper (SCADS) will be used as a secondary spin-axis orientation sensor. SCADS promises to provide the spin-axis orientation to ±0.1 degree. If sufficient flight experience is obtained with SCADS before the ATS-F&G launch, SCADS will be used as the primary sensor.

Since the spacecraft will be spun up after separation from the booster, accurate measurement of the spin-axis orientation in the transfer orbit is essential to the ATS-F&G mission. The booster auto-pilot will be programmed to provide the proper attitude for apogee burn at the time of separation; however, because of the weight and size of the payload, an error in the orientation of the momentum vector after a spinup of 12 degrees would be introduced by tipoff rates of 3 degrees/second (allowing a 2-second interval between separation and firing of the spin rockets).

ACQUISITION CONTROL

The two contractors and the Goddard study were in agreement on the acquisition sequence. This sequence is depicted in Figure VII-16. One exception to complete agreement is whether the sun should be acquired with the roll or yaw axis. General Electric favored the latter while Fairchild-Hiller and GSFC favored roll axis sun acquisition. The preferred approach is roll axis sun acquisition. The trade-offs used in reaching this decision are now presented.

If the roll axis is pointed at the sun, as presently used on OGO and recommended by the GSFC Concept Design Study and Fairchild-Hiller, the acquisition maneuver occurs near 6 a.m. or 6 p.m. local time, and involves using a spacecraft roll rate of 0.2 degree per second to search for the earth. During this maneuver, the earth sensor is performing its normal scan pattern and needs no special scan pattern for each search. The removal of the 0.2 degree per second rate during 8 degrees of rotation requires an acceleration of 10 degrees per minute^2 and a torque of 0.145 ft-lb (for a roll moment of inertia of 3000 slug-ft^2).

If the negative yaw axis is pointed at the sun, as recommended by General Electric, the acquisition maneuver occurs near local noon. This involves using an earth sensor search mode to find the earth, since the yaw axis can be as much as 23.5 degrees out of the orbit plane, and the yaw orientation is unknown. During this maneuver, however, no removal of spacecraft momentum is required; and the acquisition of the local vertical, after the sensor search mode has found the earth, could be accomplished by using only momentum wheels.

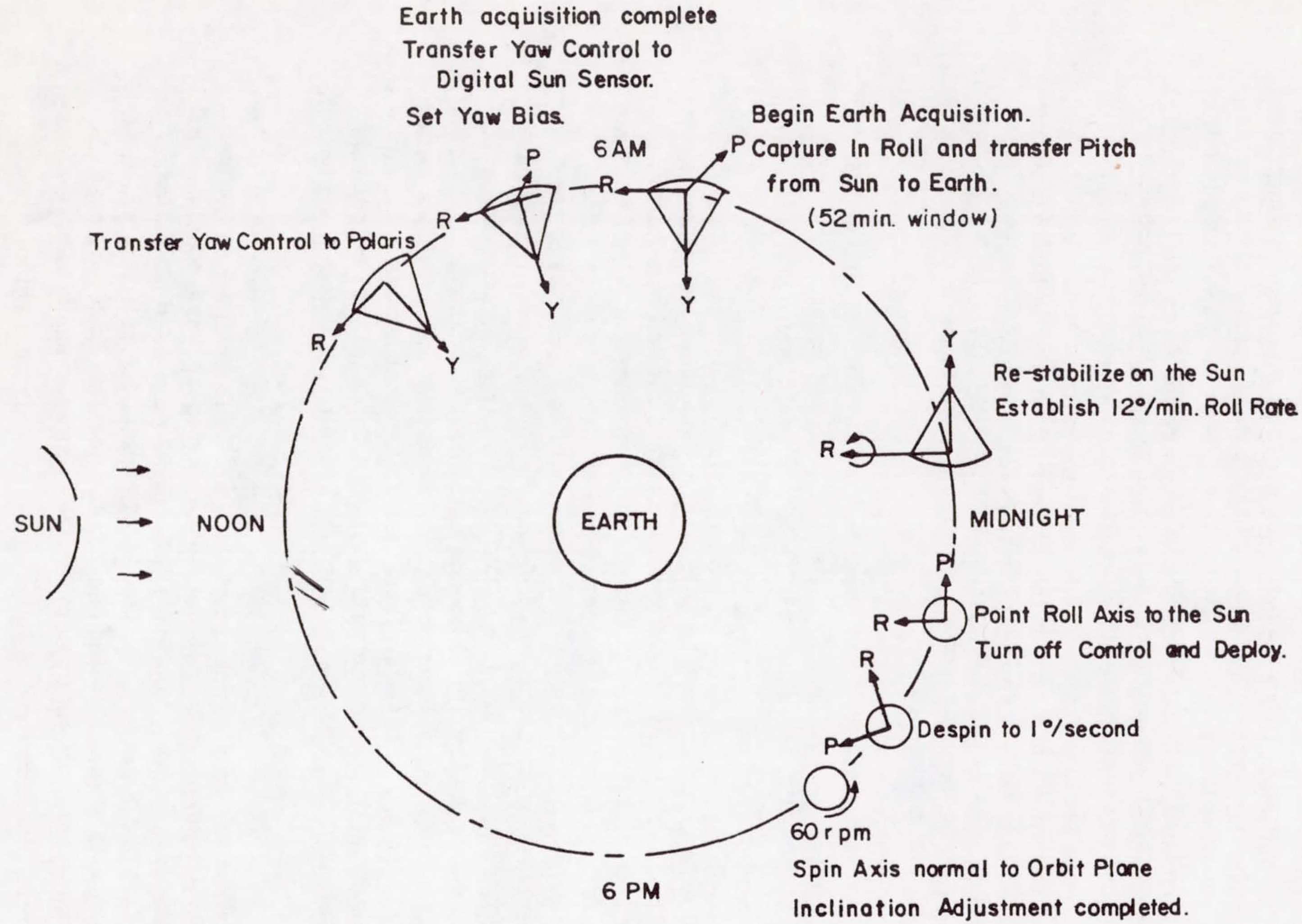

Figure VII-16. ATS-F&G attitude control acquisition sequence.

In addition to torque and sensor considerations, there is also yaw control at the end of the earth acquisition maneuver. With the roll axis pointed to the sun, yaw angle control is left on the sun during earth acquisition. Immediately following earth acquisition, the sun sensor can be biased electronically to place the roll axis in the orbit plane to acquire Polaris. With the yaw axis pointed at the sun, there is only yaw rate control during earth acquisition; and using an additional sun sensor, yaw angle control must be acquired from an arbitrary initial value before the Polaris acquisition maneuver can be initiated.

Although pointing the roll axis at the sun provides for simpler earth and Polaris acquisition maneuvers, it requires a larger roll axis maximum torque than the yaw axis orientation. The roll axis orientation enables the onboard monopulse system to be used for earth acquisition, thus there is a sensor backup for the earth acquisition mode.

The acquisition control torquers proposed by Fairchild-Hiller and GSFC use an all jet system. General Electric elected to utilize reaction wheels with jet unloading to accomplish this function. The preferred approach is to provide the all jet mode. This approach is now justified.

The GSFC study recommended a pulse-ratio modulated jet control system for the sun/earth/Polaris acquisition sequence, with a limited operational capability in the earth-pointing mode. The error signal processing would be completely isolated from the operational control and would use simple analog techniques. If necessary, the error dead zone could be increased to ± 0.2 degree if sensor noise or structural flexing makes ± 0.1 degree impractical because of excessive fuel consumption. The fuel required for acquisition and one-year backup operation is only 10 pounds using heated ammonia ($I_{sp} = 175$). The all-jet control using simple analog signal processing will provide a basic, minimum power (with jets unheated) control capability that bypasses the momentum wheels and their power converter, as well as the digital operational controller; therefore it reduces the probability of losing all three-axis control.

Lockheed proposed an ammonia jet system for coarse control and an ion jet control for the operational mode. Fairchild-Hiller proposed a hydrazine jet system for acquisition and momentum wheels for operational control, with an estimate of the fuel consumption if the jet control were used as an operational backup. General Electric proposed a jet control for sun acquisition, using wheels for earth acquisition and operational control; however, they did not consider the use of an all-jet system for the complete acquisition sequence and backup operational control.

After reviewing the various approaches proposed, it was decided that the all-jet capability yielded the highest reliability because of its simplicity and redundancy. Thus, this approach was selected.

OPERATIONAL CONTROL

The GSFC concept of using a rather complex digital controller with momentum wheel torquers as the primary operational control system and an independent, simpler all-jet control system as a backup has been chosen to ensure maximum useful spacecraft life and experimental flexibility.

Of the three study contractors, Fairchild-Hiller came closest to this concept. They proposed use of a digital controller and calculated the fuel requirements for an all-jet backup capability, but they did not carry their analysis far enough to indicate whether they actually recommend this mode of operation.

General Electric proposed an analog controller using part of the command system to generate the Polaris sensor bias signal. They did not consider the generation of other on-board low frequency signals, sensor bias trim, integral plus proportional control, or change of control system modes to an all-jet backup capability. They have no provision for closed loop onboard utilization of the interferometer error signals.

Lockheed proposed a hybrid controller using some digital techniques but still using operational amplifiers as summing points. It should also be noted that although Lockheed proposed the interferometer as the primary ± 0.1 degree reference system for generating roll, pitch, and yaw error signals; they were very brief in covering the type of controller would be used to develop these signals from the basic interferometer outputs.

The block diagram of the preferred control system showing sensors, controllers, and torquers is given in Figure VII-17. The flexibility and accuracy required in the operational mode is summarized below:

1. Generation and following of ramp inputs (up to 1 degree per minute) with a steady-state error of less than ± 0.5 degree

2. Generation and following of sinusoidal inputs (with 90-minute periods) with a steady-state error of less than ± 0.5 degree

3. Generation of Polaris apparent motion compensation bias (± 0.9 degree) and orbit inclination compensation bias, both of which have 24-hour periods

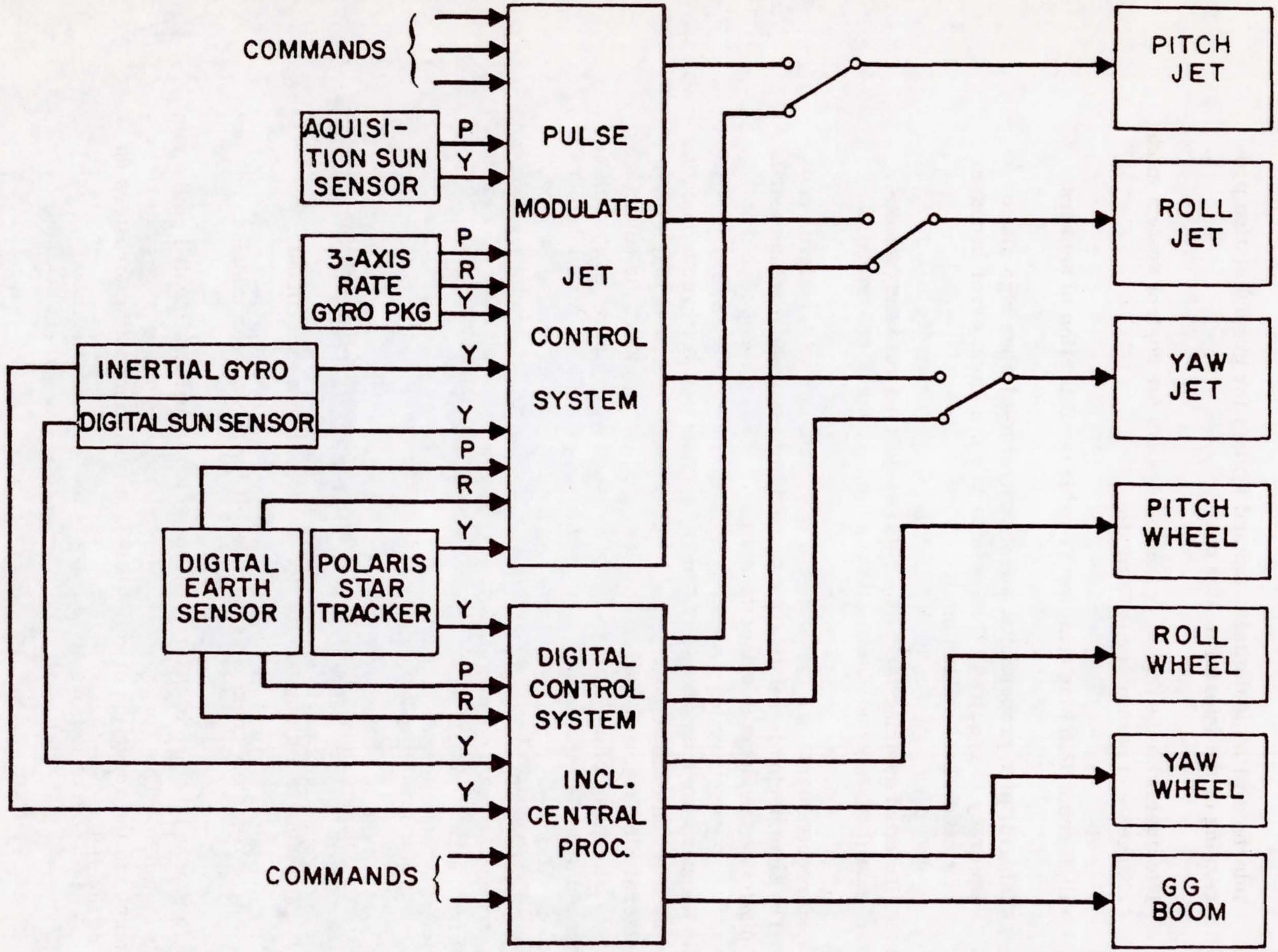

Figure VII-17. ATS-F&G control system concept.

4. Signal processing of the digital output of the earth sensor

5. Interface with the digital command system for ground station processing of the interferometer data

6. Interface with the digital command system for control system mode changes and parameter optimization

7. High resolution bias trim for in-orbit recalibration of sensors

8. Integral plus proportional wheel control used when high static accuracy is required (This results in a minimum error because of stored wheel momentum.)

9. Choice of damping technique between derived rate and pseudo-rate (both have advantages for certain modes of operation).

These requirements are most readily obviously met with a digital controller. The time period is relatively long; and time shared, medium speed, onboard digital computation is very attractive.

The torquers for the operational control system will be three momentum wheels with reaction jet dumping. This type of system was selected in the GSFC, General Electric, and Fairchild-Hiller studies. Some of the advantages for operational control of a wheel-jet hybrid system over a pure jet system are as follows:

1. Less sensitive to sensor noise

2. Less sensitive to structural vibrations and disturbances

3. Smaller orbital station-keeping disturbance

4. Gas consumption is less dependent on the operation program

5. Adaptable to a gimbaled gravity-gradient boom experiment

6. Approximately equal weight trade-off for a two-year mission.

Lockheed proposed the use of 250 micro-pound ion thrusters, located on the rim of the 30-foot dish. This concept was rejected because of the large power requirement and the reduction of reliability inherent in mounting thrusters on the rim of the dish.

The primary sensors for three-axis earth pointing control will be an optically gimbaled, infrared earth scanner (optimized for synchronous altitude) for roll and pitch, and an electronically-scanned star tracker for the yaw reference (Polaris). Because the earth sensor and Polaris tracker are recognized as long lead time items which may pace control system development, two separate SRT tasks have been initiated. SRT Task, "Vertical Sensor for ATS-F&G", will attempt to develop a high accuracy earth sensor to meet the requirements of ATS-F&G. The sensor to be developed will utilize an existing basic design but will greatly increase the accuracy. SRT Task, "Attitude Sensor for ATS-F&G", will attempt to develop a Polaris star tracker. The task will concentrate on the ability of the tracker to recognize Polaris and to lock onto it, thus giving accurate yaw control for ATS-F&G. Lockheed proposed using a spacecraft interferometer and two ground stations for the primary operational sensor. Using two lines of sight vectors, it is possible to resolve roll, pitch, and yaw errors; however, the computational requirement is complex, and the two ground stations must be synchronized so that they may time share the interferometer. A discussion of the closed-loop use of the interferometer is given in Appendix C.

Lockheed also proposed the three-axis auto-pilot, used for control during the transfer orbit as a coarse operational sensor. The stated accuracy was 0.2, 0.2, and 0.9 degree for roll, pitch, and yaw, respectively. The roll and pitch errors are based on the use of an earth sensor to null the gyro drift errors, and the 0.9-degree yaw error is the estimate of the accuracy obtained by using gyrocompassing at synchronous altitude. It should be noted that the earth sensor must be used throughout the transfer orbit; thus, it cannot be optimized for synchronous altitude.

General Electric and Fairchild-Hiller both propose the use of a Polaris sensor as the primary yaw reference and the use of an inertial three-axis package updated by the interferometer as a yaw reference backup. Lockheed proposes the interferometer as the primary yaw reference, with a two-axis gyrocompassing technique for a coarse mode of only ±0.9 degree accuracy. The Lockheed gyrocompassing accuracy is only a calculated value. Both General Electric Final Report Volume I, Book 2, pages 6-4 through 6-9, and Fairchild-Hiller Final Report Volume V, page 6-2 have included trade-offs in their mission study reports justifying the selection of the Polaris sensor over a gyrocompassing technique.

The GSFC study recommended that the primary yaw reference be a Polaris star sensor backed up by a digital sun sensor capable of being biased over a ±38 degree range and a single inertial quality gyro for inertial hold during local noon and midnight. During the ±0.1-degree roll and pitch mode, the yaw error should be less than ±0.2 degree, in order to keep the cross-coupling error less than +0.04 degree when the yaw axis is pointed at the edge of the earth. For spacecraft survival, however, it is sufficient to have the solar array pointed to within ±30 degrees of the sun, with the ability to command the roll axis to within ±3 degrees of the orbit plane for station-keeping maneuvers. In this way, the backup will provide for spacecraft survival, although its operational performance will be degraded. The inertial hold during local noon and midnight is desirable because it eliminates the yaw reacquisition after these 4-hour periods. The use of the interferometer to provide three-axis error signals is an experiment which, if it proves successful, will also serve as a backup to the Polaris sensor.

In conclusion the preferred operational control system utilized a wheel-jet hybrid system with a digital controller. An all-jet backup capability is provided. The sensors for the operational mode are earth sensor for pitch and roll and Polaris for yaw.

F. AUXILIARY PROPULSION SYSTEM ANALYSIS

The four designs for the ATS-F&G spacecraft incorporate several different auxiliary propulsion schemes. The General Electric design and the GSFC design both incorporate a hydrazine monopropellant system for station acquisition and a thermal storage resistance jet system for station-keeping and attitude control maneuvers. The Fairchild-Hiller design uses a hydrazine monopropellant system for all auxiliary propulsion requirements. The Lockheed concept uses low thermal inertia resistance jets for attitude control during the transfer ellipse and for coarse attitude control, a hydrazine monopropellant system for injection control during K motor firing and for station acquisition and change, and an ion engine system for station-keeping and fine attitude control.

STATION ACQUISITION PROPULSION SYSTEM

All four of the designs employ a hydrazine system for station acquisition. The use of hydrazine monopropellant engines is attractive because of the level of performance and reliability demonstrated by such systems. Many companies such as Rocket Research, Hamilton Standard, and TRW are now involved in the development of this type of engine for a wide range of thrust values (millipounds to hundreds of pounds). The main remaining area of development is the demonstration of performance and life over extended periods in a space vacuum environment.

Although differing in thrust levels and number of engines, the systems proposed by Fairchild-Hiller, General Electric, and Lockheed for station acquisition all favor the use of the blowdown made mass expulsion system. This system is selected by the contractors over the regulated pressure system because of the reliability gained by eliminating the regulator.

The disadvantage of the blowdown mass expulsion system is that the thrust level of a specific thruster in a multiple-thruster system cannot be accurately determined at a given time. Since all thrusters feed from a common tank, the thrust level will decay from an initial high value at a rate approximately equal to the feed tank pressure decay. For this reason, programming of thruster firing is difficult.

In addition, if spherical tankage is used, the envelope for the blowdown mass expulsion system will be larger than the envelope for the regulated pressure system because the pressurant gas and propellant would be contained in the same tank instead of in separate tanks as in the regulated pressure system.

Although the regulated pressure system uses one tank for pressurant and one for propellant and has a regulator, the system weight is comparable to that of the blowdown system. This is because of the thick-walled propellant tank used in the blowdown system to withstand the initial high pressure of the gas. The regulated pressure system tank weight has to withstand only the lower regulated pressure. The system weights and I_{sp} values proposed by the contractors are reasonable. Exact weights and I_{sp}'s can be defined only when the detailed firing times are determined.

The Fairchild-Hiller hydrazine system applies only to the spacecraft structure they have chosen because it requires access to the center of mass. Lockheed's system design is dictated to a large degree by their decision to use three-axis stabilization during injection. Consequently, the choice of the station acquisition propulsion system will come from a comparison of the General Electric and GSFC concepts.

Of these two designs, the one proposed by General Electric has more hardware and uses less propellant. It uses eight 1-pound thrusters for coning and damping control, inclination correction, and spacecraft despin. A 10-pound jet is located on the folded antenna for correction of eccentricity. The GSFC design uses two 5-pound thrusters for coning and damping control, spacecraft precession, and correction of both eccentricity and inclination. The total system weight for the General Electric design is 110-pounds, including 74 pounds of N_2H_4. The GSFC design weights include approximately 10 pounds of N_2H_4 for active damping and 12 pounds of N_2H_4 as a contingency factor (General Electric includes neither of these). The General Electric system, on the other hand, provides for spacecraft despin, which is accomplished with a 13.7 pound "yo-yo" system in the GSFC design. General Electric's launch error estimates, as well as their propellant allotment for these functions, are also larger than those of GSFC. (Latest information from NASA Lewis indicates that the GSFC hydrazine estimate for launch error correction should be increased by approximately 6 pounds.)

General Electric's use of the 10 pound radial jet on the spinning spacecraft during the transfer orbit provides a propellant savings of approximately 15 pounds. It eliminates the need for precessing the spacecraft to correct inclination and eccentricity errors. Adding electronic control circuits for the radial jet may be offset by eliminating the circuits controlling major spacecraft precessions in the GSFC concept. Locating the radial jet on the folded antenna may present a design problem. If the radial jet failed, however, the remainder of the system would possess the capabilities of the system designed by GSFC and could, if the proper controls were provided, accomplish the spacecraft maneuvers.

Because of the major precessions involved in the GSFC station acquisition sequence, it is desirable to use a constant thrust system, so that the control system will be less complex. For this reason, a pressure-regulated feed system was chosen for the thrusters in the GSFC design. General Electric chose a blowdown system, in which the thrust decays as the pressure decreases. Their choice is justified since their design does not require the accurate precessions required by the GSFC design. After examining the alternate approaches, the GSFC station acquisition concept was selected. Also, the regulated system was selected over the blowdown system.

STATION-KEEPING AND ATTITUDE CONTROL THRUSTERS

The Fairchild-Hiller spacecraft design has an accessible center of mass, hence one-pound thrusters are utilized at this location in their design. Both the General Electric and GSFC designs require two-component thrusting to achieve a resultant vector through a center at mass. The hydrazine system proposed by Fairchild-Hiller requires less power than the ammonia resistojets or the ion engines.

The back-up capability of the Fairchild-Hiller design provides redundant thrusters for attitude control functions, and an all-jet attitude control capability is available if the inertia wheel system fails (more fuel would be needed for extensive back-up use).

The use of 1-pound hydrazine thrusters for the station-keeping jets appears appropriate for the Fairchild-Hiller design. Fairchild-Hiller (volume 5) also proposes a smaller reaction control jet, using hydrazine as the propellant. They outline three different systems. A non-regulated blowdown system is shown in Figure 6-13 of their report, while Appendix 6B, page 11 shows a regulated pressure system, and page 6B-10 shows a pressure-regulated gas plenum system. Volume 5 compares the relative merits of the above systems without concluding which system should be chosen. They will withhold judgment until the spacecraft requirements become firm. However, using the ground rules on page 6B-2 of the Fairchild-Hiller report, "10 to 25 ms on-time every 300 to 700 seconds," the following can be said about the regulated and nonregulated systems (not the gas plenum system): at the duty cycle specified the catalyst bed will always be working in the cold range. A cold bed will reduce the thrust level at least 20 percent over hot-bed values. Moreover, the very short pulse widths (10 to 25 ms) combined with a cold bed will reduce the I_{sp} values to about one-third of steady-state hot-bed values. This is caused by the very high Δp values across the injector face before the chamber pressure builds up. The high Δp causes the propellant flow rate to be very high initially. The flow then reduces to design value at about the same rate as chamber pressure buildup. Therefore, I_{sp} values of less than 100 seconds can be expected with these systems at the above pulse widths and duty cycles. On the other hand, the gas plenum system appears to be attractive for low level thrust values with pulse widths and duty cycles mentioned above. I_{sp} values of 115 to 140 seconds can be expected depending on the gas temperature selected. Therefore, for short pulse widths and short duty cycles, the plenum system is preferable. For long pulse widths and varied duty cycles, the other two systems will allow higher I_{sp} values. The use of the gas plenum hydrazine system is discussed later in connection with the selection of a propulsion system for the preferred spacecraft design. Because the structural design selected is not that proposed by Fairchild-Hiller, where the center of mass would be accessible, the hydrazine system for station-keeping and attitude control thrusting was not selected.

The Lockheed design utilizes low thermal inertia resistance jets for coarse attitude control and ion engines for fine attitude control and station-keeping. Considerable back-up capability is available. The resistance jets provide back-up attitude control; the hydrazine thrusters can be used for station-keeping; and the ion engines are capable of providing for the station acquisition or station change requirements.

The Lockheed report states that the ion engines and Avco resistance jets are competitive in weight (accounting for power supply weight) for station-keeping and attitude control. Their selection of ion engines is based entirely on advancing technology. The ion engines, however, dictate to a more significant degree the design of the spacecraft power subsystem. The Avco-type resistance jets also require a high peak power. A thermal storage type resistance jet system would require lower peak power but needs more energy to heat it up to operating temperature. A more detailed discussion of the two types of resistance jets appears later in this section. The power requirements of the ion engine would eliminate it from consideration in all designs except Lockheed's (the output of the power systems of the other designs would have to be increased to accommodate the high power requirements of the ion engines). The north-south station-keeping ion engine alone requires 113 watts for 16 hours per day, and the peak power required by the entire ion engine system is 266 watts. The spacecraft power system selected will have an output of the order of 200 to 300 watts. On this basis, we believe that the ion engine cannot be justified for the ATS-F&G.

As stated, the Fairchild-Hiller and Lockheed concepts have been rejected. On this basis the comparison of the station-keeping and jet attitude control systems will be concerned only with the General Electric and GSFC designs together with an analysis of the propulsion systems which could best meet the thrust and impulse requirements imposed by the ATS-F&G mission.

The General Electric proposed station-keeping and attitude control jets have thrust values ranging from 0.86 to 1.00 millipounds. The GSFC design uses thrusters ranging from 5 to 33 millipounds. The large difference in thrust values is caused by two factors: (1) the approach to station-keeping, and (2) the earth-acquisition sequence. The GSFC design was selected under the constraint of minimum north-south station-keeping. The decision to use axial and radial thrust components to get a resultant vector through the center of mass for east-west station-keeping was greatly influenced by this constraint (how this system would change if north-south station-keeping were required is discussed later). The initial earth-acquisition sequence in the GSFC concept begins with the alignment of the spacecraft roll axis with the sun line, followed by sequential acquisition of the earth with the roll and pitch axes. The roll torque selected is that required to decelerate the spacecraft in the proper length of time for this acquisition. The pitch torque was matched in magnitude to the roll torque to satisfy the east-west station-keeping requirement. General Electric begins the earth-acquisition sequence by pointing the negative yaw axis at the sun. Their earth acquisition maneuvers can be accomplished using only momentum wheels. Their minimum thrust levels, consequently, are dictated only by the stall torques of the momentum wheels and the station-keeping firing times.

North-south station-keeping for one year requires on the order of 11,000 pound-seconds of impulse applied at the center of mass. Assuming a specific impulse of 200 seconds, this requires 55 pounds of ammonia. To perform a north-south station-keeping demonstration the easiest method is to follow the same approach as that used for east-west station-keeping: radial and axial thrusting with a resultant vector through the center of mass of the spacecraft. At this time, however, it is not certain if this technique can be used for north-south station-keeping. If the line of apsides and line of nodes do not coincide at the time of thrusting, adverse perturbations might be introduced, which could completely rule out this method.

If north-south station-keeping for a full year is a firm requirement, a method similar to that proposed by General Electric would probably be required. A thruster could be located in the antenna feed box structure to provide a net thrust through the center of mass. This technique, however, would require that either an additional propellant system be located in the earth-viewing module , or that a propellant line be run from the aft module to the additional thruster. Placing an additional tank in the feed section for north-south station-keeping purposes alone, however, is inefficient from a weight viewpoint because this thruster would require only 12.9 pounds of NH_3. Running a propellant line from the aft module to the feed box section, on the other hand, is undersirable because of pressure drop in the long feed line and increasing system complexity. Further study is required to determine whether a separate propellant system should be used or a line run from the aft module.

Before discussing the selection of the type of auxiliary propulsion system required for the ATS-F&G attitude control and station-keeping thrusting, the relative merits of the General Electric and GSFC designs should be pointed out. In general, the two designs satisfy the requirements that were established. The General Electric design is dictated by the spacecraft's earth-acquisition sequence and the one-year north-south station-keeping requirement. On this basis, their thruster layout and thrust levels appear appropriate. The GSFC design, with north-south station-keeping, presents a less complex system with more thruster back-up capability than the General Electric design. Except for the yaw couple, all jet attitude control maneuvers are performed with a single jet firing in the GSFC design. The General Electric design requires two jets for positive roll. The GSFC design, moreover, provides complete attitude control capability with one of the four thruster packages (four jets) entirely inoperative.

General Electric claims that one of the advantages of their combined attitude control and station-keeping system is that flywheel unloading is done together with station-keeping. This requires a more complex control system, and the propellant savings is only 1 to 2 pounds of NH_3, which hardly jusitifies the added complexity.

In selecting the type of auxiliary propulsion system required for the ATS-F&G mission, the four studies presented various arguments for and against a variety of propulsion systems. All recommended eliminating cold gas systems for consideration. This decision was based on the fact that the I_{sp} of heated ammonia resistance jets is two to five times that of cold gases and has a considerably lower tankage weight. For example, General Electric states that for a total impulse of 17,610 pound-seconds, resistance jets require approximately 120 pounds of ammonia with a tankage weight of 10 to 30 pounds while 440 pounds of gaseous Freon with a tankage weight of 210 pounds is needed.

With the exception of Lockheed, all contractors eliminated ion engines from consideration because of the very large power requirements.

Liquid storable bipropellants and monopropellants are not normally considered for low thrust applications because of the difficulty in handling the small liquid flow rates. Fairchild-Hiller chose a hydrazine monopropellant system to satisfy their requirement for jet attitude control functions. This decision was strongly influenced by the fact that they placed 1-pound hydrazine jets in the plane of the spacecraft center of mass for station-keeping. Without the 1-pound hydrazine system, the millipound hydrazine system must be evaluated on its own merit.

The ammonia resistance jet is favored over the plenum type hydrazine thruster for a number of reasons. The handling and storage problems are greater with the hydrazine propellant and more thermal control on the spacecraft is necessary. The performance of the hydrazine thrusters with its catalyst has not been demonstrated over an extended period of time. The I_{sp} of a plenum type hydrazine monopropellant system is estimated to be 115 to 140 seconds. The ammonia resistance jet offers the option of varying I_{sp} from 100 to 250 seconds by increasing the power input.

Various other types of propulsion systems were discussed in the individual studies. There appears to be no reason for restating their arguments. The objections to the hydrazine system, the ion engines, and the cold gas system have been stated. The two types of resistance jets considered capable of satisfying the ATS-F&G station-keeping and attitude control propulsion requirements are discussed in the following paragraphs.

Both General Electric and GSFC chose thermal storage resistance jets over the low thermal inertia resistance jets (both using NH_3) for the auxiliary propulsion system. GSFC chose the thermal storage thrusters mainly to lower the peak power requirement. Because heater cycling continues to be a development problem, it was specified that the thruster heaters would be operated at a standby power of 5 watts each.

This number is conservative and might be reduced considerably (quite possibly to zero) as heater development progresses. Since the impulse requirement for momentum storage dumping is small, it is done at the standby power. For east-west station-keeping, an additional 35 watts of power are to be added to one thruster heater. The power requirements then are: 30 watts standby (including power conditioning and feed system) and 65 watts during east-west station-keeping. If the low thermal inertia resistance jets were selected for the GSFC thrust levels, the power input for an I_{sp} of 200 seconds (same I_{sp} as estimated for east-west station-keeping thermal storage resistance jets) would be of the order of 300 watts per thruster. Two thrusters would be needed for station-keeping. This peak power requirement is too high. If the low thermal inertia jets were to be used, some limit would have to be placed on the operating temperature to stay within a reasonable power level.

General Electric chose the General Electric-type thermal storage resistance jets on the basis of total mission energy savings: 157,000 watt-hours for the General Electric system compared to 175,000 watt-hours, for the Avco-type low thermal inertia resistance jet system. At least one discrepancy in their calculations, however, is that they do not include power conditioning losses in the calculations of the G.E. system power requirements. The General Electric thrusters operate at 12 volts and 2 amperes, and, consequently, need power conpitioning to regulate the spacecraft current and voltage to the proper level. Assuming the same power conditioner efficiency of 80% that they used for the Avco jets would increase the General Electric thruster power requirement to 197,000 watt-hours. If standby power is required, moreover, the General Electric system would have an even higher watt-hour requirement.

In conclusion, therefore, the thermal storage ammonia resistance jets are recommended for the propulsion system designed by GSFC. For the General Electric design no immediate advantage is apparent with either the General Electric-type thermal storage or the Avco-type low thermal inertia thrusters. The Avco jets, contrary to General Electric's conclusion, appear to offer a total energy savings. The peak power required for the two types of jets is comparable at the thrust level selected. The energy and power requirements for the two systems could change significantly as the spacecraft design becomes final. The selection of the Avco-type resistance jets, because of their high peak power requirement at higher thrust levels, would limit the possibility of later changing the thrust level of the jets as design requirements become more definite.

In summary, the auxiliary propulsion system preferred for the ATS-F&G spacecraft is a hydrazine system for acquisition control and an ammonia resistojet system for operational control and station-keeping.

G. POWER

The four studies all selected a solar conversion power supply. As early as the proposal phase, each contractor rejected the use of an RTG as a power source. Of the four studies, General Electric, Fairchild-Hiller, and GSFC have sized their power systems at 200 to 300 watts average power level. Lockheed's proposed power system would produce an average power of 900 watts. The two extreme power requirements (Lockheed and the others) lead to different configurations of the power system. Particularly different is the solar array design. The first consideration is that the solar paddles must be located beyond the rim of the antenna to minimize shading so that a smaller battery can be used.

Whether actively oriented paddles are used depends to a great extent on the power requirements. In the case of Lockheed's requirements, a rotating paddle is easily justified on the basis of weight and cost. For lower power requirements, the other three studies come to the same conclusion: fixed paddles. The trade-off between an oriented and a fixed solar array for the spacecraft is between reliability versus weight and cost. The thermal characteristics of both types are similar and do not enter into the trade-off. Table VII-7 summarizes specific items related to the trade-off.

Table VII-7 shows that, for two spacecrafts, the costs are essentially the same for both types of solar arrays, so that for ATS-F&G the trade-off finally is between 48 pounds of additional experiments versus the risk of malfunctions with a solar array drive and slip rings. Solar array drives have been implemented successfully over long periods of time in space in the Nimbus program; however, on ATS-F&G the problem will be more difficult because the rate of rotation is only one revolution per day, about 16 times slower than the Nimbus drive. Extremely slow drives that are continuous and smooth are difficult to implement. A step-drive would require excess control system gas when the control system is operating in the all-jet mode. Because the goal of this mission is a two-year life, because of past problems with driven solar arrays, and because development of solar array drives is not a primary objective of this program, the fixed solar array was selected.

Table VII-7

Trade-off Study Summary		
Item	Fixed	Actively Oriented
Solar cell area	160 ft^2	50 ft^2
Solar panel weight	128 lb	54 lb
Solar array drive weight (2)	0	18 lb
Slip ring weight (2)	0	8 lb
Cost for tested panels for two spacecrafts	1400K	600K
Development cost of solar array drive	0	750K
Reliability	High	Requires array drive and slip rings

In the trade-off study summarized in Table VII- 7, the basis for costs were obtained from present program experience and the weights and area are justified in the GSFC Concept Study Report.

The fixed array provides a varying power output level as a function of orbital position. As the number of solar paddles in the fixed array increases, the peak-to-peak output variance decreases, but the deployment complexity and the weight increases. Because the fixed array power level varies, the batteries will experience more discharge duty cycles than those in an oriented array system. As long as the depth of discharge of these additional cycles is maintained near a maximum of 15 percent, no severe degradation of lifetime is expected in the ATS-F&G.

General Electric chose to design their system for four fixed paddles, GE Final Report Volume I, Book 2, Section 6.6.4. The justification for this decision is that it results in a less severe cycling requirement for the batteries. While this is true, it is felt that the cycling imposed by two paddles is totally acceptable. Additional reasons for a choice of two paddles are (1) two paddles require somewhat less solar cell area and (2) if for future growth additional power is required, it would surely be provided by rotating paddles provided the technology for rotating mechanisms with two year lifetime becomes available. This would mean using two paddles and, therefore, less redesign.

A discussion of voltage limiters, batteries, battery chargers, and voltage regulators is provided in the following paragraphs.

VOLTAGE LIMITERS

Voltage limiters are used to protect electronic components throughout the system from excess voltages caused by cold solar arrays. If components could be selected that would withstand the voltage excursions caused by arrays leaving the earth's umbra, voltage limiters might not be necessary. However, high voltage, high current transistors typically have slow switching times and a loss of efficiency is usually incurred when using these components in high power pulse width modulation (PWM) regulators. Filtering component weights would probably increase, but could be offset by the weight gained through removal of the voltage limiter.

Several techniques have been suggested. The partial shunt regulator described by GSFC and General Electric maintains a voltage limit by short-circuiting a portion of the array. It has the advantage of large shunt current capability at low power dissipation. The shunted portion of the array must be isolated from the main part by diodes.

Zener diode limiting, as suggested by Fairchild-Hiller, requires greater thermal dissipation, but is easier to implement because the unit can be placed outside a regulated-temperature area. The diode isolation problem is also eliminated.

BATTERIES

Nickel cadmium batteries should be used in ATS-F&G. Maximum depth of discharge should be limited to 50 percent. This is allowable since it occurs only several times a year. Lifetime would be seriously curtailed if this depth of discharge occurred daily. In case of battery failure in one string, the mission would have to be decreased. Redundancy in the form of extra capacity should be provided; this can take the form of an extra string, or higher capacity batteries in multiple battery systems. In Section 16 of the GSFC Concept Study Report, a trade-off analysis is given comparing methods to obtain redundancy and the associated weight required.

Third electrode adhydrode cells, as recommended by GSFC, should be used since these cells can provide the most positive information regarding overcharge and thereby can be used to signal full charge state. However, system tests must be conducted to determine the most favorable loading of the third electrode for the charge current reduction signal.

BATTERY CHARGERS

Battery charge regulators protect the batteries from high end of charge rates and overcharge. Two types of regulators have been proposed: taper chargers and constant current chargers.

The taper charger as described by GSFC, General Electric, and Fairchild-Hiller initially limits the charge rate until the batteries have reached a voltage that is considered safe at the battery temperature. The regulator clamps the voltage to this value and the charge current decreases as the battery continues to charge. The mode of operation is in a safe direction; the battery voltage is maintained at a safe level and the charge current decreases as the battery approaches full charge.

However, unless the system works perfectly, no cell failures, etc., the batteries can be overcharged. Although a battery failure will not necessarily occur, continued overcharge can degrade the life of the batteries. Alternatively, the current limit could be selected low enough so that the charge rates, and therefore the overcharge rates, could not degrade the batteries.

The alternative describes the essence of the constant current charger suggested by Lockheed. When long periods of time are available, the charge current is limited to rates that will not seriously damage the battery during overcharge. If the current limiting in the taper charger must be set very low, then the constant current charger should be used instead, for simplicity and reliability.

In either case some device is necessary to signal the charger to reduce the charge rate. Devices such as adhydrode cells, coulometers, or ampere-hour meters are suitable, with the adhydrode cell (third electrode) preferable.

REGULATORS

For voltage regulators, the trade-off is between an unregulated bus with a voltage regulator in each subsystem or a centrally-regulated bus. Lockheed proposed to use the unregulated bus approach. The other three studies propose using the regulated bus. Because a single regulator must be designed for peak power conditions, the unregulated bus system may be slightly more efficient from a power standpoint. However, from a total system weight standpoint, the regulated bus system is more efficient.

A bus can be closely-regulated most efficiently by non-dissipative regulators. The regulator with the greatest utilization experience is the buck type, a regulator whose input voltage is greater than its output voltage. This technique is described in the Fairchild-Hiller Final Report Volume IV, Section 4.5.4, and General Electric Final Report Volume I, Book 2, Section 6.6.6. If the input voltage falls lower than the output voltage, the regulator ceases to regulate. This system is really a voltage limiter.

A buck-boost regulator, as suggested by GSFC maintains regulation while its input voltage varies above and below its output. Its main advantage is that fewer series battery cells are required since the input voltage does not have to be maintained higher than the regulated output. For the same number of series cells, ground station control monitoring decreases greatly since low end of discharge voltages do not cause loss of regulation.

The weights and efficiencies are comparable, the buck type having an advantage.

PREFERRED SYSTEM

The preferred power system is described in the following paragraph. More detailed information is available in the GSFC Concept Study Report, Section 16.

The preferred configuration is shown by the power system block diagram in Figure VII-18. The day loads are fed directly by the solar array while night loads and power level smoothing come from the batteries. The array is split, one portion being short-circuited to maintain the voltage limit. When the sensor detects a high voltage in the unregulated bus, it shortcircuits a portion of the array. The split array feature alleviates the thermal dissipation problem. The batteries are nickel-cadmium 12-ampere-hour capacity third electrode cells. The batteries are protected by a taper charger as previously described. The bus voltage will be maintained at +28 v ± 2 percent by a buck-boost regulator. Two will be provided for reliability. This preferred concept utilizes the best features of several spacecraft power systems, all of which will have been flight-tested in time to demonstrate their usefulness for ATS-F&G.

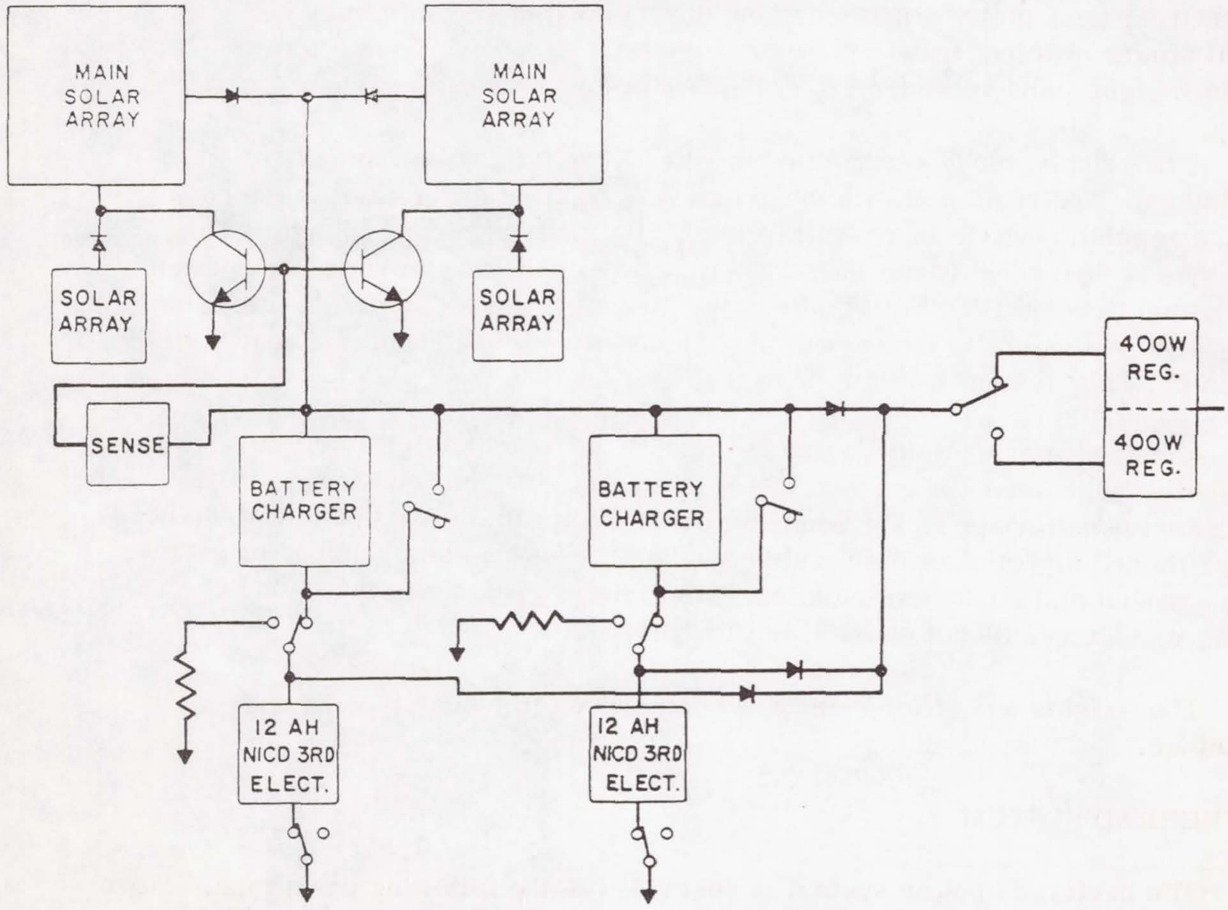

Figure VII-18. Power system block diagram.

H. TELEMETRY AND COMMAND SYSTEM

The preferred telemetry and command logic system* for the ATS-F&G has been designed to incorporate the most desirable features of the four studies, thereby providing an optimum system flexibility to meet the mission requirements. This preferred system has been described in detail in Section 17 of the ATS-4 GSFC Concept Design Study Report. This section briefly outlines the major parameters of that report and describes the trade-offs and modifications resulting from the incorporation of the contractor studies.

The primary characteristics of the telemetry and command logic system are as follows:

Command System

1. Discrete (i.e., on-off) commands

2. 10-bit digital command word

3. Command rate of 2 commands per second

4. Internal spacecraft command verification and execute indication

5. No frequency generation

6. No stored commands

Telemetry System

1. Adaptable (programmable) concept which will allow wide variations in sampling rates, points and formats

2. Two data transmission rates
   400 bits per second
   4000 bits per second

3. 10-bit sample words

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*This section describes only the telemetry and command logic. The communications system, associated with telemetry and command, and the ground data format and display techniques will be defined in detail during the next phase. The telemetry and command link calculations are shown in Appendix L.

4. 8 bit analog-to-digital conversion capability

5. Capability to accept analog, 10-bit serial digital, or 10-single bit (i.e., on-off) digital information

6. Distributed commutation to minimize harness

7. Optional ground programming mode

8. Optional auxiliary telemetry points for failure analyses

9. No low level signal conditioning

COMMAND SYSTEM TRADE-OFFS

The command system is by design constrained to real-time operation. Frequency generation requirements are minimal, and delayed time (stored) commands are not required, thereby reducing the command system complexity. The preferred system design is straightforward and implements the essential GSFC Aerospace Data Requirements. Table VII- 8 shows the command requirement breakdown for the four studies. The command systems recommended by the four studies are of the same general type, however, there are several important differences.

1. All of the systems require two types of commands, discrete (i.e., relay on-off) and digital words. The GSFC study also recommends the use of a timed-tone execute command as described in the GSFC report. It was felt that this method would be the most expedient way of operating certain desired functions. However, the alternative, proposed by General Electric, of using a digital command word to provide the control of the desired function appears to be a more desirable method of achieving this command. This decision has been predicated on the accuracy (time) required for this command and the interference with other commands. (While the timed-tone execute command is being operated, it will not be possible to enter other commands into the system.)

2. All of the systems employ digital command words, however, different techniques are utilized in the implementation of the digital command words. Lockheed's approach differs from the other three studies by making use of tone commands to obtain discrete commands and thereby deriving the command words. This is done by using three

Table VII 8

ATS-F&G Command Requirements

	GSFC	Lockheed	Fairchild	General Electric
Controls	62	26	44	185
TV subsystem	8	--	--	---
Interferometer	3	8	19	56
Thrusters	17	23	5	---
Power	24	17	20	10
Antenna	14	22	35	---
Phased array	--	69	19	8
Communications transponder	34	--	--	---
Structure	--	--	5	---
Telemetry and command	11	17	16	18
Instrumentation	--	--	--	42
Total	173	182	163	319

Note: The breakdown of commands include discretes, command words, and in the case of the GSFC study time-tone execute commands. In the case of the spacecraft command words, a single word can specify many different command functions, e.g., a single command word of 10 bits with 4 bits being used for spacecraft address can specify up to 64 different functions.

discrete commands (reset, "one," and "zero"), sent sequentially to fill a serial shift register or other storage devices with any required bit length. The other three studies use the same basic approach of examining the incoming digital command word to determine if it is a discrete or word type command. If it is a word type, it will be sent to the addressed subsystem. This method is preferred to Lockheed's because of the higher command rate, greater versatility, and capability of simplifying operational complexity

3. Spacecraft subsystem command word lengths required by the subsystems of the four studies vary (e.g., 6 bits, 10 bits, 15 bits, etc.). Determination of the actual word lengths to be implemented will by necessity be left to the next phase when finalization of all subsystem parameters will be made.

FREQUENCY REQUIREMENTS

Timing frequencies will not be provided to other subsystems by the command system. Present indications show that the requirements for these timing frequencies are minimal, therefore, this complexity can be avoided.

A master oscillator will be internal to the telemetry and command system. The various frequencies required by the telemetry and command system will be derived from the master oscillator. If at a future time it appears that frequency generation for the various subsystems is a desirable feature for either a centralized primary mode or backup mode, the necessary frequency derivations and interfaces will be included in the telemetry and command system.

STORAGE REQUIREMENTS

Stored commands are not included in the design because there is no requirement. Command capability from the ground should be virtually continuous from launch through deployment of the 30-foot dish. Stored commands will be employed only in the sense that the command will be received by the spacecraft, verified, and verification-execution indication will be sent in the telemetry bit stream.

TELEMETRY SYSTEM TRADE-OFFS

The preferred system is based upon the adaptable or programmable concept which allows wide variations in sampling rates, points and formats. This is achieved through the employment of one of several stored programs contained within the reprogrammable memory of the telemetry system.

A modified form of the adaptable telemetry concept has been recommended by the contractors, but the added flexibility of the approach outlined in Section 17 of the GSFC Report will be more valuable for this mission. The general configuration of the telemetry systems proposed by the three contractors is shown in Figure VII-19.

The system consists of a main frame commutator sampling a given number of inputs, and developing a pre-specified frame format. Sub-commutation will be developed via various combinations of sub-commutators to the depth specified in advance by that particular sub-commutator. The sampling will be done at two speeds, a low rate for predeployment and a high rate for post-deployment information. The systems are adaptable in that it is possible to speed up one of the sub-commutators to the main frame rate (preempting main frame data), and thereby provide a higher sampling rate for that particular sub-commutator. The system is inflexible because the sampling sequences and the increased sampling via the speed-up sub-commutator mode are constrained to a predetermined and prewired sequence.

The preferred data processing system will use, with minor modifications, the design of a system which will be flown on the Nimbus D spacecraft. The changes to the system will be primarily to the timing oscillator, mechanical configuration, and electrical interfaces. The preferred data processing system will be capable of transmitting 10-bit samples at two basic rates; 40 samples per second (400 bps) and 400 samples per second (4000 bps).

Lockheed indicates a total telemetry input channel requirement of 1056 channels (including 176 spaces). The composite bit rate of these channels is ≈ 34,000 bps. However, in order to fit this required sampling rate into the timing scheme of the proposed multiplexer, a bit rate of 131,072 bps will be required. A data rate of this magnitude is totally out of proportion to those recommended by Fairchild (1152 bps), GSFC (4000 bps), or General Electric (8000 bps). Furthermore, the method of obtaining the data rates in the Lockheed system indicates that little effort was given to optimizing the telemetry bit rates/subsystem sampling requirement interface.

The Fairchild bit rate of ≈ 1152 bps appears to be too low to adequately sample the subsystem telemetry data. In addition, this bit rate restricts growth capability (i.e., additional channel requirements) and flexibility in changing input channel sample rates. The rates recommended by GSFC (4000 bps) and General Electric (8000 bps) will provide adequate sampling of the subsystem information. The upper limit which will be chosen is constrained by TM antenna selection and corresponding signal-to-noise ratio received at the

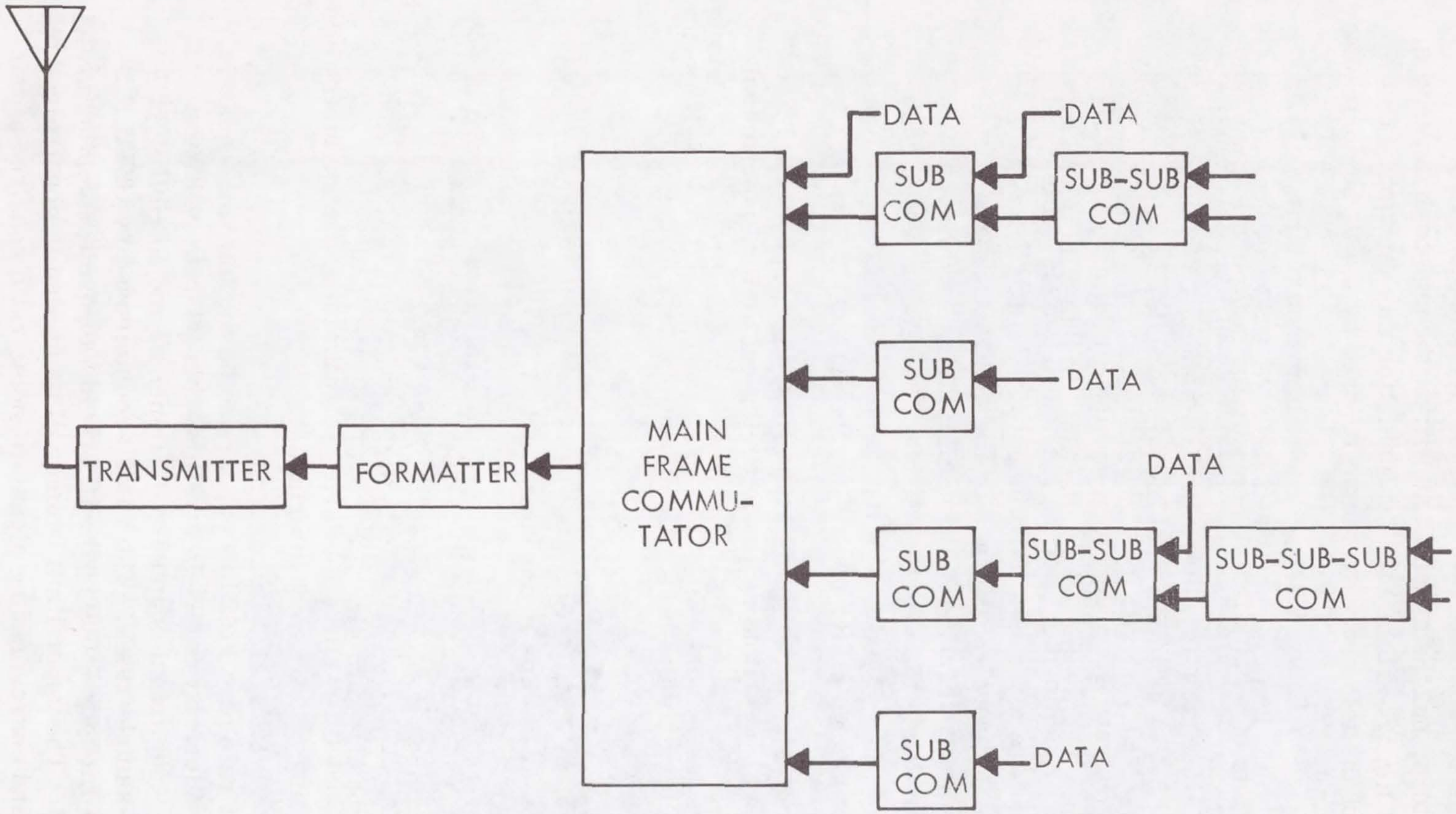

Figure VII-19. General configuration of proposed ATS-F&G telemetry system.

ground station. If spacecraft requirements are changed and if the RF parameters can be maintained, the GSFC system will be designed with a capability of sampling up to 12,000 bps by jumper selection on an external connector.

The present telemetry requirements for the ATS-F&G as tabulated for the several approaches is shown in Table VII-9 . These figures are primarily an indication of the expected total input capability of the ATS-F&G. A note of caution in examining the input channel breakdown; the contractors appear to have been inconsistent in tabulating their telemetry (and command) requirements. (General Electric's Table 5-7-1 claims a total of 544 channels, however, the numbers actually total 614. Fairchild indicates 323 channels for data, sync, etc.; however, the telemetry requirements of their Appendix 9F totals 326 points, excluding 106 predeployment points.) These inconsistencies indicate that TM requirements are a variable function, which is another reason for an adaptable system. The approaches outlined by the contractors are such that any significant changes in input channels may require major hardware changes in their systems to accommodate the new inputs. Such would not be the case with the preferred system since variations in inputs could be accommodated with a program (software) change either before or after launch.

It should be made clear at this point that the method proposed by the contractors is in itself not a poor approach. However, the size and the complexity of ATS-F&G, the changing requirements, and the necessity of having a variable bit rate; indicate that a telemetry system, adaptable to spacecraft demands, will best meet the mission requirements.

A breakdown of size, weight, power, and data rates for the telemetry and command systems of the four studies is shown in Table VII-10.

In conclusion the preferred system will offer the ATS-F&G spacecraft the flexibility required to conduct varied experimental missions. Some of the techniques described have been utilized on previous flight programs, and all techniques will have been flight tested in time for the ATS-F&G launch.

I. EXPECTED SPACECRAFT ENVIRONMENT

Conditions described in this section are indicative of service environment and are applicable to the ATS-F&G spacecraft. Safety factors for design purposes will be introduced commensurate with reliability objectives and effectiveness of design and production techniques during the next phase of the program.

Table VII-9

ATS-F&G Telemetry Requirements

	Telemetry parameters to be sampled (Analog, digital (single bit), digital word)			
	GSFC	Lockheed	Fairchild	General Electric
Controls	71	204	66	125
TV subsystem	7	---	--	---
Interferometer	50	62	34	60
Thrusters	54	68	15	8
Power	66	41	39	27
Antenna	387	111	68	257
Phased array	--	90	20	30
Communications transponder	52	---	--	---
Structure	--	44	106	---
Telemetry and command	26	260	16	7
Thermal	--	---	69	100
Total	713	880	433	614

Table VII-10

Size, Weight, Power, and Data Rate Breakdown

	Telemetry			Command				
	Size (in^3)	Wt (lbs)	Power (watts)	Size (in^3)	Wt (lbs)	Power (watts)	Pre Deployment	Post Deployment
Lockheed		*		90	** 9.50			
Prime (tlm)	2463	17	62			0.001 SS/B 1.5 w operate	192 bps	131,072 bps
Back Up (tlm)	120	9	25					
Fairchild	140	** 10	14.0 peak 2.0 S/B	150	** 9.5	0.5 S/B 11 operate	740 bps	1,152 bps
General Electric	1254	** 693	11.7 @100%$^+$ 11.7 @35%$^+$ 10.6 @5%$^+$ + duty cycle	800	** 23	2 @100%$^+$ 16 @5%$^+$ + duty cycle	500 bps	8,000 bps
GSFC	1248	** 35	6	674	** 15	4	400 bps	4,000 bps

* Non-redundant

** Redundant

NOTE: The figures given for the contractors are the best which could be obtained from their reports. The size, weight and power exclude the RF section of the Telemetry and Command system.

ENVIRONMENT DEFINITION

The spacecraft will encounter a wide range of environmental conditions during its existence from manufacture to end use. However, only a few conditions or combinations of conditions are expected to be significant to the design and operational performance of the spacecraft. For convenience of classification in design, the total lifetime environment is grouped into three phases. Phase I covers events which are essentially earth-based and long in duration, i.e., manufacture, assembly, test handling, storage, shipment, standby, and prelaunch checkout. Phase II covers events related to launch (relatively short term), i.e., boost, separation, and orbital injection. Phase III covers the orbital flight, which is a long term event.

The principal design of the ATS-F&G spacecraft will be based on Phase II and Phase III considerations. The effects of the Phase I environments will be considered in establishing ground support equipment requirements, consequently, minimizing the effects of the Phase I environments. Significant conditions of environment in the various phases are described below.

Phase I

1. Mainly temperature, humidity, pressure, and ground level solar radiation
2. Also transportation, handling and storage.

Phase II

1. Mainly aerodynamic and propulsion effects of temperature, vibration, acoustic noise, shock, and acceleration
2. Also vacuum conditions.

Phase III

1. Mainly temperature, vacuum, and operational conditions
2. Also solar irradiation, cosmic rays, charged particles, and micrometeroids.

ENVIRONMENT DURING PHASE I

The environments which would normally be experienced are covered in MIL-STD-810. Special handling, conditioning, and packaging will be used as required.

ENVIRONMENT DURING PHASE II

Temperature

The maximum temperature of the environment during launch and prior to shroud release is expected to be 50° C. The minimum is expected to 20° C.

Vibration

The main sources of vibration during the launch phase are engine acoustic noise at liftoff, aerodynamic forces near Mach 1 and near maximum dynamic pressure, and shock forces at ignition, burnout, separation, and firing of pyrotechnic devices.

The values in Tables VII-11 through VII-13 represent estimated shock and vibration flight conditions of the maximum severity which may be encountered at the ATS-F&G spacecraft adapter and booster interface ring. The levels at the ATS-F&G spacecraft and apogee motor interface during apogee motor burn are not expected to exceed the values stated for the launch condition.

The conditions of vibration encountered at ATS-F&G spacecraft and apogee motor interface, because of the apogee motor operation, are estimated to be similar to the levels shown in the above tables.

Acceleration

Expected values of thrust and lateral acceleration are given in Tables VII-14 and VII-16

Shock

Shocks caused by ignition, cutoff, staging, etc., will occur during vehicle operation. The shock environment will be equal in severity to a terminal peak sawtooth having a duration of 1.5 milliseconds and a peak level of 115 g in three axes.

Spin

Maximum nominal spin rate, which may be expected to occur simultaneously with axial acceleration due to apogee motor thrust is 70 rpm. Maximum spinup acceleration to be expected is 6.1 rad/sec^2. Maximum spindown acceleration to be expected is 10.0 rad/sec^2.

Table VII-11
Sinusoidal Vibration

Frequency (Hz)	Axis	Sweep Rate	Level (peak g)
5-250	Thrust Z-Z	4 Octaves per minute	± 1.5
250-400			± 2.5
400-2000			± 5.0
*			
5-250	Lateral X-X and Y-Y	4 Octaves per minute	± 1.0
250-400			± 2.0
400-2000			± 5.0

* Spacecraft lateral natural frequency to be above 10 Hz by design.

Table VII-12
Torsional Vibration

Frequency (Hz)	Axis	Sweep Rate	Level (rad/sec^2)
*			
20-60	Thrust Z-Z	4 Octaves per minute	± 8.6
60-150			± 17.2

* Spacecraft torsional natural frequency to be above 4 Hz by design.

Table VII-13
Random Vibration

Axis	Frequency Range (Hz)	PSD Level (g^2/Hz)	Accelera-tion (g-rms)	Duration
Thrust Z-Z and lateral X-X and Y-Y	20-150	0.01	6.1	2 minutes each axis
	150-300	*		
	300-2000	0.02		

* Increasing from 150 Hz at a rate of ± 3 db/octave.

Table VII-14

Booster Acceleration

Axis	Level (g)	Duration (minutes)
Combined *thrust and lateral	7.4 + 2.0 = 7.7	1

* 7.4 g (thrust), 2.0 g (any lateral axis), added vectorially and applied at the spacecraft adapter and booster interface.

Table VII-15

Apogee Motor Acceleration

Axis	Level (g)	Duration (minutes)
**Thrust	7.5	1

** To be applied at the spacecraft and apogee motor interface.

Table VII-16

Outer Van Allen Belt Radiation

Particle	Particle Energy (electron volts)	2-year Integrated Flux* (particles/cm^2)
Electrons	$\geq 1.6 \times 10^6$	2×10^{11}
	$\geq 40 \times 10^3$	2×10^{15}
Protons	0.1×10^6 to 5×10^6	2×10^{15}
	$\geq 30 \times 10^6$	4×10^8

* Could vary by a factor of ± 10.

ENVIRONMENT DURING PHASE III

Conditions listed in this section may be encountered by the spacecraft during operation in orbit. Equipment shall meet operative requirements after being subjected to the controlling combination of conditions during prelaunch and launch, as well as during or after the orbit conditions, as appropriate.

Temperature

The internal mean temperature of the aft-equipment module at the rear of the antenna during the orbit phase is expected to be maintained between 0 and 30 °C, by means of insulation and active thermal control. The external surfaces of the antenna and spacecraft will be subjected to radiation from the sun (422 BTU/ft^2/hr), radiation from the earth (including the albedo) (4 BTU/ft^2/hr), and radiation to space.

Ambient Pressure Environment

The spacecraft shall be capable of operation within specifications at the synchronous altitude where the ambient pressure is expected to be approximately 10^{-12} Torr.

Synchronous Orbit Radiation Environment

Radiation Due to Outer Van Allen Belt - The proton and electron particle energies and integrated flux to be encountered during a 2-year period in the synchronous orbit due to the outer Van Allen belt are given in Table VII-

Radiation Due to Solar Flares - It is expected that during a 2-year period in orbit, solar flares of 3+ magnitude, yielding an integrated flux of 5×10^9 particles/cm^2/event, will occur. Over this 2-year period, the total flux of protons having energies greater than 30×10^6 electron volts is expected to be on the order of 10^{10} protons/cm^2 during solar maximums and appreciably less (4 to 5 orders of magnitude) during solar minimums.

Operational Acceleration Environment

The following combined disturbances, which have been multiplied by 1.5, may be experienced because of antenna slewing, satellite attitude control, and station-keeping operations.

1. Angular accelerations
   Pitch : 6.7 degrees/min^2
   Yaw : 3.3 degrees/min^2
   Roll : 6.7 degrees/min^2

2. Angular velocity
   Pitch : 13.4 degrees/min
   Yaw : 13.4 degrees/min
   Roll : 13.4 degrees/min

3. Translational accelerations
   Pitch : $0.2\ ft/sec^2$
   Yaw : - - - - -
   Roll : $0.2\ ft/sec^2$

J. GROUND SUPPORT

INTRODUCTION

This section is a discussion of the telemetry, command, tracking, and ground station equipment required for the ATS-F&G mission. A number of available or obtainable means for supporting the ATS-F&G mission were considered with the selection of a preferred approach. The unified-system operating at S-band was selected as the basic approach. This approach combines the functions of tracking, telemetry, and command into a single system, thereby reducing the subsystem requirements on-board the spacecraft and on the ground.

The three study contractors presented only limited material on the ground systems. All contractors indicated their preference for the use of existing ground stations. They all submitted a list of equipment required to update the existing ground stations to be compatible with the ATS-F&G missions. Lockheed chose a 4 to 6 GHz system for primary TT&C with a backup capability at VHF. General Electric and Fairchild-Hiller both selected S-band.

GENERAL DESCRIPTION

Figure VII-20 is a functional block diagram of the information flow process for the ATS-F&G missions. Operation control is centered at the ATS operations control center. Data is transmitted to the operations control center from the ground terminals (Rosman, Santiago, etc.) and from the operational and experiments computers. Based on these and other data, mission control decisions are made and commands are sent to the spacecraft throughout its lifetime. The first and most critical phase is the injection, deployment, acquisition, and control sequence. This phase begins at launch and is completed when an operating spacecraft is accurately located in synchronous orbit. The second phase begins when an operating spacecraft is achieved and experiments are exercised. During this period, the spacecraft is operated and maneuvered to comply with the requirements of the experimenters.

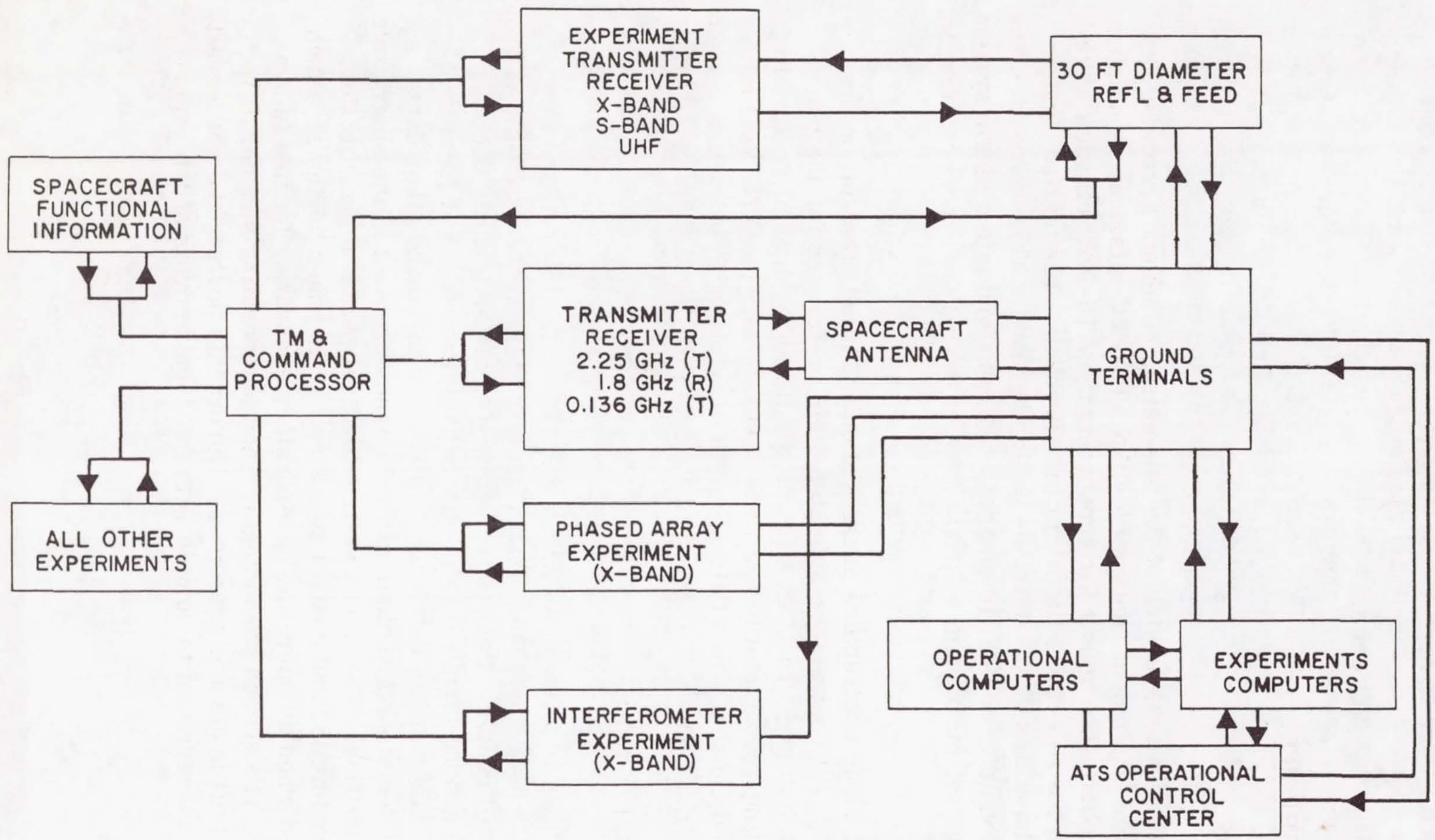

Figure VII-20. Information flow diagram.

The operational control computer is a key element during the first phase. A sun sensor aboard the spacecraft detects sun angles which are transmitted via a telemeter link through the ground terminal to the operational computer. The ground terminals also measure spacecraft signal polarization angle and ground antenna angles. From the sun data, RF polarization angle, and ground antenna angles, the operational computer determines three-axis spacecraft attitude. During this same period, a star-field-mapper experiment located on the spacecraft will detect the angle of stars with respect to the spacecraft axis. These data will be transmitted via telemeter link through ground terminals to the experiments computer. Based on these data the computer will also determine three-axis spacecraft attitude. During this period, the Goddard Range and Range Rate System will be transmitting data from the ground terminal to the operational computer. From these data the operational computer will determine spacecraft position and velocity as a function of time. Based on the derived spacecraft attitude, position, velocity, and other angle measurements, the various spacecraft maneuvers required for injection, deployment, acquisition, and control will be executed. The specific maneuvers are discussed in more detail in Section VI C (Launch Sequence and Trajectory). The orbital determination and command calculations are discussed in Section VI D (Computational Support).

TRACKING, TELEMETRY, COMMAND, AND POLARIZATION ANGLE MEASUREMENT

The Goddard Unified Tracking, Telemetry, and Command System will be used. This system operates in the S-band frequency region. Redundancy in command is provided by the use of two receivers on the spacecraft. Redundancy in telemetry, tracking, and polarization angle measurements is provided by the use of two 1-watt S-band transmitters on board the spacecraft. Further redundancy in telemetry and tracking is provided (Minitrack) at altitudes less than synchronous by a 1-watt VHF transmitter on board the spacecraft. Polarization angle measurements at VHF is not used because of the large errors introduced by Faraday rotation effects. The total system characteristics are presented in Table VII-17.

The present Applications Technological Satellite program operates in the VHF spectrum with command in the 148 to 154 MHz band and telemetry in the 135.9 to 138 MHz band. This frequency band was rejected for telemetry and command for the following reason. There are more than 10 spacecraft presently operating at the ATS-A through E spacecraft command frequency, and more than 30 spacecraft are using the 136 to 138 MHz spacecraft telemetry band. The STADAN loading study for the future shows an increase

Table VII-17

System Characteristics

Characteristics	Up-link	Down-link	Down-link
Frequency (MHz)	1760-1850	2200-2300	136-138
Transmitter power	10 kw	1 w	1 w
Transmitter antenna gain	+41.5 db	-3 db	-3 db
Antenna polarization	Circular	Linear	Linear/Circular
System noise temperature (°K)	1160	100	430°
COMMAND			
Modulation	PM	-	-
Bit rate (bps)	128	-	-
Probability of bit error	10^{-10}	-	-
TELEMETRY			
Modulation	-	PCM/PM	PCM/PM
Bit rate (bps)	-	400 or 4000	400 or 4000
Probability of bit error	-	10^{-5}	10^{-5}
TRACKING (R&RR)			
Modulation	PM	PM/PM	CW
Range tones	8, 32, 100, 800 Hz	4, 20, 100, 500 kHz	-
Range accuracy	-	±15M	-
Range rate accuracy	-	±0.1 M/sec	-
Pointing angle accuracy	-	±0.1°	±20 sec arc
TRACKING (polarization angle)			
Angle accuracy	-	±1°	-

in the number of spacecraft to be operated at these frequencies. There have been numerous situations where difficulty is encountered in commanding the ATS-B satellite, with outside interference suspected as the cause of the difficulty. Regardless of the outcome of the investigations into the cause, the example amply points out the potential problems in crowding the VHF spectrum.

The utilization of the 1760 to 1850 MHz spectrum for commanding and the 2200 to 2300 MHz range for telemetry, tracking, and polarization angle measurements poses no difficulty. This frequency is uncrowded and commonality of the tracking system transponder at S-band and the spacecraft telemetry transmitter may be achieved. In addition, ground tracking receivers and ground telemetry receiver-demodulators become common equipment.

The utilization of S-band frequencies for all primary functions, simplifies somewhat a very complex antenna design problem. Throughout the trajectory, from launch to the time when an operating spacecraft accurately located in synchronous orbit is achieved, a number of spacecraft events are involved. These events dictate antenna design requirements which are difficult to meet. The spacecraft will be maneuvered extensively during this period, thus it may achieve any attitude in space. The spacecraft will leave the launch pad in a shroud. The shroud will be removed in space exposing a stowed spacecraft configuration. Finally the spacecraft antenna will be deployed.

An omnidirectional antenna is needed to provide continuous transmission coverage while the spacecraft is maneuvered. The omnidirectional coverage must be maintained while the spacecraft is in the stowed and deployed configuration. To conduct polarization angle measurements, linearly polarized RF radiation is required, but omnidirectional antenna coverage is incompatible with a requirement of linear polarization.

The study contractors had various solutions to this problem. Switching antennas and sharing energy between two or more antennas located at various positions on the spacecraft was suggested. This problem does not lend itself to paper design or analysis. Antenna modeling employing a scaled spacecraft will be used to determine the final approach. Based on existing experience it is believed that the gain can be held higher than -3 db throughout all spacecraft attitude combinations when referenced to the several ground terminals and their different geometrical coordinates. The minimum antenna gain considered in the link calculations, Appendix D, was -10 db.

Command will be incorporated by phase modulating the up-link carrier with a subcarrier modulated by the 128 bps command code. This is demodulated in the spacecraft transponder and fed to the spacecraft command processor. (see Figure VII-19). The subcarrier is necessary to prevent the commands from appearing in the down-link tracking bandwidth. The subcarrier frequency setting will be determined by further study.

Appendix D derives the appropriate up-link signal-to-noise ratio. The command link calculations are shown with no adjustment for sharing power between the command function and the tracking function. A power division of 50 percent between command and tracking has been chosen for nominal conditions, so that for simultaneous use of command and ranging, the signal-to-noise ratios are reduced by 3 db. When simultaneous command and tracking is not required, the nominal and worst case S/N ratio margins for the up-link channel are +14.5 db and +6 db, respectively.

Two telemetry bit rates (400 and 4000 bps) are to be accommodated. The telemetry will be summed with the R&RR subcarrier and phase-modulated directly on the down-link carrier. Coherent demodulation of the signal will take place in the multifunctional receiver. Consideration of signal-to-noise ratio requirements for the specified 10^{-5} bit error probability indicates that a transponder power output of 1 watt is required. Telemetry calculations are shown in Appendix D. Nominal and worst case S/N ratio margins for the 4000 bps data rate are +6.7 db and +3.7 db, respectively.

The VHF secondary telemetry system will be designed to accommodate both the 400 and 4000 bps rates. No ground terminal additions are necessary for interfacing with these bit rates or carrier frequencies. VHF telemetry parameters are shown in Table VII-17 and link calculations are included in Appendix A. The 400 bps rate will be used prior to earth acquisition by the spacecraft at synchronous orbit or under contingency conditions. Nominal and worst case S/N ratio margins for the 4000 bps data rate from synchronous orbit are +3.6 db and -0.4 db, respectively. All margins become positive at the 400 bps data rate.

There are two systems available that measure range and range rate out to synchronous altitudes, one at S-band (the Goddard Range and Range Rate System, GR&RR) and one at C-band (the Applications Technology Satellite Range and Range Rate System, ATSR). The ATSR system and the GR&RR system are similar in function but differ in carrier frequencies.

The up-link and down-link frequencies of the ATSR system lie in frequency bands allocated for Domestic Public, Common Carrier, Fixed, and Communications Satellite. The Goddard Range and Range Rate Tracking System (GR&RR) frequency spectrum lies entirely within frequency allocations assigned for government use. Thus, the process of elimination in selecting a primary tracking system from the three available systems (Minitrack, ATSR, and GR&RR) directs that the primary tracking support for ATS-F&G be the Goddard Range and Range Rate System.

A 2-channel mode of operation is proposed since most of the trajectory will be in view of pairs of the unified S-band stations: Madagascar-Carnarvon and Santiago-Rosman. The highest major tone used will be 100 kHz. The calculations in Appendix D are based upon a sequential operation of the pairs of stations. Simultaneous use of the transponder by pairs of ground stations will lower the signal-to-noise ratios calculated in the appendix by 3 db. Based upon the signal-to-noise ratios indicated in the link calculations, the accuracy of 15 meters in range and 0.1 meter/second in range rate can easily be achieved under all conditions.

Acquisition time (from the time the spacecraft appears over the antenna horizon to the time range and range rate data is obtained) can be calculated in terms of seconds of time. (See pages 3-108 and 3-109 of the GR&RR Design Evaluation Report for various S-band acquisition times.) A more realistic statement of acquisition time, including operator reaction time is as follows:

1. Acquire VHF beacon when spacecraft appears over horizon: 30 seconds to 5 minutes depending upon accuracy of nominal predictions

2. Acquire S-band carrier: approximately 30 seconds to 1 minute, including operator reaction time. The S-band antenna is slaved to the VHF antenna

3. Acquire ranging signals and begin data readout: 30 seconds to 1 minute.

The acquire time could reach a maximum of 7 minutes, or could be as short as 1.5 minutes. However, no matter how pessimistic an acquisition time is assumed, Table VII-18 shows that visibility times for each station is in hours and not minutes (except Mojave on the first transfer orbit). For the purposes of ATS-F&G support, acquisition time for the unified S-band ground system is not considered significantly long. No onus should be put on possible acquisition times in minutes since any system which will return the same volume of information as the S-band system must have similar acquisition times. The GR&RR will provide range data with an accuracy of 15 meters rms, and range rate data with an accuracy of 0.1 meter/second rms.

Table VII-18

Station Visibility During First Transfer Orbit

Station	Symbol	Coverage Times* (Hr: Min)	Total Time Visible
Johannesburg	A	00:05-00:10 05:56-08:54 11:27-15:02	6 hr 47 min
Madagascar	B	00:06-00:58 02:44-09:40	7 hr 48 min
Carnarvon	C	00:17-10:07	9 hr 50 min
Orroral	D	00:27-07:59	7 hr 32 min
Toowoomba (ATS)	E	00:29-08:06	7 hr 27 min
Japan (ATS)	F	00:50-10:21	9 hr 31 min
Quito	G	10:42-15:46**	5 hr 4 min
Santiago	H	10:42-15:46**	5 hr 3 min
Rosman (ATS)	I	11:40-15:46**	4 hr 6 min
Mojave (ATS)	J	15:01-15:46**	45 min

* Time T = 00:00 Occurs at Point 5 in Figure VII-20, 1st Equator Crossing

** Fire Apogee Motor 15:46

A detailed description of the GR&RR as it exists at present is given in Design Evaluation Report #NASA-9731, dated November 1964 and prepared by General Electric. Following the planned conversion of the system, it will operate at the 1760 to 1850 MHz (earth-to-space) and 2200 to 2300 MHz (space-to-earth) frequency allocation as described in GSFC document Goddard Range and Range Rate System Specification S-531-P-17 Exhibit A, dated May 1966.

Tracking at 136 MHz by Minitrack stations will also be utilized. This system provides range determination by angle measurements and triangulation.

Accuracies of 20 seconds of arc (best case at zenith) can be expected for adequate signal levels during the transfer orbit. There is no interface of Minitrack with other systems, except for the possible reliance of telemetry upon the VHF system during the transfer orbit (S-band transmitter dual failure mode). In this situation Minitrack and telemetry would be required to time share the VHF spacecraft transmitter. See Appendix D for the calculations indicating the range limitations of the 1-watt Minitrack transmission system.

GROUND STATION COVERAGE

Figure VII-21 and Table VII-19 indicate the proposed ascent trajectory and identify those ground stations capable of support and the regions of potential coverage. Table VII-19 also provides the time of occurrence of these significant events.

Point 1 as listed in Table VII-19 is the Atlas burn-out, shroud separation, and Centaur first-time ignition. Point 2 is the termination of the first Centaur burn, the injection into a circular parking orbit, and the initiation of the coast phase. All tracking data covering these initial points will be available via the C-band radar systems located at Cape Kennedy and downrange. Centaur vehicle telemetry will be available via the Centaur telemetry system. Command functions will be available only to the Range Safety Officer. Selected spacecraft data during the early portion of this phase will be made available through the Fort Myers-GSFC ground station at VHF frequencies.

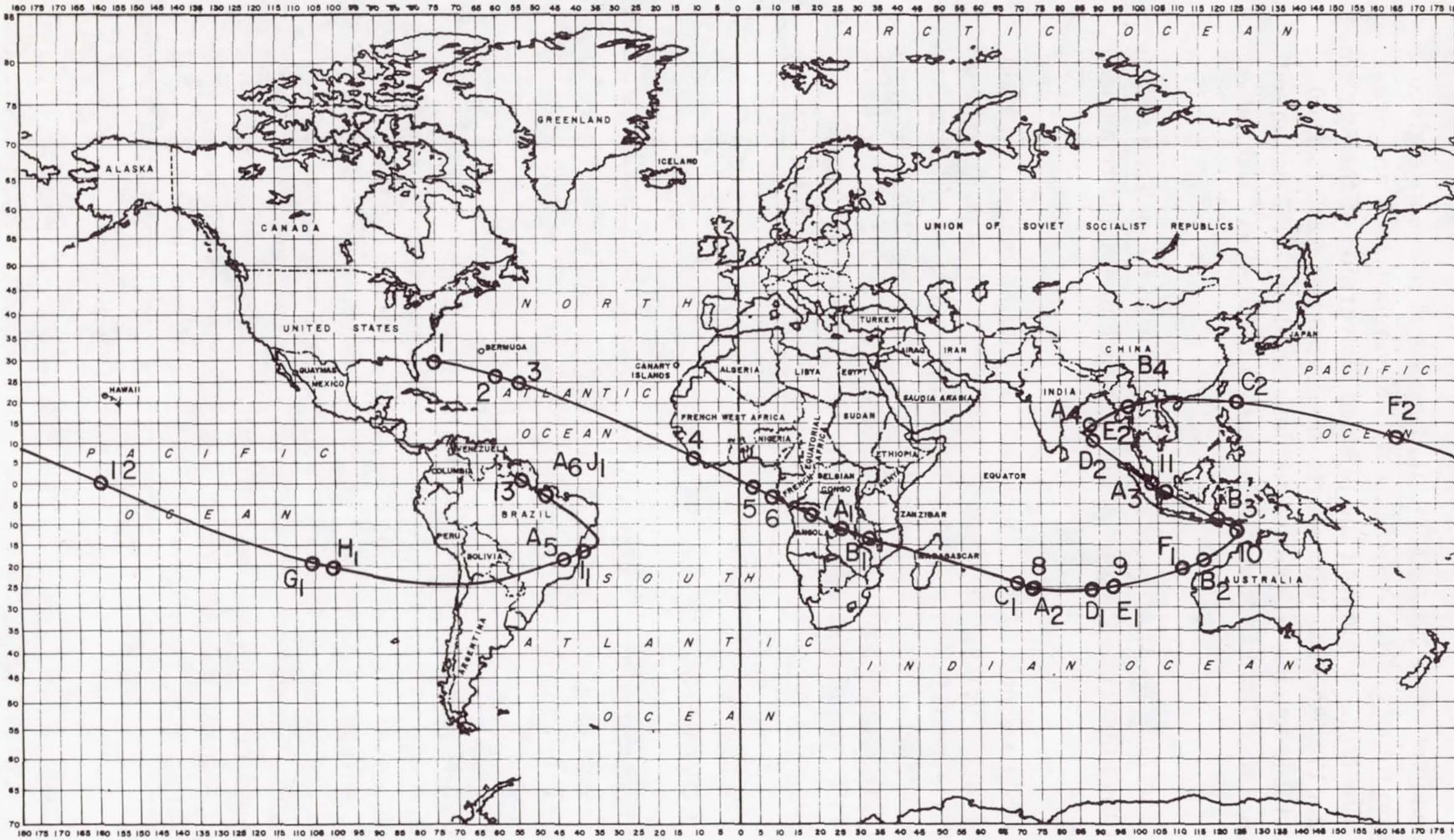

Figure VII-21. Proposed ATS-F&G ascent trajectory.

Table VII-19

Significant Events ATS-F&G Ascent Trajectory

Point Number	Event	Time (hr:min)	Long. (degrees)	Lat. (degrees)	Range (km)	S/C Height (km)
1	Atlas burnout, shroud separation, Centaur lst ignition	00:27	N/A	N/A	N/A	146
2	Termination of lst Centaur burn, injection into circular parking orbit, initiation of coast phase	00.18	N/A	N/A	N/A	182
3	Coast phase continued	00.16	N/A	N/A	N/A	182
4	Coast phase continues	00.02	N/A	N/A	N/A	182
5	Initiation of Centaur 2 nd burn, lst perigee, plane change	00.00	4.49	-0.03	N/A	182
6	Completion of Centaur 2nd burn, transfer orbit established, re-orientation of vehicle through 175° of pitch axis	00.015	N/A	N/A	N/A	215
A_1	Johannesburg acquires	00.05	27.61	-9.70	1911	478
B_1	Madagascar acquires	00.06	31.98	-11.34	2020	602
C_1	Carnarvon acquires	00.17	70.37	-21.19	5956	2843
8, A_2	Johannesburg loses	00.19	75.38	-21.76	6735	3342
D_1	Orroral acquires	00.27	90.81	-22.33	9300	5406
9, E_1	Toowoomba acquires	00.29	93.76	-22.23	9837	5924

Table VII-19 continued)

Point Number	Event	Time (hr:min)	Long. (degrees)	Lat. (degrees)	Range (km)	S/C Height (km)
F_1	Japan acquires	00:50	112.40	-19.48	15,574	11,061
B_2	Madagascar loses	00:58	116.10	-18.32	17,470	12,834
10	N/A					
B_3	Madagascar acquires	02:44	122.67	- 8.19	33,834	28,701
11	1st Apogee	05:15	105.50	- 0.06		35,775
A_3	Johannesburg acquires	05:56	100.20	- 2.10	40,509	35,284
D_2	Orroral loses	07:59	87.45	- 9.19	32,511	27,413
E_2	Toowoomba loses	08:06	87.08	- 9.69	31,718	26,633
A_4	Johannesburg loses	08:54	87.18	-13.80	24,985	20,053
B_4	Madagascar loses	09:40	98.30	-19.48	15,726	11,180
C_2	Carnarvon loses	10:07	126.09	-22.32	8189	4504
F_2	Japan loses	10:21	164.63	-15.97	3479	1176
12	2nd perigee	10:30	-156.01	- 0.95		185
G_1	Quito acquires	10:42	-104.51	-18.04	4091	1563
H_1	Santiago acquires	10:43	-101.00	-18.84	4210	1778
A_5	Johannesburg acquires	11:27	- 42.44	-18.47	17,149	12,501

Table VII-19 (continued)

Point Number	Event	Time (hr:min)	Long. (degrees)	Lat. (degrees)	Range (km)	S/C Height (km)
I_1	Rosman acquires	11:40	-38.20	-16.69	19,948	15,199
J_1	Mojave acquires	15:01	-46.70	- 2.04	40,396	35,180
A_6	Johannesburg loses	15:02	-46.82	- 1.99	40,431	35,207
13	2nd apogee	15:46	-52.51	- 0.19		35,775

Point 5, second initiation of Centaur burn and its plane change, and point 6, completion of Centaur second burn, transfer orbit establishment, and reorientation of the vehicle through 175 degrees of the pitch axis, are not visible to any ground station because of the spacecraft 100 nautical mile altitude. However, ship coverage is possible for these points, if desired. Spacecraft appearance above the Johannesburg horizon is 5 minutes after point 5 and is 6 minutes after point 5 for Madagascar.

Upon reaching an altitude sufficient for visibility at Johannesburg (point A_1) and Madagascar (point B_1), the spacecraft remains visible to more than one ground station until it passes Japan in its descent toward second perigee. After disappearance from view of the Japan ATS station, visibility does not occur again until points G_1 and H_1, off the coast of South America, are reached.

Table VII-14 lists station coverage times during the transfer orbit.

Ground Terminal

Table VII-20 lists the equipment operational at the stations with ATS-F&G visibility. The stations presently compatible (after some modifications such as polarization angle measurement equipment) with the proposed S-band tracking, telemetry, and command system are Madagascar, Carnarvon, Santiago, and Rosman. Of the 10 stations that have visibility during the first transfer orbit, the four previously mentioned have a total visibility time of 26.8 hours (42 percent), compared to the 10-station total visibility of 63.9 hours (see TableVII-18), with critical visibility by Rosman and Santiago at the second apogee. The ATS-equipped stations of Toowoomba, Japan, Rosman, and Mojave have a total visibility time during the first transfer orbit of 21.8 hours (34 percent), with Rosman and Mojave visibility at the second apogee.

The five Goddard Range and Range Rate stations at Rosman, Madagascar, Carnarvon, Santiago, and Alaska will be converted to the new frequency allocations of 1760 to 1850 MHz, earth-to-space, and 2200 to 2300 MHz, space-to-earth. (Alaska will not be utilized for ATS-F&G support.) Under this conversion, it is planned also to replace the dual 14-foot parabolodial S-band antenna, one receive and one transmit, with a single 30-foot diameter paraboloidal antenna having a Cassegrainian feed system. This modification is necessary only to Rosman, Carnarvon, and Madagascar, since Santiago and

Table VII-20

Station Capability

Station	Tracking			Telemetry		Command	
	No. Links	Type	Freq. (MHz)	No. Links	Freq. (MHz)	No. Links	Freq. (MHz)
Johannesburg	1	Minitrack	136	2	136	2	148
				1	136 or 400		
Madagascar	1	GR&RR	S-Band	2	136	1	148
	1	Minitrack	136				
Carnarvon	1	GR&RR	S-Band			1	148
Orroral	1	Minitrack	136	1	136 or 400 or 1700	3	148
				3	136		
Toowoomba (ATS)	1	ATS	4000/6000	1	136*	1	148
Quito	1	Minitrack	136	2	136	2	148
				1	136 or 400		
Santiago	1	GR&RR	S-Band	1	136 or 400	2	148
	1	Minitrack	136	2	136		
Rosman (STADAN & ATS)	1	GR&RR	S-Band	1	136 or 400 or 1700	4	148
	1	ATS	4000/6000	1	136 or 400 or 1700 or 4000*		
				2	136		
Mojave (STADAN & ATS)	1	Minitrack	136	2	136	2	148
	1	ATS	4000/6000	1	4000*		
Ft. Myers	1	Minitrack	136	2	136	2	148

* 4000 MHz - Communications Experiment Down Link
136 MHz - Telemetry

Alaska have the single 30-foot diameter antenna. Advantages gained over the use of the dual 14-foot antenna assembly are increased gain due to aperture enlargement and reduction of system noise temperature. The Cassegrainian feed is superior in reducing antenna backlobes and thus reducing terrestial noise radiated into the system through the antenna backlobes.

The polarization angle measurement system at 2200 to 2300 GHz is not an operational system at this time. However, no major difficulties are anticipated in its development since the proposed system is a modification of the successful polarization tracking system developed for the ATS-B satellites, adapted to S-band frequencies.

To provide an adequate down-link telemetry margin, it is necessary to install cooled parametric amplifiers with an effective amplifier noise temperature of 25°K (see Figure VII-21). Although the cooled parametric amplifier as one unit has an effective noise temperature of 25°K, it is not expected that the system noise temperature can be reduced much below 100°K. The 100°K-temperature was used in the link calculations, Appendix D.

All of these modifications, polarization angle tracking at S-band for Santiago, Rosman, Carnarvon, and Madagascar; conversion of all GR&RR's to the new S-band frequency allocations; changing of the 14-foot reflectors at Rosman, Carnarvon, and Madgascar; and installation of the cooled parametric amplifiers at all four stations will be completed and operational before January 1970.

Equipment is presently installed in all STADAN stations which is capable of decommutation of any format conforming to the Goddard aerospace telemetry standards at the 400 and 4000 bps rates proposed for ATS-F&G. Twenty channels of real-time display and/or computer entry is also provided. The large volume of data to be provided by the ATS-F&G spacecraft on a 24-hour per day basis over its projected 2-year lifetime, makes it imperative that methods for data compression and rejection of low value data be fully exploited. Intensive consideration will be given to the feasibility and methods of implementation of such techniques.

The preceding sections have given the outline of a unified S-band tracking, telemetry, and command system. It is also desirable to incorporate the attitude determination system with the TT&C system. This combination is feasible, and it can be shown that the resulting system will provide an accuracy of ±1 degree for the polarization angle measurement.

Establishment of the VHF ground stations is considered relatively straightforward due to the multiplicity of existing VHF ground equipment and the predominant experience in the VHF spectrum. See Figure VII-22 for the block diagram of the VHF ground equipment.

SUMMARY

The Goddard unified tracking, telemetry, and command system will be used. Two command receivers on the spacecraft will operate in the 1760 to 1850 MHz region. The frequency selected for telemetry, tracking, and attitude determination is 2200 to 2300 MHz with a VHF backup for tracking and telemetry. Two transmitters will be provided in the S-band and one in the VHF region. The command data rate is 128 bps. The telemetry data rate is 400 bps prior to earth acquisition of the spacecraft and 4000 bps after earth acquisition for the primary telemetry system.

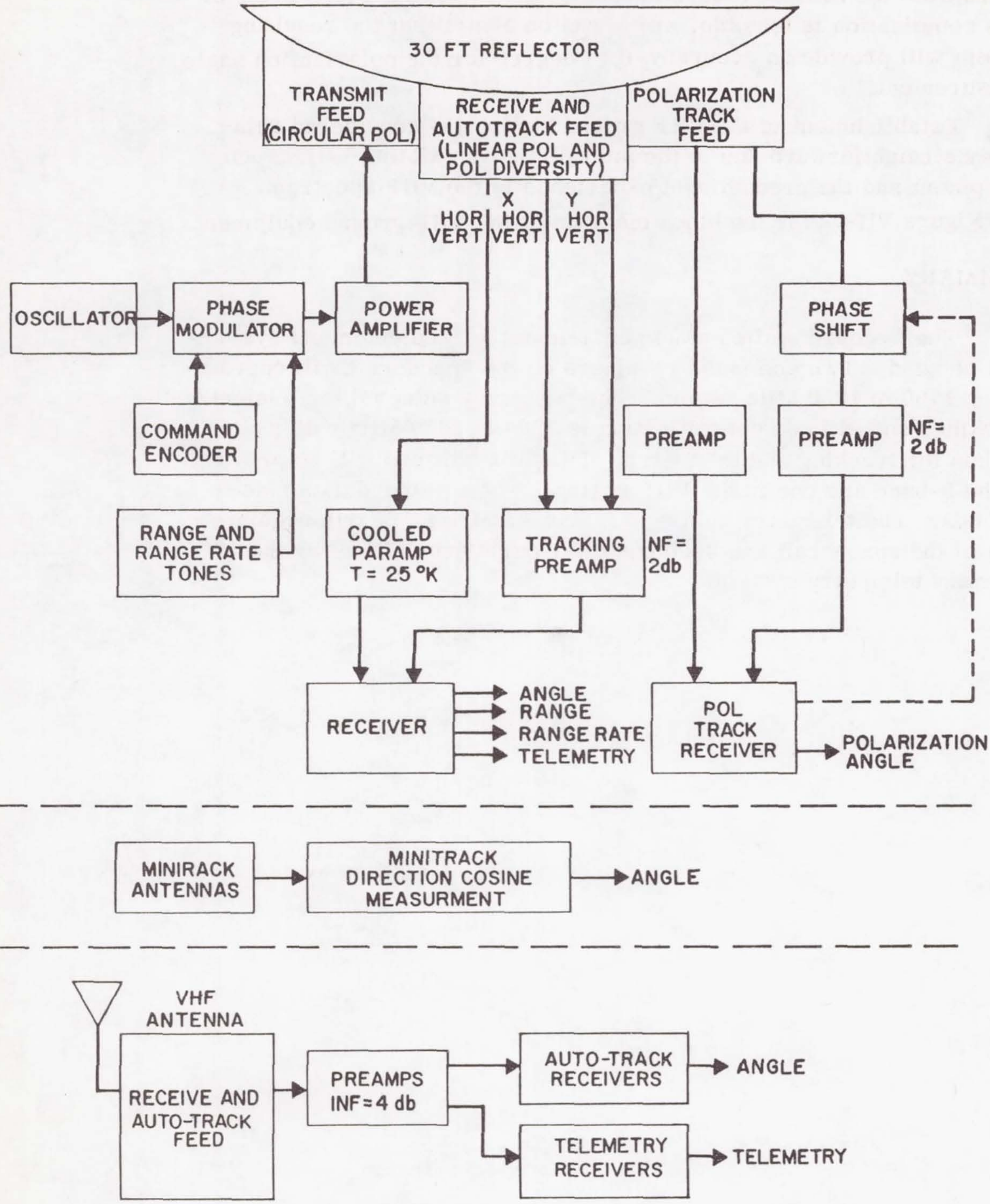

Figure VII-22. VHF ground equipment.

APPENDIX A

ANTENNA FEED MAST TRADE-OFF AND SELECTION

The RF feed which illuminates a parabolic reflector must be supported by a structure. In ground-based radar and communication antennas, this structure or mast frequently consists of three trusses which form a tripod support, with the feed located at the apex of the tripod. The legs of the tripod are attached to the reflector either at its periphery or on a circle whose diameter is one-half to two-thirds the size of the dish diameter. Such a mast is near optimum from the RF performance viewpoint, since energy scatter and blockage are minimized. Because of packaging constraints imposed by the launch vehicle shroud on the spacecraft, it would be necessary to deploy the mast and feed if such a structure were used. Failure of this deployment sequence would be catastrophic because all three mission objectives would be essentially lost.

The alternate approach is a mast designed so that it can be fabricated and adjusted in the factory. This approach requires that the mast struts be attached to the reflector on a circle whose diameter is about 80 inches, since the stowed reflector must still fold over the mast. The disadvantage of this approach is that the RF performance is degraded compared to the standard feed support.

During the course of the study, the trade-off (operational reliability vs RF performance degradation) was discussed with each of the contractors. Scaled measurements were made by both the government team and the three mission contractors. A summary of the measurement data is given below.

The Fairchild-Hiller Space System Division, through their subcontractor, Airborne Instrument Laboratory, Mineola, New York, used a 6-foot diameter reflector with an $f/d = 0.3$ and a scaling factor of 5. The measurements were made on a 150-foot range. With this short range the antenna was in the near field. As it was necessary to refocus the feed, the data are subject to some interpretation. Scaled measurements were made for 0.8-, 2.1-, 2.4-, 3.4-, and 7.9-GHz full-scale frequency. Table A-1 is a tabulation of main beam change, sidelobe levels, and sidelobe level change plotted against the full-scale frequency. The measurements were made using the reflector and a 3-V tripod, as shown in Figure A-1.

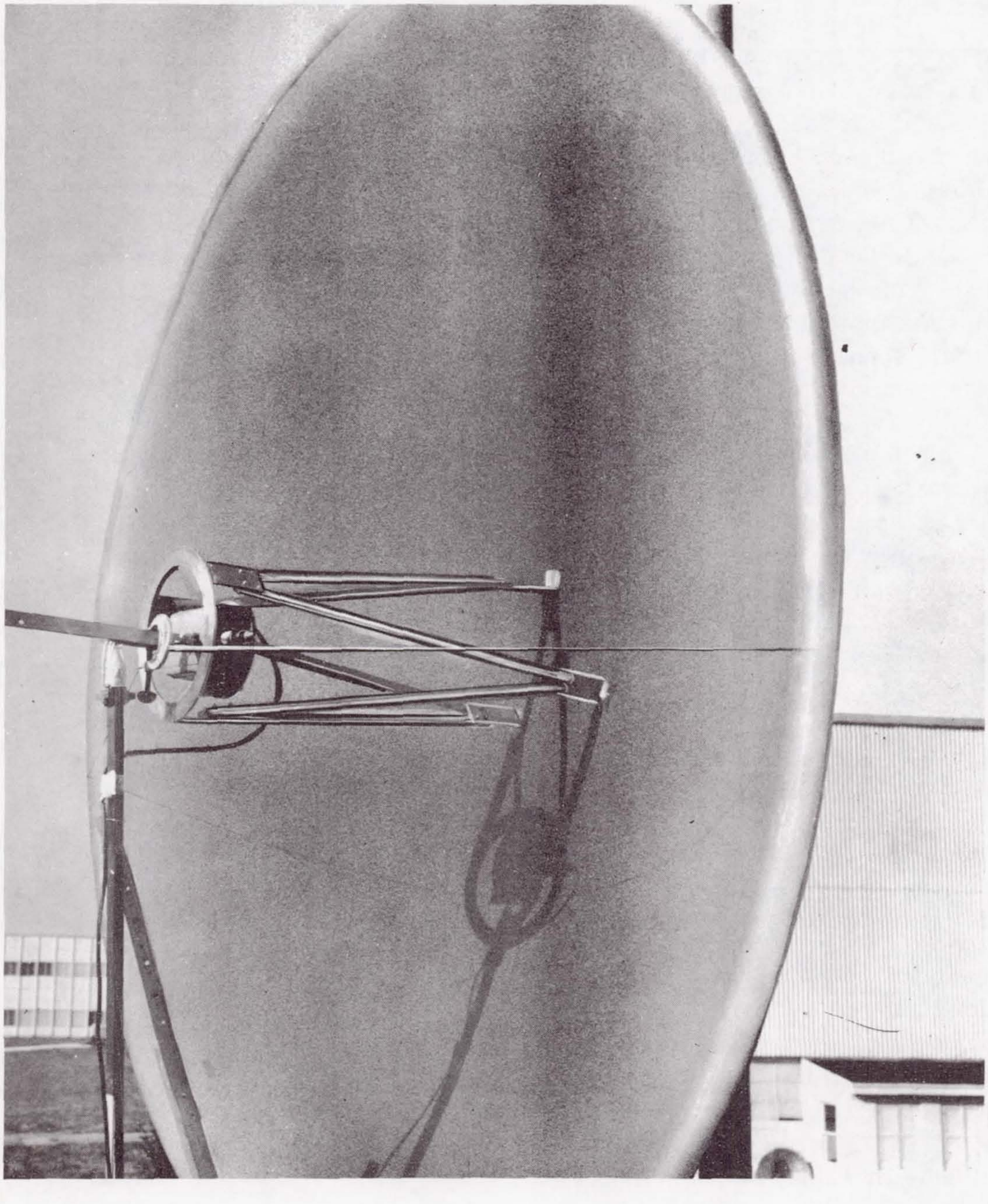

Figure A-1. Reflector with 3-V tripod.

Table A-1

Fairchild-Hiller Measurements

Frequency GHz	Gain Change (db)	Sidelobe Level (db)		Sidelobe level Change (db)	
		Right	Left	Right	Left
0.8	-2.25	-13.5	-14.5	-2	-5
2.1	-1.25	-9.0	-9.0	-1/4	+3/4
2.4	-1.00	-19.0	-15.0	+3	+6
3.4	-2.50	-14.5	-13.5	-1	+1
7.9	-1.50	-14.0	-16.5	-2.25	+.75

The General Electric Company conducted model measurements utilizing the 450-foot range at the Martin Antenna Laboratory, Baltimore, Maryland. Because with this range the antenna measurements were made in the far field, no interpretation is necessary. A 39-inch dish with a f/d ratio of 0.385 was used. The primary feed was designed for a 13-db edge illumination. Scaled measurement was made for full scale frequency of 0.8 GHz. Twenty-six antena patterns were taken with two strut configurations. Table A-2 shows the gain change, sidelobe level, right and left, and sidelobe level change, right and left, for the 3-V tripod mast and the 4-strut mast. A picture of the 3-V tripod mast is shown in Figure A-2. Figure A-3 shows the 4-strut mast.

The Lockheed Missiles and Space Company used a 24-inch diameter reflector with an f/d ratio of 0.36 and a scaling factor of 15. Measurements were made for 100-MHz full-scale frequency. Three metal rods, for equivalent full-scale diameters of 5.65 and 9.375 inches, were used to model the feed mast. The rods were positioned parallel to the axis of the reflector, running from the reflector surface past the aperture of the feed horn and touching the edge of the horn on three sides. The equivalent full-scale dimensions of the feed horn were 67.5 inches by 82.5 inches. The results are tabulated in Table A-3.

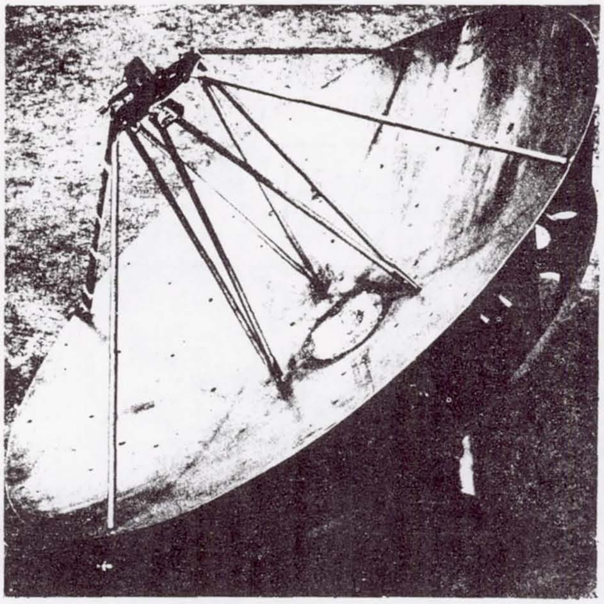

Figure A-2. 3-V tripod mast.

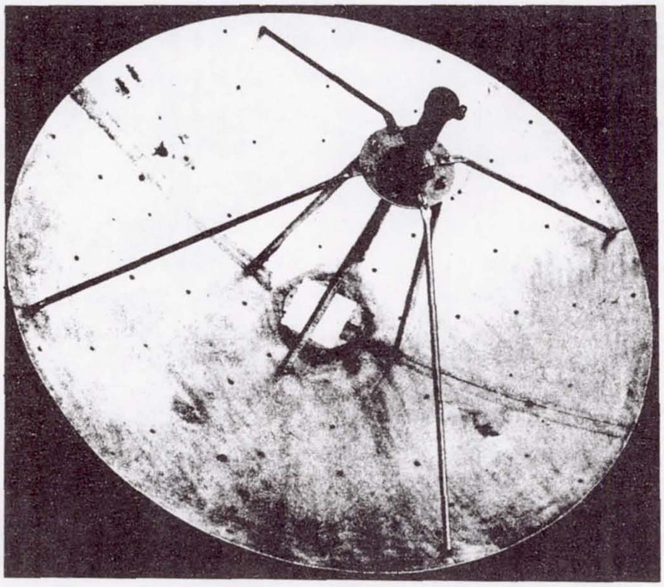

Figure A-3. 4-Strut mast.

Table A-2

General Electric Company Measurements
(Frequency 800 MHz)

	Gain Change (db)	Sidelobe Level (db)		Sidelobe Level Change (db)	
		Right	Left	Right	Left
3-V Tripod Mast	1.0	24	27.00	-2.5	0
4-Strut Mast	2.2	21	24.25	-2	-3

Table A-3

Lockheed Missiles and Space Company Measurements

Rod diameter (inches)	Gain change (db)
5.65 inches	-1
9.375 inches	-3

Measurements were made for the Goddard team by their contractor, Keltec Industries, Alexandria, Virginia, using a 6-foot diameter reflector with a f/d ration of 0.5. Sufficient distance was available on the measuring range to permit all measurements to be made in the far field of the antenna. A 4-V mast structure was employed, with equivalent full-scale dimensions for the strut diameter of 4.375 inches and strut base attachment diameter of 84 inches. Antenna pattern measurements were made to equivalent full-scale frequency of 7 GHz. A photograph of the model antenna is shown in Figure A-4.

Figure A-4. 4-V mast structure.

The data for these measurements are shown in Table A-4.

Table A-4

GSFC Measurements
(Frequency 7 GHz)

Gain Change (db) with mast added		Sidelobe Level (db) without mast		Sidelobe Level with mast (db)	
		Right	Left	Right	Left
4-V Mast	-1.2 db	-19.8	-19.8	+5.7	+5.7

Based on the data shown in Tables A-1 through A-4, it is reasonable to assume that the gain degradation which results from using a feed mast structure which can be fabricated and adjusted in the factory is about 1 db throughout the frequency range from 100 GHz (when compared to a conventional mast configuration). These data also indicate that the sidelobe degradation is in the order of 2 to 4 db at worst throughout the same frequencies.

On the basis of these findings, it was decided that the best trade-off is to accept this minor gain loss and sidelobe degradation, rather than to accept the risk of a failure of the feed mast deployment sequence.

APPENDIX B

ATTITUDE CONTROL DURING THE TRANSFER ORBIT, APOGEE MOTOR BURN, AND EARTH/POLARIS ACQUISITION

For the ATS-F&G mission, several phases of attitude control must be exercised to correspond with the phases of the mission. These phases are

1. Control in the Hohmann transfer orbit after separating from the launch vehicle

2. During apogee motor burn and vernier orbit corrections

3. Earth/Polaris acquisition

4. Operational control.

The operational attitude control mode is used after injection into a near-synchronous orbit, completion of initial orbit corrections, and deployment of parabolic antenna and solar paddles. The purpose of the analysis contained herein is to compare spin stabilization with three-axis stabilization for Phases 1, 2, and 3. The comparison is made on the basis of control system weight, power, and reliability. The reference three-axis stabilization systems chosen for comparison are the Lockheed concept which uses an auto-pilot with earth sensor reset and the Boeing Burner II stage which uses an auto-pilot without an earth sensor.

A spin stabilized vehicle attitude control system requires the use of spinup devices (spintable or motors attached to vehicle), despin devices (motors or yo-yo's) and spin axis precession control devices in order to maintain control over the spacecraft attitude. In addition for ATS-F&G, nutation control is required because of the nature of the spacecraft body during Phases 1, 2, and 3 i.e., unstable moment-of-inertia ratio and semiflexible structure. Studies have shown that both precession and nutation damping can be performed with the same propulsion system i.e., one set of thrusters is required for Phases 1 and 2 and an additional set used for Phase 3.

A three-axis stabilized vehicle of the ATS-F&G type during Phases 1, 2, and 3 requires that three sets of thrusters be used, i.e., low or intermediate thrust levels during Phase 1, high thrust level during Phase 2 and low or intermediate thrust levels during Phase 3. Note also that during Phase 3, a set of four thrusters is operated in pairs in an inverse modulation mode to eliminate rotational moments when performing orbit changes and corrections.

The proposed Lockheed system differs from the Burner II system in one major respect. Although both guidance systems used strapped down gyros with bias drifts of 1 degree per hour, they differ in their approach to cancelling this drift during the time spent in the transfer orbit. The Lockheed system uses an earth sensor and an earth pointing mode during part of the transfer orbit. During this time the gyro bias is nulled out, and just before apogee kick motor firing the system is switched to the inertial hold mode with the commanded input being the desired kick stage attitude. Burner II uses no earth sensor. For synchronous orbit injection Boeing recommends the use of a slow spacecraft roll while the control system is holding the roll axis pointing in the desired attitude for kick motor firing. Boeing claims that the slow roll would provide open loop compensation of the gyro drifts, and for 5.25 hours in the transfer orbit the accuracy at injection is comparable to the Lockheed system. The sequential order of events for all three control systems is shown below.

Spin Stabilization Sequence

1. Spinup after separation from launch vehicle
2. Active nutation damping
3. Spin-axis orientation measurement and vernier attitude correction
4. Apogee motor firing
5. Spin-axis precession
6. Initial orbit corrections
7. Despin
8. Three-axis stabilization on the sun
9. Earth acquisition maneuver
10. Polaris acquisition maneuver

Lockheed Auto-Pilot Sequence

1. Earth acquisition after separation from launch vehicle
2. Initiate course limit cycle if injection is to occur on first apogee - or initiate pitch tumble to conserve power if injection is to occur on second apogee
3. Earth reaquisition after one orbital revolution if injection is to occur at second apogee
4. Command injection attitude
5. Maintain attitude during apogee motor burn
6. Command local vertical attitude
7. Maintain attitude during vernier orbit corrections

Burner II Sequence

1. Command injection attitude
2. Initiate slow spacecraft roll to cancel effects of gyro drifts
3. Maintain attitude during apogee motor burn
4. Command local vertical attitude
5. Separate Burner II from spacecraft
6. Initiate spacecraft attitude control system
7. Maintain attitude during vernier orbit corrections

The comparison of the control system weights is given below (not including thruster hardware).

Spin Stabilization

	Component	Weight (lb)
1.	Nutation sensor	4
2.	Sun Sensors	5
3.	Star field mapper	4
4.	Body rate gyros	8
5.	Electronics	5
		26

Lockheed Three-Axis Auto-Pilot

	Component	Weight (lb)
1.	Inertial gyro package	20
2.	Earth sensor*	10
3.	Electronics	10
		40

*The earth sensor for gyro reset must operate at altitudes between 90 and 19,323 n.m while in the transfer orbit, thus a sensor optimized for use at the synchronous altitude cannot be used.

Burner II Launched ATS-F&G

(Guidance and control system - 35 lb)

While in the transfer orbit power is limited since the vehicle solar paddles are not deployed. The spin stabilized control system required 15 watts, and the Lockheed three-axis auto-pilot system required 50 watts. A Burner II launched ATS-F&G spacecraft requires no spacecraft power during the transfer orbit since it carries its own primary batteries and power regulation system. It might be noted, however, that approximately 14 pounds of batteries must be added to the standard Burner complement in order to have sufficient power for the 5.25-hour coast in the transfer orbit.

The propulsion system hardware weights are outlined below. The weights given include thrusters, plumbing, and electrical components.

Spin Stabilization System

	Component	Weight (lb)
1.	Ammonia Resistance Jet System	
	4 multi-jet thrusters	6
	Power conditioning	10
	Storage and feed system	30
		46
2.	Monopropellant Hydrazine System	
	2 5-lb_f thrusters	3.5
	Storage and feed system	19.1
	Nitrogen gas for pressurization	.23
		22.83
3.	Spin and Despin Devices	
	4 spin rockets	13.52
	2 yo-yo's	13.70
		27.22
	Total hardware weight	96.05

Lockheed Three-Axis Auto-Pilot System

	Component	Weight (lb)
1.	Ammonia Resistance Jet System	
	6 thrusters	6
	Power conditioning	15
	Storage and feed system	34
		55

2. Monopropellant Hydrazine (N_2H_4) System

a. Initial acquisition and injection

4 60-lb_f thrusters	24.0
4 1-lb_f thrusters	6.0
Feed system	7.0
	37.0

b. Vernier orbit correction and adjust

4 1-lb_f thrusters	6.0
Storage and feed system	29.0
	35.0

Total N_2H_4 System weight	72.0
Total Prop. Hardware weight	127.0

For a Burner II launched spacecraft, propulsion for Phases 1 and 2, except vernier orbit corrections, is supplied by the systems carried on the Burner II stage. Propulsion for Phases 3 and 4 and vernier orbit corrections would be supplied by spacecraft systems such as items 1 and 2-b in the Lockheed approach. The Burner II supplied propulsion system consists of four hydrogen peroxide thrusters for pitch and yaw control during the firing of the kickmotor and eight cold-gas nitrogen thrusters for roll control and pitch and yaw control during coast. For a mission such as ATS-F&G which requires a long coast and has a spacecraft of the ATS-F&G form, the Burner II propulsion systems have to be modified in order to achieve the mission; i.e., the thrust levels increased and the propellent loading increased.

The propulsion system dry weight for such a mission is 61 pounds. Assuming the spacecraft supplied propulsion to be items 1 and 2-b of the Lockheed approach, then the system dry weight breaks down as follows:

Burner II	61 lb
Resistance jet system	55 lb
Monopropellant hydrazine system	35 lb
	151 lb

The comparison of total fuel required by Phases 1, 2, and 3 is given below. Two cases are given for spin stabilized control. Case I corresponds to ± 1.5 out-of-plane and ± 0.5 in-plane spin-axis orientation errors. These are the errors obtained using the sun sensor/RF polarization method used on ATS-A through E. Case II corresponds to ±0.2 out-of-plane and ±0.2 in-plane errors, which are the errors expected with the star field mapper.

Spin Stabilization System

Case I

Total propellant weight - 97 lb

Case II

Total propellant weight - 67 lb

Lockheed Three-Axis Auto-Pilot System

Total propellant weight - 63 lb

Burner II Launched ATS-F&G

A summary of the weight and power trade-off is given below.

		Spin	Lockheed Auto-Pilot	Burner II
1.	Spacecraft Power (watts)	15	50	0
2.	Weight (lb)			
a.	Sensors and Electronics	26	40	35
b.	Thrusters	96	127	151
c.	Total fuel	67 to 97*	63**	67**
d.	Primary batteries	0	0	21
		189 to 219	230	274

*Case I - 67 lbs; Case II - 97 lbs (worst case spin-axis orientation)

**Assumes that spacecraft is injected on the first apogee and includes enough fuel to place the spacecraft on station and remove injection errors (see Table B-1 for injection errors).

It should be noted, however, that the Burner II stage in its present form has never been used for a synchronous injection and would require modifications for use with ATS-F&G. On the basis of modifications Burner II cannot be considered an "off-the-shelf" flight proven system. The modifications required by Burner II for use with ATS-F&G are as follows:

1. New payload support structure
2. Increase size of hydrogen peroxide tanks
3. Increase size of nitrogen tanks
4. Upgrade attitude control thrust level
5. New Intervalometer (timer) to allow 6-hour coast
6. Additional batteries.

The effect of these modifications has been considered in generating the calculations presented above.

In summary, the weight of the Burner II control system (primary batteries, electronics and propulsion) is 274 pounds compared to 230 pounds for the Lockheed system. The studies previously exhibited indicate that the weight reduction resulting from elimination of the earth sensor on Burner II is insignificant when compared with the extra weight required for primary batteries and attitude control systems that are not optimized for the specific mission. Also the reliance on open loop gyro drift compensation constrains the injection to occur at the first apogee. The open loop compensation technique is more sensitive to larger than nominal errors than a system using a closed loop earth reference gyro nulling technique. It should be noted also that the significant increase in total system weight for Burner II over the Lockheed concept is based on the higher specific impulse of the propellant (hydrazine over peroxide), use of primary batteries, and the fact that the Lockheed concept presents an integrated system approach. Thus, the comparisons performed indicate that there is no advantage in using Burner II in lieu of the Lockheed system.

Based on system weight, power, and reliability, it is concluded that the spin stabilized transfer orbit control is preferred. The weight advantage of spin stabilization is 57 pounds. The power advantage of spin stabilization is 35 watts. The reliability of the spin stabilized mode is greater because fewer in-line thrusters are required for successful operation. The three-axis control mode requires successful operation of four 60-pound and four 1-pound hydrazine thrusters and no redundancy is provided. The spin stabilized mode only requires one 5-pound hydrazine thruster; two are provided for redundancy.

Table B-1

ATS-4 Injection Errors

I. Spinning Injection

Sun sensor/polarization measurement system

$e = 0.0210$	ΔV_{in}	=	107 fps
$i = \pm 0.7065$	ΔV_{out}	=	124 fps
	ΔV_{total}	=	231 fps

Star field mapper system

$e = 0.0174$	ΔV_{in}	=	88 fps
$i = \pm 0.233$	ΔV_{out}	=	41 fps
	ΔV_{total}	=	129 fps

II. Lockheed Concept*

$\Delta V_{total} = 120$ fps

III. Burner II

$e = 0.017$	ΔV_{in}	=	86 fps
$i = \pm 0.33°$	ΔV_{out}	=	58 fps
	ΔV_{total}	=	144 fps

*Final study report does not resolve the errors into in-plane and out-of-plane components.

The effect of these modifications has been considered in generating the calculations presented above.

APPENDIX C

CLOSED LOOP USE OF THE INTERFEROMETER

The resolution of roll, pitch and yaw errors from knowledge of the direction cosines of two line-of-sight vectors from two ground stations involves 42 multiply and 18 add/subtract operations per error update. This assumes that the direction cosines of the L.O.S. vectors in the orbit reference frame are computed on the ground and then stored in a spacecraft memory, with the direction cosines of the L.O.S. vectors in body coordinates available as outputs from the interferometer.

Since the interferometer is an experiment in attitude-sensing independent of the 30-foot dish, the two experiments should be as isolated as possible. Therefore, the interferometer will not be used as the primary attitude reference for the evaluation and use of the 30-foot dish. The attitude sensing experiment involves telemetry of the basic RF phase measurements from a fine and a coarse baseline; therefore, for closed-loop operation of the attitude control system using the interferometer, the attitude errors will be determined on the ground. For static pointing, an error update every 5 seconds in roll and pitch and every 10 seconds in yaw is adequate, in view of the low control system bandwidth required. This experiment will investigate static pointing to any point on the earth.

For the special case where the yaw angle is independently controlled and where pointing the yaw axis at the ground station is desirable, the interferometer phase errors are directly related to the roll and pitch errors, and no error resolution is required. The special mode will provide closed-loop onboard attitude control, and will be designed to provide the ultra-precise (± 0.02 degree) static pointing required for some of the experiments planned to measure some of the characteristics of the 30-foot dish.

Lockheed proposed the use of the interferometer for the ± 0.1 degree attitude control mode, whereas the Goddard study and the two other industrial contractors propose a non-RF technique, using a Polaris sensor and an earth sensor. The Lockheed approach makes one experiment dependent on another and is therefore rejected.

APPENDIX D
LINK CALCULATIONS FOR TRACKING,
TELEMETRY, AND COMMAND SUBSYSTEMS

The contents of this appendix are detailed link calculations pertaining to tracking, telemetry, and command. Some parameters such as bandwidth have been chosen for their consistency with orbit dynamics, bit rates, etc. Other parameters such as antenna gain are fixed by the existing antenna parameters, or in the case of the spacecraft, by anticipated performance. The parameters and their justifications are indicated in this appendix under the appropriate subsections.

COMMAND

Command calculations in Table D-1 are based on a ground transmitter capability of 10 kilowatts minimum at S-band. Ground antenna gain specification in the frequency range of 1760 to 1850 MHz is 41.5 db. A polarization loss of -3 to -5.5 db has been assumed. The maximum deviation of 2.5 db for the transmit axial ratio of the polarization ellipses is taken as a very pessimistic tolerance. The nominal value of 3 db for polarization loss must be taken into account for the linearly polarized spacecraft S-band antenna. Path loss is assumed for the maximum range (synchronous altitude to farthest visible station).

The spacecraft antenna gain has been given as -10 db relative to isotropic over more than 90 percent of the sphere. Over 10 percent of the sphere antenna gain can be less than -10 db, but this 10 percent is not contained within a continuous 10-percent segment of the sphere. Instead it will be distributed in relatively narrow nulls over large portions of the sphere. Because of the assumption of pessimistic tolerances for the other minimum conditions and because of the relatively low probability of antenna nulls, it is not considered feasible or practical to accommodate these nulls by increased powers or gains.

Spacecraft cable loss is assumed to be 2 db for the lengthy runs of cable between antenna and receiver. Because of necessary redundancy, two S-band receivers are operated simultaneously with their inputs connected by a power splitter. This 3-db loss is shown as the spacecraft power splitter loss of 3 db. Unless another S-band antenna system is included in the design (not considered justifiable at this time), the passive power splitter is a necessary part of the system.

Table D-1

S-Band Uplink Command/Ranging			
	Minimum	Nominal	Maximum
Ground transmit power	+40 dbw	+40 dbw	+40 dbw
Ground antenna gain (30-foot)	+41 db	+41 db	+42 db
Effective radiated power	+81 dbw	+81 dbw	+82 dbw
Path-loss (1800 MHz)	-189 db	-189 db	-149 db
Polarization loss	-5.5 db	-4 db	-3 db
Spacecraft antenna gain	-10 db	-3 db	-1 db
Spacecraft cable loss	-2 db	-2 db	-2 db
Diplexer loss	-0.5 db	-0.5 db	-0.4 db
Spacecraft power splitter	-3 db	-3 db	-3 db
Total received power (A)	-129 dbw	-120.5 dbw	-76.4 dbw
Noise figure	+7 db	+7 db	+7 db
IF noise power (BW-400 kHz) (B)	-141 dbw	-141 dbw	-141 dbw
Ratio $\frac{(A)}{(B)}$	+12.0 db	+20.5 db	+64.6 db
Ratio required (discriminator threshold)	+6 db	+6 db	+6 db
Minimum power required for transponder operation	-135 dbw	-135 dbw	-135 dbw
Signal margin	+6 db	+14.5 db	+58.6 db

The spacecraft IF bandwidth of 400 kHz covers the ±100 kHz maximum range tones on the range and range rate up-link and the maximum expected Doppler of ±85 kHz. With a spacecraft noise figure of 7 db, available noise power in the spacecraft IF bandwidth is calculated to be -141 dbw. A threshold is set in the spacecraft IF bandwidth to enable turn on of the S-band transmitter for ranging, telemetry, and polarization angle tracking. (A VHF beacon remains on continuously during the transfer and synchronous orbits until station acquisition, at which time it will be commanded off.) This S-band IF threshold is established at a received power level of -135 dbw, providing a +6-db discriminator margin which gives a 10^{-10} or better command error rate. The mechanics of turn on are a decrease in IF noise in a narrow bandwidth positioned slightly above the maximum signal frequency expected. As signal power is introduced into the IF bandwidth, the noise (limiter action) is suppressed. Reliable, non-random operation of this type of turn on device is assured for signal power levels several times greater than the IF noise power. This level has been set at -135 dbw (6 db above the noise power). The level can be reduced somewhat at the expense of more likely accidental turn on of the S-band transmitter.

For suitable discriminator and command decoder operation, the signal-to-noise ratio at the input to the discriminator (signal-to-noise ratio in the IF bandwidth of 400 kHz) would be greater than 6 db. For weak signals compared to the noise, the signal output of the discriminator is proportional to the square of the input signal-to-noise ratio and thus behaves like a half-wave AM detector. For large signal-to-noise ratios, the discriminator tends to suppress the noise in the vicinity of the carrier in the low-pass filter output and improves signal-to-noise even beyond the ratio of the IF filter bandwidth to low-pass filter bandwidth preceding the command decoder (400 kHz/2 kHz = 200). The additional improvement above the 23 db ratio of bandwidths is proportional to the square of the modulation index B. Little improvement is obtained since B has been set to a relatively low value of 1.2 in the command transmitter. This restricts the command frequency spectrum to occupy a narrow portion of the total spectrum available. If command and ranging occur simultaneously, the command will be retransmitted on the down-link subcarrier along with the range tones. Refer to Section VII H (TM & Command) for additional details. The command spectrum, for simultaneous use with range and range rate, should lie below 4 kHz to avoid appearing in the range tone spectrum and should lie above 100 Hz to avoid appearing in the subcarrier tracking loop. The

100 Hz subcarrier tracking bandwidth has been chosen to cover maximum expected dynamics of the transfer orbit. The command bit rate has been set by the project at 128 bits per second. A subcarrier command frequency will be chosen which is acceptable for phase modulating the command carrier to circumvent the problem of command and ranging self-interference.

Table D-1 is based upon sequential use of ranging and commanding. With simultaneous use and a power division of one half between command and range and range rate, the signal margins must be decreased by 3 db. System design will incorporate facility of simultaneous usage since nominal conditions allow it. In critical operations such as commanding apogee motor firing, maximum transmitter power will be available to the command function.

RANGE AND RANGE RATE

Table D-1 also applies to the range and range rate uplink. In the minimum case condition (-129 dbw signal power available at the spacecraft receiver input) the signal-to-noise ratios out of the transponder for the downlink transmission have been calculated for the carrier, subcarrier, major tone, and minor tone. The ratios are as follows:

Carrier	S/N--+42.6 db (200 Hz maximum tracking BW, two sided)
Subcarrier	S/N--+35.0 db (200 Hz maximum tracking BW, two sided)
Major tone	S/N--+49.4 db (2 Hz maximum tracking BW, two sided)
Minor tone	S/N--+38.0 db (2 Hz maximum tracking BW, two sided)

These are the limits of maximum signal-to-noise ratio at the ground under worst conditions for the tracking bandwidths listed. It is apparent even in the worst case that a retransmission of the range tones originally at a -129 dbw spacecraft receiver input level shows no degradation in range and range rate accuracy compared to the down-link condition (Table D-2).

Table D-2 considers the case of the sole use of the down-link by one station for range and range rate. Polarization tracking will not affect the calculated signal-to-noise ratios since it operates only with the down-link carrier, and polarization tracking utilizes a separate antenna feed network. Tolerance of cable, diplexer, etc., has been discussed previously.

Table D-2

S-Band Range and Range Rate Down Link			
(One Station Alone)	Minimum	Nominal	Maximum
S/C transmit power (dbw)	0	0	0
S/C transmitter degradation (db)	-1	-0-	+1
S/C cable loss (db)	-2	-2	-2
S/C diplexer loss (db)	-0.5	-0.5	-0.4
S/C antenna gain (db)	-10	-3	-1
Effective radiated power (dbw)	-13.5	-5.5	-2.4
Path loss (db) 2250 MHz	-191	-191	-151
Ground antenna gain (30-ft) (db)	+41	+41	+42
Total received power (dbw)	-163.5	-155.5	-111.4
Received carrier power (dbw)	-165.9	-157.9	-113.8
Received subcarrier power (dbw)	-173.5	-165.5	-121.4
Received major tone power (dbw)	-179.1	-171.1	-127.0
Received minor tone power (dbw)	-190.5	-182.5	-138.4
System noise temperature (^{O}K)	100	100	100
System receiver noise Density $\frac{(dbw)}{(Hz)}$	-208.6	-208.6	-208.6
Carrier tracking S/N (db)	(200 Hz) +19.7	(20 Hz) +37.7	(20 Hz) +81.8
Subcarrier tracking S/N (db)	(200 Hz) +12.1	(20 Hz) +30.1	(20 Hz) +74.2
Major tone tracking S/N (db)	(2 Hz) +26.5	(0.2 Hz) +44.5	(0.2 Hz) +88.6
Minor tone tracking S/N (db)	(2 Hz) +15.1	(0.2 Hz) +33.1	(0.2 Hz) +77.2

The process of phase modulation in the spacecraft and on the ground is such that the following division of total power occurs in the down-link:

1. Carrier power 2.4 db below total available power
2. Subcarrier power 10 db below total available power
3. Major tone power 15.6 db below total available power
4. Minor tone power 27 db below total available power.

These values were used in Table D-2 to compute the tracking signal-to-noise ratios.

Carrier and subcarrier tracking bandwidths under worst case conditions are taken as 100 Hz. The setting of this third-order tracking loop bandwidth is determined by the expected rate of change of the Doppler frequency. ADCOM, Inc., in work sponsored by NASA/GSFC, has calculated the response of various third-order loops to frequency ramps described in terms of the two-sided noise bandwidth (B_n). Figure D-1 taken from the ADCOM final report and shows the phase error as a function of B_n and the time following the beginning of the ramp. It is apparent that neither the third-order loops nor the second-order loop are capable of minimizing the transient. The Mallinckrodt third-order loop is that loop presently implemented in the field and is the most useful in terms of loop stability. If one assumes a one-second time to allow the transient to subside, for a phase error of 1/2 radian (adequate to keep the loop locked), $B_n = 12$ Hz and the frequency ramp causing the phase error = 144 rad/sec^2 = 23 cycles/sec^2.

An antenna horizon for low spacecraft altitudes where maximum rate of change of frequency occurs, it is possible to obtain frequency ramps (at 2250 MHz) on the order of 8000 cycles/sec^2.

Figure D-2 shows the phase error for a Mallinckrodt third-order loop for a $B_n = 1000$ Hz. It is apparent that from time of .01 seconds to 500 seconds, the phase error is well below 1/2 radian in response to a frequency ramp of 10^6 rad/sec^2. It is anticipated that a frequency ramp of this magnitude will not occur for any duration, so that the increase of phase error for large time is not significant. The increase of phase error for large time is the result of imperfect integration in the loop.

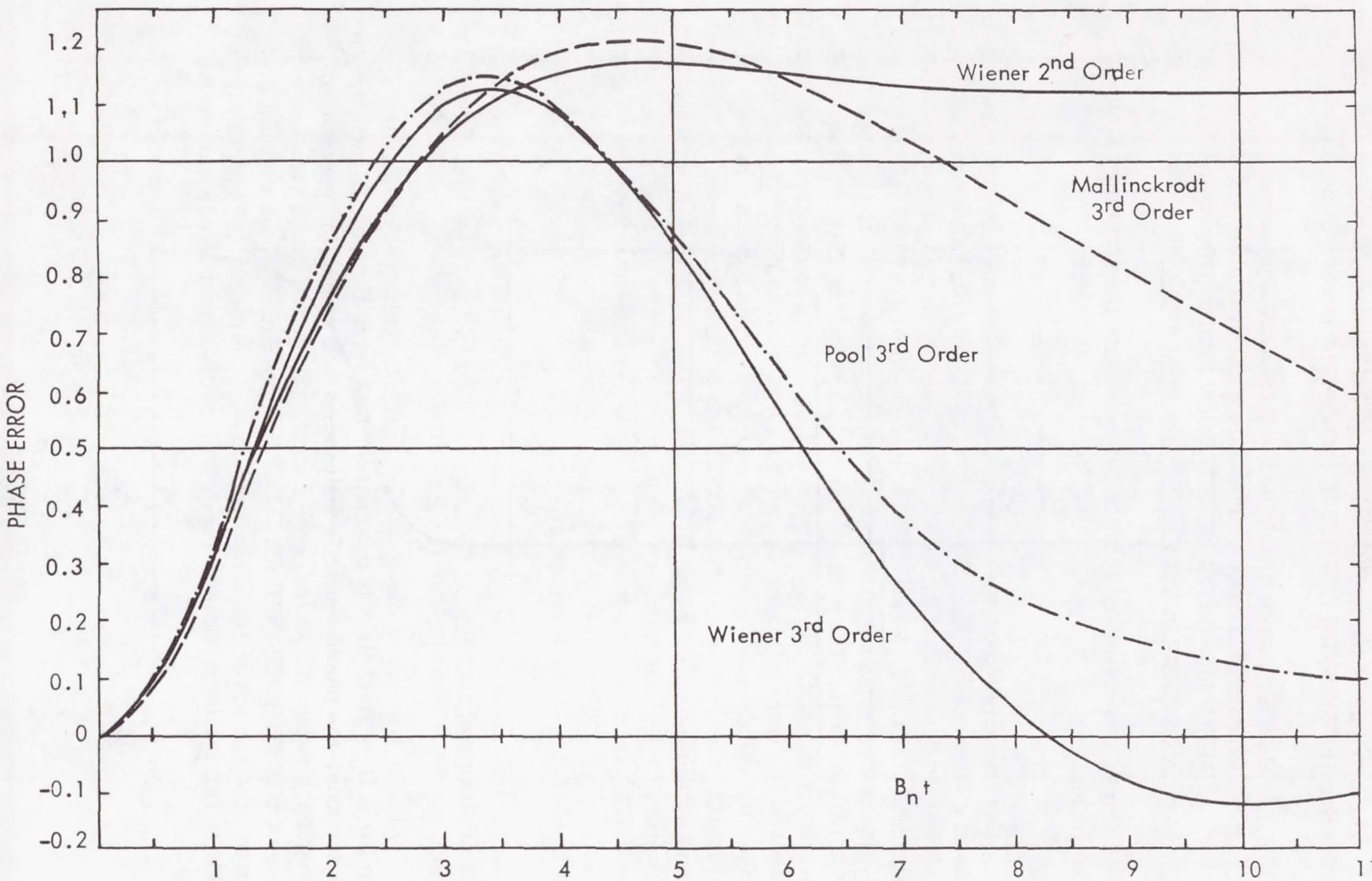

Figure D-1. Error response to a frequency ramp $r = B_n^{\,2}$ rad/sec^2, B_n in Hz.

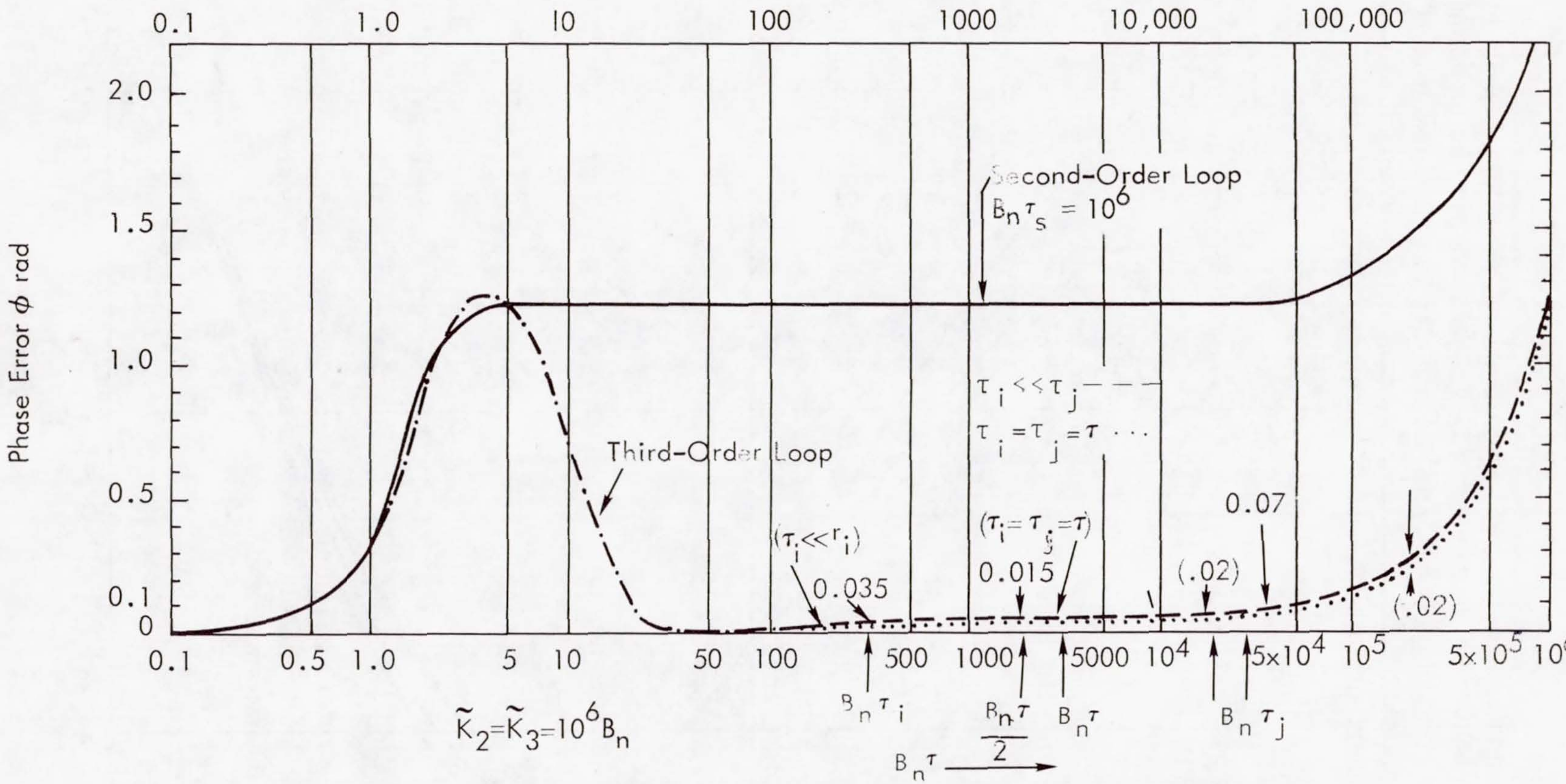

Figure D-2. Normalized error response to frequency ramp, $r = B_n^2$ rad/sec^2.

A compromise setting of 100 Hz tracking bandwidth (200 Hz two-sided noise bandwidth) has been decided upon to prevent an inclusion of the telemetry data sidebands in the tracking loop (located at odd multiples of 400 Hz) and yet maintain a capability to respond with negligible phase error to frequency ramps of $4(10^4)$ rad/sec^2.

TELEMETRY

Table D-3 covers link calculations for both S-band telemetry and the redundant VHF telemetry. Major differences between the two frequency bands are

1. Spacecraft cable loss is lower at VHF
2. No VHF diplexer is necessary
3. Noise figure of the VHF system is 4 db while the S-band system noise temperature is 100^oK
4. Carrier-tracking bandwidth for VHF is 10 times smaller than S-band.

Up to this point no mention has been made of the PCM format. It is undesirable to place the telemetry spectrum within the down-link subcarrier range and range rate tracking-loop bandwidth (100 Hz). This requires putting the telemetry on a down-link subcarrier to remove it from the range rate subcarrier tracking bandwidth. This, in essence, is just what the process of split phase modulation does; it moves the telemetry baseband spectrum to the bit rate frequency, and the bit rate frequency becomes the telemetry subcarrier frequency. In addition, split phase, because one transition occurs per bit, allows easier synchronization of data handling equipment on the ground.

Setting of the modulation index resulted from the following consideration. In square-wave phase modulation (the case considered here), the power in the carrier component passes from a maximum at a one-sided phase excursion of 0 radians ($\beta = 0$) to zero at a one-sided phase excursion of $\pi/2$ radians ($\beta = \pi / z$). The sideband power varies as the square of a sinusoid whose argument is the one-sided phase excursion, the modulation index, B.

Table D-3

Telemetry Link Calculations												
VHF TM							S-Band TM					
400 bps			4000 bps				400 bps			4000 bps		
Min.	Nom.	Max.	Min.	Nom.	Max.		Min.	Nom.	Max.	Min.	Nom.	Max.
0	0	0	0	0	0	S/C transmit power (dbw)	0	0	0	0	0	0
-1	0	+1	-1	0	+1	S/C transmitter degradation (db)	-1	0	+1	-1	0	+1
-1	-1	-1	-1	-1	-1	S/C cable loss (db)	-2	-2	-2	-2	-2	-2
--	--	--	--	--	--	S/C diplexer loss (db)	-0.5	-0.5	-0.4	-0.5	-0.5	-0.4
-10	-3	-1	-3	-1	-1	S/C antenna gain (db)	-10	-3	-1	-3	-1	-1
-12	-4	-1	-5	-2	-1	Effective radiated power (dbw)	-13.5	-5.5	-2.4	-6.5	-3.5	-2.4
-167	-161	-127	-167	-167	-167	Space loss (db)	-191	-185	-151	-191	-191	-191
+20	+21	+22	+20	+21	+22	Ground antenna gain (30-ft) (db)	+41	+41	+42	+41	+41	+42
-159	-144	-106	-152	-148	-146	Total received power (dbw)	-163.5	-149.5	-111.4	-156.5	-153.5	-151.4
-169	-154	-116	-162	-158	-156	Received carrier power (PCM/PM, β=1.2)	-173.5	-159.5	-121.4	-166.5	-163.5	-161.4
-159.4	-144.4	-106.4	-152.4	-148.4	-146.4	Received data power (dbw)	-163.9	-149.9	-111.8	-156.9	-153.9	-151.8
-200	-200	-200	-200	-200	-200	Receiver noise density (dbw/Hz) NF=4 db for VHF T_e=100°K for S-band	-208.6	-208.6	-208.6	-208.6	-208.6	-208.6
-174	-174	-174	-164	-164	-164	Noise in detection bandwidths = 1/2 x bit rate x 2-sided (dbw)	-182.6	-182.6	-182.6	-172.6	-172.6	-172.6
+14.6	+29.6	+67.6	+11.6	+15.6	+17.6	Telemetry data S/N (db)	+18.7	+32.7	+70.8	+15.7	+18.7	+20.8
-187	-187	-187	-187	-187	-187	Noise in carrier tracking bandwidth (dbw) 20 Hz (2-sided) 200 Hz	-185.6	-185.6	-185.6	-185.6	-185.6	-185.6
+18	+33	+71	+25	+29	+31	Telemetry carrier S/N (db)	+12.1	+26.1	+64.2	+19.1	+22.1	+24.2
+12	+12	+12	+12	+12	+12	Minimum telemetry data S/N required, $P_e = 10^{-5}$ (db)	+12	+12	+12	+12	+12	+12
+2.6	+17.6	+55.6	-0.4	+3.6	+5.6	Telemetry data S/N margin (db)	+6.7	+20.7	+58.8	+3.7	+6.7	+8.8

The close relationship of square-wave phase modulation to amplitude modulation (the sideband voltage components maintain a sin x/y relationship) results from the phase angle produced by the square waves modulation remaining stationary with respect to the carrier over each half period of the square wave.

B has been set at 1.2 radians to retain a carrier component 10 db below the sideband power. This level is adequate for phase-lock loop tracking as the calculations in Table D-3 indicate.

POLARIZATION ANGLE MEASUREMENT

Table D-4 contains the signal-to-noise calculations for the polarization angle measurement. A 2-db system noise figure is suitable for this system and the calculations do not warrant the inclusion of two additional cooled parametric amplifiers in the system design.

Polarization angle measurement and range and range rate measurement must occur together to obtain attitude as a function of position in the orbit. Hence, the polarization measurement uses the highest powered component of the down-link range and range rate spectrum, the carrier. The carrier is 2.4 db below the total received power.

MINITRACK

Minitrack link calculations given in Table D-5 are based upon a 1-watt VHF spacecraft beacon. The VHF antenna system, because of its status as a backup system, is not expected to have better omnidirectional antenna coverage than the S-band antenna system. Table VII-4 of Section VII D (Spacecraft Transponder-Tracking, Telemetry and Command) shows that the maximum gain attainable when two antenna elements are used to achieve maximum coverage is approximately -1 db except in the interference region where it may locally exceed this magnitude. This low value of the antenna gain is a direct result of the equal division of the transmitter power between the two antenna elements. Approximately the same quantities and percentages have been assumed for the VHF antenna system:

-10 db relative to isotropic for better than 90 percent of sphere
- 6 db relative to isotropic for better than 50 percent of sphere
0 db relative to isotropic for better than 10 percent of sphere.

Table D-4

S-Band Polarization Angle Measurement			
	Minimum	Nominal	Maximum
S/C transmit power (dbw)	0	0	0
S/C transmitter degradation (db)	-1	0	+1
S/C cable loss (db)	-2	-2	-2
S/C diplexer loss (db)	-0.5	-0.5	-0.4
S/C antenna gain (db)	-10	-3	-1
Effective radiated power (dbw)	-13.5	-5.5	-2.4
Path loss (db)	-191	-191	-151
Ground antenna gain (30-foot) (db)	+41	+41	+42
Total received power (dbw)	-163.5	-155.5	-111.4
Received carrier power (dbw)	-165.9	-157.9	-113.8
Polarization system Noise figure (db)	+2	+2	+2
System receiver noise density $\frac{(dbw)}{(Hz)}$	-202	-202	-202
Carrier tracking bandwidth*(Hz) (2-sided)	1	1	1
S/N, polarization angle measurement (db)	+36.1	+44.1	+88.2

*Considered an adequate tracking bandwidth for major portions of the transfer ellipse, from ATS-B experience. Further study will determine optimum tracking bandwidth in terms of oscillator stability and drift rates.

Table D-5

Minitrack - 136 MHz			
	Minimum	Nominal	Maximum
Spacecraft transmit power (1 watt)	-1 dbw	0 dbw	+1 dbw
Spacecraft antenna gain	-10 db	-6 db	0 db
Spacecraft antenna cable loss	-1 db	-1 db	-1 db
Effective radiated power	-12 dbw	-7 dbw	0 dbw
Path loss	21,000 n. m. -167 db	10,500 n. m. -161 db	210 n. m. -127 db
Ground antenna gain (fine)	+11 db	+12 db	+13 db
Ground antenna gain (ambiguity)	+6 db	+7 db	+8 db
Received signal power Fine antenna Ambiguity antenna	 -168 dbw -173 dbw	 -156 dbw -161 dbw	 -114 dbw -119 dbw
Minitrack system Digitized data	Accurate to 20 seconds of arc (best case at zenith) for signal levels greater than -150 dbw (fine antennas) and -155 dbw (ambiguity antennas).		
Manual reading	Signal levels greater than -155 dbw (fine antennas) and -160 dbw (ambiguity antennas).		

Spacecraft cable loss of 1 db results from the lengthy cable runs necessary from the transmitter to the antenna.

Path loss will vary during the transfer orbit over a 40-db range with maximum at approximately 21,000 nautical miles and minimum at approximately 210 nautical miles. It is apparent that the Minitrack system will not give range information out to synchronous altitudes. However, the Minitrack transmitter is included as part of the spacecraft equipment for the following reasons:

1. Source of transmitter redundancy for telemetry (at VHF)
2. Source of data for range calculation for large segments of the transfer orbit
3. Beacon for S-band acquisition aid
4. Source of telemetry information from Fort Myers, Florida station during lift-off.

APPENDIX E

COMMUNICATIONS EXPERIMENTS AND DEMONSTRATION CONSIDERATIONS

INTRODUCTION

This section is both an outline of specific demonstration-type communications experiments to be considered for the ATS-F&G program and a survey of potential applications of ATS-F&G high-gain spacecraft antenna technology Under the first heading, such systems as television relay to small ground terminals, multiple-access "man-pack" satellite communications, experimental weather forecast/facsimile broadcast, aircraft-satellite communications, and satellite-satellite data acquisition links have been analyzed Under the second heading, certain advanced systems such as TV and FM direct broadcast, libration point communications to the hidden side of the moon, and deep-space applications of ATS-F&G large aperture antenna technology have been looked at briefly to determine channel capacities, power requirements, and other link parameters. While essentially a revision, summary, and extension of Section 20-A of the GSFC study report (Reference A), it draws on and provides critiques of those areas of the three contractor studies (Reference B, C, and D) dealing with ATS-F&G communications possibilities.

In this study specific ATS-F&G communications frequency assignments of 401 MHz (up-link) and 466 MHz (down-link) at UHF, and 8.0 GHz (up-link) and 7.3 GHz (down-link) at X-band, have been assumed. This choice fairly well encompasses the range of frequencies at which it is technically feasible to operate a large aperture, three-axis stabilized parabolic reflector in synchronous orbit Below UHF, apparent antenna gain (assuming constant radiation efficiency) falls off rapidly to the point where the large dish has no advantage over smaller, cheaper, and simpler structures. Above X-band, the surface tolerances attainable with a large light-weight structure in a thermal environment must be expected to degrade the radiation efficiency of the antenna to the point where smaller and highly rigid structures would be more effective.

In addition to the specific frequencies identified above, hypothetical communications capabilities at 860 MHz, 1.5 GHz, 1.7 GHz, and 2.3 GHz are also considered here in connection with certain potential ATS-F&G applications which are not compatible with the specific UHF and X-band assignments assumed. The eventual choice of communications frequencies for future ATS-F&G

programs will, of course, be dictated by the experiments and demonstrations selected. While it is obvious that no single spacecraft would be designed to handle all the frequencies considered here, it is quite reasonable to assume that two different spacecrafts (i. e., ATS-F and ATS-G) could be designed to cover most of them.

ANTENNA AND TRANSPONDER CHARACTERISTICS AND FUNDAMENTAL LINK PARAMETERS

Assuming nominal deployment of the antenna and successful three-axis stabilization of the ATS-F&G spacecraft, the anticipated electrical characteristics of the large aperture parabolic reflector (see Section VII B) at the frequencies of interest are summarized in Table E-1. The data given, particu-

Table E-1

Electrical Characteristics of ATS-F&G Large Aperture Antenna

Operating Frequency	Apparent Gain		Beamwidth	
	50% efficiency (db)	Anticipated (including feed losses) (db)	50% efficiency (degrees)	Anticipated (degrees)
401 MHz	28.7	27.0	5.6	6.0
466 MHz	30.0	28.6	4.8	5.0
860 MHz	35.4	33.5	2.5	2.6
1.5 GHz	40.2	37.7	1.4	1.5
1.7 GHz	41.3	38.6	1.3	1.4
2.3 GHz	43.8	41.2	0.94	1.1
7.3 GHz	53.9	49.0	0.31	0.35
8.0 GHz	54.7	50.0	0.28	0.32

larly antenna gains and beamwidths at X-band, reflect the multiple-mission nature of the ATS-F&G spacecraft and are somewhat poorer than those which could be achieved in a system optimized for operation at a specific frequency. In such "dedicated" systems, it does not seem unreasonable to postulate achieving antenna efficiencies of 50 percent. For reference purposes, Table E-1 also shows the gains and beamwidths which would be obtained with 50 percent efficiency.

At a synchronous orbit altitude of 36.3×10^3 km and a minimum earth elevation angle of 5 degrees, the maximum earth-spacecraft slant range is 41.1×10^3 km The resulting maximum free space losses at the several operating frequencies are given in Table E-2, together with approximate diameters of main-beam earth coverage at the sub-satellite point, computed from the anticipated 3-db beamwidths of Table E-1

Table E-2

Spacecraft-to-earth Path Losses and Approximate Earth Coverages

Operating Frequency	Free Space Loss (db)	Diameter of Earth Coverage (km)
401 MHz	-176.9	3700
466 MHz	-178.2	3160
860 MHz	-183.2	1640
1.5 GHz	-188.1	940
1.7 GHz	-189.1	930
2.3 GHz	-193.0	730
7.3 GHz	-202.1	220
8.0 GHz	-202.9	180

If the antenna is grossly defocused at UHF, global coverage could be achieved at a cost of about 10 db in apparent antenna gain. For world-wide data acquisition systems, this type of sub-optimum operation would have certain advantages. However, since beam-steering is available to center the main-lobe of the undefocused beam anywhere on the surface of the earth, defocusing with its concomitant penalty of 10 db in system channel capacity would not appear to have any particular merit in an assessment of ATS-F&G UHF capabilities. Therefore, defocused UHF operation will not be considered further in this study.

All ATS-F&G communications operations discussed here presuppose inclusion in the spacecraft of some type of frequency translating transponder. For the specific UHF and X-band cases assumed, detailed descriptions of proposed transponder designs are given in Section VII. This section also discusses briefly possible spacecraft subsystems for communications at the several hypothetical frequencies assumed; i.e., 860 MHz, 1.7 GHz, and 2.3 GHz. Table E-3 below summarizes the specific spacecraft transponder characteristics for the communications subsystems described in detail in Section VII-D.

Table E-3

Summary of Assumed Transponder Characteristics

Communications Subsystems	Receive Frequency	Transmit Frequency	Maximum 3-db Bandwidth (MHz)	Receiver Noise Figure (db)	Maximum CW Power Output (watts)
UHF	401 MHz	466 MHz	1.0	3	40
X-band	8.0 GHz	7.3 GHz	40	7	24

Table E-4 below summarizes supplementary spacecraft communications capabilities assumed for hypothetical ATS-F&G operations at frequencies incompatible with the specific transponder characteristics given in Table E-3.

Table E-4

Supplementary ATS-F&G Communications Subsystems Characteristics			
Operating Frequency	Function	Receiver Noise Figure (db)	Maximum CW Power Output (watts)
860 MHz	Transmit	---	100
1.7 GHz	Receive & Transmit	3	1
2.3 GHz	Receive	3	---

The various antenna, transponder, and link parameters presented in Tables E-1 through E-4 define the ATS-F&G communications subsystems to be dealt with in this study. The next step is to consider the analysis of specific missions.

GENERAL COMMENTS ON COMMUNICATIONS SYSTEM ANALYSIS

In each of the several systems discussed below, the analysis centers around the question of available versus required channel capacity (Note 1). A tacit assumption, then, is that system performance is limited by white gaussian noise alone. In specific instances, it is possible, and indeed probable, that non-gaussian noise of man-made origin will be many decibels more severe than receiver thermal noise. In such situations it is clear that system performance may be limited by the former rather than by the latter. In such cases, the gross analysis provided here is, of necessity, inadequate and over optimistic.

A preliminary study (Reference 1) has pointed out the magnitude of the man-made noise problem in any future VHF broadcast satellite system (particularly in urban areas). In any event, it is obvious that local RF interference must be taken into account in establishing the practicality of a given operation at a given site. Several considerations, however, discourage serious treatment of man-made noise limited systems in this study. First, available data on man-made noise at the frequencies of interest (see Table E-1) are too meager to be of any real value for purposes of gross system design. Second, it is in general quite difficult to predict and characterize the performance of communications systems perturbed by thermal noise other than gaussian. Finally, communications system performance is always eventually limited by the thermal noise of the receiver itself, and unless enough signal power is provided to override this interference, there is no hope of operating in the exacerbated environment of man-made noise. Man-made noise, therefore, is disregarded in the analyses of potential ATS-F&G applications discussed in this report.

ATS-F&G MISSION OBJECTIVES INVOLVING SMALL (MOBILE) TERMINALS

Preliminary ATS-F&G mission objectives call for the demonstration of spacecraft stabilization techniques in a series of experiments. These experiments will be performed in conjunction with the large aperture spacecraft antenna system, providing communications, position determination, data acquisition, and control functions for small (mobile) terminal applications. A number of such experiments have been considered for the program and are discussed in the following pages, grouped according to type of operation under one of seven classifications:

1. Satellite TV broadcast and relay
2. Satellite FM voice broadcasting
3. Man-pack type multiple-access communications
4. Air-traffic communications and control
5. Satellite data acquisition
6. Satellite position determination
7. Satellite RFI measurements*.

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*See Note 5. The material on ATS-F&G RFI measurements was prepared too late for inclusion in the main text of this appendix of the Analytic Report; because of its potential importance, however, it is presented in a final note.

SATELLITE TV RELAY

General

For purposes of classification, TVtransmission systems involving communications satellites may be divided into two categories: those serving home viewers indirectly (through a central signal redistribution scheme) and those serving home viewers directly (direct broadcast TV) Up to the present, of course, all systems have been of the former type because of the severely limited effective radiated power (erp) available with both subsychronous satellites (Telstar and Relay) and synchronous satellites (Syncom and Early Bird). In every case both sensitive (high gain/low noise) ground receiving facilities and the use of wideband frequency modulation (trading bandwidth for signal-to-noise ratio) have been required for satisfactory video performance on the critical satellite-to-ground down-link.

The ATS-F&G large aperture antenna in conjunction with the several proposed communications subsystems will develop erps several orders of magnitude greater than those achieved in any previous satellite relay system. It is, therefore, of interest to investigate the extent to which this capability can be used to provide TV transmission to small ground terminals including, if possible, direct broadcast to the home.

Summary of Results in Reference A

Four system configurations from Reference A were chosen for study and are listed in Table E-5.

System 1, which was included mainly for purposes of completeness, had been analyzed previously (at a somewhat different frequency) by Gould (Reference 2) who termed it a "Spacious Fantasy." System 2 was presented as a space application of the well-developed techniques of UHF line-of-sight TV relay (Reference 3). Systems 3 and 4, although similar to previous

Table E-5		
Summary of Systems Studied in Reference A		
System	Frequency	General Description
1	466 MHz	TV direct-broadcast to home receivers using conventional vestigal sideband amplitude modulation (VSB-AM)
2	466 MHz	TV relay to small community type central receivers using VSB-AM
3	466 MHz	TV relay to central receivers using frequency modulation
4	7.3 GHz	TV relay to central receivers using frequency modulation

satellite TV relay operations, clearly reflected the advantages of ATS-F&G erps in terms of enormously reduced ground receiving facility complexity and cost.

The signal characteristics and system design objectives assumed are discussed in Reference A and will not be reproduced here. It was noted in each case that 466 MHz was not an appropriate frequency for any of the three UHF systems analyzed. That frequency was used for purposes of gross analysis because a usable spacecraft transponder was assumed available at 466 MHz, and because the UHF TV broadcast band of 460 to 890 MHz being immediately adjacent suggested that system technical feasibility could be investigated successfully, despite the fact that eventual frequency allocation would be a severe problem. Brief summaries of the four analyses are given below.

Analysis of System

System 1 was over 30 db short of its design objective and hence must be considered entirely unfeasible with the ATS-F&G RF power levels presently under consideration. The home receiver assumed was a typical UHF unit with a noise figure of 12 db, driven by an elaborate 15-db gain antenna. While it is true that an inexpensive antenna-mounted preamplifier could significantly improve the 12-db noise figure assumed, it is obvious that even a 6-db improvement would do little to make up the 30-db deficit in this system. The analysis of System 1 was checked by comparing the field strength* at the ground produced by an ATS-F&G-type spacecraft radiating 40 watts of power at 466 MHz with the field strengths recommended for various classes of TV service at Channel 14 (see Table 20-7 of Reference A). The field strength produced, approximately 25 v/meter, is roughly 30 db below that required for strong secondary reception and at least 50 db below that required for first-class principal city reception in a severe environment of man-made noise.

Analysis of System 2

The only difference between Systems 1 and 2 was the complexity of the ground receiving facility assumed. A 3-db receiver and a 25-db gain UHF array were postulated as feasible for a small central receiving system. The resulting analysis indicated a deficit of 20 db in system channel capacity relative to the assumed design objective.

Analysis of System 3

Wideband FM was chosen as the down-link modulation technique for System 3, and the same receiving system as that for System 2 was assumed. A positive margin of several decibels existed over postulated design objectives, reflecting the efficiency of modulation techniques which trade bandwidth for performance (Note 1). The RF Bandwidth in System 3 was nearly 30 MHz, equivalent to five 6-MHz UHF channels stacked side by side.

Analysis of System 4

This last system used the same FM parameters as System 3, at a down-link frequency of 7.3 MHz. A minimal microwave receiving facility (2-foot dish and 1000^{o}K receiver temperature) were assumed; and as with System 3, a positive margin of several decibels was obtained relative to design objectives.

*See Note 2 for a concise summary of the relationships between synchronous satellite erp and electric field strengths at the ground.

SYSTEM 5 - A PROPOSED EXPERIMENT FOR ATS-F&G

Of the four systems examined in Reference A and discussed briefly above, only System 4 (FM-TV relay at 7.3 GHz) showed any promise of eventual fruition in a program of ATS-F&G demonstration-experiments. Systems 1, 2, and 3 were seen to be unpracticable because of insufficient down-link channel capacity, frequency allocation problems, or both .

The concept of System 5 proposed here is quite similar to that of both System 3 of Reference A and an 800-MHz FM-TV relay system discussed in Reference C. The frequency selected is 860 MHz, which lies roughly in the center of a block of approximately 14 6-MHz channels (i.e., numbers 70 through 83) presently allocated for point-to-point TV relay operations of the type discussed in Reference 3. At least two considerations suggest the possibility of securing permission, at least on a temporary basis and within the geographical United States, for such an operation in the vicinity of 860 MHz. First, the proposed demonstration-experiment would be in general philosophical accord with the operations for which this band is presently reserved; i.e., point-to-point TV relay rather than TV broadcast to home users. Second, it can be shown that it is unlikely that an 860-MHz ATS-F&G wideband operation with the RF power proposed (100 watts) would cause harmful interference to existing services. An analysis which supports this assertion is presented in Note 3.

The basic FM parameters of the proposed system are the same as those of Systems 3 and 4 of Reference A. One hundred watts of spacecraft power and a 1000°K receiving system with a 5-foot dish are the basic assumptions. The link analysis is detailed in Table E-6.

With a spacecraft dedicated solely to TV relay operations of the type outlined in the above calculation, it would be entirely feasible to assume a somewhat higher antenna gain than the 33.5 db used in Table E-6. This would increase existing margins by several decibels (see Table E-1). It is felt that System 5 is of great interest for purposes of time zone TV relay operations to small central receivers of the CATV type (Reference 6). Specialized receiving equipment required to convert existing CATV-type stations to handle wideband FM signals is discussed below.

Table E-6

Analysis of System 5	
Parameter	Link Calculation
Satellite transmit power (100 watts)	+ 20 dbw
Transmit antenna gain (Table E-1)	+ 33.5 db
Erp	+ 53.5 db
Path loss	-183.2 db
Receive antenna gain (5' dish)	+ 20.2 db
C, received carrier power	-109.5 dbw
η_o receiver noise density (1000° K)	-198.6 dbw/Hz
C/η, available channel capacity (see Note 1)	+ 89.1 db-Hz
Required channel capacity (Reference 4)	+ 88.7 db-Hz
Margin over reference system	+ 0.4 db
Output weighted S/N ratio	+ 52.4 db
Linear fade range before FM threshold	+ 4.4 db
Carson's rule RF bandwidth (Reference 5)	28.8 MHz

RECEIVER CONSIDERATIONS FOR SYSTEMS 3, 4, and 5

The wideband FM television relay systems discussed here are extremely efficient so far as information theory is concerned (see Appendix C-2 of Reference 1) and require the use of specialized FM receivers with bandwidths of 30 MHz. Such receivers are not, in general, available on an off-the-shelf basis at this time. Therefore, the question of ready availability is a definite factor in establishing the overall feasibility of the systems proposed. Wideband receivers of the type required have already been successfully developed in limited quantities in connection with existing satellite relay systems. One such receiver is the RCA MM1200 furnished under contract NAS 5-3984 as terminal equipment for NASA's Rosman and Mojave ground stations.

Thus, while it is true that at present there exists no specific source of high-quality 30-MHz FM discriminators, there is little doubt that such equipment could be produced to supply any demand which might arise in future systems of FM television relay to small central receiving stations. Hence, development of terminal equipment would not be a pacing item in such systems.

Typical CATV stations are usually equipped with sensitive receiving antennas, and the 5-foot dish proposed in the calculation of Table E-6 would appear to be a modest choice in view of existing antenna investments (Reference 6).

SATELLITE FM VOICE BROADCASTING

Direct FM voice broadcasting from ATS-F&G in the commercial band (88 to 108 MHz) was considered briefly in Reference B and was not dealt with in depth in References A, C, and D. Spacecraft antenna gain at this frequency (assuming 50-percent efficiency) is only about 17 db, and could be achieved by much simpler methods than an ATS-F&G 30-foot parabola. Hence, this service, although of potential interest for other spacecraft antenna configurations, is not discussed further in this analytic report. It may be mentioned in passing that commercial-band FM satellite broadcast is presently being studied for system feasibility under NASA contracts NASW-1475 and NASW-1476.

It is readily demonstrated without resorting to link calculations that the proposed ATS-F&G UHF subsystems do not develop enough erp at either 466 or 860 MHz to make commercial quality FM broadcast to home receivers (using inexpensive converters and conventional tuners) technically feasible. As noted in the discussion of direct broadcast TV, ATS-F&G can be expected to develop maximum field strengths of some 25 μv/m or 28 dbu at 466 MHz. The FCC (Reference 7) recommends absolute minimum field strengths of 34 dbu for FM broadcast in the commercial band (88 to 108 MHz). Extrapolating this figure to 466 MHz by the square of the frequency ratio (reflecting the smaller effective area of a simple dipole receiving antenna at the high frequency) suggests required field strengths of the order of 50 dbu, some 22 db greater than the field strength available with the proposed spacecraft power. At the upper end of the UHF TV band (near 860 MHz), essentially the same situation will prevail. These deficits are not so great as to make commercial quality FM broadcast to specialized (high gain/low noise) home receiving systems technically unfeasible, but they are certainly large enough to obviate service to the average home user. For quality service of the latter type, satellite RF levels in the kilowatt range would be required, working into an ATS-F&G-type antenna.

General Electric (Reference C) has proposed a somewhat different quality broadcast service in which UHF home television receivers, audio sections only, would be used to receive FM transmissions direct from ATS-F&G. The analysis of General Electric (presented below) is considered misleading, as it fails to recognize that the real problem in a system of this type is to provide sufficient total signal power to drive the second detector of intercarrier television receivers in a linear fashion. In Note 4 it is shown that a video carrier channel capacity of 82.5 db-Hz is required to ensure linear second detector operation. A small fraction of this channel capacity (75.5 db-Hz) is required for the actual FM carrier to achieve a 50-db post-detection signal-to-noise ratio in a band of 50 to 15,000 Hz using standard FM parameters for video sound transmission ($\Delta f = \pm 25$ kHz). Both carriers are required to operate the usual intercarrier sound system; and combining the numbers, an overall channel capacity of 83.3 db-Hz is required for the service. Assuming a system operating at channel 14 and an elaborate UHF home television receiver of the type used in the TV direct broadcast system (see above or Table 20-6 of Reference A), a deficit of 8.9 would exist. For a less elaborate home receiver, the deficit would be correspondingly greater, indicating the general technical unfeasibility of the operation.

Nevertheless, noncommercial-type FM broadcast services at UHF remain of considerable interest for potential industrial, government, and military users, who could afford a somewhat more sensitive receiving facility than the home user. Possible applications of an ATS-F&G-type UHF FM broadcast system include 24-hour central weather reporting and subscription facsimile services. The latent demand for the latter type of service has been pointed out by A. G. Cooley (Reference 8).

The following material parallels (except for numerical details) the corresponding discussion in Reference A. Design objectives are taken from military standards for FM broadcasting in the UHF band (Reference 9) rather than from FCC standards because of the somewhat specialized nature of the proposed operation. The information signal is assumed to be either 4-kHz speech or a 4-kHz voice frequency facsimile system. The facsimile system proposed would have typically a scan rate of 120 lines/minute, a scan width of 18 inches, with a resolution of 100 elements per inch. The keying frequency would then be given by

$$(2 \text{ lines/second})\ (18 \text{ inches/line})\ (100 \text{ elements/inch}) = 3.6 \text{ kHz}$$

and hence would effectively constitute a 4-kHz voice frequency signal.

Military design objectives for transmission of 4-kHz signals by FM in the UHF band are a signal-to-noise ratio of 47.8 db and a maximum RF bandwidth per emission of 40 kHz. The signal-to-noise ratio quoted is for transmission of a sinusoidal signal of the maximum amplitude consistent with system capacity and unweighted noise measured in a 4-kHz band. This is equivalent to +38 dba_0 (U.S. telephone terminology-FIA weighting at the zero-transmission point) and/or 25,000 picowatts (European telephone terminology - psophometric weighting at the zero transmission point). (For details, see References 10, 11, and 12.)

The assumed ground facility has a gain of 15 db and a noise temperature of 1000°K. Two formulas from FM theory are required to complete the analysis:

Equation 1: Carson's rule $B_{rf} = 2\,(\Delta f + B_i)$ (Reference 5)

Equation 2: FM output signal/noise ratio $= \frac{3}{2}\left(\frac{\Delta f}{B_i}\right)^2 \frac{C}{\eta B_i}$ (Reference 12)

The calculation is summarized in Table E-7.

Table E -7

Analysis of UHF FM Broadcast Service

Parameter	Link Calculation
B_i intrinsic bandwidth	4 kHz
Brf, Carson's rule bandwidth (Reference 9)	40 kHz
Δf, allowable peak deviation	16 kHz
$\Delta f/B_i$, allowable index of modulation	4
Design objective output signal/noise ratio	47.8 db
C/η, required channel capacity (solving Eq. 2)	+ 70.0 db
Satellite transmit power (40 watts - Table E-31)	+ 16.0 db
Transmit antenna gain (Table E-1)	+ 28.6 db
Path loss (Table E-2)	-178.2 db
Receive antenna gain (UHF array)	+ 15.0 db
C, received signal power	-118.6 dbw
η, receiver noise density	-198.6 dbw/Hz
C/η available channel capacity	+ 80.0 db
Margin above performance objective	+ 10.0 db
Margin above 10-db FM threshold	24.0 db

The analysis shows the proposed system to have a substantial margin over design objectives. The simplicity of the required ground receiving equipment would recommend this type of demonstration-experiment for inclusion in the ATS-F&G program. The relatively broad earth coverage afforded by the 30-foot dish at 466 MHz (Table E-2) is still another argument in favor of this system.

MULTIPLE-ACCESS "MAN-PACK" COMMUNICATIONS

Two "man-pack" multiple-access voice communications systems meeting military design objectives have been studied for possible experimental use with the ATS-F&G. Important characteristics of the two systems are summarized in Table E-8. Except for numerical details, this is the same material which was given in Reference A. Together with the subsequent discussions of ATS-F&G aircraft communications, data collection, and position location applications, it is presented here in revised form for purposes of completeness.

Table E-8

Characteristics of "Man-pack" Systems

System	Multiplexing Technique	Number of Duplex Channels	Band of Operation	Modulation Technique
1	Frequency Division Multiplex (FDM)	10 (with companding)	UHF	SSB
2		10 (without companding)	X-band	FM

Both systems would require essentially linear transponders to avoid severe intermodulation effects (References 11 and 13); design objectives for both systems have been taken directly or extrapolated from Reference 9.

ANALYSIS OF SYSTEM 1

Table E-9 summarizes the analysis of System 1. A thorough discussion of the loading factors and peak factors used in computations for SSB-FDM systems is to be found in Reference 12. Syllabic speech compandors are required in System 1 to bring performance up to the design objective. The conservative 16-db compression/expansion improvement assumed, as well as the multichannel peak and loading factors for compandored channels, has its source in a paper by Rizzoni (Reference 14). Without compandors, the system would be many decibels short of the military design objective but would still be a useful voice channel.

Frequency stability of the inserted carrier is an important consideration in the detection of SSB signals. Assuming a maximum allowable inaccuracy (in precision or stability) of several tens of hertz, the requirement would be an easily achieved part in 10^{-7}.

Table E-9

Analysis of System 1, Down-link

Parameter	Link Calculation
Design objective for overall system noise performance	+ 38 dba_0
Equivalent TT/noise ratio, 4 kHz unweighted (References 10 and 15)	+ 42.5 db
Down-link TT/noise objective, allowing 3 db for up-link noise and intermodulation effects	+ 45.5 db
Down-link TT/η design objective, assuming 16-db subjective companding improvement	+ 65.5 db-Hz
Peak satellite erp (3 db greater than CW erp of Table E-7	+ 47.6 dbw
Multichannel peak factor, 10 compandored channels	+ 12.3 db
Mean satellite erp, 10 compandored channels	+ 35.3 dbw
Loading factor, 10 compandored channels	+ 2.5 db
Satellite TT erp	+ 33.8 dbw
Free space loss	-178.2 db
Receiving antenna gain (UHF array)	+ 12.0
Received TT power	-133.4 dbw
η, receiver noise density (1000° K)	-198.6 dbw/Hz
Available TT/η, down-link	+ 65.2 db-Hz
Margin over design objective	- 0.3 db

The analysis for the up-link parallels that for the down-link and will not be given in detail. The essential result is that a single channel SSB transmitter, operating into the UHF array used in the down-link, would operate at a mean power of about 2.5 watts and would require a peak rating of some 60 watts. This analysis assumed that all 10 duplex channels were on the air or in use at the same time, and the loading and peak factors used took into consideration the statistics of ordinary talkers. In a modest multiple-access operation of this type, the use of these statistics is probably pessimistic and leads to an overdesigned system. At any rate, the present results indicate the general technical feasibility of a compandored UHF SSB-FDM "man-pack" multiple-access ATS-F&G communications experiment.

ANALYSIS OF SYSTEM 2

For System 2, the assumed "man-pack" receiver parameters are a 2-foot dish and a noise temperature of 1000° K. As with System 1, the down-link is analyzed in detail and discussion of the up-link is limited to a statement of ground transmitter power requirements. A conservative 2-db back-off from the rated 24 watts of TWT output power is assumed, ensuring essentially linear amplification of the 20 (equal power) FM carriers, corresponding to 10 duplex FM channels (Reference 16). See Table E-10.

Per channel, up-link CW power requirements, assuming no interference from other sources, would be roughly 2 watts working into an on-beam 2-foot dish. Power control could be a consideration in system operation, for if one user were to operate so that his up-link transmission arrived at the satellite many decibels stronger than the FM carriers of the other users, potential "power-grabbing" would occur. For the relatively large number of simultaneous carriers considered, however, the problem should not be too serious.

The minimum RF bandwidth of an X-band multiple-carrier FM "man-pack" system would be 800 kHz, assuming 40 kHz per carrier (Reference 9). To reduce intermodulation effects, it would probably be worthwhile to use somewhat more spectrum spacing the carriers instead of separating them by 40 kHz. Babcock (Reference 17) has studied the spacing problem, and his "optimum" spacings to avoid intermodulation spectra lead to somewhat inefficient use of the spectrum. Doyle (Reference 13) concludes that elaborate spacing schemes are not worth the trouble they cause. Thus, it may be argued that the most convenient and practical solution would be to assume 50-kHz carrier separations in keeping with general FCC recommendations for fixed and mobile FM voice links (Reference 18).

The excellent technical quality attainable with relatively simple although admittedly specialized "man-pack" systems of the two types discussed here suggest that such an operation at X-band or UHF be considered as a demonstration-experiment in the ATS-F&G program. It should be noted that the comments relative to weather broadcast and facsimile services earlier in this document apply to the "man-pack" systems discussed, as do the statements regarding earth coverage of the 30-foot dish (Table E-2).

Figure E-1 illustrates receive and transmit spectra for the two FDM "man-pack" systems presented in this portion of the study.

Table E-10

Analysis of System 2 (Down-Link)	
Parameter	Link Calculation
Required C/η per channel, assuming no up-link noise and intermodulation effects (from Table E-7)	+ 70.0 db-Hz
Required C/η per channel, allowing 3 db for up-link noise and intermodulation effects	+ 73.0 db-Hz
Available satellite erp (2 db back-off from figure computed from Tables E-1 and E-3)	+ 60.8 dbw
Available satellite erp per channel (20 carriers)	+ 47.8 dbw
Free space loss	-202.1 db
Receive antenna gain (2-foot dish)	+ 30.4 db
C, received carrier power per channel	-123.9 dbw
η, receiver noise density (1000° K)	-198.6 dbw/Hz
Available C/η, per down-link channel	+ 74.7 db-Hz
Margin over required C/η	1.7 db

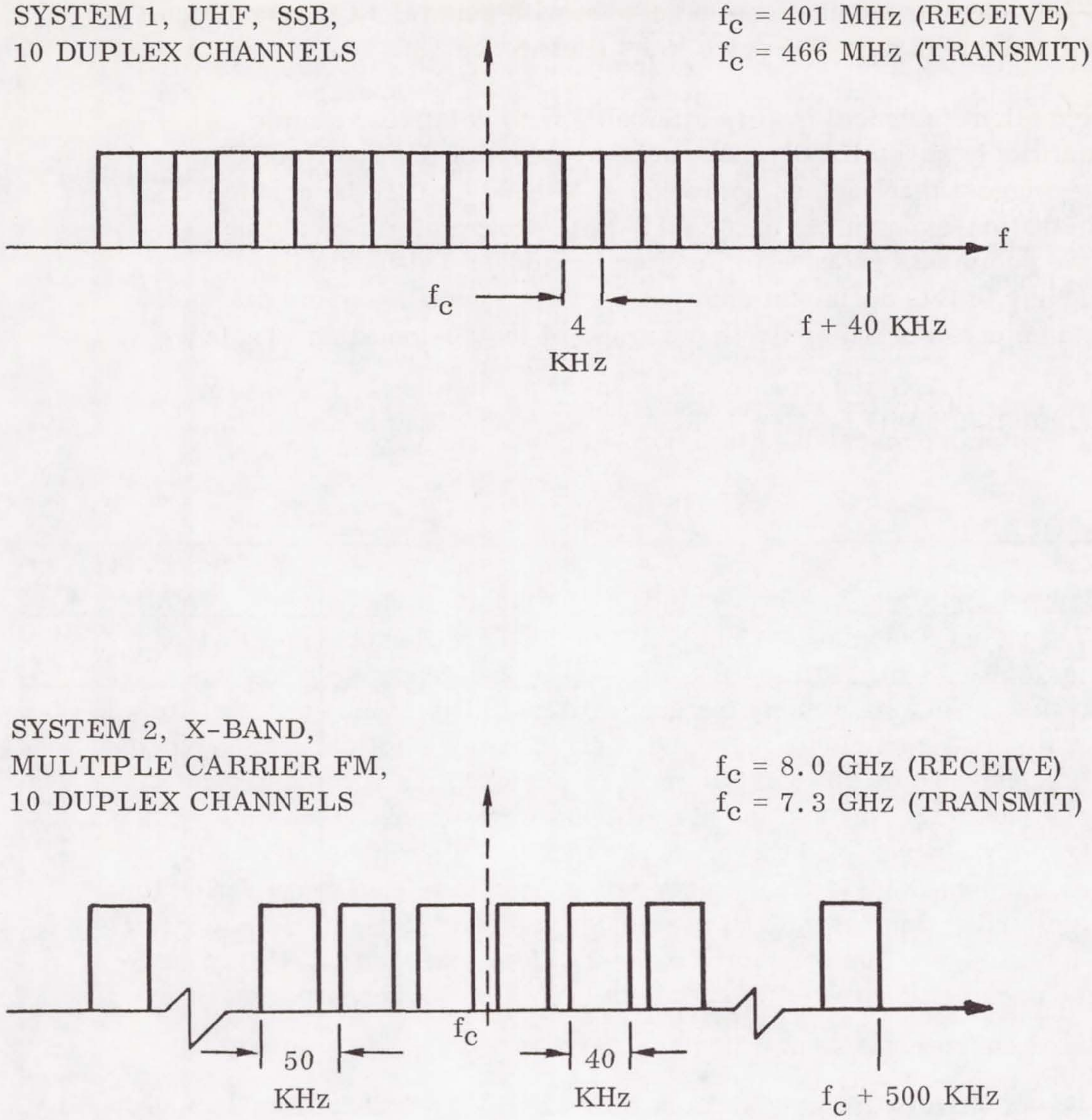

Figure E-1. Receive and transmit spectra for FDM multiple-access communications, Systems 1 and 2

AIR TRAFFIC COMMUNICATIONS AND CONTROL

The following discussion is not intended to be a comprehensive treatment of ATS-F&G potentialities and capabilities for systems of air traffic communications and control. The field is so new, and the requirements and characteristics of future operational systems (including frequency assignments) so vaguely defined that a detailed analysis would be beyond the scope of this in-house study. Rather, the discussion is limited to a general statement of the background of aircraft-satellite communications, a comparison of ATS-B VHF Repeater and ATS-F&G UHF channel capacities in the satellite-to-aircraft link, and to a brief analysis of a hypothetical S-band (1.7 GHz) link between an ATS-F&G spacecraft and an in-flight airplane. Except for numerical details, this section duplicates corresponding material in Reference A.

BACKGROUND

Early attempts at aircraft-satellite communications conducted jointly by Air Transport Association (ATA), Bendix, NASA-Hughes, and Pan American (subsequently termed the ATA tests) and reported in Reference 19 were successful in establishing the general feasibility of using a satellite relay for aircraft communications and control over remote areas. Teletype transmissions at rates up to 100 wpm were achieved on both the ground-satellite-aircraft up-link and the aircraft-satellite-ground down-link via a VHF telemetry channel of the Syncom II satellite. Performance was highly variable, but enough good copy was received in both modes of operation to encourage further interest in and development of aircraft-satellite communications technology. One such development, of course, is the VHF repeater which has been demonstrated in successful tests of the recently launched ATS-B spacecraft (Reference 20).

The communications package of the ATS-B VHF transponder has an overall bandwidth of some 100 kHz and develops an erp of +23.5 dbw as compared to an erp of only -3.5 dbw in the original Syncom II VHF system, affording a 27-db increase in the channel capacity of the critical satellite-aircraft link, and opening the way for FM voice communications and other wideband signaling techniques.

ATS-B AT VHF VERSUS ATS-F&G AT UHF

ATS-B, with its phased-array antenna giving global coverage and developing +23.5 dbw of erp at 136 MHz, has a channel capacity of some

+58 db-Hz in a typical link to an airborne receiver having a 3-db gain antenna, 3-db polarization loss, and a system temperature of 1000° K. Analyses in Reference 20 and preliminary experiments conducted by FAA, private industry, and the military indicate that this link (satellite-aircraft links are in general many decibels weaker than aircraft-satellite links) can support an "acceptable" 4-kHz voice channel using FM parameters close to those used in the systems detailed in Tables E-7 and E-10 of this study.

By way of comparison, ATS-F&G with its 44.6 dbw of erp at 466 MHz in a similar link would have a channel capacity of about +66 db-Hz, somewhat fatter but still below that required to meet military design objectives for the FM voice link.

ATS-F&G coverage at 466 MHz is somewhat short of global (without defocusing the beam), but studies (Reference 21) have shown that by positioning the spacecraft in equatorial orbit near 87 degrees W longitude and pointing the large aperture antenna at 54 degrees W longitude by 20 degrees N latitude, the main-beam would virtually blanket the North Atlantic, presumably a requirement for any future operational system in that part of the world.

Since ATS-F&G UHF frequencies are not used for present air-traffic control and since future systems are unlikely to use these frequencies (Reference 22), the fact that ATS-F&G is only at best some 8 db better than ATS-B in the critical satellite-aircraft link does not suggest a bright future for ATS-F&G in the area of UHF air-traffic communications and control. Beam-defocusing, of course, would produce world-wide coverage and reduce ATS-F&G channel capacity to an aircraft below that of ATS-B.

S-BAND AIRCRAFT-SATELLITE COMMUNICATIONS WITH ATS-F&G

Much of the variability in performance in the ATA tests (Reference 19) was attributed to multipath propagation due to specular sea-water reflections. Even in the presence of large received signal-to-thermal-noise ratios, intersymbol interference and resulting teletype error rates were at times completely unacceptable. The severity of multipath interference in any satellite-aircraft link over water, and the validity of the geometrical model of Figure E-2 (taken from Reference 23) have recently been confirmed in Air Force tests conducted by Lincoln Labs at M.I.T. (Reference 24). Further evidence of the impact of multipath on the VHF aircraft-satellite communications is anticipated in the results of present and future experiments with ATS-B.

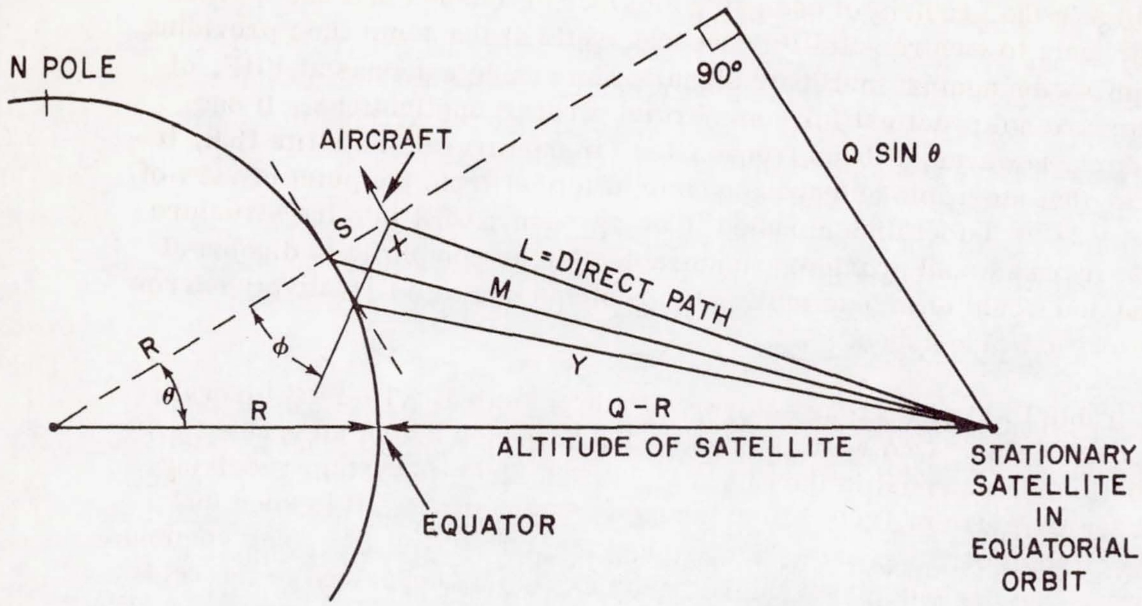

Figure E-2. Geometrical model of multipath.

L = Direct Path

Y + X = Approximate Reflected Path

Q $\cong$ 6.5R for synchronous altitude

S = Aircraft Altitude

$$M = \sqrt{R^2 + Q^2 - 2QR\cos\theta}$$

$$L \cong M - \frac{S}{M}\left[Q\cos\theta - R\right] \qquad X = \frac{S}{\cos\phi}$$

$$Y \cong M - \frac{SQ}{M}\sin\theta\tan\phi \qquad D = \text{path difference}$$

$$D = X + Y \;\lrcorner = \frac{S}{\cos\phi}\left[1 + \frac{Q\cos(\theta+\phi) - R\cos\phi}{M}\right]$$

pt. of min. multipath difference. $\sin\phi = \frac{Q}{M}\sin\theta$

$$\therefore D_{min} = \frac{25}{M}(Q\cos\theta - R) \quad \left[D_{min}\right]_{max} = 25 \cong 60\mu s \text{ for 30 k ft alt.}$$

and N - S flight

To a large extent, the difficulty in overcoming multipath problems at VHF lies in the problem of designing an aircraft antenna that has a broad enough beam to ensure satellite coverage while at the same time providing discrimination against multipath signals. Steerable antennas at VHF, of course, are not practical for commercial aircraft applications. If one considers, however, S-band frequencies for the aircraft-satellite link, it is clear that steerable antennas become practical from the point of view of size. A 2- or 3-foot dish mounted in an aerodome built into the structure of the aircraft would provide considerable gain as compared to dipoles at S-band and would eliminate multipath problems due to its relatively narrow beam and low side-lobes.

Table E-11 is a gross analysis of a hypothetical ATS-F&G aircraft link at 1.7 GHz.* One watt of satellite RF power, a 3-foot steerable parabolic dish mounted in the airplane, and an aircraft system receiving noise temperature of 1000°K are the basic assumptions. It is seen that ample channel capacity exists for supporting high quality FM voice communications. One serious disadvantage to the system as described is the relatively narrow beamwidth of the ATS-F&G 30-foot antenna at 1.7 GHz, corresponding to a coverage of some 1000 km at the subsatellite point. Any operational satellite-aircraft system would probably work with a less directional spacecraft antenna, making up the difference by using increased RF power. The calculation does show, however, the technical feasibility of such a communications link.

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*A frequency allocation for air-traffic communications and control already exists around 1.5 GHz. It is felt that the results of ATS-F&G experiments at 1.7 GHz could be extrapolated to the lower frequency with complete confidence.

Table E-11

Hypothetical ATS-F&G Link to In-flight Aircraft at S-band	
Parameter	Link Calculation
Satellite transmit power (1 watt)	+ 0.0 dbw
Satellite antenna gain (Table E-1)	+ 38.6 db
Satellite erp	+ 38.6 dbw
Nominal path loss	-189.1 db
Aerodome losses (assumed)	- 3.0 db
Aircraft receive antenna gain (3-foot dish at 1.7 GHz)	+ 21.5 db
C, received signal power	-132.0 dbw
η, receiver noise density (1000^{o}K)	-198.6 dbw/Hz
C/η, S-band satellite-aircraft channel capacity	+ 66.6 db-Hz

DATA ACQUISITION

The sensitive receiving capability of the proposed ATS-F&G communications subsystems combined with the flexibility of a high-gain steerable antenna in geosynchronous orbit suggests several novel applications in the area of data acquisition. Three potential applications are discussed below: satellite-to-satellite relay of Interrogation, Recording, and Location System (IRLS) data from a Nimbus-type spacecraft to ATS-F&G, satellite-to-satellite relay of S-band video from a Nimbus-type spacecraft to ATS-F&G, and launch phase Apollo support S-band data links. Again, except for numerical details, this material duplicates corresponding material in Reference A.

ATS-F&G IRLS APPLICATIONS

The IRLS experiment (Reference 25) is a proposed data collection operation in which many sensors monitoring oceanographic, geophysical, and meteorological conditions in remote areas around the world (unmanned sea buoys, platforms, or balloons) relay their data to a central point through a sub-synchronous satellite of the Nimbus series. The spacecraft would have a UHF transponder to collect signals at 466 MHz and would relay them at 401 MHz to a central point via NASA Command Data Acquisition stations located at Fairbanks, Alaska, and Rosman, North Carolina.

The pseudo-noise ranging technique to be used in the IRLS is not compatible with the concept of direct interrogation of platforms by a geosynchronous satellite, although enough channel capacity would exist to make some operations feasible. The propagation delays involved in such a link would produce time ambiguities which could not be resolved without major redesign of the IRLS. However, addition of a 401-MHz transmit capability to the Nimbus spacecraft would enable ATS-F&G, as presently conceived, to monitor IRLS platforms indirectly, replacing the direct Nimbus-Rosman link with a Nimbus-ATS-F&G Rosman link. This link would have the advantage of greatly increasing the percentage of time that Nimbus would be available to interrogate platforms. The concept of using 401 MHz as both a receive and transmit frequency for Nimbus, while somewhat unusual, is technically feasible since all IRLS transmissions to and from Nimbus are in bursts, permitting time-division-multiplexing of spacecraft receive and transmit functions.

The nominal channel capacity of the platform-Nimbus up-link is given as 64.1 db-Hz (Reference 25). Using this as a design objective for the proposed Nimbus-ATS-F&G link, the required Nimbus erp is computed in Table E-12.

Table E-12

Nimbus-ATS-F&G IRLS Relay, Power Requirements

Parameter	Link Calculation
Design objective C/η ratio	+ 64.1 db
η, ATS-F&G receiver noise density (600° K)	-200.8 dbw/Hz
Nominal Nimbus-ATS-F&G path loss	-176.0 db
ATS-F&G receive antenna gain	+ 27.0 db
Required Nimbus erp	+ 12.3 dbw
Available Nimbus erp (approx.)	+ 14.0 dbw
Margin	+ 1.7 db

The next system to be considered is hypothetical in that it assumes the existence of an ATS-F&G wideband receiving capability at 1.7 GHz. Present Nimbus operations involve transmission of reduced data rate recorded video signals to the Fairbanks and Rosman 85-foot receiving facilities at S-band (1.7 GHz). Satellite transmitter power is nominally 5 watts, and the nominal antenna gain is 3 db.

Following the line of reasoning used in the IRLS-ATS-F&G application discussed above, Table E-13 compares the channel capacity of the existing Nimbus-Rosman link with that attainable in a hypothetical Nimbus-ATS-F&G S-band link. Beam steering of the large aperture antenna is assumed, keeping the subsynchronous satellite always in the main-beam of the ATS-F&G receiving system.

Table E-13

Nimbus-ATS-F&G versus Nimbus-Rosman S-band Channel Capacities

Parameter	Nimbus-Rosman	Nimbus-ATS-F&G
Link frequency	1.7 GHz	1.7 GHz
Nimbus erp	+10.0 dbw	+10.0 dbw
Free space loss	-------*	-189.1 db
Receive antenna gain	-------*	+38.6 db
C, received signal power (Reference 26)	-99.0 dbw (typical measured value)	-140.5 dbw
System noise temperature	250°K (typical)	$+600^{\circ}$K
η, receiver noise density	-204.6 dbw/Hz	-200.8 dbw/Hz
C/η, link channel capacity	+105.6 db-Hz	+60.3 db-Hz

*Path loss and antenna gain are not indicated for the Nimbus-Rosman link because actual measured received signal strengths were available.

The enormous difference in channel capacities, some 45 db, reflects the higher gain, lower system temperatures, and shorter path length involved in the Nimbus-Rosman link. Nevertheless, the channel capacity to ATS-F&G could support digital data rates of 4×10^5 bps with error rates under 10^{-3}, enough for non-real-time transmissions. Real-time video in such a link would require a high gain antenna on the Nimbus spacecraft or orders of magnitude greater RF power levels.

APOLLO S-BAND LAUNCH PHASE SUPPORT

During the critical launch phase of the Apollo mission, the spacecraft will be stationed in a subsynchronous parking orbit for a sufficiently long time to make continuous communications a severe problem. An ATS-F&G spacecraft could be of considerable value in supplementing the presently planned Apollo communications network of ships, aircraft, and ground stations during launch phase. The following gross analysis indicates the technical feasibility of such an operation with an ATS-F&G spacecraft equipped with a 2.3-GHz receive capability. (See Table E-14)

Table E-14

Approximate Data Rate for Hypothetical Launch-phase Apollo-ATS-F&G Data Relay Link

Parameter	Link Calculation
Frequency (Apollo unified S-band system)	2.3 GHz
Apollo spacecraft erp (assumed)	+12.0 dbw
Free space loss to ATS-F&G (max.)	-193.0 db
ATS-4 receive antenna gain (50% efficiency assumed for dedicated mission)	+43.0 db
C, received signal power	-138.9 dbw
η, receiver noise density (600°K)	-200.8 dbw/Hz
C/η, link channel capacity	+62.8 db-Hz
Energy per bit/η ratio for bit $P_e = 10^{-3}$ (Reference 27)	+6.5 db
Attainable Apollo ATS-F&G data rate (56.3 db)	4×10^5 bits/sec

POSITION DETERMINATION

Consideration of potential ATS-F&G applications in the area of position determination will be limited to a brief discussion of ATS-F&G usage in the Omega Position Location Equipment (OPLE) experiment (Reference 28). The proposed OPLE system is a world-wide data collection operation which has a precision position location capability ($\pm$3km) based on use of the Omega Navigation System recently developed by the U. S. Navy. Involved are a large number of free-floating meteorological platforms and weather balloons, dispersed over the earth and synoptically monitored by a geo-synchronous satellite which relays collected data to NASA's Rosman receiving facility. The existing VLF Omega system would provide each platform (or balloon) with information as to its position. This information, together with real-time meteorological sensor data, would be transmitted upon interrogation to the monitoring satellite for relay to Rosman.

Two frequency configurations have been discussed, a preliminary system at VHF (149 MHz up / 136 MHz down) and an operational system at UHF (401 MHz up / 466 MHz down). This latter scheme is, of course, one which has been selected for ATS-F&G UHF operations, and hence it is clear that ATS-F&G might serve as the spacecraft relay in an OPLE experiment.

Using platform and satellite parameters from Reference 28, Table E-15 compares the up-link channel capacities of the OPLE UHF system as proposed and the OPLE system working with ATS-F&G. Once more, this is essentially the same material as was presented in Reference A. The table shows that the OPLE system, working with ATS-F&G at UHF, would have an up-link some 12 db better than that discussed in Reference 28. This would mean that platform RF power, for example, could be reduced to the milliwatt region or, alternately, much higher data rates could be used.

A requirement to have truly world-wide coverage, however, would entail defocussing of the ATS-F&G beam, reducing the advantage to only 2 db. Nevertheless, it is clear that ATS-F&G as proposed would be a more than suitable spacecraft for OPLE UHF experiments.

Table E-15

OPLE Up-link Comparison		
Parameter	OPLE as Proposed	OPLE with ATS-F&G
Platform transmit power	+2.6 dbw	+2.6 dbw
Platform antenna gain	0.0 db	0.0 db
Path loss	-176.9 db	-176.9 db
Satellite antenna gain	+14.0 db	+27.0 db
Misc. losses	-2.0 db	-2.0 db
C, received signal power	-162.3 dbw	-149.3 dbw
Receiver noise temperature	500°K	600°K
η, receiver noise density	-201.6 dbw/Hz	-200.8 dbw/Hz
C/η, up-link channel capacity	+39.3 db-Hz	51.5 db-Hz

SUMMARY

A relatively broad spectrum of ATS-F&G applications, experiments, and demonstrations has been in widely varying depth, all involving communications between the large aperture antenna in synchronous orbit and small, including mobile, terminals. A great many assumptions entered into the analysis of any single system, and it is felt that a detailed side-by-side tabulation of all systems considered, including identification of all assumptions, would be unwieldy and somewhat misleading. Nevertheless, it is desirable to summarize the work in some concise way. Therefore, Table E-16 has been prepared, identifying the system studied, noting the spacecraft RF power proposed, stating the design objective or reference system used, and indicating the results of the analysis in terms of performance margins, plus or minus, with respect to the design objective or reference system. Brief comments on overall system feasibility are also included for completeness.

Table E-16

Summary of ATS-F&G Experiments Involving Small Terminals			
ATS-F&G Application	Spacecraft RF Power (in watts if applicable)	Design Objective Source or Reference System	Margin with respect to Design Objective or Reference System, and/or General Comments
Direct broadcast TV (system 1 - 466 MHz)*	40	Downgraded CCIR standards for TV relay systems	-31.5 db**; completely unfeasible with present ATS-F&G RF power levels
UHF TV relay to small central receivers (System 2 - 466 MHz)*	40	CCIR standards for TV relay systems	-19.3 db**; at best, a marginal operation; frequency a problem
FM-TV relay to small central receivers (System 3 - 466 MHz)*	40	CCIR standards for TV relay systems	+4.1 db**; frequency allocation problems for this service would be most severe
FM-TV relay to small central receivers (System 4 - 7.3 GHz) *	24	CCIR standards for TV relay systems	+4.9db**; a promising ATS-F&G application
FM-TV relay to small central receivers (System 5 - 860 MHz)	100	CCIR standards for TV relay systems	+0.4 db; coordination with existing services seems feasible
FM voice broadcast to home receivers with UHF tuners (466/860 MHz)	40	FCC field strength requirements for home receivers	At least 20 db more spacecraft power required for quality service.

*Analysis given in Reference A and results only quoted here.

**ATS-F&G antenna efficiency of 50% assumed in these applications.

Table E-16 (Cont'd)

UHF FM voice broadcast to home TV sets, audio sections only	-----	50 db output S/N ratio	Spacecraft RF levels in the kilowatt range required for quality service
UHF FM voice broadcast to specialized home receivers	40	Applicable military standards for 4 KHz UHF FM voice broadcast	+10.0 db; an ATS-F&G application worthy of serious consideration
UHF "Man-pack" communications (System 1 - SSB with compandors at 466 MHz)	40 (for 10 duplex channels)	Applicable Military standards for voice communications	-0.3 db; a generally attractive ATS-F&G application
X-Band "Man-pack" FM communications (System 2 - 7.3 GHz)	24 (for 10 duplex channels)	Applicable Military standards for voice communications	+1.7db; again, a promising ATS-F&G application; area coverage limited
UHF air traffic communications and control	40 (per channel)	ATS-B links with aircraft at VHF	+8.0 db; a poor frequency for the proposed service
S-band air traffic communications and control	1.0 (per channel)	ATS-B links with aircraft at VHF	+11.3 db; experiments performed at 1.7 GHz could be used to justify use of the 1.5-GHz band for this service.
Satellite-to-satellite IRLS data relay	-----	Platform-Nimbus link	Nimbus erp of 12.3 dbw at 401 MHz would be required
Satellite-to-satellite video data relay	10	Nimbus-Rosman S-Band link	4×10^5 bps feasible
Satellite-to-satellite Apollo launch phase data relay	12 (assumed)	---------	4×10^5bps feasible **
Position location systems	2.6 (platform power)	Proposed OPLE system	+12.2 db; platform power could be reduced an order of magnitude.

** ATS-F&G antenna efficiency of 50% assumed in these applications.

CRITIQUE OF CONTRACTORS' FINAL REPORTS IN THE AREA OF COMMUNICATIONS EXPERIMENTS

The three contractors in the initial ATS-F&G study (Fairchild-Hiller, General Electric, and Lockheed) were asked to consider ATS-F&G 30-foot antenna communications capabilities at the frequencies summarized in Table E-17.

The three contractors examined the communications capabilities of the 30-foot antenna in varying depth; separate critiques of their efforts are given below.

Table E-17

Contractors' ATS-F&G Frequency Assignments

Frequency Band	Assignment	
	Receive	Transmit
VHF	-	100 MHz
UHF	-	800 MHz
S-band	1.7 & 2.1 GHz	2.3 GHz
X-band	8.0 GHz	7.3 GHz

FAIRCHILD-HILLER

The Fairchild-Hiller study (Reference B) was brief and to the point. Few details of the various link analyses were included, but the results, conveniently summarized in a single graph, appear correct in all essentials. Where comparisons are possible, the results are substantially in agreement with Table E-16.

GENERAL ELECTRIC

The General Electric discussion (Reference C) was the most elaborate and perhaps least satisfactory of the three contractor efforts. The system of FM direct broadcast to home TV sets, audio sections only, at 800 MHz, which General Electric proposed was supported by an inaccurate and misleading analysis. This analysis failed to recognize the fact that intercarrier receiver video carrier power requirements for linear second detector operation are much more severe than the FM sound carrier power requirements to achieve design objective, post-detection, signal-to-noise ratios. A more realistic analysis of General Electric's proposed system is given in Note 4, with the results quoted above (under Satellite FM Voice Broadcasting), and summarized in Table E-16.

The other systems discussed by General Electric appear to have been given more careful study. With the reservation that all of General Electric's assumptions involving receiver antenna gains and noise temperatures seem consistently optimistic, the several analyses can be endorsed as substantially correct.

LOCKHEED

The Lockheed discussion (Reference D) was limited to a calculation of receive and transmit power requirements to attain specified RF signal-to-noise ratios in a noise bandwidth equal to 10 percent of the carrier frequency in question. To the extent that nothing appears wrong with any of their calculations, Lockheed's brief discussion can be given blanket approval.

ADVANCED MISSIONS

The several applications discussed in the previous section concerned the category of communications between an ATS-F&G spacecraft in synchronous orbit and small (not necessarily earthbound) terminals. This section of the study considers very briefly non-geocentric space applications of ATS-F&G antenna technology. The two systems discussed are libration point communications to the hidden side of the moon and interplanetary probes. Antenna efficiencies of 50 percent are assumed rather than the conservative values associated with the gains and beamwidths of Table E-1. This material, presented for completeness, duplicates discussions in Reference A.

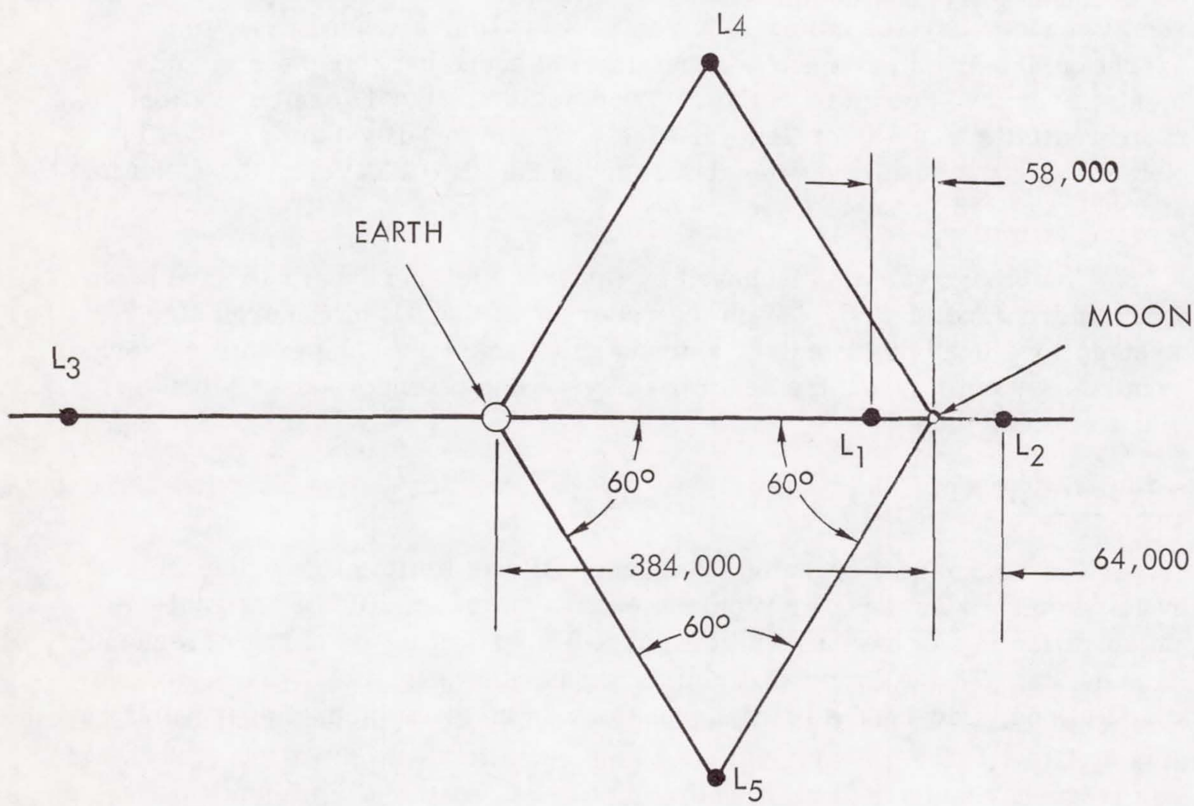

Figure E-3. Geometry of the five earth-moon libration points, labelled L_1, L_2, L_3, L_4, and L_5. (Distances in kilometers).

LIBRATION-POINT LUNAR COMMUNICATIONS

Figure E-3 shows the geometry of the five earth-moon libration points (References 29 and 30). Given the proper initial conditions, a light mass positioned at any of these points will rotate with the moon about the earth, remaining fixed in earth-moon space.

Since the moon's period of revolution about the earth is equal to its rotational period, any body that remains fixed in earth-moon space will be a synchronous satellite of the moon. This suggests several applications of ATS-F&G large aperture antenna technology. Two hypothetical systems have been selected for study.

Lunar-earth Communications via L_4 or L_5

The first system is that of a lunar base on the hidden side of the moon having a 10-foot dish and 10 watts of power communication with earth via a satellite located at L_4 or L_5. The satellite has an ATS-F&G type 30-foot dish for receiving signals from the lunar base at 8.0 GHz, and relays them at a level of 10 watts to earth at a frequency of 7.3 GHz, via a second, smaller (5-foot) dish. The earth receiving station is assumed to have an 85-foot dish and a noise temperature of 50°K. The channel capacity of the overall link is computed in Table E-18.

This channel capacity is within 1 db of that required to support a CCIR quality video link using wideband FM (Table E-6), or from another point of view, is sufficient to permit digital data rates up to 100 megabits with bit error rates under 10^{-3} (Reference 27).

Point-to-point Lunar Communications via L_2

In this hypothetical system, an ATS-F&G type spacecraft is assumed to be located at L_2. The maximum frequency at which the 3-db mainbeam of a 30-foot dish at L_2 will give global coverage of the moon is about 800 MHz. Choosing this frequency and assuming 1 watt per channel, spacecraft transmit power, a gross estimate of channel capacity in a link to the lunar surface is computed in Table E-19. The up-link, assumed reciprocal, is not analyzed.

This channel capacity is essentially that required to meet military design objectives for FM voice communications (Tables E-7 and E-10). In terms of digital transmission, such a link could accommodate data rates in excess of a megabit with error rates under 10^{-3}.

Table E-18

Lunar-Earth Channel Capacity Using Relay at L_4 or L_5	
Parameter	Lunar-Earth Channel Capacity
Lunar base xmit power (10 watts)	+10.0 dbw
Xmit antenna gain (10-ft dish @ 8.0 GHz)	+45.5 db
Lunar base erp	+55.5 dbw
Path loss to libration point (384,000 km)	-222.5 db
Receive antenna gain (30-ft dish @ 8.0 GHz)	+54.7 db
C, received signal power at satellite	-112.3 dbw
η, satellite noise density (600°K)	-200.8 dbw/Hz
C/η, lunar-satellite channel capacity	+88.5 db-Hz
Satellite xmit power (10 watts)	+10.0 dbw
Xmit antenna gain (5 ft dish @ 7.3 GHz)	+33.7 db
Satellite erp	+43.7 dbw
Receive antenna gain (85 ft dish @ 7.3 GHz)	+63.5 db
C, received carrier power	-114.6 dbw
η, receiver noise density (50°K)	-211.6 dbw/Hz
C/η, satellite-earth channel capacity	+97.0 db-Hz
Overall link channel capacity	88.0 db-Hz

INTERPLANETARY MISSIONS

The material presented in this section is intended to indicate the RF power levels required for typical interplanetary missions using an ATS-F&G type transmitting antenna on the probe spacecraft. The enormous distances involved in such missions preclude real-time transmission of wideband data unless both high-gain antennas and substantial spacecraft power are available.

Three typical real-time digital signals are considered at widely varying data rates. They are vocoded speech at 2.4 kbps, 6-bit PCM speech at 48 kbps, and 4-bit PCM slow-scan TV at 480 kbps. A link frequency of 7.3 GH_Z is assumed, and the earth receiving system is taken to be an 85-foot dish (63 db gain at 7.3 GH_Z) and 50^oK system noise temperature.

Spacecraft RF power levels required to achieve the three data rates while maintaining a bit error rate under 10^{-3} are shown in Table E-20 for four interplanetary missions.

Table E-19

Channel Capacity of Point-to-point Lunar Communications System Using L_2

Parameter	Channel Capacity
Satellite xmit power	+0.0 dbw
Xmit antenna gain (30 ft dish @ 800 MHz)	+35.0 db
Satellite erp	+35.0 dbw
Path loss to lunar surface (64,000 km)	-187.1 db
Receive antenna gain (assuming 10 ft dish)	+25.6 db
C , received signal power	-126.5 dbw
η , noise density (assuming 1000^oK)	-198.6 dbw/Hz
C/η, channel capacity	+72.1 db-Hz
Overall link channel capacity (reciprocal up-link)	+69.1 db-Hz

Table E-20

Power Requirements for Several Interplanetary Missions Using ATS- Type Transmitting Antennas					
Mission	Distance (km)	Free Space Loss (db)	Power Required (dbw) 2.4 kbps	48 kbps	480 kbps
Venus-earth (min)	3.5×10^7	262	-26.3	-13.3	-3.3
Mars-earth (min)	5.3×10^7	264	-24.3	-11.3	-1.3
Venus-earth (max)	2.5×10^8	277	-11.3	+1.7	+11.7
Mars-earth (max)	5.4×10^8	284	-4.3	+8.7	+18.7

NOTE 1

Throughout this study, the parameter C/η, received signal power divided by (white, gauss) noise power per unit bandwidth, has been termed "channel capacity" and used extensively as a common denominator in analyzing and comparing a broad spectrum of communications systems. This note is included for the benefit of the reader who may not be familiar with this approach to characterizing communications system capability.

C/η, which is measured in units of time^{-1}, bandwidth, or some dimensionless quantity per unit time, is also sometimes called "noise bandwidth" (it is that bandwidth W in which the signal-to-noise power ratio $C/(\eta W)$ is unity), and has also been discussed as a "system capacity quotient" (Reference 31). The importance, both theoretical and practical, of the C/η ratio in communication system analysis and design may be appreciated by appeal to a simple argument involving Shannon's famous theorem from information theory (Reference 32).

For present purposes, this theorem may be stated non-rigorously as follows. Given a transmission channel characterized by a rectangular bandpass W, an average communication signal power C, and an average additive white gauss noise power N, the function

$$S(W, C, \eta) = W \log(1 + C/N)$$

defines the maximum information transmission rate which can be supported without error. The function or quantity S has the units of information per unit time, where the actual information measure is arbitrary and depends on the logarithm base, as yet unspecified.

Shannon's theorem does not tell one how to build equipment to approach this maximum transmission rate, nor does it spell out what other factors (transmission and encoding-coding delays) may be involved in realizing S. However, considering S as a function of W only (C and η, the noise power per unit bandwidth or spectral density considered fixed), the fact that S(W) is strictly monotonic increasing with W suggests that for fixed signal power C and constant spectral density η, S(W) can be maximized by using signaling or modulation techniques which use greater and greater bandwidth.

Working with Shannon's "natural units" of information, in the limit for large W, using the relationship

$$\ln(1 + x) \cong x, \qquad x << 1,$$

one has that

$$\lim_{W \to \infty} S(W) = S_\infty = C/\eta \qquad \text{natural units per second.}$$

Information theory, then, asserts that in a communications system with fixed power and fixed noise spectral density but variable bandwidth, the information transmission rate is ultimately limited by the ratio C/η and will be approached as one goes to larger and larger transmission bandwidths. Examples of signaling techniques which trade signaling bandwidth for performance are, of course, wideband FM and the various digital systems (PCM, delta modulation, etc.).

The Parameter C/η used in this study is thus seen to be a special value of Shannon's function, i.e., S_∞, the "infinite bandwidth channel capacity."

NOTE 2

This note is included to present in concise form the relationships between power density in watts/m^2 (or dbw/m^2), electric field strength in volts/m (or dbu) and the power density and field strength developed by a given satellite erp at geosynchronous altitude. This material has important application in the analysis of direct broadcast (TV or FM) satellite systems.

If E is the electric field intensity in volts/m, it can be expressed in dbu (decibels relative to one microvolt/m) by the formula

$$E_{dbu} = 20 \log_{10} E + 120. \qquad (1)$$

If E is the electric field intensity in volts/m, the power density P in watts per meter squared is given by the formula

$$P = \frac{E^2}{120\pi}. \qquad (2)$$

If E is the electric field intensity in volts/m, the power density in dbw per meter squared is given by the formula

$$P_{dbw/m^2} = 10 \log_{10}\left[\frac{E^2}{120\pi}\right] = 10 \log_{10} P. \qquad (3)$$

If W is the synchronous satellite erp in dbw, the power density produced at the ground is given by the formula

$$P_{dbw/m^2} = W + 10 \log_{10}\left[\frac{1}{4\pi R^2}\right] = W - 163.3, \qquad (4)$$

where the value of R used is the maximum slant range 41.1×10^3 km.

Figure E-4 plots the relations between formulas (1), (2), and (3).

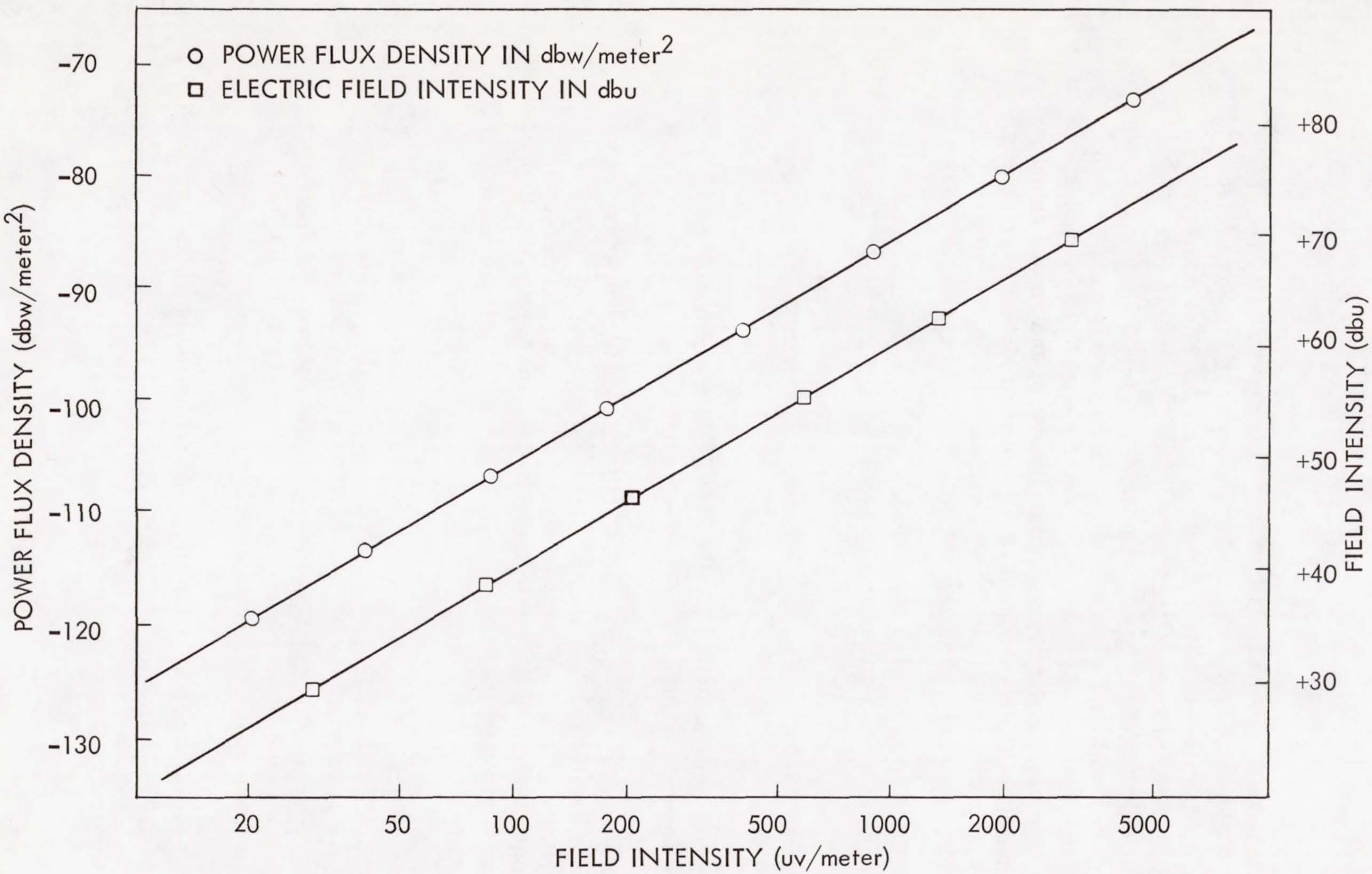

Figure E-4. Showing Interrelationships between μv/m, dbu, and dbw/m^2.

NOTE 3

In the discussion of the 860 MHz ATS-F&G FM-TV relay operation (System 5) proposed in the text, it was asserted that the wideband signal radiated from the spacecraft would not cause harmful interference to existing ground-based line-of-sight UHF TV relay systems (at channels 76 through 82) of the type discussed in Reference 3. The analysis given in this Note, while not a study in depth, does provide a useful quantitative estimate of the interference which would be produced by such an experiment. The basic assumptions on signal spectra and ground antenna discrimination follow closely the work of Medhurst on mutual interference between communications satellites and terrestial common -carrier microwave relay systems (Reference 33). The three key ideas are the following:

1. The wideband FM signal radiated from the spacecraft has a continuous gaussian spectrum with no spikes and, in particular, no residual carrier. This is assured by the relatively large deviation ratio of 5.4 (see Table E-6).

2. The receiving antenna of the ground-based TV relay operation is essentially an isotropic radiator for overhead signals.

3. In any 6-MHz sub-band of the 28.8-MHz Carson's rule RF bandwidth of the satellite signal, the filtered FM signal has all the essential characteristics of receiver thermal noise. If the spectral density of the satellite signal at the "receiver input" is substantially less than that of the receiver's own thermal noise, the satellite signal will not cause harmful interference to the terrestial system.

The only critical assumption in this analysis is that the signal radiated from the spacecraft is fully modulated at all times. An unmodulated ATS-F&G FM carrier at 860 MHz could cause very severe interference to any terrestial TV relay system operating on channels 78 or 79. For this reason it would be necessary to make some provision for fully modulating the spacecraft RF carrier at all times.

The (one-sided) spectral density function S(f) of the modulated spacecraft RF signal is modeled as

$$S(f) = \frac{1}{\sqrt{2\pi\Delta f^2_{rms}}} \exp\left[-\frac{(f-fo)^2}{2\,\Delta f^2_{rms}}\right]$$

where the mean of the distribution is

$$f_o = 860 \text{ MHz}$$

and the standard deviation Δf_{rms} may be estimated by using the fact that 99.9 percent of the area under the normal curve lies to the left of the 3 standard deviations point. Assuming that Carson's rule bandwidth of 28.8 MHz includes 99.9 percent of the total signal power,

$$\Delta f_{rms} = 4.8 \text{ MHz.}$$

The maximum value of the signal spectral density function occurs at 860 MHz and is

$$S(f)_{max} = S(f-fo) = \frac{1}{\sqrt{2\pi\Delta f^2_{rms}}} \quad ,$$

and may be seen to be down from the unmodulated carrier by an amount

$$10 \log_{10}\left[\frac{1}{\sqrt{2\pi\Delta f^2_{rms}}}\right] = 70.8 \text{ db.}$$

This is illustrated in Figure E-5.

By way of comparison, assuming a uniform distribution of signal power over the Carson's rule bandwidth would lead to a spectral density down

$$10 \log_{10}\left[28.8 \times 10^6\right] = 74.6 \text{ db,}$$

from the unmodulated carrier power. The more realistic gauss model is seen to be a more pessimistic one.

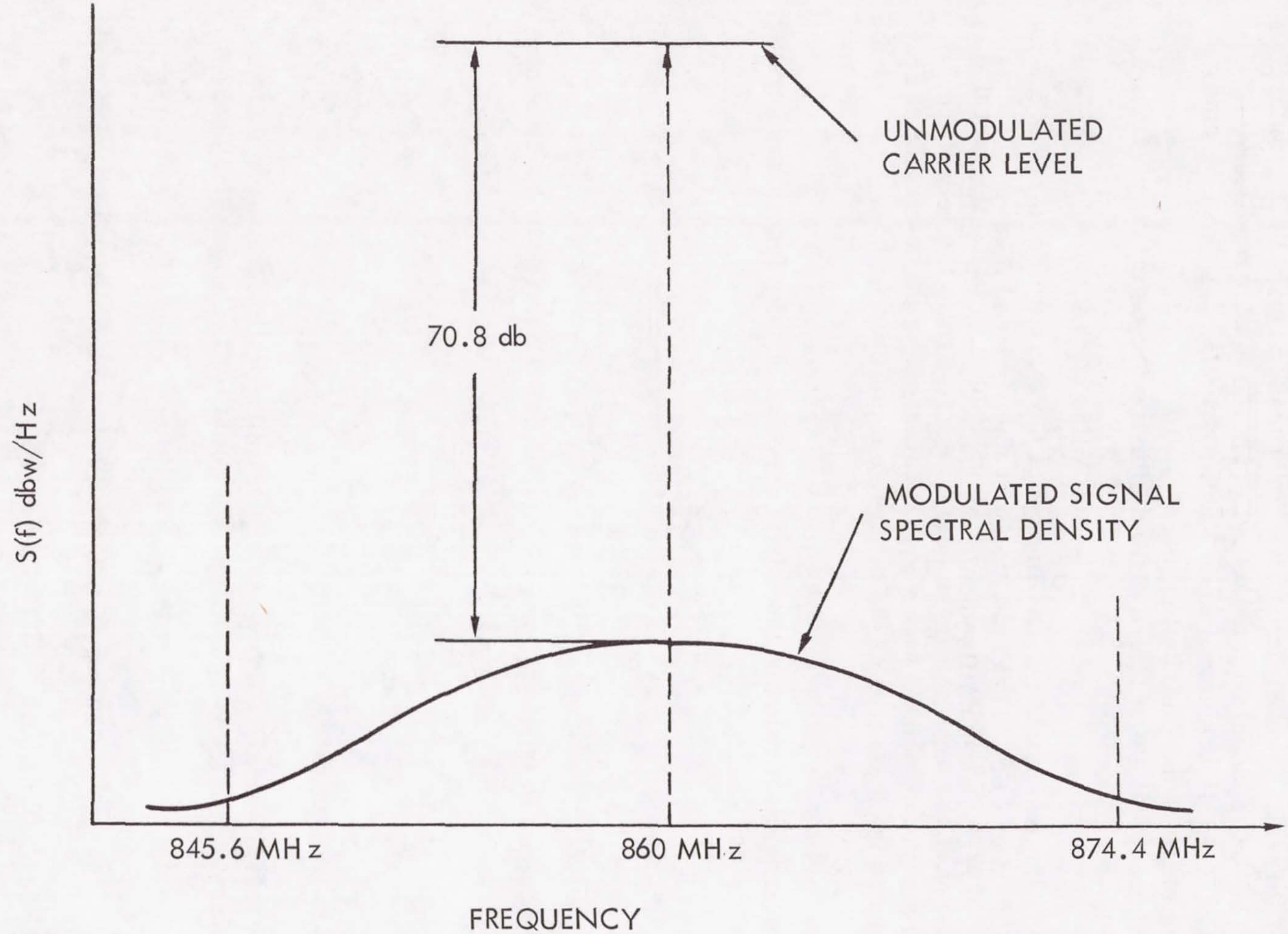

Figure E-5. Signal spectral density function, maximum value.

Drawing on Table E-6 and assuming an isotropic ground receiving antenna for the terrestial system (i.e., for "overhead" signals in the main-beam of the ATS-F&G spacecraft antenna pattern), the maximum received signal spectral density is calculated to be -200.5 dbw/Hz.

Assuming a typical noise figure of 10 db for the terrestial receiver, its thermal noise is -193.9 dbw/Hz.

Combining the two numbers, one finds that the terrestial service is degraded by less than 1 db. This indicates that the proposed ATS-F&G FM-TV relay operation at 860 MHz should not cause harmful interference to a terrestial UHF TV relay system with the parameters assumed.

NOTE 4

General

General Electric has proposed (Reference C) an ATS-F&G communications demonstration in which the audio sections of commercial-type UHF television sets would be used to receive high-quality FM program material broadcast from the satellite at 800 MHz. Their discussion includes a rough analysis which concludes that several tens of watts of satellite RF power would suffice to provide quality service working into unmodified conventional UHF TV receivers equipped with modest antennas.

The technical feasibility of FM sound broadcast to home TV sets, audio sections only, is considered in this Note. It is concluded that while the idea may be of interest for future broadcast satellite applications, General Electric's analysis is inaccurate, misleading, and in general technically inadequate. Specifically, it appears that the satellite RF power requirements discussed by General Electric are low by more than an order of magnitude, and that a truly high-quality FM broadcast system of this type would call for satellite RF levels in the kilowatt range.

This Note, therefore, has been included to identify and comment on what are viewed as several weak points in General Electric's analysis, and to clarify the matter of power requirements for FM sound broadcast to home-type TV receivers.

Television Signal Characteristics and Intercarrier Sound Detection

A number of different television transmission systems are in use in different parts of the world (Reference 2). The following remarks, based largely on Reference 34 apply in particular to the U. S. 525 line compatible color system using vestigial-sideband amplitude modulation (VSB-AM) for the visual signal and FM for the sound signal.

The received television signal (prior to conversion to IF and ignoring noise and other interference) may be described as the real part of a complex signal defined by

$$f_{rf}(t) = A\left[1 + K_1 V(t)\right]\exp\left[i2\pi f_c t\right] + \delta(t)$$
$$+ B\exp\left[i2\pi(fc + \Delta f)\,t + iK_2 \int s(t)dt\right], \qquad (1)$$

where

$v(t)$ = baseband video signal
$s(t)$ = baseband sound signal (pre-emphasized)
f_c = reference carrier frequency
Δf = 4. 5 MHz
K_1, K_2 = modulation constants
$\delta(t)$ = distortion term
$A/B \cong 1$ (NAB Handbook, page 1-206 - see Reference 3) .

The first term represents a pure double-sideband amplitude modulated transmission; the term $\delta(t)$ represents the distortion introduced by attenuating most of the lower sideband prior to transmission. Hence, the first two terms combined represent the VSB-AM signal actually transmitted. The third term is, of course, the FM sound signal. The essential characteristics of the sound transmission system are:

1. FM with 100 percent deviation of $\pm$ 25 KHz

2. Audio response flat 50 to 15, 000 Hz within limits and using 75 μ sec pre-emphasis of Figure E-6.

The frequency spectrum of the received signal is shown in Figure E-7.

In the earliest commercial TV receivers, detection of the visual and audio signals was accomplished independently, paralleling the way the signals were transmitted. After frequency conversion, a frequency discriminator was used to detect the sound signal, and an envelope detector was used to recover the picture signal. Independent tuned IF strips, separated by 4. 5 MHz, were used to avoid intermodulation of picture and sound. This scheme is illustrated in Figure E-8.

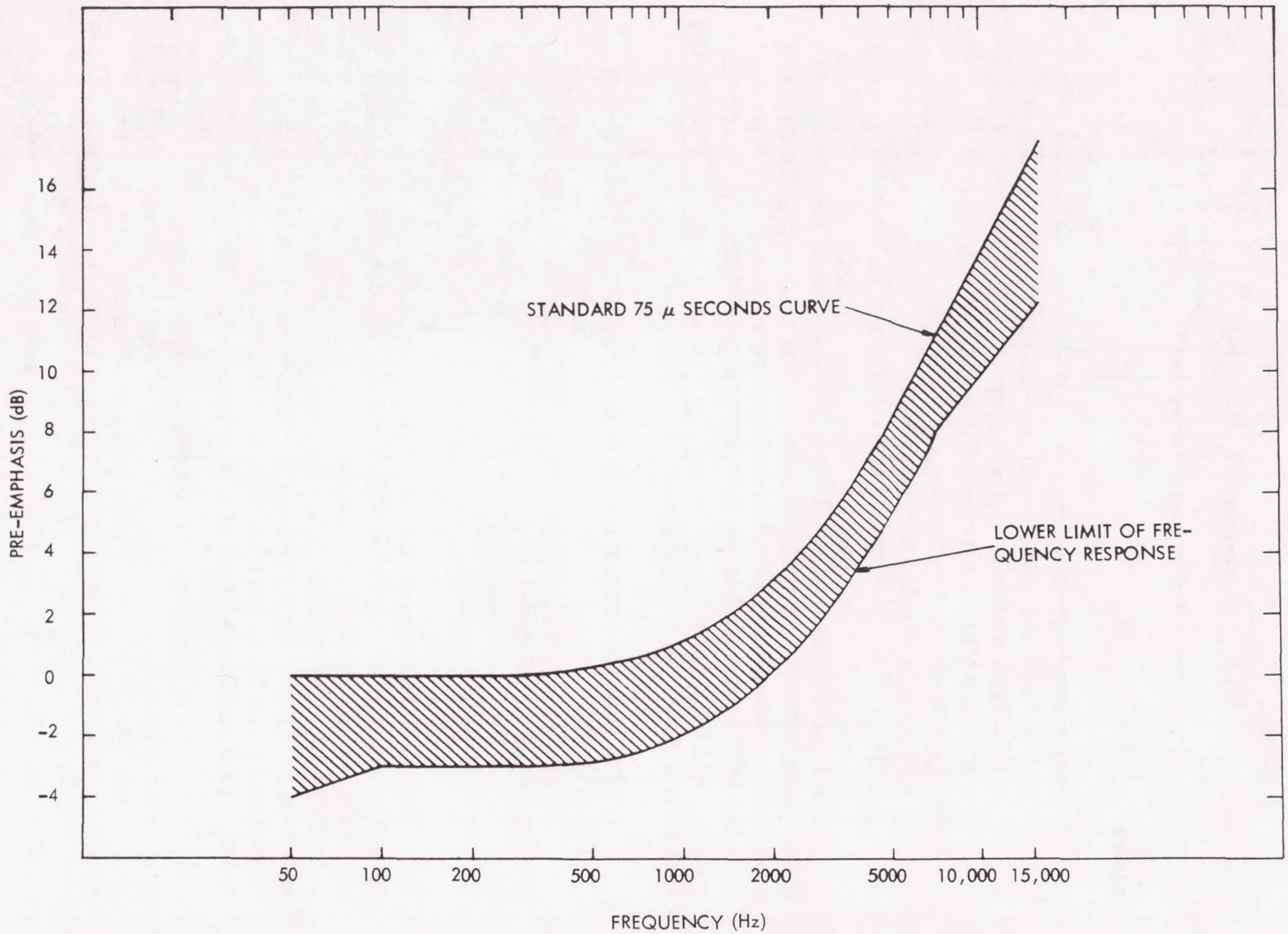

Figure E-6. Standard FM transmitter pre-emphasis characteristics for time constant of 75 μ seconds; frequency response limits given by shaded area.

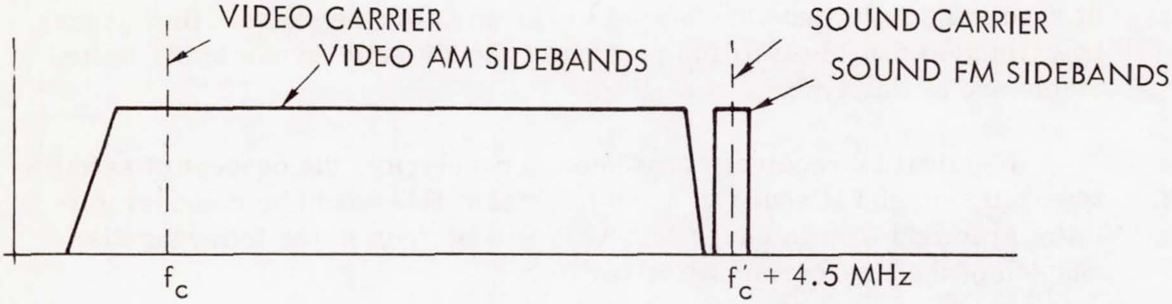

Figure E-7. Spectrum of video signal at RF.

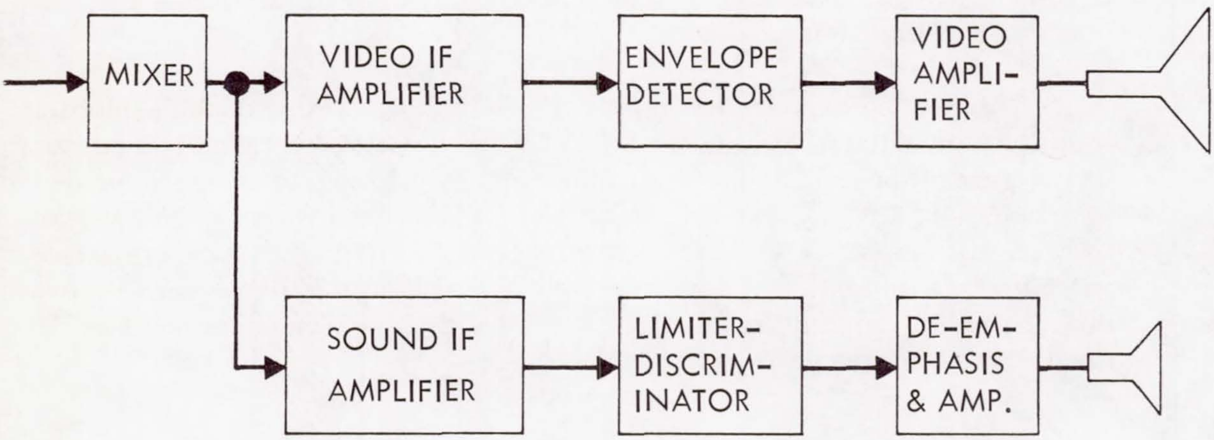

Figure E-8. Dual-IF type TV receiver.

Usually termed the dual IF design, this type of receiving system declined in popularity in the late 1940's in favor of what is generally called the intercarrier receiver. The virtual independence of picture and sound reception in the dual IF receiver was apparently something of an unwarranted luxury in mass produced home television sets, and the cheaper and simpler intercarrier design has been used in the vast majority of receivers built since the second world war. At the present time it may be estimated that close to 100 percent of the TV sets in use in the United States are of this type.

Had dual IF receivers continued in popularity, the concept of satellite transmission of FM sound to home television sets would be considerably more practical than in fact it is. This will be seen in the following discussion of the intercarrier receiver.

Briefly, the intercarrier receiver operates on the principle that the sound carrier, separated by 4.5 MHz from the video carrier (see Figure E-7) may be considered a sideband of the video carrier. If the sound carrier could be attenuated relative to the video carrier, it could be envelope-detected (with tolerable distortion and intermodulation) along with the video sidebands proper to yield an FM signal at an IF frequency of exactly 4.5 MHz, independent of local oscillator drift in the frequency converter of the home receiver.

This can be explained heuristically as follows. Pure double-sideband amplitude-modulated signals, with depth of modulation less than 100 percent can in general be detected without distortion by a properly designed envelope detector. Attenuation or removal of one sideband (leaving the carrier and one full sideband) produces a signal that cannot be envelope-detected without some distortion. The relative magnitude of this distortion, however, depends on the depth of modulation; for small depths of modulation, the distortion is correspondingly small.

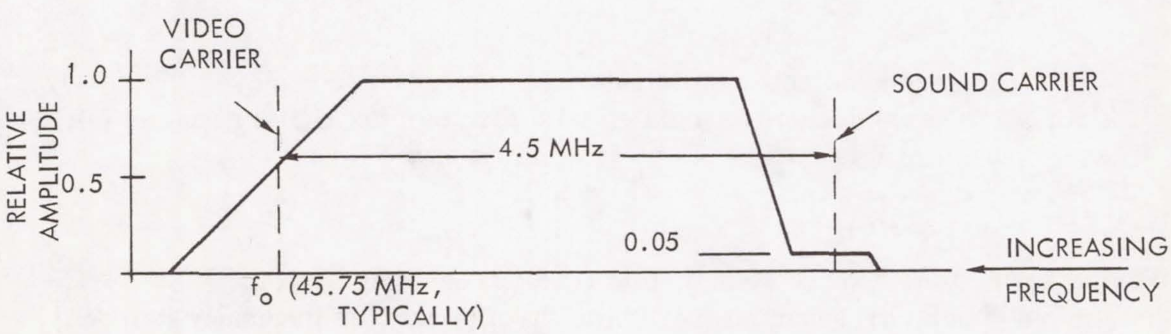

Figure E-9. Sound carrier attenuation scheme for intercarrier receivers.

The required attenuation of the sound carrier in the intercarrier receiver is accomplished by shaping the IF bandpass characteristic according to the scheme shown in Figure E-9.

The overall signal (at IF) may be described by (again using complex notation)

$$f_{if}(t) = \alpha\left[1 + K_1 V(t)\right] \exp\left[i2\pi f_o t\right] + \delta_2(t)$$
$$+ \beta \exp\left[i2\pi (f_o - \Delta f) - iK_2 \int s(t)\, dt\right], \tag{2}$$

where $\frac{\alpha}{\beta} \geq 10$

fo = 45.75 MHz (typical IF frequency)

$\delta_2(t)$ = distortion term

The conversion process has inverted the spectrum relative to that shown in Figure E-7 and the signal is ready to be envelope detected.

Equation 2 may be rewritten as

$$f_{if}(t) = \alpha\left\{1 + K_1 V(t) + \frac{\beta}{\alpha} \exp\left[i2\pi\Delta ft - iK_2 \int s(t)\, dt\right]\right\}\exp\left[i2\pi_{fo} t\right]$$
$$+ \delta_2(t) . \tag{3}$$

The second detector of the intercarrier receiver removes the slowly fluctuating real envelope associated with the complex signal given by (3), and a component of this real envelope is a signal of the form

$$\cos\left[2\pi\Delta ft + \Theta + K_2 \int s(t)\, dt\right],$$

where Θ is an arbitrary phase. Note that the carrier frequency is exactly 4.5 MHz, a value set and controlled at the transmitter. The fact that the sound signal carrier frequency after the second detector is independent of local oscillator drift in the receiver is a major advantage of the intercarrier audio system.

The presence of the FM sound carrier in the baseband video signal is a relatively unimportant distortion which can be remedied by a 4.5-MHz frequency trap. In any event, the effects of the sound carrier on the video signal are not severe because of the 20-db attenuation of the former relative to the latter, prior to the envelope detector. It is noted in passing that the attenuation of the sound carrier is accomplished at the receiver rather than at the transmitter to provide maximum audio signal-to-noise ratio. A block diagram of a typical intercarrier receiver is shown in Figure E-10.

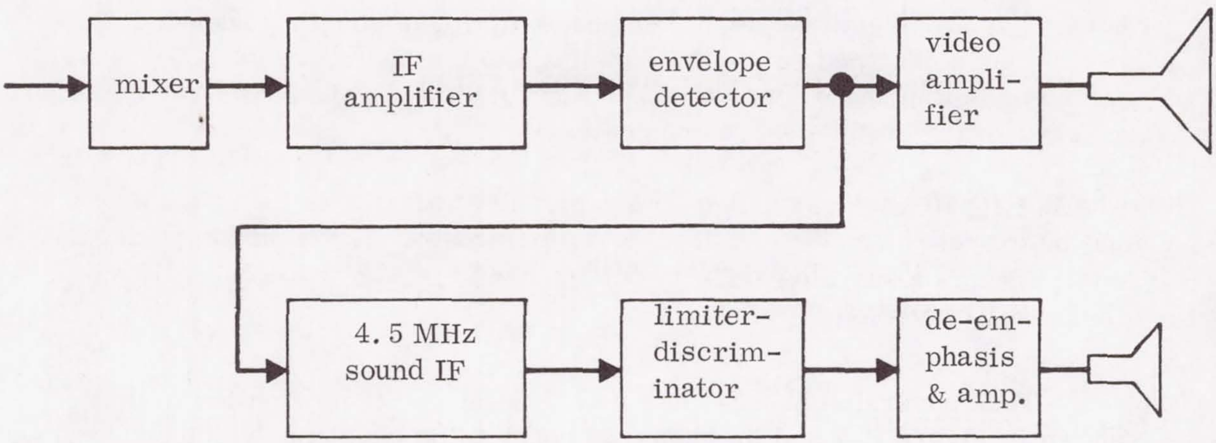

Figure E-10. Intercarrier-type TV receiver

Under the condition that the (unmodulated) video carrier-to-thermal-noise power ratio just before the envelope detector exceeds 10 db, it may be assumed that the conventional diode detector will successfully recover the real envelope associated with (3). The video carrier-to-noise ratio at that point may be computed by estimating the equivalent noise bandwidth of the receiver bandpass characteristic at 4.5 MHz. Certainly no qualitative error is made in this choice. The criterion for linear operation of the diode detector is then (Reference 35)

$$10 \log_{10} \left[\frac{\alpha}{2} / \eta_{if} \right] - 10 \log_{10} (4.5 \times 10^6) \geq 10 \text{ db.} \qquad (4)$$

Referencing everything to the input terminals of the receiver, the required video carrier-to-receiver noise density ratio in db is given by

$$\frac{Cv}{\eta}_{min} = 10 + 10 \operatorname{Log}_{10} (4.5 \times 10^6) + 6 \text{ db,} \qquad (5)$$

where 6 db has been included to take into consideration the 50 percent attenuation in video carrier amplitude due to the IF bandpass characteristic (see Figure E-9).

The criterion for successful envelope detector operation is then

$$\frac{Cv}{\eta} \geq 82.5 \text{ db} \qquad (6)$$

where Cv is the actual received (unmodulated) video carrier power and η is the receiver noise spectral density referred to the input terminal; i.e., the produce of Boltzmann's constant and the receiver equivalent receiver noise temperature (typically 4000°K or greater.)

For input video carrier-to-thermal-noise ratios less than 82.5 db, it must be expected that the envelope detection process will be accompanied by "small signal suppression" effects (Reference 35, leading to generally unsatisfactory operation.

Nothing has been said so far about the requirements on sound-carrier-to-noise-ratios. Only if the relationship in (6) is satisfied can one assume that the intercarrier detector will present the limiter-discriminator with a usable 4.5 MHz FM sound carrier. Assuming that (6) is satisfied, one may assume that the thermal noise spectral density in the vicinity of the 4.5 MHz carrier is not degraded by the second detector, and use elementary FM theory to relate the post-detection signal-to-noise design objective to the FM carrier-to-noise-density ratio at the input of the receiver.

We assume a design objective post-detection signal-to-noise ratio of 50 db and a modest gain of 10 db from the 75 μ seconds de-emphasis network after the discriminator (Reference 3, page 3-3). The modulating signal is a sinusoid producing a 100 percent swing of the carrier ($\pm$ 25 kHz, and the noise power is to be integrated over the audio bandpass of 50-15,000 Hz. The relationship between the output signal-to-noise ratio and the predetection sound carrier-to-noise-density ratio Cs/η is given by Reference 12.)

$$40 \text{ db} = 10 \text{ Log}_{10} \frac{3}{2} \left(\frac{\Delta f^2}{f_{max}} \right) \frac{Cs}{\eta f_{max}} \qquad (7)$$

With $\Delta f = 25$ kHz and $f_{max} = 15$ kHz, one has that we require

$$\frac{Cs}{\eta} \geq 75.5 \text{ db} \qquad (8)$$

Comparing (8) with (6), it is seen that the real problem in providing FM sound broadcast to home TV sets is to furnish the requisite unmodulated video carrier to drive the intercarrier second detector. The sound carrier power required is only one-fifth (7 db) of the required video carrier power.

The essential conclusions of the above analysis have been verified experimentally (Reference 36) by investigating the audio threshold characteristics of a high-quality intercarrier television receiver.

The test setup used is indicated in Figure E-11. Independent VHF signal generators were used to simulate the frequency modulated audio carrier and unmodulated video carrier of a Channel 2 television signal. Provision was made for modulating the audio carrier at a one kHz rate with a frequency deviation of ± 25 kHz.

Post-detection signal-plus-noise to noise ratios for the one kHz audio modulation were measured as a function of input audio carrier C_a for different values of input video carrier power C_v. The predetection noise was receiver thermal noise alone, and the post-detection noise bandwidth was essentially the 50-15,000 Hz bandpass (modified by 75 μ sec de-emphasis) used above.

The family of curves in Figure E-12 confirm, in particular, two points stressed in the above discussion. First, it is clear that both carriers must be supplied if intercarrier TV sets are to be used to receive FM sound broadcasts. Removal (or in this case, severe attenuation) of either carrier caused a virtual loss of output signal.

Second, for high-quality reception of FM sound broadcasts using intercarrier TV sets (i.e., output audio signal-to-noise ratios of the order of 50 db), the audio carrier power need only be a relatively small fraction of the video carrier power.

Figure E-12 also reveals an interesting and important phenomenon in the operation of intercarrier receivers which was not brought out in the simplified analysis above. Namely, it is evident that either the video signal or the audio signal can play the role of RF carrier in driving the second detector. This fact raises the following question. For a given design objective output signal-to-noise ratio, what is the ratio of video carrier to audio carrier power which will minimize total system RF power?

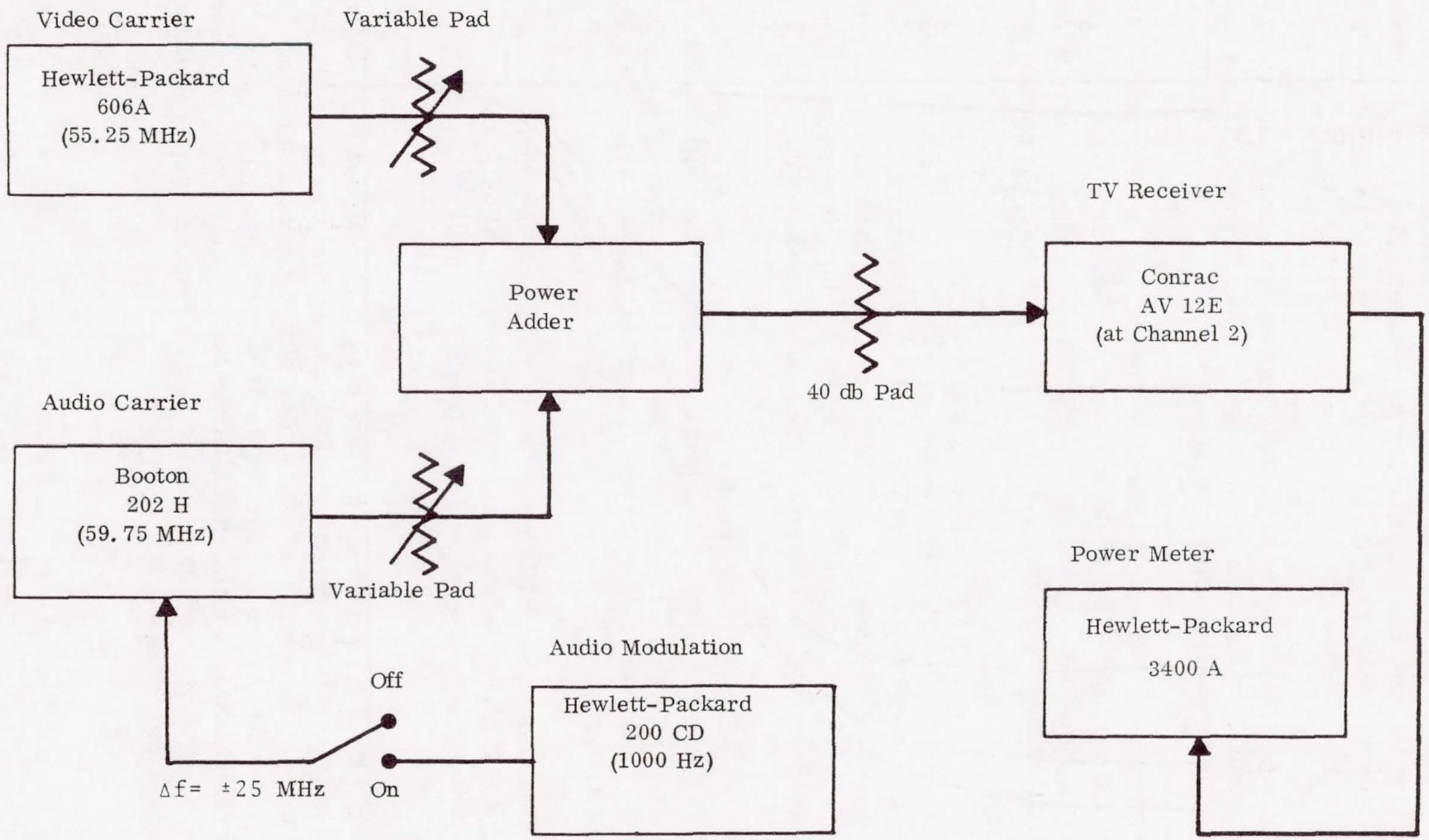

Figure E-11. Test setup.

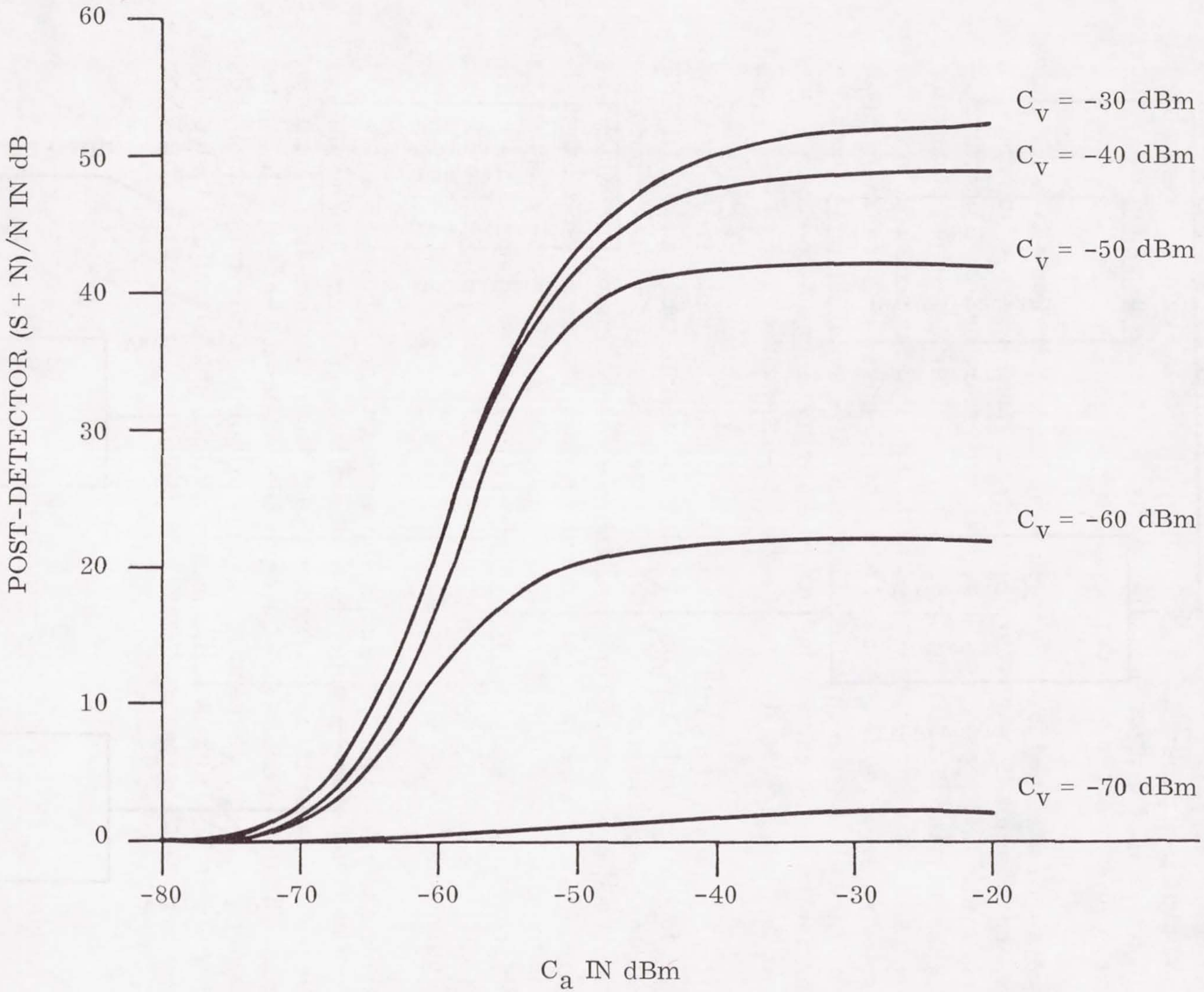

Figure E-12. Receiver audio threshold characteristics.

The curves of Figure E-12 indicate that for large output signal-to-noise ratios, the available power should be concentrated in the video carrier with only a small fraction allocated for the audio carrier. This arrangement will minimize RF power requirements in high-performance systems.

In systems designed for intermediate or low output signal-to-noise ratios, however, the situation is somewhat different. In these systems, the optimum division of available RF power would appear to be more or less a one-to-one ratio of video carrier to audio carrier. The reason for this is not clear from the above analysis, and is probably related to the fact that the second detector in intercarrier receivers may be viewed as a simple mixer rather than an envelope detector. At any rate, the question of optimum division of RF power between video carrier and audio carrier for the receiver tested has been shown to depend largely on the design objective output signal-to-noise ratio.

A final point which should be noted is that although the measurements were made with frequency deviations of ± 25 kHz, an attempt was made to drive the discriminator of the intercarrier receiver over the full ± 75 kHz deviations of conventional FM broadcast. This resulted in excessive distortion, which was to be expected in view of the fact that the audio section was designed to handle FM signals with deviations of only ± 25 kHz. Since the receiver under test was a high-performance unit, this is interpreted as an indication of the undesirability of using full ± 75 kHz deviations in systems of FM broadcast to home TV sets, audio sections only.

Avins (Reference 37) has studied the audio characteristics of intercarrier receivers in considerably greater depth than attempted here. His useful and carefully written paper takes into account such factors as receiver fine tuning and intermodulation effects which have not been discussed here at all. Avins' work, together with the elementary analysis and preliminary experimental data presented in this Note, tends to discourage serious consideration of UHF direct broadcast to home TV sets, audio sections only, from synchronous satellites with erps below the 70- to 80-dbw range.

Critique of General Electric's Analysis

Table E-21 below reproduces (in considerable extra detail) General Electric's analysis of the FM broadcast system using home TV sets, audio sections only (Table 8.7-3 of Reference C).

Table E-21

General Electric's Analysis	
Parameter	Link Calculation
P xmit	+ 17 dbw
Xmit antenna gain	+ 35 db
erp	+ 52 db
Path loss	- 184 db
Receive antenna gain	+ 3 db
Receive losses	- 2 db
C_a received sound carrier power	- 131 dbw
η, receiver noise density (10 db NF)	- 193 dbw/Hz
C_a/η, link channel capacity	+ 62 db-Hz
B_{rf}, system bandwidth (100 kHz)	+ 50 db-Hz
C_a/η, predetection carrier-to-noise ratio	+ 12 db
S/η, audio signal-to-noise ratio	+ 50 db

A number of objections can be raised to the above analysis, and several are detailed below.

1. No provision appears to have been made for supplying the video carrier (modulated or unmodulated) required to operate the envelope detector in intercarrier receivers. The indicated sound carrier-to-noise-density ratio in the General Electric analysis is 62 db-Hz, whereas it was shown above that an 82.5 db-Hz unmodulated video carrier-to-noise-density ratio was required merely to ensure linear operation of the intercarrier detector. Working back through General Electric's link calculation, this would mean that satellite transmit power would have to be increased by over 20 db to supply the requisite video carrier, bringing the system well into the kilowatt range.

2. The FM parameters chosen are apparently those for standard FM broadcast ($\Delta f = \pm 75$ kHz), not those used for TV sound broadcast ($\Delta f = \pm 25$ kHz), although General Electric has assumed an RF bandwidth consistent with the latter (100 kHz). Allowing 10 db for gain from the de-emphasis network, and assuming $\Delta f = \pm 75$ kHz, one gets essentially a 50-db post-detection signal-to-noise ratio. Using, however, the more appropriate TV sound parameter of $\Delta f = \pm 25$ kHz, and allowing 10 db de-emphasis gain, one gets a post-detection signal-to-noise ratio of about 40 db, still an excellent system, but not so good as truly high-quality FM design objectives of 50 to 55 db.

3. The system proposed (considering $\Delta f = \pm 25$ kHz) is operating just above the 10 db threshold of the discriminator. That is to say, there is no fade margin provided at all. Using the 200-kHz RF bandwidth consistent with General Electric's apparent Δf of ± 75 kHz, the proposed system is below FM threshold and will not work at all.

4. The indicated noise figure of 10 db is very optimistic for commercial-type UHF receivers at 800 MHz. A 12- or 14-db noise figure would be more in keeping with measured receiver characteristics (Reference 2).

Apart from their actual link analysis (shown above to be something less than satisfactory), General Electric goes on to present data of FM receiver performance as a function of field strengths at the receiver. This data, apparently, is substantially correct when applied to conventional FM receivers operating in the FM band of 88 to 108 MHz. The numbers given in Tables 8.7-4 and 8.7-5 of Reference C do not apply to reception of FM broadcasts at 800 MHz using unmodified intercarrier TV receivers.

The above remarks do not exhaust the criticisms which could be made (e.g., the formula on page 8.7-16 appears wrong). However, it is felt that the critical problem areas in the General Electric analysis have been identified and commented on in sufficient detail.

NOTE 5

With the exception of the brief analysis in NOTE 3, the question of mutual radio-frequency interference (RFI) between proposed ATS-F&G operations and existing terrestial communications systems has not been dealt with in any depth in either this report or by References A through D. This Note has been prepared for the dual purpose of summarizing existing regulations relating to such interference, and indicating how an ATS-F&G type spacecraft could be a valuable tool in a program of actual measurements of communications satellite-terrestial system mutual RFI.

Summary of existing regulations on mutual RFI

The general area of mutual RFI can be divided naturally into the four specific types listed below:

1. Satellite emissions interfering with terrestial systems

2. Satellite system ground transmitter emissions interfering with terrestial systems

3. Terrestial system emissions interfering with satellite system spacecraft receivers

4. Terrestial system emissions interfering with satellite system ground receivers.

Recognizing the need to protect existing terrestial servies from future satellite systems and vice-versa where band-sharing was involved for purposes of spectrum economy, the CCIR* took prompt action in adopting recommendations dealing with each of the four RFI types listed above. For example, Recommendation 358 of Reference 38 effectively set provisional upper limits on synchronous satellite erps for systems in the 1 to 10-GHz band wherever band-sharing was involved. Similarly, Recommendation 406 of the same document set upper limits on both the erp and RF power of line-of-sight radio relay system transmitters, effectively constraining such systems to use high-gain antennas

* International Radio Consultative Committee of the International Telecommunications Union

to avoid radiating signals which might interfere with satellite receivers on the same frequency. These two Recommendations, apparently based on the work of Medhurst (Reference 33), are reproduced below in essentially complete form.

CCIR Recommendation 358 (excerpted from Reference 38)

For communication-satellite systems which use wide-deviation frequency modulation, the power flux density set up at the surface of the Earth by the emissions of a satellite should not exceed:

-130 dbw/m^2 for all angles of arrival,

and that signals radiated by a satellite should be continuously modulated by a suitable waveform if necessary, so that the power flux density measured in any 4 kHz bandwidth particularly during periods of light loading should not exceed:

-149 dbw/m^2 per 4 kHz for all angles of arrival;

that for communication-satellite systems using other types of modulation, the power flux density set up at the Earth's surface by the emissions of a satellite, measured in any 4 kHz bandwidth, should not exceed:

-152 dbw/m^2 per 4 kHz for all angles of arrival.

CCIR Recommendation 406 (excerpted from Reference 38)

The maximum erp of any radio-relay system transmitter and its associated antenna should not exceed 55 dbw; the power delivered to the antenna input by any transmitter should not exceed 13 dbw.

As regards mutual RFI of types (2) and (4), no specific upper limits were set on erps or RF levels (except as already contained in Recommendation 406) because such RFI is so strongly dependent on the physical proximity of the interfering and interfered with systems. A so-called "coordination distance" technique was recommended for locating satellite ground terminals relative to terrestial radio-relay systems and vice-versa, rather than specific numbers.

Below 1.0 GHz, no limits were placed on satellite erps, in part because band-sharing of space and terrestial services was not anticipated except in certain common-carrier microwave bands above 1.0 GHz such as 3.7-4.2 GHz (used for satellite-to-earth transmissions) and 5.925 - 6.625 GHz (used for earth-to-satellite transmissions). In particular, no recommendations concerning broadcast satellites (voice or TV) were made at all, although the subject was discussed in several study reports included in Reference 38.

It is readily seen that the several frequencies considered for ATS-F&G communications operations either fall below 1.0 GHz or, as with the S-band and X-band cases, do not involve band-sharing with existing terrestial services, and hence no significant mutual RFI problems are anticipated with the frequencies presently under consideration.

ATS-F&G and RFI Measurement Experiments

The provisional limits given in CCIR Recommendations 358 and 406, reproduced above, may be considered extremely conservative. While generally compatible with first generation satellite systems (Relay, Syncom, Telstar, etc.) which employed sensitive (high gain/low noise) ground receiving facilities, it is readily seen that any attempt to design wideband communications satellite links to small terminals on a shared-frequency basis is out of the question if one is to comply with these CCIR recommendations. For example, the FM-TV relay system discussed in Table A-16 above (System 4 - 7.3 GHz) lays down a total flux density of approximately -100 dbw/m^2 as compared with the CCIR limit of -130 dbw/m^2. (This calculation may be made using the data in Tables A-1 through A-3 and the information in Note 2). In terms of flux density per 4 kHz, using the method of Note 3, one finds the number -134 dbw/m^2/4 kHz, as compared with the CCIR limit of -149 dbw/m^2/4 kHz. If modulation conditions are such that the gaussian distribution of Figure A-5 is not a valid model of the FM spectrum, the flux density per 4 kHz may significantly exceed -134 dbw/m^2, and render the problem even more severe.

The proposed ATS-F&G service does not involve band-sharing with terrestial services, and hence these large flux densities should not cause any problems. It is obvious, however, that extrapolation of System 4 into one of the neighboring shared bands could be a difficult matter.

It has been realized for some time that Recommendation 358, and particularly its limitation on total satellite erp, is unduly restrictive. It appears that this recommendation will be replaced by a single limitation on satellite erp per 4 kHz. Reference 39 contains the following single recommendation:

That in frequency bands in the range 1 to 10 GHz shared between communication-satellite systems and line-of-sight radio-relay systems, the maximum power flux density produced at the earth's surface by emissions from a space station for all conditions and methods of modulation, should not exceed

$$(-152 + \frac{\theta}{15})\ \text{dbw/m}^2 \text{ in any 4 kHz band where}$$

θ is the angle of arrival of the wave in degrees above the horizontal; that the aforementioned limit should be assumed to relate to the power flux density under free-space propagation conditions.

For an "overhead" satellite ($\theta = 90$ degrees) the permissible flux density would be -146 dbw/m^2/4 kHz, a figure which ATS-F&G System 4 FM-TV emissions would exceed by many db.

Both the original CCIR recommendation and the new one to replace it would appear, then, to inhibit seriously the development of educational-type TV relay systems using satellites (Reference 40) and similiar wideband operations to small terminals wherever band-sharing with terrestial microwave systems is involved. In as much as the CCIR limits are based on statistical analyses involving many "worst-case" assumptions rather than on actual satellite-terrestial system RFI measurements, it would be desirable to conduct such measurements to see if the present limits could not be relaxed further. This general lack of empirical data on such interference is due, of course, to the fact that no satellite has yet been flown for the specific purpose of measuring mutual RFI. As noted above, existing communications satellites under modulated conditions develop flux densities at the ground which are not significantly at variance with present CCIR limits, and hence there has been no real opportunity to make systematic RFI measurements.

An ATS-F&G type spacecraft equipped with only a conventional RF transmitter (i.e., +10 to +13 dbw) could, as shown above, develop modulated condition flux densities at the ground orders of magnitude greater than those specified by the CCIR. Because the ATS-F&G beam is extremely narrow at microwave frequencies (Table E-1), these high flux densities could be localized to a relatively small area on the surface of the earth for making controlled RFI measurements.

Similarly, if the ATS-F&G spacecraft had a wideband (hundreds of MHz) linear frequency translating repeater receiving, say, in the 4.4- to 4.7- GHz common-carrier band, it would be possible to monitor at the ground any interfering signals on the ground-to-satellite up-link to determine the severity of this type of RFI.

An ATS-F&G experiment for RFI measurements would require somewhat different communications subsystems than those discussed in Section VII-D of this Analytic Report. No problems are anticipated in designing appropriate subsystems, and to the extent that there exists a need for actual data on communications satellite - terrestial system mutual RFI, it is recommended that such an experiment be studied for inclusion in the ATS-F&G flight program.

REFERENCES

Lettered References

A. "ATS-4 GSFC Concept Design Study," Document X-730-67-10, GSFC, January 1967.

B. "ATS-4 Study Program Final Report," Contract NASW-1411, Fairchild-Hiller Space Systems Division, Volume 6; December 1966.

C. "Final Study Report," Contract NASW-1410, General Electric Spacecraft Department, Volume I, Book 3; November 1966.

D. "Advance Study of an Applications Technology Satellite (ATS-4) Mission, Final Report," contract NASW-1214, Lockheed Missiles and Space Company, November 1966.

Numbered References

1. "Technical and Cost Factors that Affect Television Reception from a Synchronous Satellite." Final Report, Contract NASW-1305, Jansky and Bailey Systems Engineering Department, Atlantic Research Corp., June 30, 1966.

2. R. G. Gould, "TV Broadcast from an Earth Satellite," IRE Trans. Comm. Sys., Vol. CS-10, No. 2, June 1962.

3. B. W. St. Clair, "UHF Television Translators," NAB Engineering Handbook, A. P. Walker, McGraw-Hill, New York, 1960.

4. Unpublished work of J. E. Miller, Communications Research Branch, Goddard Space Flight Center.

5. Transmission Systems for Communications, Third Edition, Bell Telephone Laboratories, Winston-Salem, 1964 (page 468).

6. W. A. Rheinfelder, CATV System Engineering, Tab Books, Thurmont, Maryland, January 1966.

7. H. T. Head, "The Measurement of FM and TV Field Strength," NAB Engineering Handbook (see Reference 3).

8. A. G. Cooley, "The Possible Use of Facsimile in Broadcasting," NAB Engineering Handbook, (see Reference 3).

9. "Military Communication System Technical Standard," MIL-STD-188B, Washington, D. C., February 24, 1964.

10. H. C. Franke, "Noise Measurements on Telephone Circuits," Tele-Tech and Electronic Industries, March 1955.

11. P. J. Heffernan, "Analysis of Non-linear Noise in FDM Telephony Transmission over an SSB-PM Satellite Communication System," NASA TN-2365, Washington, D. C., July 1964.

12. J. Fagot and P. Magne, Frequency Modulation Theory, Pergamon Press, New York, 1961 (Eq. 2, 112, P. 118).

13. W. Doyle, "Crosstalk of Frequency-Multiplexed Signals in Saturating Amplifiers," RAND Memorandum RM-3576-NASA, April 1963.

14. E. M. Rizzoni, "Compandor Loading and Noise Improvement in Frequency Division Multiplex Radio-Relay Systems," Proc. IRE, Vol. 48, No. 2, February 1960.

15. W. J. Bray, "The Standardization of International Microwave Radio-Relay Systems," Proc. IEE., Paper No. 3412E, March 1961.

16. E. D. Sunde, "Intermodulation Distortion in Multicarrier FM Systems," IEEE Int. Con. Rec., Part 2, March 1965.

17. W. C. Babcock, "Intermodulation Interference in Radio Systems," BSTJ, January 1953.

18. Frequency Allocations, RCA Frequency Bureau, Washington, D. C. December 1965.

19. "VHF Aircraft Satellite Relay - Final Report of Flight Tests," Bendix Radio Division, Baltimore, Maryland, April 1965.

20. "VHF Repeater Experiment," Final Report, NASA Contract NO. 5-9593, Hughes Aircraft Company, February 1, 1967.

21. W. Sollfrey, "Earth Coverage patterns with High-Gain Antennas on Stationary Satellites," RAND Memorandum RM-4894-NASA, February 1966.

22. O. J. DeZoute, "Future Satellite Communication Subsystem Investigation," FAA Report No. RD-65-77, July 1965.

23. "First Quarterly Report," Contract NAS 5-10255, Technical Communications Corporation, October 1966.

24. Unpublished results of measurements made by M.I.T. Lincoln Labs with their LES 3 and 4 satellites.

25. J. R. Cressey and G. D. Hogan, "The Interrogation, Recording and Location System Experiment." Proc. National Telemetry Conference, Paper WA 61, 1965.

26. NDHS Pass Summary, Interrogation Orbit #105R (unpublished data sheets, Tracking Systems Directorate, GSFC).

27. J. G. Lawton, "Comparison of Binary Data Transmission Systems," Proceedings of the Second National Convention on Military Electronics (MIL-E-CON), Washington, D. C., June 1958, P. 54.

28. C. Laughlin, G. Hilton, R. Hollenbaugh and R. Lavigne, "Meteorological Experiment Using the Omega System for Position Location," NASA Document X-731-65-416, GSFC, October 1965.

29. L. Steg and E. Shoemaker, "Libration Point Satellites," in A Review of Space Research, NAS-NRC Publ. 1079, 1962.

30. D. Jamison, "Synchronous Communication Satellites of the Moon," Proc. IEEE, Vol. 54, No. 7, July 1966.

31. J. W. Schwartz, J. M. Aein, and J. Kaiser, "Modulation Techniques for Multiple Access to a Hard-Limiting Satellite Repeater," Proc. IEEE, Vol. 5, No. 5, May 1966.

32. C. E. Shannon, Mathematical Theory of Communication, University of Illinois Press, Urbana, 1949.

33. R. G. Medhurst and J. H. Roberts, "Expected Interference Levels due to interactions between Line-of-Sight Radio Relay Systems and Broadband Satellite Systems," Proc. IEE, Vol. 111, No. 3, March 1964, P. 519.

34. D. L. Fink, Television Engineering, McGraw-Hill, New York, (second edition) 1952.

35. W. B. Davenport and W. L. Root, Random Signals and Noise, McGraw-Hill, New York, 1958; pp. 266 and 308.

36. GSFC internal memorandum, P. J. Heffernan to M. B. Weinreb, February 1, 1967.

37. J. Avins, "Sound Signal-to-Noise Ratios in Intercarried Sound Television Receivers," IEEE Trans. on Broadcast and Television Receivers, Vol. BTR-9, July 1963, p. 9.

38. "Documents of the Tenth Plenery Assembly" CCIR, Geneva, (Vol. IV) 1963.

39. Documents submitted by Study Group IV to the Eleventh Plenary Assembly of the CCIR, Oslo, (Document IV/1034-E), July 1966.

40. FCC Docket 16495.

APPENDIX F

ELECTRIC PROPULSION EXPERIMENT

The electric propulsion experiment will serve to demonstrate the in-space operational feasibility of an advanced colloid microthruster. Colloid thrusters fall within the category of electrostatic thrusters because the exhaust beam kinetic energy is obtained by the acceleration of charged particles within the electrostatic fields maintained by an accelerating electrode structure. Colloid thrusters differ from ion engines in that the charged particles are multimolecular rather than atomic.

Of the several colloid thruster concepts which have been experimentally investigated over the past 7 years, only one has yielded significant performance. The characteristic feature of that concept is the use of a charged particle source, which employs the phenomenon of electrostatic spraying of liquids. The propellant, a low vapor pressure, moderately conductive liquid, is sprayed from the tips of metallic capillary tubes which are maintained at a high electrical potential. The capillary potential imposes large electrical stresses in the liquid meniscus at the capillary tip, causing the meniscus to rupture and eject extremely small, charged droplets. The very low power required for this charged particle generation technique is the feature which makes colloid thrusters highly competitive with ion engines, especially at thrust levels below 1 millipound.

Conventionally, electrostatic thruster operation consists of three separate processes: (1) charged-particle generation, (2) acceleration, and (3) exhaust beam neutralization. Within the past 2 years, the state-of-the-art in charged particle generation by electrostatic spraying has advanced to the point where the particle formation and acceleration process can be accomplished simultaneously. The advancement has eliminated the requirement for an accelerating electrode structure and, with it, the problem of accelerating electrode erosion, which has been one of the more serious developmental problems of ion engines. There is presently under development a colloid microthruster, which uses a thermionic neutralizer. This microthruster is scheduled for experimental test aboard a DOD satellite and promises a power requirement of approximately one-fifth that of ion engines of comparable thrust level. Nearly half its power is required for the neutralizer.

An advanced, autoneutralizer version of the colloid microthruster produces positively and negatively charged particles simultaneously, but from adjacent capillary tubes, thereby eliminating the requirement for a thermionic neutralizer and reducing the power requirement by almost one-half. There are presently two Air Force sponsored research efforts devoted to this thruster concept.

The primary object of this experiment aboard the ATS-F&G is to determine the adequacy of the autoneutralizer characteristic of the advanced microthruster. This determination can be made implicitly from a measurement of the thrust output. The sizing (determination of the thrust output) and location of the experiment aboard the spacecraft must be consistent with the thrust measurement requirement. If desired, the thruster may be sized to serve an operational function such as north-south station-keeping. A thrust level of 200 micropounds would require less than 20 watts (perhaps as few as 10 watts) at 800 to 1000 seconds specific impulse. The propellant requirement would be 6-1/2 to 8 pounds per year of continuous operation. The total package weight could be expected to be 15 to 20 pounds.

Summarizing, the object of the experiment is to test the effectiveness of an autoneutralizer (defined in the text) as compared to the effectiveness of a thermionic neutralizer. If the operation of the autoneutralizer is found to compare favorably it will offer the advantages of low power consumption and the elimination of problems associated with accelerating-electrode erosion.

APPENDIX G

RADIO INTERFEROMETER ATTITUDE SENSOR EXPERIMENT

INTRODUCTION

The radio interferometer is defined by the mission study work statement as follows:

> An interferometer system configuration, geometry, and electrical/mechanical design will be selected so as to fully demonstrate the capabilities and limitations of an on-board interferometer, as a spacecraft attitude determination device. The interferometer antenna system will operate in a frequency range consistent with the attainment of the maximum resolution and accuracy performance characteristics required for the spacecraft orientation control system.

The RF interferometer can provide the spacecraft with a highly ac curate attitude sensor with a very wide field of view. Since the mission requires demonstration of attitude control accuracy of ± 0.1 degree in all three axes, the sensor must be somewhat more accurate. Studies have shown that an interferometer accuracy of 0.03 degree with a field of view of 35 degrees can be achieved within the ATS-F&G time frame. In addition to attitude determination, the interferometer will also measure pointing angles to the ground stations. These pointing angles will aid in steering the beam of the parabolic reflector. This experimental interferometer sensor will also serve as a backup system to the conventional attitude sensors.

GENERAL DESCRIPTION

The basic interferometer consists of two antennas separated by a known distance and aligned in a plane normal to the yaw axis of the spacecraft. When the antennas are illuminated by RF energy from a ground station, a phase difference occurs between the signals in the two antennas. (The phase difference will be zero if the ground station is contained in the plane which perpendicularly bisects the line joining the two antennas.) By measuring this phase difference, the angle at which the signal is incident upon the interferometer baseline can be determined. By mounting two basic interferometers orthogonal to each other and to the spacecraft yaw axis, the angle of incidence with respect to the yaw axis can be measured. It is necessary to measure angles of incidence from two widely separated ground stations and to know the spacecraft ephemeris in order to compute the spacecraft attitude.

Due to the experimental nature of the interferometer and the effort to save spacecraft weight, the system will not have on-board computation facilities. The angle data from the phase detectors will be telemetered to the ground where the attitude will be computed. An on-axis error signal will be provided to the control system for closed-loop spacecraft attitude control.

GSFC/CUBIC CORPORATION/LOCKHEED MISSILES AND SPACE COMPANY CONCEPT

The interferometer system proposed by Lockheed is very similar to the system being developed by Cubic Corporation under contract to the GSFC Communications Research Branch. This is a receive type system, employing orthogonal baselines and coarse (2 wavelengths) and fine (16 wavelengths) spacings. The system is designed to operate with two ground stations separated in frequency by about 15 MHz. Using broadband circuitry both signals are received by each channel and converted to intermediate frequencies. A double local oscillator produces two signals separated by 10 kHz. One frequency is mixed with the signal in the reference channel and the other frequency is mixed with the signal in the measurement channel. Each channel contains two signals separated by 15 MHz. These are added and filtered as shown in Figure G-1. The output of the detector is a 10 kHz signal with a phase relationship to the 10 kHz reference signal which is proportional to the phase relationship of the microwave signals at the antenna apertures. A digital phase locked loop counts the relative phase of each channel. The digital signal is then telemetered to the ground for additional processing. This system is capable of providing an accuracy of 0.03 degree and because two ground stations are used, the system provides a complete attitude determination.

The ground stations for the GSFC concept operate at 8.4 GHz. These ground stations employ a 3-foot diameter parabolic reflector with a manually positioned reflector feed at 300 watts of RF power output. The Lockheed concept uses a 10-foot diameter reflector at 5 GHz and 1 kw of RF power. Initial calibration of the interferometer requires use of the star field scanner.

The monopulse system will provide boresight recalibration after the 30-foot reflector has been evaluated, however, the interferometer will be required during reflector evaluation.

FAIRCHILD-HILLER SYSTEM

Of the four different interferometer systems discussed in the Fairchild-Hiller report, one is chosen as best and will be reviewed herein. This system employs five antennas forming two orthogonal baselines. Each baseline contains a fine antenna and a coarse antenna. A reference antenna is located at the vertex of the two baselines. For an operating frequency of 8 GHz, the fine element spacing is 34 inches and coarse spacing is 4.5 inches. The antennas are mounted on a superinsulated aluminum box beam for mechanical and thermal stability. The outputs from the antenna elements are processed through a mixer to an IF of 30 MHz where a switch selects two of the signals to be connected to the phase meter. The phase meter, after amplifying the two signals, are processed through a mixer to give an output frequency of 100 kHz. A zero crossing detector in the reference channel starts a counter. A similar detector in the other channel stops the counter. With a clock rate of 36 MHz the system can resolve 1 electrical degree. With a baseline of 34 inches (23 λ), the system resolution is approximately 0.04 degree of space angle. Control and sequencing circuitry is provided to step the IF switch to each antenna element and to switch to another L.O. frequency so that a second ground station may be accommodated. The stepping rate is 2 kHz.

Rigid coaxial transmission line is used for interconnecting microwave circuitry. Antennas have a rectangular aperture tapering to a circular throat and then into a rectangular-to-coaxial transition segment. The RF mixers will use Schottky diodes to achieve an improvement in noise performance over conventional diodes. IF amplifiers must be phase matched.

The ground station required is a 40-foot diameter dish with a circularly polarized feed and 10 kw of RF power. This results in a system signal-to-noise ratio of 30 to 40 db.

The antenna configuration and operating frequency of the Fairchild-Hiller system are approximately the same as the GSFC/Cubic/Lockheed system. However, because of their choice of an IF bandwidth of 100 kHz, the ERP of the ground station is quite large (i.e., 97 dbw vs 52 dbw for the GSFC approach). Because of the low dynamics which the spacecraft will experience and the necessary transmitter stability, a bandwidth of approximately 100 Hz would be quite reasonable. This would result in a 30-db reduction in required transmitter ERP.

A disadvantage of the 100-kHz base frequency is that a clock rate of 36 MHz is necessary for the required resolution. Clock rates of this magnitude dictate the use of current mode logic which is more complex and requires more power than lower frequency logic.

It seems that the switching arrangement proposed by Fairchild-Hiller is not as reliable as a hardwired system. Because of the need for sequencing and switch driving circuitry the weight and volume savings will be minimal. The switching scheme does not have the redundancy inherent in the hardwired, multiple receiver approach. Phase matching of the IF circuitry is also made difficult with the switching approach.

GENERAL ELECTRIC SYSTEM

The system proposed by General Electric is a receive type system operating at a frequency of 10 GHz. The fine antenna system consists of four antennas on two orthogonal baselines. The antennas on each baseline are separated by 46.2 inches (39 wavelengths). The coarse system also consists of four separate antennas similarly mounted, separated by 2.7 inches (2.3 wavelengths). This results in a field of view of 23 degrees and an accuracy of 0.015 degree. A pilot tone, separated from the carrier frequency by 10 kHz, is added to each channel ahead of the first mixers. In each channel the two signals are fed to a double conversion receiver and the beat signal of 10 kHz is detected. The relative phase of the 10-kHz signal in each channel is proportional to the relative phase of the 10-GHz signal at each antenna. The 10-kHz signals from opposite channels are processed through zero crossing detectors and counter circuitry to yield a digital signal proportional to the pointing angle to the transmitter. The counter operates at a clock rate of 2.5 MHz. Signals will be furnished to the spacecraft control system for attitude control.

While it appears that the General Electric pilot tone concept will work, it will also add unnecessary complexity to the system. With good circuit design the phase stability of the IF portion of the receivers can be held to much less than 1 electrical degree. The principal sources of error are as follows:

1. Differential phase shift in the microwave circuitry due to thermal/mechanical effects

2. Alignment error of the antennas

3. Signal-to-noise ratio

4. Electro-magnetic coupling to structures in the vicinity of the antennas.

By adding the additional pilot tone circuitry, General Electric has only increased the chance of microwave differential phase shift.

By using a separate receiver for each channel, General Electric has enhanced the reliability of the system. General Electric has provided for a better phase match between the various elements by hard connecting each receiver, as opposed to switching receivers to the antenna or mixers.

It seems that General Electric has not adequately identified problem areas associated with the antenna. Maintaining antenna phase symmetry across the interferometer field of view and decoupling the antennas from adjacent structures are very important considerations. The General Electric system requires time synchronization of the ground transmitters, which is to be avoided if possible.

APPENDIX H

SCANNING CELESTIAL ATTITUDE DETERMINATION SYSTEM (SCADS) EXPERIMENT

INTRODUCTION

A star mapping system is proposed to provide a simple means for determining three-axis satellite attitude information for ATS-F&G during the transfer elipse mode, circular orbit, and the three-axis stabilized mode. The SCADS concept consists of a single on-board sensor head and a ground-based computer data reduction system. The sensor head, by observing the known star field, provides signals from which the star field can be positively identified and three-axis vehicle attitude information derived.

The primary objective of the experiment is to utilize SCADS to provide three-axis attitude to within 0.1 degree or better and spin period to 0.03 percent during the transfer and synchronous-spin-stabilized orbital phases. The attitude data provided by SCADS will be used to supply the direction of the incremental velocity necessary to produce the synchronous orbit. A second objective of this experiment is to utilize SCADS to provide three-axis attitude data during the three-axis controlled synchronous mode to within 0.01 degree. The attitude data will be used to evaluate the control system performance.

GENERAL DESCRIPTION

The basic scanning instrument is comprised of a lens, an opaque reticle with a hairline radial slit, and a photodetector. The reticle will be rotated by the ATS-F&G spacecraft during the spin stabilized modes and by a motor during the three-axis synchronous mode. As the reticle rotates, the instantaneous star image within the optical field of view will pass through the slit and impinge onto the photodetector. In this manner, the photodetector produces an output for each star viewed by the scanning instrument.

Computerized Star Identification

The first step in the computer solution to determine attitude is that of star identification. The star identification problem consists of establishing a pairing of a transit time with the name (number) of the star which furnishes that transit. The right ascension and declination of stars brighter than a threshold setting controlled by ground command are examined to ascertain which of these stars could yield a measured transit time. The relative star magnitudes assist in the star identification when the assumed pointing direction is unknown. When the approximate pointing direction is given (within 5 to 10 degrees), the relative star magnitudes are not required. After the star identification phase is completed, the parameters which define attitude may be computed.

SCADS System Sub-components

The two basic parts of the SCADS system are the scanning sensor head with the associated signal processing electronics and a power supply. The components contained within the sensor head include a lens system, photomultiplier tube, motor with gearhead, reticle, and angle encoder. The volume required for the sensor head will be approximately 40 cubic inches, based on a housing 4-1/2 inches in diameter and 8-3/4 inches in length. The electronics-package dimension will be 2 inches by 6-1/2 inches by 6 inches. The complete SCADS system will weigh approximately 7 pounds.

APPENDIX I

WHEEL-GRAVITY-GRADIENT BOOM HYBRID EXPERIMENT

The technology for passive gravity-gradient attitude control systems is being fully developed by ATS-A, D, and E. Passive techniques produce an attitude control system with a response time related to orbital frequency, high static errors when subjected to disturbance torques, and no off-vertical pointing capability. The wheel-gravity-gradient boom hybrid is an active control system in that the attitude reference is not derived from the gravity field, thus it does not suffer the performance limitation of a passive system. The gravity-gradient boom is gimbaled with 2 degrees of freedom and serves two functions: (1) to provide a source of reaction torque for attitude maneuvers (thus reducing the required wheel torque) and (2) to provide a source of external torque to prevent wheel saturation and to minimize the momentum storage requirement. The wheel and control electronics provide control system damping; therefore, a low torque passive damper is not required.

A system study is being conducted by the Westinghouse Air Arm Division. This study will produce a recommended system design, a recommended gimbal design, and a digital computer simulation of the control system. The contractor was given the ATS-F&G performance specitication, and it is anticipated that the wheel-gravity-gradient boom system will satisfy the ATS-F&G operational control requirements.

Since the wheel-gravity-gradient boom hybrid attitude control system is a new technology (not proven in flight), it will be carried on ATS-F&G as an experimental system and will be deployed only after the major antenna experiments have been carried out. The concept is shown in Figure I-1.

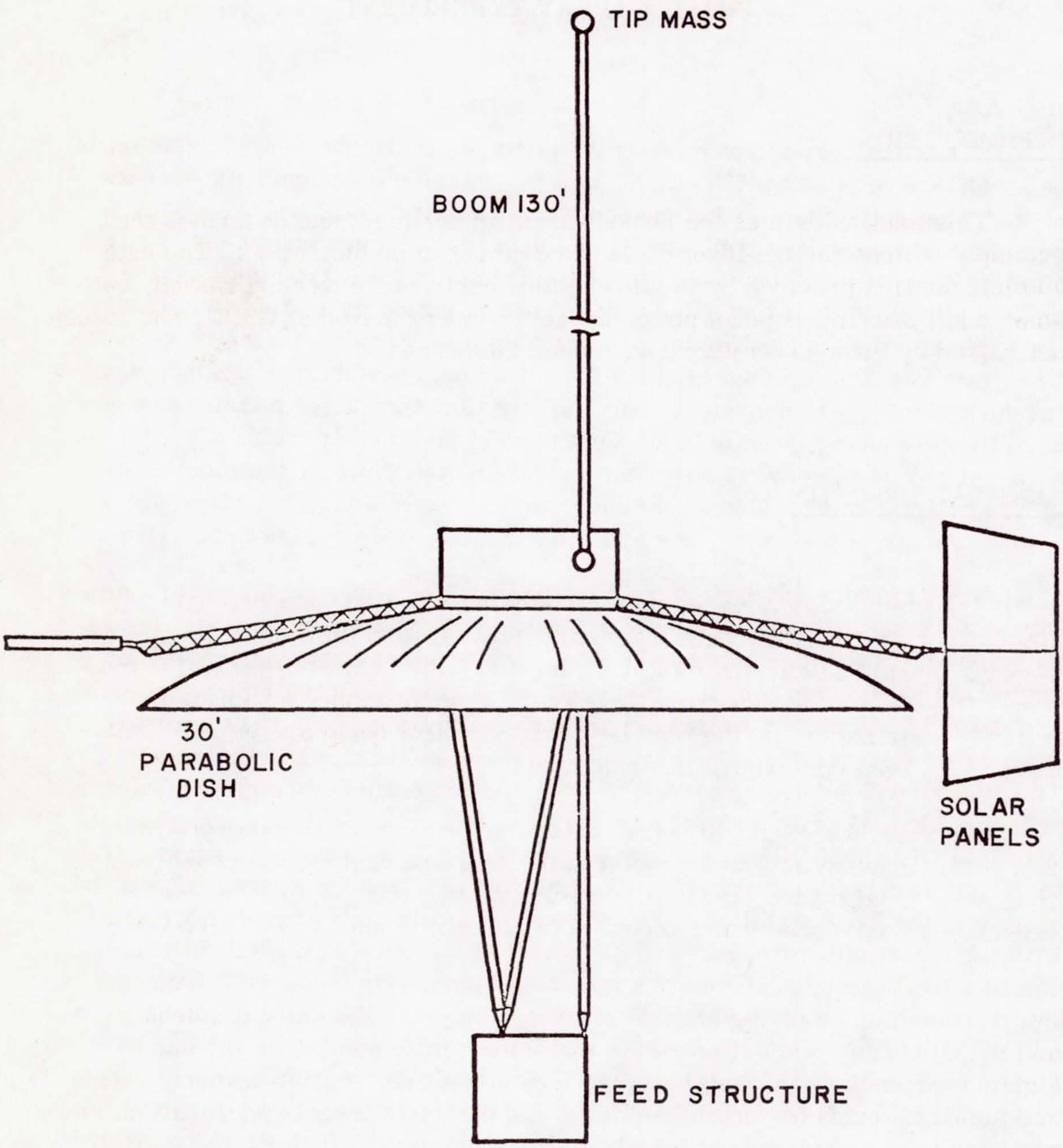

Figure I-1. Wheel-gravity-gradient boom attitude control system.

APPENDIX J

PHASED-ARRAY EXPERIMENT

INTRODUCTION

This section defines the phased-array experiment and describes the principal options for the form of the experiment implementation. The data and information provided by the three study contractors were reviewed, but some additional independent power budgets were prepared at GSFC, and issues not raised by these contractors have been introduced.

GENERAL DESCRIPTION

The phased array has characteristics making it particularly well-suited for use in a satellite communication relay. The primary objective of the experiment will be to demonstrate the use of inertialess scanning over wide angles (up to ± 30 degrees) in a communication relay between other spacecraft and ground stations. A subsidiary objective will be to begin the transition from ground proven phased-array techniques to space qualified hardware. The system will be designed to provide a minimum of two complete transmit/receive channels or four beams in space. It could be used as a repeater between APOLLO and other spacecraft in earth orbit and ground stations. This technique will increase the effective time during which the spacecraft will be visible at the ground station. The phased array also can provide a weight trade-off alternative to precision stabilization systems in the satellite and can provide the angle information that would otherwise be derived from the interferometer. Many scientific spacecraft use omnidirectional antennas, and signal budget calculations show that under these conditions the use of higher frequencies (X-band) becomes less attractive. Signal-to-noise ratios and signal margins for omnidirectional and directive target spacecraft antennas have been calculated for a high quality voice link at both S- and X-band frequencies. The results slightly favor the use of S-band where omnidirectional antennas are used on the target spacecraft, while with directive antennas on the target spacecraft the differences in S/N ratio are sufficientlv small that the choice should probably be based on other criteria. (See Table J-1.)

Table J-1
Power Budgets

Omnidirectional Antenna on Target Spacecraft		
Characteristic	S-Band +2.25 MHz	X-Band +8.0 MHz
Target S/C transmitter power (2 watts)	+33.0 dbm	+33.0 dbm
S/C cable loss	-1.5 db	-2.0 db
Target S/C antenna gain	-4.0 db	-5.0 db
ERP	+27.5 dbm	+26.0 dbm
Space loss (R_{max} = 27k mi)	-192.2 db	-203.5 db
ATS-F&G antenna gain (4-ft aperture)	+27.0 db	+38.0 db
Received power	-137.7 dbm	-139.4 dbm
Noise power density (1 Hz)	-174.0 dbm	-174.0 dbm
Receiver noise figure	+4.0 db	+6.0 db
Receiver noise power (1 kHz)	-140.0 dbm	-138.0 dbm
Telemetry S/N ratio	+2.3 db	-1.4 db
Directive Antenna on Target Spacecraft		
	+2.25 MHz	+8.0 MHz
Target S/C transmitter power (2 watts)	+33.0 dbm	+33.0 dbm
S/C cable loss	-2.0 db	-3.0 db
S/C antenna gain	+20.0 db	+30.0 db
ERP	+51.0 dbm	+60.0 dbm
Space loss (R_{max} = 27k mi)	-192.2 db	-203.5 db
ATS-F&G antenna gain	+27.0 db	+38.0 db
Received power	-114.2 dbm	-105.5 dbm
Noise power density (1 Hz)	-174.0 dbm	-174.0 dbm
Receiver noise figure	+4.0 db	+6.0 db
Receiver noise power (1 kHz)	-140.0 dbm	-138.0 dbm
Telemetry S/N ratio	+25.8 db	+32.5 db

Implementation of the phased array could take several forms. The choices of the contractors were:

1. Retrodirective array

2. Corporate feed array

3. Lens Approach.

Retrodirective Array

Of these, the retrodirective array is the most versatile for the present application and has the greatest growth potential. A pilot signal from the target spacecraft is received at the phased array which transforms the angle of arrival phase information to its conjugate value so that a high gain beam is directed to the target spacecraft. A ground station desiring to receive the target spacecraft transmission directs a pilot signal to the ATS-F&G spacecraft, which in turn directs a high gain beam, containing the information content of the target spacecraft signal, at the ground station. The system is, therefore, well-adapted to operation with synchronous, non-synchronous, stabilized, and unstabilized vehicles and does not require monopulse tracking, beam switching, preprogramming, or conventional controls. This system does require a phase-matched channel, a receiver, and a transmitter for each antenna source element; but a component failure does not result in catastrophic array failure. Undesirable features of the retrodirective system are the high prime power consumption and a need for flight qualified component improvement.

Corporate Feed Array

The corporate feed array represents a conventional approach to the solution of the communications problem. This method utilizes power dividers, diplexers, and circulators in combination with variable phasors to direct the beams. Undesirable grating lobes may be generated when the beams are scanned. Active acquisition and autotrack capabilities are required to exercise continuous control over the phasor elements.

Lens Approach

The lens approach is very similar to the corporate feed array. Access to individual antenna elements is obtained through a combination of space and corporate feeding. Disadvantages of the lens approach include: the physical size of the lens, its susceptibility to mechanical shock, and a limited growth potential due to the problem of the size of the antenna feed elements.

APPENDIX K

REFLECTOR ANTENNA BEAM SCANNING

The S-band scanning system consists of an array of n feed horns which can generate independent transmit and receive beams at any of the n-1 beam positions. The phase centers of the pairs of horns in the feed array are equally spaced along a straight line in the focal plane of the parabolic reflector. The effective phase center which determines the main beam position is located midway between any two of the adjacent feed horns. The receive and transmit beams can be pointed in desired directions by transmitting a command to the spacecraft to select the correct effective phase center. The phase center will be electrically positioned in one axis and mechanically in the other to point the transmit and receive beams.

A continuous one-way communications link can be obtained between two stations (ground or space) by using this scanning technique. A continuous two-way communications link can be obtained by using a more complex ferrite switching network. The maximum beam scan limits are imposed by theoretical and practical considerations. Figure K-1 illustrates the maximum angle ψ between the receive and transmit beams and hence the maximum separation between any two stations.

For two stations whose included angle is equal to or less than $\psi/2$, the following procedure is used to provide a continuous one-way communications link:

1. The control station transmits a command to the satellite to point the spacecraft antenna boresight at the station desiring to transmit data. This can be accomplished by computing the required satellite position and transmitting the required control data to the satellite or using the on-board monopulse system.

2. When the spacecraft boresight is pointing at the desired station, the effective phase center for the transmit beam is selected and the phase center is rotated (by mechanically moving the feed assembly or rotating the spacecraft) about the spacecraft boresight until the transmit beam is pointed at the receiving station.

For two stations at or near the limit of scan (included angle is greater than $\psi/2$) , the following procedure is used to provide a continuous one-way communications link:

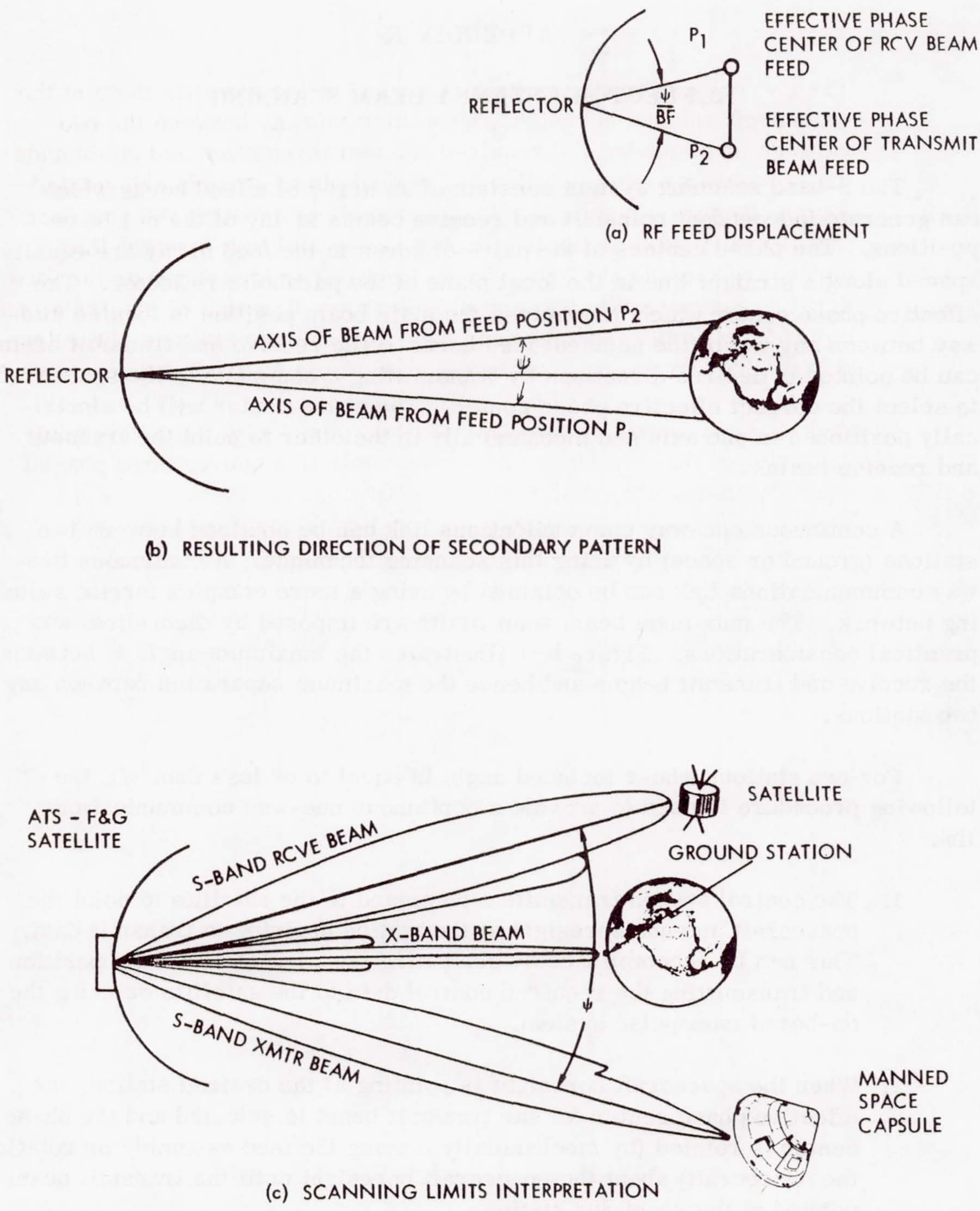

Figure K-1. Geometry of scanning of the reflectors.

The control station transmits a command to the satellite to point the spacecraft antenna boresight at a station midway between the two stations or computes the required spacecraft position and commands the spacecraft to this position. The feed phase centers are rotated (by mechanically moving the feed assembly or rotating the spacecraft) until the direction of electrical scan is parallel to a straight line connecting the two stations. The spacecraft is translated or the feed array is mechanically translated so that the direction of electrical scan coincides with the straight line connecting the two stations. The transmit and receive beams are then pointed at the desired stations.

The versatility inherent in this type of beam positioning combines many of the characteristics that normally are present only in a conventional phased array or a non-scanning high-gain parabolic reflector, not both.

APPENDIX L

LOCAST EXPERIMENT

OBJECTIVE

The objective of the Location of and Communication with Aircraft by Satellite Transponder (LOCAST) experiment is to define the requirements and parameters of an operational air traffic control system capable of worldwide applications. This system would provide two-way voice and digital data communications between all cooperating aircraft and their associated ground control facilities by means of a satellite borne transponder at synchronous altitude. Real-time surveillance over all aircraft would be provided through continuous position tracking and automatic reporting from an on-board telemetry sensor.

GENERAL DESCRIPTION

The LOCAST communications system would consist of duplex radio links from the ground control stations to all participating aircraft and return. Each ground control facility would be equipped with a high-gain antenna for transmitting to and receiving from the satellite. Up to ten independent ground control facilities could operate simultaneously in the communication mode. Any of the ground control facilities could function as a master control if provided with the timing, tracking, and computing equipments necessary for position location. The satellite would transmit to and receive from the aircraft in the 1.5 GHz (L-band) frequency range. All satellite equipment requirements are within the present capabilities of the Applications Technology Satellite (ATS) program. Participating aircraft equipped with a high-gain L-band antenna could receive any of the ground control voice and data transmissions. Any of the aircraft could transmit voice and data to all ground facilities, and more than 200 aircraft could be accommodated on an operational basis with a single satellite.

POSITION LOCATION

The position location scheme can be divided into two areas: (1) ranging by satellite alone, and (2) satellite-augmented ranging from VLF ground transmitters. The two-way range for the ground station-to-satellite-to-aircraft and return is measured at the same time as the two-way range from the ground station-to-satellite and return by means of standard side-tone ranging techniques. From these two measurements, and the known aircraft altitude, the range from the sub-satellite point to the aircraft can be determined. A circular line of position (LOP) is thereby defined which contains the aircraft position and which is centered at the sub-satellite point.

A second LOP is defined by the range of the aircraft from a VLF ground transmitter. This range measurement is also made at the ground station on signals relayed from the aircraft by means of the same satellite transponder. If the Omega navigation network becomes operational, the Omega VLF transmitters will be used, but in the event that they are not, any suitable VLF transmission standard emitting a single tone in the 10 to 14 kHz frequency range can be used. Single frequency transmitters will produce a set of LOPs so that the resulting ambiguity must be resolved by other means. The proposed primary method of doing this in the LOCAST system is by continuous tracking from a known starting position.

A second and completely independent method of position location was also incorporated into the LOCAST design for the purpose of increasing the overall reliability of aircraft surveillance and to provide a secondary method of ambiguity resolution. This subsystem, designated SINC (Satellite Inertial Navigation Control), makes use of the measured range and range-rate between the aircraft and the satellite along with the telemetry aircraft velocity vector derived from an on-board inertial platform. These measurements alone will define the position of the aircraft to an accuracy sufficient for ambiguity resolution and will also permit ground observation of on-board navigation equipment performance.

The ATS-F&G LOCAST experiment will require a specially designed transponder in the satellite, a single ground control facility of minimum size, and a maximum of four cooperating aircraft operating simultaneously. The control facility would be instrumented to demonstrate the feasibility of a complete air traffic control system on a global scale with present technology. In particular, the minimum control facility constructed for purposes of the LOCAST experiment, could be expanded to a system capable of two-way voice and digital communications, and simultaneous location and tracking of more than 200 aircraft utilizing the same satellite. In addition, the proposed experiment will be instrumented to provide a basis for determining the accuracy of other previously proposed position-location schemes (i.e., multiple satellite ranging and doppler systems).