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ELECTROMECHANICALLY STEERED ANTENNA SYSTEMS (EMSAS) FOR USERS OF TDRSS

PERFORMANCE INTERFACE DOCUMENT

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PERFORMANCE INTERFACE DOCUMENT
FOR
USERS OF TRACKING AND DATA RELAY SATELLITE SYSTEM (TDRSS) ELECTROMECHANICALLY STEERED ANTENNA SYSTEMS (EMSAS)

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This document has been prepared to complement information found in the Tracking and Data Relay Satellite System (TDRSS) User's Guide (STDN 101.2)

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All measurement values are expressed in the International System of Units (SI) in accordance with NASA Policy Directive 2220.4, paragraph 4.

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SUMMARY

The Tracking and Data Relay Satellite (TDRS) System (TDRSS), a major National Aeronautics and Space Administration program, will become operational with the launch of the first TDRS in 1983. This program caused major changes in spacecraft operations. User satellite antenna system performance is very crucial for achieving reliable TDRSS link performance at the desired data rate. This Technical Memorandum contains guidelines that will assist the TDRSS user in selecting and procuring the correct high-gain antenna system.

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ILLUSTRATIONS

ACRONYMS

ADR	Achievable data rate
AM/PM	Amplitude modulation/phase modulation
AR	Axial ratio
ATS	Applications Technology Satellite
BER	Bit error rate
DPSK	Differential phase shift keying
DSN	Deep Space Network
EIRP	Effective isotropic-radiated power
EMSA	Electromechanically steerable antenna
EMSAS	Electromechanically steerable antenna system
ESSA	Electronic Switching Spherical Array
FOV	Field of View
GSFC	Goddard Space Flight Center
GSTDN	Ground Spaceflight Tracking and Data Network
HGA	High-gain antenna
HPBW	Half-power beamwidth
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
KSA	Ku-band single access
LHCP	Left-hand circular polarization
MSFN	Manned Spaceflight Network
NASA	National Aeronautics and Space Administration
Nascom	NASA Communications Network
NCC	Network Control Center
NGT	NASA Ground Terminal
PA	Power amplifier
PFD	Power flux density
PN	Pseudonoise
RF	Radio frequency
RFI	Radio-frequency interference
RFPIP	RF performance improvement package
RHCP	Right-hand circular polarization

SMA	S-band multiple access
SMM	Solar maximum mission
SNR	Signal-to-noise ratio
SSA	S-band single access
STADAN	Space Tracking and Data Acquisition Network
STDN	Spaceflight Tracking and Data Network
STS	Space Transportation System
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TWT	Traveling wave tube
UQPSK	Unbalanced quadrature phase shift keying
Usat	User Satellite
VSWR	Voltage standing wave ratio
ZGTF	Zero Gravity Test Facility
ZOE	Zone of exclusion

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USERS OF TRACKING AND DATA RELAY SATELLITE SYSTEM (TDRSS)
ELECTROMECHANICALLY STEERED ANTENNA SYSTEMS (EMSAS)

INTRODUCTION

HISTORICAL BACKGROUND

The success of all space missions has been based on the ability to gather data in space and return that data to Earth-based investigators. The National Aeronautics and Space Administration (NASA) tracking and data acquisition stations, located in various parts of the Earth, underwent three distinct evolutions. These were as follows:

- Space Tracking and Data Acquisition Network (STADAN)–STADAN, completed in 1958, was used for tracking unmanned spacecraft in Earth orbits from ground facilities that used sensitive receivers and powerful transmitters.

- Manned Space Flight Network (MSFN)–During the Mercury, Gemini, and Apollo programs of the early 1960's, MSFN was used to provide two-way contact between the ground, the sea, and space for the astronauts.

- Deep Space Network (DSN)–Implemented in the early 1960's, DSN used parabolic dish antennas in three stations located approximately 120° of longitude apart, to support NASA lunar and planetary missions. DSN is still operational and continues to support planetary missions under Jet Propulsion Laboratory (JPL) management.

The Spaceflight Tracking and Data Network (STDN), operational from May 1971 to the era of the Tracking and Data Relay Satellite (TDRS) System (TDRSS) Network, is a combination of the STADAN and the MSFN.

The Ground Space Tracking and Data Network (GSTDN) is composed of 14 fixed and portable land-based stations that serve as direct support to NASA's Earth-orbiting scientific and applications spacecraft and manned spaceflight programs such as the Space Shuttle. Network operation and control and the associated central computing facility for operation and analysis are located at Goddard Space Flight Center (GSFC).

The STDN's most significant disadvantage is that the antennas are bound to the Earth's surface which creates a limited field of view (FOV). This limited FOV is caused by the antenna which usefully transmits and receives signals only when a spacecraft is in view. Therefore, the average coverage a spacecraft may expect to receive from the tracking stations is limited to approximately 15 percent of its orbit.

The TDRSS Network will become part of the new STDN on operational acceptance and will substitute the space segment and one ground station for many of the ground stations within the STDN. Some of the ground stations such as Bermuda and Merritt Island remain in operation for launch support.

TDRSS NETWORK

It was recognized that the GSTDN's limitations could be removed through a new network that used geostationary satellites rather than ground stations for tracking and communicating with user spacecraft. This network could provide coverage for almost the entire orbital period of a user spacecraft, support a number of spacecraft simultaneously, and have a high assurance of availability. Several NASA studies showed that this network was feasible by using state-of-the-art technology developed in the middle to late 1970's and by using the ATS-6 and the Nimbus-6 spacecraft as demonstration models. This research led to the development of the TDRSS Network.

The impact on the user spacecraft is the requirement for increased effective isotropic radiated power (EIRP), higher-gain antennas, higher power transmitters, and more sensitive receivers.

TDRSS NETWORK CAPABILITIES

The TDRSS Network consists of the TDRSS and a series of associated ground-based organizations. The Network Control Center (NCC), located at GSFC, manages the entire TDRSS Network. The NCC is linked with the NASA Ground Terminal (NGT) and collocates and interfaces with the TDRSS White Sands Ground Terminal (WSGT). Communications between the NGT, the NCC, and the users are through the NASA Communications Network (NASCOM), headquartered at GSFC. Other elements of the TDRSS Network include certain Earth stations that are needed during the launch and transfer orbit phases of user spacecraft. GSFC is responsible for developing, operating, and managing the TDRSS Network.

The TDRSS Network provides adequate performance margins, operational flexibility, and high reliability for supporting projected Space Shuttle payloads and free-flying spacecraft during the 1980's. Signal processing is not performed on board the TDRS. The satellite acts as a bent-pipe repeater that relays signals or data between the user spacecraft and the ground terminal. The space segment is simply designed and highly reliable. The TDRSS Network has three primary capabilities:

- Tracking–The TDRSS network determines the precise location of orbiting user spacecraft by measuring range (distance) and range rate (velocity) with respect to the TDRS's.

- Telemetry and Data--Each user spacecraft transmits telemetry signals that indicate certain operational parameters (e.g., power level and temperature). The spacecraft also transmits data signals that correspond to the scientific or applications information collected by the spacecraft instruments. The telemetry and data signals are relayed by the TDRS's from the user spacecraft to the WSGT for ultimate use by GSFC and the user community.

- Command–The WSGT, which sends command signals through the TDRS's to user spacecraft, orders the spacecraft to perform certain functions such as aim a camera and fire a thruster. The commands are originated by GSFC or Johnson Space Center (JSC) for unmanned or manned spacecraft, respectively.

To relay large amounts of scientific data from the Space Shuttle or investigatory spacecraft the Network provides:

- Simultaneous service to mul ·iple spacecraft and support for the scientific user community
- Accurate orbit determination for placement and position of user spacecraft, including determination of the range (distance of the user spacecraft from the relay satellite and ultimately from the ground terminal) and the range rate (velocity at which the user spacecraft is moving).
- Single-access service or multiple-access service, offering options to user spacecraft for radio frequencies (RF's), bandwidths, and rates of data transmission between the spacecraft and the TDRS's.
- Forward-link service provides an uplink from the WSGT to the TDRS and a downlink to the user spacecraft for commands. These links pass through either the multiple-access S-band RF or the single-access S- and Ku-band RF's.
- Return-link service provides an uplink from the user spacecraft to the TDRS and a downlink to the WSGT for telemetry and data. Up to 20 links are available on the multiple-access RF.
- Pseudorandom noise (PN) coding applies to all data streams for transmission and some reception. PN coding, which is primarily used for meeting Earth flux density regulations, provides some security from interception and protection against additive noise disturbances. For example, telemetry data generated by the user spacecraft are transmitted in a data stream with PN coding to the TDRS at S-band RF. The TDRS transmits these data to the WSGT through the Ku-band return link. When the data reach the ground station, an algorithm (i.e., a mathematical procedure) is applied to decode the data. The data are then sent to the user by Nascom.

TDRS OPERATIONS

The TDRS's operate from geostationary orbits. Using the satellites for tracking and data transfer allows for a greater FOV. To achieve cost savings by using only one ground station and two relay satellites, it is necessary to locate the satellites at certain fixed positions relative to the ground station. This configuration, however, has an undesired result: the Earth blocks either relay satellite from viewing the user spacecraft for a portion of the spacecraft's orbit. This blockage, which specifically applies to user spacecraft with orbital heights of less than 1200 km, may extend to 15 percent of the orbit. The blockage occurs over the Indian Ocean and is termed the "Zone of Exclusion" (ZOE). A spacecraft that orbits above 1200 km receives 100-percent coverage from the TDRSS Network.

The TDRSS Network provides bent-pipe communications links from user spacecraft to relay satellite to ground station and vice versa. This state-of-the-art telecommunications service is based on a successful experiment that involved the Applications Technology Satellite 6 (ATS-6) and the Nimbus-6 spacecraft during the mid 1970's. The ATS-6's Tracking and Data Relay Experiment demonstrated that the Earth-orbiting Nimbus-6 could be commanded from a ground station by the

ATS-6 Earth-synchronous satellite. Data generated on board the Nimbus-6 were sent back to the ATS-6 and were relayed by the ATS-6 to the ground. In addition, the range and range-rate measurements of the Nimbus-6 through the ATS-6 determined the relative distance and velocity of the two spacecraft. These measurements were then compared with similar direct measurements made by the ground station.

PERFORMANCE CONSIDERATIONS

IMPACTS OF ANTENNA SYSTEM PERFORMANCE ON ACHIEVABLE DATA RATE

To achieve reliable TDRSS link performance at the desired data rate depends on how well a User Satellite (Usat) antenna system performs. Users employing TDRSS S-band or Ku-band return-link service at medium-to-high data rates usually will require a high-gain antenna (HGA). A steerable system is necessary because of the narrow beamwidths associated with high RF gain. The primary antenna system used for meeting this need is the electromechanically steerable (gimballed) antenna (EMSA).

Antenna system performance is principally directed towards TDRSS return-link service performance because the resulting RF gain values required for return-link medium-to-high data rate service are far greater than the gain needed for supporting the required command data rates in forward-link service. Figure 1 shows an example of user antenna gain requirements versus link-data rates for the various TDRSS forward-link services.

TRADEOFFS BETWEEN ANTENNA GAIN AND TRANSMITTER POWER

Usat antenna system performance for the TDRSS return-link services is a key factor for maintaining and pointing the necessary EIRP toward a TDRS for achieving the desired data rate at an acceptable bit error rate (BER) under dynamic tracking conditions. The EIRP is the sum (in dB) of the antenna RF gain, G, and the transmitter power, P, available at the antenna input port (transmitter power less RF transmission line and rotary joint losses). After the required EIRP has been estimated by the methods described in the section on "Relating Required Antenna Gain and Transmitter Power to Achievable Data Rate," there is an important tradeoff between transmitter power and antenna gain. This tradeoff process causes numerous effects such as:

- Impact on prime power requirements

- Weight and moment impacts

- Impact on gimbal steering angles

- Tradeoff between locating the transmitter power amplifier (PA) and its associated components on the movable portion of the antenna system to minimize RF line and rotary joint losses—at the expense of increased moment—or locating the PA at some fixed location away from the antenna assembly, necessitating a cable wrap for bypassing the gimbals

- Antenna minor lobe performance and transmitter power impact on power flux density limits

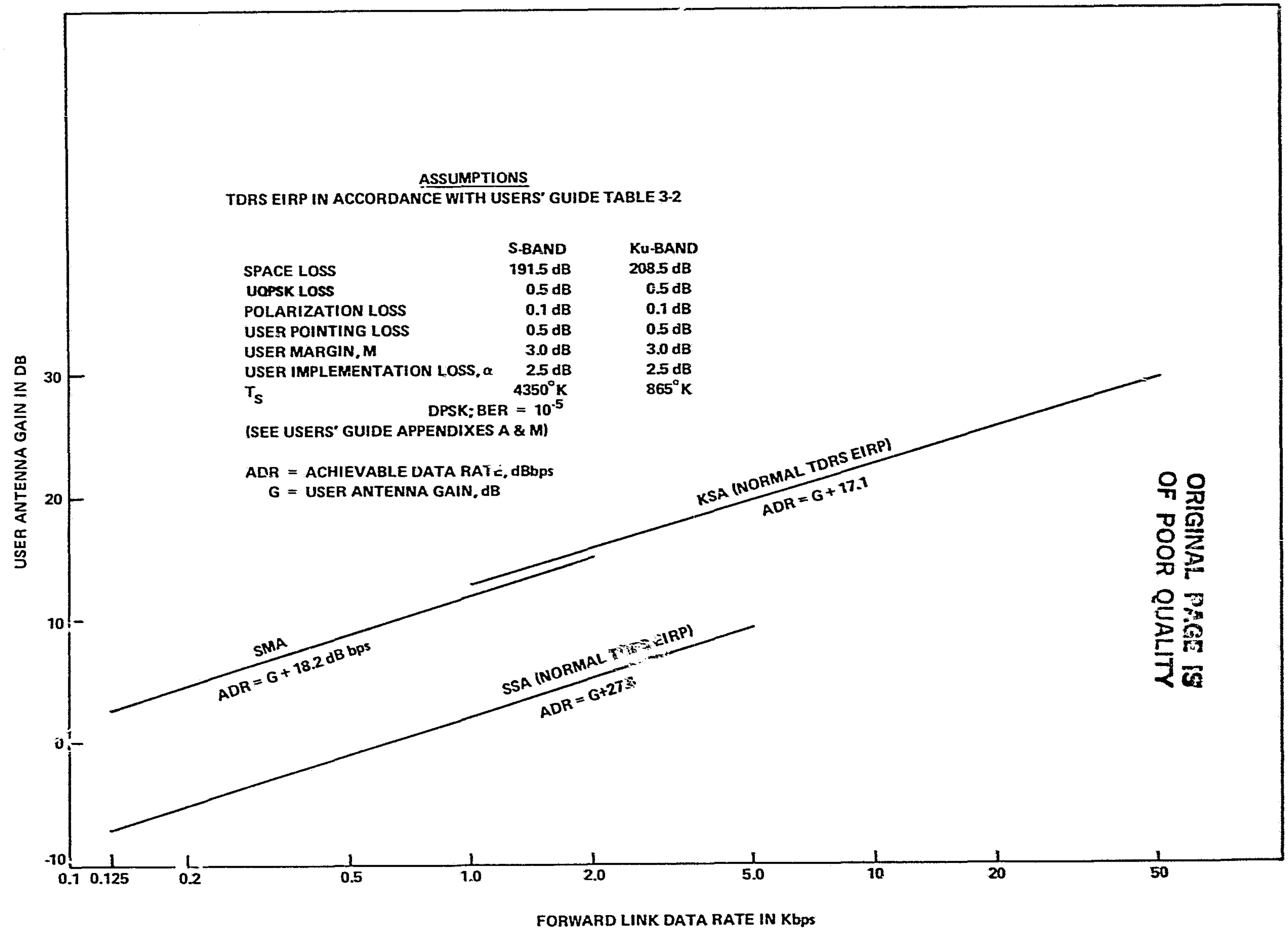


Figure 1. Example TDRSS Forward-link Service User Antenna Gain Versus Data Rate

- State-of-the-art limitations on achievable transmitter power output with high reliability

The highest available nominal power output of the NASA GSTDN/TDRSS transponder, which operates at S-band, is 5 W (7 dBW). At present, solid-state S-band PA's which may be interposed between the transponder and the antenna, are in the final stages of advanced development at GSFC. These amplifiers have nominal 10 W (10 dBW), 20 W (13 dBW), and 30 W (14.8 dBW) outputs. The PA's can be packaged in a variety of configurations that include a low-noise receiver preamplifier and a diplexer. These configurations, referred to as RF Performance Improvement Packages (RFPIP's), are small, light-weight units of high efficiency.

The maximum RF power output currently achievable at Ku-Band is 20 W (13 dBW) using travelling wave tube (TWT) amplifiers of proven high reliability, and 5 to 8 W (7 to 9 dBW) using solid-state power amplifiers which may prove to be highly reliable. Solid-state power amplifiers with 20 W output are projected for the late 1980's. Solid-state power amplifiers are easier to implement than TWT amplifiers at the antenna input port location because of their lighter weight, smaller volume, and less sophisticated power supply requirements. A frequency upconverter is required between the S-band output of the NASA GSTDN/TDRSS transponder and the Ku-band PA for Ku-band operation.

RELATING REQUIRED ANTENNA GAIN AND TRANSMITTER POWER TO ACHIEVABLE DATA RATE

User operations with TDRSS are detailed in the "Tracking and Data Relay Satellite System (TDRSS) Users' Guide," STDN 101.2, hereafter referred to as the "Users' Guide." The Users' Guide contains the methods used for determining the user EIRP required to achieve the desired data rate for the return-link services. The Users' Guide also includes an example of a return-link calculation that provides a convenient checklist of all the loss items that usually need to be considered.

Figure 2 shows the range of parabolic-dish and array antenna sizes that are necessary to accommodate TDRSS S-band and Ku-band return-link services. In addition, the figure uses the state-of-the-art transmitter power ratings described in the section on "Tradeoffs Between Antenna Gain and Transmitter Power," and the antenna-size gain relationship discussed in the section on, "Antenna System RF Performance."

The antenna diameters range from 0.7 to over 2.0 m with S-band gain values in the 20- to 36-dB range and Ku-band gain values ranging from 24 to 49 dB. The curves of data rate versus EIRP are derived from Table 3-6 of the Users' Guide which has been appropriately corrected for the particular user unique constraint parameters used as an example in this figure. These curves can be altered so that a user can match his unique constraints. The figure also indicates the points on the curves corresponding to the existing flight-qualified tape recorder playback rate of 2.56 Mbps, the 20 Mbps playback rate of recorders currently being developed, and the 150 Mbps of recorders playback rate that are planned for development.

In addition, Figure 2 shows the remarkable reduction in S-band dish size that can be achieved by using the advanced technology currently under development (i.e., high power PA's that can be located at the antenna input port). For example, at 3 Mbps the dish diameter is reduced from 2.7 m using a conventional NASA GSTDN/TDRSS 5-W transponder connected to the antenna with a 10-ft RF transmission line, to 0.75 m using a 20-W PA located directly at the antenna input port. The antenna weight/moment/cost advantage is obvious.

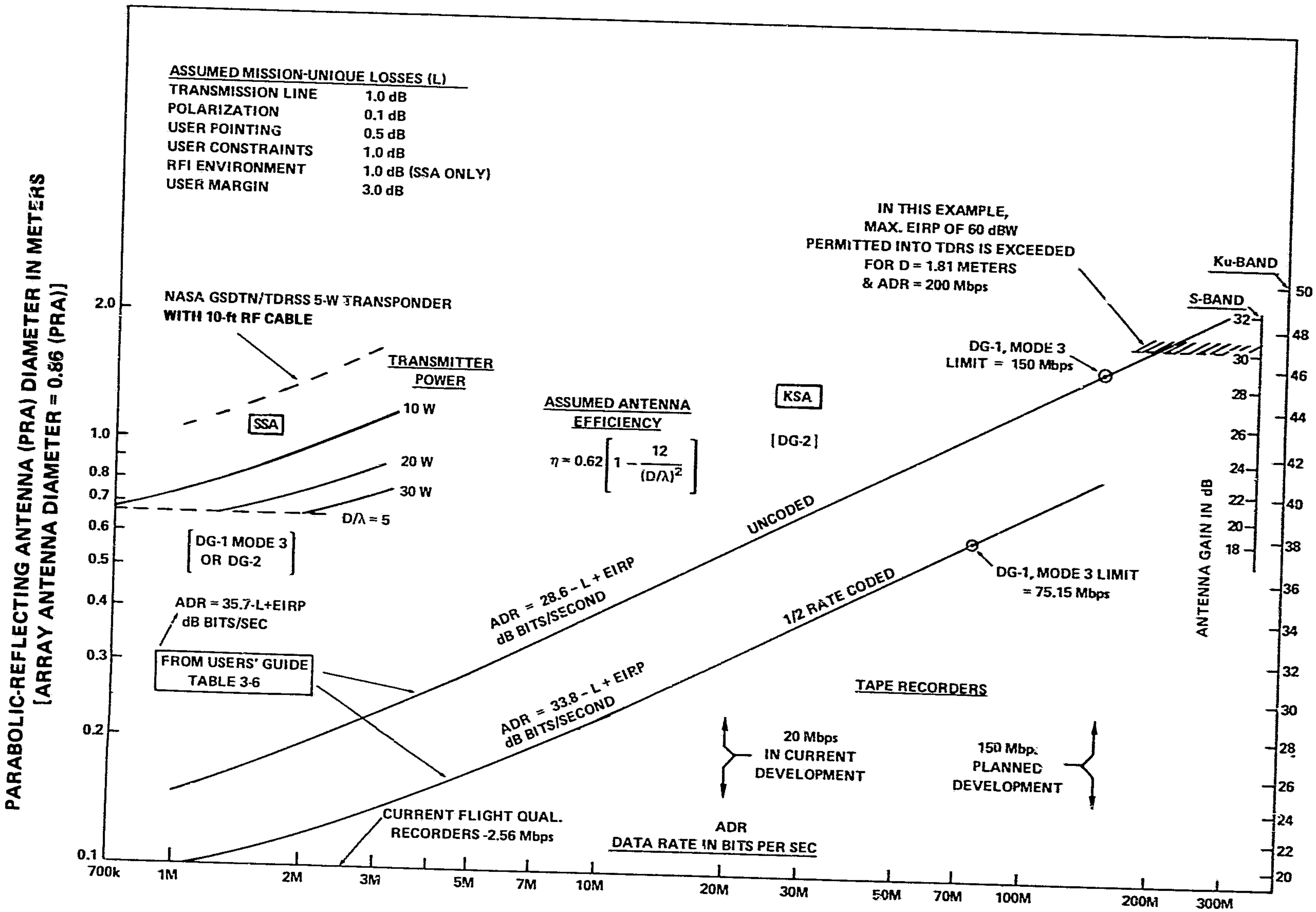


Figure 2. Example TDRSS Return Services Data Rate Versus Antenna Gain and Antenna Diameter

ANTENNA SYSTEM RF PERFORMANCE

Antenna Size and Gain Consideration

Antenna gain, G, for a parabolic-reflector (dish) antenna is given as

$$G = 10 \log \left[\frac{\eta \pi^2 D^2}{\lambda^2} \right]$$

referenced to an isotropic radiator (a hypothetical radiator with unity gain (0 dB) that radiates equal powers over 4π steradians) where D is the diameter in the same units as the operating wavelength, λ, and η is the efficiency that accounts for the feed system illumination taper and spillover of energy past the dish and feed components.

Although the user is not limited to using a dish antenna for a mechanically steered system, this antenna has proved very practical for the range of gain required. Space-qualified hardware has been supplied for existing missions, but, a planar array could be used as an alternative. Planar arrays, which typically have an efficiency of 10 to 20 percent greater than a dish antenna for the same physical aperture size, have the potential for a size, weight, and shortened moment from gimbal axes advantage. This advantage, however, may be offset by a small reduction in steering angle caused by physical shape.

The minimum dish size (thus the lowest available gain) is limited at S-band by geometric optics considerations. The maximum dish size is determined by the 60 dBW EIRP limitation into TDRS and the transmit power used at Ku-band.

Considering the minimum dish size, the D/λ ratio can be reduced to about 6 by using a design that ensures reasonable efficiency. As the D/λ ratio is further reduced, the efficiency rapidly decreases with a corresponding rapid rise in minor-lobe levels that could possibly impact power flux density limitations. See the section on "Minor Lobe Performance." (η versus D/λ is approximated by the empirical formula shown in Figure 2.)

Referring to the return link example of Figure 2, a 0.88-m diameter would be required (G = 21.1 dB, $D/\lambda = 5.75$, and $\eta = 0.395$) for the maximum S-band single access (SSA) 3.15 Mbps, DG-2 return-link data rate with 20-W power. With 30-W power, a 0.77-m diameter would be required (G = 22.9 dB, D/λ - 6.61, and $\eta = 0.45$). At Ku-band in this example the larger of these two antennas with 20-W power would just be able to handle the maximum available 150-bps rate for 1/2 rate coded Ku-band single access (KSA) DG-2 service ($\eta = 0.62$). In reference to the example parameters chosen, the maximum uncoded KSA DG-2 300-Mbps rate could not be achieved without exceeding the 60-dBW EIRP allowed toward TDRS. The maximum rate for 60-dBW EIRP would be 200 Mbps and would require a 1.81-m dish (G = 47.0 dB, $D/\lambda = 90.6$, and $\eta = 0.62$) unless the user was willing to reduce his margin M from 3 to 1.2 dB to achieve the 300 Mbps maximum. Because this antenna is fairly large, the user may choose a planar array with a 10 to 20 percent higher efficiency. Assuming a 15 percent higher efficiency, the required physical aperture area would be 2.24 m^2 with a 1- by 2.25-m flat array. Many large antennas of this type have been built and qualified (e.g., the synthetic aperture radar antenna flown on the second Shuttle mission) with the radiator elements, RF transmission lines, and matching networks etched into the supporting substrate.

Antenna Beamwidth Considerations

The antenna half-power (3 dB) beamwidth (HPBW) is approximately related to the gain, G (numeric), by

$$\mathrm{HPBW} = \left[\frac{41253\eta}{G}\right]^{1/2}$$

Probably TDRSS should be autotracked for a user antenna having a HPBW 1° or less because of user and TDRS ephemerides and timing accuracy limitations. Even with a larger HPBW (smaller diameter), the user may want to autotrack because the magnitude of ground software support required for command pointing may be unpalitable or signal level changes induced by nonprecise pointing and tracking may cause errors in the TDRSS autotrack system. A 1° HPBW, based on a 62-percent efficiency, is characteristic of antennas having a gain of 44 dB and a diameter of 1.3 m at Ku-band (15 GHz).

HPBW and beam-nose shape are included in the estimation of user pointing loss. A parabolic approximation to the true main-beam shape is sufficiently accurate to estimate the pointing loss from the mechanical pointing error budget. Once the HPBW is estimated or measured, the pointing loss is related to the mechanical pointing error by

$$\text{Pointing loss in dB} = 3.00\left[\frac{2\,(\text{pointing error in degrees})}{{}^\circ\,\mathrm{HPBW}}\right]^2$$

Polarization Considerations

Circular polarization is always used. The polarization sense may be either left-hand circular polarization (LHCP) or right-hand circular polarization (RHCP) depending on the TDRSS application.

The maximum polarization axial ratio (AR) must be specified (usually over the HPBW angle) because polarization loss enters into the link performance calculations. Figure 3 shows the polarization coupling loss as a function of user AR for both SSA and KSA services. Sometimes, the polarization sense may be deliberately chosen to minimize interference to other users. Figure 3 indicates the minimum achievable isolation between two antennas that have oppositely sensed polarizations. Both sets of curves assume a maximum AR of 1.5 and 1.0 dB at S-band and TDRSS Ku-band for the TDRS antennas, respectively. Surprisingly, little same sense coupling loss occurs with a relatively poor user AR. When isolation from the user signals is required by using the opposite sense, however, the achievable isolation is a very sensitive function of the user AR.

Minor-Lobe Performance

The power flux density at the Earth's surface from space segment emitters in the frequency bands shared by terrestrial systems is limited by international agreement. Both the TDRSS SSA and KSA frequency bands fall into this category. Figure 4 shows the EIRP limitations for both bands as a function of satellite orbital height, elevation, and off-nadir angles.

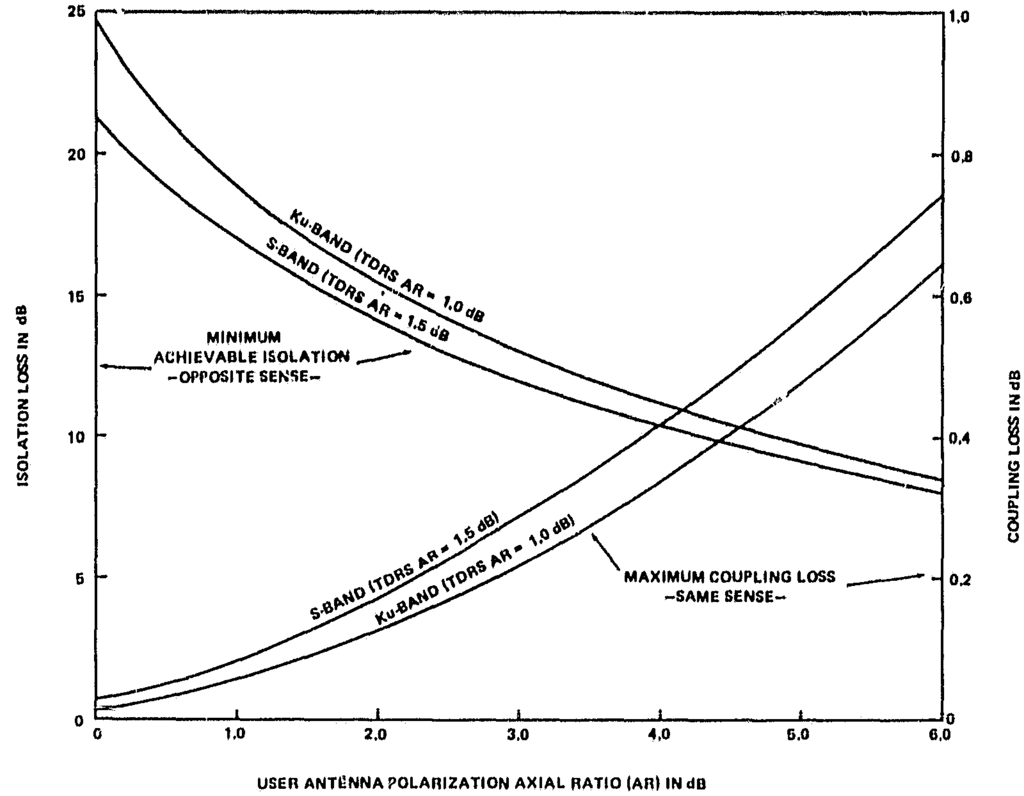


Figure 3. User/TDRS Elliptical Polarization Isolation and Coupling Losses

The user's main beam is never pointed toward the Earth's surface during routine service. Antenna sidelobe and back-lobe performance and transmitted power are the determining factors. The curves of Figure 4 are based on a uniformly spread signal by the PN code. EIRP in this figure is transmitter power plus side-lobe gain (total of all polarizations) referred to an isotropic antenna in dB. Because density limitations are expressed in terms of the maximum occurring in any 4-kHz band, uncoded high data-rate user signal emission spectra must be examined in detail to ensure compliance.

Figure 5 shows how minor-lobe maxima are generally distributed. Two types of feed arrangements, front feed and Cassegrain (double reflecting), are commonly used. Dual-frequency antennas (i.e., antennas serving S-band and Ku-band simultaneously) usually apply one feed type at S-band and the other feed type at Ku-band with special treatment of the subreflector to reduce blockage for the front-feed frequency. These antennas can operate at lower efficiencies because of increased blockage by the feed components and supports, and can produce higher than normal spurious side lobes and back lobes by reradiation from these structures. The side lobes near the main beam show a rapid drop in level as the angle from boresight increases. Feed spillover to the rear part of

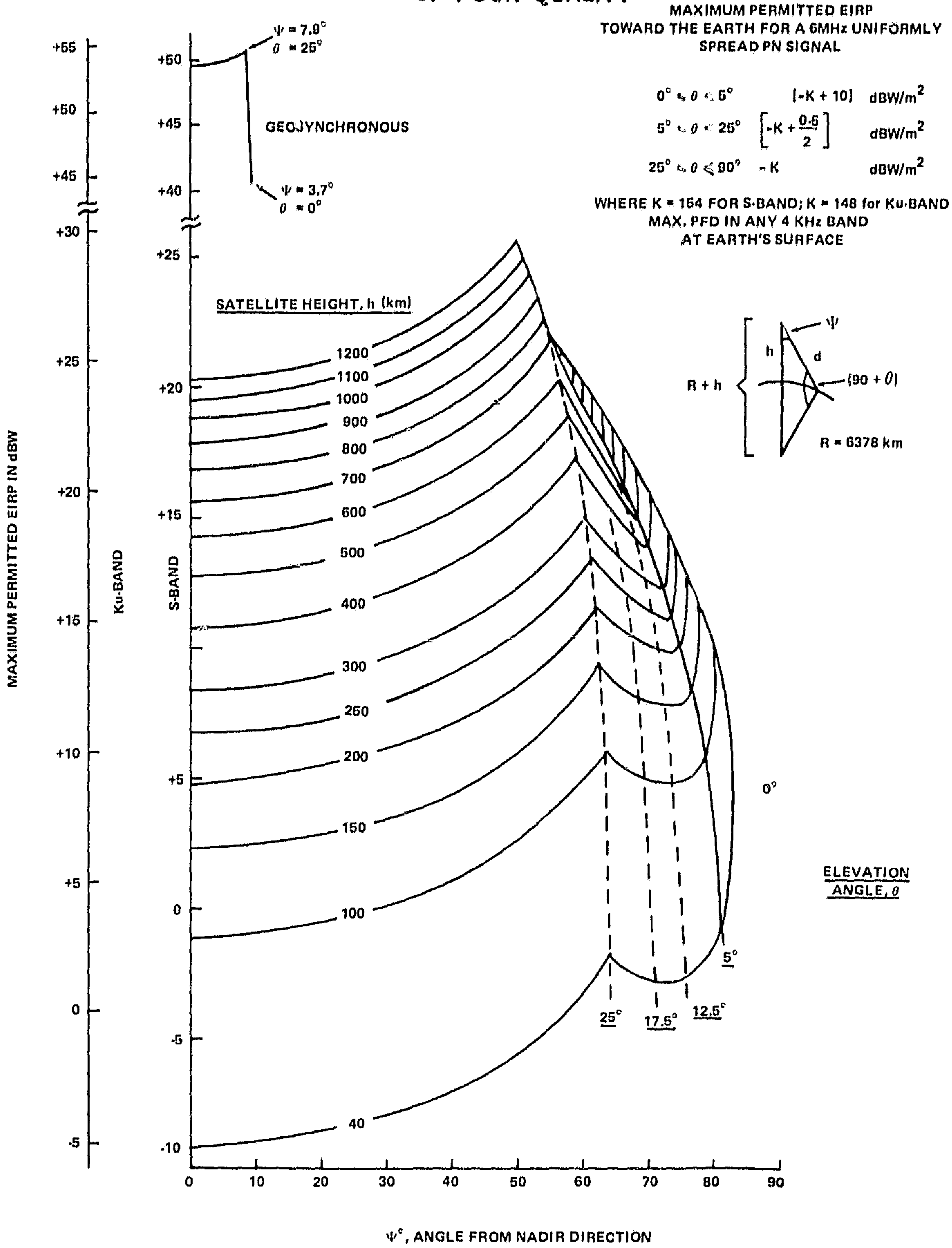


Figure 4. Maximum Permitted EIRP Toward the Earth for a 6-MHz Uniformly Spread PN Signal

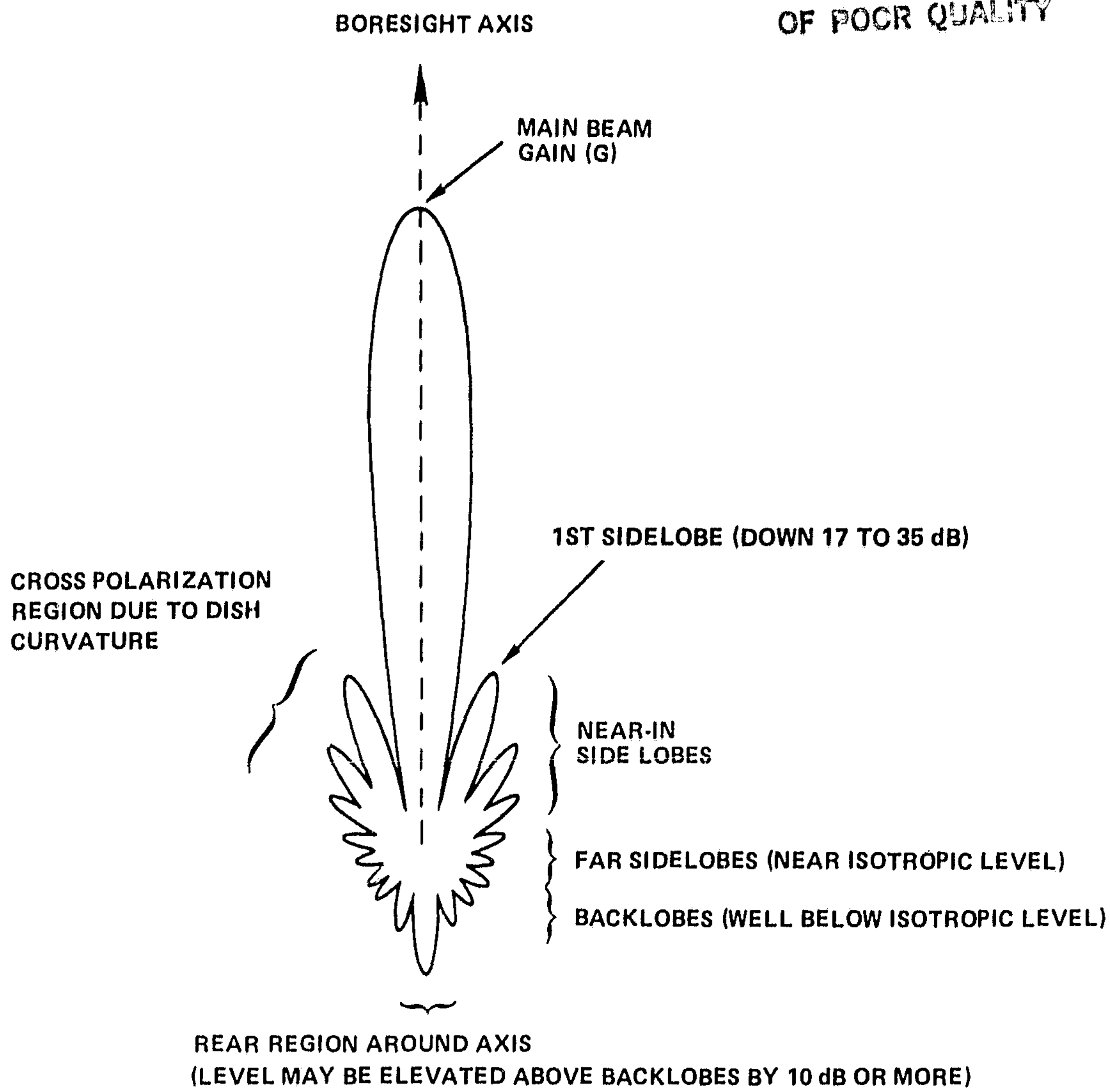


Figure 5. Generalized Antenna Lobe Structure for Antennas of Large D/λ

the main dish tends to be greater when using a front feed instead of a Cassegrain. Primary feed spill-over past the subreflector in a forward direction is characteristic of the Cassegrain arrangment. Both feed systems, however, have very broad cross-polarization lobes that occur about 60° from boresight because of main reflector curvature. Control of these elements is necessary in the design to ensure that off-boresight radiation toward the Earth is maintained at a value that satisfies the power flux density limitations.

In addition, Figure 5 shows how the side lobes and back lobes are distributed. The first side lobes adjacent to the main beam are generally of the greatest amplitude and lie between 17 and 35 dB below the main beam depending on the shape of the illumination taper. Because the first few side lobes of the higher-gain antenna are quite close to the main beam, the Earth would not be illuminated during routine TDRSS service. At larger angles from the main beam but still in a relatively forward direction, the side lobes are usually at the isotropic level (0-dB gain). Lower levels can be expected in the rear region where the feed components are "out of sight." Directly to the rear, the level may rise appreciably because of the ring of illumination around the dish periphery. The level expected in this region is a function of illumination taper and is generally lower with a Cassegrain feed than with a front feed. In addition, the oppositely sensed polarization in this region is often the stronger component.

Usually arrays have nearly uniform amplitude side lobes in the 25 to 35 dB range decaying as the cosine of the angle off-beam maximum.

For small parabolic reflector antennas with a D/λ less than 10, side-lobe and back-lobe control may be difficult. Therefore, the user should carefully watch the high PA power/low antenna gain combination, especially at S-band. Furthermore, the close-in side lobes may illuminate the Earth during routine service.

Note that the power flux density restrictions at the Earth's surface do not respect polarization, so minor lobe levels should be specified in terms of all polarization components because these exhibit strong cross-polarization components.

Input Impedance and VSWR Consideration

The RF impedance of the antenna input port should be matched carefully to the power source. Usually, the connection between the power amplifier and the antenna is by a coaxial cable at S-band and by either a coaxial cable or waveguide at Ku-band. RF rotary joints may be used. Impedance mismatch is indicated by a voltage standing wave ratio (VSWR) greater than unity.

A VSWR that is not greater than 1.5 should be satisfactory. The VSWR must be specified over the operating bandwidth, and care taken to ensure that the VSWR curve over the operating bandwidth versus frequency is smoothly varying function so that with the rest of the RF system the overall gain slopes are small enough to meet the Users' Guide constraint limits for amplitude modulation/ phase modulation (AM/PM) conversion, differential gain, and allied signal characteristics.

GIMBAL DRIVES

To ensure reliable, long lifetime gimbal drives, the following two approaches have been successfully used for space antenna systems:

- Direct drive (no gearing), brushless dc torque motor
- Harmonic drive gear reduction, stepper motor

Bearings and Lubricants

Ball bearings are the choice for low friction and long lifetime. Bearings are sized to survive the launch vibration environment with a maximum Hertzian stress of 350,000 psi. Usually, thin race (torque tube) geometry is preferred to maximize the number of load-sharing balls and to minimize weight. Material is 440C stainless steel.

Bearings and harmonic drives are lubricated with either of two proven low-vapor pressure oils—a synthetic fluorocarbon oil (i.e., Krytox) or a highly refined mineral oil (i.e., Apeizon C KG-80). The lubricant is vacuum impregnated into sythane ball separators.

To provide adequate angular repeatability bearing fits and preload provisions must be carefully considered and constraints and loads caused by thermal changes and gradients minimized. The general rule is to use the lightest preload and to minimize constraints to the extent allowable.

Shaft Angle Sensors

Gimbals angles must be accurately measured for feedback and telemetry for both open and closed loop tracking systems. Angle sensors should be noncontacting for long, trouble free lifetimes. Either optical or inductive approaches are suitable.

Torque Margin

Although normal gimbal friction torques are very low, a substantial torque margin should be designed for the drive motors because reserve torque is often the difference between success and failure when unexpected loads are encountered (i.e., uncaging, thermal control problems, or bearing deterioration). Although an optimum margin cannot be stated, a capability of three times the worst predicted load at end of lifetime is the minimum.

Gimbal Pointing Performance

The following maximum gimbal pointing errors can be expected:

(1) Response to a commanded angle (3 Σ error, degrees)

	Absolute	Repeatability
Boom	±0.2	±0.1
Gimbal	±0.1	±0.1
Total	±0.2	±0.14

(2) Closed loop tracking

- Accuracy determined by error signal characteristics
- Follow-up within ±0.01° is typical

DEPLOYABLE MASTS/BOOMS

Deployable masts and booms for antenna systems are considered as nearly mission-unique items. Basic masts and booms have been space qualified; however, size, hinge points, attachment points, and caging are expected to vary according to spacecraft structure and antenna system location.

IMPLEMENTATION

GENERAL

EMSA's consist of a variety of configurations that include a fan-beam antenna and one gimbal, a reflector, and an array-type pencil beam antenna with two or more gimbals (usually two). The most common gimbal combinations are crossed axes (X-Y arrangement) or axes intersecting in a "tee" arrangement (elevation over azimuth). The gimbals are attached to a deployable beam or mast that is accurately deployed in relation to the spacecraft during postlaunch operations. Associated electronic packages that drive the gimbals and interface with the spacecraft are in the spacecraft body. To keep antenna sizes to a minimum, RF packages may be located at or near the antenna. See the section on "Performance Considerations" for the RF performance and gimballed drives. The electronics subsystem's three principal functions are:

- To develop the drive power so that the gimbals point to the antenna
- To process gimbal position/desired pointing direction errors
- To interface the command and telemetry signals to the spacecraft

There are three principal approaches to steering an antenna:

- The antenna is command pointed by data generated on board, on the ground, or a combination of both
- The antenna is program-tracked by an onboard antenna processor that calculates where the antenna should point at regular intervals
- The antenna autotracks on an RF signal received from the data relay satellite that produces an error signal continually processed to point the antenna at the data relay satellite

Combinations of these three basic methods can also be implemented.

A number of flight-worthy antenna systems and subsystems have been developed. Appendixes A through H contain data on these systems. Only minor design modifications are believed necessary for meeting the requirements of most future missions. Antenna and RF component sizes and location, however, can be expected to vary when configuring future spacecraft that may require thermal, structural, control and dynamic modelling, program modes, and redundancy philosophy. In addition, location and deployment/attachment would cause minor changes to the basic existing designs.

Requirements that would cause dramatically new designs are not foreseen. Two possible economical designs that may evolve into flight-capable hardware are a low-torque noise-limited scan angle antenna system and a cost-effective autotrack antenna system.

LOCATION/COORDINATES/ALIGNMENT/SWEPT VOLUME

Selecting a location for the antenna system is a major activity during the early phases of a spacecraft project. Usually, the antenna must be deployed away from the spacecraft body, because the antenna's size could be 0.5 to 9 ft across with a typical swept volume of 0.1 to 300 ft^3 for conventional gimbaled antennas. This type of location would ensure that physical interference with parts or appendages of the spacecraft would not occur. Furthermore, the antenna's beam should never be directed toward parts of the spacecraft (degradation of RF performance would occur or possible degradation of the part).

Because swept volume is usually greater than a hemisphere, the location should be on the side of the spacecraft that views either or both of the data relay satellites. Data relay satellite access time for these desired locations should be established in early planning phases to ensure adequate contact time and to ensure that onboard communications and data-handling subsystems are not seriously impacted and are of a practical design.

Once the location is defined, both the antenna coordinates relative to the spacecraft and the accuracies of alignment of these sets of coordinates must be defined. Stiffness and accuracy requirements of the deployment boom or mast must be determined so that alignment errors can be found and controlled and torque and momentum perturbations induced back into the spacecraft are limited and definable. The usual method for determining these alignment errors and perturbations is by controlling manufacturing tolerances, by using optical cubes (i.e., antenna and spacecraft), and by testing the total antenna systems in a zero-g test facility that is capable of measuring both amplitude and frequency components of the perturbations generated, and modal frequencies of the support structure. At present, this facility does not exist, however the concept will be demonstrated in fiscal year 1983.)

OPERATION

To meet launch environmental and safety requirements the usual antenna configuration that evolves from the mission gain, contact time, and location requirements must be caged to the spacecraft body during launch operations. Once the spacecraft is at or near final orbit and attitude becomes stabilized, the antenna is uncaged, deployed from the spacecraft, operationally validated, and routinely operated to meet mission data acquisition requirements. The caging points and design are expected to be unique, but the holding and release methods will have to be proved.

LIFE

The antenna system is designed to reliably operate, assuming that a 50-percent duty cycle ("on/off") of continuous operation during the lifetime of the space mission and its test and storage periods before launch. A design margin of two or more lifetimes is used to ensure that this critical system operates.

SINGLE-POINT FAILURES

Single-point failures are minimized in accordance with good spacecraft design practices. Redundant components of a EMSAS consist of electronics packages, motors, position sensors, and duplex bearings. Nonredundant components consist of the deployable boom or masts, the actual antenna, and a separate pair of gimbals. Should these nonredundant components become redundant, a completely separate EMSAS with electronic cross strapping is the best approach because of cost and complexity.

TEST REQUIREMENTS

Tests are performed to ensure the following:

- Adequacy of the functional performance requirements
- Capability to withstand environmental conditions
- Life expectancy
- Unit performance during pre- and post-integration and prelaunch activities

The project will determine the amount of testing for the antenna system and its components and the extent of modifying or designing additions to the antenna system. To define the extent of required testing, previous testing and test levels should be compared. It should be emphasized that if this system does not function the mission will probably fail.

Testing that can be performed after integration onto a spacecraft is limited to operability not capability. Some operability testing require special tools that allow for positioning the antenna as if it were in space (e.g., location, alignment, and steering). Ground-test equipment should be modified or designed for simulating the specific spacecraft interface level, acceptance testing and for monitoring and supporting the system spacecraft level testing.

HANDLING AND STORAGE

Every antenna system should have handling and storage tools and containers. The antennas, which will be large and fragile, require a container that can be safely handled by one or more persons. The complete antenna system, for this same reason, will not be installed onto the spacecraft except for special tests and prelaunch assembly. This storage container would have multiple purposes such as handling, storage, and shipping.

INTERFACES

MECHANICAL

All EMSAS's will require some means of attaching and positioning the gimbals and antenna to the spacecraft body while the spacecraft is in orbit and during space transportation system (STS) payload launch operations. Booms/masts, cages, latches, and other mechanisms and devices may be used. The EMSAS could include these items if a simple attachment location could be defined on the spacecraft body at an early period, the attachment/deployment subsystem could be coordinated between the designers and manufacturers of the spacecraft and the EMSAS, and the attachment/deployment subsystem could remain the spacecraft manufacturer's responsibility. Variations of existing designs would be adequate for meeting the uniqueness of the mission.

Because the release and positioning of the EMSAS's are critical to the mission, relevant items must be carefully designed and extensively tested. Alignment measurements of the operational position of the gimbals and antenna must be completed to remove fixed, accumulated assembly and integration errors if programmed track-type antenna control is used.

The alignment between the gimbals, gimbal interfaces, and spacecraft support structures simply requires locating the holes and alignment pins in the mating surfaces to ensure minimum tolerance buildup and misalignments with respect to the spacecraft.

The entire assembly should be attached to rigid and fixed structural members of the spacecraft to ensure that minimal pointing errors and unknown modes are set up by steering or slewing the antenna.

THERMAL

Thermal designs of existing antennas and gimbals that are now in orbit or planned to be are probably adequate for most other missions. Each mission should have these designs validated to ensure that the components remain within operational temperature limits and that distortion of antenna masts does not cause the antenna to point outside the allowable pointing errors.

The antenna system is thermally isolated from the spacecraft to the maximum extent possible. The complex thermal model of the antenna system is simplified to include only a few nodes for incorporating into the entire spacecraft thermal model.

ELECTRICAL

The electrical interfacing of the EMSAS connectors, power, grounding, command and data handling, and timing are considered of only usual concern to ensure that they are compatible with the total spacecraft. Modifications and additions of electronics and thermal controls to current designs are a minor effort as compared with the total effort of building and testing an entire EMSAS.

The EMSAS should have an antenna control processor to ensure that it can be completely tested as a spacecraft subsystem to minimize integration complexities.

Ground-support equipment should emulate spacecraft electrical (nonRF) interfaces and monitor spacecraft engineering-type data during testing and integration stages.

Two RF interfaces that should be considered are: the microwave signal interface with onboard microwave components by RF transmission lines that require compatible connectors and the effects of scattering or reflections from spacecraft structure and appendages.

TORQUE DISTURBANCE

The motion of the EMSAS antenna dish with the mast deployed will result in torque disturbances on the spacecraft that interface with jitter-sensitive scientific instruments. In addition, this motion may cause a degradation in pointing accuracies of controlled systems. For example, a simplified mathematical model indicates that a step input to the antenna position is seen by the control system as a disturbance with amplitude at least 27 percent greater than actual input. The problem is further compounded by antenna motion which can excite modal frequencies in the antenna support structure and in the spacecraft. Figure 6, which is constructed from the solar maximum mission (SMM) data, shows the ratio or "gain" in single axis spacecraft angular motion caused by antenna "jitter" (i.e., small amplitude angular motion) at various frequencies from 0.1 to 100 rad/sec. (For example, the antenna jitter of SMM is estimated from 0.8 to 1.0 arc-sec in amplitude.) Should the frequency of this jitter be very near or equal to either the frequency of the flexible appendage or the antenna support structure, a resonant condition would exist and the spacecraft would experience a jitter gain factor of 2.5 (flexible appendage) to 4.25 (antenna support). For some spacecraft such as Space Telescope this jitter would degrade pointing accuracy to unacceptable values.

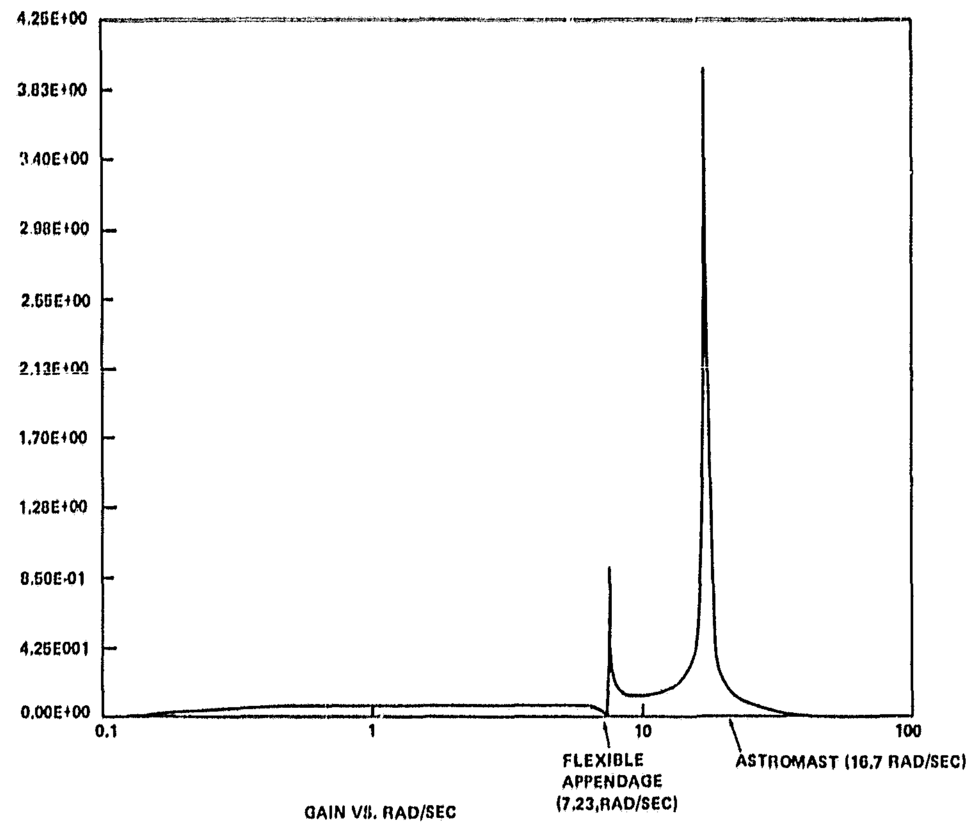


Figure 6. Gain in Single Axis Spacecraft Angular Motion Caused by Antenna Jitter

The problem of torque noise or jitter is impossible to eliminate, however, if the modal frequencies of the antenna support mast and the spacecraft can be determined, control compensation techniques can be used to diminish the response at these frequencies.

Spacecraft modal frequencies are difficult to predetermine except by an involved computer simulation. The antenna and antenna support structure can be physically tested to determine the modal frequency content and to determine the characteristics and effectiveness of control compensation. To perform these tests it is necessary to construct a test facility, the "Zero Gravity Test Facility" in which the force of gravity can be unloaded as much as possible. (Report of "Proof of Concept," will be completed in August 1983.)

APPENDIXES

The appendixes show that users requiring a high-gain antenna system do not have to support a significant development program. In recent years, functional flight components and systems have been developed that need only minor modifications to meet the uniqueness of a mission.

Letters were sent to the following representative industrial organizations (copy of letter, Appendix I) requesting data about representative flight components of electromechanically steered antenna systems. The list was compiled in the order of receipt of information when the information contained data of flight-proven designs:

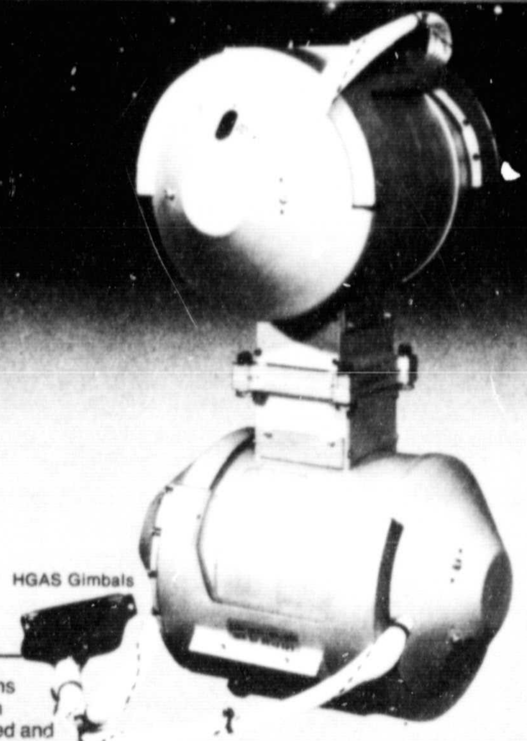
Appendix	Organization	Type of Response
A	Sperry Flight Systems	Letter, information
B	General Electric Valley Forge Space Center	Brochure
C	ASTRO Research Corporation	Letter, information
D	Ball Aerospace Systems Division	Letter, information
E	Harris Corporation GESD	Letter, information
F	Hughes Aircraft Company Space & Communications Group	Phone (no information available)
G	Fairchild Space and Electronics Company (FSEC)	Letter, development information
H	TRW Incorporated Space Technology Group	None
I	Letter of Solicitation	––

Each appendix does not include all the electromechnical flight hardware designs of the source company, but it does show that they have an interest in supplying future needs. A "no response" cannot be interpreted as no future interest in supplying flight components, however, it can be assumed that the source company was not interested in contributing to this document.

APPENDIX A

SPERRY FLIGHT SYSTEMS

High-Gain Antenna Systems



The Sperry Flight Systems High-Gain Antenna System (HGAS) has been developed and space qualified for spacecraft requiring a communications link with the Tracking and Data Relay Satellite System (TDRSS). It is capable of transmitting and receiving in both the S and Ku bands with a pointing accuracy of .64 degrees. The modular design allows for many different mission-unique configurations; the gimbal drive and electronics may be adapted to various payloads, including antennas, platforms, and solar panels.

- Solar Maximum Mission (High-Gain Antenna System)
- Space Telescope (gimbal drive and electronics)

- High accuracy
- Low friction

- Shuttle launch compatibility
- Minimal disturbance to spacecraft
- Redundant pointing electronics

- Excess of 10-year life test for low earth orbit pointing
- Proven thermal control for external operation

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Pointing Accuracy	• Pointing accuracy within 0.64 degree; fixed geometric errors and variable error sources combined. Accuracy within 0.30 degree could be achieved.
Perturbations to Spacecraft	• Reaction torque is less than 3.0×10^{-2} N-M peak spike, 1.3 seconds between pulses. • Reaction torque impulse is less than 3.2×10^{-3} N-M-S peak, approximate sawtooth of same polarity with 1.3 seconds between peaks. • Momentum storage is less than 2.5×10^{-2} N-M-S.
Thermal Control	• Primary system is passive through material and surface coating selections to ensure an operational HGAS at an ambient where adjacent components are at 180° Celsius. A backup system is implemented with heaters near the gimbal bearings in both drives and in the steering control electronics.
Power	• Nominal Operating (steering or slewing) 17.4 watts Nominal Heaters (both gimbal drives) 6.2 watts
Gimbal Rates	• 30 deg/min slew and 1.2 deg/min steer maximum with electronically limited continuous motion.
Antenna and RF Parameters	• Parabolic reflector utilizing the S-band single access link of the TDRS system Antenna Gain = 27 dB at Band 2 Frequencies: Band 1 = 2106.4 ± 20 MHz; Band 2 = 2287.5 ± 20 MHz System Gain = 25.0 dB with cable and RF coupler losses in Band 2 Power Handling = 10 watts cw minimum VSWR = 2.0 in band 1; 1.5 in Band 2
Reliability	• Probability of success of .98 for a 3-year mission obtained through the use of redundant pyrotechnics, electronics, drive motors, resolver and wiring. The single channel probability of success is .887.
Operational Duty	• A 3-year orbit life on a 50 percent duty cycle subsequent to a ground storage of up to 18 months. The life capability of the HGAS is estimated to be 10 years or more.
Weight	• Gimbals = 11 pounds each Astromast and supporting structure = 30 pounds Electronics = 11 pounds per channel Antenna = 25 pounds (nominal)

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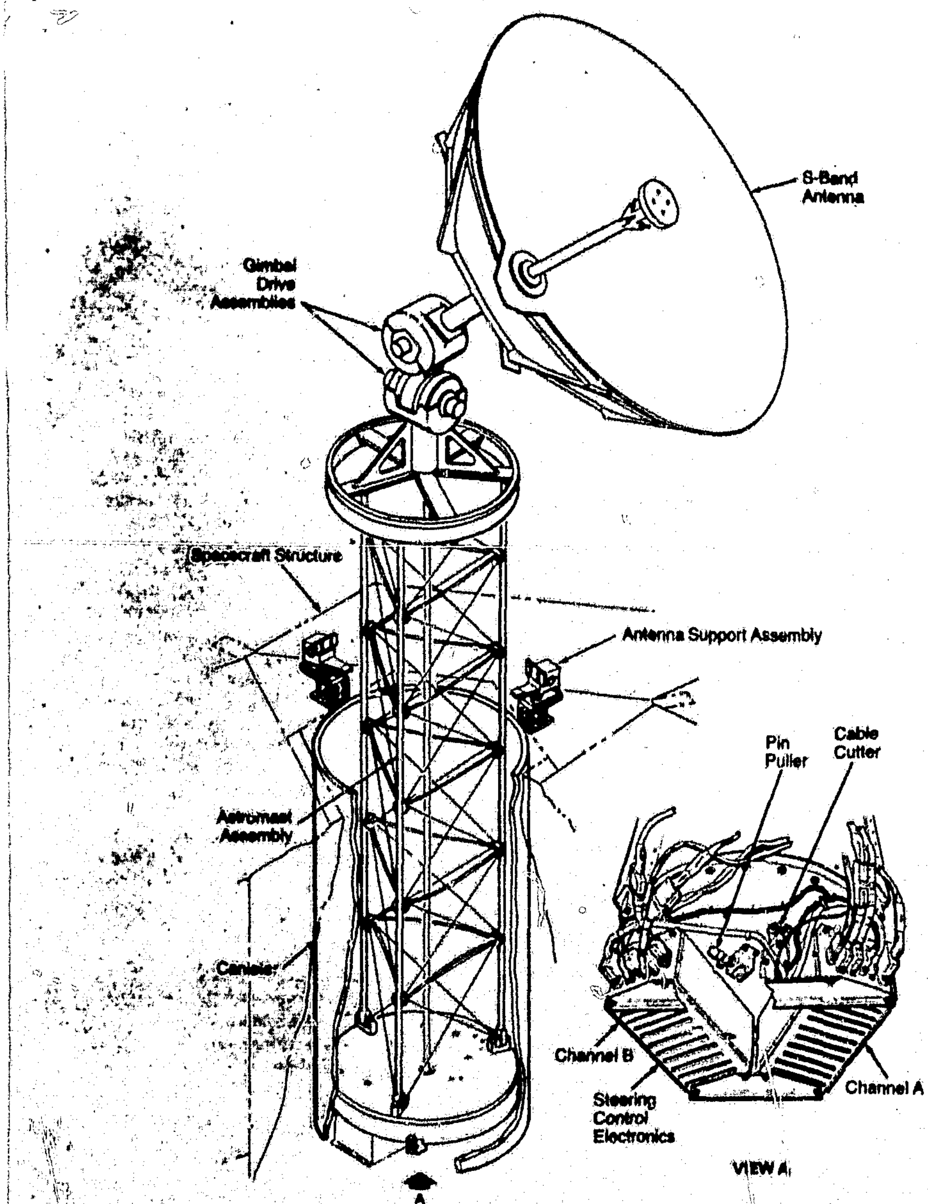

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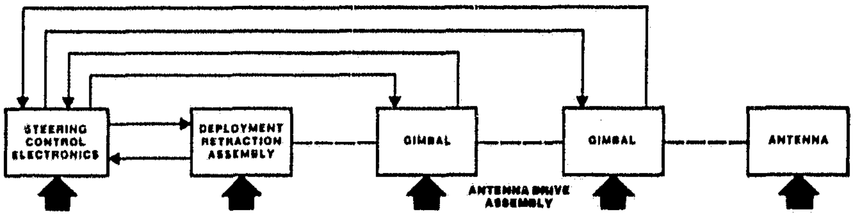
System Versatility
Modular Design Approach
Single or Dual System



- Analog Inputs
- Variable Accel and Velocity Limits
- Microprocessor Controller
- Hybrid Design for Lighter Weight, Increased

- Fixed or No Mast
- Different Lengths
- Various Stiffnesses
- Deploy Only

- Different Torque Output
- Limited or Full Rotation
- Different Rates
- Other Thermal Ambients

- Higher Power
- Various Payloads (Antennas, Platforms, Solar Panels)

The HGAS consists of four major components:

- A 27-dB-gain parabolic reflector that is used for spacecraft communications via the S-band Single Access (SSA) link of the TDRSS in the radio frequency bands of 2106.4 ± 20 MHz and 2287.5 ± 20 MHz.
- An antenna drive assembly capable of steering and pointing at a TDRS, or slewing to a TDRS prior to communication, through a solid cone of ± 110 degrees. All electromechanical hardware is dual redundant, and the direct drive is used to minimize disturbances to the spacecraft.
- A mission-unique deploy, retract, and jettison assembly, which stows the antenna and its drive for launch, deploys the system in orbit, and is capable of jettisoning all protruding elements in the event of a failure to stow for shuttle recovery.
- A dual-redundant set of steering control electronics which receives and processes commands and sends telemetry signals to and from the spacecraft's data bus. Controls for both gimbals, for deployment and retraction, and for an active thermal system are included.

The HGAS is stowed within the spacecraft at launch and later deploys to a fixed relationship with respect to the spacecraft's coordinate axes to assure proper execution of commands. The HGAS may also be retracted to the stowed position for recovery by the space shuttle.

The commands for steering the HGAS can be from an operations control center (for real-time control) or from an on-board computer (for steering through many successive orbits). The 16-bit position command contains the address and axis position and rate for each axis (Z-Y). When an error is detected (that is, when the antenna is not within the arc limits of the digital words), the error is processed in the steering control electronics. The antenna drive assembly then rotates and steers the antenna until it is within the digital-word arc limit. The antenna's rotation to correct any combination of errors causes perturbation of the spacecraft within the disturbance torque and momentum storage limits listed in the summary of characteristics.

A thermal control system is incorporated in the HGAS gimbals, control electronics, and actuation assembly; this system maintains required temperatures during all phases of flight and storage.

For more information on the High-Gain Antenna System (and the rest of the Space Systems product line), call (602) 869-2474 or write:

Sperry Flight Systems
Space Systems Marketing
P. O. Box 21111
Phoenix, AZ 85036

SPERRY FLIGHT SYSTEMS IS A DIVISION OF SPERRY CORPORATION

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May 1982
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APPENDIX B

GENERAL ELECTRIC
VALLEY FORGE SPACE CENTER

HIGH GAIN ANTENNA

DESCRIPTION: LIGHT WEIGHT ANTENNA DISH & FEED HORN FOR USE IN S-BAND COMMUNICATION BETWEEN IN-ORBIT SPACECRAFT AND TDRSS (TRACKING & DATA RELAY SATELLITE SYSTEM) SPACECRAFT

FEATURES:

- FLIGHT QUALIFIED
- STATE-OF-THE-ART PRODUCT
- LIGHT WEIGHT
- HIGH GAIN
- S-BAND
- DESIGNED FOR COMMUNICATION WITH TDRS SPACECRAFT
- TWO BROADCAST BANDS
- SIMPLICITY OF DESIGN
- VERSATILE DESIGN CAN ACCOMMODATE ANY FREQUENCY BAND
- DESIGN INCORPORATES GE EXPERTISE IN ANTENNA DESIGN
- INHERENT LOW COST DUE TO REPETITIVE PRODUCT
- GE CAN ALSO PROVIDE GIMBAL AND ELECTRONICS (OPTIONAL)

HIGH GAIN ANTENNA
TECHNICAL DATA

	SOLAR MAXIMUM MISSION	SPACE TELESCOPE & SUBSEQUENT
MATERIAL	KEVLAR	GRAPHITE FIBER EPOXY
WEIGHT		
SPEC	10 LB (4.5 KG)	7 LB (3.2 KG)
ACTUAL	10 LB (4.5 KG)	6.8 LB (3.0 KG)
DIAMETER	50.75 IN (128.9 CM)	50.75 IN (128.9 CM)
F/D	.25	.25
FREQUENCY (MHZ)	2086 TO 2126***	2245 TO 2265*
	2267 TO 2307**	2277.5 TO 2297.5**
NOMINAL GAIN (db)		
SPEC	27 PEAK	26.4 OVER 1° CONE
ACTUAL	27 PEAK	26.4 OVER 1° CONE: 26.6 PEAK
FEED HORN	CROSS DIPOLE FEED WITH REFLECTING CUP	
OUTPUT	CIRCULARLY POLARIZED LOW SIDELOBE PATTERNS	
HARDWARE	ANTENNA DISH & FEED HORN	
BEAMWIDTH (3 db)	~8°	~8°

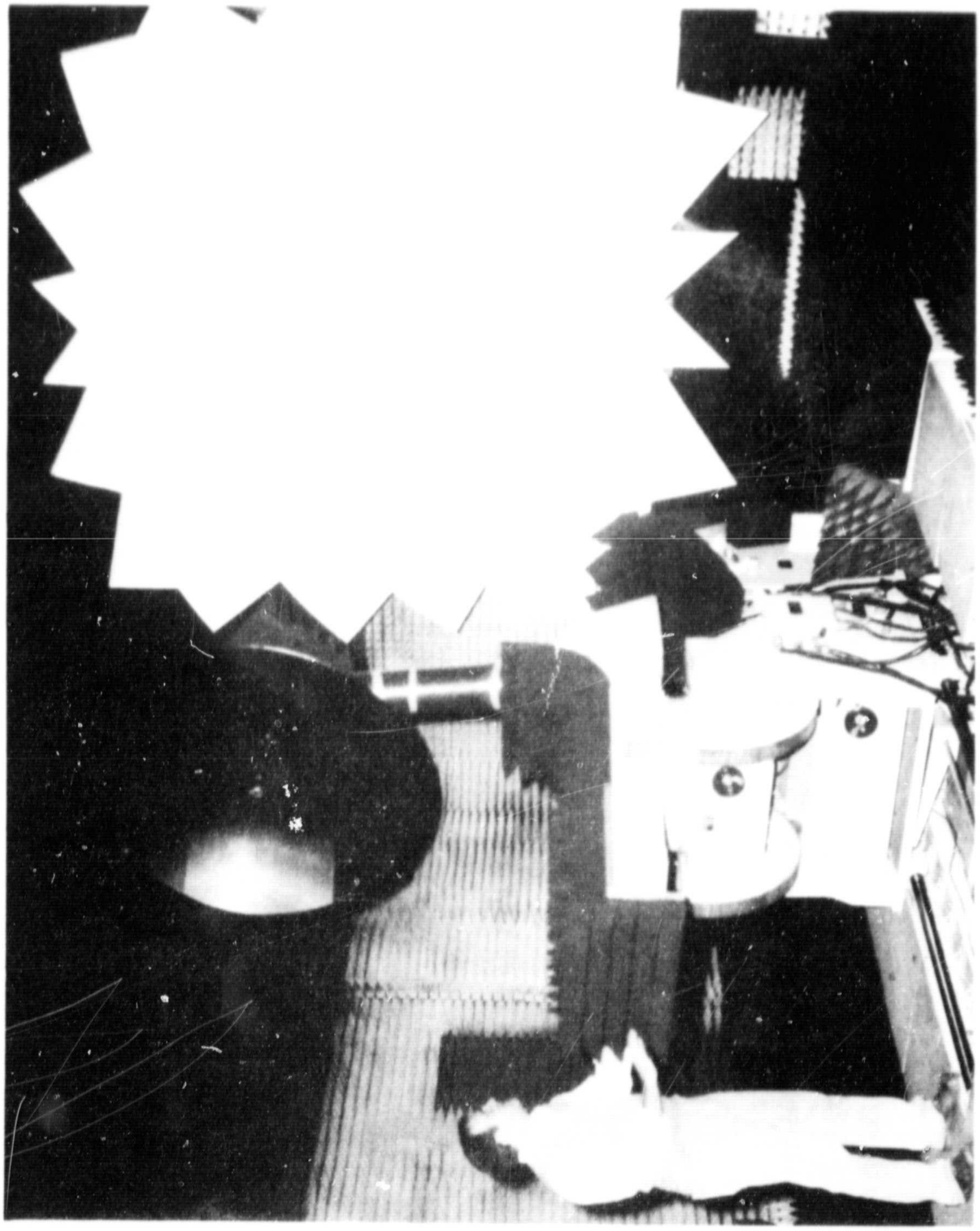
* TRANSMIT (TDRS SINGLE ACCESS CHANNEL)
** TRANSMIT (TDRS MULTIPLE ACCESS CHANNEL)
*** RECEIVE (TDRS MULTIPLE ACCESS CHANNEL)

HIGH GAIN ANTENNA

HARDWARE FLIGHT HISTORY:

- SOLAR MAXIMUM MISSION
  - IN ORBIT*
  - HARDWARE: ANTENNA DISH & FEED HORN
  - QUANTITY: ONE
- SPACE TELESCOPE
  - TO BE LAUNCHED AT FUTURE DATE
  - HARDWARE: ANTENNA DISH & FEED HORN
  - QUANTITY: TWO
- NOTE: GE HAS ALSO DESIGNED & FABRICATED ANTENNAS (DISH & FEED HORN, GIMBAL, & ELECTRONICS) FOR NUMEROUS OTHER PROGRAMS

*TDRS NOT YET IN ORBIT



APPENDIX C

ASTRO RESEARCH CORPORATION

DEPLOYMENT/RETRACTION MECHANISM FOR SOLAR MAXIMUM MISSION HIGH GAIN ANTENNA SYSTEM

By

Neal Bennett, Sperry Flight Systems
Peter Preiswerk, ASTRO Research Corporation

ABSTRACT

Accurate steering of a spacecraft communication antenna requires a stable platform. A mechanism called a Deployment/Retraction Assembly (DRA) which provides not only a stable, but a deployable platform for the High Gain Antenna System (HGAS) aboard the Solar Maximum Mission (SMM) spacecraft is described. The DRA also has the capability to retract the system upon command.

INTRODUCTION

The SMM spacecraft scheduled for launch into a 357 mile orbit in October 1979 will have aboard a high gain S-band antenna system capable of communicating with and tracking the TDRS system. This antenna system, called HGAS, must be stowed within a required envelope in the aft end of the spacecraft and withstand launch by a Delta launch vehicle. The spacecraft attitude does not allow the antenna to view the relay satellites when stowed. Consequently, once in orbit, the HGAS must be deployed to a position that allows the antenna to communicate with and track the relay satellites. The HGAS is shown in the deployed condition in Figure 1. When deployed, the deploy mechanism must maintain accurate support alignment for the antenna and articulation system while being exposed to the orbital space environment. Space Shuttle recovery of the SMM is planned and to facilitate this the HGAS is required to retract within its launch envelope so that the SMM spacecraft can fit within the Shuttle bay. If retraction is not possible, all portions of the HGAS outside the recovery envelope must be jettisoned from the spacecraft.

The DRA design described in this paper was selected based on the flight experience of the concept and its potential to satisfy the stringent HGAS requirements described above. Similar structures have flown successfully on the Air Force S-3 satellite and NASA's Voyager 1 and 2 as magnetometer booms of 20-foot length, 7-inch diameter and of 43-foot length, 9-inch diameter respectively. The deployable portion of the DRA, the Astromast, provides an ultralight, low profile structure with the deployed stiffness and stability required of HGAS.

DRA DESIGN DESCRIPTION

The DRA is a major subassembly mechanism of the High Gain Antenna System, weighing less than 23 pounds. The total deploy stroke is 60 inches, which positions the antenna at a point relative to the spacecraft that allows a view of the TDRS system. It has overall stiffness properties that yield major HGAS deployed resonant frequencies in excess of 8 Hz. Deployed alignment stability is expected to be better than .2 degrees over the required temperature range and deploy/retract cycle life. The required cycle life for ground operation will be approximately 30 cycles and for space, 1 cycle.

The DRA itself consists of five major subassemblies: (1) An Astromast Assembly which is the basic deploying and retracting support structure for the antenna and articulation system, (2) A servo assembly, which restrains the Astromast during deployment in a controlled manner, and provides the force required for retraction, (3) hardware that interfaces the HGAS with the SMM spacecraft, (4) a jettison mechanism that is capable of jettisoning certain portions of the HGAS, and (5) an antijettison caging mechanism that inhibits the jettison mechanism in the stowed condition.

ASTROMAST

The Astromast provides the key function in the DRA of structural support to the deployed antenna and articulation system. Its construction provides for maximum stiffness, minimum weight, and minimum volume.

Figure 2 is a layout of the Astromast showing the truss type construction. The basic members are the three main longitudinal members (longerons), triangular frames separating the longerons (batten frames), and pretensioned diagonal members connecting adjacent longerons and batten frames. The diameter through the longerons is 18.5 inches and the length between longeron pivot points is 62.16 inches. There are 5 batten frames, forming 6 bays, each 10.36 inches long. All members are fabricated from unidirectional S-glass/epoxy laminate to take advantage of the inherent high stiffness-to-weight ratio and thermal stability. The total weight of the Astromast is 3.7 pounds.

Stowed, the Astromast is coiled into a height of only 2.3 inches. The longerons develop, like coiled springs, a force tending to deploy the system and are restrained by a central lanyard. Figures 3 through 6 show the deployment sequence, as demonstrated by an engineering model, beginning with the fully stowed condition. Two different stages during the transition from stowed to deployed are shown in Figures 4 and 5. Since all or some portion of the longerons still form a helix, the Astromast is relatively weak during the transition phase. The maximum stiffness and strength properties are not achieved until full deployment, shown in Figure 6. Note the restraining lanyard located in the center. The top plate, representing the interface to the antenna and articulation system, rotates about the longitudinal axis a total of 382.5 degrees as the Astromast deploys.

In the deployed state, the longerons provide axial load capacity and bending stiffness, the battens stabilize the structure while in an elastically buckled condition, and the diagonals provide shear and torsional stiffness. Though the DRA will not be exposed to direct sunlight, the thermal alignment stability is designed to remain within .2 degrees with as much as 270 degrees R temperature differential between the two diagonals of each bay panel.

SERVO MECHANISM

The DRA servo mechanism controls the rate of deployment and provides the retracting force. A pully containing the restraining lanyard is attached to the output shaft of a simple worm gear assembly. The worm is casehardened steel and the driven helical gear is cast bronze; the mesh as well as the gear bearings are lubricated by Braycote 3L-38RP grease. The worm is driven by a brush type DC gear head motor, producing a total speed reduction of 2433:1. Motor brushes are redundant and of the longlife dry-lube type consisting of 85 percent silver, 12 percent molydisulfide, and 3 percent carbon. Figure 7 shows the location of the servo mechanism on the bottom of the DRA.

For launch the Astromast is not held in position by the lanyard, but by a pyrotechnic pin puller which absorbs the load directly. The pin puller is shown in Figures 7 and 9. Once the spacecraft is stabilized in orbit, the dual initiated pin puller is fired and the deploy sequence is started by commanding the DC motor on. The lanyard is in turn played out at a controlled rate allowing the DRA to deploy. Redundant microswitches are used to indicate the deployed state and are used to switch off the motor. A special bridle system, shown in Figure 8, is used in conjunction with a change in effective lanyard pully radius to prevent the DRA from "snapping" into the deployed state. This bridle system also serves to generate the initial rotation of the DRA about the longitudinal axis when retraction is commanded. Redundant stowed status microswitches, located so that they sense contact of the top structure with the base plate, are used to turn off the servo mechanism once stowage is complete.

INTERFACE HARDWARE

The interface between the DRA mechanism and the SMM spacecraft consists of a lightweight, stiff aluminum cylinder 28.3 inches long and 21 inches diameter, called the canister, shown in Figure 9. Three hat section stringers run the length of the canister carrying loads from the three Astromast longerons directly to three I-beams in the aft end of the spacecraft. The canister not only provides the static and dynamic interface with the spacecraft, but also acts as a guide tube during jettison.

JETTISON MECHANISM

The DRA is capable of jettisoning the antenna, articulation system, Astromast, servo mechanism, and control electronics. The jettison mechanism will only be activated in the event the DRA cannot be retracted to the fully stowed condition for Shuttle recovery. This capability is provided by three ball release/jettison spring assemblies, shown in Figure 10. A lightweight but structurally sound mechanism has resulted through extensive use of 7075 aluminum. A pyrotechnic cable cutter, shown in Figure 7, severs 3 stainless steel cables that release spring loaded plungers, unloading a set of steel balls in sockets. Once the ball loads are released, 3 jettison springs eject the base plate and all assemblies attached to it out of the canister. The spring stroke is 2.5 inches providing a terminal velocity of 12 in./sec. Spring forces are designed to yield a net force through the HGAS center of gravity to minimize the tendency to rotate. In addition to the jettison release cables, the pyrotechnic cable cutter severs all electrical and RF cables interfacing the HGAS to the SMM spacecraft.

CAGING MECHANISM

To prevent inadvertent jettisoning of the HGAS after Shuttle recovery, the jettison capability is inhibited by a unique but simple caging mechanism shown in Figure 9. Caging occurs only in the stowed condition, but the mechanism allows the DRA to deploy and retract normally. It consists of a pivoted wedge which, when the DRA is fully retracted, will not allow the base plate to move relative to the canister. Consequently, if the cable cutter is accidentally fired, the jettison springs are inhibited from forcing the baseplate out of the canister. After approximately .75 inches of deployment, the caging wedges are released, and jettison is possible.

CONCLUDING REMARKS

A unique mechanism, called a Deployment/Retraction Assembly, has been described that is capable of deploying and retracting the S-band antenna and associated articulation system aboard the SMM spacecraft. Once deployed, it provides a stable and stiff structure from which the antenna can track and communicate with the TDRS system.

At the time of writing, engineering model tests are underway and the protoflight DRA is being fabricated. Engineering models of the worm gear assembly, the jettison mechanism, and the Astromast have been tested and the following has been demonstrated:

- Worm gear design has load/cycle capability in excess of that required for ground and space operation.
- Jettison mechanism ball release concept works successfully.
- The Astromast has adequate cycle life and stiffness to meet mission requirements.

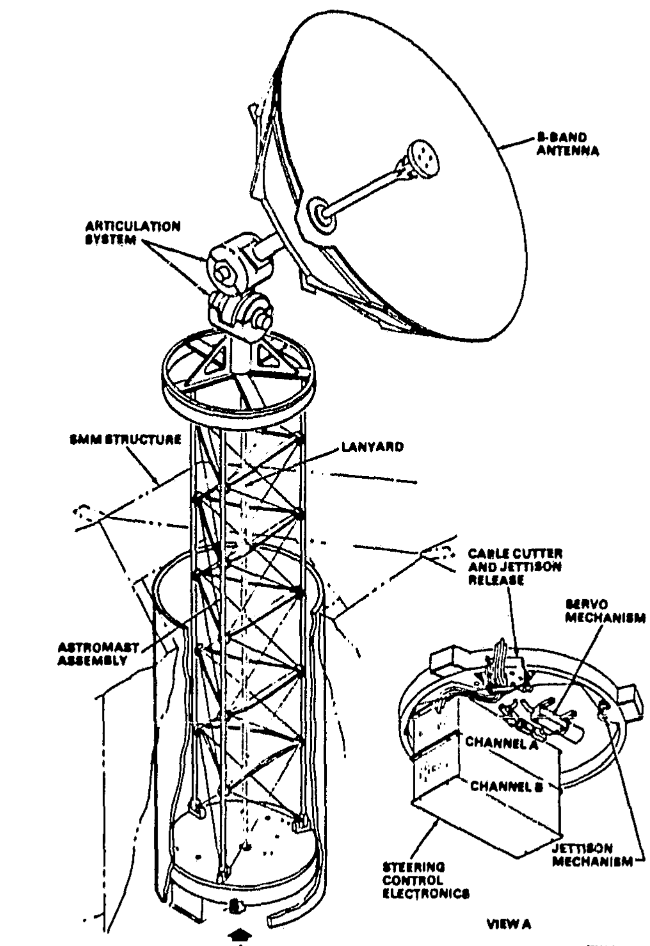


Figure 1
HGAS Deployed

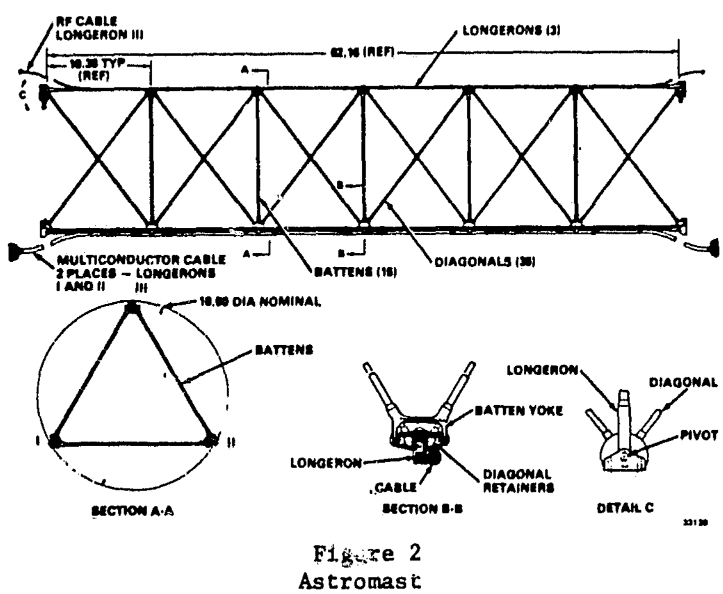


Figure 2
Astromast

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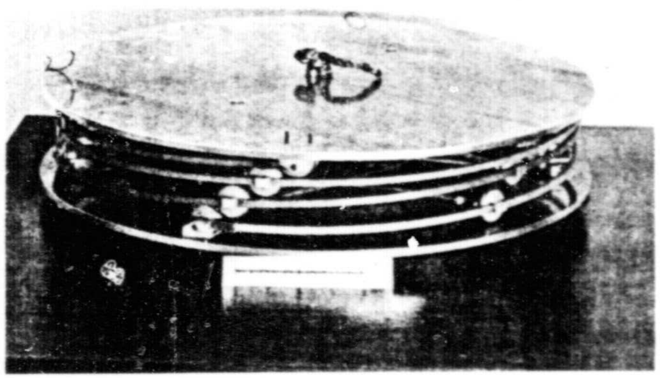


Figure 3 Astromast stowed.

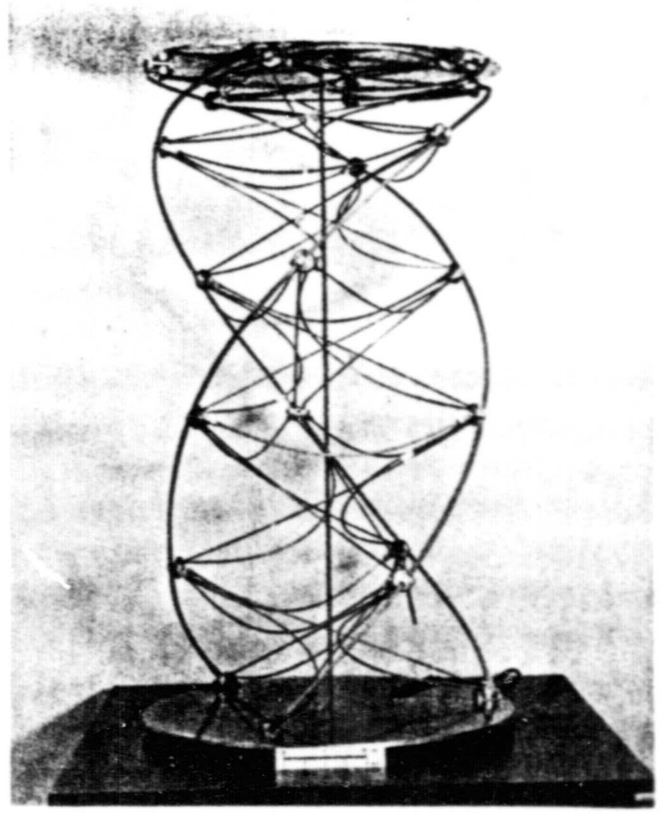


Figure 4 Astromast initial deployment stage.

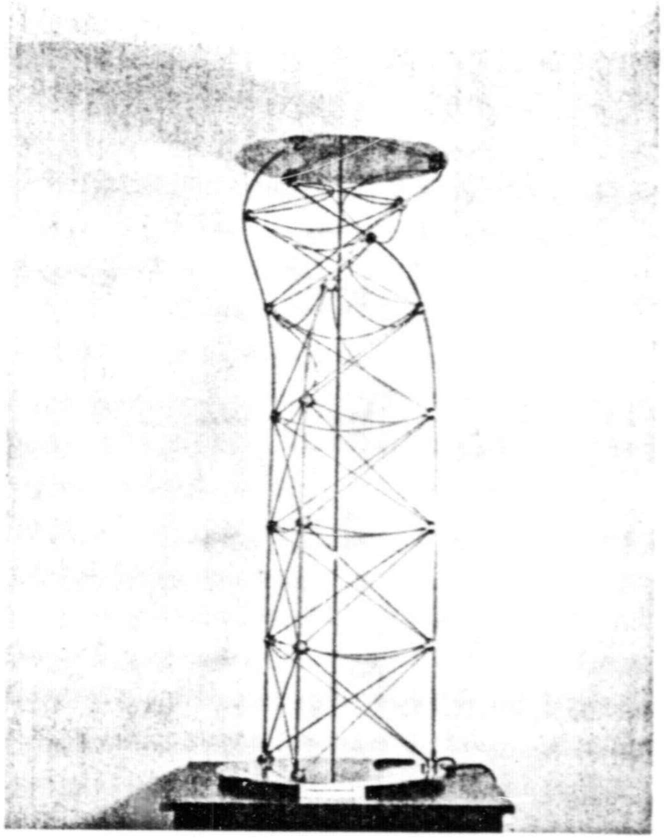


Figure 5 Astromast deploying.

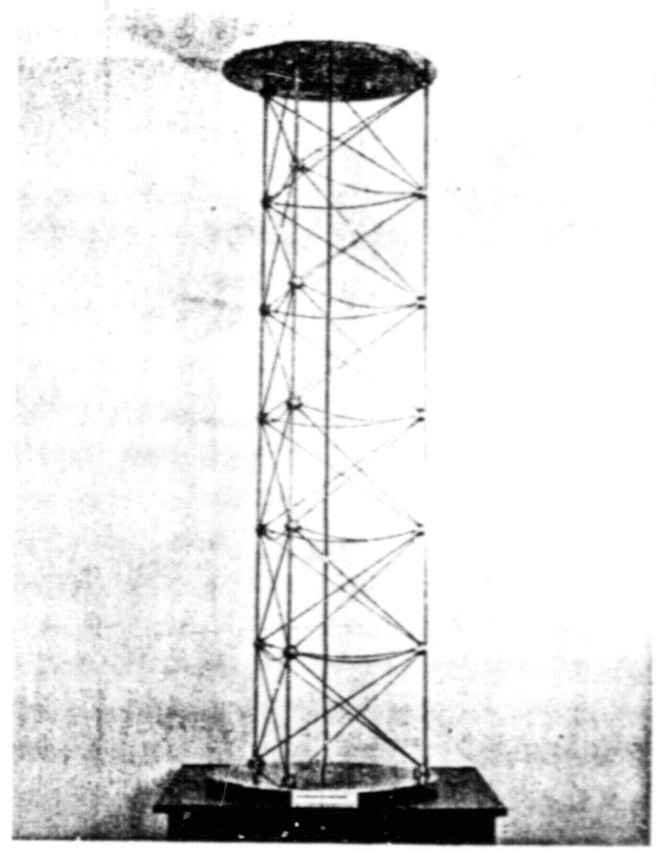


Figure 6 Astromast fully deployed.

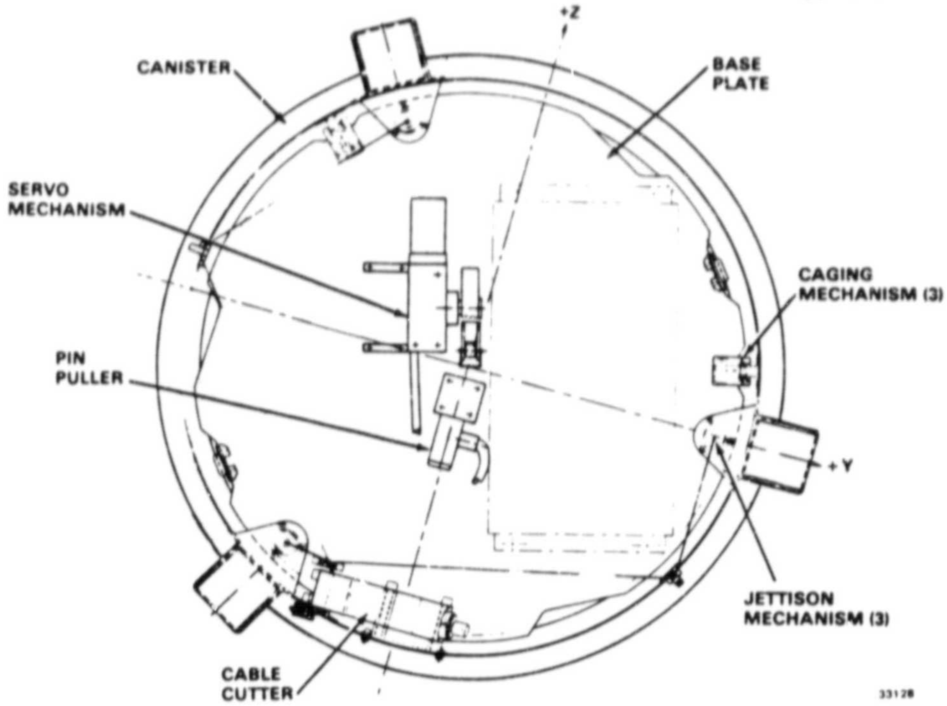


Figure 7
Bottom View of DRA

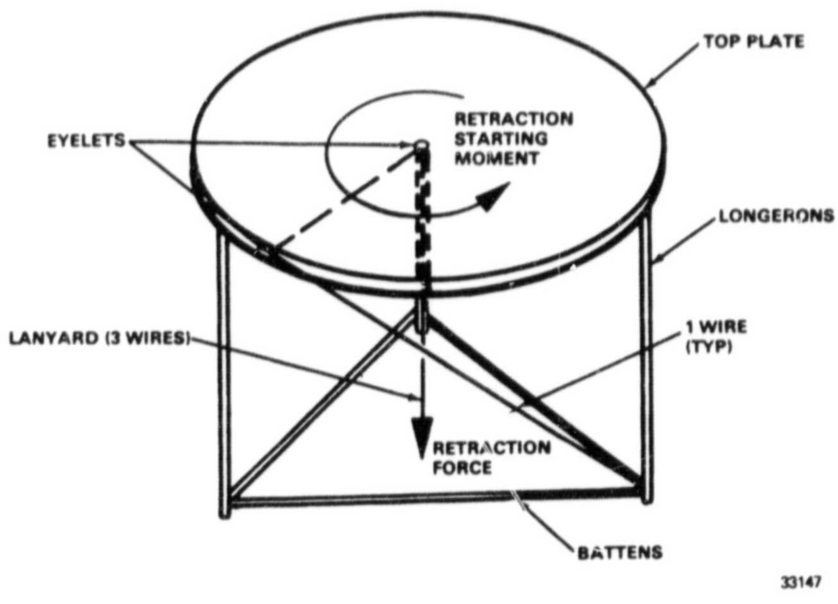


Figure 8
Bridle System

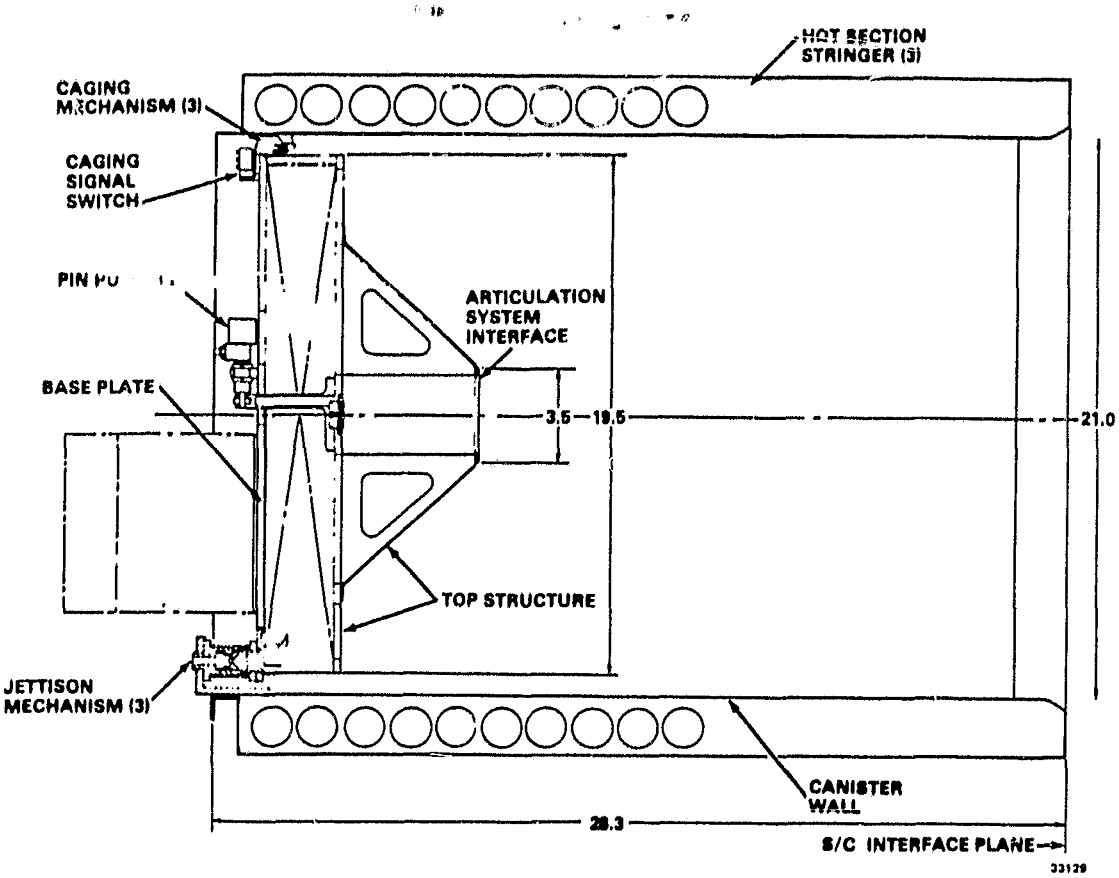


Figure 9
Side View of DRA

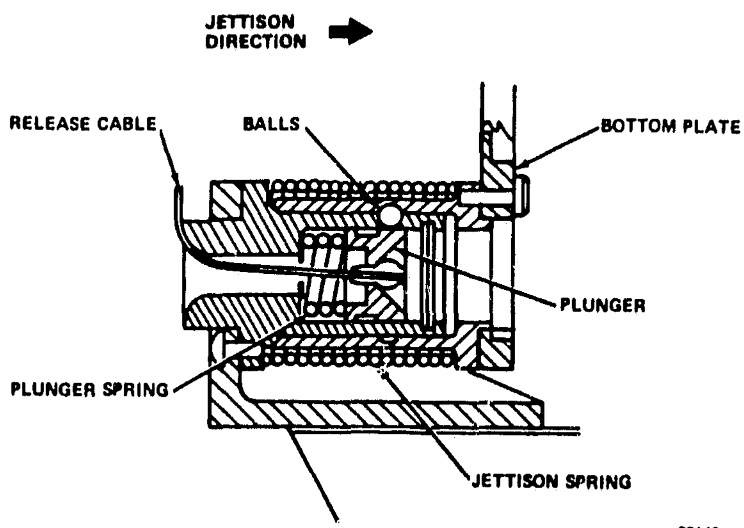


Figure 10
Jettison Mechanism

APPENDIX D

BALL
AEROSPACE SYSTEMS DIVISION

SPACE-PROVEN ELECTRO-MECHANICAL SYSTEMS

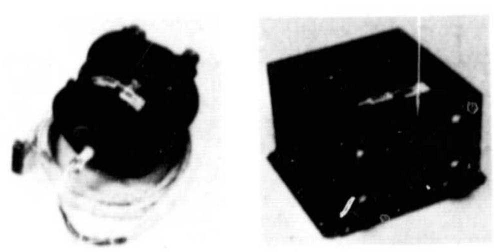
BASD has been developing electro-mechanical systems for despinning and pointing scientific instruments, antennas, solar arrays, and satellite despun platforms since 1957. BASD has remained at the forefront of this field, developing a full range of component, analytical, electronic and lubrication technologies. Over 150 cumulative years of space operation by 14 qualified drives in over 50 flights reflect their superior record of performance and reliability. Operating lifetimes of over nine years prove their long-life capabilities.

Beginning with DC torque motor drives for rocket control systems, followed by the despin and instrument pointing drives for the seven BASD-built Orbiting Solar Observatories (OSO's), BASD established its drive system capabilities in the early years of America's space program. Our Vac Kote lubrication technology, that originated with the OSO program, has been a major factor in the success of all our drives.

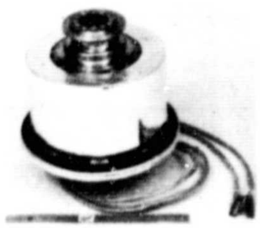
The technology BASD applies to electro-mechanical systems includes direct drive dc torque motors (brush and brushless, continuous and limited rotation, slow and high speed), stepper motors, harmonic drive and spur gear speed reducers, and resolver, potentiometer, magnetic pickup, optical encoder and synchro angle readout methods. We design and build both open and closed loop electronic control systems for these drives. These systems feature the latest in analog and digital integrated circuits and pulse width modulation techniques for efficient operation.

As one of a few companies that produces drive systems as a product for other aerospace companies, the Electro-Mechanical Products Group of BASD is experienced with a broad range of customer documentation, contractual and quality control requirements, and all relevant NASA and military specifications. We are sensitive to the importance of meeting schedules, and much of the work is done on a fixed price basis to protect the customers' financial risk. Superior technology, responsiveness to customer needs, extensive inflight space experience - these are the hallmarks of the Electro-Mechanical Products Group of BASD.

PAYLOAD RETENTION LATCH ACTUATOR

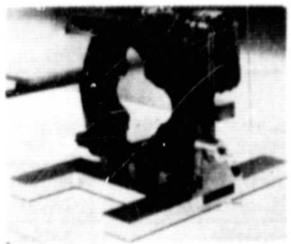


GPS SOLAR ARRAY POINTING SYSTEM

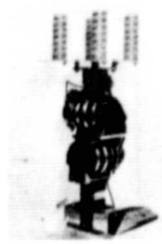


HELIOS ANTENNA REFLECTOR DESPIN DRIVE

DSCS-II PHASE 1 AND 2 PLATFORM DESPIN DRIVE



ACTIVE KEEL ACTUATOR



NIMBUS 2-AXIS ANTENNA POINTING SYSTEM

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FLIGHT MODEL EXPERIENCE

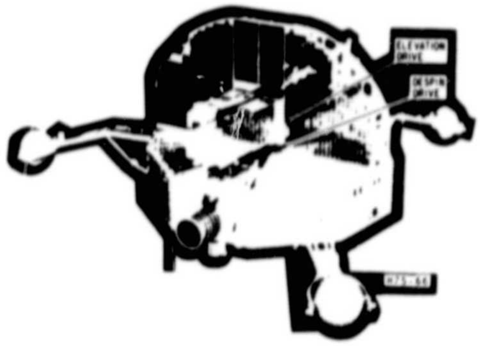
Designation	Description	Customer	Number of Units (Including Non-Flight)	Flight Status	Cumulative Period of Inflight Performance through 1980
	POINTING SYSTEMS:				
EMS-101 EMS-105	• Elevation Drive Assembly for OSO's 1-7 Spacecraft and P78-1 Spacecraft	BASD	11	Eight successful flights	330 months
EMS-121	• Biaxial Antenna Pointing System for Nimbus-F Spacecraft	GE	3	One successful flight	18 months
EMS-131	• Biaxial Solar Pointing System for AE-C, D, and E Spacecraft	RCA	4	Three successful flights	130 months
EMS-151	• Solar Pointing System for SAM-II on Nimbus-G	University of Wyoming	2	One successful flight	26 months
EMS-155	• Solar Pointing System for SAGE on AEM-B	NASA/Langley	2	One successful flight	23 months
EMS-171	• Gimballed Mirror Assembly for Airborne Laser Communication Experiment	McDonnell Douglas	1	Several Successful flights-1980	N/A
EMS-173	• Gimballed Mirror Assembly for Satellite Laser Communication Experiment	McDonnell Douglas	1	1983 Launch	N/A
	SOLAR ARRAY DRIVE SYSTEMS:				
EMS-201	• Solar Array Drive Assembly for Classified Program	TRW	9	Several flown	Unknown
EMS-221	• Solar Array Drive and Power Transfer System for NDS (GPS) Satellite	Rockwell International	12	Six Successful flights	138 months
EMS-231	• Solar Array Orientation System for NTS-II (GPS) Satellite	Naval Research Laboratory	2	One successful flight	43 months
EMS-241	• Solar Array Drive System for P80-1 Satellite	Rockwell International	1	1983 Launch	N/A
	DESPIN DRIVE SYSTEMS:				
EMS-301 EMS-305	• Azimuth Drive Assembly for OSO's 1-7 Spacecraft	BASD	11	Seven successful flights	330 months
EMS-311	• Mechanical Despin Assembly for Skynet 1 and 2/NATO 1 and 2	Ford Aerospace	12	Four successful flights	241 months
EMS-313	• Mechanical Despin Assembly for SIRIO Spacecraft	Selenia-Italy	4	One successful flight	41 months
EMS-321 EMS-323	• Despin Mechanical Assembly for DSCS-II Spacecraft	TRW	22	Ten successful flights	385 months
EMS-331	• Despin Drive Assembly for Helios A and B Spacecraft	MBB-Germany	5	Two successful flights	134 months
EMS-333	• Antenna Drive Mechanism for Maritime Observation Spacecraft	Mitsubishi Electric-Japan	1		In Development
EMS-335	• Despin Drive Assembly for Planet-A Spacecraft	Nippon Electric Japan	1		In Development
	ACTUATORS:				
EMS-401	• Drive Unit Actuator for Classified Program	HESD	6	Classified Program	Unknown
EMS-410	• Payload Retention Latch Actuator for Space Shuttle	Rockwell International		September 1981	N/A
EMS-420	• Active Keel Actuator for Space Shuttle	Rockwell International	2	September 1981	N/A
EMS-450	• Rendevous Radar Deployment Actuator for Space Shuttle	Rockwell International		1982	In Development

DB-212
(1 of 2)

OTHER DRIVE EXPERIENCE

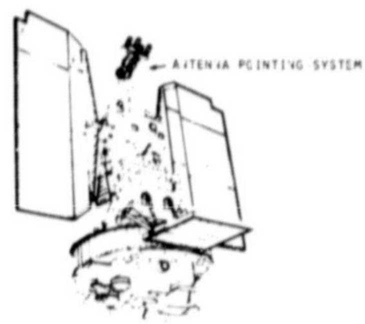
Designation				Number of Units
EMS-111	Biaxial Antenna Drive	Antenna Pointing System	Bell Telephone Laboratories	1
EMS-141	Biaxial Solar Pointing System	Solar Instrument for Aircraft and Balloon Flights	Jet Propulsion Laboratory	1
EMS-161	Geared Stepper Drive for High Inertia Loads	Antennas or Solar Array Pointing System	BASD	1
EMS-211	High Power Capacity Orientation Drive and Power Transfer Assembly (ODAPT)	Large Space Station	Lockheed-MSC	1
EMS-251	Geared Stepper Drive	Solar Array, Antenna, or Experiment Pointing System	BASD	1
EMS-501	Satellite Vacuum Test Fixture	NASDA (Japan) Test Facility	Hitachi-Japan	1

EMS-101 ELEVATION DRIVE ASSEMBLY FOR ORBITING
EMS-105 SOLAR OBSERVATORY (OSO) 1-7 SPACECRAFT



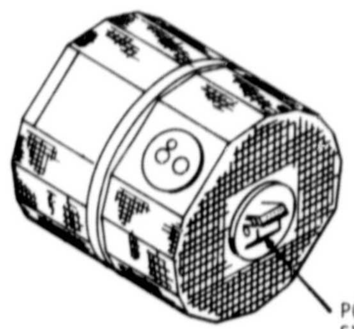
The elevation axis drive is the second-half of the solar science instruments pointing system. After the despin drive provides azimuth orientation of the upper platform, the elevation axis then rotates to position the optical axes of the pointed experiments to the center of the sun. Offset pointing capability is available. Coarse and fine BASD sun sensors supply the error signal.

EMS-121 BIAXIAL ANTENNA POINTING SYSTEM FOR NIMBUS-F SPACECRAFT



This biaxial antenna pointing system consists of a gimbal assembly (GA) and a drive electronics package. The GA is mounted at the very top of the spacecraft and orients a high gain antenna toward the Applications Technology Satellite (ATS-F).

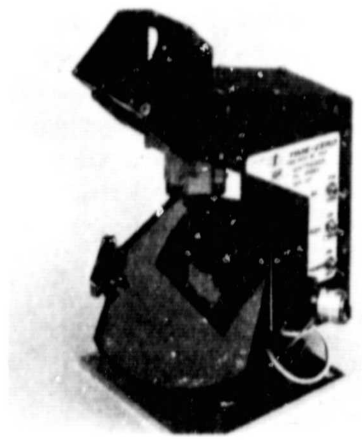
EMS-131 BIAXIAL SOLAR POINTING SYSTEM FOR ATMOSPHERE EXPLORER (AE) C, D, AND E SPACECRAFT



This system consists of a solar pointing platform, a control electronics package, and a fine sun sensor assembly. The purpose of the SPS on the three AE satellites is to automatically acquire and point an extreme ultraviolet spectrophotometer towards the sun.

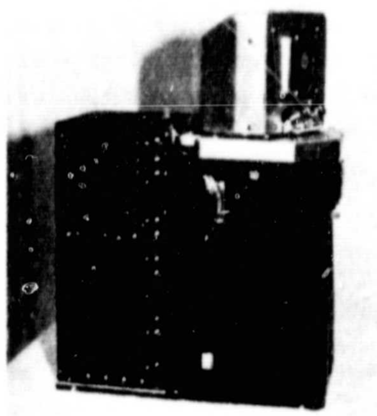
EMS 141 BIAXIAL SOLAR IMAGING POINTING SYSTEM

This drive is used in the Mark I Aircraft Interferometer built by Jet Propulsion Laboratory to measure the atmosphere trace constituents. It is used on high altitude aircraft and balloon flights.



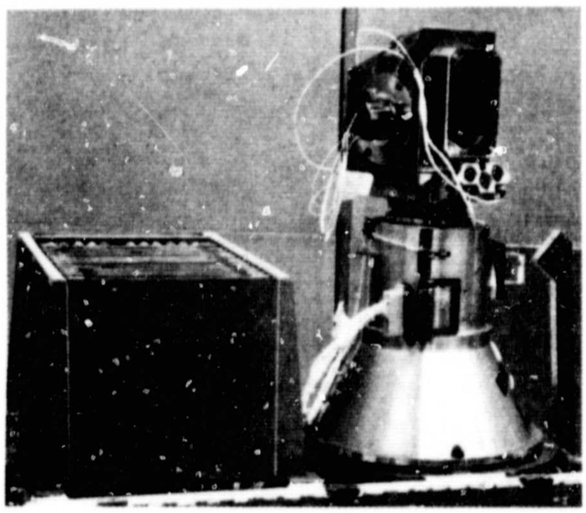
EMS 151 SOLAR POINTING SYSTEM FOR STRATOSPHERIC AEROSOL MONITOR (SAM) ON NIMBUS-G

This biaxial pointing system on the Nimbus-G spacecraft points at the sun during sunrise and sunset. The solar image is reflected from the scan mirror through a Cassegrain telescope to the science detector aperture. The image is scanned across the aperture by the motion of the scan mirror.



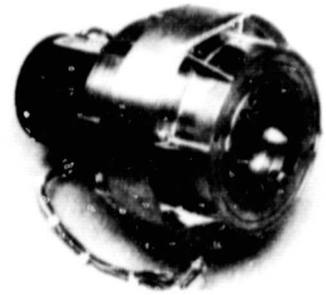
EMS 155 SOLAR POINTING SYSTEM FOR STRATOSPHERIC AEROSOL AND GAS EXPERIMENT (SAGE) ON AEM-B

This biaxial pointing system on the Applications Explorer Mission (AEM-B) spacecraft points at the sun during sunrise and sunset. BASD also provided the science experiment which evaluates aerosols and ozone constituents in the earth's upper atmosphere.



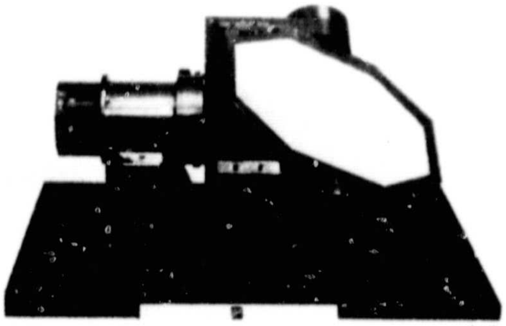
EMS 161 GEARED STEPPER DRIVE

This geared stepper drive is designed to move a high inertia load in small steps. It is light-weight, has excellent lateral rigidity, and includes redundant precision potentiometers for position feedback signals. This drive is well-suited for pointing large antennas or orienting large solar arrays.



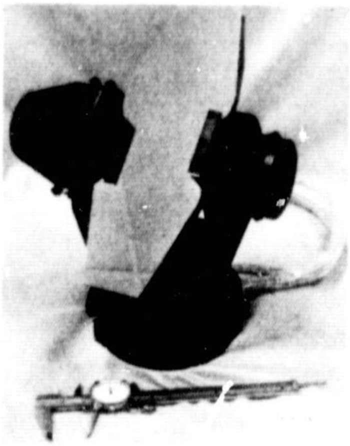
EMS 171 GIMBALLED MIRROR ASSEMBLY FOR AIRBORNE LASER COMMUNICATION EXPERIMENT

This system consists of a two-axis gimbal assembly and an optically flat mirror. The airborne system tests the feasibility of using an accurately pointed moving mirror to redirect laser transmissions from a moving source. The system has been successful on a military aircraft.



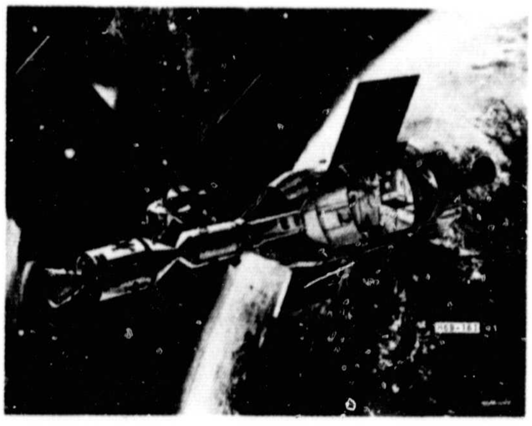
EMS 173 GIMBALLED MIRROR ASSEMBLY FOR SATELLITE LASER COMMUNICATION EXPERIMENT

This system consists of a two-axis gimbal assembly and an optically flat mirror. It is a derivative of the airborne gimballed mirror assembly. This system is installed in a spacecraft and carried into orbit by the Space Shuttle. It is used to redirect the line-of-sight of laser transmissions.



PRIMARY MIRROR POINTING SYSTEM FOR APOLLO TELESCOPE MOUNT (ATM)/HARVARD COLLEGE OBSERVATORY (HCO)

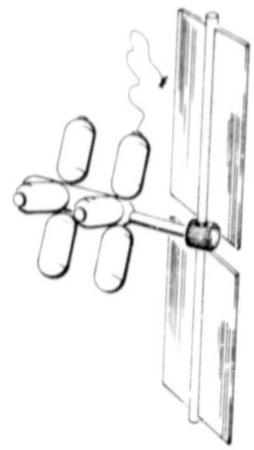
The pointing system which controls the primary mirror of the HCO/ATM instrument supports the mirror frame on flexure pivots, eliminating friction and hysteresis. The mirror is controlled to form a raster scan pattern of 60 lines that are five arc-minutes long and separated by five arc-seconds. It can also point to any chosen spot within the five arc-minute square area.



EMS 201 SOLAR ARRAY DRIVE ASSEMBLY FOR CLASSIFIED PROGRAM

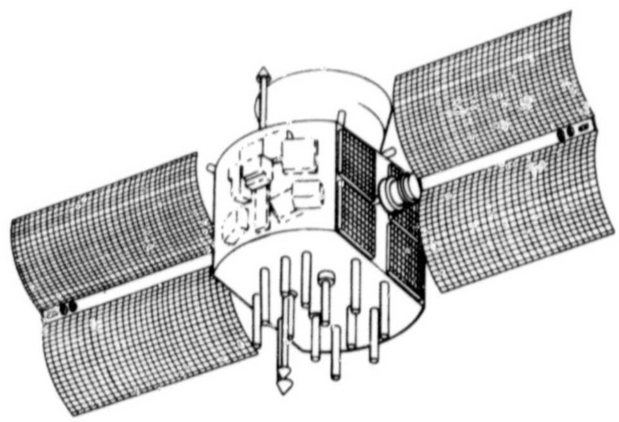
BASD developed, built, and tested several of these drive assemblies to orient two separate solar panels toward the sun. Each assembly consists of a drive, a slipring module, a resolver module, and a drive shaft. This assembly has been used in a classified application.

EMS 211 HIGH POWER CAPACITY ORIENTATION DRIVE AND POWER TRANSFER ASSEMBLY (ODAPT)



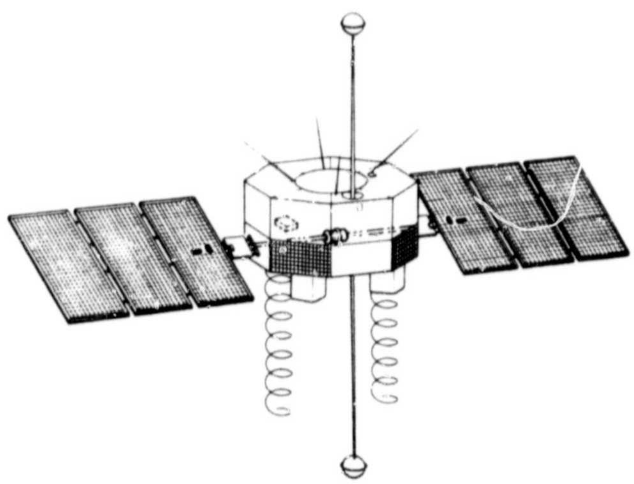
The ODAPT assembly was built to point very large solar arrays toward the sun and transfer the 100kw of electrical power generated by the arrays across the rotating interface to the Space Shuttle-launched Modular Space Station. A clear passageway through its center is provided for cargo and astronauts. The ODAPT design permits continuous (360-degree), bidirectional solar tracking in two axes.

EMS 221 SOLAR ARRAY DRIVE AND POWER TRANSFER SYSTEM FOR NAVIGATIONAL DEVELOPMENT SATELLITE (NDS) GLOBAL POSITIONING SYSTEM (GPS) SATELLITE



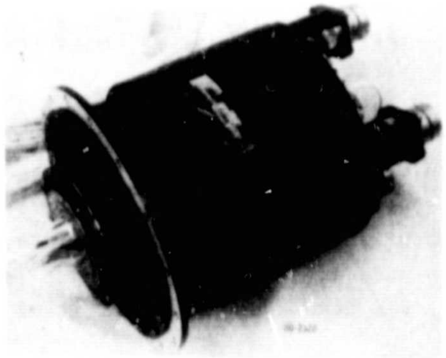
The two independent solar arrays of the Navigational Development Satellite (NDS) are each oriented by individual drive units and their related sun sensors and control electronics. The closed loop pointing of the arrays by this drive system, with each drive assembly operating in synchronism with the other, maximizes power output through accurate control of the solar array/sun angle.

EMS 231 SOLAR ARRAY ORIENTATION SYSTEM FOR NAVIGATIONAL TECHNOLOGY SATELLITE-II (NTS) GLOBAL POSITIONING SYSTEM (GPS) SATELLITE



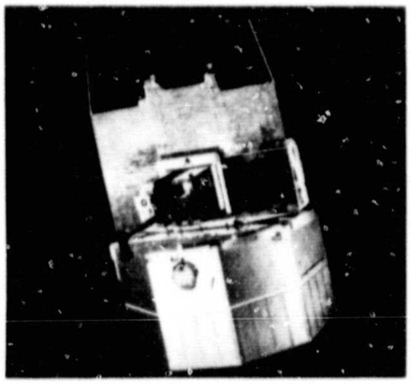
On the Navigational Technology Satellite component of the Global Positioning System, a single drive unit orients the two solar arrays. This drive, with its associated control electronics and sun sensors, operates in a closed loop mode to acquire the sun automatically and position the arrays for maximum power output.

EMS 241 SOLAR ARRAY DRIVE SYSTEM FOR P80-1 SATELLITE



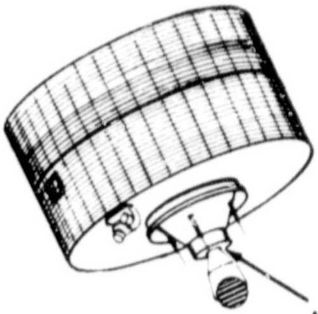
This solar array drive system is designed for large arrays and has a current transfer capacity of 50 amperes. The system includes a drive, electronics and sun sensors. The reference position is incremented at nominal spacecraft orbit rate, and is trimmed by means of the sun sensors to maintain sun pointing.

EMS-301 AZIMUTH DRIVE ASSEMBLY FOR ORBITING
EMS-305 SOLAR OBSERVATORY (OSO 1-7 SPACECRAFT)



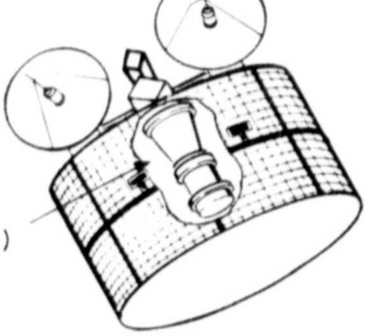
The azimuth axis drive despins the sail, which contains the solar array and the solar instrument package. Coarse and fine BASD sun sensors supply the error signals. In addition to general position orientation for the solar array, the drive is responsible for precision pointing and offset pointing for the science instruments.

EMS-311 MECHANICAL DESPIN ASSEMBLY FOR SKYNETS
EMS-313 1 AND 2/NATO'S 1 AND 2 SATELLITES, AND FOR SIRIO SPACECRAFT



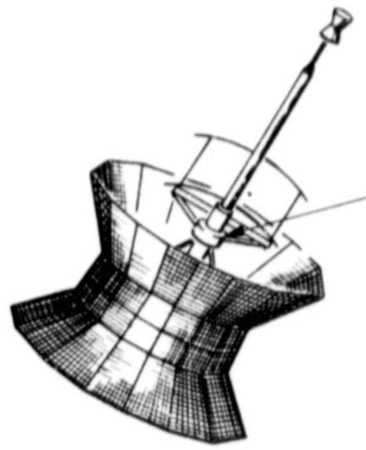
This RF antenna despin drive is used on the United Kingdom communication satellite, Skynet, and NATO 1 and 2 satellites. It is designed to perform for five years, continuous duty. The drive is also used on the Italian SIRIO experimental satellite. One drive on Skynet ran for 9.4 years before being deactivated by ground command.

EMS-321 DESPIN MECHANICAL ASSEMBLY FOR DEFENSE
EMS-323 SATELLITE COMMUNICATIONS SYSTEM (DSCS) II SPACECRAFT



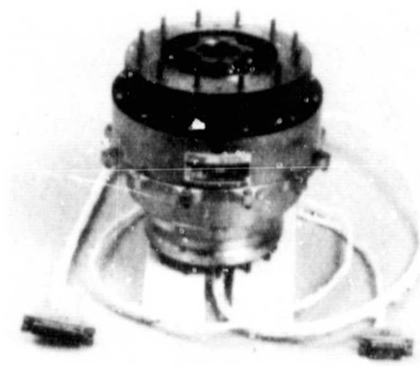
The despin mechanical assembly is the vital link between spinning and despun sections of the DSCS-II spacecraft. It supports and drives the despun platform and provides multiple electrical power and signal paths across the interface.

EMS-331 DESPIN DRIVE ASSEMBLY FOR HELIOS A AND B SPACECRAFT



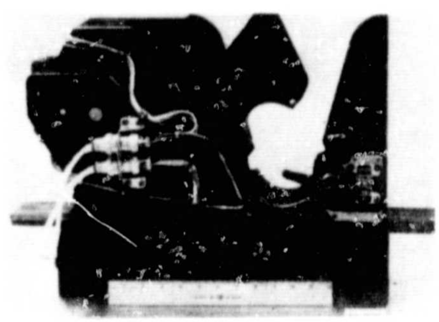
This drive was developed to despin the antenna reflector on the HELIOS A and B satellites. The HELIOS orbits within 26.9 million miles of the sun, requiring the drive to operate in a widely varying thermal environment. A special feature is an extremely low magnetic dipole moment.

EMS 333 ANTENNA DRIVE MECHANISM FOR MARITIME OBSERVATION SPACECRAFT



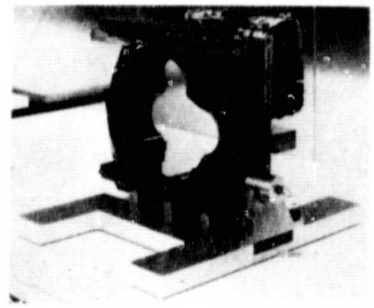
This system was developed for Mitsubishi Electric Corporation, and it is used to rotate the scanning microwave radiometer antenna on the Japanese Maritime Observation Satellite (MOS). This drive mechanism operates between 10 and 30 rpm on the three-axis stabilized spacecraft. It is a derivative of a drive developed for the HELIOS spacecraft.

EMS 410 PAYLOAD RETENTION LATCH ACTUATOR FOR SPACE SHUTTLE



This actuator is designed to secure payloads in the Shuttle Orbiter bay weighing up to 65,000 pounds. It attaches to the sill longeron in the Shuttle bay and provides restraint in two axes. It is used in conjunction with the Active Keel Actuator (EMS 420) described below.

EMS 420 ACTIVE KEEL ACTUATOR FOR SPACE SHUTTLE



This actuator attaches to the keel of the Shuttle Orbiter bay and secures payloads weighing up to 65,000 pounds. Two to four EMS 410 Actuators and one EMS 420 Actuator are used in concert. Redundancy is provided in all electrical and mechanical systems to minimize failures.

APPENDIX E

HARRIS CORPORATION
GESD

TRACKING AND DATA RELAY SATELLITE SINGLE ACCESS DEPLOYABLE ANTENNA

Summary

The Tracking and Data Relay Satellite (TDRS) utilizes two 4.8 meter deployable antennas for accessing and communicating with high data rate user satellites. These single access antennas operate at the dual frequencies of S- and Ku-band, with full tracking capability at Ku-band. The stringent gain and tracking requirements at Ku-band require state-of-the-art designs in the deployable reflector and the dual frequency feed system. To meet these requirements, special surface shaping techniques and thermally stable materials were developed for the deployable reflector. A unique dual frequency cassegrain feed system utilizing sidelobe cross-over for tracking was developed to meet the tracking requirements. The feed system tracking performance is essentially insensitive to reflector distortions. This paper summarizes the design and performance of the TDRS single access antenna design.

I. Reflector Design

Figure 1 illustrates the deployable reflector design. The reflector utilizes eighteen graphite fiber reinforced plastic (GFRP) ribs to shape and support the reflective mesh surface. The number of ribs is based on a trade-off considering surface tolerance and weight. As the number of ribs increases, the surface error decreases, while weight increases. The minimum number of ribs consistent with the surface tolerance requirements is, therefore usually selected. The ribs are circular in cross-section tapering in diameter from the root to the tip. The reflective mesh surface is attached to the ribs by adjustable standoffs and therefore the tolerance on rib shape is not a critical parameter. The ribs are typically fabricated to a constant radius of curvature rather than a parabolic shape.

The reflective mesh consists of gold-plated wire which is knitted into a soft (low spring rate), elastic mesh. The mesh opening size can be varied to ensure adequate RF reflectivity for a given requirement. The required reflector surface tolerance is achieved with minimum weight through the use of a secondary drawing surface technique. A series of circumferential quartz cords is attached to the back of the ribs by adjustable standoffs. A second series of quartz cords is attached to the front mesh surface. These "front" cords run parallel to the "back cords". The front and back cords are connected by a series of tie wires. By properly adjusting the rib stand-off heights, the back cord

Figure 1. TDRSS Antenna Subsystem in STV Chamber

geometry, and these individual tie wires, an accurate surface contour is achieved.

For setting the reflector surface, the contour is measured in the face-up and the face-down positions. The measured face-up and face-down positions are then averaged to determine the "zero-gravity" surface contour. This contour is then compared on a point-by-point basis with the desired parabolic contour and surface adjustments made as necessary to achieve the desired manufacturing contribution to the total surface tolerance budget. The setting process is iterative. Two to three setting iterations yield a high accuracy contour.

Deployment of the reflector surface is achieved in a totally controlled manner to ensure no degradation of the accurate reflector surface occurs and to essentially eliminate any transfer of stored energy to the spacecraft. The mechanical deployment system (MDS) consists of a carrier mounted to the moving section of a recirculating ballnut pair on the ballscrew shaft. Connected between the carrier and the ribs are pushrods that transmit the required force and motion to deploy the ribs. As the carrier moves along the screw shaft, the ribs are rotated from their stowed to their fully deployed position. Latching in the deployed position is accomplished by driving the carrier and linkages through an overcenter condition (relative to the rib pivot position).

The feed support structure provides the primary structure for the stowed antenna as well as serving as the structure for support of the dual frequency feed and subreflector. This support structure consists of a 6 member GFRP truss structure and monocoque (single skin) quartz radome structure. The subreflector is a sandwich construction of skins and a honeycomb core.

Thermal control of the reflector ribs and feed support structure is accomplished with multi-layered insulation blankets. These blankets utilize inner layers of embossed aluminized kapton and an outer layer of kapton with vapor deposited aluminum striping. The percentage of VDA striping is based on the average solar absorptivity (αs) and emissivity (ε) values desired. The number of layers is selected to provide a desired thermal time constant and to minimize distortions due to diametral temperature gradients.

II. Feed Design

Five-Horn Cassegrain feeds are very desirable for communications antennas for which the tracking requirements are usually not very stringent. The simplicity of the sum channel makes possible greater bandwidth and higher efficiency. The historical problem has been the tradeoff between a sufficiently large sum horn for good illumination efficiency and sufficiently small error horn spacing for adequate error channel secondary pattern crossover. When the error horn spacing is fixed so that the error horn secondary pattern crossover is at the first null (or on the main offset beam), the sum horn aperture size is such that large spillover past the subreflector is incurred. In order to make the sum horn larger (to reduce spillover), the error horn spacing must be increased resulting in a crossover on the first sidelobe. Generally, crossover on the first sidelobe or beyond has been avoided because of the sensitivity of the low-level sidelobes to various factors, including reflector distortion, frequency change and blockage. The sensitivity of tracking performance to reflector distortion is especially important for spaceborne antennas.

A Five-Horn Cassegrain feed has been developed for the TDRSS 4.8-meter deployable antenna, which uses the up-taper of the dual shaped reflectors in conjunction with a special error horn design to produce a stable tracking crossover on high-level sidelobes. This permits use of a larger sum horn and greater spillover efficiency.

The error horns are rectangular with the narrow dimension in the plane of the associated tracking crossover. Because the error horns have a much broader illumination in this plane than the sum horn, the up-taper which produces uniform reflector illumination (maximum gain) for the sum channel results in a highly inverted distribution for the error horns, and hence the desired high sidelobes.

III. Antenna Performance

Orbital antenna gain performance is predicted both with loss budgets and with a composite computer math model. Although the composite math model provides a more accurate prediction, the loss budget has a diagnostic advantage in that the significance of the various components is revealed by inspection.

The loss budget for the 4.8 meter antenna is shown in Table I. The defocus, roughness, and pointing error losses are based on the budgeted worst case distortions. The mesh reflectivity loss is based on measured values for the gold-plate. Radome loss uses calculated flat-plate approximations and measurements on similar radomes. The blockage loss is based on the aperture blockage resulting from the feed support structure. The feed related losses are based on measured range test results with the breadboard feed.

The defocus, roughness, and pointing error values for the reflector are calculated analytically. The pillow effects result from a natural phenomenon. Due to the pretension of the mesh and the doubly curved parabolic surface, the mesh tends to "pillow" toward the focal point. This phenomenon is described by the partial differential equations of membrane theory. The secondary drawing surface technique significantly reduces this error but the mesh between tie points still bulges toward the focal point, forming small "pillows" between adjacent tie points.

The largest source of surface tolerance is the manufacturing (or setting) tolerance. This error source represents the physical ability of technicians to adjust the surface and is also constrained by the time available to make iterative settings. Typically 2 to 3 iterations are required to achieve this precision.

The thermal distortion effects of the reflector and feed support are determined using analysis techniques. Cord creep results from a slight permanent preset of the quartz cords when they are initially pretensioned. Preconditioning of the cords can be accomplished if this source of error becomes significant. Dryout shrinkage error results from changes in the dimensions of the GFRP ribs and feed support when moisture absorbed by the GFRP outgasses in the orbital environment. Both the cord creep and GFRP dryout error have been verified by element tests of these materials.

Other Pertinent Antenna Data is shown below:

Weight:	53 lbs.	
Diameter:	16 Feet	
Stowed Volume:	Cylindrical:	31-inch diameter, 100-inch length
Interface:	18-inch, 3-bolt circle	

ORIGINAL PAGE IS OF POOR QUALITY

Table I. Antenna Loss Budget

	2.025	2.2	2.3	11.7	14.0	13.75	14.896	15.121
DEFOCUS	--	--	--	0.11	0.16	0.15	0.18	0.19
ROUGHNESS	0.01	0.01	0.01	0.36	0.51	0.49	0.58	0.60
MESH REFLECTIVITY	0.03	0.03	0.03	0.46	0.53	0.53	0.57	0.58
SCALLOP LOSS	0.23	0.23	0.23	0.19	0.19	0.19	0.19	0.19
RADOME LOSS	0.05	0.05	0.05	0.20	0.20	0.20	0.20	0.20
BLOCKAGE	0.22	0.22	0.22	0.22	0.22	0.22	0.22	0.22
FEED LOSS	0.58	0.58	0.61	0.50	0.30	0.30	0.30	0.30
VSWR	0.15	0.15	0.15	0.10	0.05	0.05	0.05	0.05
SPILLOVER-TAPER	1.78	1.95	2.24	1.11	0.72	0.72	0.72	0.72
	3.05	3.22	3.54	3.25	2.88	2.85	3.01	3.05
100% GAIN	40.08	40.80	41.18	55.31	56.87	56.72	57.41	57.54
NET GAIN	37.03	37.5	37.64	52.06	53.99	53.87	54.40	54.49
SPEC	36.0	36.8	36.8	50.0	52.8	52.6	53.6	53.6
MARGIN/MEASUREMENT TOLERANCE	1.0	0.8	0.8	2.1	1.2	1.3	0.8	0.9

APPENDIX F

HUGHES AIRCRAFT COMPANY (HAC)
SPACE AND COMMUNICATIONS GROUP

Information not available.

APPENDIX G

FAIRCHILD SPACE AND ELECTRONICS COMPANY

A HIGH STRENGTH, TORSIONALLY RIGID, DEPLOYABLE AND RETRACTABLE MAST FOR SPACE APPLICATIONS

Lamont DiBiasi and Richard Kramer*

ABSTRACT

The era of retrieving and/or servicing satellites in orbit has mandated that extendable elements, such as those used to deploy solar arrays, thermal radiators, communications antennas, instruments and numerous other appendages have inherent in their design a highly reliable retraction capability. Throughout the past year a structural mast has been developed which during and after full deployment produces a supporting structure with the characteristics of a high bending moment capability, high stiffness and, particularly important for instrument deployment, a high degree of position repeatability and torsional rigidity. These features have been accomplished while providing an easily retractable mast with a high life cycle capability. Since these properties are consistent throughout the full range of deployed lengths, partial deployments of retractions can be utilized for check-out, balance, fine tuning or whatever other reason may be deemed necessary for operation modes or spacecraft stability.

INTRODUCTION

The mast, shown in Figures 1 & 2 to be discussed is of a triangular cross-section and is formed by the interlocking of three identical strips of material along their common edges. The interlocking of the edges is achieved by a meshing of a series of socket-type inserts permanently attached to rolled tabs alternately spaced along the length of each strip of material. Since all the edges are locked internal to the deployment mechanism, a fully formed mast with full mechanical properties is always being subjected to any induced loading. In the stowed configuration, the material is unlocked and therefore it is very flexible. This allows great packaging freedom since the three separate storage spools of material can be located remote and in any orientation with respect to the locking station within the mechanism. The packages for a given mast can therefore range from a long cylindrical configuration to a flat rectangular one. This paper will cover the design requirements, design philosophy, fabrication methods and test methods and results for a specific model mast.

*Fairchild Space Company, Germantown, Maryland 20874-0811

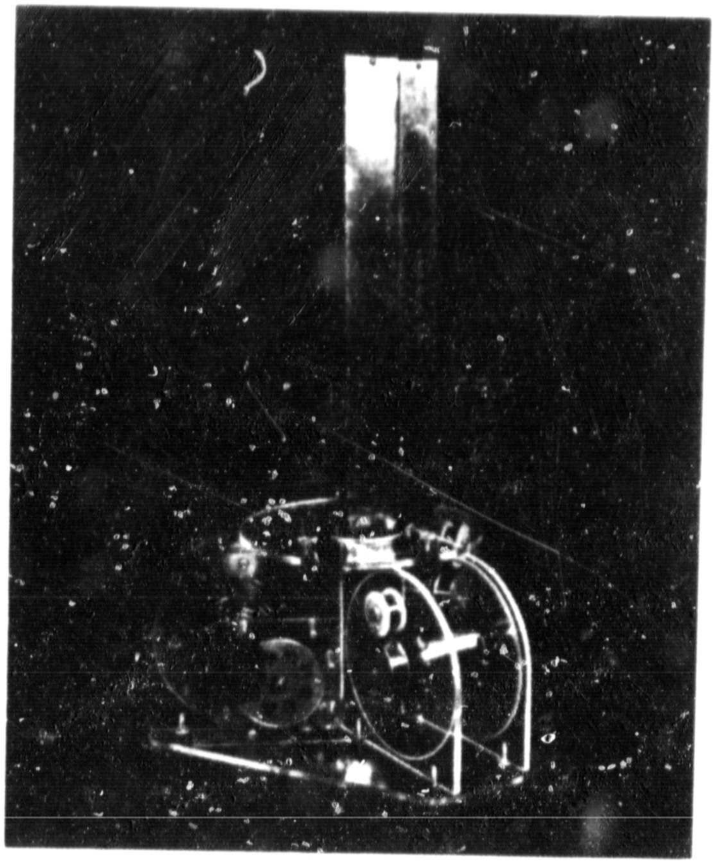


Figure 1

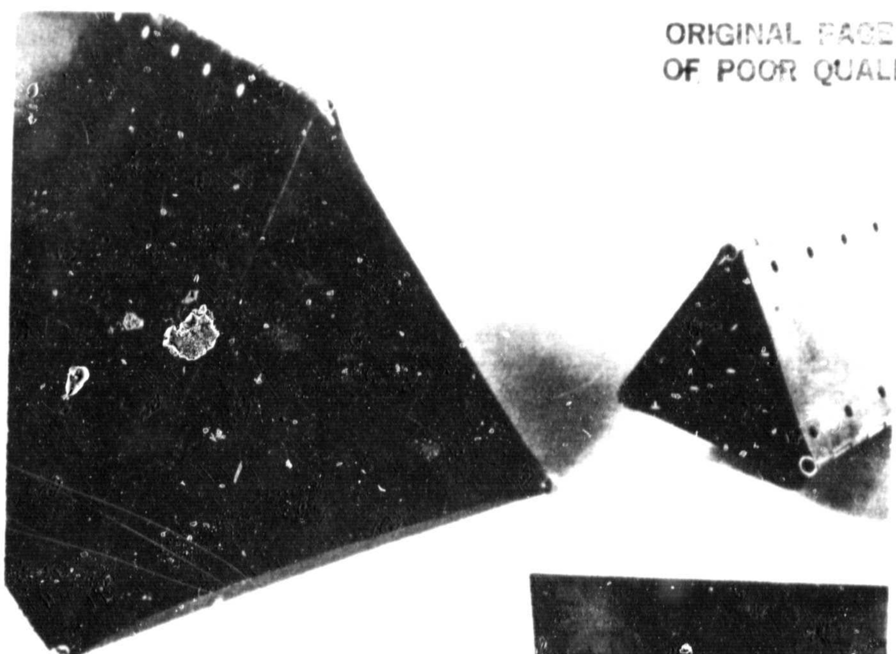


Figure 2

APPENDIX H

TRW INCORPORATED
SPACE TECHNOLOGY GROUP

No response.

APPENDIX I

LETTER OF SOLICITATION

National Aeronautics and
Space Administration

Goddard Space Flight Center
Greenbelt, Maryland
20771

NASA

Reply to Attn of 727

Dear ___:

The NASA/Goddard Space Flight Center plans to issue in the Fall of 1982 a Performance and Interface Document (PID) entitled "Electro-Mechanically Steered Antenna Systems for TDRSS Users."

The document is intended to provide general performance capabilities and interfacing information to prospective TDRSS User mission programs which are in early studies and design phases. Since there have been a number of flight capable designs completed and tested, it is our intent to publish as appendices inputs from those organizations that have designed, built and tested systems that are adaptable to missions other than those for which they were intended.

Therefore, the purpose of this letter is to solicit from you the essential electrical, mechanical and thermal performance and interface data on the flight systems units you consider applicable. Should you not have delivered a complete system the applicable four major subassemblies (antenna, gimbals, mast and drive electronics) can be supplied.

The government will not be responsible for costs incurred or interpretations of your inputs. We view these data are readily available and are or can be easily assembled in your desired format. The information is requested on nine 8 1/2" x 11" pages maximum including drawings and pictures.

It would be appreciated if your response describes, as a minimum, the following list of items:

a. The hardware actually delivered, or to be delivered.

b. The estimated ultimate performance/interface capability of the basic design approach for both command-pointed and autotrack operational systems. The range of reflector sizes (or other directional antenna configurations) the gimballs are capable of handling for low earth orbiting spacecraft should be included. Achievable overall pointing accuracies should be included as well as the suitability and adaptability to Delta and Shuttle vehicle launched missions.

We would appreciate your response within 3 weeks of the receipt of this letter. To assure no translation errors, should your response need be transposed for printing purposes, we will send you a draft of your inputs for editing corrections. Two weeks will be allotted for this process.

I thank you for your consideration and cooperation. We believe that issuance of this PID will mutually benefit both industry and government in assuring that the best electromechanical steered antenna systems to match missions requirements will be selected based on performance and economic considerations.

Should you desire to further discuss this request please call me at 301-344-9067.

Sincerely,

Original Signed by

Richard P. Hockensmith
Microwave Instrument and
RF Technology Branch
Instrument Division