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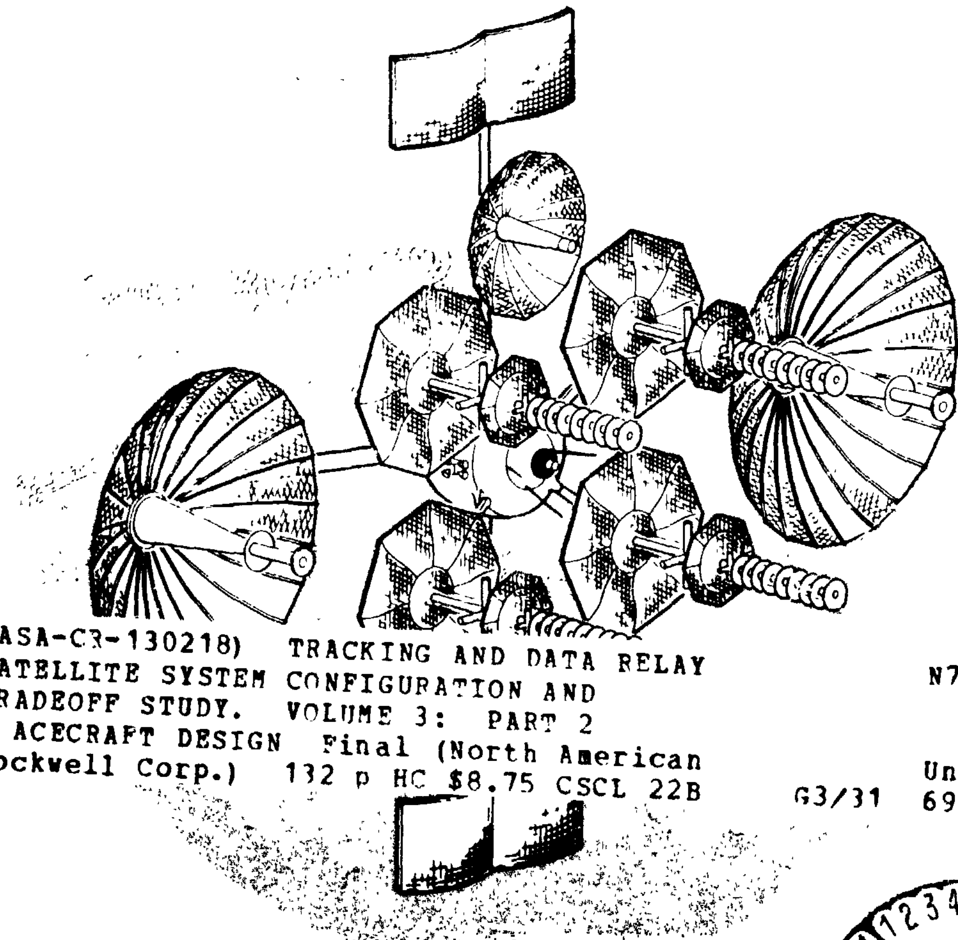
SD 73-SA-0018 3

PART II FINAL REPORT

TRACKING & DATA RELAY SATELLITE SYSTEM CONFIGURATION & TRADEOFF STUDY

VOLUME III SPACECRAFT DESIGN (PART II)



APRIL 1973

SUBMITTED TO
GODDARD SPACE FLIGHT CENTER
NATIONAL AERONAUTICS & SPACE ADMINISTRATION

Space Division
Rockwell International

12214 Lakewood Boulevard, Downey California 90241

IN ACCORDANCE WITH
CONTRACT NAS5-21705

SD 73-SA-0018-3

PART II FINAL REPORT

TRACKING & DATA RELAY SATELLITE SYSTEM CONFIGURATION & TRADEOFF STUDY

VOLUME III

PART II SPACECRAFT DESIGN

Tom Hill

T. E. Hill
TDRS STUDY MANAGER

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FOREWORD

This report summarizes the results of the study conducted under Contract NAS5-21705, Tracking and Data Relay Satellite Configuration and Systems Trade-Off Study--3-Axis Stabilized Configuration. The study was conducted by the Space Division of North American Rockwell Corporation for the Goddard Space Flight Center of the National Aeronautics and Space Administration.

To ensure that the NASA would receive the most comprehensive and creative treatment of the problems associated with the definition of an optimum TDRS system concept, North American Rockwell entered into subcontracting agreements with the AIL Division of Cutler-Hammer and the Advanced Systems Analysis office of Magnavox. In this teaming relationship NR performed as the prime contractor with responsibility for study management, overall system engineering, TDR spacecraft and subsystem design, network operations and control, reliability engineering, and cost estimating. AIL was responsible for RF link analysis, the on-board telecommunications subsystem design and ground station RF equipment design. Magnavox was responsible for telecommunications system analysis, user spacecraft terminal design, and ground station signal processing.

The study was in two parts. Part I of the study considered all elements of the TDRS system but emphasized the design of a 3-axis stabilized satellite and a telecommunications system optimized for support of low and medium data rate user spacecraft constrained to be launched on a Delta 2914. Part II emphasized upgrading the spacecraft design to provide telecommunications support to low and high, or low, medium and high data rate users, considering launches with the Delta 2914, the Atlas/Centaur, and the Space Shuttle.

The reporting for both parts of the study is as follows:

Part I SD 72-SA-0133	Part II SD 73-SA-0018
1. Part I, Summary (-1)	1. Study Summary (-1)
2. System Engineering (-2)	2. Telecommunications Design (-2)
3. Telecommunications Serivce System (-3)	3. Spacecraft Design (-3)
4. Spacecraft and Subsystem Design (-4)	4. Cost Estimates (-4)
5. User Impact and Ground Station Design (-5)	
6. Cost Estimates (-6)	
7. Telecommunications System Summary (-7)	

This report consists of four volumes: Volume I, Study Summary: Volume II, Part II Telecommunications Design; Volume III, Part II Spacecraft Design, and Volume IV, Study Cost Analysis; This volume summarizes the activities and results of both Part I and Part II, as does Volume IV. The detailed technical material developed during Part I was extensively reported in the Part I Final Report and is not repeated in this report. Volumes II and III are technical reports covering only Part II. The reader is referred to the Part I Final Reports for detailed considerations of mission analysis, network operations and control, telecommunications system analysis, telecommunications subsystem design (baseline), spacecraft mechanical and structural design (baseline), spacecraft subsystem design and analysis, reliability, user spacecraft impact (baseline), and ground station design: except as they were influenced by Part II design and analysis activities.

Acknowledgement is given to the following individuals for their participation in and contributions to the conduct of this study:

North American Rockwell

M. A. Cantor	System Engineering and Spacecraft Design--Project Engineer
A. A. Nussberger	Electrical Power
W. C. Schmill	Electrical Power
R. E. Oglevie	Stabilization and Control
A. F. Boyd	Stabilization and Control
R. N. Yee	Propulsion
A. D. Nusenow	Thermal Control
T. F. Rudiger	Flight Mechanics
J. W. Collins	Satellite Design
P. H. Dirnbach	Reliability
W. F. Deutsch	Telecommunications Design
S. H. Turkel	Operations Analysis and Cost
A. Forster	Cost
A. F. Anderson	Integration

AIL-Division of Cutler-Hammer

T. T. Noji	Telecommunications Design
L. Swartz	Telecommunications Design

The Magnavox Company

D. M. DeVito	Telecommunication System Analysis
D. Cartier	Ground Station Design
R. H. French	Operations Analysis
G. Shaushanian	User Transponder Design

In addition we acknowledge significant support and contributions from Mr. Peter Sielman of AIL and Dr. Neil Birch of Magnavox.

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1.0 INTRODUCTION

This volume documents the results of the Part II spacecraft design studies of the TDRSS configuration and tradeoff studies. During Part II, primary emphasis was placed on Phase I, the design of an uprated TDRS to support low, medium, and high data rate users but still within the constraints of a launch on a Delta 2914, as was the baseline spacecraft designed in Part I. In Phase II of Part II, a spacecraft launched on an Atlas/Centaur and multiple launches of the TDRS from the Shuttle using an Agena Tug were investigated.

The uprated TDRS for the Delta 2914 launch required no changes in operations or system engineering from the baseline (Part I) system. The spacecraft design was affected only by replacing the 2 m fixed Ku-/S-band antennas with 3.8 m furlable antennas and relocating the TDRS/GS antenna from the centerline to the solar panel strut and increasing its size from 0.9 m to 1.8 m. The solar panels were increased .186 m^2 (2 ft^2) to minimize operating restrictions at end of life and the skin thickness reduced from 0.008 to 0.006 and the glue from 0.008 to 0.005. Other than the telecommunications subsystem, the only changes in equipment were the removal of two amp-hour meters from the electrical power system and 16 explosive valves (one at each thruster) in the propulsion system to minimize the decrease in weight contingency. Incorporating these design changes results in a weight contingency of 26.2 kg (57.5 lb).

The telecommunications subsystem was configured to have essentially the same reliability for the uprated design as the baseline; therefore, spacecraft and system reliability does not change. In addition, the telecommunication components using high power were mounted directly on the louver panels to maximize heat rejection and eliminate potential hot spots.

The increased capability of the Atlas/Centaur permits launch of a spacecraft with additional communications capability. Five 3.8 m (12.5 ft) furlable antennas, larger solar panels, more batteries, and a "Senior AGIPA" LDR antenna were put on the spacecraft. Four of the large antennas serve four MDR users (or three MDRU and 1 HDRU) and one transmits/receives to/from the ground. This version has ample weight contingency.

Studies for a Shuttle launch considered launching three spacecraft on one launch with an Agena tug. Two versions were studied. A minimum cost version packaged three uprated baseline spacecraft with the Agena placing them in synchronous orbit (no apogee kick motor). The weight contingency on each spacecraft is 178 kg (393 lb). A high capability version considered three of the five-antenna spacecraft placed in orbit by the Agena tug. The weight contingency was reduced to 50.6 kg (111 lb) for each spacecraft. Use of an apogee kick stage will increase the allowable weight of each spacecraft by approximately 230 kg. This was not needed by the selected designs (which have adequate communications capability) and the kick stage would complicate spacecraft operations and increase cost.

2.0 DELTA 2914 UPRATED SPACECRAFT DESIGN

The first and primary task of Part II was to uprate the Delta-launched TDRS design to support LDR, MDR, and HDR users. The basic support requirements were changed to eliminate LDR voice and to include S-band voice for Shuttle and support of HDR users.

These changes required modifications to the telecommunications system including the antennas, the electrical power requirements, and the spacecraft weights. Telecommunication system changes are described in Volume II, and the resulting power and weight values are summarized in Table 2-1.

2.1 SYSTEM ENGINEERING

Uprating the TDRS has no effect on the mission analysis or network operations and control as defined in Volume II of the Part I final report for the baseline spacecraft, and that analysis is applicable to the uprated vehicle. Additional analyses were conducted to establish the relationship between antenna elevation angle, station location and satellite inclinations, and prediction of solar outages.

2.1.1 Ground Elevation Angle versus Station Latitude

The effect of latitude on ground station elevation angle was computed. This is shown in Figure 2-1 for TDRS orbit inclinations of 0, 2.5, and 7.5 degrees. Although the base system assumes the ground station to be at GSFC a tradeoff can be made between desired ground elevation angle as shown in this figure and increased TDRS spacing (as shown in Figure 2-4 of Report SD 72-SA-0133-2). The figure shows that for a station at the latitude of White Sands, the elevation angle is approximately one degree higher than at Rosman (11 degrees versus 10 degrees). At Houston, it increases another 0.5 degree to 11.5 degrees. At Goddard's latitude the elevation is approximately 1.5 degree less than Rosman.

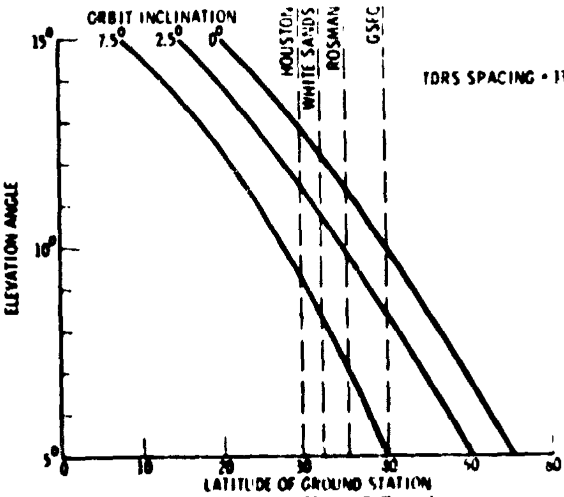

Figure 2-1. Ground Elevation Angle Versus Latitude

Table 2-1. Telecommunication Weight and Power Summary

		Weight		Power (Watts)	
		kg	lb	Maximum	Average
LDR					
Receiver		4.1	9.0	9.1	9.1
Transmitter	(30 db EIRP)	2.5	5.5	*	116.0**
	(27 db EIRP)				61.0**
Antenna	(including support arms)	14.4	31.7		
MDR/HDR #1					
Receiver	(S and Ku)	4.5	9.9	8.2	8.2
Transmitter	S-Data	6.4	14.1	15.0	15.0
	S-Voice/Data			66.0	66.0
	Ku-Data			5.1	5.1
	Ku-Video			35.8	35.8
Antenna	(including support strut)	17.9	39.4	24.0	4.0
MDR/HDR #2					
Receiver	(S and Ku)	4.5	9.9	8.2	8.2
Transmitter	S-Data	6.4	14.1	15.0	15.0
	Ku-Data			5.0	5.0
(Will not transmit voice or video at same time and #1)					
Antenna	(including support arms)	17.9	39.4	24.0	4.0
TDRS/GS					
Receiver		2.2	4.8	5.3	5.3
Transmitter		9.6	21.1		
MDR	(17.5 dB margin)			6.0	6.0
HDR	(7.5 dB margin)			20.8	20.8
HDR	(17.5 dB margin)			50.6	50.6
Antenna		7.5	16.5	9.5	--
Frequency Source		3.5	7.7	8.0	8.0
TT&C					
Processor		4.4	9.7	10.0	10.0
Transceiver		1.8	4.0	13.5/4.5	0.5
S-Band Tracking/Order Wire		2.6	5.7	7.9	2.0
DC Cabling		6.0	13.2		
RF Cabling and Waveguide		6.0	13.2		
Total		123.6	272.5		

* Peak power of 205 watts or 403 watts can be put through this system for up to 2 to 5 minute emergencies, when user is in severe RFI environment.

** Includes 10 percent for power conditioning

2.1.2 Solar Outage

The "solar outage" which is created when the sun, TDRS, and ground station tracking cone are in line, also was determined. This outage occurs twice a year. For an 18.3 m (60 ft) ground antenna the cone angle is approximately 0.10 degree. This produces an outage of approximately 2.5 minutes on one day twice a year. For a TDRS spacing of 130 degrees (65 degrees relative longitude from ground station) this occurs 15 days before and after each equinox for Rosman, and 13 days for White Sands. It occurs at a local time of approximately 7.3 hours for the east satellite and 16.7 hours for the west satellite.

2.2 SPACECRAFT DESIGN CONCEPTS

Three concepts were investigated in Part II for the Delta 2914 launch. The first concept, which was looked at in greatest detail, was an uprated version of the Part I baseline concept but with the capability to support high data rate (HDR) users as well as medium and low data rate users. The two other concepts were investigated to a much lower level of detail. These are an austere version of the uprated concept with no LDR service and a version to service MDR/HDR users through two high performance antennas with the low and medium data rate users requiring multiple access serviced by an S-band array. These configurations are described in the following sections.

2.2.1 Uprated TDRS Configuration

2.2.1.1 Spacecraft Design

The baseline TDRS configuration generated in Part I was reconfigured by replacing the MDR 1.98 m diameter dishes by 3.8 m diameter furlable dishes to permit high data rate (HDR) operation. To facilitate the increased TDRS/GS link requirements the TDRS/GS antenna was increased from .9 m diameter to 1.8 m and relocated from the spacecraft centerline to one of the solar panel support struts. With the larger antennas, the solar shadow lines from the dish required moving the solar array panels further outboard. The LDR UHF/VHF array remains unchanged but the LDR transmit mode of operation was changed to FFOV from steered beam which necessitated the electronic changes described in Volume II.

Figure 2-2 illustrates the uprated configuration, and Figure 2-3 shows the overall changes between the baseline and uprated concepts.

To incorporate the support requirements of the HDR users, the size of the parabolic reflector antennas were increased from 1.98 m to 3.8 m diameter. The receivers and transmitters were uprated to allow simultaneous operation on both Ku- and S-band.

The change in antenna diameter had the greatest impact on the spacecraft configuration. The 1.98 m diamater solid reflector antennas used in Part I were the largest solid-face reflector possible to package in the Delta 2914 shroud. The requirement for HDR capability and the higher Shuttle requirements in the uprated TDRS necessitated using furlable or folding antenna reflectors to provide the full 3.8 m diameter.

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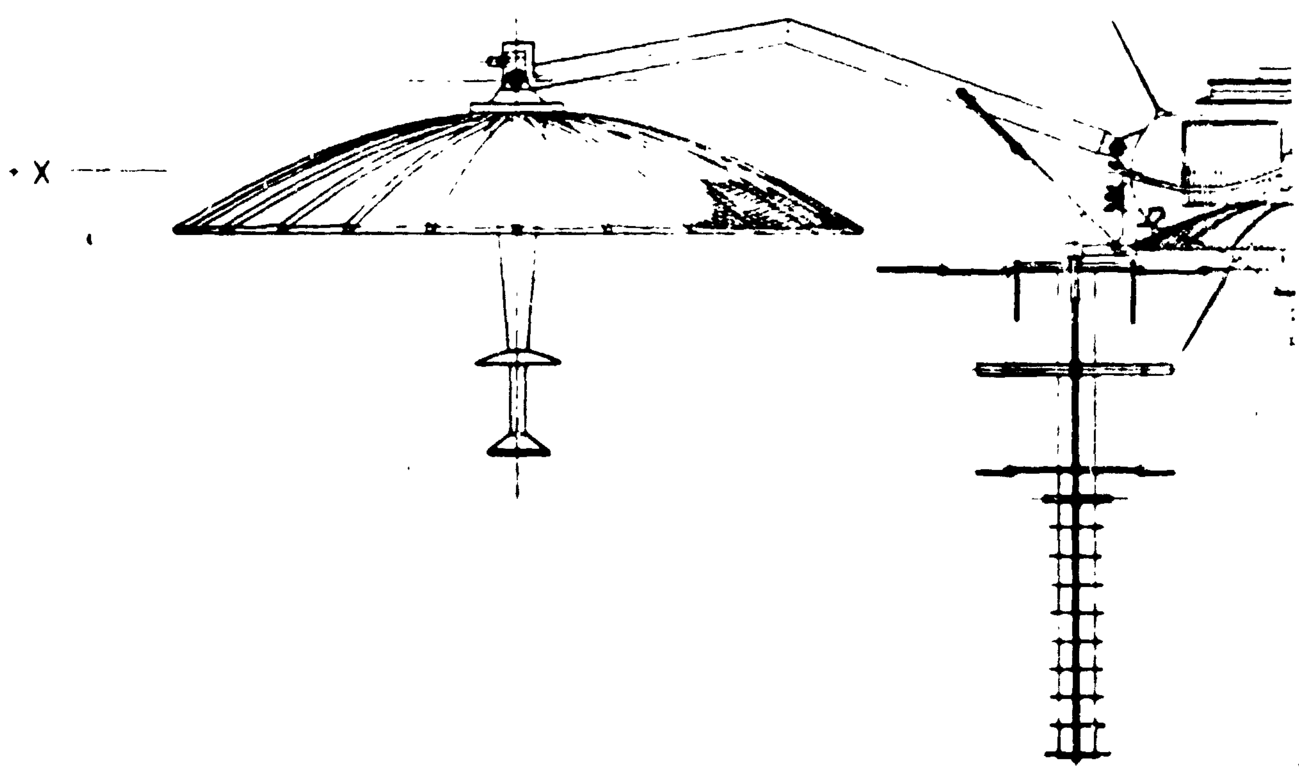

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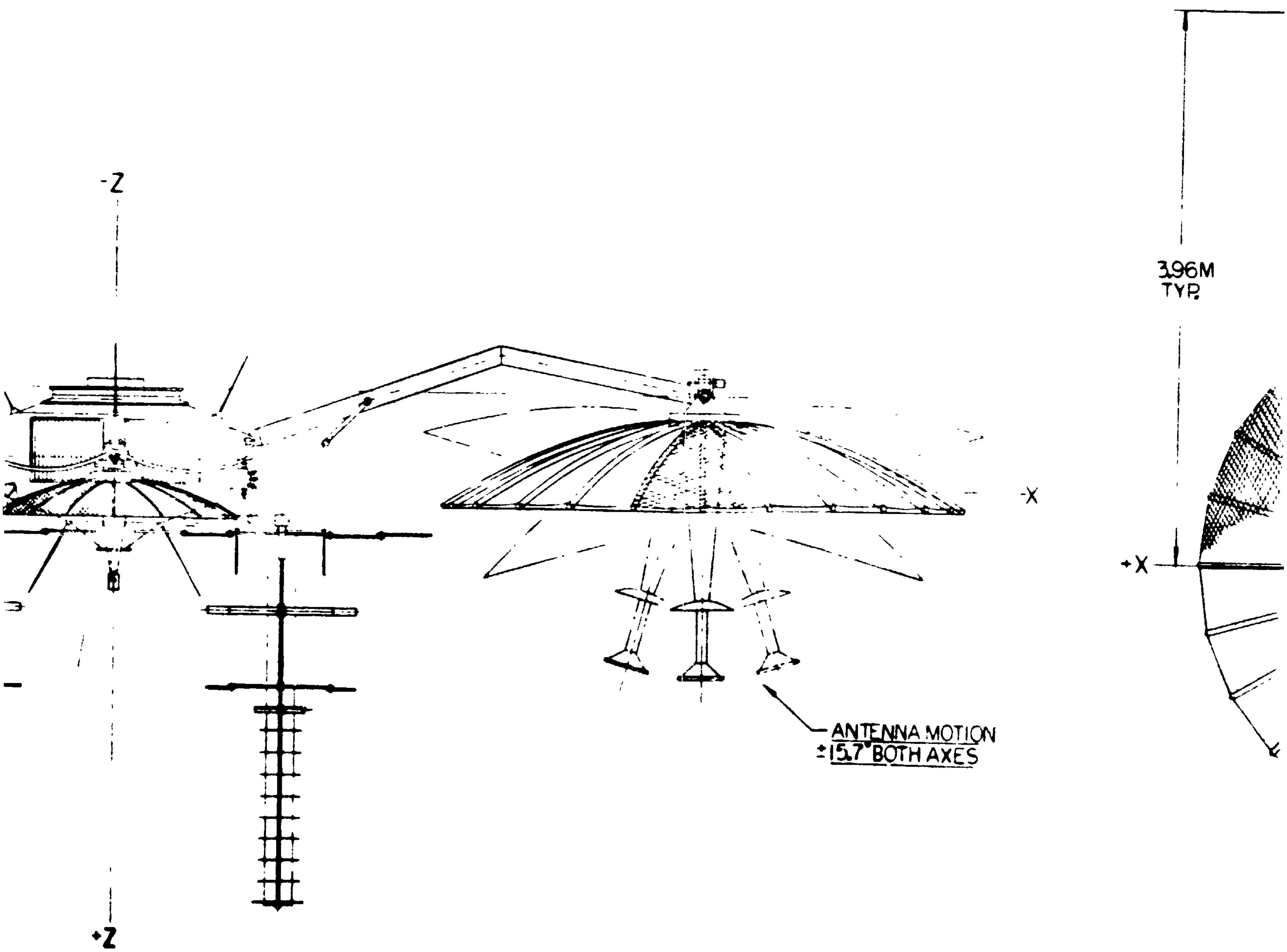

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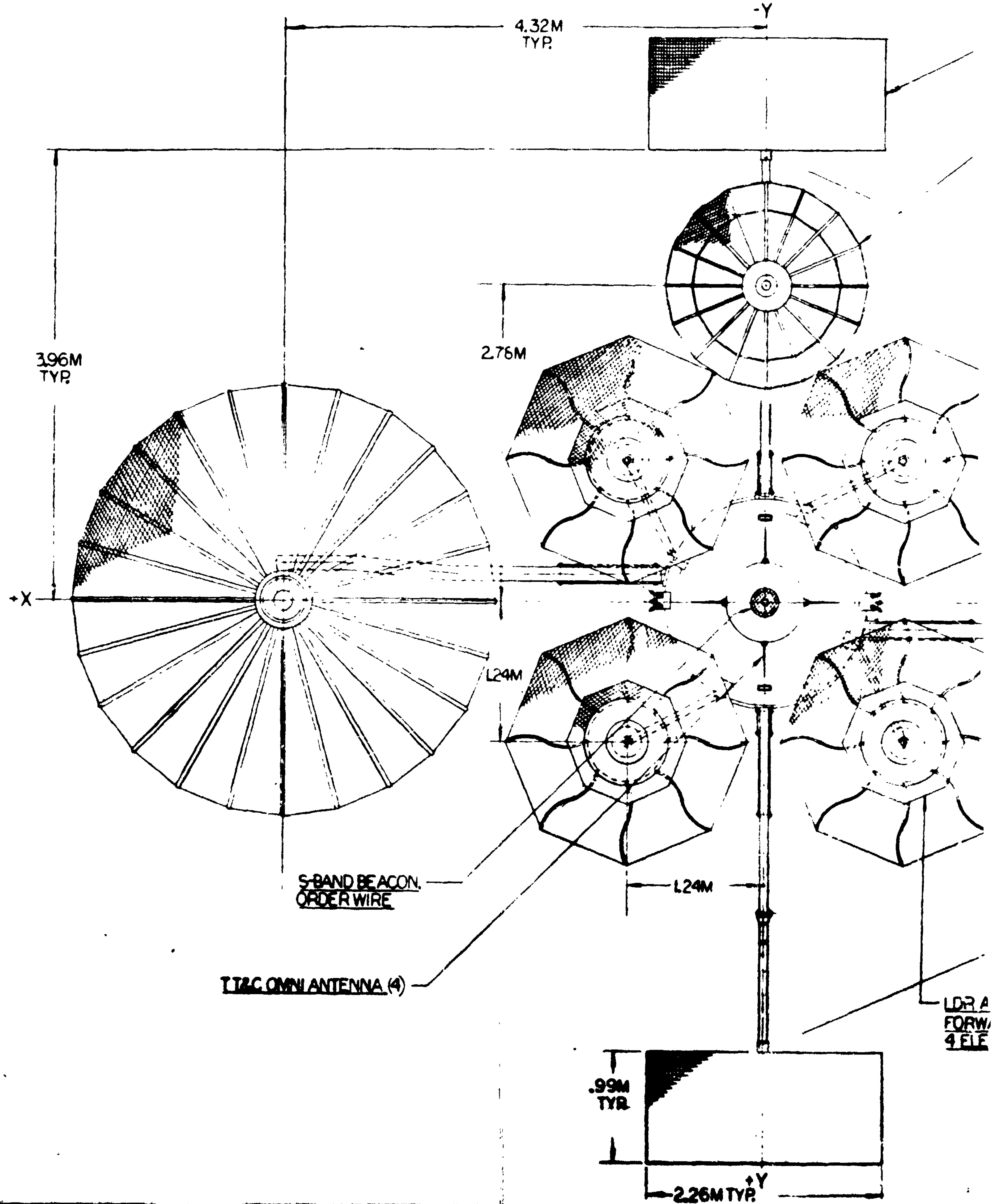

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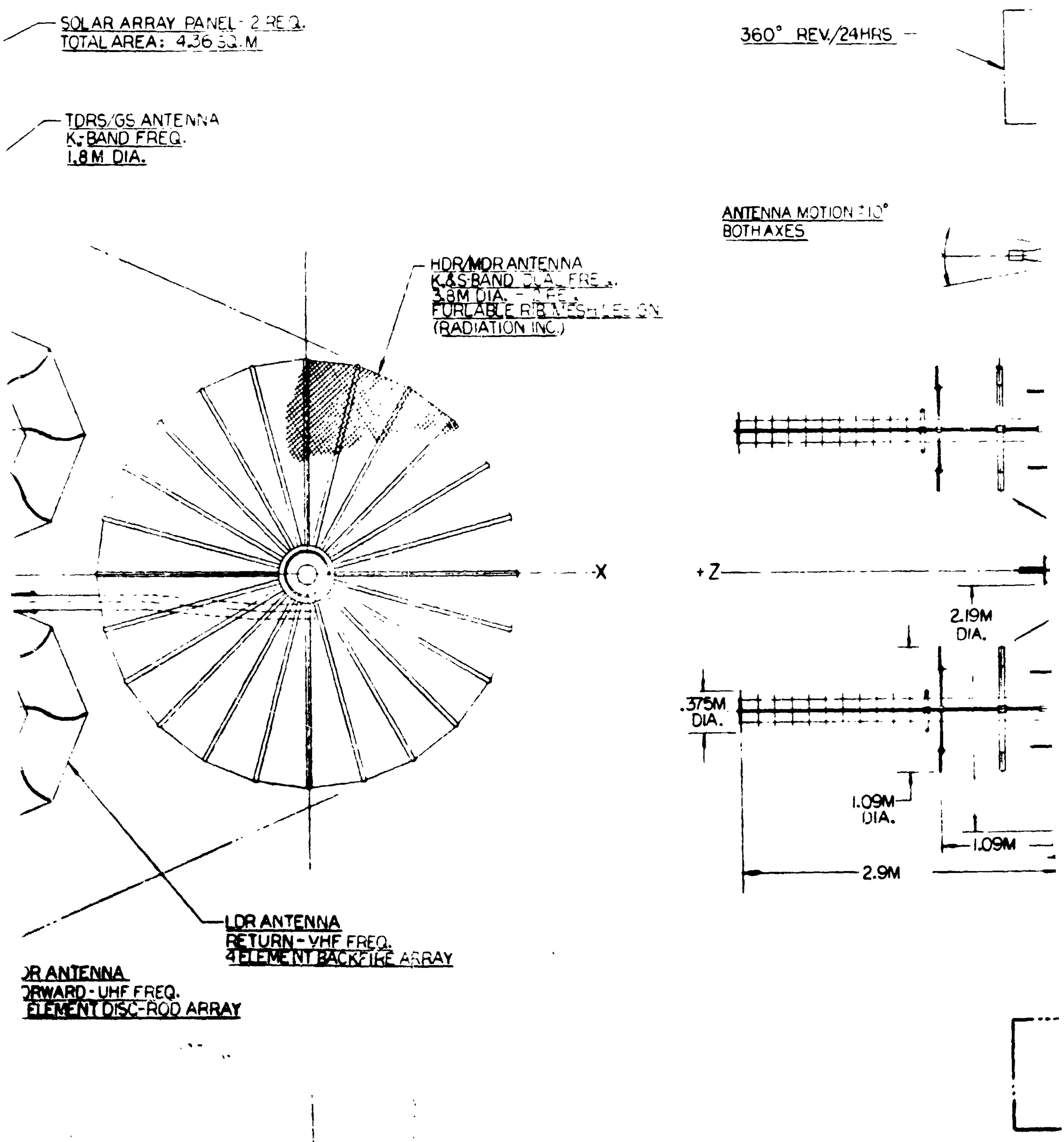


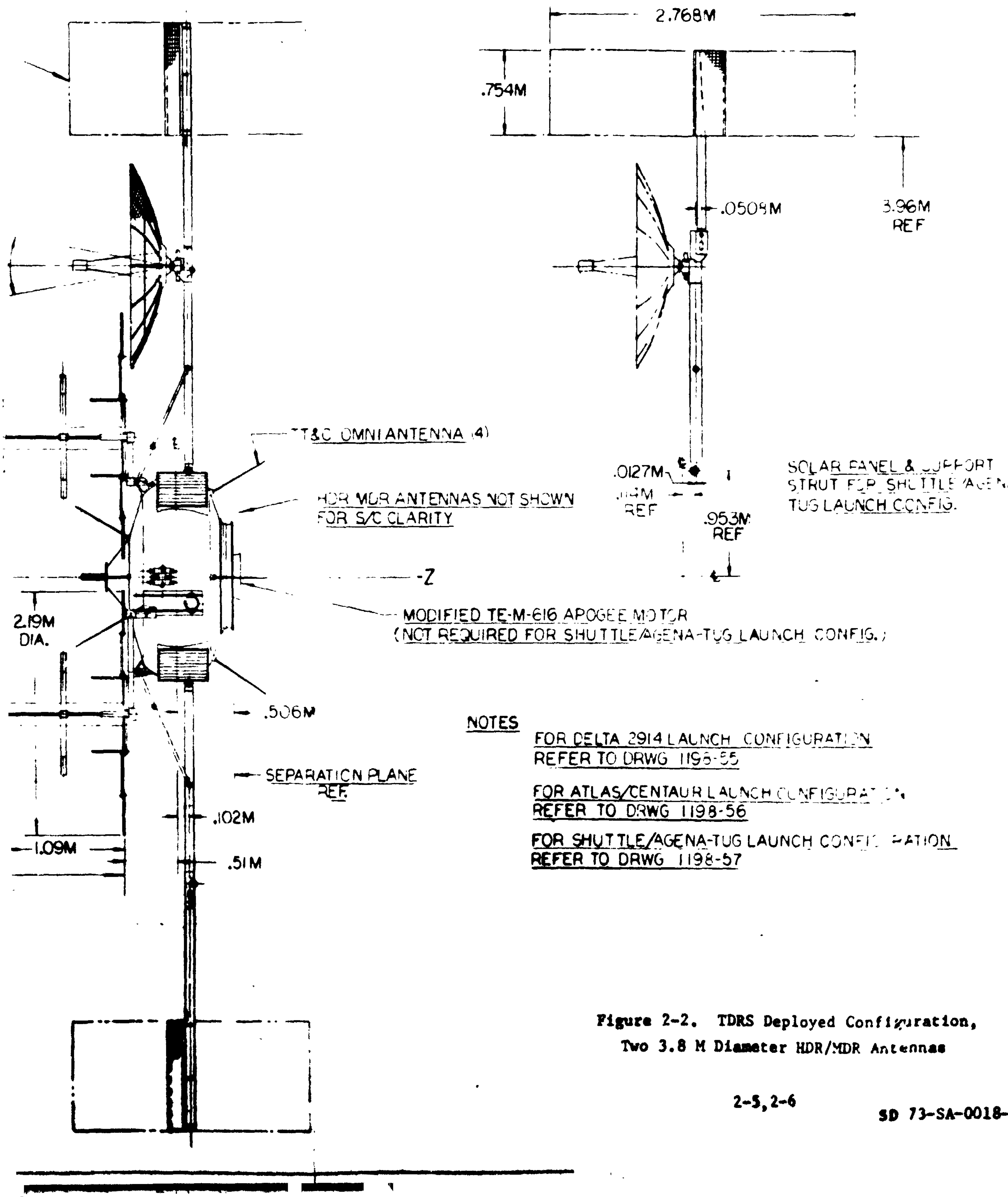
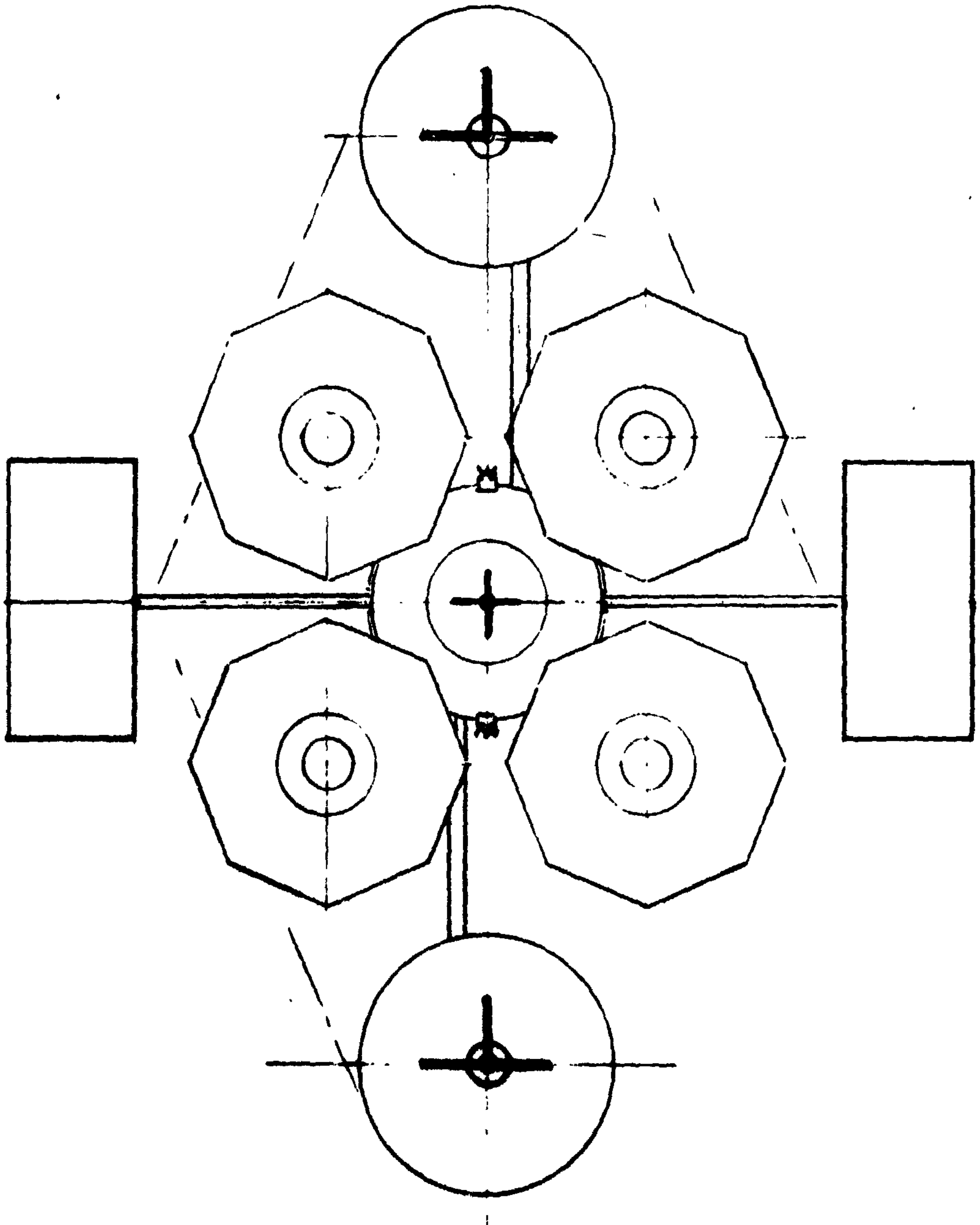


Figure 2-2. TDRS Deployed Configuration, Two 3.8 M Diameter HDR/MDR Antennas



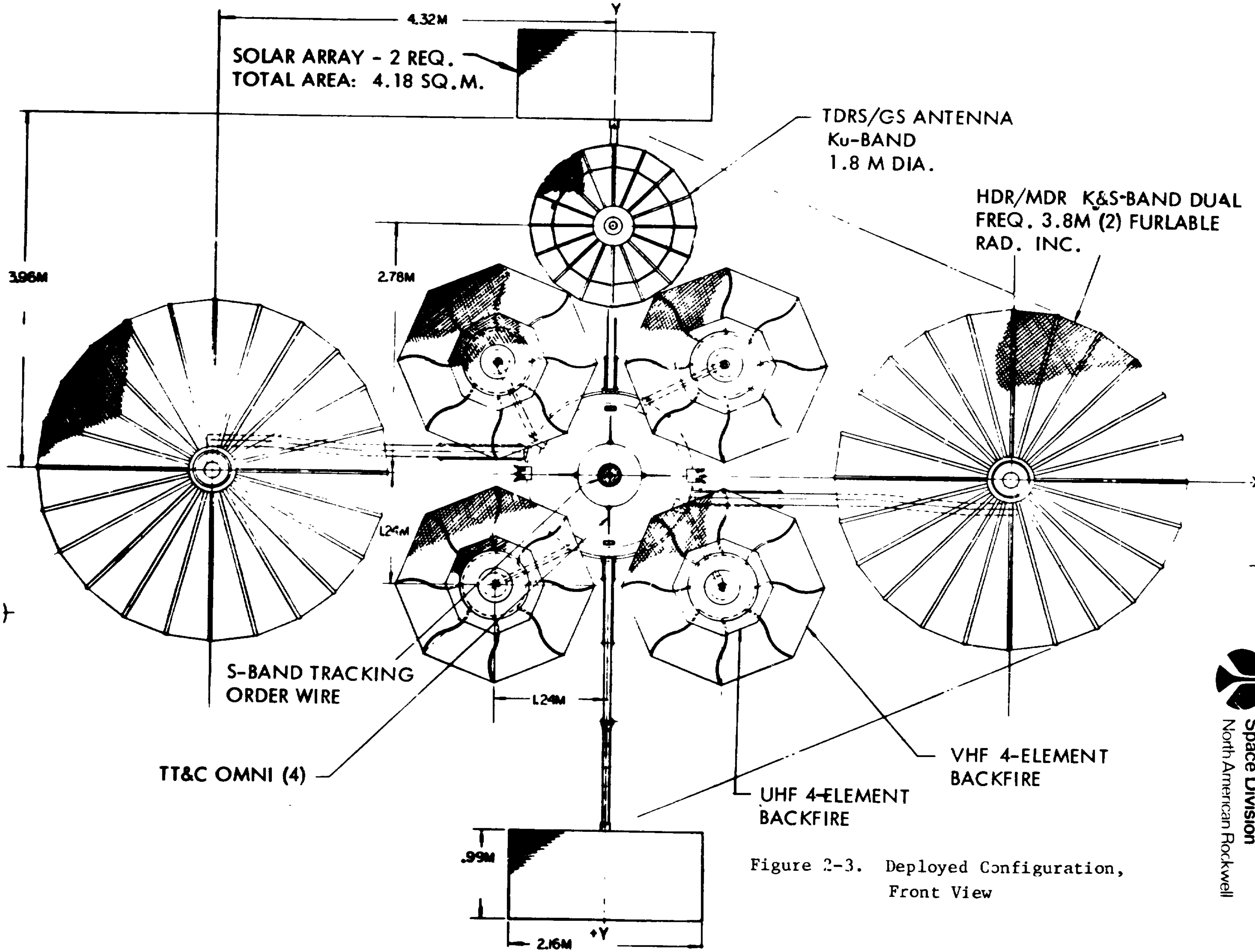


Figure 2-3. Deployed Configuration, Front View

The uprated TDRS is packaged for launch in the shroud fairing of the Delta 2914 as shown in Figure 2-4.

The LDR UHF and VHF array elements are compressed and retracted to their positions ahead of the spacecraft body. The HDR/MDR antennas are furled into their package configuration, and their support struts are swung forward around and in front of the LDR elements.

The TDRS/GS mesh antenna is rotated forward with the lower strut of the solar panel strut system as it folds above and below the spacecraft body so that the TDRS/GS antenna is positioned above the furled HDR/MDR antennas with its feed support cone extending between them.

As the solar panel struts are folded forward and toward the spacecraft body, the solar array panels fold down and around the body behind the LDR elements leaving the gap between panels on the sides for clearance with the HDR/MDR antenna support struts and clearance for operation of the attitude stabilization and control thrusters prior to deployment of the solar panels and antennas.

Launch locks and latches restrain and position the various structures in relation to clearance with the Delta fairing and to withstand launch environment loads and vibration.

The rear of the spacecraft inner cone surrounding the apogee motor has a machined ring surface at its rear face for attachment and matching with the 37-31A attach fitting on the Delta third stage motor.

With the allowable spacecraft payload on the Delta 2914 plus the CTS apogee motor of 334.8 kg, the TDRS spacecraft weight of 308.5 kg provides a contingency of 26.3 kg. This is approximately 10 percent of the dry weight of the spacecraft.

After the spacecraft reaches synchronous orbit and becomes three-axis-stabilized, ground commands initiate the release of the solenoid-operated latches of the solar panel strut system. The spring-loaded struts extend and lock into their deployed positions. The solar panel halves rotate forward around their hinge lines with spring loaded hinges to assume a flatter shape for increased efficiency.

The latches for the HDR/MDR antenna support struts are similarly activated and the support struts rotated back on each side of the spacecraft with spring-loaded joints until they lock in the deployed position. The tips of the furled antennas are released from their latches to the support struts and the antenna gimbals drive the furled antennas to their neutral forward pointing position. The tension cable restraining the ribs is severed by the guillotine cutters on command and the ribs deployed back to their stops by the antenna deployment mechanism in the antenna hub. This deployment takes approximately 90 seconds.

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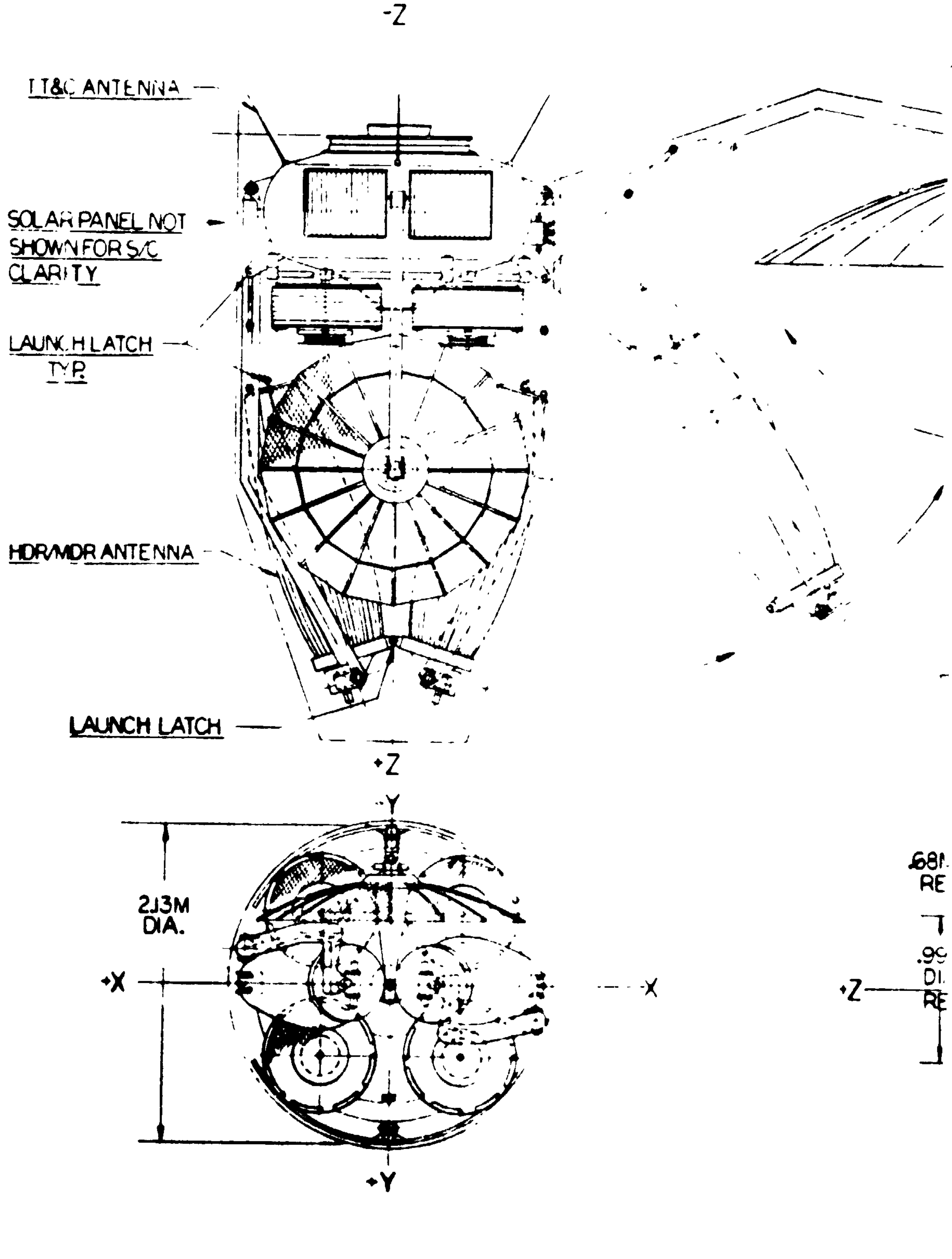

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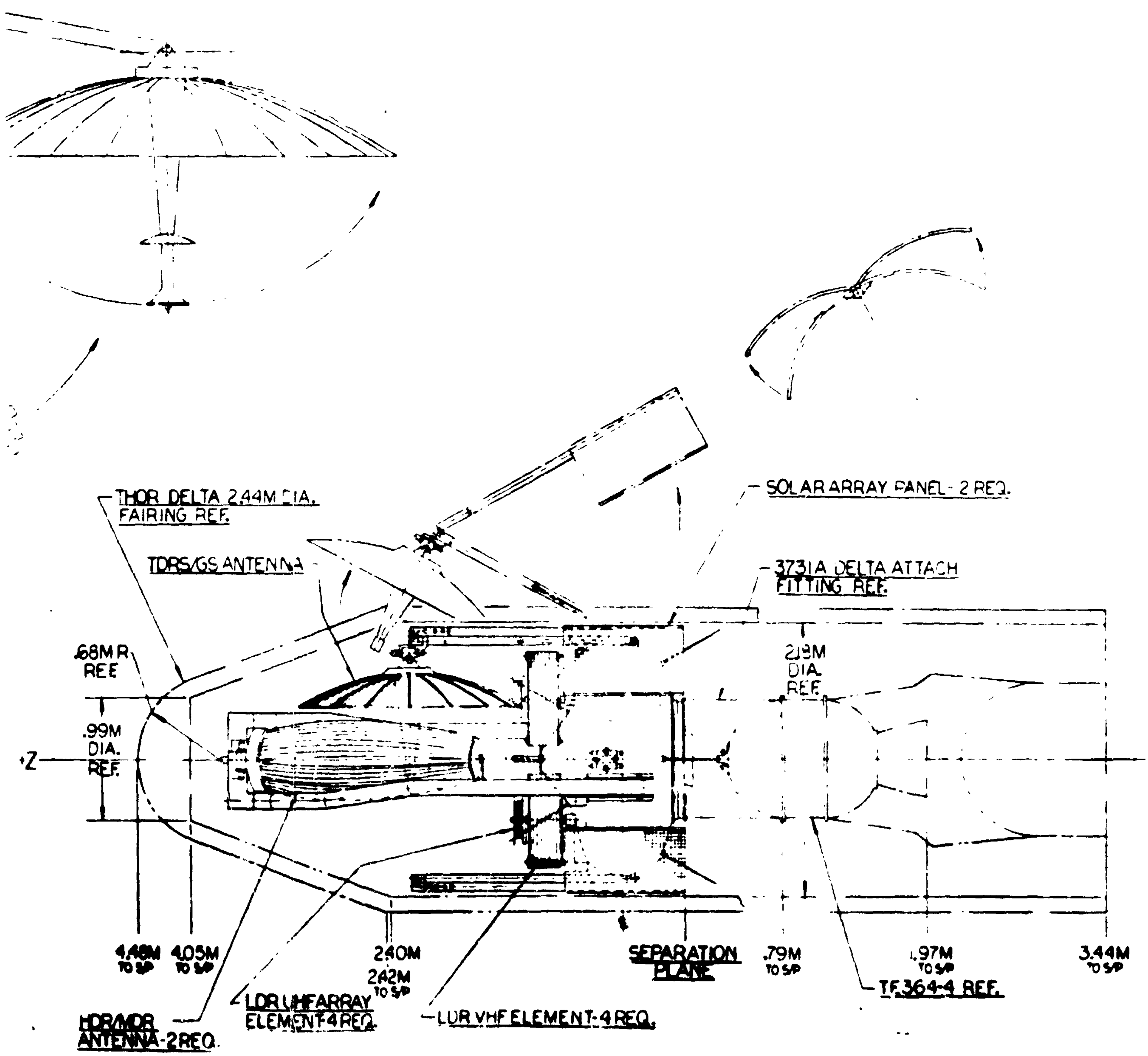

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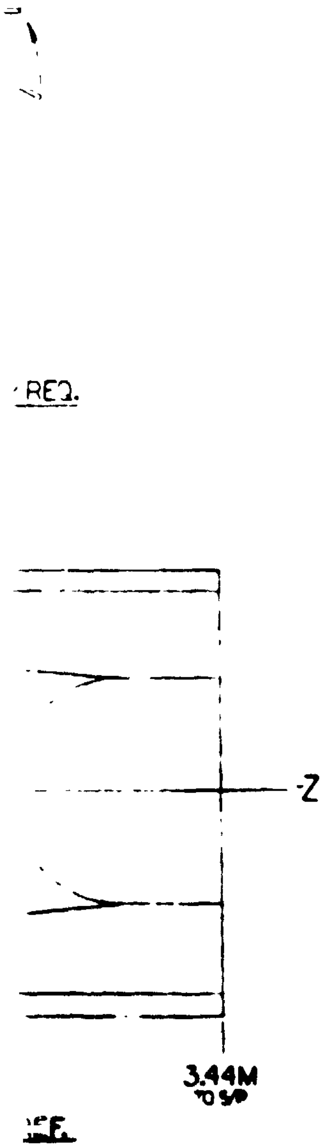


NOTES

FOR DEPLOYED CONFIGURATION
REFER TO DRWG 1198-54

Figure 2-4. TDRS Configuration, Delta 2914 Launch, Two 3.8 M Diameter HDR/MDR Antennas

While the HDR/MDR antennas are being deployed, the LDR antenna system is deployed by first initiating the solenoid release of the latches on the support arms of the elements. As the spring-loaded arms swing out to a position close to their final locked position, a cable-operated release system is activated by the arms and the ground plane extension arms on the element are released to extend the mesh ground planes to their full diameters. The electrically powered actuator in the STEM unit at the hub of the element is activated and extends the center STEM forward to the full length of the deployed element, carrying the UHF discs and dipole element forward in equally spaced increments with a thin dacron cord system.

The TDRS/GS antenna is automatically deployed into its neutral, or forward looking position, by the actuation of the solar panel strut system since it is secured to the top of the inner solar panel strut.

The solar panel drive actuators are energized to rotate the panels to acquire the sun and to remain in rotation to maintain the panels normal to the sun line. The TDRS/GS antenna is aligned to the proper coordinates to acquire the ground station antenna and the TDRS achieves its operational status.

All joints in the deployment of the antennas and solar panels are designed to avoid undue impact loading of the spacecraft during deployment. Shock absorbing devices within the joint latch designs are included to reduce the rate of closure while still assuring positive lock at all joints. Proper choice of lubrication and bearing design is made to assure reliability of deployment devices in the space environment.

Increasing the spacing between solar panels to clear the solar shadow line of the larger HDR/MDR antennas increased the length of the solar panel support struts, and the location of the TDRS/GS antenna on one of the struts increased the size and weight of the strut system.

The solar panel area was increased from 4.18 square meters to 4.36 square meters to minimize operating restrictions at end of life. To accommodate the increased solar array area, the substrate design was changed to a lightweight construction to minimize weight increase.

The solar cells and covers increased by 1.04 kg and the substrate reduced in weight 1.85 kg by incorporating lighter weight construction. The support strut system weight increased 1.95 kg because of the extended length and the mounting of the TDRS/GS antenna on one of the struts. Weights of solar array system components are shown in Table 2-2.

Table 2-2. Solar Array System Weight

	Kg
Solar cells, covers, wiring, etc.	9.21
Substrate	7.48
Solar array drive system (2)	6.80
Support struts and linkage	4.22
Total	27.71

In the HDR/MDR system, the weights of the receiver and transmitter electronics were incresed by 1.0 kg and 1.29 kg, respectively, over the Part I system to include simultaneous operation of Ku- and S-band with each antenna. Each lightweight mesh/rib furlable antenna is 2.47 kg heavier than the smaller solid-face antenna. Table 2-3 summarizes the MDR/HDR telecommunications weights.

Table 2-3. HDR/MDR Weights

		Kg
No. 1 receiver		4.5
No. 2 receiver		4.5
No. 1 transmitter		6.4
No. 2 transmitter		6.4
Antenna (2) (2 x 17.9)		35.8
Reflector	7.09	
S-band feed	1.04	
Ku-band feed	.95	
Gimbal	2.26	
Control/electronics	2.26	
Rotary joints	.95	
Support strut	3.35	
Total	17.9	
		57.6

Incorporating the HDR capability in the TDRS/GS link resulted in an increase in the .9 m TDRS/GS antenna to 1.8 m diameter and the TDRS/GS transmitter changed from solid state to a redundant two-channel TWT design. The larger 1.8 m diameter antenna could not be mounted in the same position as the smaller Part I antenna support struts of the stowed HDR/MDR antennas and blocked the field of view of the horizon sensors. A position on the upper solar panel support link provided the best support to the antenna and permits packaging with the stowed LDR elements and the HDR/MDR antenna without requiring furling of the TDRS/GS antenna.

The antenna is a rib-mesh design developed by Radiation Systems Division of Radiation, Inc., Melbourne, Florida, similar to their furlable design except that the ribs remained fixed, providing a lightweight nonfurlable antenna.

The rib-mesh antenna is formed by 16 rigid ribs of 0.50 inch diameter thin wall aluminum tubes which support and contour the elastic mesh surfaces. To obtain the required surface tolerance a double mesh design is employed. (This construction is similar to that described in Section 2.2.1.2.) To provide thermal control, each rib is wrapped with multilayer (superinsulation) blankets. The ribs are supported in a rigid box hub structure which also serves as the interface flange and mounts to the antenna gimbal. The gimbal unit is flange-mounted to the front of the inner solar panel support link strut just below the strut folding joint.

The antenna is offset from the spacecraft centerline, unbalancing the symmetry of the spacecraft from both a mass and solar pressure viewpoint. However, the lightweight, high-porosity rib-mesh design minimizes the unbalance compared to a solid-face reflector of similar size. The solar pressure may be balanced by slightly reducing the porosity of the mesh surfaces of the large ground planes of the two LDR VHF array elements on the side of the spacecraft opposite the TDRS/GS antenna to match the reflectivity of the TDRS/GS antenna mesh.

The mass unbalance in both the deployed and stowed configuration by the offset of the TDRS/GS antenna is balanced by reversing the location of one of the HDR/MDR antenna support struts to the side opposite the TDRS/GS, thereby locating the mass of both HDR/MDR support struts on the side away from the TDRS/GS antenna. The further balancing required for stowed configuration is achieved by locating both the gimbal control/electronics modules (4.52 kg) at the gimbals axes. The weight of the transmitter was increased 5.08 kg for two channels with the TWT, and 1.71 kg to incorporate dual TWT's in each channel. The lightweight mesh/rib antenna is .89 kg heavier than the smaller solid-face antenna used in the Part I baseline design. Table 2-4 shows the TDRS/GS system weights.

Table 2-4. TDRS/GS Weights

		kg
Transmitter		9.6
Receiver		2.2
Antenna		7.5
Reflector	2.44	
Ku-band feed	.95	
Gimbal	1.47	
Control/electronics	2.27	
Rotary joints	.39	
	7.5	
Total		19.3

The transmitter/divider components of the LDR system were redesigned to combine the transmitter and divider network, reducing electronics weight by 1.81 kg (Table 2-5).

Table 2-5. LDR System Weight

		kg
Transmitter (4)		2.5
Receiver (4)		14.1
Antenna (4)		
Element	2.26	
Stem Unit	.91	
Strut	.41	
	3.58	
Total		21.0

In the Tracking/Order Wire System, a separate receiver was installed for the order wire to reflect the difference in frequency from the tracking transponder. The two units use the same antenna. Weights are shown in Table 2-6.

Table 2-6. Tracking and Order Wire System Weight

	kg
Transceiver	2.5
Antenna helix	.1
Total	2.6

The dc wiring, RF cabling and waveguide necessary for the interconnection of the systems and the frequency source increased because of the relocation of the larger antennas and the increase in bands in the frequency source required for the HDR frequencies The RF cabling and waveguide increased in weight by 1.49 kg and the frequency source by .99 kg (Table 2-7).

Table 2-7. Miscellaneous Support System Weight

	kg
DC wiring	6.0
RF cabling and W/G	6.0
Frequency source	3.5
Total	15.5

A summary of the communication system weights is shown in Table 2-8.

Table 2-8. Communication Systems Weight

	kg
HDR/MDR	57.6
TDRS/GS	19.3
LDR	21.0
TT&C	7.6
Tracking and orderwire	2.6
Frequency source	3.5
Cabling W/G and wiring	12.0
Total	123.6

2.2.1.2 Antenna Description

The space-to-space antenna design chosen for suitability, proven hardware development, and lightweight construction was a furlable rib-mesh design developed by Radiation Systems Division of Radiation, Inc., Melbourne, Florida.

The basic design is illustrated in Figure 2-5. A number of rigid, parabolic-shaped ribs are mounted radially about a central hub and deployment mechanism. The RF reflective surface is provided by attaching a low spring rate, elastic, metallic mesh to the ribs. The use of this "soft" mesh with the rigid ribs results in a rib-dominated reflector surface that is relatively unaffected by changing mesh forces and orbital thermal variations throughout the antenna life. The rigid ribs provide sufficient stiffness to the reflector surface to allow meaningful RF range testing of the reflector in a gravity environment.

In the past, the surface accuracy of this type of design was directly proportional to the number of ribs because the largest contribution to surface error was the "reverse bulge" effect of the mesh between the ribs. The general nature of this effect is shown in Figure 2-6. The mesh membrane is pulled tight between the two curved, relatively rigid ribs. Due to the curvature of the ribs, the mesh takes a doubly curved shape, bowing in toward the concave side of the reflector.

This error can be eliminated by the "double mesh" concept shown in Figure 2-7. The concept utilizes a second mesh as a drawing surface for contouring the front reflector mesh. The second mesh is attached to the back of the ribs and is tied to the front mesh by tensioned wires. By properly tensioning these tie wires, the reflector surface can be contoured to a precision parabolic shape. Since the double mesh design approach eliminates the dependence of surface accuracy on the number of ribs, reflector designs can be tailored to meet a wide range of surface tolerance and structural requirements with a light weight. Deployment of the reflector is controlled by a mechanical deployment system (MDS), with redundant mechanical and electrical drive motors. Figure 2-8 shows the stowed reflector layout.

Figures 2-9 and 2-10 illustrate a test model antenna in the deployed configuration, that clearly shows the porosity of the thin mesh membrane that provides the reflective surface for RF operation, and a deployment sequence in a vacuum test chamber as the antenna is deployed from the packaged configuration to the fully deployed position.

The surface accuracy of the reflector is closely controlled by the fabrication and measurement methods employed and the double mesh design.

1. Component of overall RMS surface error due to fabrication and assembly is independent of reflector size with the use of double mesh and adjustable tie wires.

Figure 2-5. Basic Features of Deployable Reflector Design

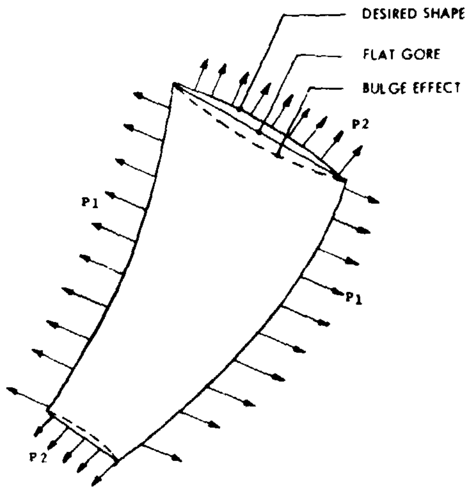


Figure 2-6. Reverse Bulge Effect

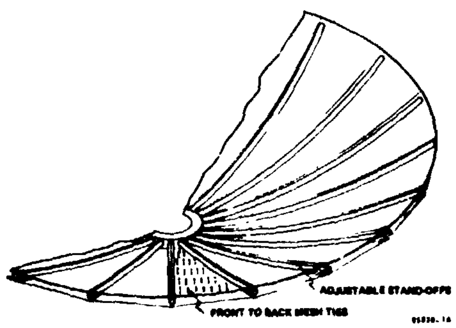


Figure 2-7. Double Mesh Antenna Design

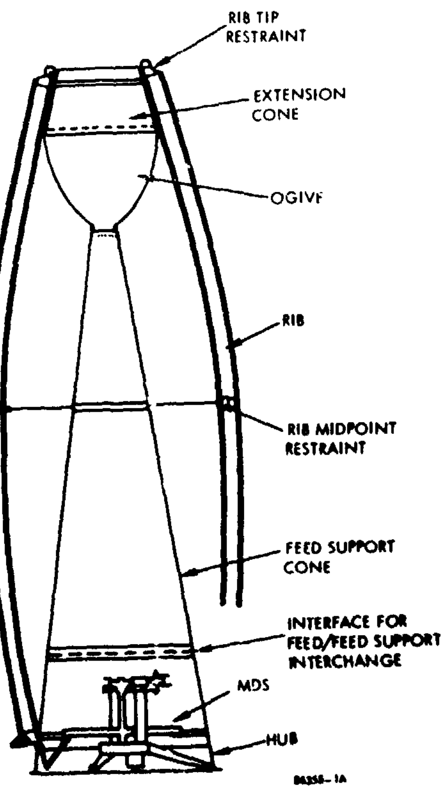


Figure 2-8. Stowed Reflector Layout

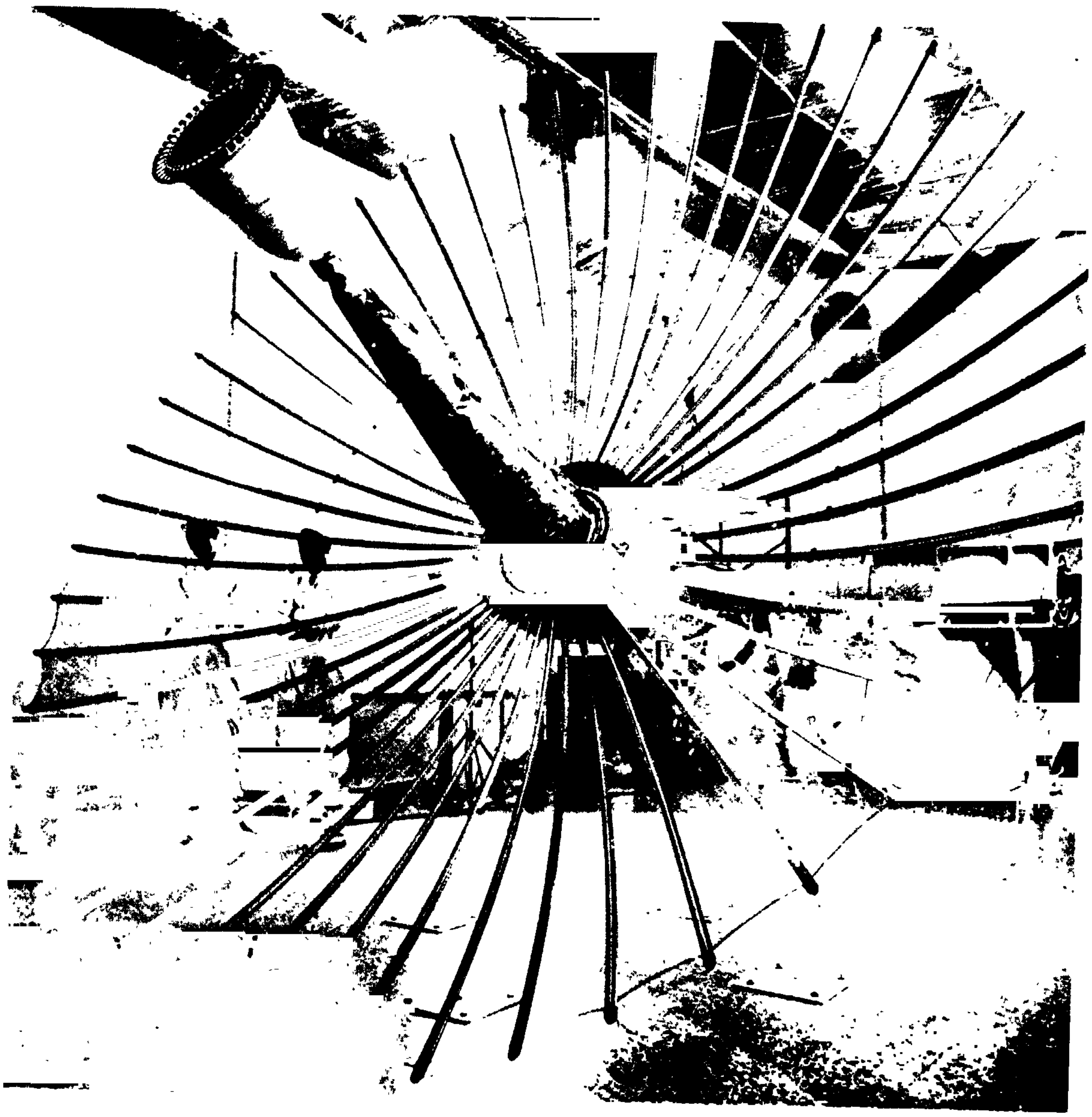


Figure 2-9. Deployed Test Model Antenna

2. Component of total RMS (orbital conditions) due to gravity deflection in a zero gravity environment is minimized by setting the reflector surface in a face-side position at assembly.

3. Component of total RMS error due to thermal effects (on orbit) is minimized by using appropriately plated mesh and thermal control of ribs, feed cone, and hub.

Figure 2-11 indicates surface RMS error and gain loss due to surface RMS error plotted against aperture diameter. For the 3.8 m (12.5 ft) diameter antenna, an RMS surface error of .018 inch can be achieved and at 15 GHz frequency less than .25 db gain loss is experienced.

The TDRS antenna design consists of a 3.8 m diameter, deployable, parabolic reflector with dual feeds, operating at Ku- and S-band frequencies, supported on a central conical structure. This aluminum feed support cone forms the main structure to which all components are attached.

The parabolic reflector is formed by 12 rigid ribs of 1.50-inch diameter thin wall aluminum tubes which support and contour the elastic mesh surface. The number of ribs was based on a tradeoff of surface tolerance and weight. The rib diameter f 1.50 inches was chosen to provide a high deployed stiffness and to allow gravity testing of the reflector in any orientation without external fixturing. Each rib is attached and pivoted about a point which is in-line with the feed cone to eliminate load eccentricity. The mesh is constructed from 7-strand bundles of 0.7 mil Chrome I-R wire knitted into a wire screen. The metallic mesh material selected ensures no degradation of mesh strength properties throughout the 5-year orbital operation. After knitting, the mesh is plated with electroless nickel, gold, and vapor-deposited aluminum. The nickel/gold plating provides the necessary properties for electrical reflectivity, while the outer aluminum plating provides the thermal and environmental control (hardening) necessary for the orbital environment. The finished mesh is a low spring rate, elastic material. The use of this "soft" mesh with the rigid ribs results in a rib-dominated reflector surface which is relatively unaffected by changing mesh forces and orbital thermal variations throughout the antenna life.

Deployment of the reflector from the stowed to the fully deployed position is controlled precisely to eliminate the transfer of any deployment forces to the spacecraft. The controlled deployment also prevents impact loading of the rib structures, assuring the preset parabolic surface is not distorted by the deployment action. The deployment mechanism utilizes redundant energy drive systems to rotate a ball screw within a recirculating ball nut. The resultant linear motion of the ball nut rotates each rib from the stowed to deployed position through the individual linkages to each rib. The primary drive of this system is a constant torque spring motor which also provides a preload on the mechanism in the stowed condition. The 5-inch spring motor provides sufficient energy to deploy the antenna in any orientation under a one-g condition. In a zero gravity condition, the spring motor can provide more than twice the deployment energy requirements. The backup drive system consists of two miniature torque motors driven through 60:1 ratio

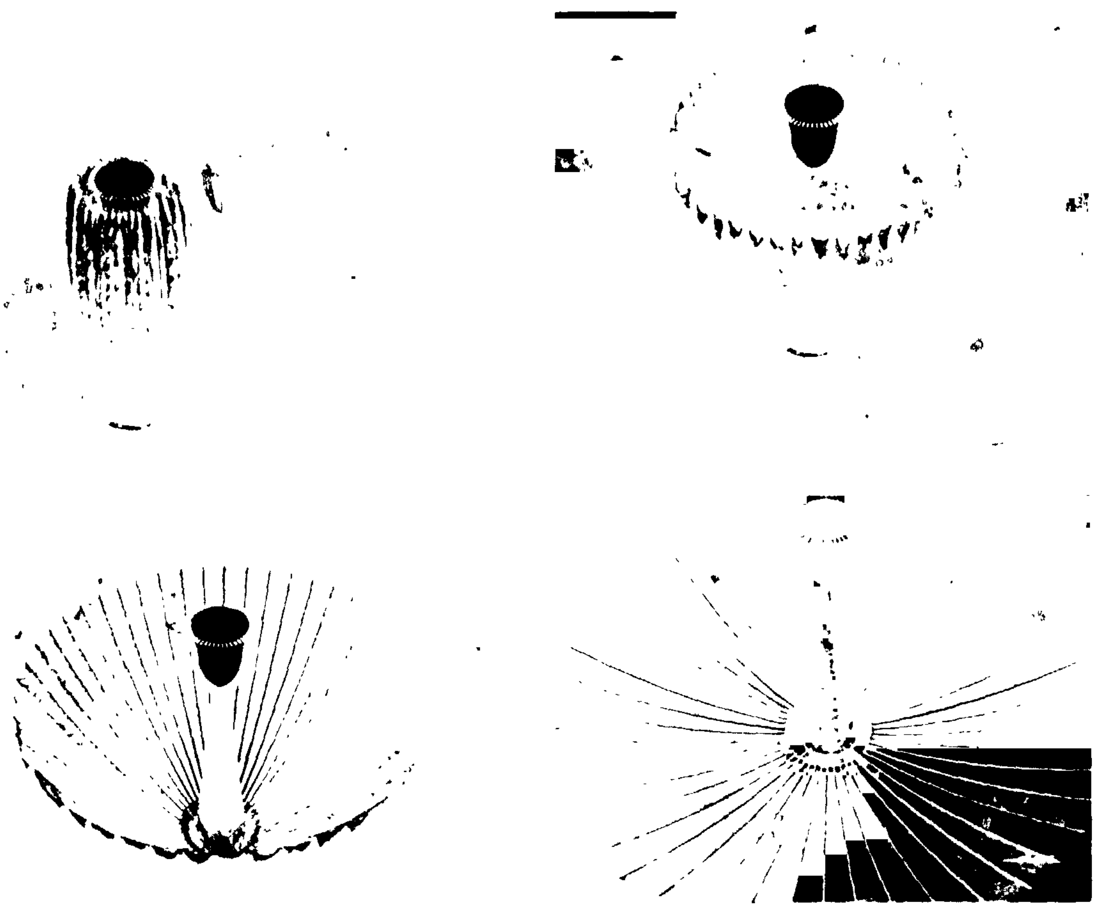


Figure 2-10. Deployment of a Rib-Dominated Rib-and-Mesh Reflector

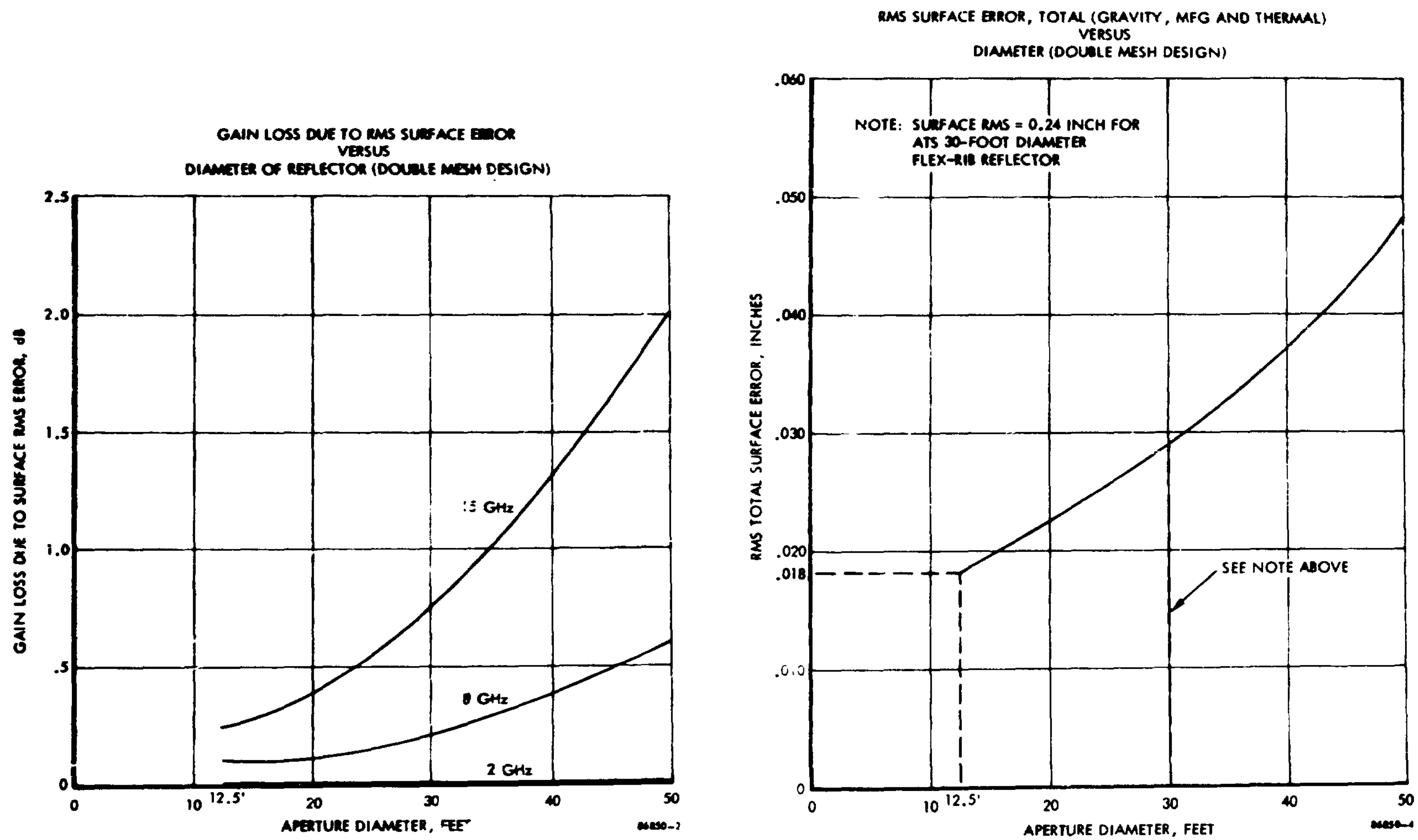


Figure 2-11. RMS Surface Error and Gain Loss Due to Surface RMS Error Plotted Against Aperture Diameter

gear system. The torque motors normally function as dynamic brakes, controlling the deployment rate and requiring no electrical power. If called upon to deliver power (by the deployment control unit), the motors can increase the torque to the ball screw by as much as a factor of four.

Latching in the deployed condition is accomplished by driving the ball nut carrier and linkages through an over-center condition (relative to the pivot arms). In this condition the mesh tension forces, rib loads, spring motor, and pivot arm prelowad, all force the carrier against a mechanical stop. Any external loads which tend to restow the antenna (such as vibration) only serve to further increase the loading of the ball-nut carrier against the mechanical stop. This toggle action eliminates the requirement for further latching devices in the deployed condition (e.g., a mechanical brake or one-way wrap clutch) and improves reliability. A reverse torque of 8 inch-pounds on the ball-screw is required to back drive the mechanism through the latching toggle action. A secondary advantage of the toggle latching is the convenience during ground testing and handling. The antenna can be remotely stowed during ground testing by reversing the current to the electric motors.

All rib and linkage bearings are designed with simple parallel redundancy which greatly reduces the probability of any bearing exhibiting undesirable friction changes. In the event of a high friction condition, the deployment system transfers the full deployment force to the lagging member to overcome the increased friction. All moving, sliding, and rolling joints are lubricated with appropriate solid dry lubrication systems for maximum reliability under worst-case environmental conditions.

The reflector ribs are restrained in the stowed configuration at both the midsection and the tips. This restraint system design forces the stowed antenna to act as a single structural element and results in a high stowed resonant frequency with minimum launch stress. To maintain structural efficiency, the stowed rib tips are restrained by a moment-resisting joint with a preload maintained by a tensioned cable around the rib tips. On deployment command, a redundant set of guillotine cutters severs the cable. The severed cable is instantly cleared from the ribs and held captive by 12 leaf springs located directly above the rib tips. The rib midsections are restrained to the feed support cone by a spar-supported hoop constructed of high modulus ($E = 8 \times 10^6$ lb/in^2) fiberglass reinforced epoxy. In a stowed condition, each rib midsection is restrained to the hoop by a ball end stem mated into a socket on the dielectric hoop. The ball joint is preloaded by deflecting the rib tips into their restraint after the rib midsections contact the dielectric hoop.

Deployment is initiated by cutting the rib tip restraint cable and continues for approximately 90 seconds. Deployment is complete when the ribs come to rest against their stops and the mechanical linkages toggle and preload each rib against its stop by a predetermined amount. This preloading develops a moment resisting joint which effectively eliminates any joint looseness, maintains a high deployed natural frequency, and ensures surface repeatability.

The deployment control unit, located at the base of the feed support cone, sequences and controls the deployment process on receipt of the deployment command from the spacecraft. This unit also provides telemetry to indicate deployment initiation, progress, and completion.

The antenna feed systems are located and supported in the feed cone support structure as illustrated in Figure 2-12. The S-band feed is located at focus and the cassagrain Ku-band feed is mounted with the dichroic subreflector.

2.2.1.3 Weight Summary

Table 2-9 summarizes the weight changes between the uprated TDRS and the baseline. The weight of the uprated TDRS is shown in Table 2-10. The weight contingency is 26.3 kg, reduced from the 37.2 kg in the baseline.

2.2.2 Uprated TDRS Without UHF/VHF LDR System

A study was made of a TDRS with the UHF/VHF system eliminated to save weight and reduce complexity.

Figure 2-13 shows removing the LDR array permits bringing the HDR/MDR antennas 1.31 m closer to the spacecraft centerline (from 4.32 m to 3.01 m distance). The solar panels can be brought inboard from 3.96 m to 3.44 m with a slight reduction of the weights of the antenna and solar panel support struts. The main weight reduction is in the saving of 21 kg by elimination of the LDR array and electronics.

All other systems remained unchanged from the uprated TDRS spacecraft. The loss of all LDR capability does not make this modification appear viable.

Table 2-11 summarizes the weight of this modified uprated TDRS.

The uprated TDRS without the LDR/UHF/VHF array is packaged in a similar fashion to the uprated TDRS previously described.

Without the LDR elements and their swing arms, the space available ahead of the spacecraft body increases for use for the HDR/MDR antennas. As they are on slightly shorter support struts than the uprated TDRS, the furled antennas swing in slightly closer to the spacecraft body, with the TDRS/GS antenna folded down as before, above the furled antennas. The solar panels fold down upon their support struts in a similar fashion to the uprated TDRS and fold around the spacecraft body as before.

Similar locks and latches restrain and position the antennas and panels in their stowed position within the Delta 2914 fairing.

The same 37-31A attach fitting to the third-stage motor of the Delta 2914 is used to position the spacecraft on the launch vehicle.

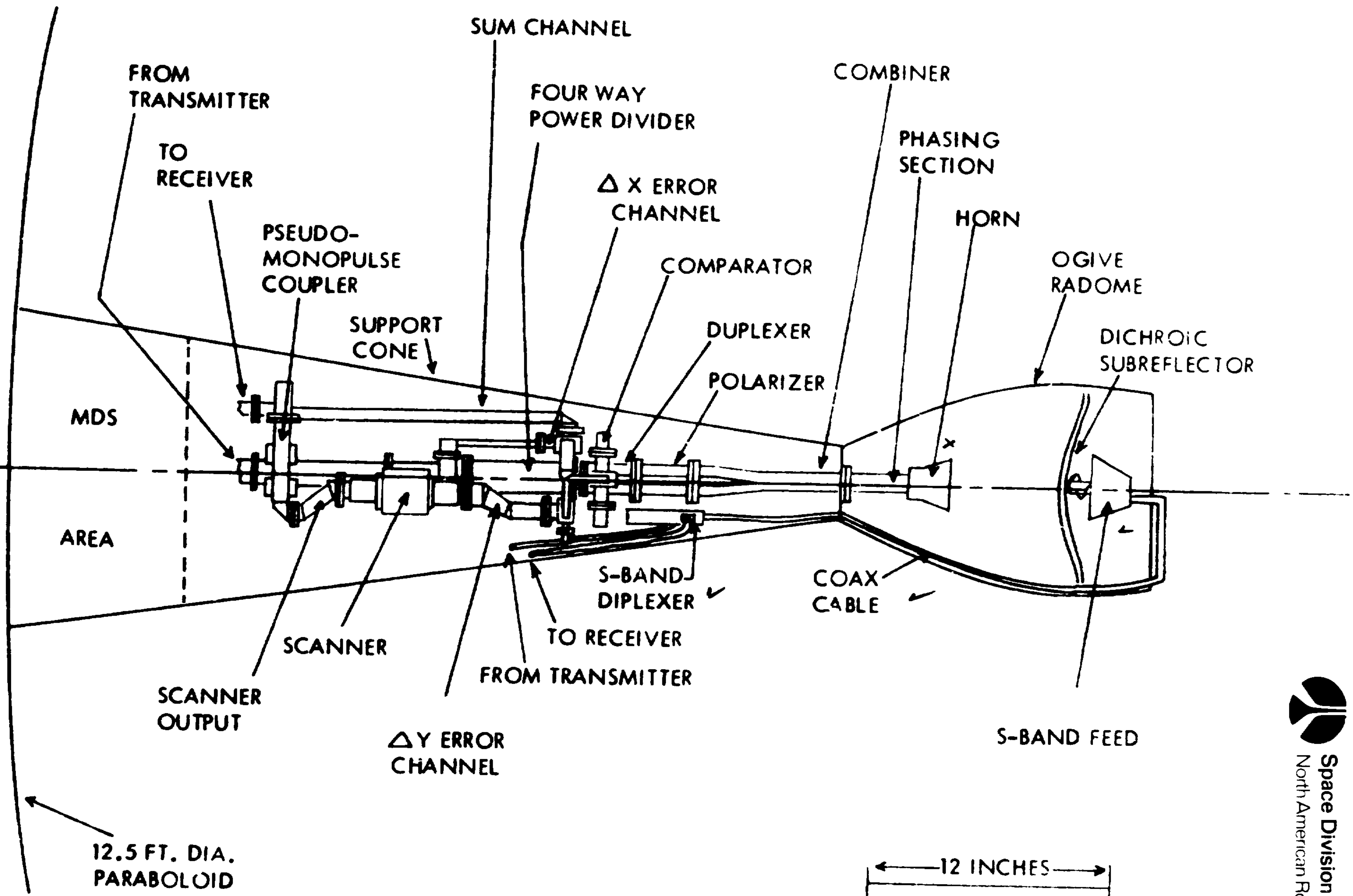


Figure 2-12. Feed System Arrangement

Table 2-9. Uprated TDRS Weight Changes

	KG	LB
1. COMMUNICATION SYSTEMS	(+14.53)	(+32.05)
ANTENNAS	(+ 5.72)	(+12.63)
HDR/MDR INCREASE TO 3.8M DIA	+4.94	+10.88
TDRS/GS INCREASE TO 1.8M DIA	+0.79	+ 1.75)
ELECTRONICS	(+ 8.80)	(+19.42)
LDR TRANS/DIVIDER (COMBINED)	- 1.8	- 4.0
MDR RECEIVERS (S & Ku BAND)	+ 1.0	+ 2.2
TRANSMITTERS (S & Ku BAND)	+ 1.9	+ 4.12
TDRS/GS TRANSMITTER (TWT)	+ 6.96	+15.34
FREQ SOURCE	+ 1.0	+ 2.2
RF CABLING & WAVEGUIDE	+ 1.5	+ 3.3
REMOVE Ku-BAND BEACON	- 1.7	- 3.74
2. ELECTRICAL POWER SYSTEM	(- 0.69)	- 1.5
LENGTHEN SOLAR PANEL STRUTS	+ 1.95	+ 4.3
INCREASE SOLAR PANELS 0.186M^2 & LIGHTER CONST	- 0.8	- 1.8
ELIMINATED AMP HR METERS (2)	- 1.8	- 4.0
3. AUXILIARY PROPULSION SYSTEM	(- 2.9)	(- 6.4)
ELIMINATED N/O EXPLOSIVE VALVES AT THRUSTERS (16)	- 2.90	- 6.4
TOTAL	+10.95	+24.15
CONTINGENCY (PHASE I BASELINE = 37.2 KG (82 LB)	26.3	57.8

Table 2-10. Uprated TDRS Weight Summary

	Part I Baseline		Uprated TDRS	
	Weight		Weight	
	lb	kg	lb	kg
Communications				
Electronics	122.2	55.5	141.9	64.4
Antennas	117.9	53.5	130.3	59.1
Attitude stabilization and control	57.7	26.2	57.7	26.2
Electric power	97.0	44.0	93.0	42.2
Solar array	58.6	26.6	61.1	27.7
Structure	91.0	41.3	91.0	41.2
Thermal control	23.9	10.8	23.9	10.8
Auxiliary propulsion hardware	38.4	17.4	32.0	14.5
	606.7	275.0	630.9	286.1
Propellant + N_2 (2-65°--15-day staion changes)	49.3	22.4	49.3	22.4
Total spacecraft	656.0	297.6	680.2	308.5
DELTA 2914 VEHICLE				
Total spacecraft	656.0	297.6	680.2	308.5
Contingency	82.0	37.2	57.8	26.3
Allowable PL (Delta 2914 + CTS apogee motor)	738.0*	334.8*	738.0	334.8
Empty apogee motor case	50.0	22.7	No Change	
Initial on orbit	788.0	357.5		
Burned-out insulation	8.0	3.6		
Apogee motor propellant	688.0*	312.1		
Synchronous orbit injection	1484.0	673.2		
Transfer orbit propellant	6.0	2.7		
Delta sep. weight (27° transfer orbit)	1490.0	675.9		

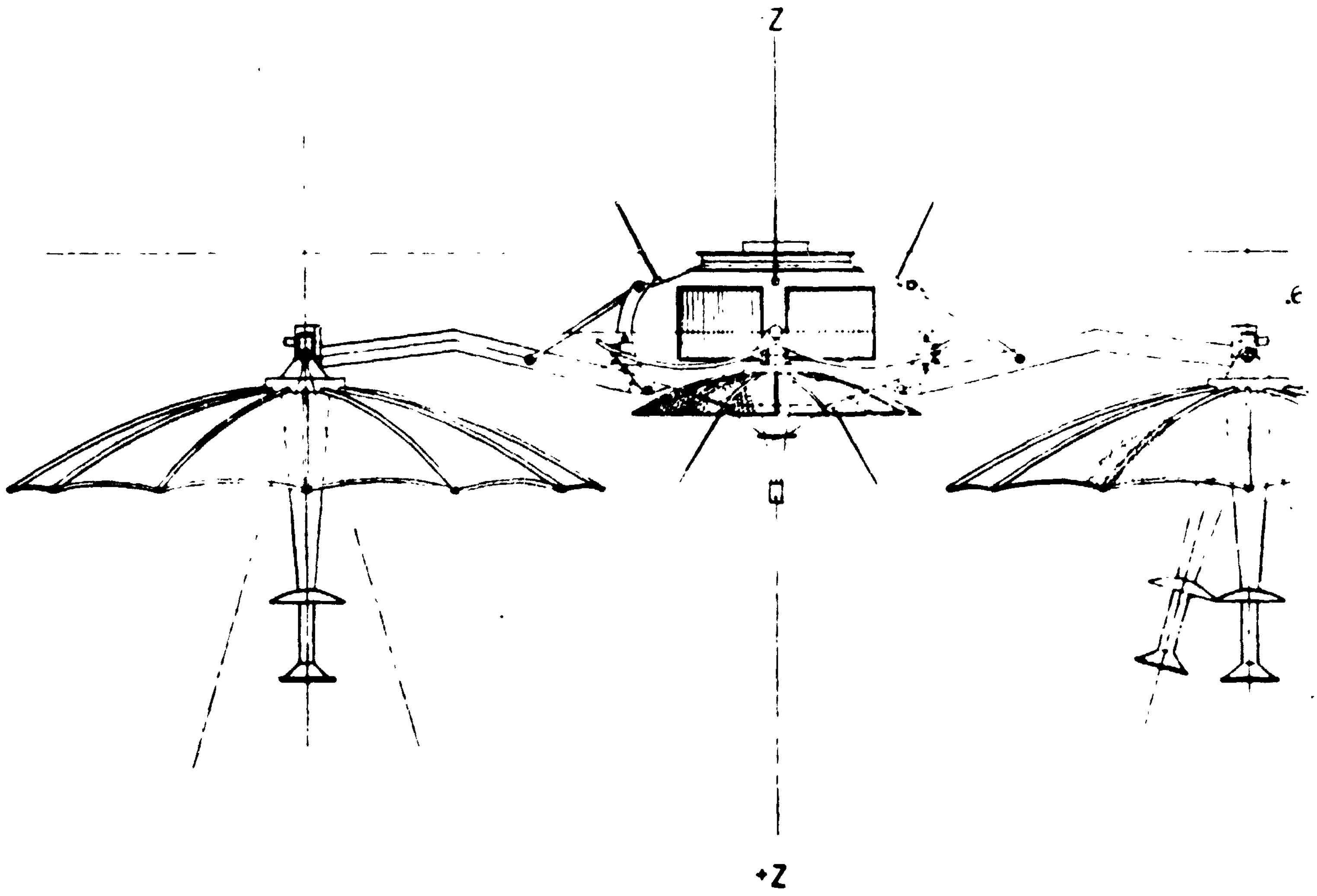
Considering the same allowable payload of 334.8 kg for the Delta 2914 plus the CTS apogee motor as before, and with a spacecraft weight of 387.6 kg, the contingency becomes 47.3 kg or 17 percent of the spacecraft dry weight (Table 2-11).

Table 2-11. Uprated TDRS Without LDR UHF/VHF Array Weight

	Weight (kg)
Communications	
Electronics	57.8
Antennas	44.8
Attitude stabilization and control	26.2
Electric power	42.2
Solar array	27.7
Structure	41.2
Thermal control	10.8
Auxiliary propulsion hardware	14.5
	265.2
Propellant + N_2 (2-65 deg--15-day station changes)	22.4
Total spacecraft	287.6
Total spacecraft	287.6
Contingency	47.2
Allowable payload (Delta 2914 + CTS apogee motor)	334.8
Empty apogee motor case	22.7
Initial on orbit	357.5
Burned-out insulation	3.6
Apogee motor propellant	312.1
Synchronous orbit injection	673.2
Transfer orbit propellant	2.7
Delta separation weight (27-deg transfer orbit)	675.9
5 deg/day drift orbit	

The deployment sequence is identical to that described for the uprated TDRS but with the elimination of the LDR antenna system and its sequential operations.

FOLDOUT FRAME



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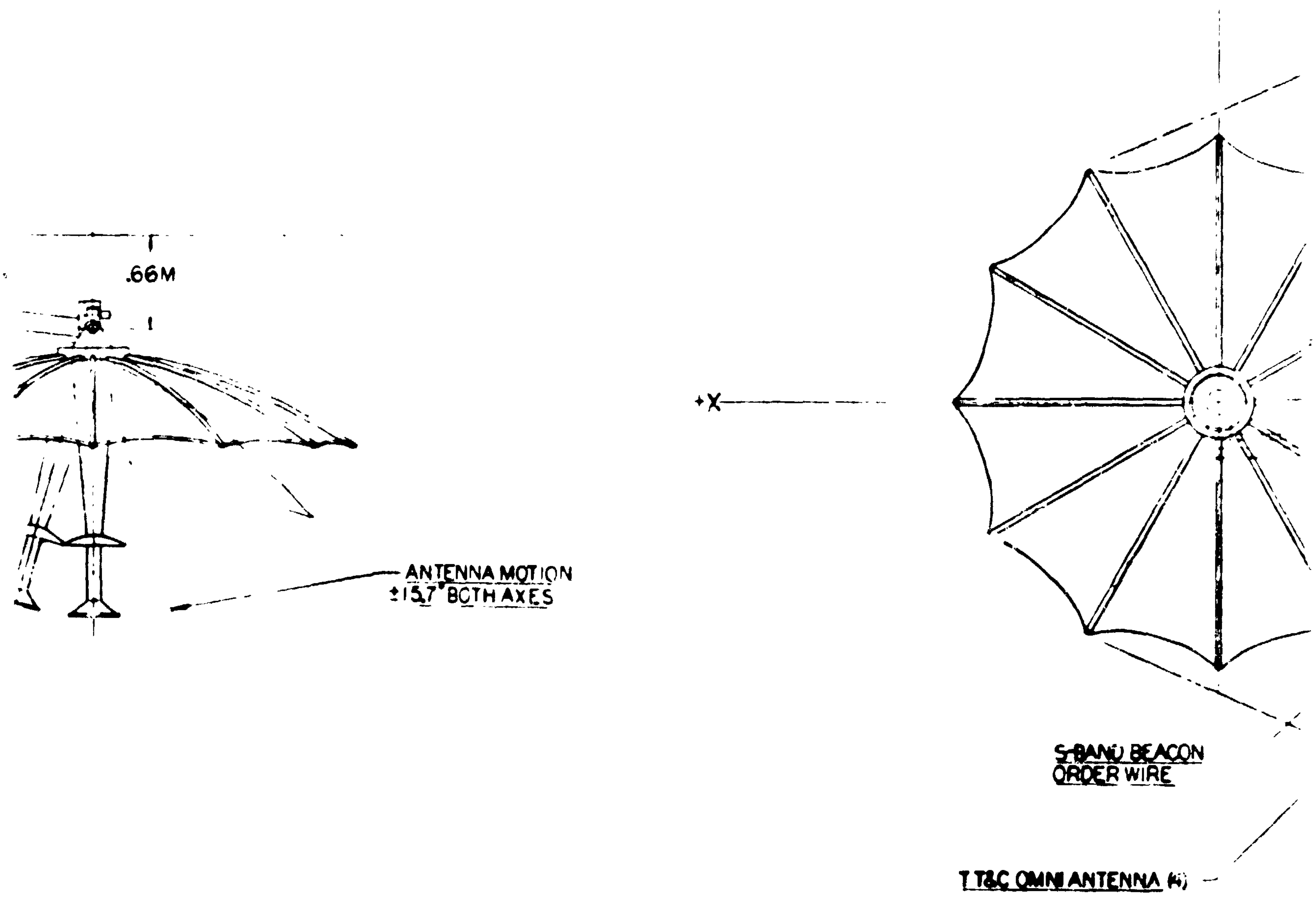

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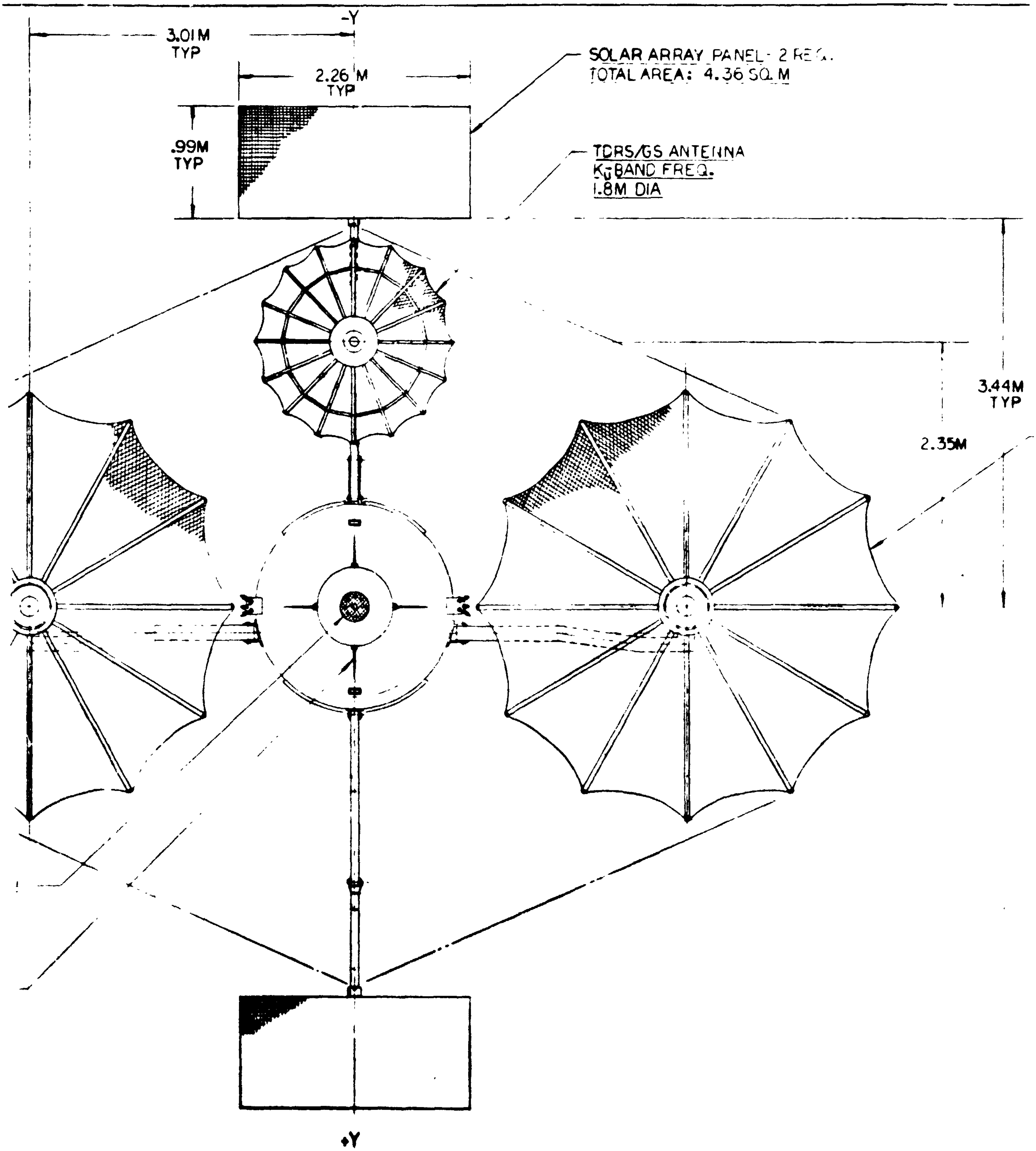

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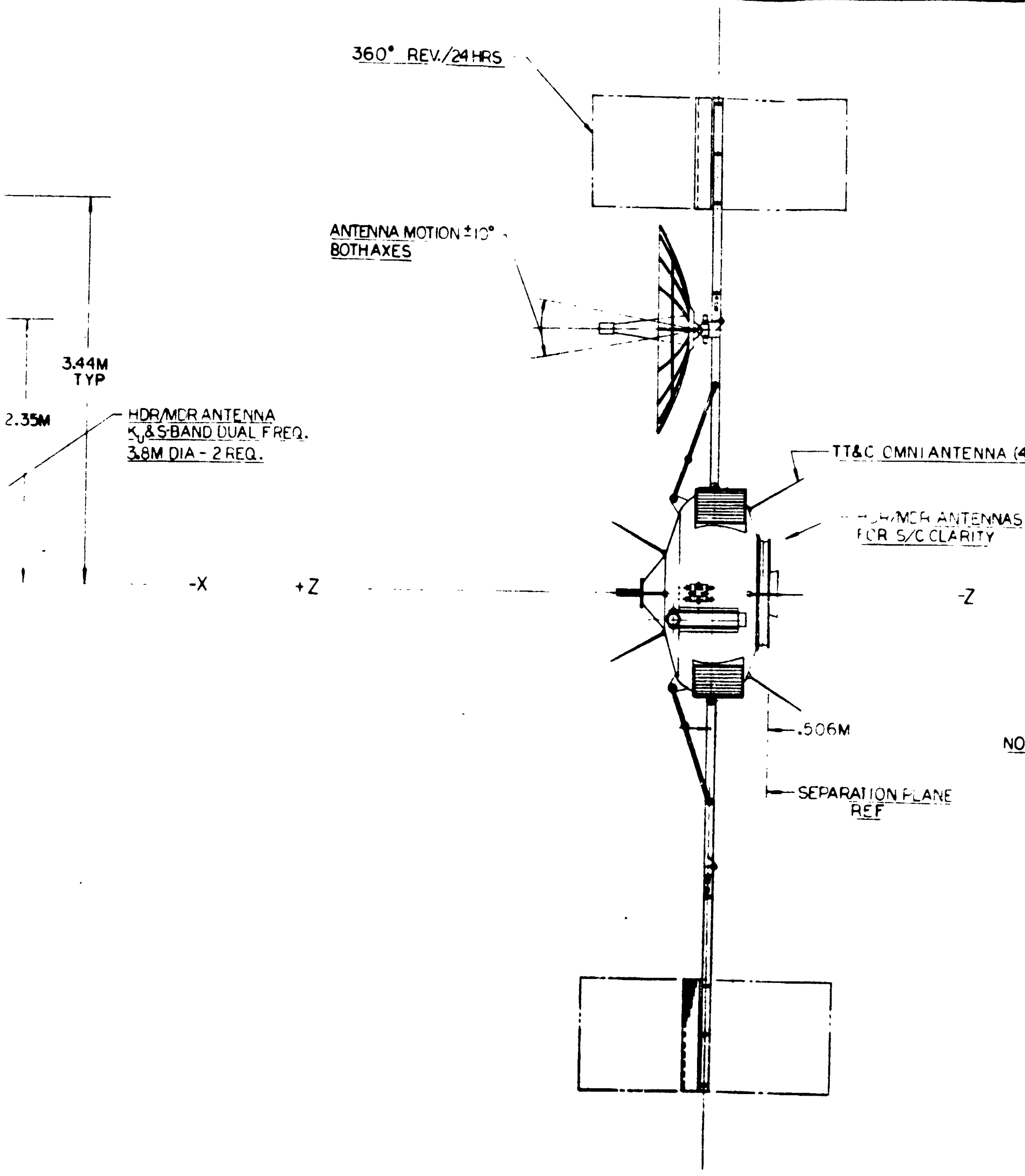
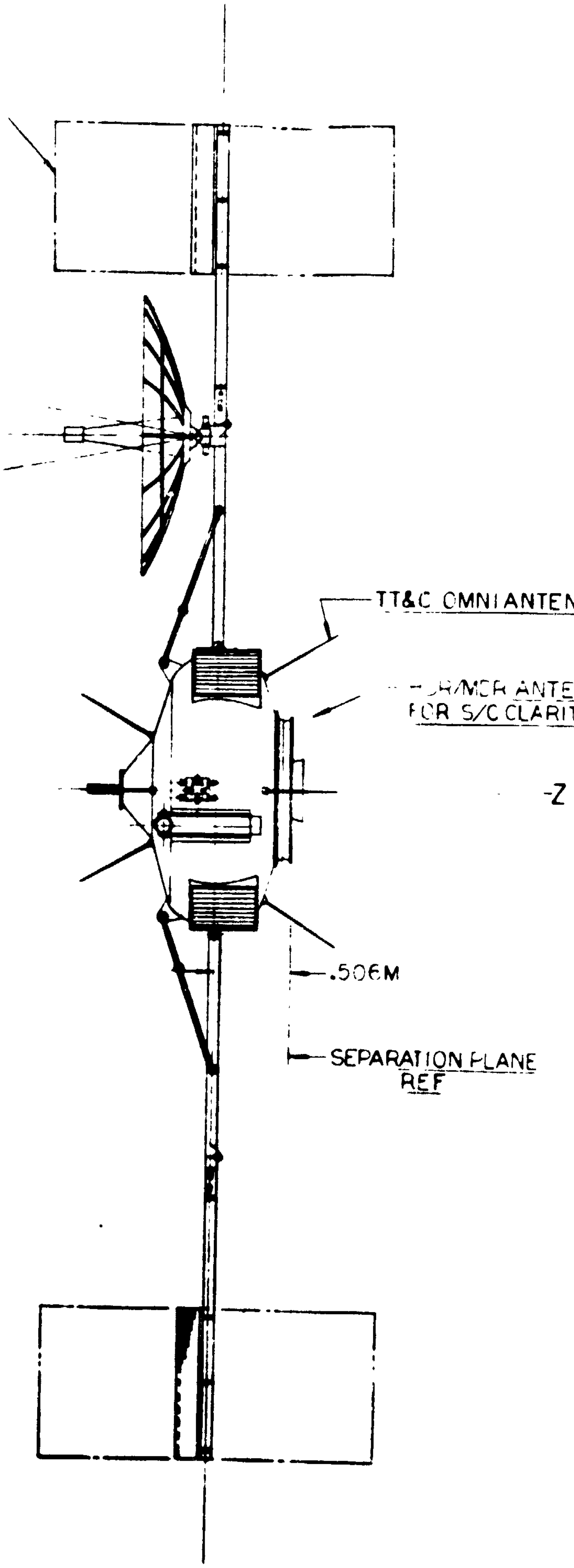




NOTES

FOR DELTA 2914 LAUNCH CONFIGURATION REFER TO DRWG 1199-55. PACKAGING WOULD BE SIMILAR.

Figure 2-13. TDRS Deployed Configuration, Two 3.8 M Diameter HDR/MDR Antennas, No LDR UHF/VHF Arrays

2.3 SUBSYSTEMS

The requirements of the uprated TDRS have only minimal effect on the baseline subsystems other than the telecommunications. The increased antennas result in a reduction in most power requirements and a reassessment of the power system and operations procedure showed adequate capability. Table 2-12 shows these changes in power. Most of them are due to changes in requirements and antenna size. The stepper motors for the antenna drive and the solar panel drive were changed to eliminate power requirements when the motors are not impulsing, resulting in a low duty cycle and an average power reduction of 11 watts.

Table 2-12. Electrical Power Changes (Watts)

ITEM	CHANGE	BASELINE	UPRATED	DELTA
LDR Forward Link	Baseline (1 data + 1 voice, steered beam) Uprated (1-FFOV, no voice)	115.8	116/61	+0/-55
MDR/HDR Receiver (Both)	Uprated S + Ku; Baseline S or Ku	12.4	16.4	+4.0
MDR/HDR Transmit	Baseline 2 m dish; uprated, 3.8 m			
S-Band Data		47.5	15.0	-32.5
S-Band Voice		83.0**	66 (peak)	-17.0
Ku-band Data		13.2	5.0	- 8.2
Antenna Track (each)	Eliminate motor power when not impulsing	7.0	4.0	- 6.0
Solar Panel Drive		6.5	1.5	- 5 0
TDRS/GS Transmit	Baseline, .09 m dish; uprated, 1.8 m			
MDR - 7.5 dB		11	NA	
MDR - 17.5 dB		49*	6	
HDR - 7.5 dB		NA	20.8	
HDR - 17.5 db		NA	50.6	+39.6
Frequency Source	Added frequencies required	4.8	8.0	+ 3.2
Ku Acquisition Beacon	Baseline--continuous; uprated--eliminate	8.3	0	- 8.3
S-Band Tracking/Orderwire	Baseline--continuous; uprated--orderwire cont.	7.9	2.0	- 5.9

*38 W taken from batteries
**25% duty cycle

The solar panel was increased in area and the skin and glue thickness reduced. Two amp-hour meters were removed from the EPS and 16 explosive valves from the auxiliary propulsion system to reduce weight.

There is no change in the attitude stabilization and control system. Thermal control was reinvestigated. Although the overall power is no more than in the baseline, the high power components were repositioned to alleviate any potential hot spots caused by local increases in power, and to minimize the size of the radiators.

2.3.1 Electrical Power System

The electrical power system (EPS) for the baseline TDRS is described in Section 8.0 of SD 72-SA-0133. The only changes made to this baseline configuration were to increase projected solar panel area by .186 m^2 (2 ft^2) to increase the capacity by 19.2 watts at BOL and 16.5 watts at EOL, and to remove the two amp-hour meters. The solar panel skin was reduced from .008 mil to .006 mil, and the adhesive to .005 mil, resulting in a net weight saving of 3.3 kg. The removal of the amp-hour meters removed an additional 1.8 kg. All other components in the EPS are the same as shown in SD 72-SA-0133. The modified system is shown in Figure 2-14 with weights shown in Table 2-13.

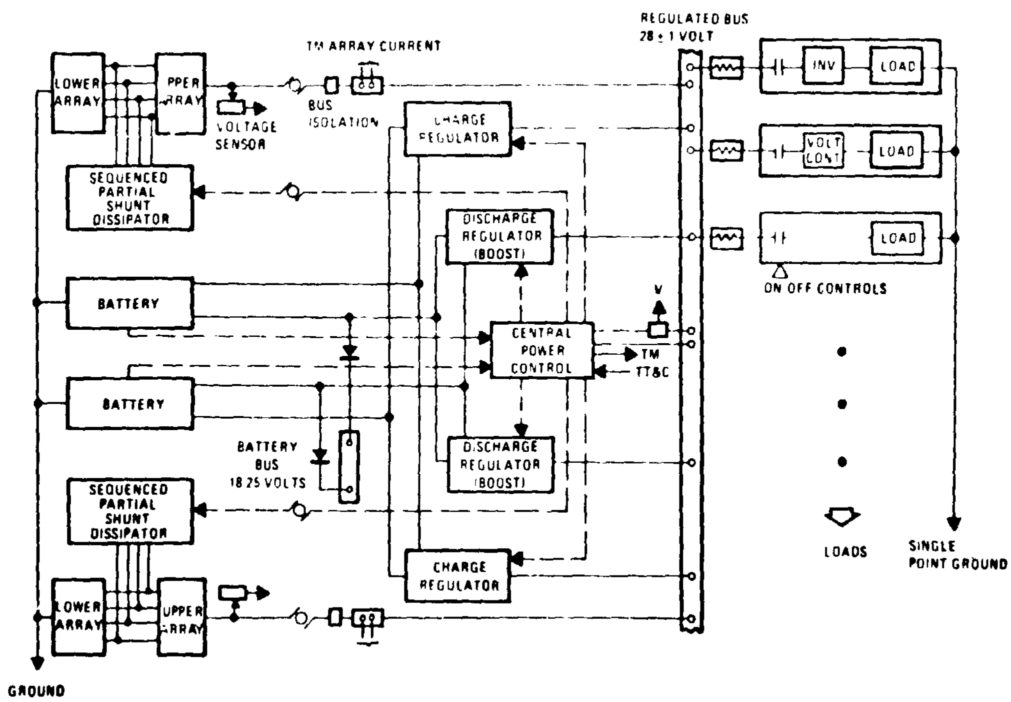


Figure 2-14. Electrical Power Subsystem Block Diagram, Uprated TDRS

Table 2-13. Electrical Power Subsystem Weights

Components/Assemblies	Weight kg	Weight lb	Potential Supplier
Solar Array	(27.7)	(61.1)	
Panels (2)	16.7	36.8	EOS, Ferranti
Drive mecnanism (2)	6.8	15.0	BBRC, SPAR, GE
Linkage and fitting (2)	4.2	9.3	NR
Power Conditioning and Distribution	(22.1)	(48.7)	
Charge and discharge	5.1	11.3	
Central control and logic	2.3	5.1	GE
Packaging	2.2	4.9	
Shunt dissipators	1.1	2.4	
Power conditioner voltage	2.3	5.0	
Cabling	9.1	20.0	NR
Energy Storage			
Batteries (2)	(20.1)	44.3	GE
Total	69.9	154.0	

The method of analysis used for the uprated TDRS is slightly different than in the baseline case. Because of the large number of possible modes of operation (i.e., combinations of rain margin, S/Ku, data/voice/video, high/low power forward LDR, etc.) and the variation in available power due to solstice/equinox, this analysis does not consider "average" power values but an actual time profile is used. Similarly, the power available for the whole five-year lifetime is determined and compared with power required for each mode.

The analysis shows the baseline system with .186 m^2 added solar area is limited only for one mode; e.g., where a 17.5 dB rain margin is required for HDR, the LDR forward link is on high power (30 db EIRP), S-voice is transmitted forward from one high-gain antenna, and Ku-video from the other. In this mode, during the last year and one-half of life, continuous transmission cannot be made for a period of approximately 20 to 60 days at each solstice. However, at worst case end of life, this mode can be maintained continuously for 12 hours with battery augmentation, at which time certain operations must be reduced to permit battery recharging. This assumes continuous heavy rain at the ground station and continuous HDR transmission for this period.

Restrictions during the eclipse periods are treated separately.

2.3.1.1 Determination of Power Loadings

Table 2-14 shows the power requirements for the telecommunication system. Other subsystem power requirements are shown in Table 2-15.

Table 2-14. Telecommunication System Power Requirements

Component	Prime Power (Watts)	
	Peak	Average
1. LDR		
Receiver	9.1	9.1
Transmitter	403.6/205.2/106.0(1)	106.0/56.0/31.6(2)
2. MDR/HDR No. 1		
Receiver	8.2	8.2
Transmitter		
*S-band	66.0/15.0(3)	66.0/15.0(3)
*Ku-band	35.8/5.1(4)	35.8/5.1(4)
Antenna drive	24.0	4.0
3. MDR/HDR No. 2		
Receiver	8.2	8.2
Transmitter		
*S-band	66.0/15.0(3)	66.0/15.0(3)
*Ku-band	35.8/5.1(4)	35.8/5.1(4)
Antenna drive	24.0	4.0
4. TDRS/GS		
Receiver	5.3	5.3
Transmitter		
HDR + FDM/FM channel	50.6(5)	20.8(6)
FDM/FM channel only	6.0(5)	6.0(5)
5. Frequency source	8.0	8.0
6. TT&C		
Processor	10.0	10.0
Transceiver	13.5/4.5	0.5(7)
7. TDRS tracking/order wire		
Transponder	7.9	2.0(8)

NOTES
(1) Emergency steered beam mode provides EIRP of +42, +39, or +36 dBw
(2) F-FOV mode provides EIRP of +30, +27, or +24 dBw at 26 deg FOV. Values shown must be increased 10 percent for power conditioning.
(3) S-band mode emits EIRP of +47 and +41 dBw to support manned and unmanned user, respectively.
(4) Ku-band mode emits EIRP of +53.6 and +23.6 dBw to support video to manned user and 1 kbps to unmanned user, respectively.
(5) With 17.5 dB rain margin
(6) With 7.5 dB clear weather margin
(7) Transmitter normally turned off "on-station"
(8) Transmitter turned on only periodically; receiver is always on

Table 2-15. Subsystem Power Requirements (Watts)

	Peak	Average
Attitude stabilization and control	100.5	16.5
Thermal control	25.2	2.0
Solar panel drive and EPS controls	28.2	10.7
Line losses		40.0

2.3.1.2 Daylight Operations

Table 2-16 shows eight of the most critical power cases for the various operating daylight modes, and two cases with forward voice, video, and high power forward LDR turned off to provide maximum power for rapid battery charging.

Modes 1 through 8 assume 100-percent continuous forward voice at S-band on one of the MDR/HDR transmitters. The EIRP on the forward LDR is 27 db and 30 db. The TDRS/GS rain margin is also varied as shown and the second forward link assumes either S-band data, Ku-band data, or Ku-band video.

The table shows that at end of life (5 years) all modes have an adequate power margin at equinox and all except Mode 1 have a margin at solstice. If it is necessary to transmit both voice and video during a heavy rain, transmitting forward LDR on low power (Mode 4) provides an adequate margin. If high power forward LDR is needed, replacing Ku video with S-band data or Ku-band data on the forward link (Mode 2 or 3) provides a margin.

Should it be necessary to use Mode 1, however, battery implementation can be provided for up to 12.5 hours before reaching 60-percent DOD. (This assumes a heavy rain at the ground station requiring more than 7.5 dB margin for this time period.) Voice, video, and high power forward LDR can then be closed down for approximately 2.8 hours to fully recharge the batteries, or less time if a partial charge is acceptable. If forward voice is maintained 100 percent during the charge period, five hours is required to fully recharge batteries.

Figure 2-15 shows the power available from the 4.37 m^2 (47 ft^2) solar panels over the five-year lifetime for launch at equinox and solstice. An adverse 2-1/2 degrees inclined orbit was conservatively assumed for the whole lifetime and the power reduced accordingly. The power required for each mode also is shown. The periods requiring battery augmentation for Mode 1 are shown shaded.

2.3.1.3 Eclipse Operations

The TDRS is in the earth's shadow for up to 1.2 hours for 45 days twice a year at each equinox and must operate on batteries during this period. Figure 2-16 shows the time in eclipse. Weight limitations dictate minimum battery weight and all-up capability cannot be maintained throughout the whole eclipse period. The second TDRS is available, however, and both TDRS's are

Table 2-16. Electrical Load Chart (Watts)

Item \ Mode Number		1	2	3	4	5	6	7	8	9*	10*
SUBSYSTEMS	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)
Attitude Stabilization and Control	16.5										
Thermal Control	2.0										
Solar Panel Drive and EPS Control	10.7										
TT&C	10.5										
COMMUNICATIONS		(317.2)	(296.4)	(286.5)	(262.2)	(241.4)	(287.4)	(266.6)	(273.6)	(190.4)	(160.6)
LDR 30 dB EIRP	125.1	125.1	125.1	125.1			125.1	125.1	125.1		
LDR 27 dB EIRP					70.1	70.1				70.1	70.1
MDR/HDR #1 S-Data	27.2									27.2	27.2
MDR/HDR #1 S-Voice/Data	78.2	78.2	78.2	78.2	78.2	78.2	78.2	78.2	78.2		
MDR/HDR #1 Ku-Data	17.3										
MDR/HDR #1 Ku-Video	48.0										
MDR/HDR #2 S-Data	27.2		27.2			27.2		27.2		27.2	27.2
MDR/HDR #2 Ku-Data	17.3			17.3							
MDR/HDR #2 Ku-Video	48.0	48.0			48.0		48.0		48.0		
TDRS/GS MDR (17.5 dB)	11.3								11.3		
TDRS/GS HDR (7.5 dB)	26.1						26.1	26.1			26.1
TDRS/GS HDR (17.5 dB)	55.9	55.9	55.9	55.9	55.9	55.9				55.9	
Frequency Source	8.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
S-Band Track/Order Wire	2.0										
SUBTOTAL		356.9	336.1	326.2	301.9	281.1	327.1	306.3	312.3	230.1	200.3
System Losses		40	40	40	40	40	40	40	40	40	40
TOTAL		397	376	366	342	321	367	346	352	270	240
EOL Power Available ***											
Equinox	417 →										
Solstice	375 →										
Power Margin**											
Equinox		20	41	51	75	96	50	71	65	147	177
Solstice		-22	-1	9	33	54	8	29	23	105	135

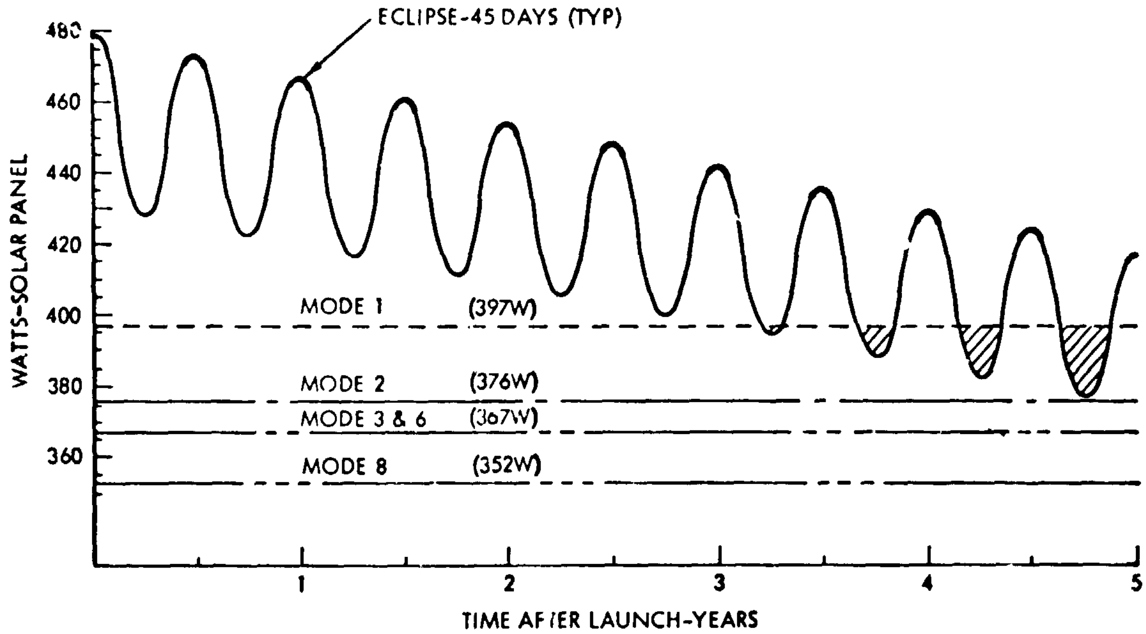
Modes 1-6, 9 are HDR rain margin = 17.5 dB; Modes 6, 7, 10 HDR rain margin = 7.5 dB; Mode 8 is MDR rain margin = 17.5 dB

Modes 1-3, 6-8 are LDR forward with 30 dB EIRP; Modes 4,5,9, 10 are LDR forward with 27 dB EIRP

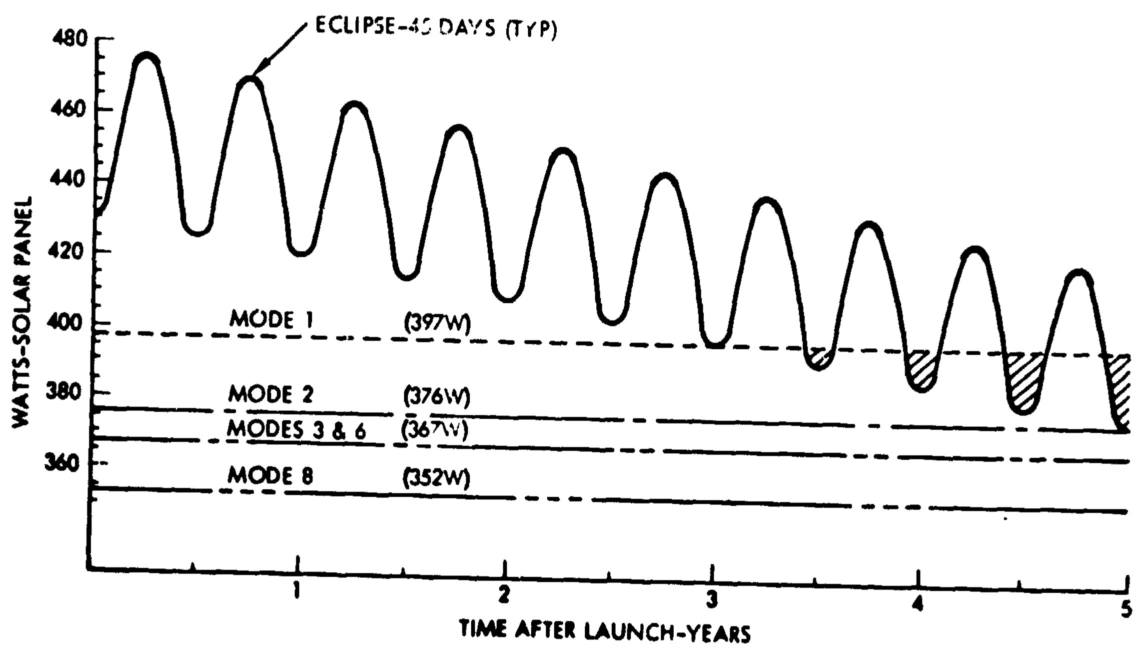
* Reduce operations (i.e., no voice or video) for rapid battery charge if needed.

** For battery charge, additional service, or margin. If minus, must be supplied by battery on duty cycle. If minus, or insufficient to charge batteries in reasonable time, some service (e.g., voice, video, high power LDR) must be reduced until batteries are fully charged.

*** Power at BOL = 487 watts, equinox
= 436 watts, solstice



(a) Launch at Equinox



(b) Launch at Solstice

Figure 2-15. Power Available/Required

not simultaneously in shadow. This section identifies the operational limitations that must be imposed during eclipse.

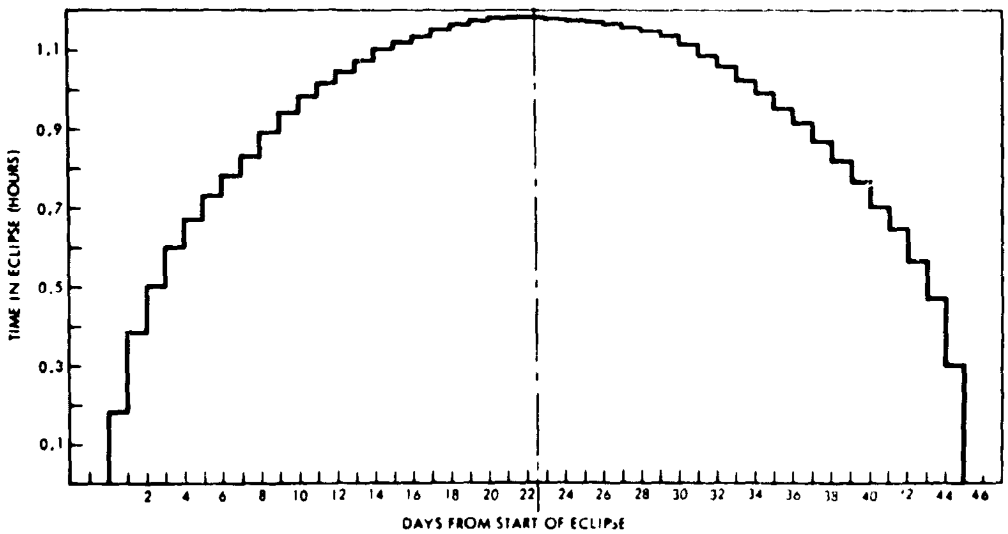


Figure 2-16. Time in Eclipse

Basic eclipse operations assume the items in Table 2-17 operating. This results in three cases as shown.

Two 16-cell, 12-amp-hour batteries are used for energy storage. They have a capacity of 460 watt-hours. Limiting maximum depth of discharge to 60 percent results in 276 watt hours of usable energy. Table 2-17 shows the basic cases all have a margin for the worst 1.2-hour eclipse. However, as shown in Figure 2-16, extra battery capability is available on days other than maximum eclipse for Case E-1 and at all times for Cases E-2 and E-3. The excesss can be used either for S-voice or high power forward LDR.

The difference in power is:

S-voice/data = 78.2, S-data = 27.2; delta = 51 watts

LDR forward high power = 125.1, low power = 70.1; delta = 55 watts. These values are nearly the same and it is assumed that a net increase of 60 watts (including losses) is required for each of the increases.

Table 2-17. Eclipse Operations

Item	Power (watts)
Subsystems	39.7
LDR (27 dB EIRP)	70.1
Frequency source and S-band track/order wire	10.0
MDR/HDR No. 1 S-data	27.2
MDR/HDR No. 2	OFF
Subtotal	147.0

TDRS/GS			
MDR	17.5 dB rain margin	11.3	One only
HDR	7.5 dB rain margin	26.1	
HDR	17.5 dB rain margin	55.9	

Case Number	E-1	E-2	E-3
TDRS/GS			
Mode	HDR	HDR	MDR
Margin	17.5	7.5	17.5
Power	203	173	158
System losses	20	17	16
Total power	223	190	174
Total energy (w-h)*	268	228	210

*Maximum 1.2-hour eclipse

Figure 2-17 shows the battery depth of discharge for the three basic cases in Table 2-17. The space above the line is the power available for voice or high LDR. The power available is shown in Figure 2-18. Also shown are the days when full service with S-voice or high forward LDR, or both of them, cannot be used. During the period when full service cannot be supplied, partial voice and/or high power forward LDR can be used.

Based on the extra energy available in the battery shown in Figure 2-17, Figure 2-19 shows the time during the eclipse period that either voice or high power forward LDR can be used. The distance above the base curve indicates the time both can be used.

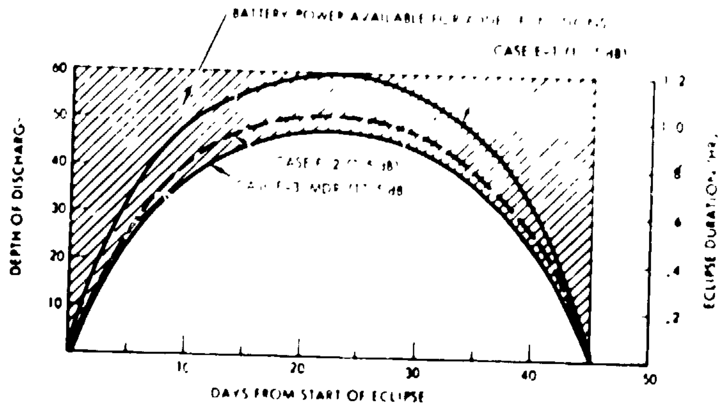
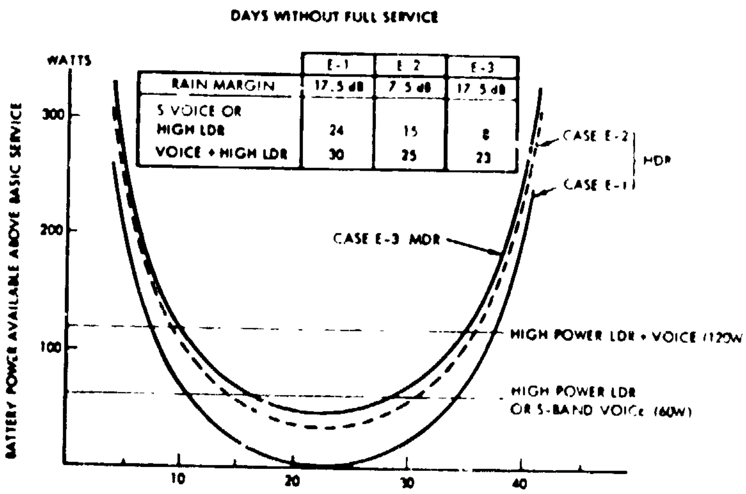


Figure 2-17. Battery DOD Versus Eclipse Period



	E-1	E-2	E-3
RAIN MARGIN	17.5 dB	7.5 dB	17.5 dB
S VOICE OR HIGH LDR	24	15	8
VOICE + HIGH LDR	30	25	23

Figure 2-18. Extra Battery Power Available

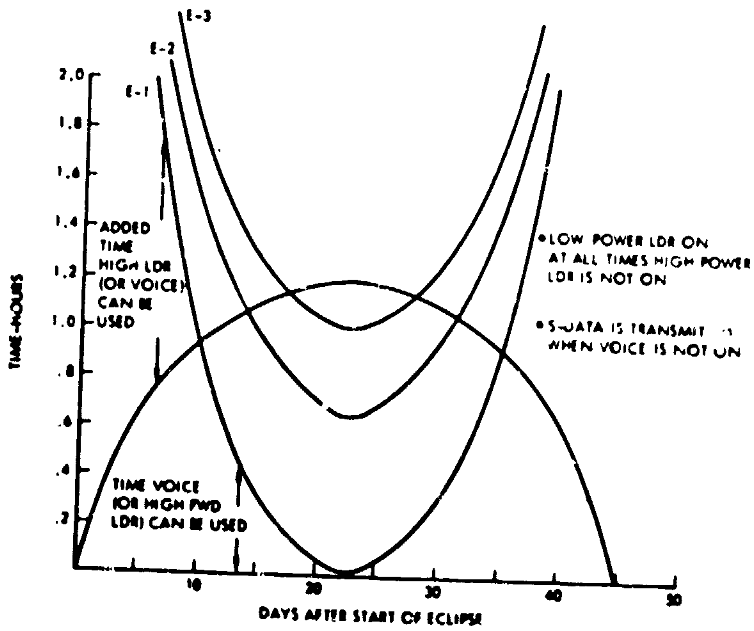


Figure 2-19. Eclipse Time Voice and/or High Power LDR Can be Used

This figure shows that 100-percent voice is not obtainable during the entire eclipse period if low power forward LDR is on continuously. However, by turning off the low power forward LDR part of the time, 100-percent voice can be obtained during eclipse. The time when LDR must be off, if 100-percent voice is used, is shown in Figure 2-20.

Two factors modify these constraints. The first is that they are based on maximum DOD of 60 percent. This can be exceeded on occasion during the satellite life without appreciable affect on the battery. In emergencies, therefore, the limitations can be exceeded. Secondly, a second TDRS is available out of the eclipse and will be able to command users later in the users' orbits.

2.3.1.4 Battery Charge Times

When the TDRS comes out of eclipse, the batteries will be discharged--possibly as much as 60 percent. Prudent procedure dictates recharging as soon as possible. The time to recharge the batteries is a function of power available for charging, depth of discharge, and battery temperature. This relationship is shown in Figure 2-21 and modified in Figure 2-22 to give a direct reading for 40- and 60-percent DOD.

The battery depth of discharge for basic eclipse operations (no voice or high power forward LDR) is shown in Figure 2-17. The power available to charge the batteries after coming out of eclipse can be obtained from Table 2-17. This shows (in conjunction with Figure 2-22) that the batteries cannot be charged in a reasonable time in Modes 1, 2, 3, 4, 6, and 8, indicating forward voice should be discontinued (or put on a duty cycle until the battery is charged. Similarly, LDR forward could remain on low power to minimize recharge times. These cases are shown by Modes 9 and 10, where 147 and 177 watts are available for battery charge. If high power forward LDR or voice is desired, these values would be reduced approximately 55 watts, resulting in an extra hour of charge time for 60-percent DOD.

Recharging time for Mode 9 (TDRS/GS on HDR with 17.5 dB rain margin) is 2.75 hours from 60-percent DOD, and for Mode 10 (rain margin = 7.5 dB) it is 2.2 hours.

2.3.1.5 Power Profile

A typical power profile is shown in Figure 2-23. A worst case condition is shown consisting of Eclipse Case E-1 at maximum eclipse with the battery discharged 60 percent; Mode 9 (17.5 dB rain margin) for recharge; and Mode 1 (voice + Ku video + LDR forward 30 dB EIRP + 17.5 dB rain margin) for the remainder of the day with end-of-life power.

For this worst case, there is a 20-watt power margin at EOL equinox; but, as shown in Figure 2-15 and Table 2-17, the solstice condition cannot be met at EOL without battery augmentation.

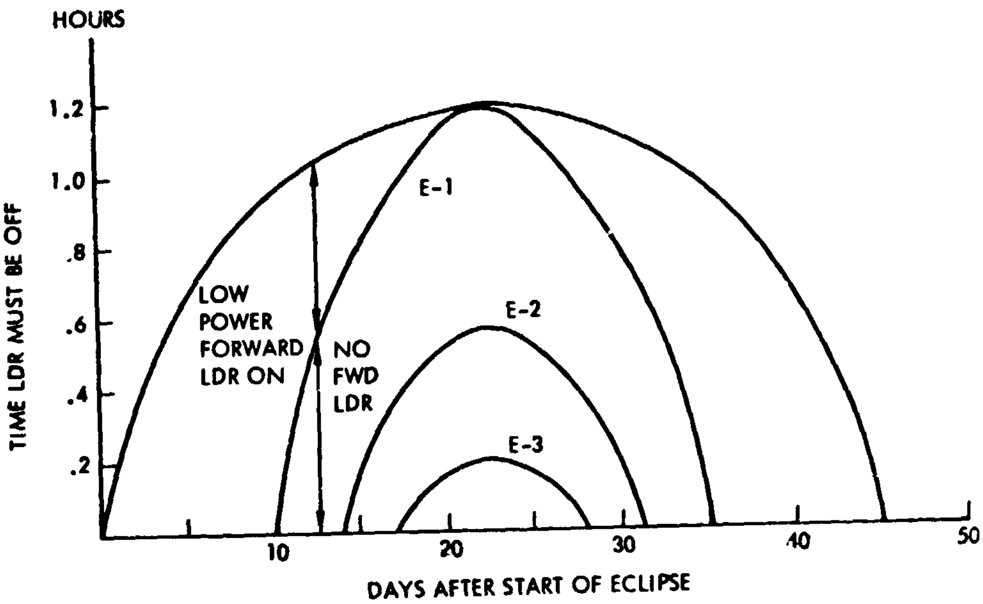


Figure 2-20. Eclipse Time LDR Must Be Off (100 Percent Voice)

2.3.2 Auxiliary Propulsion System

The uprated TDRS requirements do not affect the auxiliary propulsion system. However, the increase in the telecommunications weights cause a decrease in the spacecraft weight contingency. To make up part of this reduction, the explosive valves put at each thruster were removed in the baseline design. These valves were to prevent a runaway thruster should the thruster valve fail open. As the thruster valve is redundant against both open and closed failures, such a double failure is unlikely.

In the baseline, an adequate weight contingency allowed insertion of these explosive valves. In the uprated TDRS, they were eliminated to save 2.9 kg (6.4 lb). The new system is shown in Figure 2-24 and the weights in Table 2-18.

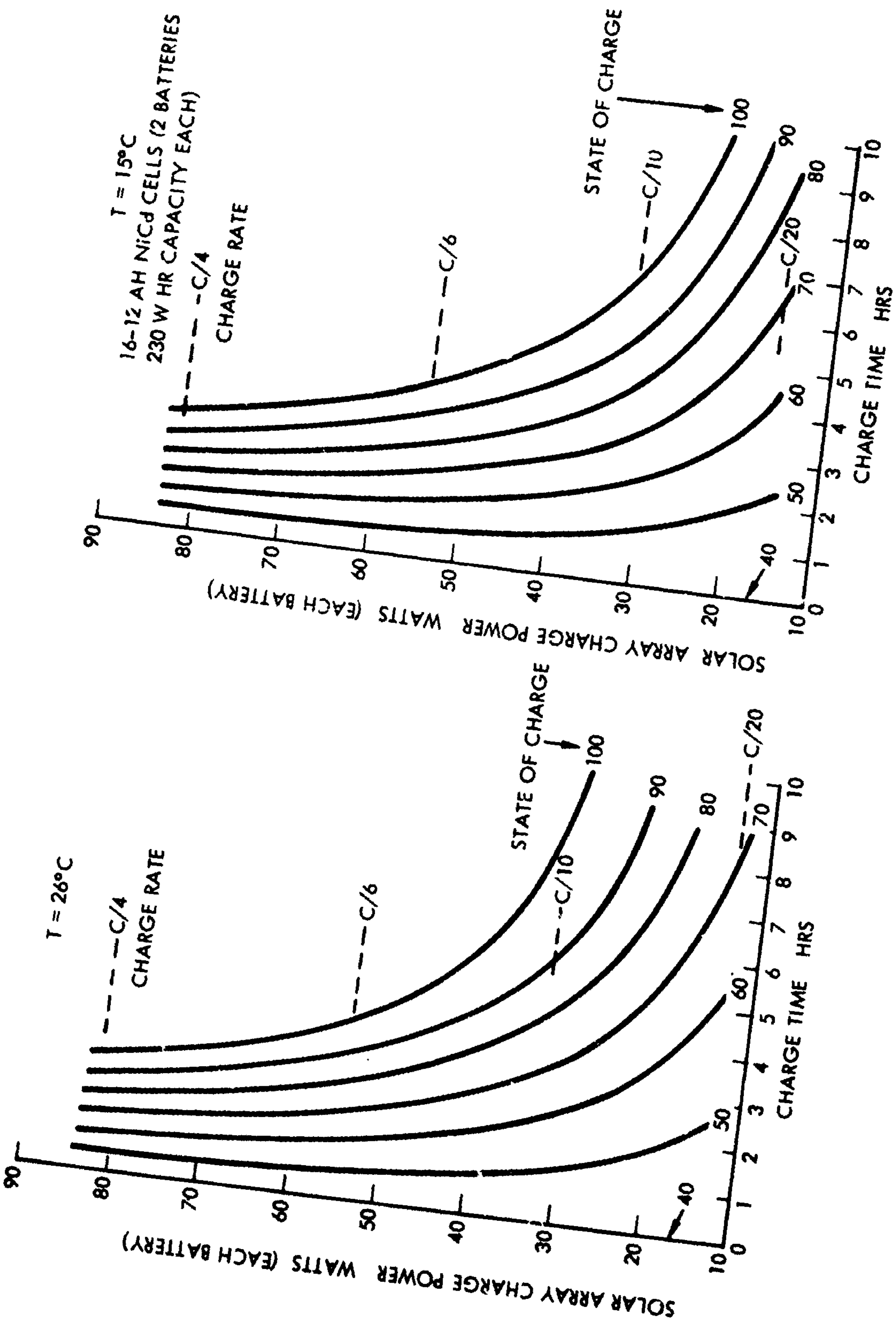


Figure 2-21. Battery Charge Parameters

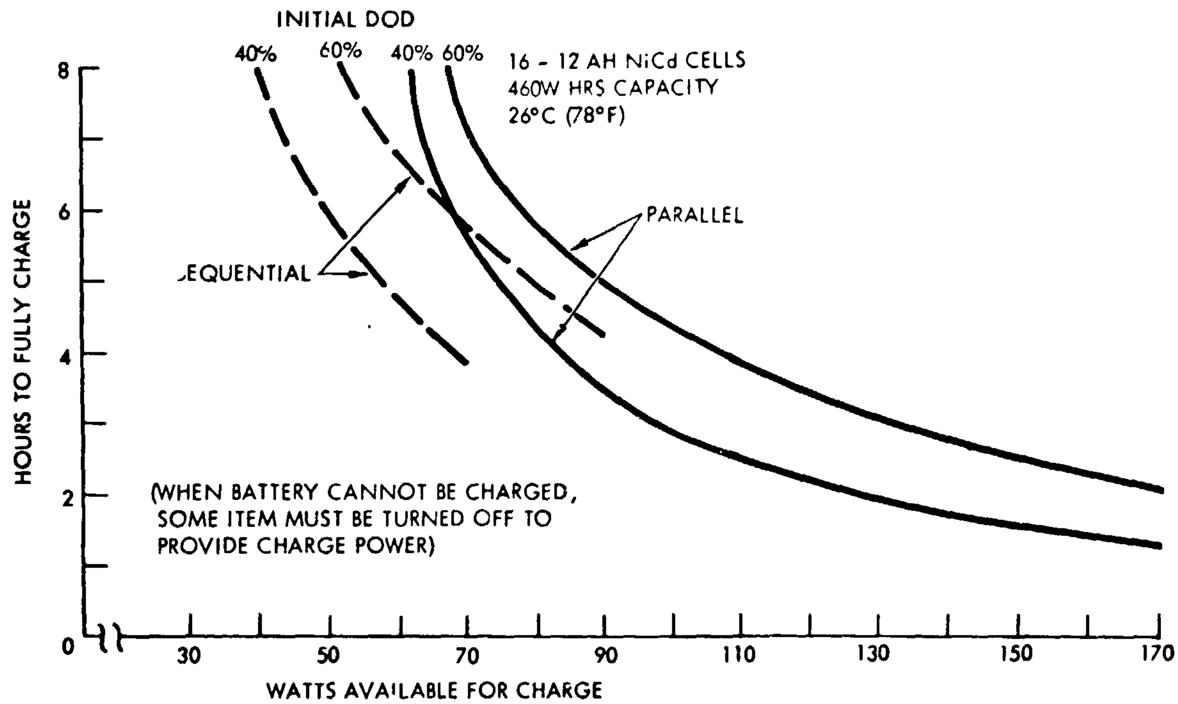


Figure 2-22. Battery Recharge Time

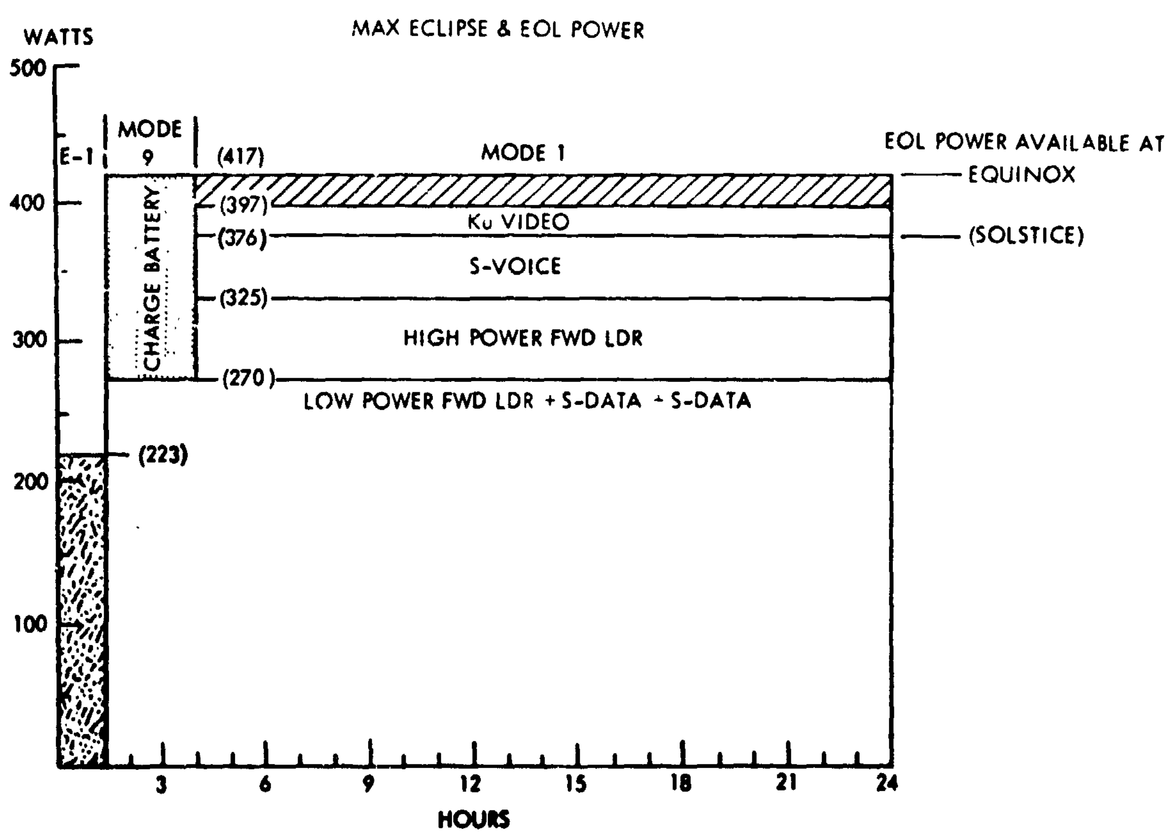


Figure 2-23. Typical Power Profile, 17.5 dB Margin

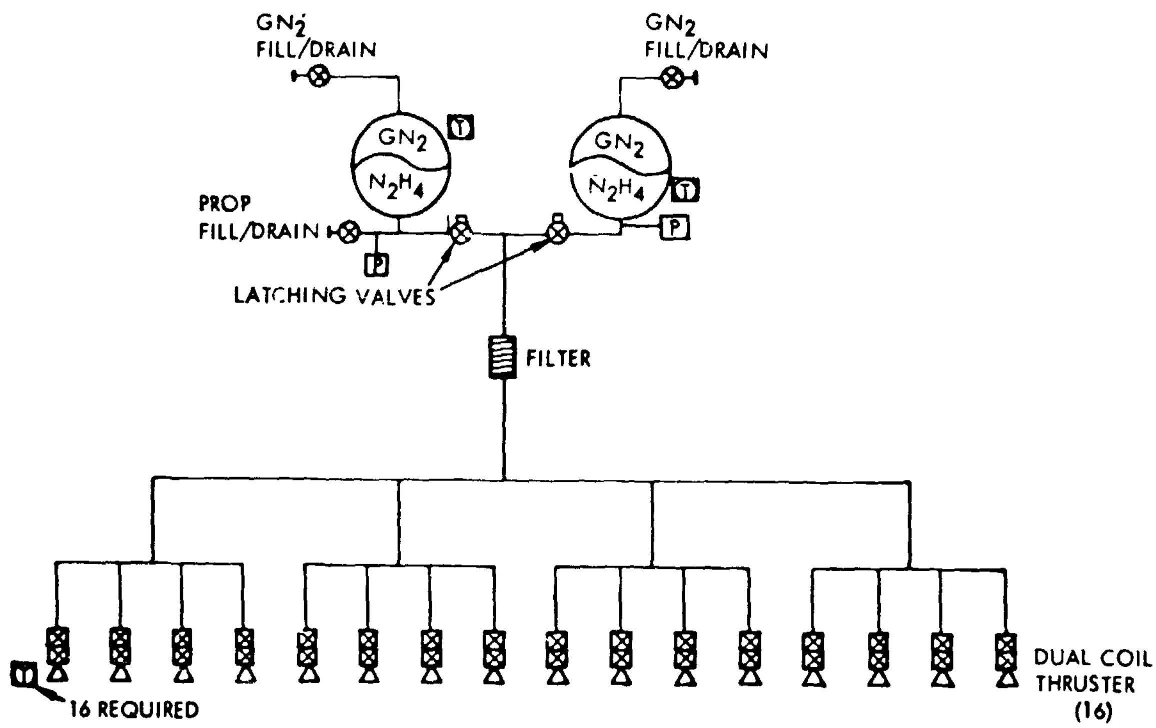


Figure 2-24. Auxiliary Propulsion System Baseline Configuration

Table 2-18. Auxiliary Propulsion System Weights

Component	Manufacturer	Number Required	Weight kg	Weight lb	Previous Use
GN_2 fill/drain	TRW	2	0.27	0.6	Intelsat IV, RAE-B
Prop tank w/EPT-10 diaph.	PSI	2	4.99	11.0	CTS
Latching valve	HRM	2	0.54	1.2	Similar to SMS and RAE-B
Prop. fill/drain	TRW	1	0.14	0.3	Intelsat IV, RAE-B
Filter, 15-micron ABS	Vacco	1	0.18	0.4	Surveyor, Intelsat IV, Mariner, Saturn, LEM
Pressure transducer	Bourns	2	0.27	0.6	Saturn, Scout, Shrike
Temperature transducer (tanks)	Gulton	2	0.27	0.6	Apollo
Temp. transducers (thrusters)	Genisco	16	0.18	0.4	
Thruster	Hamilton-Std.	16	4.35	9.6	CTS, Solrad
Wiring and lines		-	1.36	3.0	
Thruster housing		2	0.68	1.5	
Trapped propellant		-	1.36	3.0	
Total			14.60	32.2	
GN_2			0.27	0.6	
Propellant			25.08	55.3	
Total			40.00	88.1	

2.3.3 Thermal Control

2.3.3.1 Introduction

Due to changes of the Part II design of the TDRS communications and power systems, the heat rejection loads increased from the Part I requirements listed in report SD 72-SA-0133. In addition, in the Part I baseline design analysis the equipment layout was inefficient for transfer of the heat loads. In particular, the temperature differences between the transmitters and TDRS radiation panels contributed to over-limit temperatures when the radiators were overly stressed with solar loading under certain short duration seasonal and orbital conditions. These results led to relocating the LDR and MDR transmitters to the louver panels and relocating other equipment items to equalize power loads among the quads.

This report presents the results of this redesign and resizing the thermal control subsystem. The heat rejection requirements are re-evaluated for the uprated TDRS power and communication subsystems. The communications system transmitters were relocated and remounted to preferred positions on the radiator panels, and the radiator panel resized to meet the increased heat rejection requirements. The methods and results of thermal analysis of the uprated design are presented to show detailed equipment temperatures under worst case conditions.

2.3.3.2 Uprated Requirements

The heat rejection requirements of the uprated TDRS are presented in Table 2-19. The power dissipations of all inboard mounted equipment are itemized. RF power is accounted for in the entries for the transmitters. Wiring losses are distributed and are considered negligible with regard to radiator panel sizing. On the other hand, equipment dissipation which would leak overboard through view ports or masts is conservatively included in Table 2-19, e.g., horizon sensors, panel drive. The operating modes selected are those with maximum power demand, and assumed to load up each quad to maximum possible totals. Therefore redundant systems are double entered when located in different quads since this local operating mode is realistic.

2.3.3.3 Equipment Configuration

As a direct result of the baseline design analysis, the communications system equipment was relocated. (Section 9.2.4, SD 72-SA-0133) The uprated equipment layout is shown in Figure 2-25. The major changes are to directly mount the LDR, MDR, and TDRS/GS transmitters onto the thermal control radiator panels. The LDR transmitters are co-located on quad 3 to accommodate the phased operating mode and so maintain a steady dissipation loading. The reconfiguration of the equipment was accomplished to satisfy the balance, functional, and cabling requirements as well as thermal control.

Table 2-19. Power Dissipation (Watts)

Quad 1.

Front side	
Horizon sensor	0.25
Reaction wheel	3.00
MDR XMTR	52.00
TDRS/GS RCVR	5.30
MDR RCVR	8.20
Back side	
Solar panel drive	1.63
MDR electronics	4.00
Σ	74.38

Quad 2.

Front side	
Horizon sensor	0.25
Reaction wheel	3.00
TDRS/GS XMTR	40.60
TDRS Track RCVR	5.30
LDR RCVR	9.10
Freq. source	8.00
TDRS Track XMTR	2.00
Back side	
Solar panel drive	1.63
ACS electronic	3.00
Σ	72.88

Quad 3.

Front side	
Freq source	8.00
LDR RCVR	9.10
LDR XMTR	66.00
Horizon sensor	0.25
Reaction wheel	3.00
Back side	
ACS gyro	2.00
Pwr module	10.00
Σ	98.35

Quad 4.

Front side	
MDR RCVR	8.20
LDR RCVR	9.10
MDR XMTR	52.00
Horizon sensor	0.25
Reaction wheel	3.00
Back side	
T&C	10.50
LDR RCVR	9.10
Pwr module	10.00
Σ	102.15

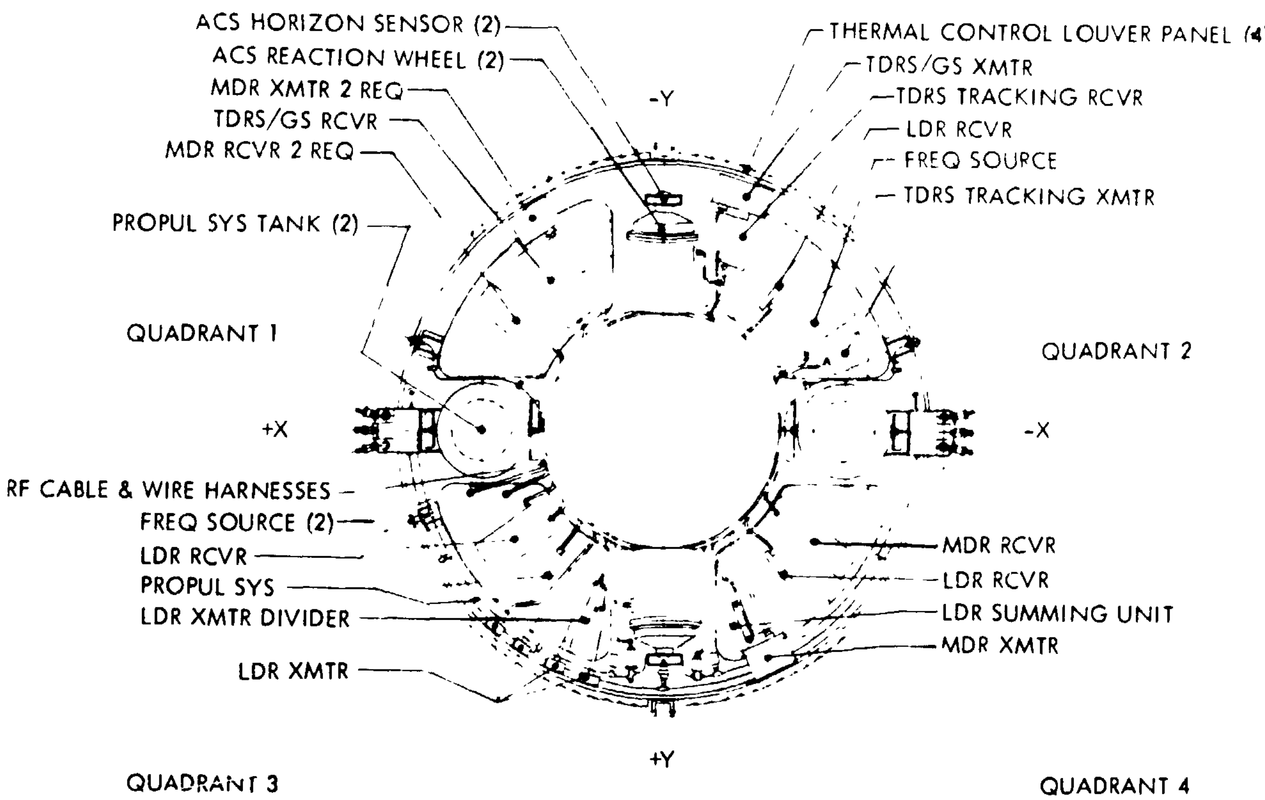


Figure 2-25. Equipment Layout - Front View

2.3.3.4 Analysis

The uprated design analysis follows from the baseline design. Three sections are presented to show the uprated thermal control design: (1) radiator panel sizing, (2) temperature differences between separated equipment and the radiator panels for radiative heat transfer, (3) temperature differences between equipment and mounting shelves or panels for differing concentrated heat loads over larger areas. The results of the three sections are then combined to provide a detailed temperature profile of quad 4 under worst case conditions.

Radiator Panel Sizing. The TDRS baseline thermal control panels were sized to provide maximum heat rejection under solar incidence on the louvers within the constraints of the antenna and array clearance envelope. Under certain seasonal and orbital conditions, the solar loads reduce panel heat rejection at nominal temperature to less than required levels. In these conditions, panel temperatures increase over nominal values to provide the heat rejection. The higher temperatures occur as well on inboard equipment. Several approaches for alleviation of the problem were stated, but only one provides adequate thermal dissipation for the uprated TDRS, namely a radiation window in each quad.

Each quad is assumed to have independent heat rejection capability, since radiant heat transfer of the equipment loads over large distances requires large temperature differences. The minimum heat rejection of a quad with the baseline louver size of .334m^2(3.6ft^2) is 44.4 w for a panel temperature of 30^{o}C when the solstice solar vector lies in the XY plane. The worst case uprated heat rejection load is 102.1 watts in quad 4, (Table 2-19). The area of a radiation window, (insulation cutout) required to reject the additional 57.7 watts is .134m^2 (1.44 square feet) with no solar incidence.

To provide the best supplementation to the louvered panels, the .134m^2 window is located on the forward shell surface opposite the equipment shelf. For quads 2 and 3, the area is apportioned with a separate window on the rear face of the shell opposite the battery. The outboard thermal control coating is back surface microquartz mirrors, and the inboard coating is thermal black paint such as 3M black velvet. These locations allow adequate clearance for inboard mounted louvers, but requirements for inboard louvers are not established

Maximum heat rejection performance for the quad window is 55.6 w at 30^{o}C, while minimum performance with full flood sunlight is 44 w. Since supplemental heat rejection is required only for worst case localized quad solar loading, the nominal performance of the system will be enhanced. It was also assumed that the insulation blanket is perfect and that penetration heat leaks are small. These heat leaks are estimated to be 10 w per quad and to reduce slightly radiator sizing. The effects of these leaks are, however, included in baseline cold case analysis of the transfer orbit and eclipse phases.

Temperature Differences - Equipment to Radiators. Shelf mounted equipment require temperatures higher than the radiator panels to radiatively transfer the power dissipation. The temperature difference primarily depends on the power load and secondly on the separation distance, because of the large reradiation (insulated) surface area. It is assumed that the quad 4 compartment enclosed by the equipment shelf, the forward shell, and the propellant tank and reaction wheel form a radiation enclosure. The equipment and shelf are the heat transfer source, the louvered radiator panel is the sink, while the remaining insulated shell area is a reradiating surface. The radiation network is shown in Figure 2-26.

The significant approximations are:

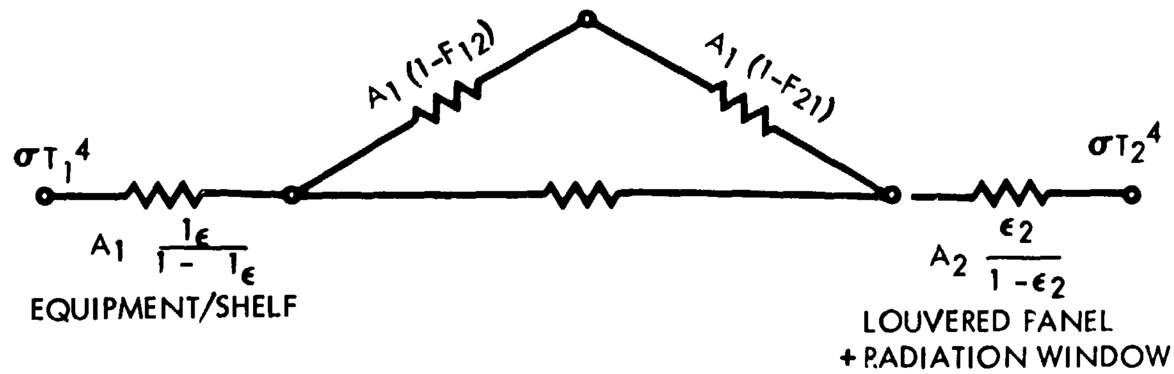
1) The equipment dissipation is radiated from one shelf side only to the quad louvered radiator half area only, i.e., no conduction through the shelf for radiation in the rear compartment.

2) The effective equipment shelf fin radiating area is .093m^2(1.0ft^2) and is analyzed in the next section.

3) The surfaces have near black diffuse emittances.

In quad 4 the MDR and LDR receivers radiate over large separations from the louvered panel and acquire the higher temperatures. The transmitters are direct mounted to the louvered panels and are considered in the next section.

Figure 2-27 shows the temperature difference to transfer heat loads within a range of view factors between 0.1 and 0.25. The specific view factor between the MDR receiver and the radiators is approximately 0.13, and the view factor for the LDR receiver is 0.23. The temperature differences for radiative transfer of these loads are shown on the Figure.



A_1 = .093M^2 (1 FT^2) EFFECTIVE SHELF-FIN AREA

A_2 = .3M^2 (3.2 FT^2) LOUVER PANEL HALF AREA PER QUAD PLUS QUAD WINDOW

ϵ_1 = .95 INBOARD SURFACE EMITTANCE, SHELF

ϵ_2 = .95 INBOARD SURFACE EMITTANCE, RADIATOR

Figure 2-26. Radiation Network of Quad Enclosure

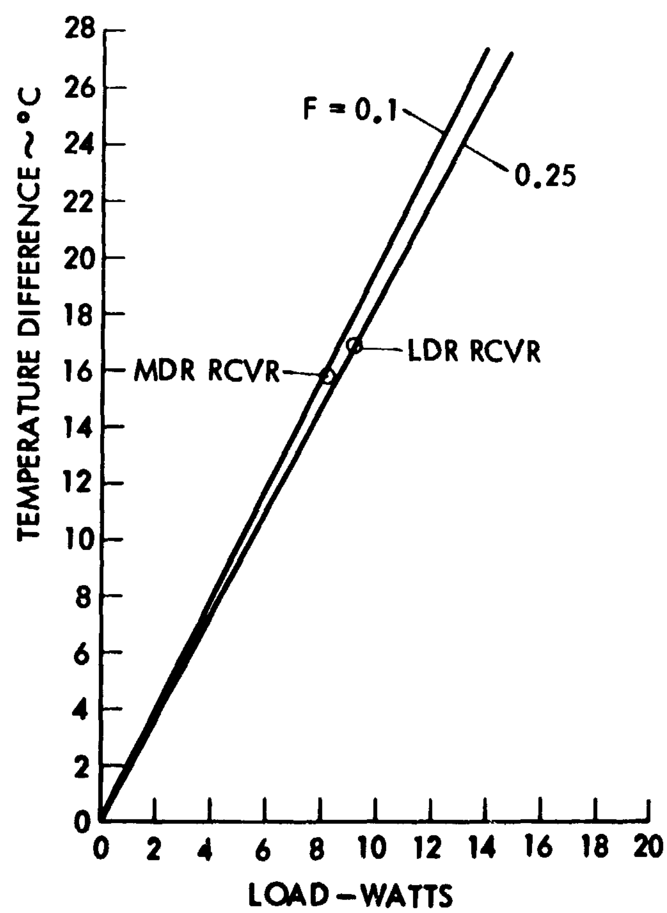


Figure 2-27. Radiative Temperature Difference Vs. Heat Load

Temperature Differences - Equipment to Shelf. Electronic equipment heat loads are diffused from the component base over a section of the mounting surface. The equipment shelf radiation fin characteristics were evaluated for a unit area of $.093m^2(1ft^2)$. Table 2-20 lists the important fin parameters and the resulting temperature differences between the equipment base and the fin extremity. Aluminum substrates were assumed.

Table 2-20. Equipment Shelf Radiation Fin Characteristics (Unit Areas $.093m^2$)

Item	Load w.	Net Radiated Flux w/cm^2	Fin Length cm	Fin Thickness mm	Temperature Difference °C
1. MDR RCVR	8.2	0.0086	7.6	0.635	2.3
2. LDR RCVR (on edge)	9.1	0.0096	15.3	0.635	9.4
3. LDR RCVR (on base)	9.1	0.0096	7.6	0.635	2.6
4. MDR XMTR (louver panel)	56.0	0.0420	15.3	5.00	5.0

Results. The equipment temperature is obtained from the results of the previous sections. The differences are accumulated and summed to the 30°C average radiator temperature. The calculated temperatures are presented in Table 2-21 and also shown in Figure 2-28 according to equipment layout.

Table 2-21. Quad 4 Equipment Temperatures

Item	Temperature Differences (°C) Radiation	Fin	Temperature (°C)
1. MDR RCVR	15.8	2.3	48.1
2. LDR RCVR	16.9	2.6	49.5
3. MDR XMTR	-	5.0	35.0
4. T&C Logic	18.0	2.0	50.0

2.3.3.5 Conclusion

The results of the thermal analysis of the uprated design indicate that equipment temperature limit requirements are satisfied under combined worst case conditions of environmental and operational heat loads. Further analysis should be made to fully assess the effects of the increased radiator size during the transfer orbit. Preliminary estimates indicate modest make up heating requirements are well within the availability from the folded arrays. No additional requirements are found for the eclipse mode. The assumptions used in this analysis are conservative since they are based on worst case conditions and no leakage.

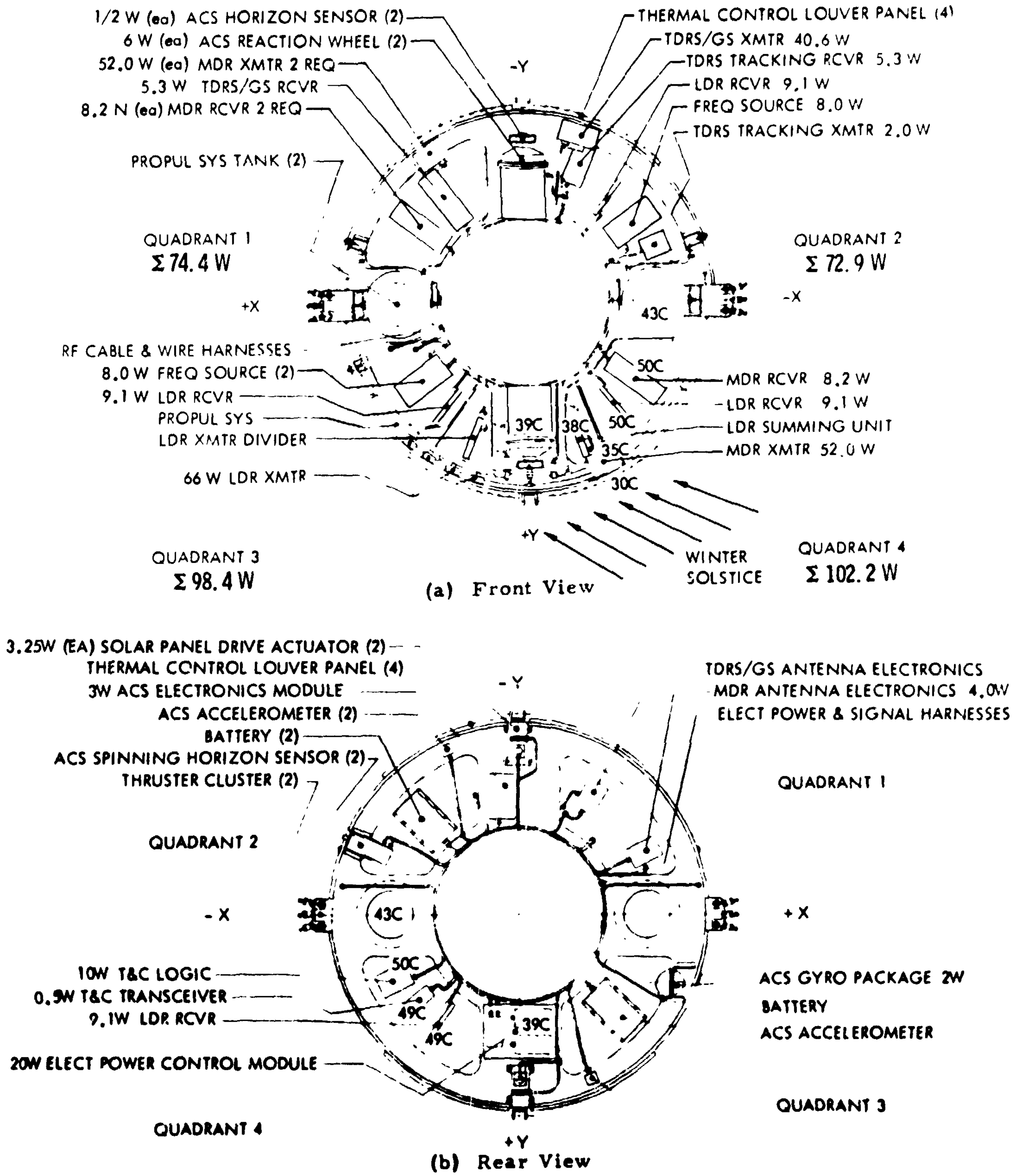


Figure 2-28. Thermal Control of Equipment Shelf

2.4 RELIABILITY

The reliability analysis performed for the baseline TDRS and reported in the "Final Report of the TDRS System Configuration and Tradeoff Study" (SD 72-SA-0133-4) is applicable to the uprated TDRS configuration with the exception of the communication subsystem. Therefore, only the reliability analysis of the communication subsystem, considering the differences between the baseline and the uprated designs, are discussed in this report. Overall changes in system reliability also are included.

The relationship of system versus satellite reliability is shown in Figure 2-29 which includes the effect of the original number of satellites purchased. In developing the curves a booster reliability of 0.95 and an apogee motor reliability of 0.98 were used. The curves show the probability of mission success where mission success is defined as follows.

- LDR Forward Link: Ability to transmit with one of four transmitter V/H channels
- LDR Return Link: Ability to receive with one of four receiver H and one of four receiver V channels
- MDR/HDR Transponder: Ability to service two MDR or one MDR and one HDR users simultaneously
- TDRS/GS Transponder: Ability to transmit and receive all data to the ground with 100-percent duty cycle and 17.5 dB margin
- Frequency Source: Supply discrete frequencies for master oscillators with 100-percent duty cycle
- Location Transponder: Ability to transmit and receive for a total of 4000 hours during 5-year mission (2 hr/day = 3650 hr).
- Order Wire: Ability to receive Shuttle orders for a total of 16,800 hours during 5-year mission (100 Shuttle flights of 7-day average duration = 16,800 hr) 100-percent duty cycle while Shuttle is in orbit
- Subsystems: Provide spacecraft support for 5 years; reduced forward link capability is permitted during eclipse

Based on a predicted reliability of 0.898 for the communication subsystem, the satellite reliability is calculated as 0.800. Table 2-22 shows a comparison of reliability assessments for the baseline and uprated satellite designs.

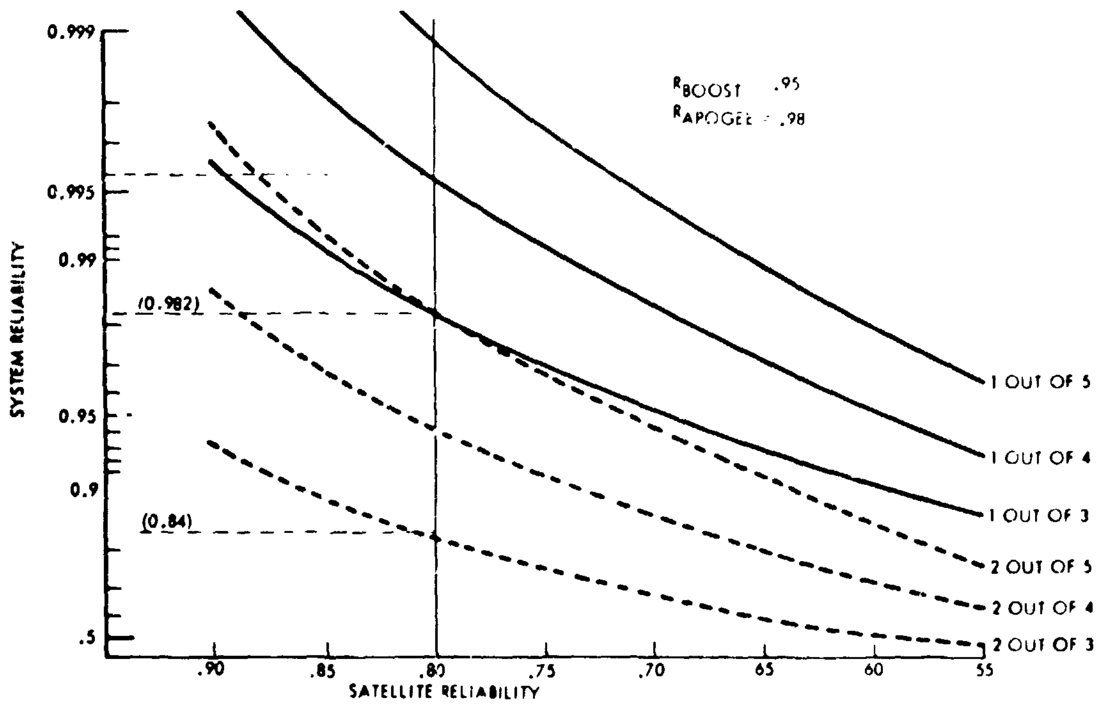


Figure 2-29. System Reliability Versus Satellite Reliability

Table 2-22. Preliminary Subsystem Predictions

Subsystem	Baseline	Uprated
Tracking, telemetry and command	0.966	0.966
Communications	0.904	0.898
Structure and mechanisms	0.999	0.999
Attitude control	0.962	0.962
Auxiliary propulsion	0.998	0.998
Electrical power	0.962	0.962
Thermal control	0.999	0.999
Total satellite	0.805	0.800

The reliability calculations for the LDR receiver are based on the definition for mission success where the operation of any one vertical plus any one horizontal channel (i.e., one out of four receivers) are adequate to service 20 LDR users. If it is determined that these conditions are inadequate, then the following numbers for the LDR receiver assembly are applicable:

1 out of 4 (present definition of success)	=	.99992
2 out of 4 required	=	.99586
3 out of 4 required	=	.92907
4 out of 4 required	=	.50407

If it is necessary to operate either three or four receivers (last two cases shown above), NR recommends the addition of a redundant channel to each one of the LDR receivers. In case of either vertical or horizontal channel failure in a given receiver, the redundant unit would be switched to take its place. This concept increases the overall LDR receiver reliability from .92907 to .99787 and from .50407 to .92577, respectively, with a weight increase of approximately 2 kg.

As can be seen, the differences in satellite reliability are negligible a and have no effect on the calculations for probability of mission success as presented in SD 72-SA-0133.

2.4.1 Communication Subsystem Reliability Analysis

The communication subsystem consists of the LDR transponder, the MDR/HDR transponder, the TDRS/GS transponder, a frequency source, a location transponder, and the TDRS antennas. Figure 2-30 shows the reliability logic diagram of the overall subsystem, where the shaded areas indicate deletions from the baseline design.

The redesign of the LDR transponder is caused by changing the mode of operation from electronically steered to a fixed field of view.

The MDR transponder was redesigned to accommodate HDR traffic. Improvements in the MDR receiver increased the overall transponder reliability.

The TDRS/GS transponder was completely redesigned. Due to additional power requirements for HDR and the 17.5 dB rain margin, the solid-state power amplifiers in the baseline design were replaced with redundant TWT amplifiers. The low MTBF of the TWT's led to a design where two tubes are hard-wired to each amplifier. A command turns on the filament voltage of one of the two redundant tubes, thus increasing considerably the overall transponder reliability as shown in Figure 2-31.

Redundant receivers operating at a different frequency than the tracking receiver were added to the location transponder to handle the order wire exclusively. The overall function was relabeled "TDRS Tracking/Order Wire Transceiver." The reliability of this function was calculated as 0.9955 by changing the total tracking time to 4000 hours and assuming 16,800 hours operation for the order wire receiver. The times used for the reliability calculations are based on a maximum of two hours per day tracking time and 100 Shuttle flights of 7-day duration each, which is a conservative estimate for the 1977 to 1982 period. (Current mission models show approximately 60 flights with an average stay of 2 to 4 days.)

Since there is no requirement for the Ku beacon in the uprated configuration, it was deleted from the design.

The reliabilities of the major subsystem components are indicated in Figure 2-30. The overall subsystem reliability is 0.8978. The calculations include a 10-percent contingency applied to the failure rate summations for each component to allow for any additional parts required during more detailed design definition.

The updated FMEA for the communication system is shown in Table 2-23. The results revealed no single failure points other than the components listed in the reliability logic diagrams as "common." The "common" circuits consist of passive networks like power or voltage dividers which are highly reliable. Since the probability of failure is negligible, duplication of these circuits is unnecessary.

2.4.2 Reliability Analysis of Remaining Subsystems

The uprated configuration includes a few changes from the baseline design in the remaining subsystems as stated in the spacecraft design section. An assessment of the changes and their impact on reliability was made and it was concluded that the effect is negligible and can be disregarded for the purpose of the prediction analysis.

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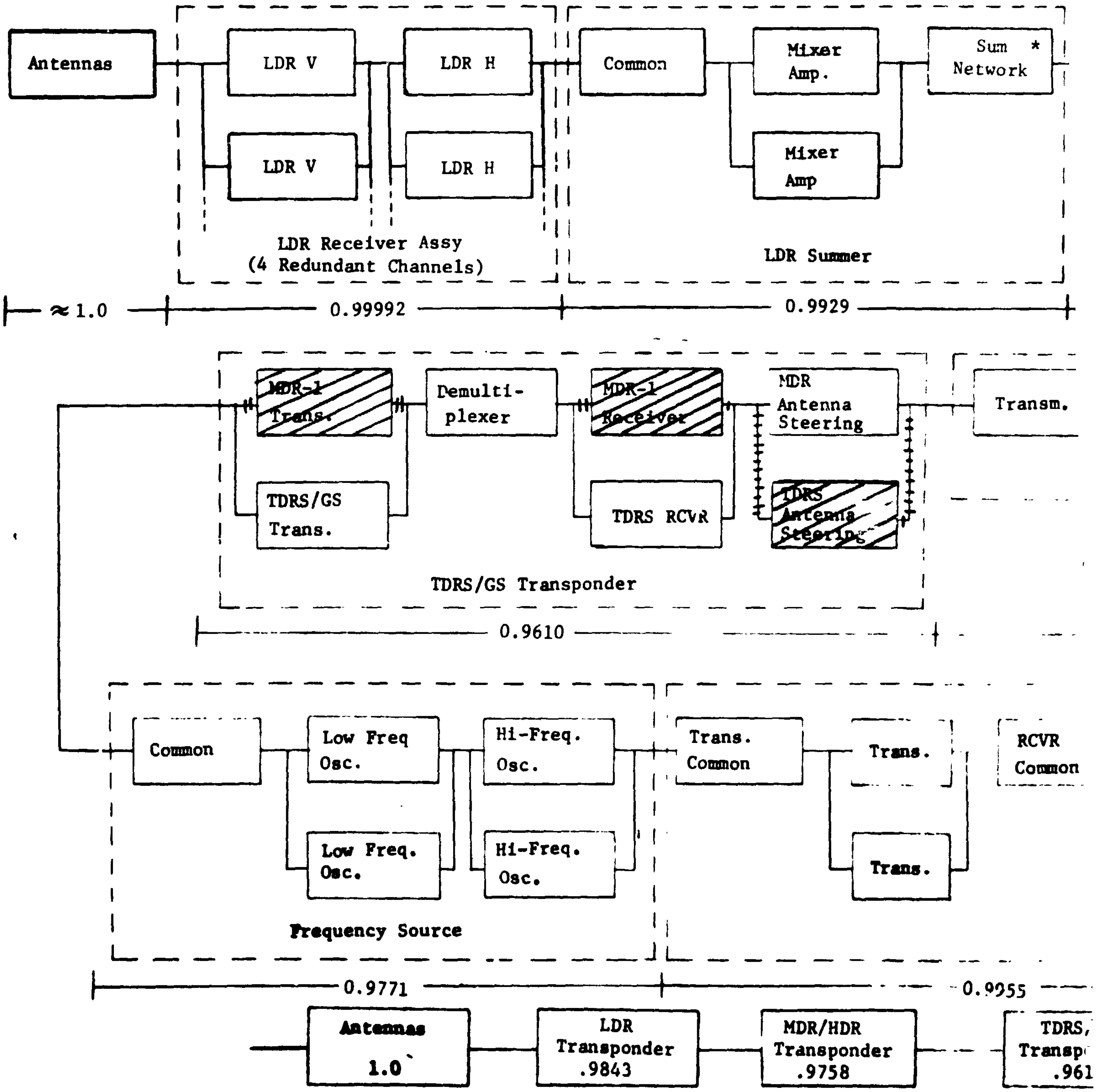
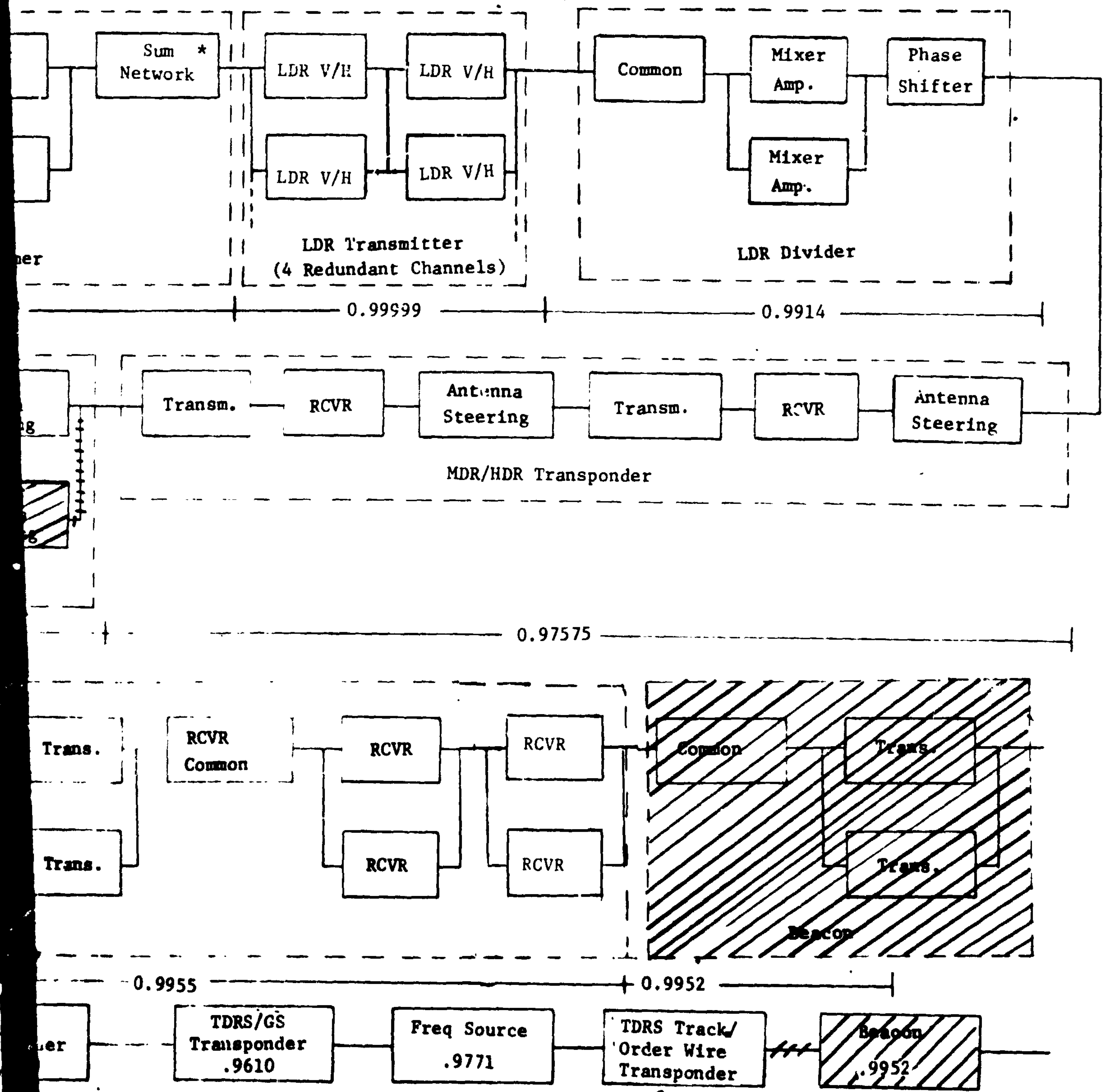


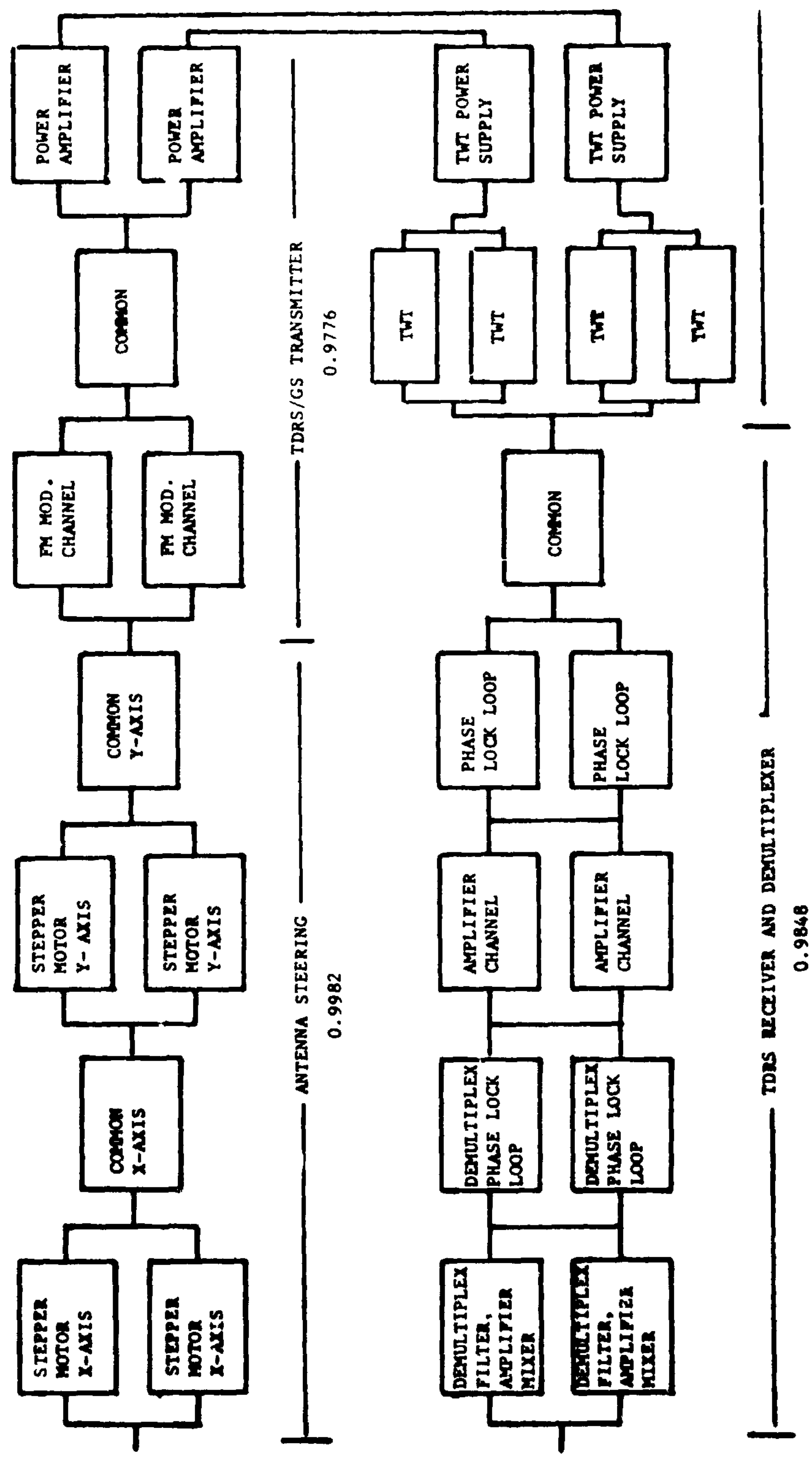
$R_{Total} = R_{ANT} \times R_{LDRTR.} \times R_{MDRTR} \times R_{TDRSTR.} \times R_{FS} \times R_{L}$

Notes: *Included with LDR receiver, transmitter respectively.
Shaded boxes indicate deletion from baseline configuration



$\ldots_{TRSTR.} \times R_{FS} \times R_{LT} \times R_B = 0.8978$

Figure 2-30. Reliability Diagram of Communications Subsystem



$R_{TOTAL} = R_{ANT.STEER.} \times R_{EMIT} \times R_{RECEIV} = 0.9610$

Figure 2-31. TDRS/GS Transponder Reliability Logic Diagram

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Table 2-23. Failure Mode Effects Analysis

PROGRAM: TDRS
SUBSYSTEM: COMMUNICATION

CRITICALITY I: SINGLE FAILURE PRECLUDES MISSION SUCCESS
CRITICALITY II: ALL OTHER FAILURES

ITEM IDENTIFICATION/ QUANTITY	FUNCTION	FAILURE MODE	CRIT. CATE-GORY	FAILURE EFFECT	ALTERNATE MEANS OF OPERATION	REMARKS
LDR TRANSPONDER						
LDR RECEIVER/SUMMER/8	RECEIVES LOW DATA RATE SIGNALS FROM USER SATELLITES	LOSS OF FUNCTION	II	SLIGHT SIGNAL DEGRADATION	USE REMAINING CIRCUITS	
LDR TRANSMITTER/ DIVIDER/4	TRANSMITS LOW DATA RATE COMMANDS TO USER SATELLITES	LOSS OF FUNCTION	II	NONE	USE REMAINING CIRCUITS QUADRUPLE REDUNDANCY	
RF SWITCH (COMMON)/1	SWITCHES TO REDUNDANT CIRCUITS IN CASE OF MALFUNCTION	FAILS TO TRANSFER	II	NONE	USE REDUNDANT PATHS	ALL RF SWITCHES WILL EITHER BE REDUNDANT OR SWITCHES WITH REDUNDANT COILS WILL BE UTILIZED
POWER DIVIDER (COMMON) 9	PASSIVE DIVISION OF POWER	LOSS OF FUNCTION	I*	LOSS OF SIGNAL	NONE	
MDR/HDR TRANSPONDER						
MDR/HDR TRANSMITTER/2	TRANSMITS MEDIUM AND HIGH DATA RATE COMMANDS TO USER SATELLITES	LOSS OF FUNCTION	II	CAN COMMUNICATE WITH ONLY ONE USER AT ANY GIVEN TIME	USE REDUNDANT UNIT	WITH THE EXCEPTION OF TWO POWER DIVIDERS ALL PARTS WITHIN THE MDR TRANSMITTER ARE COMPLETELY REDUNDANT. TWO FAILURES HAVE TO OCCUR BEFORE ONE LINK CAPABILITY IS LOST
MDR/HDR RECEIVER/2	RECEIVES MEDIUM AND HIGH DATA RATE SIGNALS FROM USER SATELLITES	LOSS OF FUNCTION	II	CAN COMMUNICATE WITH ONLY ONE USER AT ANY GIVEN TIME	USE REDUNDANT UNIT	WITH THE EXCEPTION OF ONE DIPLEXER EACH IN THE Ku AND S BAND RECEIVER ALL PARTS ARE REDUNDANT. TWO FAILURES HAVE TO OCCUR BEFORE ONE LINK CAPABILITY IS LOST
MDR MOTOR (ANTENNA DRIVE)/8	DRIVES GIMBAL MECHANISM	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	USE REDUNDANT UNIT	
GIMBAL MECHANISM; 2 AXIS (COMMON)/1	GIMBALS MDR/HDR ANTENNA TO POINT AT USER SATELLITES	LOSS OF FUNCTION	I*	LOSS OF ONE MDR/HDR LINK	USE THE SECOND MDR/HDR LINK	ALL PREDOMINANT FAILURES HAVE BEEN PROTECTED BY REDUNDANCY COMM. SS. WILL DEGRADE TO ONE LINK CAPABILITY.
RF SWITCH (COMMON)/2	SWITCHES TO REDUNDANT CIRCUITS IN CASE OF MALFUNCTION	FAILS TO TRANSFER	II	NONE	USE REDUNDANT PATHS	ALL RF SWITCHES WILL EITHER BE REDUNDANT OR SWITCHES WITH REDUNDANT COILS WILL BE UTILIZED.
POWER DIVIDER/HYBRID COMMON/18	PASSIVE DIVISION OF POWER	LOSS OF FUNCTION	I*	LOSS OF ONE LINK	USE SECOND LINK	COMM. SS. WILL DEGRADE TO ONE LINK CAPABILITY.
DIPLEXER (COMMON)/4	FILTERS INCOMING SIGNALS	LOSS OF FUNCTION	I*	DEGRADED LINK CAPABILITY	USE SECOND LINK	
TDRS/GS TRANSPONDER						
TDRS/GS TRANSMITTER/ 1	TRANSMITS DATA TO GROUND	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	SWITCH TO EITHER MDR/HDR TRANSMITTER FOR BACKUP	
TDRS/GS RECEIVER/1	RECEIVES DATA FROM GROUND	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	SWITCH TO EITHER MDR/HDR RECEIVER FOR BACKUP	
RF SWITCH (COMMON)/6	SWITCHES TO REDUNDANT CIRCUITS IN CASE OF MALFUNCTION	FAILS TO TRANSFER	II	NONE	USE REDUNDANT PATHS	ALL RF SWITCHES WILL EITHER BE REDUNDANT OR SWITCHES WITH REDUNDANT COILS WILL BE UTILIZED.
TDRS MOTOR (ANTENNA DRIVE)/2	DRIVES GIMBAL MECHANISM	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	USE REDUNDANT UNIT	

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RF SWITCH (COMMON)/2	SWITCHES TO REDUNDANT CIRCUITS IN CASE OF MALFUNCTION	FAILS TO TRANSFER	II	NONE	USE REDUNDANT PATHS	ALL RF SWITCHES WILL EITHER BE REDUNDANT OR SWITCHES WITH REDUNDANT COILS WILL BE UTILIZED.
POWER DIVIDER/HYBRID COMMON/18	PASSIVE DIVISION OF POWER	LOSS OF FUNCTION	I*	LOSS OF ONE LINK	USE SECOND LINK	COMM. SS. WILL DEGRADE TO ONE LINK CAPABILITY.
DIPLEXER (COMMON)/4	FILTERS INCOMING SIGNALS	LOSS OF FUNCTION	I*	DEGRADED LINK CAPABILITY	USE SECOND LINK	
TDRS/GS TRANSPONDER						
TDRS/GS TRANSMITTER/1	TRANSMITS DATA TO GROUND	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	SWITCH TO EITHER MDR/HDR TRANSMITTER FOR BACKUP	
TDRS/GS RECEIVER/1	RECEIVES DATA FROM GROUND	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	SWITCH TO EITHER MDR/HDR RECEIVER FOR BACKUP	
RF SWITCH (COMMON)/6	SWITCHES TO REDUNDANT CIRCUITS IN CASE OF MALFUNCTION	FAILS TO TRANSFER	II	NONE	USE REDUNDANT PATHS	ALL RF SWITCHES WILL EITHER BE REDUNDANT OR SWITCHES WITH REDUNDANT COILS WILL BE UTILIZED.
TDRS MOTOR (ANTENNA DRIVE)/2	DRIVES GIMBAL MECHANISM	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	USE REDUNDANT UNIT	
GIMBAL MECHANISM; 1 AXIS (COMMON)/1	POINTS TDRS ANTENNA TO EARTH	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	SWITCH TO MDR/HDR DISH FOR BACKUP	THE ANTENNA POINTING OPERATION IS PERFORMED ONLY ONCE DURING THE 5 YEAR MISSION.
FREQUENCY SOURCE						
LOW FREQUENCY SOURCE /2	PROVIDES LOCAL OSCILLATOR REFERENCE SIGNALS	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	USE REDUNDANT UNIT	
HIGH FREQUENCY SOURCE /2	PROVIDES LOCAL OSCILLATOR REFERENCE SIGNALS	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	USE REDUNDANT UNIT	
RF SWITCH (COMMON)/13	SWITCHES TO REDUNDANT CIRCUITS IN CASE OF MALFUNCTION	FAILS TO TRANSFER	II	NONE	USE REDUNDANT PATHS	ALL RF SWITCHES WILL EITHER BE REDUNDANT OR SWITCHES WITH REDUNDANT COILS WILL BE UTILIZED.
LOCATION TRANSPONDER						
LOCATION TRANSMITTER/2	TRANSMITS ACQUISITION & TRACKING BEACON SIGNALS	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	USE REDUNDANT UNIT	
LOCATION RECEIVER/2	RECEIVES TRACKING BEACON & ORDER WIRE	LOSS OF FUNCTION	II	LOSS OF REDUNDANCY	USE REDUNDANT UNIT	
RF SWITCH (COMMON)/3	SWITCHES TO REDUNDANT CIRCUITS IN CASE OF MALFUNCTION	FAILS TO TRANSFER	II	NONE	USE REDUNDANT PATHS	ALL RF SWITCHES WILL EITHER BE REDUNDANT OR SWITCHES WITH REDUNDANT COILS WILL BE UTILIZED.
DIPLEXER (COMMON)/1	FILTER RF SIGNALS	LOSS OF FUNCTION	II	LOSS IN TRACKING ACCURACY	USE SINGLE CHANNEL RANGING WITH EITHER Ku BAND OR VHF	

*A DETAILED JUSTIFICATION FOR RETAINING THESE SINGLE POINT FAILURES IN THE DESIGN IS PRESENTED IN SECTION 10.4 OF SD 72-SA-0133

2.5 ALTERNATE DESIGN - UPRATED TDRS WITH S-BAND ARRAY

2.5.1 Spacecraft Design

A brief study was performed on a conceptual arrangement of the uprated TDRS with an S-band array replacing the LDR UHF/VHF array for servicing LDR and MDR users.

Figure 2-32 illustrates the concept with the S-band array body mounted on the front of the spacecraft and the HDR/MDR and TDRS/GS antennas grouped around the body similar to the previously described uprated TDRS. The HDR/MDR antennas were brought closer to the spacecraft body with the elimination of the LDR UHF/VHF array with its large ground planes, reducing support strut length and weight. The solar panels were also located closer to the spacecraft centerling because of the lowering of the solar shadow lines from the HDR/MDR antennas. The solar panel support struts were also shortened, reducing their weight.

The TDRS/GS antenna diameter was increased from 1.8M to 2.0M to minimize link power. Eliminating the LDR UHF-VHF elements permitted this increased diameter without affecting the packaging.

The spacecraft body shape was modified to a flat front end which provides the mounting surface for the S-band array elements. The overall length of the body was slightly increased to provide adequate clearance between the front face of the body and the apogee motor. The additional weight of the modified body shape was more than compensated for by eliminating the LDR UHF/VHF array support fittings.

The S-band array consists of 31 single helix elements, 29 receiving and 2 transmitting. Each element is a thin wall dielectric material tube supporting a conductive material tape wound in a helix on the tube outer surface. A ground plane of larger diameter is located at the bottom of each tube. Directly behind and integral with the element, the receiver unit is mounted to the inner wall of the spacecraft front face. Clearance holes are cut in the face to install the elements from the rear with screws through lugs on the perimeter of the receivers securing the unit to the body. Figure 2-32 illustrates details of the element and its installation.

The transmitter elements are identical to the receivers except that the transmitter units are remotely located from the elements. These higher powered heat producing units are located on the equipment mounting surfaces of the upper and lower thermal control louver assemblies along with the transmitters of the HDR/MDR and TDRS/GS systems. To provide minimum length cabling between transmitter elements and transmitter units, the 2 transmitter elements are located at the top and bottom of the array as shown in Figure 2-32.

This S-band array design and method of support and installation results in a very light weight system necessary to maintain the payload contingency with the Delta launch capabilities without sacrificing performance in the other communications systems. The estimated weight for the S-band array system is shown below.

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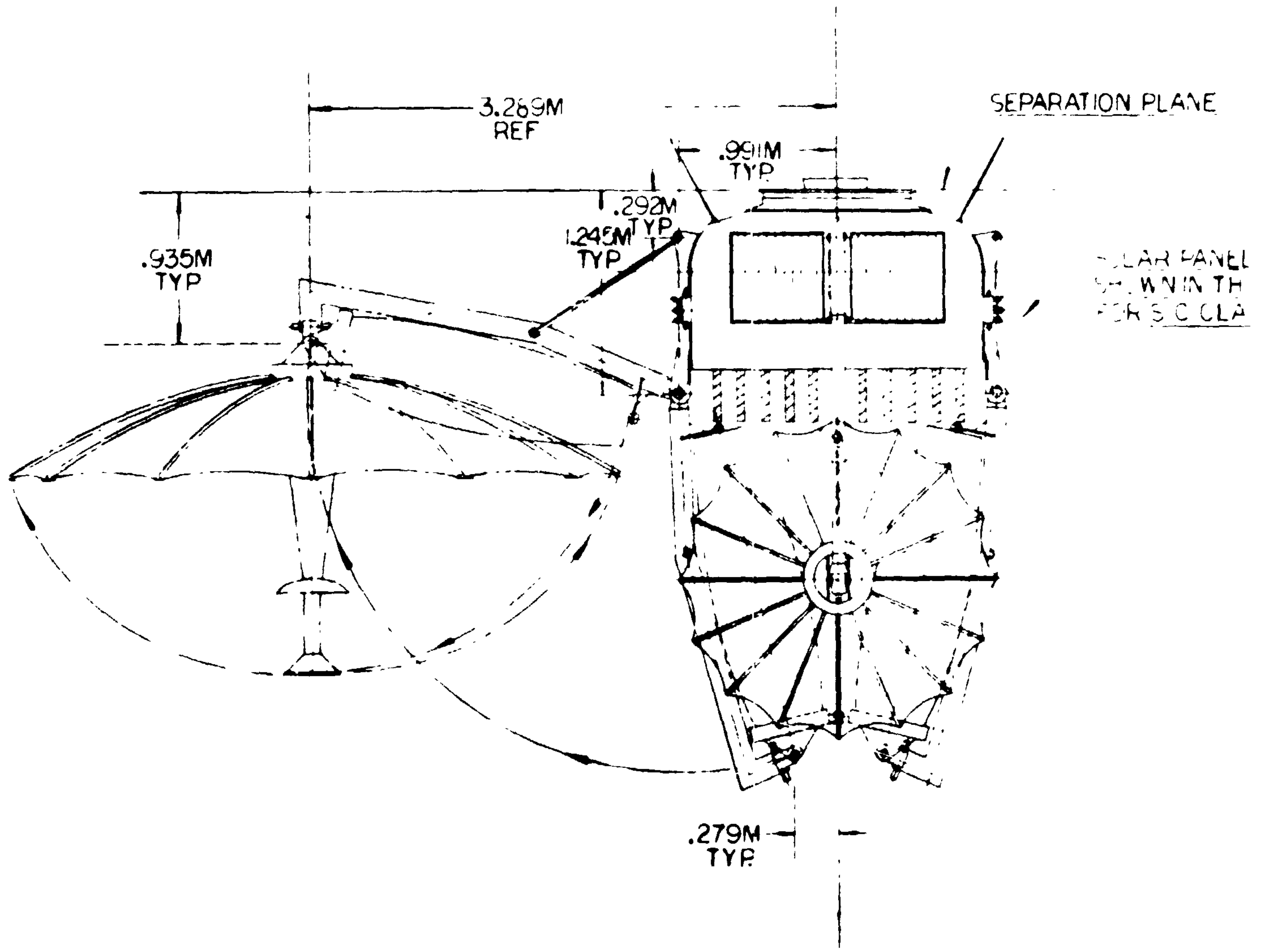

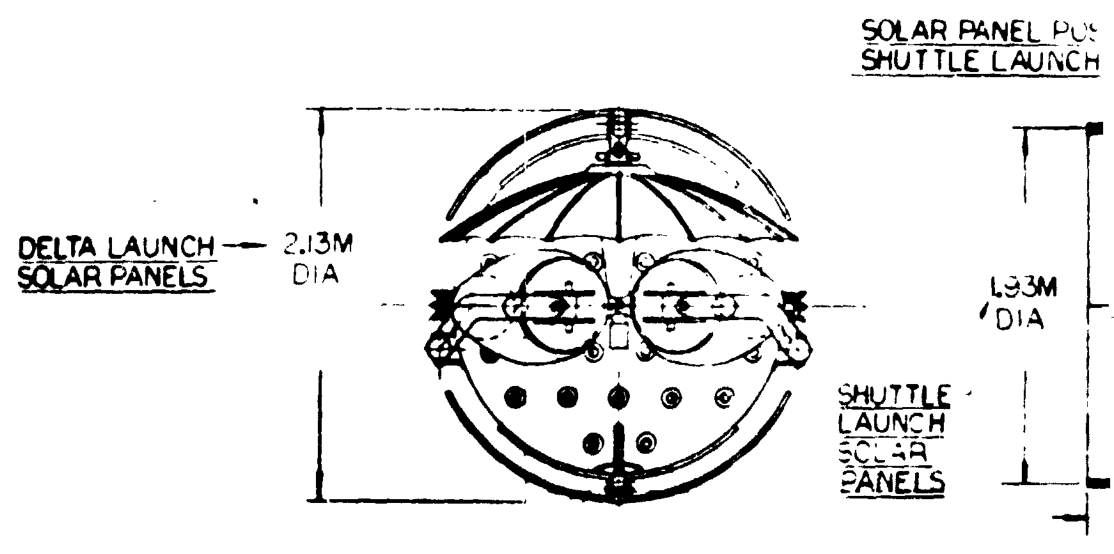

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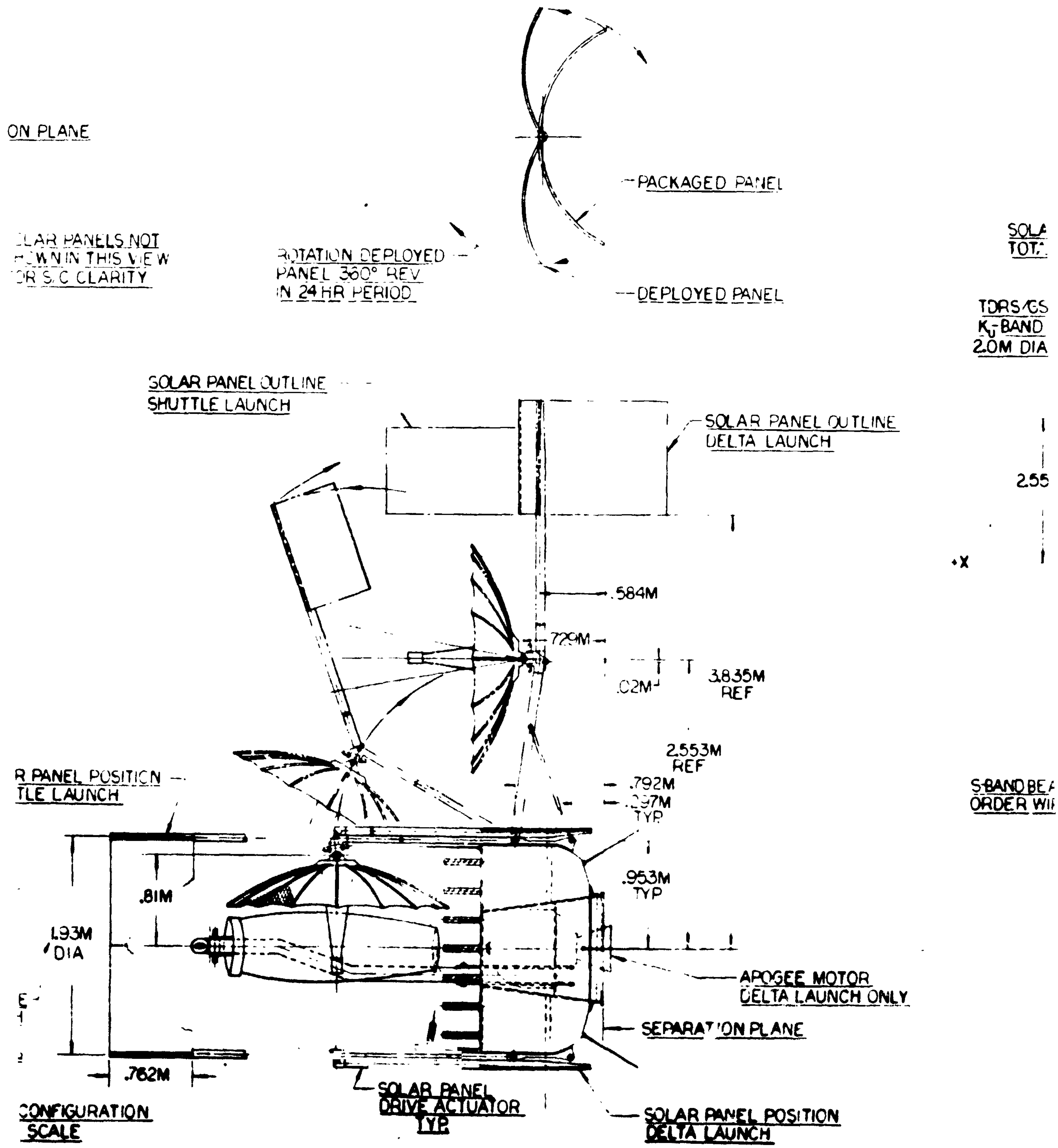
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TDRS/GS
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SHUTTLE LAUNCH
SOLAR PANEL OUTLINE
DELTA LAUNCH
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.792M
.297M
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S-BAND BEA
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.953M
TYP
.81M
1.93M
DIA
APOGEE MOTOR
DELTA LAUNCH ONLY
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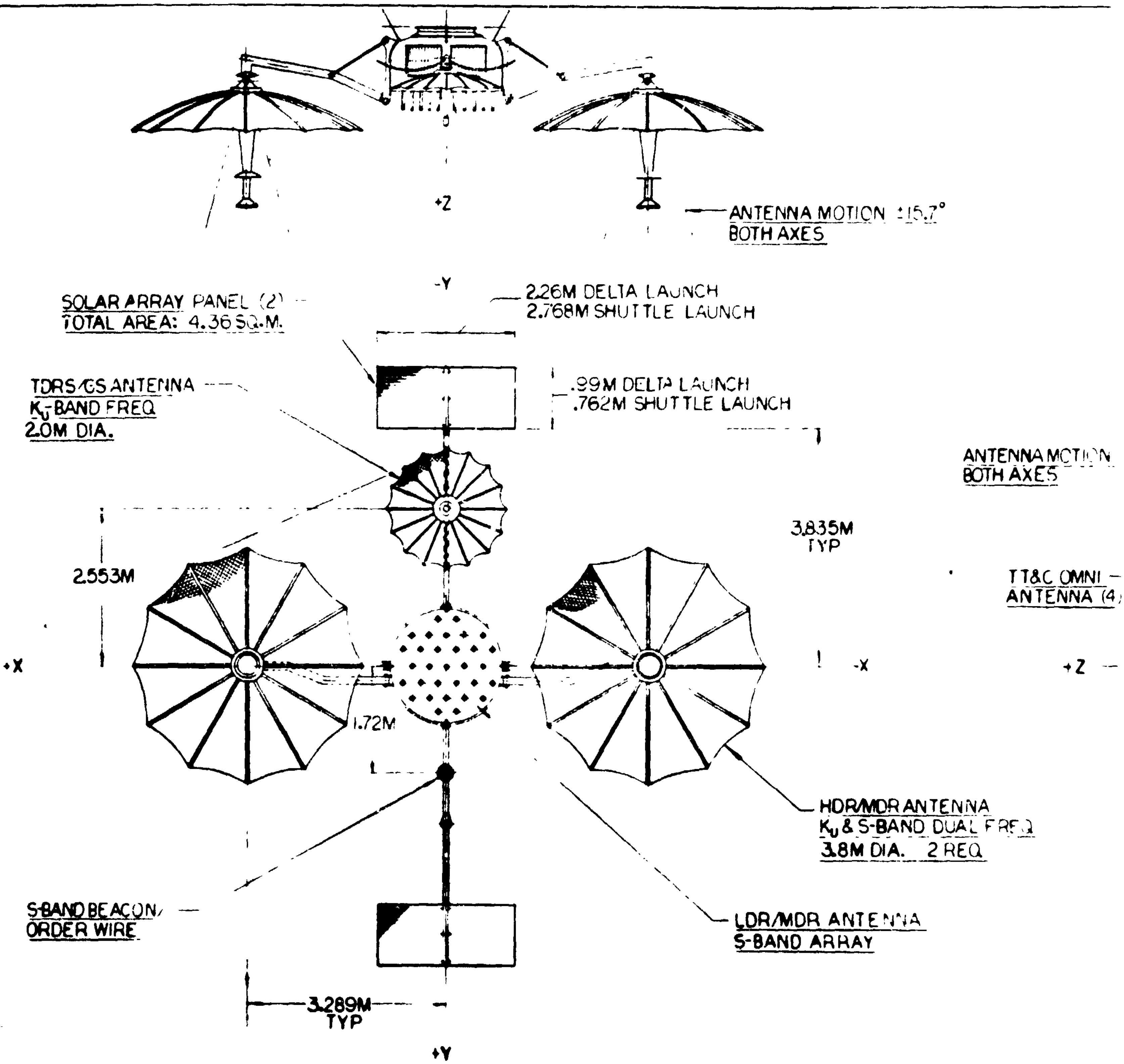
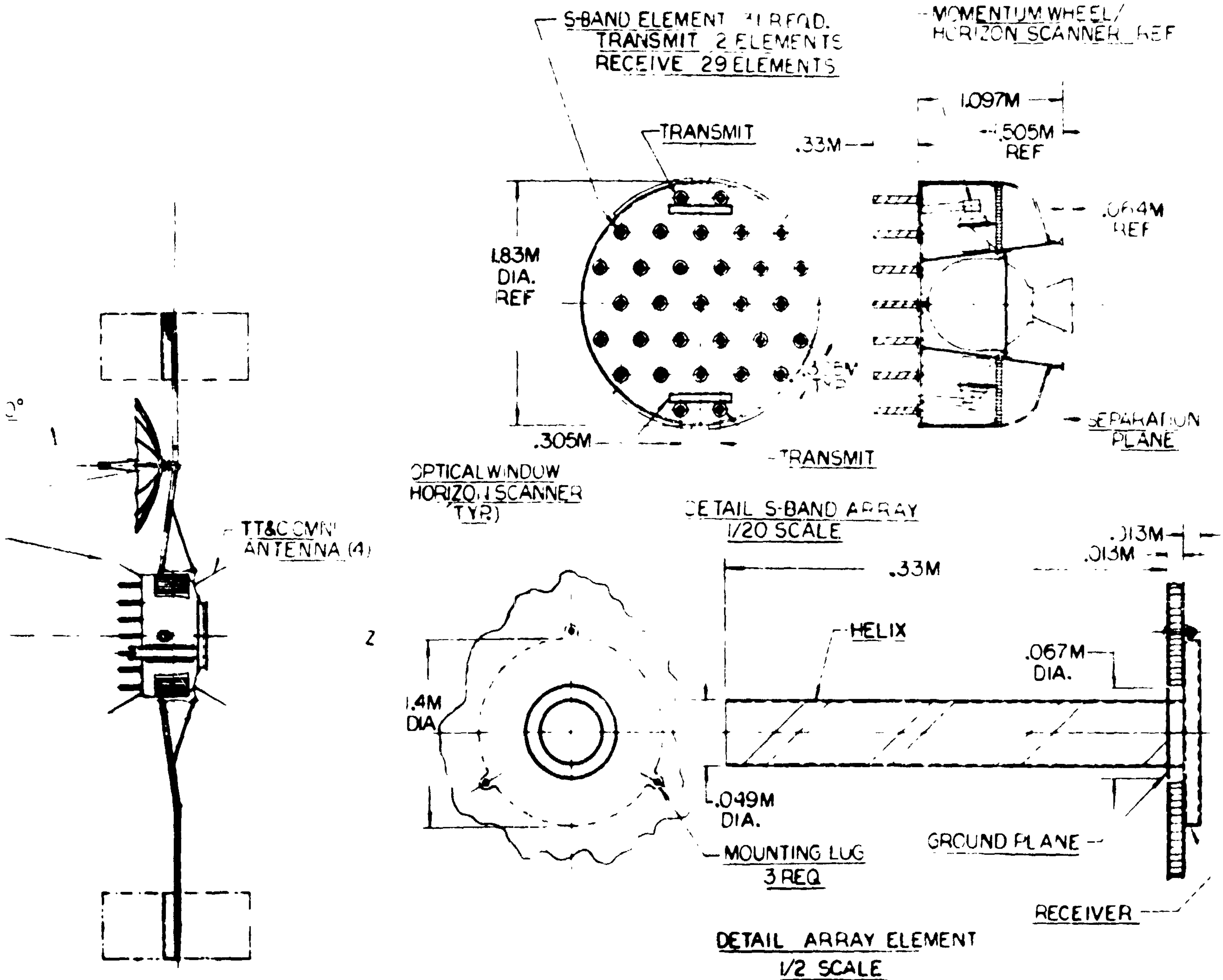


DEPLOYED CONFIGURATION

1/40 SCALE



NOTES

FOR DELTA 2914 LAUNCH CONFIGURATION
SIMILAR TO REF DWG 198-55

FOR SHUTTLE/AGENA-TUG LAUNCH CONFIGURATION
SIMILAR TO REF DWG 198-57

Figure 2-32. TDRS Uprated Configuration with S-Band Array

S-Band Array System Weight

	Kg
Element (31)	1.32
Receivers (29)	13.15
Transmitters (2)	6.40
FDM Module	0.64
Local Freq. Reference	6.19
Total	28.70

Packaging of this concept for the Delta launch is similar to the uprated TDRS. Without the LDR UHF/VHF elements the packaging density is reduced and the 2.0 M TDRS/GS antenna stows as shown in Figure 2-23. Deployment is also similar to the uprated TDRS but simpler by the elimination of the LDR UHF/VHF elements with their deployment and extension. The body mounted S-band array is a fixed structure and require no deployment.

The weight of the uprated TDRS with the S-band array changes in several areas in both the communications systems and other subsystems as a result of the changes in configuration.

In the communications subsystems, the HDR/MDR system was reduced 1.09 Kg by shortening the antenna support struts. The larger TDRS/GS antenna is equivalent to the uprated TDRS/GS antenna because the rib and hub weights were reduced to compensate for the increase of .076 M in rib length. The TT&C subsystem increases 1.04 Kg because of the additional channels required to monitor and control the S-band array. The S-band array increased 6.80 Kg over the previous LDR UHF/VHF system because of the great increase in number of receivers and elements. The communication system therefore increased by 6.75 Kg over the uprated TDRS communication system.

The solar array panel was reduced 1.61 Kg by shortening the support struts and incorporating a light weight magnesium grid substrate design developed by Electro-Optics, Santa Barbara, California.

The spacecraft body structure was reduced 3.23 Kg with the body shape modification weight increase negated by the elimination of the LDR UHF/VHF support fittings on the body and a reanalysis of component detail and hardware definition.

The thermal control system was reduced 2.11 Kg by incorporating a foam louver blade design developed and spacecraft flight proven by RCA/Astro-Electronics Division, Princeton, N.J.

The weight summary for this TDRS concept is given in Table 2-24.

Table 2-24. Weight of Uprated TDRS With S-Band Array

	Weight (kg)
Communications	
Electronics	45.01
Antennas	85.2
Attitude stabilization and control	26.2
Electric power	42.2
Solar array	26.09
Structure	38.04
Thermal control	8.73
Auxiliary propulsion hardware	14.5
	285.97
Propellant + N_2 (1-65 deg, 20-day station change)	17.82
Total spacecraft	303.79
Delta 2914 vehicle	
Total spacecraft	303.79
Contingency	30.95
Allowable payload (Delta 2914 + CTS apogee motor)	334.8*
Empty apogee motor case	22.7
Initial on orbit	357.5
Burned-out insulation	3.6
Apogee motor propellant	312.1
Synchronous orbit injection	673.2
Transfer orbit propellant	2.7
Delta separation weight (27 deg transfer orbit)	675.9
*5 deg/day drift orbit	

As noted in the table, to maintain a contingency of 10% of the dry weight of the spacecraft, one station change was eliminated (leaving one station change of 65 degrees in 20 days).

Slight modification of the solar panel geometry and packaging as indicated on Figure 2-23 permits Shuttle/Agena launch of this concept similar to the uprated TDRS discussed in Paragraph 4.2.2. of this report. As noted, no apogee motor would be required as the Agena Tug carries the three TDRS spacecraft into synchronous orbit.

Weight changes resulting from the solar panel geometry rearrangement would be minimal and since the contingency for the uprated TDRS in Shuttle is very large, the effect is minimum.

2.5.2 Operations Limitations

As with the uprated TDRS, minor operations limitations must be imposed on the S-array design due to power requirement. The S-array limitations will be less severe during daylight but somewhat more severe during eclipse and for battery charge conditions.

Table 2-25 shows the electrical loads for the S-array configuration of 29 receivers and two transmitters. A comparison of this table with the equivalent one for the uprated TDRS shows the S-array uses 3 watts less than the UHF/VHF array at 30DB EIRP and 12.7 watts less on the TDRS/GS link due to a 10% increase in the diameter of this antenna. As there is no equivalent to a low power forward LDR in the S-array configuration, modes 4 and 5 are not applicable and the power available for battery charge in modes 9 and 10 is reduced, resulting in longer times for battery charge. If necessary, this time can be reduced by shutting off the forward S-array increasing the battery charge power to approximately 10 watts higher than on the uprated design.

There are no daylight limitations on the S-array concept except for mode 1 which has a negative margin at EOL solstice of 6 watts. This can readily be made up by battery augmentation for 46 hours before 60% DOD is reached. This condition occurs for approximately 30 days at EOL and assumes 17.5 dB rain margin is required with HDR to GS plus full forward voice power (i.e., not push-to-talk) for the continuous time period.

Eclipse power use is shown in Table 2-26. The basic operating modes assume all subsystems operating with the S-array on receive, antenna #1 transmitting S-data and receiving, antenna #2 receiving, and transmission from TDRS to GS on HDR (17.5dB margin) for Mode E-1, HDR (7.5dB margin) for Mode E-2, and MDR (17.5dB margin) for Mode E-3. An analysis is made to determine the allowable duty cycle of either forward S-array or forward S-voice on antenna #1. The forward S-array uses 47.5 watts and the S-voice uses 51W. For ease of analysis and presentation, these are assumed to be equal at 55 watts including losses.

Table 2-25. Electrical Load Chart (31-Element S-Array Configuration)

ITEM / MODE NO.			1	2	3	6	7	8	9*	10*
SUBSYSTEMS		(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)	(39.7)
Attitude Stabilization and Control		16.5								
Thermal Control		2.0								
Solar Panel Drive and EPS Control		10.7								
TT&C		10.5								
COMMUNICATIONS			(301.4)	(280.6)	(270.7)	(279.3)	(258.5)	(268.4)	(227.6)	(207.5)
S-ARRAY		122.0	122.0	122.0	122.0	122.0	122.0	122.0	122.0	122.0
MDR/HDR NO. 1	S-Data	27.2							27.2	27.2
	S-Voice/Data	78.2	78.2	78.2	78.2	78.2	78.2	78.2		
	Ku-Data	17.3								
	Ku-Video	48.0								
MDR/HDR NO. 2	S-Data	27.2		27.2			27.2		27.2	27.2
	Ku-Data	17.3			17.3					
	Ku-Video	48.0	48.0			48.0		48.0		
TDRS/GS	MDR (17.5 dB)	10.2						10.2		
	HDR (7.5 dB)	21.1				21.1	21.1			21.1
	HDR (17.5 dB)	43.2	43.2	43.2	43.2				43.2	
Frequency Source		8.0								
S-Band Track/Order Wire		2.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
SUBTOTAL			341.1	320.3	310.4	319.0	299.2	308.1	267.3	247.2
System Losses			40.0	40.0	40.0	40.0	40.0	40.0	40.0	40.0
TOTAL			381.1	360.3	350.4	359.0	339.2	348.1	307.3	287.2
EOL Power Available***										
Equinox		417 →								
Solstice		375 →								
Power Margin**										
Equinox			36	57	61	58	78	69	110	130
Solstice			-6	15	25	16	36	27	68	87

Modes 1 - 3, 9 are HDR rain margin = 17.5 dB; Modes 6, 7, 10 HDR rain margin = 7.5 dB; Mode 8 is MDR rain margin = 17.5 dB

* Reduced operations (i.e., no voice or video) for rapid battery charge if needed

** For battery charge, additional service, or margin. If minus, must be supplied by battery on duty cycle. If minus, or insufficient to charge batteries in reasonable time, some service (e.g., voice, video, forward S-array) must be reduced until batteries are fully charged.

*** Power at BOL = 487 watts, equinox
= 436 watts, solstice

Table 2-26. Eclipse Power Use (31 Element S-Array)

	Watts
Subsystems	40
Freq. Source and S-Band Track/Ord.Wire	10
	50
29 S-Array Receivers	42
FDM Module	1
Local Freq. Ref.	31
MDR/HDR Recvrs and Antenna Track (#1 & #2)	24
	148W
MDR/HDR #1 Data Transmit	15
	163W

TDRS/GS (Transmit + Receive)

E-3	MDR	(17.5dB)	one only	10.2W
E-2	HDR	(7.5dB)		21.1W
E-1	HDR	(17.5dB)		43.2W

Total Eclipse Power Requirements

E-3 163 + 10.2 = 173.2W x 1.10* = 190W
E-2 163 + 21.1 = 184.1W x 1.10 = 203
E-1 163 + 43.2 = 206.2W x 1.10 = 227

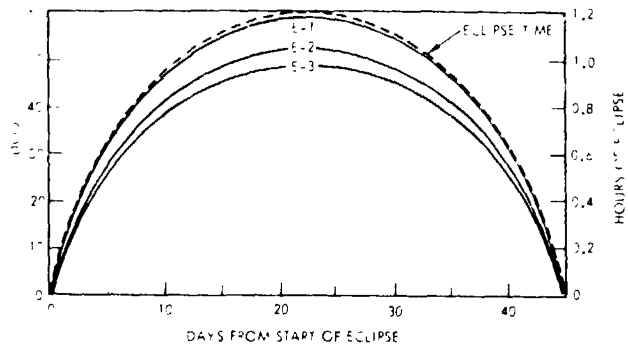
* Losses assumed to be 10%

Additional Items

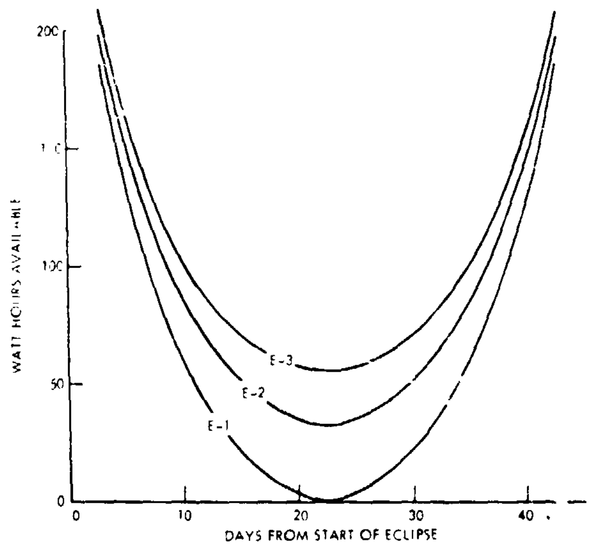
Forward S-Array = 47.5W
Forward S-Voice = 66W Δ = 51W
(Assume each one to be 55W incl. 10% for losses)

Figure 2-33(a) shows the battery DOD versus days from start of eclipse. Figure 2-33(b) shows the extra energy available in the battery (to 60% DOD) for forward S-array or S-voice. Figure 2-33(c) shows the time available for forward S-array and S-voice as a function of days from start of eclipse. The distance below the lines E-1, E-2 and E-3 are the times S-voice <u>or</u> forward S-array can be used. At maximum eclipse for HDR (17.5dB margin), no voice can be used for the 1.2 hours. For E-2, a 50% duty cycle is permissable and for MDR, voice cannot be used for only 0.2 hours. In occasional emergencies these restrictions can be exceeded resulting in higher than 60% battery DOD.

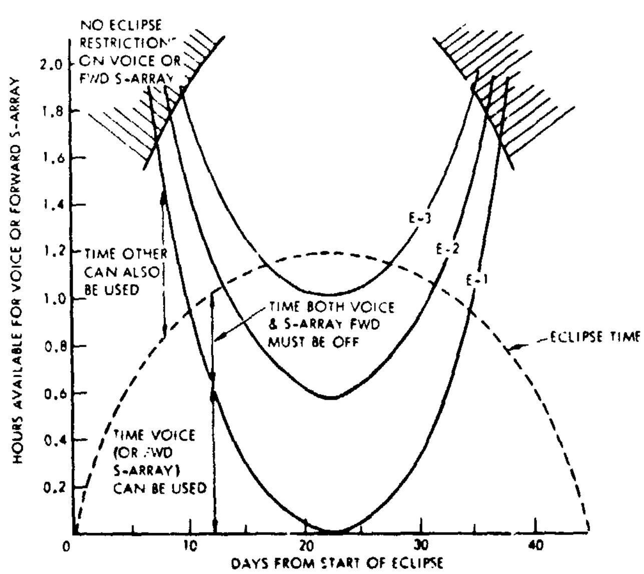
When the E-1, E-2 and E-3 curves are below the Eclipse Time curve, the distance between the curves indicates the time voice must be off. When they are above the Eclipse Time curve, the distance indicates the time forward



(a) Battery Depth of Discharge



(b) Extra Energy Available



(c) Time of Voice or Forward S-Array

Figure 2-33. Eclipse Operating Restrictions - S-Array Concept

S-array can also be on if voice is on during the entire eclipse. The crosshatched areas in the corners indicate the time when both forward S-array and S-voice can be on during the entire eclipse.

3.0 ATLAS-CENTAUR SPACECRAFT CONCEPT

A study was made of a TDRS concept to be launched on an Atlas-Centaur. A version with increased telecommunications capability with five 3.8 M antennas was conceived and is shown in this section. Only one such spacecraft can be within the shroud, packaged for each launch, even though there is an ample weight margin. This does not appear to be a cost-effective booster.

3.1 SYSTEM ENGINEERING

The geosynchronous orbit payload capability of the Atlas-Centaur launch vehicle is shown in Figure 3-1 as a function of final orbit inclination. The apogee kick motor was assumed to be the TE 364-4 with an inert weight of 78 kg and an I_{sp} of 284 seconds. The payload shown is above the weight of the empty apogee motor. The curve shows that for the 2-1/2 degrees final inclination orbit, the payload is 915 kg (2020 lb).

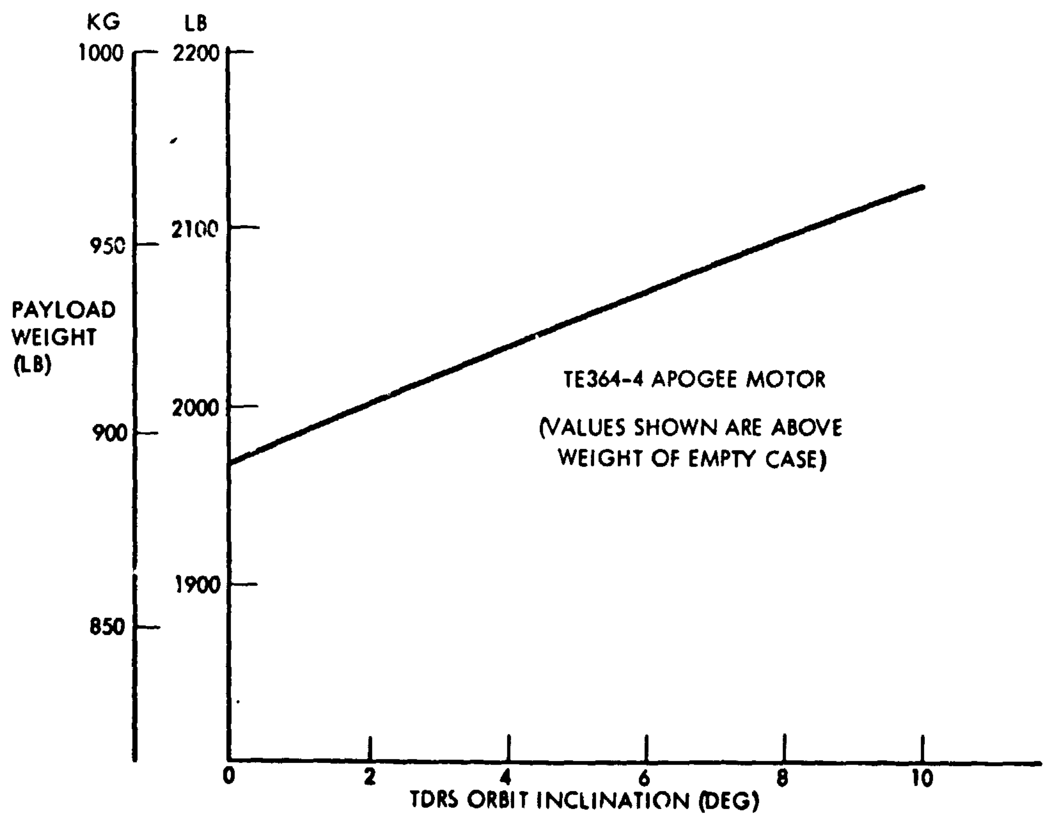


Figure 3-1. Geosynchronous Orbit Payload Capability (Atlas-Centaur)

The operations required for launch with the Atlas-Centaur will be essentially similar to those specified for the Delta. One added requirement will be the need to spin up the TDRS after separation from the Centaur. Unlike the Delta (which has a spin table to spin up its payload to 90-100 rpm) the Centaur spins up the payload to 1 rpm prior to separation.

3.2 SPACECRAFT DESIGN CONCEPT

With consideration of the much greater payload capabilities of the Atlas/Centaur, an effort was made to extend the capability of the TDRS in the size and number of antennas to a maximum concept that still could be packaged in the launch vehicle. Several attempts were made with many symmetrical and unsymmetrical arrangements of varying diameter parabolic antennas and the high performance Senior AGIPA array before the final concept was generated. In the choice of this concept, every effort was made to maintain simplicity by duplication of multiple identical units, to eliminate solar pressure variations by a symmetrical and balanced arrangement, and to maintain a light but feasible weight.

3.2.1 Spacecraft Design

The configuration in Figure 3-2 has five 3.8 M diameter parabolic antennas equally spaced between the five elements of the Senior AGIPA UHF disc-on-rod and VHF backfire arrays. The 3.8 M antennas are located to provide clearance of beam widths with the LDR arrays.

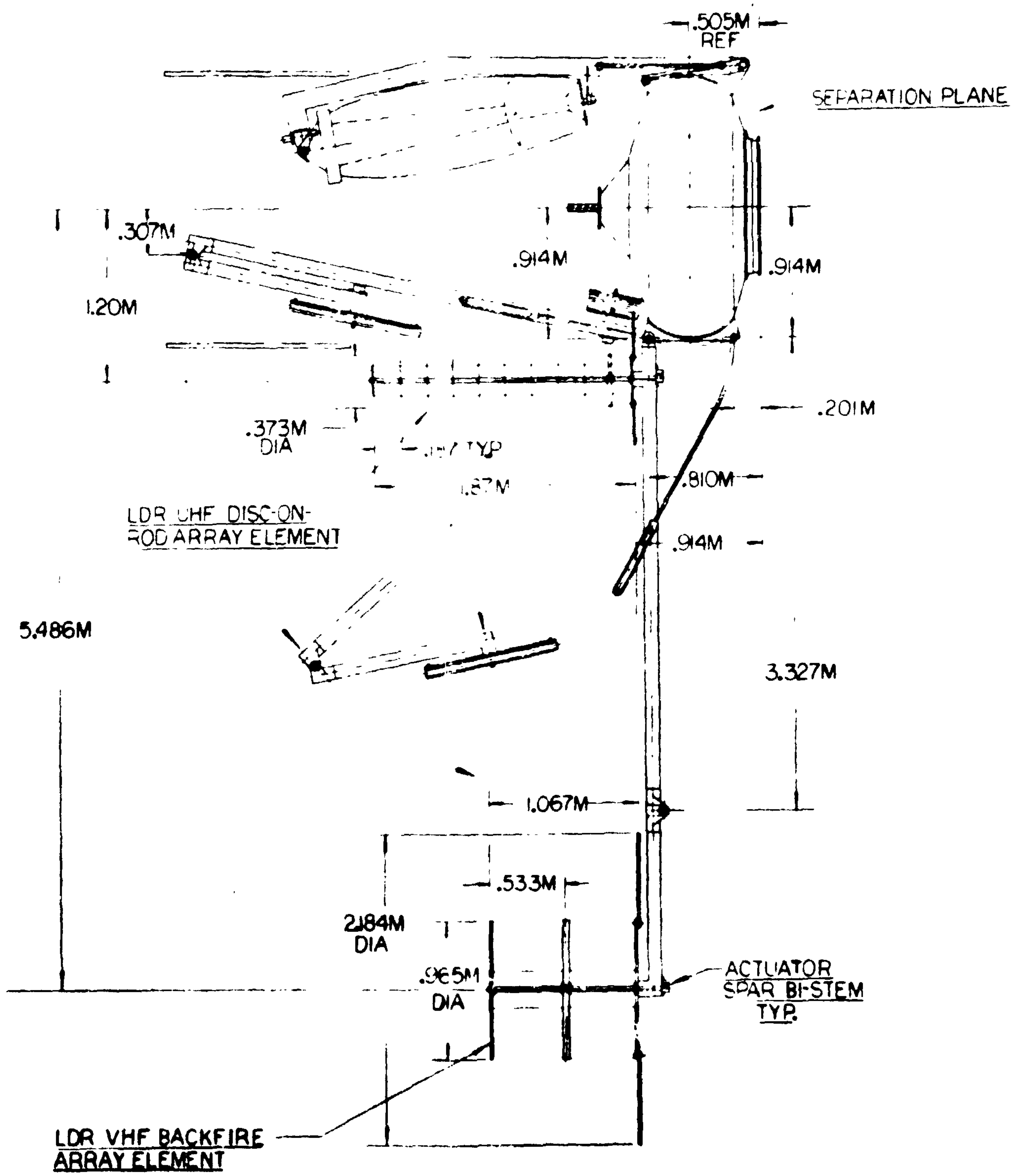
Four of the 3.8 M diameter antennas serve HDR/MDR users and one is the TDRS/GS antenna. As they are all identical except for the omission of the -band frequency feed in the TDRS/GS antenna, multiple switching of antennas from HDR/MDR to TDRS/GS operation is possible with high redundancy in case of any antenna failure.

The solar panels are located beyond the 23-1/2 degree solar shadow line on strut extensions on one 3.8 M antenna support strut and on one LDR UHF/VHF array support strut. Below the attach point of these solar panel struts the main support strut and deployment system are identical for all 3.8 M antennas and for all LDR UHF/VHF array elements.

As shown in the sectional views on Figure 3-2, all 3.8 M antennas, (which are of the furlable rib-mesh design by Radiation Systems, Melbourne, Florida, and described in Section 2.2 of this report) are folded down and the support strut pivots forward to the stowed position. As shown in Section A-A, the antenna with the solar panel extension strut has the solar panel strut and the solar panel fold down over the stowed antenna.

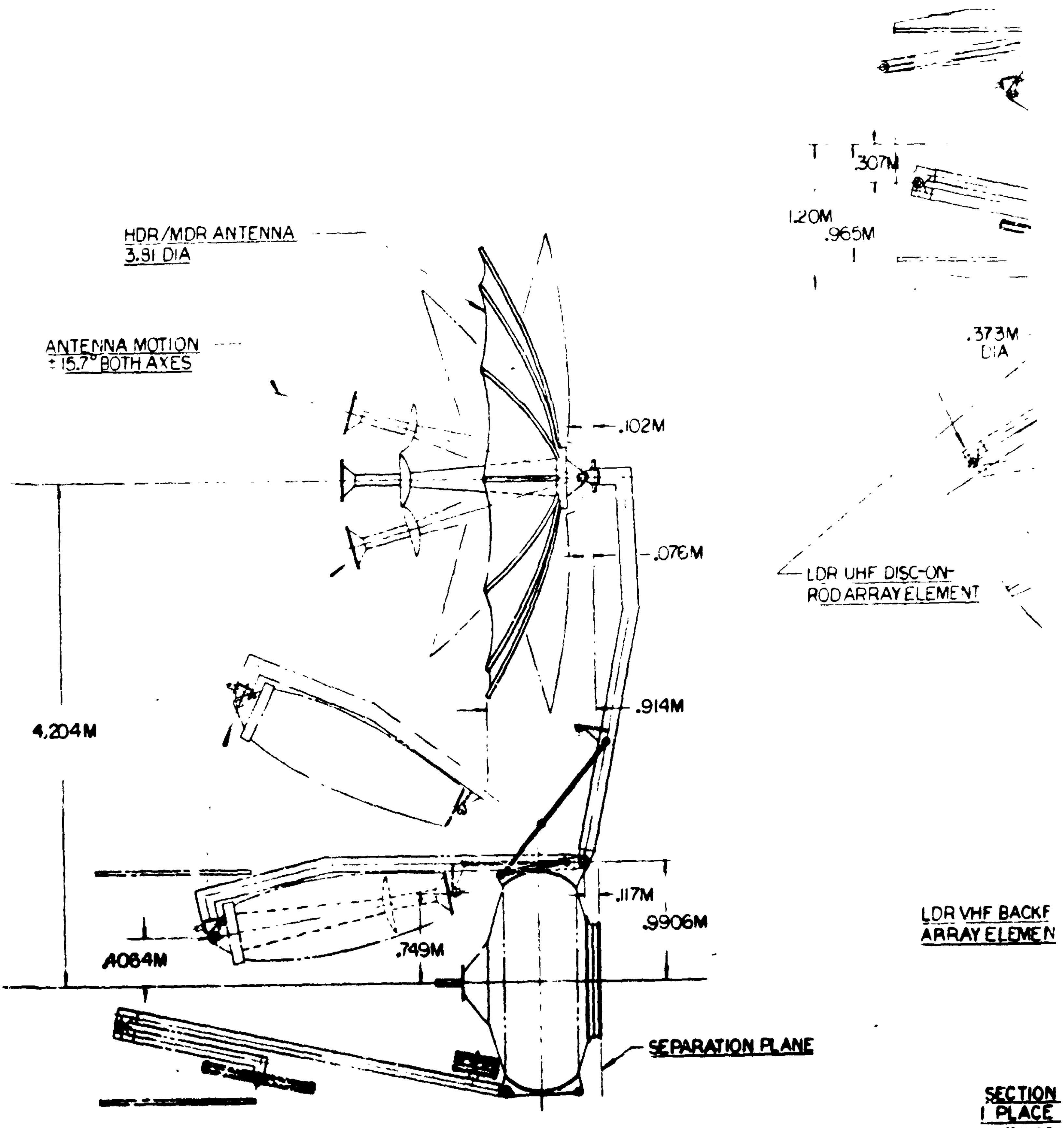
The senior AGIPA LDR array consists of five equally spaced UHF disc-on-rod and VHF backfire arrays mounted on folding support struts as shown in Section D-D and B-B.

FOLDOUT FRAME



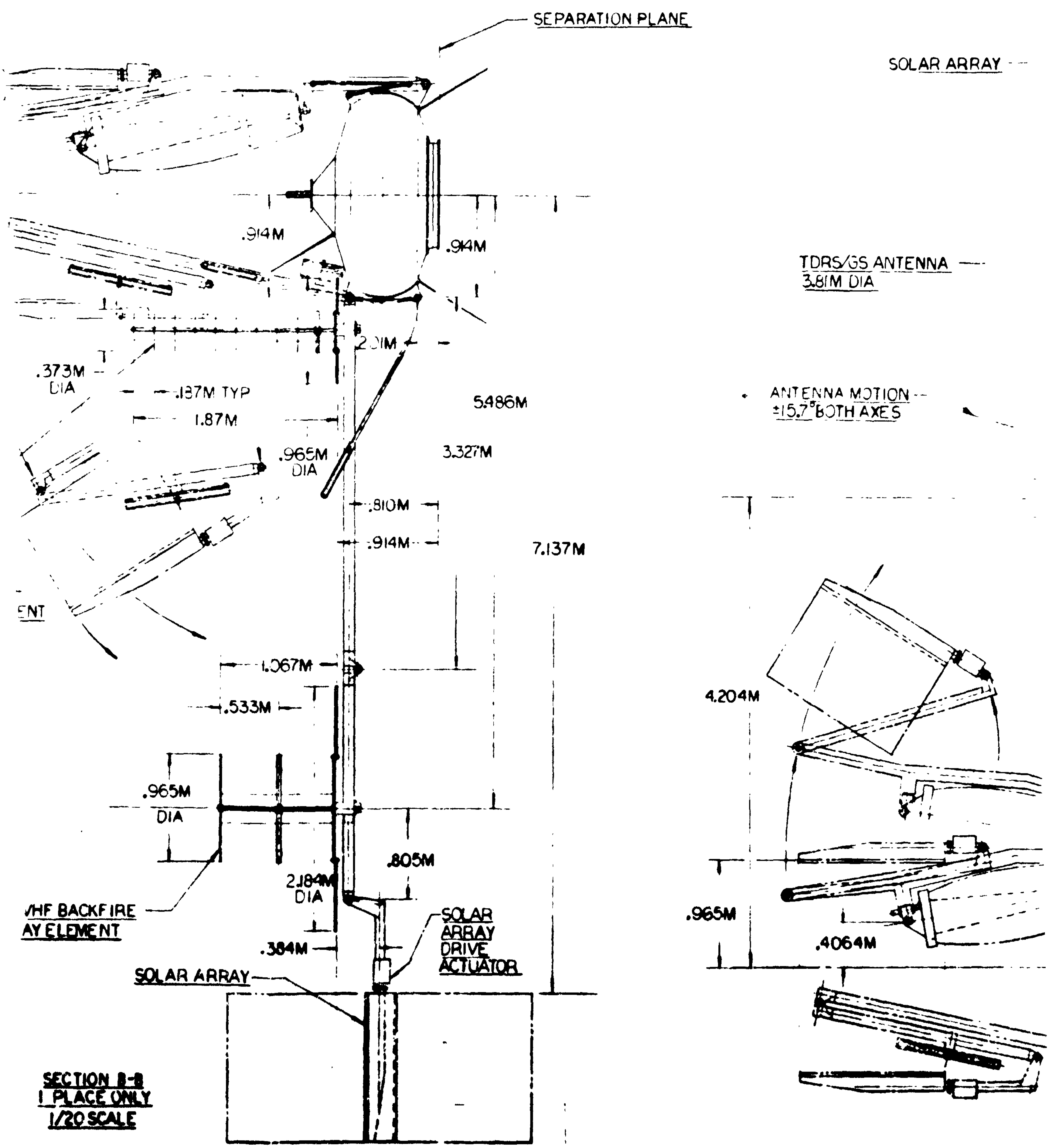
SECTION D-D
TYPICAL 4 PLACES
1/20 SCALE

FOLDOUT FRAME 2

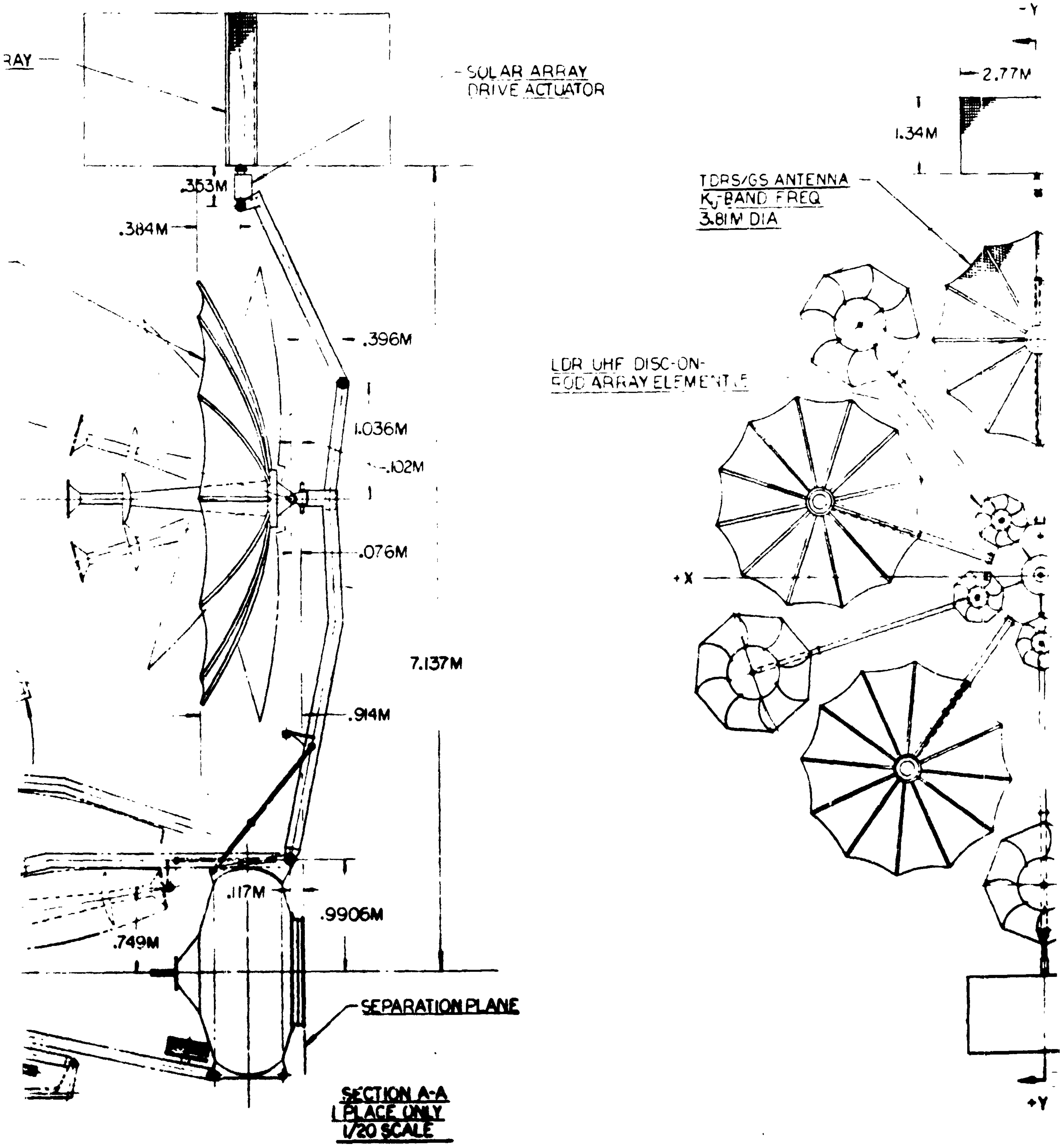


SECTION C-C
TYPICAL 4 PLACES
1/20 SCALE

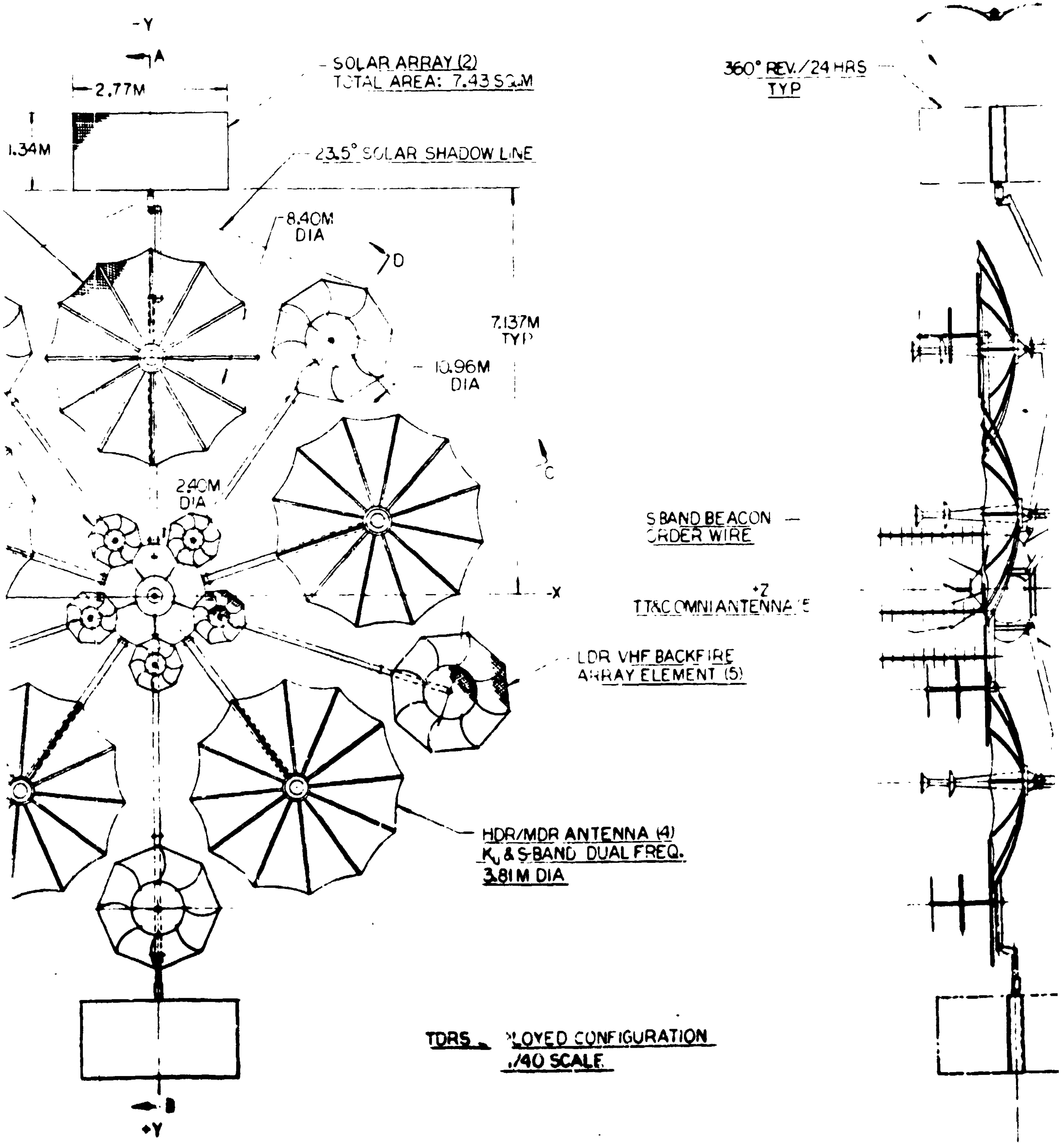
FOLDOUT FRAME 3



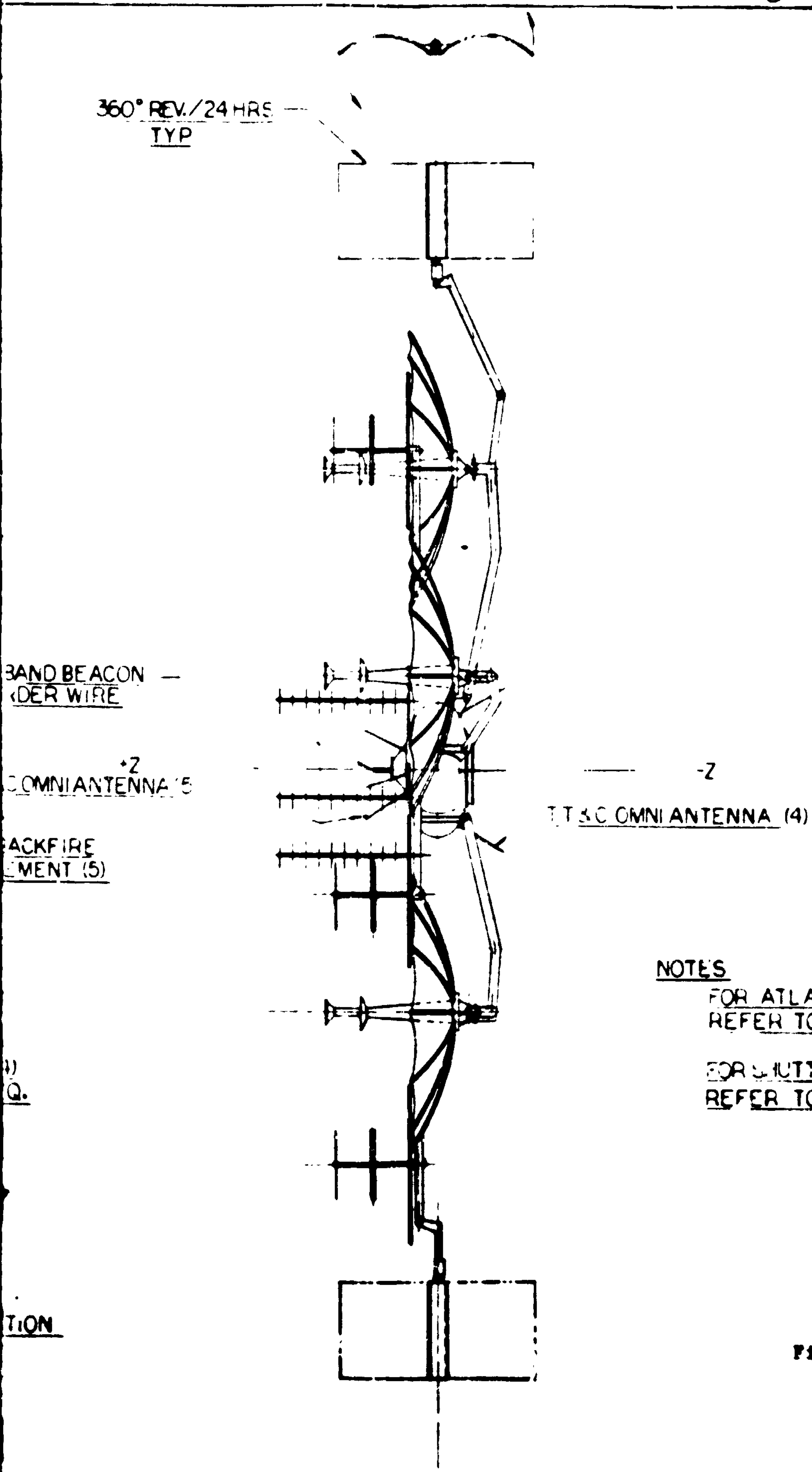
FOLDOUT FRAME 4



SECTION A-A
1 PLACE ONLY
1/20 SCALE



TDRS ... LOYED CONFIGURATION
./40 SCALE



NOTES
FOR ATLAS/CENTAUR ... CONFIGURATION REFER TO DRWG 1198-60

FOR SHUTTLE/AGENA-TUG LAUNCH CONFIGURATION REFER TO DRWG 1198-61

Figure 3-2. TDRS Deployed Configuration Five 3.81 M Diameter Antennas

The UHF disc-on-rod array element has a furlable ground plane of mesh and frame, with the discs and dipole element spaced equally along an extendable Spar bi-stem unit. When stowed, the unit is packaged within the 0.375 M diameter by folding the outer portion of the ground plane and closely stacking the discs and dipole along the retracted stem unit to a height of approximately 0.15 M.

The VHF backfire array element also has the outer portion of the mesh ground plane fold to a diameter of 0.965 M and the dipole and front plane mesh disc are spaced along an extendable Spar bi-stem unit. When stowed, the unit fits within the 0.965 M diameter and the front plane and the dipole are closely stacked along the retracted stem unit to a height of approximately 0.09 M.

The LDR element with the solar panel and its extension support is folded forward on its strut in the identical geometry as the other elements but with the solar panel and its support linkage folded down over the element system as shown in Section B-B of Figure 3-2. With all the main antennas mounted on the support struts, the front of the spacecraft body is available for mounting of the S-band tracking and order wire antenna and the forward looking TT&C omni-antenna.

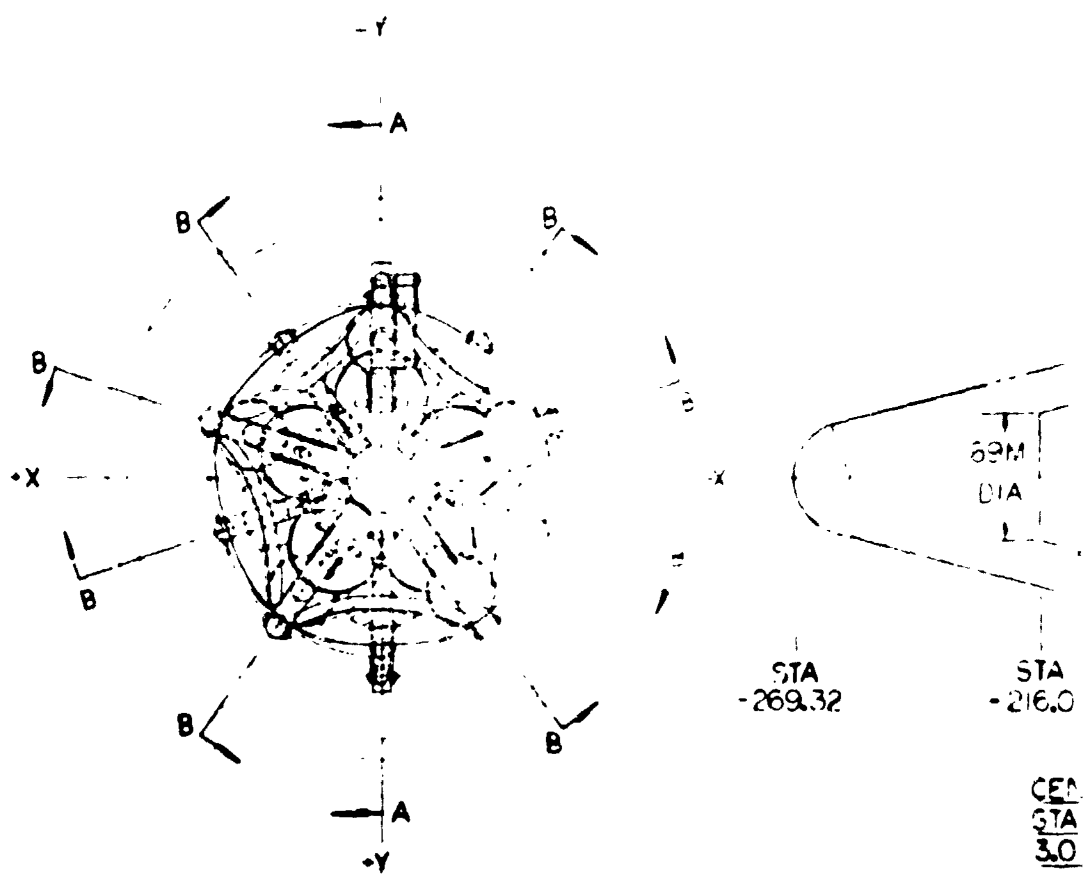
The greatly increased capability of this concept is not only apparent in the increase of performance realized with the LDR senior AGIPA array and the doubling of the number of HDR/MDR antennas from two to four, but the two sidemost HDR/MDR antennas along the X-X axis of the TDRS can easily be adapted by an increased gimbal travel to track and communicate with user spacecraft up to and including synchronous orbit altitude or with a third TDRS on the other side of the world, with the remaining HDR/MDR antennas still furnishing full communications capability equivalent to the uprated TDRS concept.

In the launch configuration, Figure 3-3, the five 3.8 M diameter antenna are folded and their supporting struts pivoted forward as shown in Section B-B of Figure 3-3. The senior AGIPA array elements are retracted upon their support struts and the struts folded down and forward between the stowed antennas. The TDRS/GS 3.8 M antenna support strut linkage has the solar panel support extension which folds down over the antenna and locates the stowed solar panel forward of the antenna as shown in Section A-A of Figure 3-3. The other solar panel, supported by the strut extension on the support strut for the lower LDR AGIPA elements, is folded forward and up to match the position of the upper stowed solar panel so that the outer edges contact along the spacecraft centerline making a cylindrical shape for the two stowed solar panels.

Launch latches between the hubs of the folded antennas and between the support struts and the spacecraft body provide restraint for launch-induced loadings. The LDR element support struts are locked to the spacecraft body in a similar fashion and the solar panels are latched together at their contacting edges to maintain position during launch.

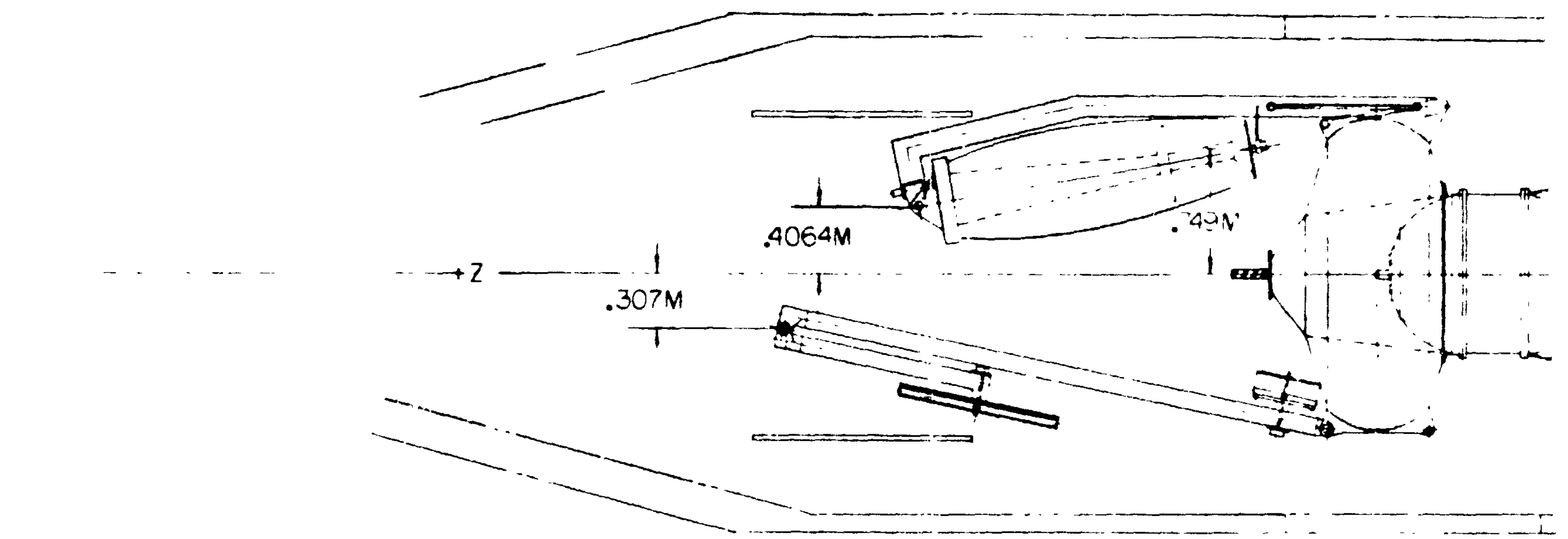
The Centaur D-1A fairing is extended by a standard barrel extension, with a length of 1.52 M, to provide clearance for the solar panels with the inside of the fairing. The spacecraft and apogee motor are mounted to the Centaur with the spacecraft adapter as shown.

FOLDOUT FRAME

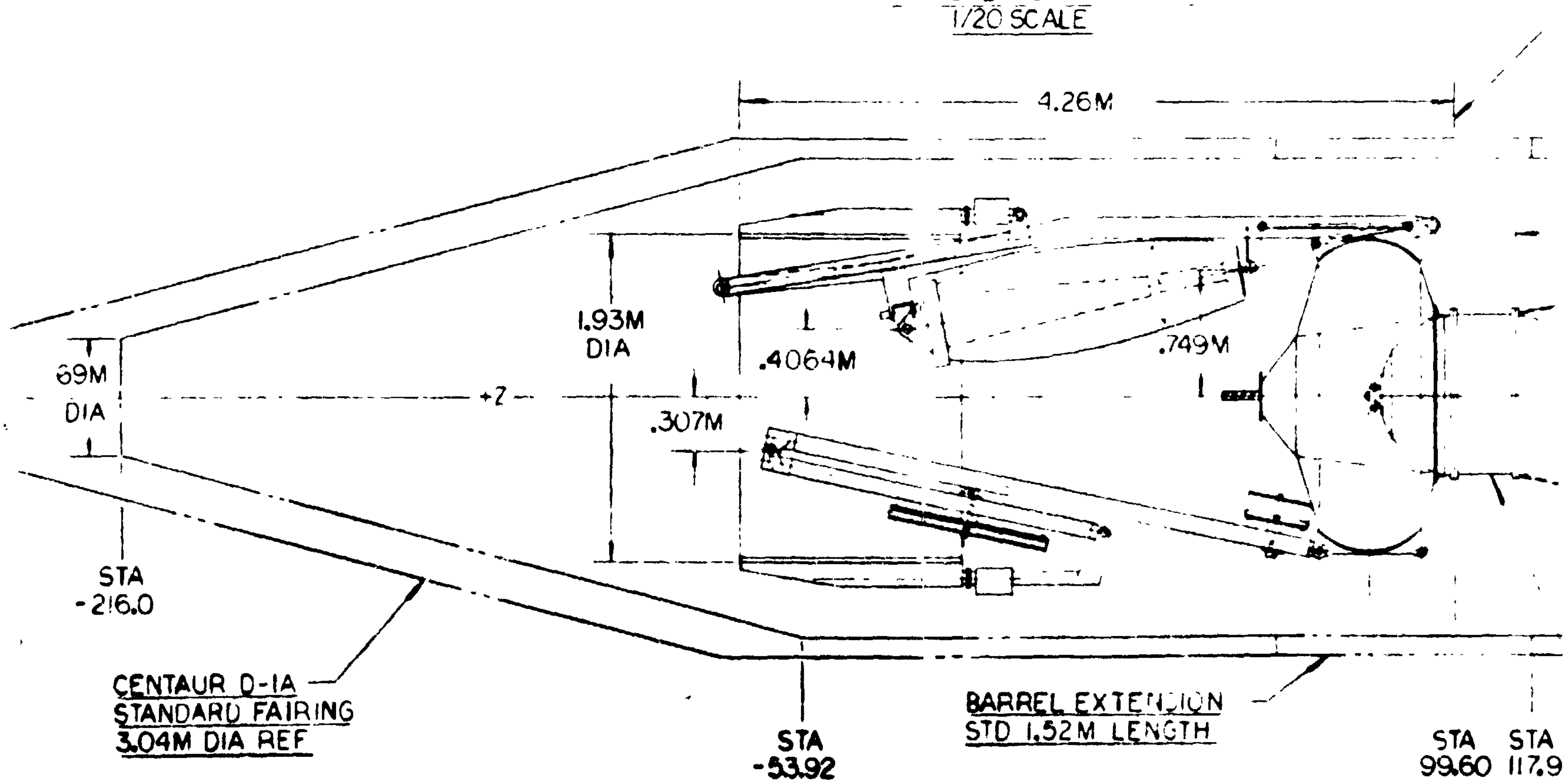


FRONT VIEW
1/20 SCALE

FOLDOUT FRAME 2



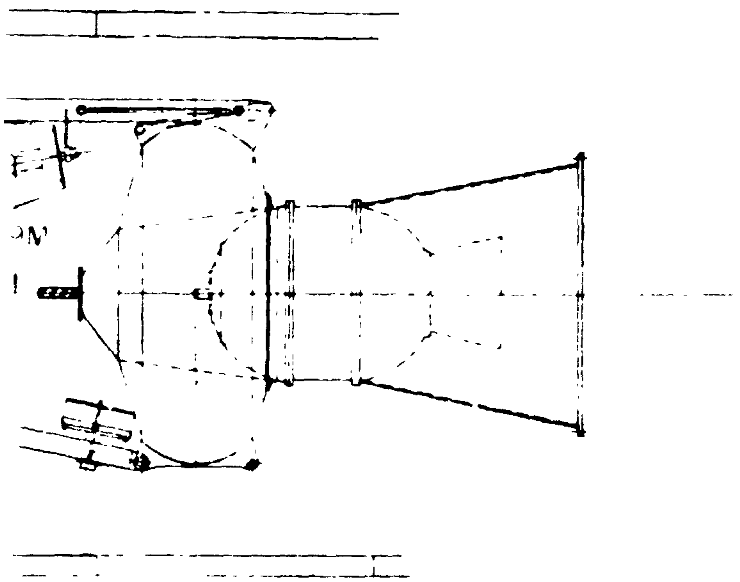
SECTION B-B
TYPICAL FOUR PLACES
1/20 SCALE

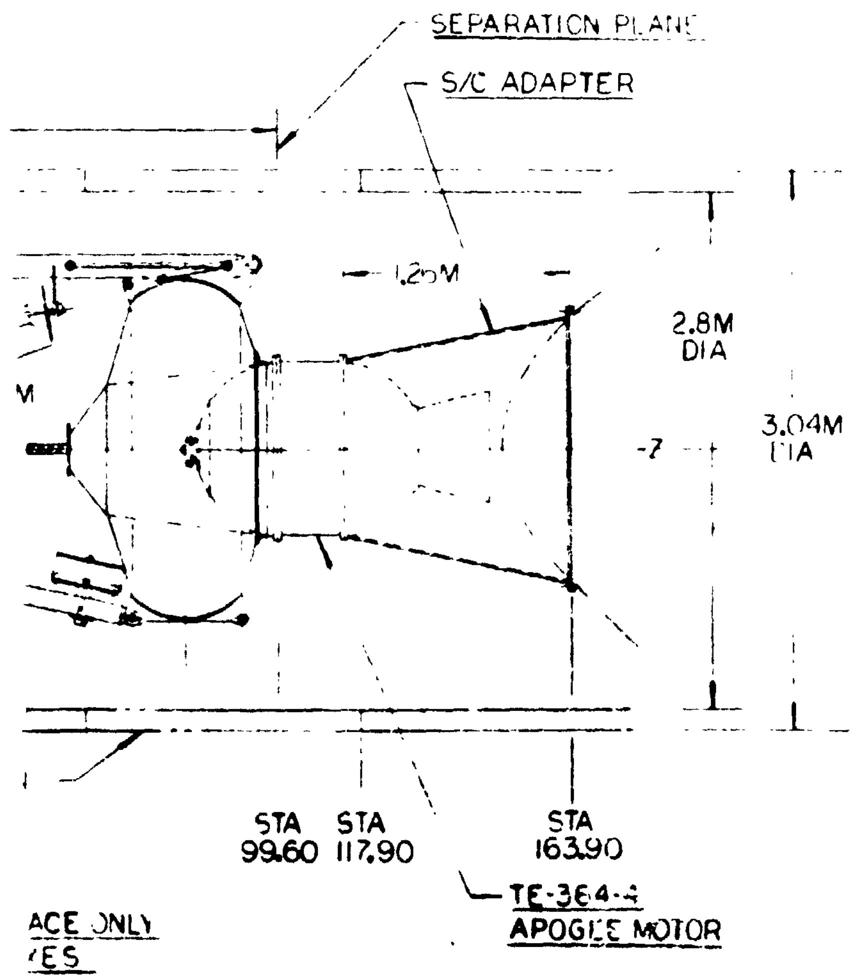


SECTION A-A ONE PLACE ONLY
IN PLANE OF Y-Z AXES
1/20 SCALE

FOLDOUT FRAME 3

PRECEDING PAG





NOTES
FOR DEPLOYED
REFER TO DWG

LAUNCH VEHICLE

Figure 3-3. TDRS
Five

FOLDOUT FRAME 4

PRECEDING PAGE BLANK NOT FILMED

NOTES

FOR DEPLOYED CONFIGURATION
REFER TO DRWG 1196-59

LAUNCH VEHICLE: ATLAS SLV-3D/CENTAUR D-1A

Figure 3-3. TDRS Configuration, Atlas/Centaur Launch, Five 3.81 M Diameter Antennas

In the deployment sequence, the two strut systems supporting the solar panels are released initially by ground-activated commands to the solenoid-operated latches and the solar panels deploy along with the TDRS/GS antenna and the lower LDR element struts to clear for subsequent deployment of the LDR elements on their struts.

The remaining HDR/MDR antennas are then deployed to their extended positions, all folded antenna-to-strut latches are released and the antennas are driven by their gimbal drives to their neutral forward-looking positions. The antenna reflectors are then released and deployed to their full diameters.

Simultaneously, the LDR UHF and VHF elements are activated when their struts extend and lock in their full out position, and the spring-loaded ground plane arms extend the ground plane rims to full diameter. The STEM actuator at the base of each element is energized and extends the elements to their full lengths.

The extended solar panels are driven by their drive systems to acquire and align with the solar line normal to the panels. They are then driven to maintain position with one-revolution-per-day rate. The TDRS/GS antenna is then aligned to the proper coordinates to acquire the ground station antenna and the TDRS achieves its operational status.

3.2.2 Subsystem

3.2.2.1 Communication Systems

The communications system consists of the HDR/MDR, the LDR, the TDRS/GS, TT&C and tracking and order wire similar to the uprated TDRS.

System. The HDR/MDR system is equivalent to the uprated TDRS concept except for the increase in number of antennas and their receivers and transmitters. It uses the same type and diameter of antenna as in the uprated TDRS design, with identical receivers and transmitters.

The weight of the system is shown in Table 3-1.

The TDRS/CS system consists of a 3.8 M diameter Ku-band antenna and the receiver and transmitter. Because of the increased aperture of the antenna, the transmitter design was changed back to the solid state type instead of the TWT design of the uprated TDRS design. The receiver is identical to that of the uprated TDRS.

The weight of the system is shown in Table 3-2.

The LDR system consists of the senior AGIPA array of five elements of VHF disc-on-rod and VHF backfire on 2.4 M diameter and 0.96 M diameter, respectively. The receivers and transmitters are identical to the uprated TDRS but increased in number to five.

The weight of the system is shown in Table 3-3.

Table 3-1. HDR/MDR System Weight

		Kg
HDR/MDR No. 1		
Receiver		4.1
Transmitter		6.1
Antenna and support strut		20.1
Reflector	7.03	
Feeds	2.00	
Electronics/control	2.26	
Gimbal drive	2.26	
Rotary joints	0.96	
Support strut (5.00" x 0.020" wall + (2) fittings)	2.68	
Springs	0.57	
Braces	0.95	
Hardware	1.36	
	20.07	
		30.3
HDR/MDR No. 2		
Receiver		4.1
Transmitter		6.1
Antenna and support strut		20.1
		30.3
HDR/MDR No. 3		
Receiver		4.5
Transmitter		6.4
Antenna and support strut		20.1
		31.0
HDR/MDR No. 4		
Receiver		4.5
Transmitter		6.4
Antenna and support strut		20.1
		31.0
Total		122.6

Table 3-2. TDRS/GS System Weight

Receiver		Kg
Receiver		2.2
Transmitter		6.5
Antenna and support strut		
(This antenna is identical to the HDR/MDR antenna except for the elimination of the S-band feed system)		
HDR/MDR antenna	20.1	
Less S-band feed	1.04	
	19.06	
Total		27.7

Table 3-3. LDR System Weight

		Kg
Receiver (5)		5.1
Transmitter (5)		3.2
Antenna and support strut (5)		45.7
VHF backfire element		
Ground plane	1.10	
Front R.Fl. plane	0.57	
Dipole element	0.25	
Stem and drive	0.69	
	2.60	
UHF disc-on-rod element		
Ground plane	0.47	
Discs (10)	0.92	
Dipole element	0.004	
Stem and drive	0.68	
	2.09	
Support strut		
Tube (4.0 in. x 0.032 in. wall)	3.3	
Fittings (3)	0.68	
Hardware	0.46	
	4.45	
Total	9.15	
Total		54.0

The TT&C system is identical to the uprated TDRS design system except for a change in the forward looking omni-antennas from four elements to five to provide symmetrical packaging with the five 3.8 M antennas and the five LDR array elements and an increase of 1.6 kg in the processor weight to provide for the additional HDR/MDR antennas. The new weight is 9.8 kg.

The tracking and order wire system is identical to the uprated TDRS design with the antenna mounted on the front center of the spacecraft. The weight of the system is identical to the uprated TDRS system at 2.6 kg.

The wiring, cabling, and W/G runs necessary to interconnect the systems and the frequency source unit are included in this section.

The frequency source is identical to the uprated TDRS design.

The weight of these items is 15.5 kg.

The total weight of the communications systems (Table 3-4) is 231.6 Kg.

Table 3-4. Communications Systems Summary Weight

	Kg
HDR/MDR	122.6
TDRS/GS	27.7
LDR	54.0
TT&C	9.3
S-band tracking and order wire	2.6
Frequency source	3.5
Wiring, cabling	12.0
Total	231.6

3.2.2.2 Electrical Power System

The electrical power system is the same as the uprated TDRS design system except that the battery capacity and solar panel area were increased to accommodate the electrical power loads of the greater number of HDR/MDR antennas in this TDRS concept. Additional battery weight of 15.9 kg was added to the system.

The weight of the system is shown in Table 3-5.

Table 3-5. Electrical Power System Weight

	Kg
Power conditioning and distribution	5.1
Charge and discharge	5.1
Central control and logic	2.3
Packaging	2.2
Shunt dissipators	1.1
Power conditioning voltage	2.3
Cabling	9.1
Energy storage	
Batteries (2)	36.0
Total	58.1

The solar array system was changed from the uprated TDRS design by an increase in solar panel area from 4.18 M to 7.43 M to provide the greater electrical capacity for the greater number of HDR/MDR links. To maintain low weight, the substrate incorporates a light weight magnesium grid design developed by Electro-Optics, Santa Barbara, California, to provide a 2.47 kg per M solar panel.

The support strut linkage supporting the solar panels above the TDRS/GS antennas and below the lower LDR array element are designed to fold the curved solar panels in the stowed configuration around the stowed antennas and to form a cylindrical shape by their outer edges contacting along the centerline of the spacecraft.

The weight of the system is shown in Table 3-6.

Table 3-6. Solar Array System Weight

	Kg
Solar array panel No. 1	
Solar panel	9.07
Support strut	2.13
Fittings (2)	0.68
Springs, hardware	0.45
Drive actuator	3.40
	15.73
Solar array panel No. 2	
Solar panel	9.07
Support strut	0.86
Fittings (2)	0.68
Springs, hardware	0.68
Drive actuator	3.40
	14.69
Total	30.42

3.2.2.3 Spacecraft Body Structure

The body structure is the same as in the uprated TDRS design except for the location and number of attach fittings for the antenna support struts. Additional weight of 4.5 kg was included for the increased number of fittings resulting in a total structural weight of 45.8 kg.

4.2.2.4 Thermal Control System

No change in the thermal control system was made over the uprated TDRS design. The system weight is 10.8 kg.

3.2.2.5 Auxiliary Propulsion System

The system is the same as the uprated TDRS system except for the additional propellant required because of the increase in spacecraft weight due to additional antennas. Additional propellant weighing 3.29 kg is added to the system, giving a weight of 14.60 kg for hardware, 0.27 kg of N_2 pressurant and 28.37 kg propellant for a total of 43.24 kg.

3.2.3 Weight Summary

The weight of the maximum capability TDRS design is shown in Table 3-7.

Table 3-7. Maximum Capability TDRS Weight Summary

Communications systems	231.6
Attitude and control	23.21
Electric power	58.1
Solar array	30.42
Structure	45.8
Thermal control	10.8
Aux. propulsion hardware	14.6
	414.4
Propellant + N_2 (2-65° - 15 day sta. change)	25.7
Total spacecraft	440.1
For Atlas/Centaur launch	
Total spacecraft	440.1
Contingency	476.2
Allowable P/L	
Atlas/Centaur + TE-364-4	916.3

4.0 SHUTTLE-AGENA SPACECRAFT CONCEPTS

The concept of multiple TDRS launches from the Space Shuttle was investigated. The Shuttle alone cannot place a payload at synchronous orbit and an additional boost stage (Tug) or a boost stage plus an apogee kick motor is required. Both the Agena stage and the transtage were considered for the Tug, both with and without an apogee kick motor.

Two spacecraft concepts were considered for the Shuttle launch. The uprated baseline which can support two MDR/HDR users serves as a minimum cost version. The five antenna spacecraft configured for the Atlas-Centaur launch also can be launched (three at a time) by the Shuttle. This can support four MDR/HDR users with an adequate weight margin, but will have a higher cost. Both versions can be placed in synchronous orbit by the Agena without need for a kick stage.

Both versions of the TDRS launched from the Shuttle are essentially the same as those launched from the Delta 2914 and the Atlas-Centaur with the apogee motors removed and minor structural revisions to accommodate the removal. Subsystems will be the same except the spin operations during launch will be eliminated, permitting removal of the spin attitude control sensors and a reduction in the propellant for the spin phase and orbit injection correction. This propellant either can be removed and the weight contingency increased or it can be kept aboard to increase the on-orbit expendables.

4.1 SYSTEM ANALYSIS

Tug versions of both the Agena and transtage were considered for multiple TDRS launch. Lockheed is developing an Agena with an I_{sp} of 324 seconds by using N_2O_4/MMH which can put 1500 kg (3300 lb) into 0-degree inclined synchronous orbit. The transtage can insert 1365 kg (3000 lb) into this orbit. (A slightly higher load can be inserted into the TDRS inclined subsynchronous orbit.) The transtage costs slightly less than the Agena but weighs approximately 6000 kg (13,000 lb) more and requires a heavier adapter both to the Shuttle and to the TDRS. Considerably more Agenas have been used, and it is felt to be more reliable. For these reasons, the Agena was selected for the TDRS Tug.

Figure 4-1 shows the payload capability of the Agena with an I_{sp} = 310 seconds and 324 seconds to synchronous orbit, both with and without a TE-M-616 apogee motor. Both TDRS versions are within the range of direct insertion by the Agena with I_{sp} = 324 seconds without need for the apogee motor.

The Agena will separate from the Shuttle, ignite, and carry the TDRS to synchronous altitude without spinning. At synchronous altitude on the first or second apogee the Agena will burn again and go into a circular 2-1/2 degree inclined orbit with approximately 30 m/sec (100 fps) eastward drift. This drift is equal to 10 degrees per day and increases the Agena payload capability by 25 kg (55 lb) as shown in Figure 4-2. The Agena and TDRS drift to the assigned stations where each TDRS is released and stops its own drift.

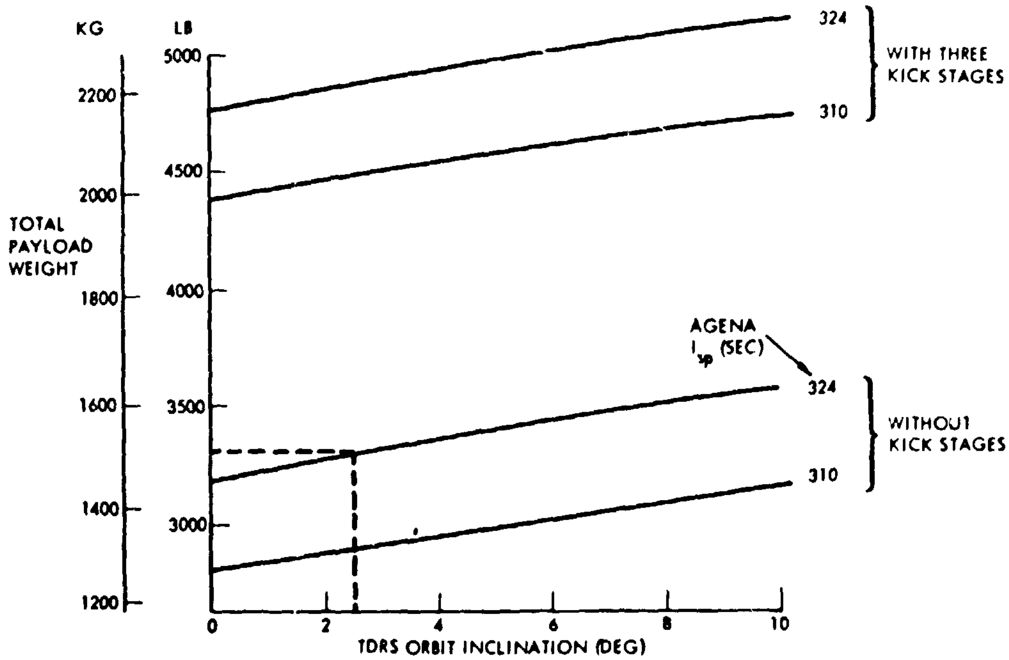


Figure 4-1. Payload Into Geosynchronous Orbit (Shuttle/Agena)

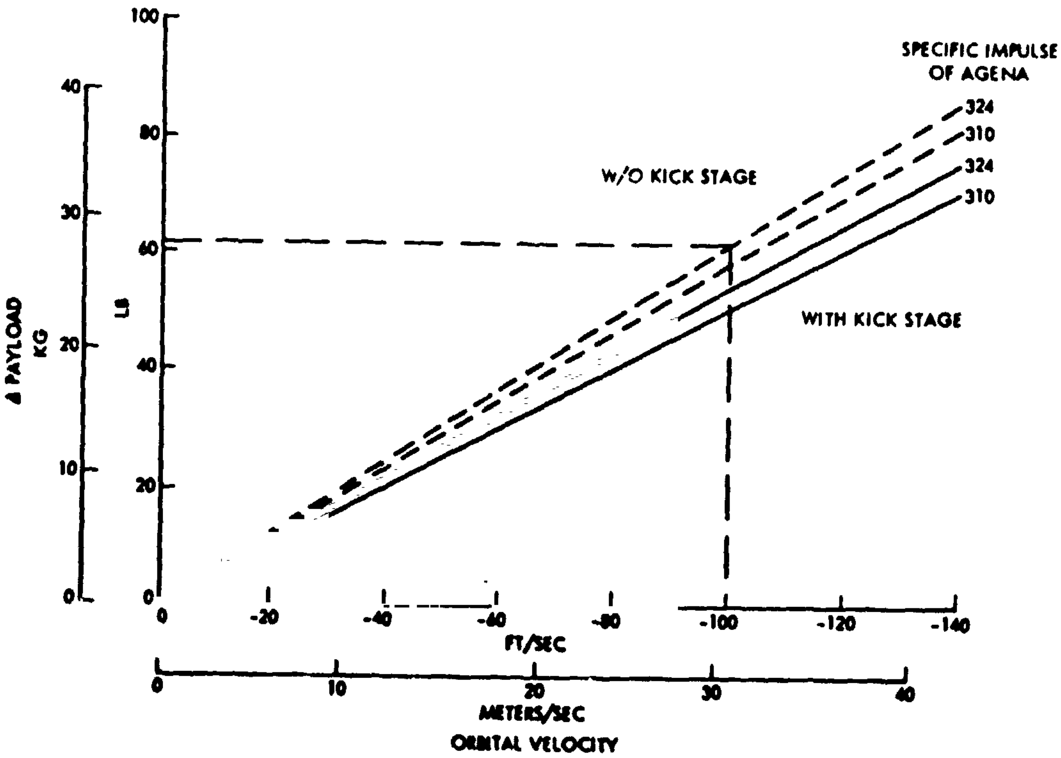


Figure 4-2. Shuttle/Agena Payload Increment

The total payload capability from Figures 4-1 and 4-2 is: 1504 kg (3315 lb) to 2-1/2 degree inclined orbit plus 25 kg (55 lb) for the drift, resulting in a capability of 1529 kg (3370 lb).

4.2 SPACECRAFT DESIGN CONCEPTS

4.2.1 Maximum Capability TDRS Concept

This TDRS concept is similar to the concept described in Section 3.2.1. In the deployed configuration there is no difference; but in the launch configuration in the Shuttle bay, an arrangement permitting three TDRS's to be launched on one Agena Tug is shown in Figure 4-3. The three TDRS spacecraft are positioned side-by-side at 120-degree spacing around the centerline of the Shuttle bay on an adapter that mounts to the front of the Agena Tug.

Each TDRS spacecraft is packaged around its own centerline in a similar fashion to the Atlas-Centaur packaging described in Section 3.2.1. As shown in the front view of Figure 4-3, the orientation of the X-X axis for each spacecraft is along each radial 120-degree line from the center of the adapter. This positions protuberances such as the solar panel actuators and support struts to be clear of the minimum clearance locations with the side of Shuttle bay and TDRS-to-TDRS spacecraft.

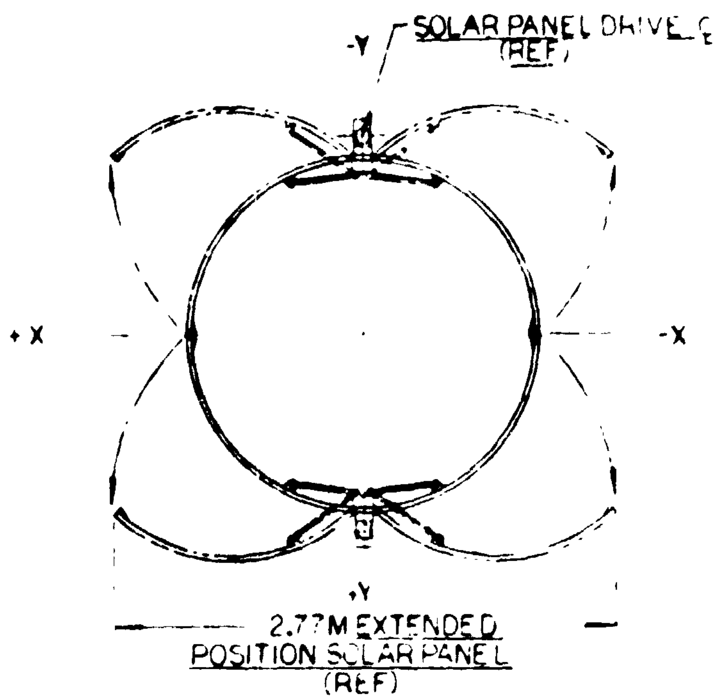
The Agena Tug is located and positioned in its support cradle in the aft end of the Shuttle bay. With the three TDRS spacecraft mounted upon the adapter, there remains approximately 7 meters of clearance to the forward bulkhead of the Shuttle bay. This area can be used for other experiments or scientific packages that might be carried in the same Shuttle launch.

Separation from the Agena Tug, after reaching synchronous orbit, is achieved by one-at-a-time release of the V-clamps securing the TDRS's to the Agena adapter and differential spring loading of the separation springs between the spacecraft and the adapter. A slight rotation and outboard velocity vector are achieved to clear each TDRS from the adjacent spacecraft. Control of this rotation is maintained by the spacecraft's auxiliary propulsion system to avoid tumbling at separation. Each TDRS is released at the proper sequence and position in orbit to allow drift and braking with its thrusters to stop at its desired position.

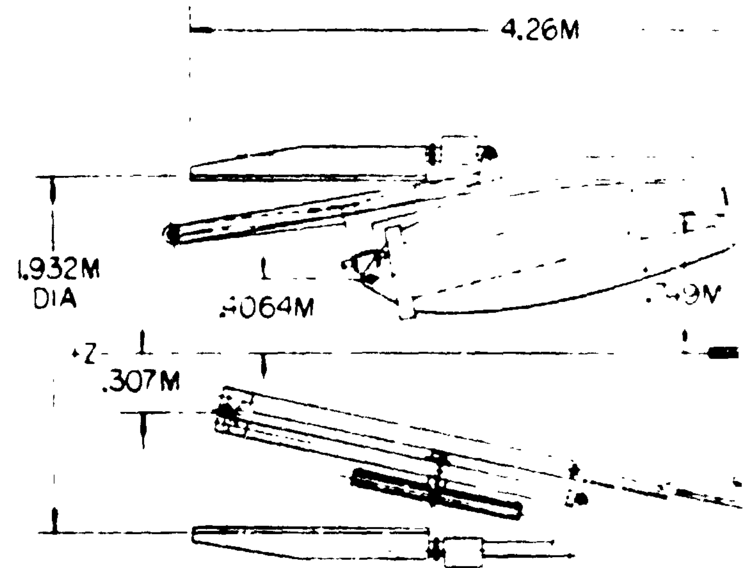
Deployment of each TDRS is achieved in a fashion similar to that described in Section 3.2.1.

The weight of each TDRS is the same as the Atlas/Centaur TDRS except that the spinning sensors were eliminated from the system. The Agena Tug places the spacecraft into orbit without the apogee kick motor and the spacecraft is not spun up. This reduces the attitude control system by 3.0 kg, and the total spacecraft weight to 440 kg.

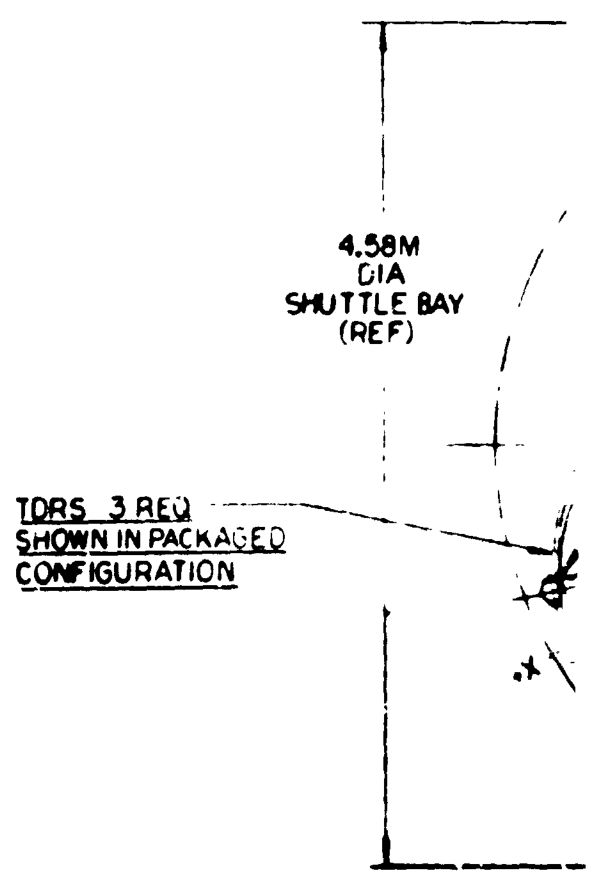
With the allowable payload of the Shuttle-Agena Tug of 1529 kg as defined in Section 4.1, the weight of the adapter of 68 kg is subtracted to give 1461 kg payload for three TDRS spacecraft. Each spacecraft allowable payload becomes 487 kg, and with a spacecraft weight of 440 kg, the contingency is 47 kg per spacecraft, or approximately 11 percent of dry weight.

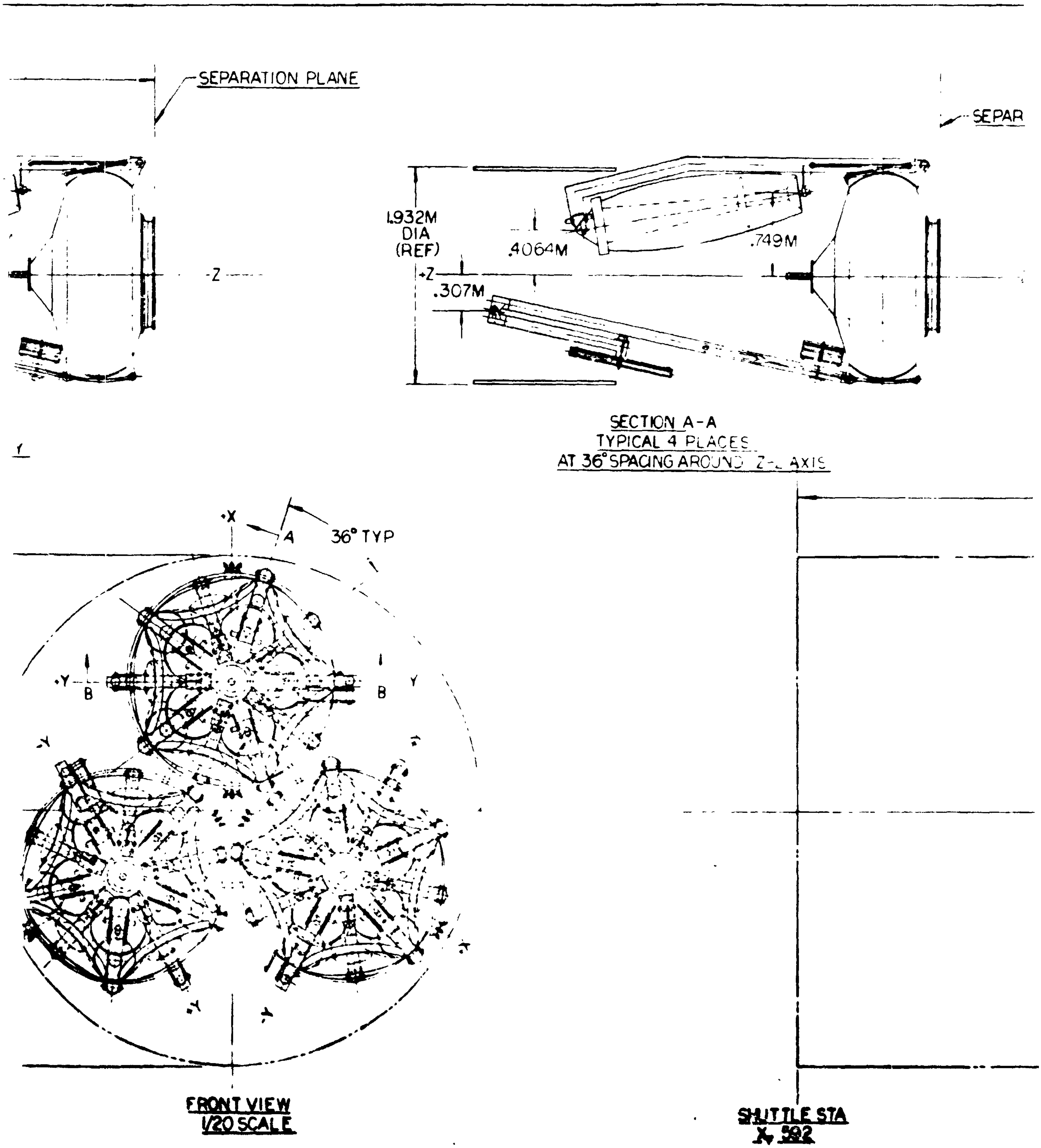


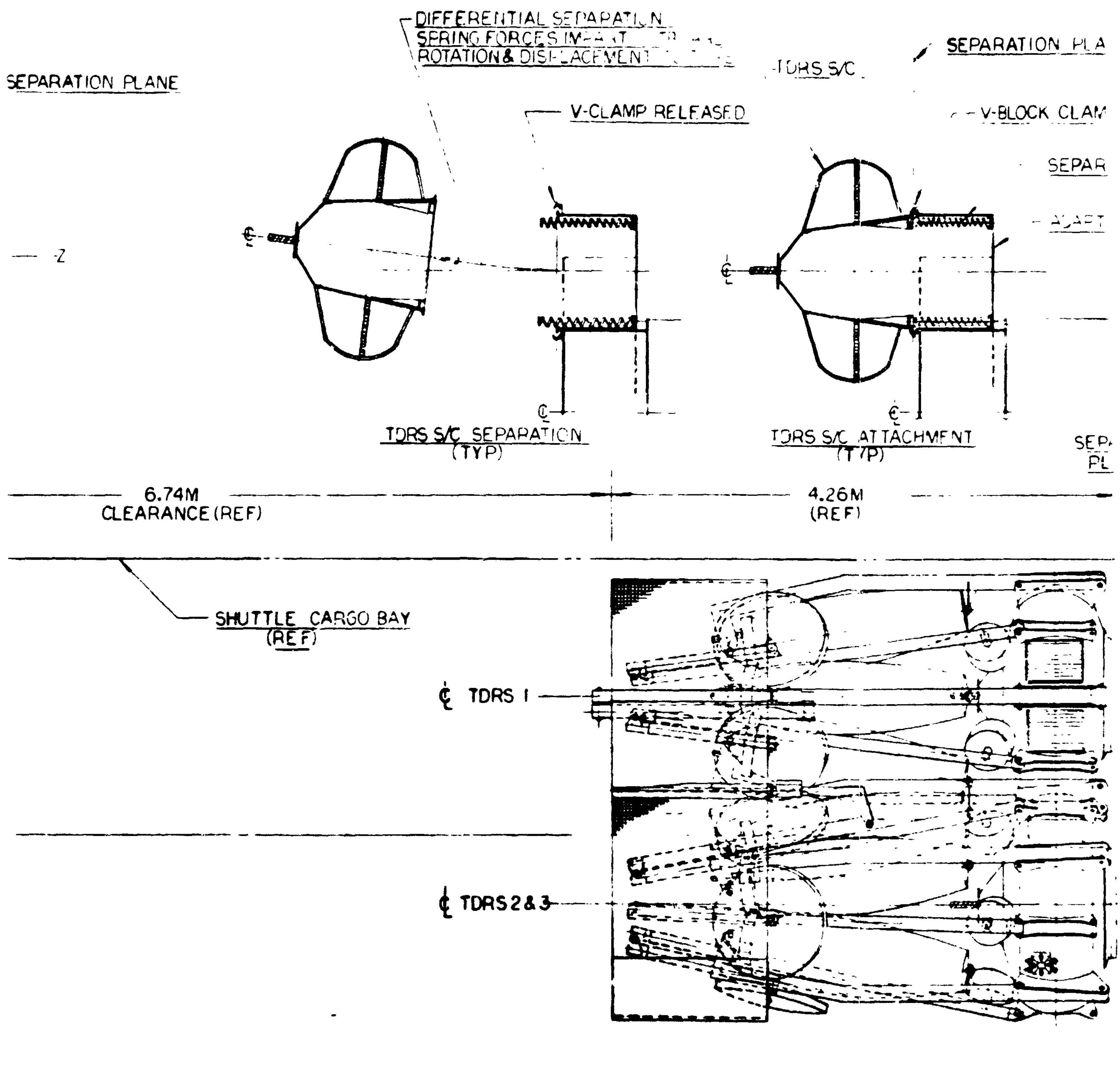
FRONT VIEW
SOLAR PANELS ONLY



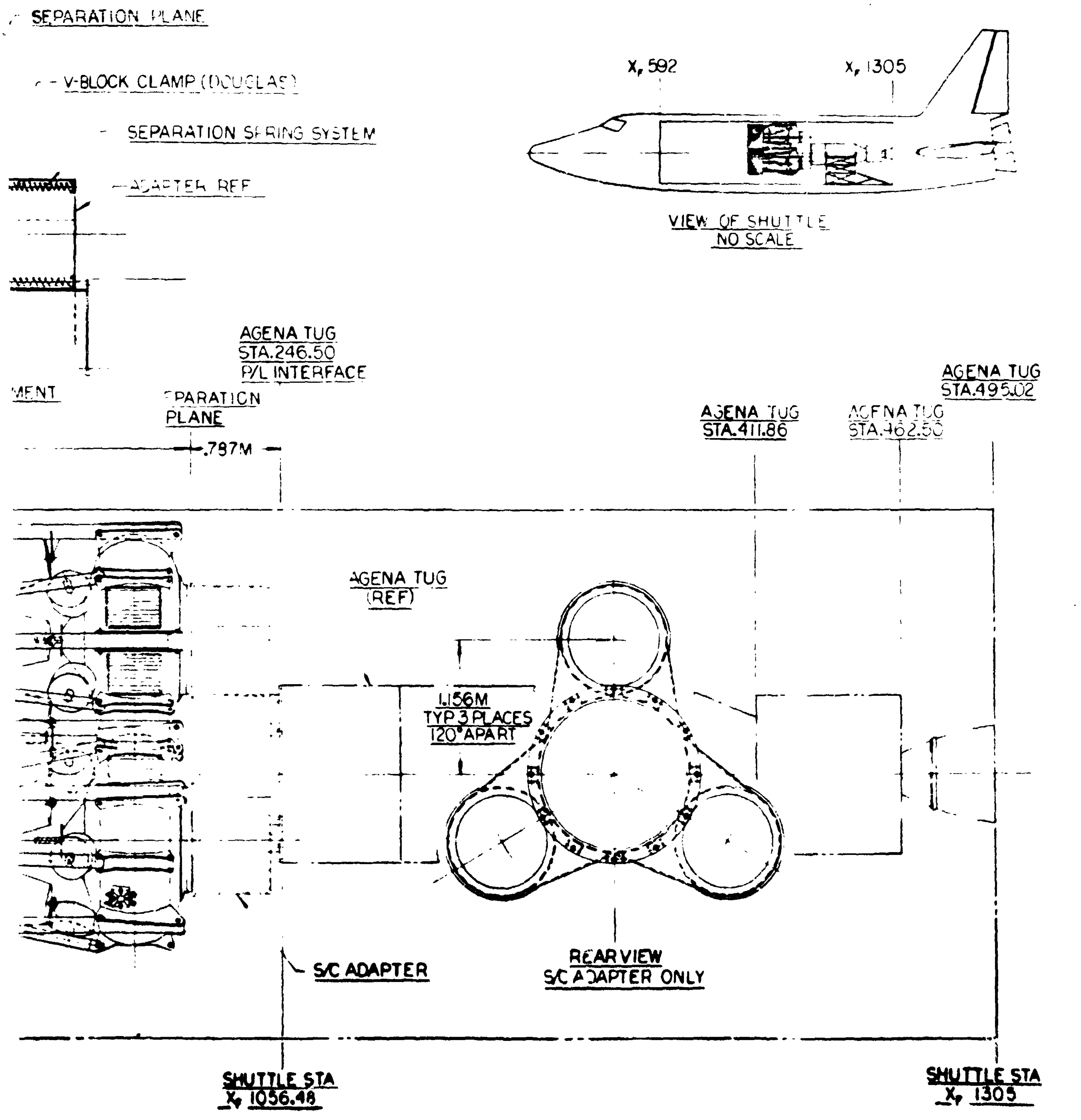
SECTION B-B
IN PLANE OF Y-Z AXES ONLY
1/20 SCALE

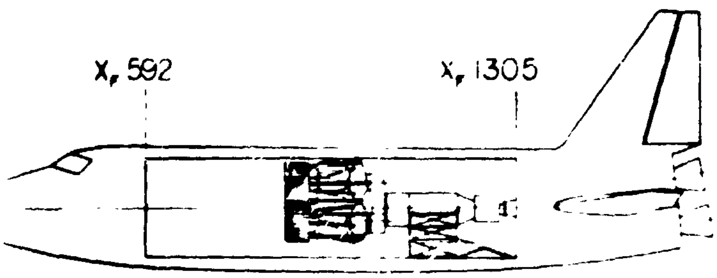


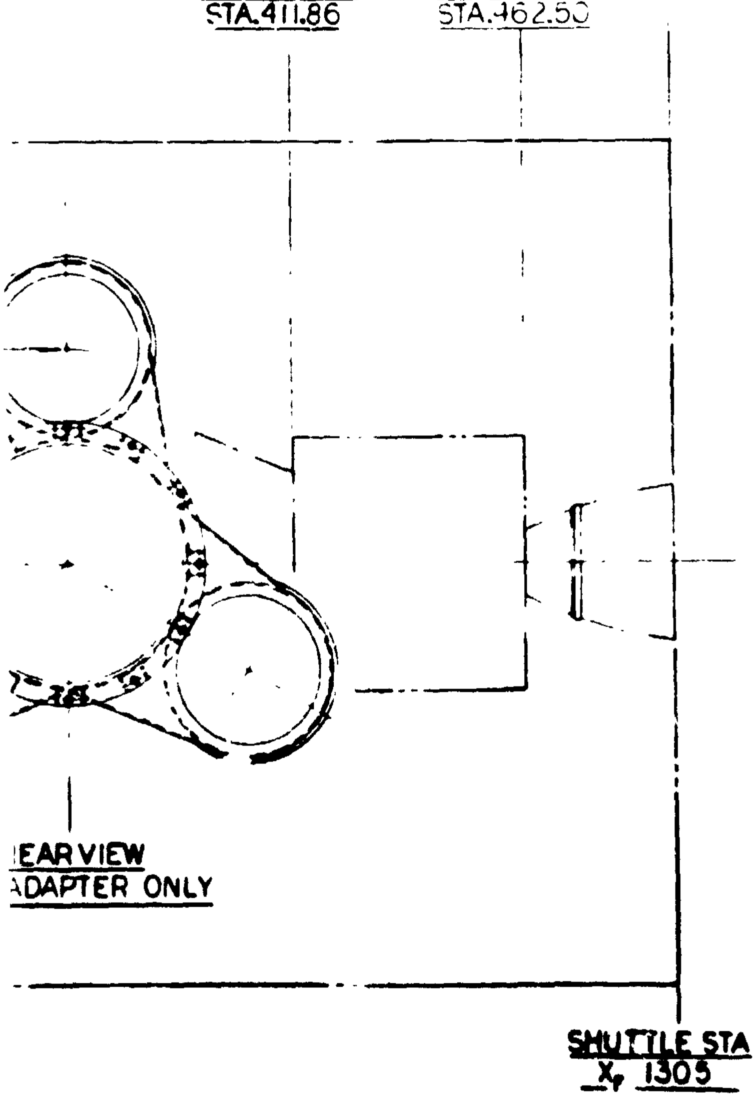

SEPARATION PLANE
SEPAR
-Z
1.932M
DIA
(REF)
.4064M
.749M
+Z
.307M
SECTION A-A
TYPICAL 4 PLACES
AT 36° SPACING AROUND Z-Z AXIS
+X
A
36° TYP
+Y
B
B
Y
FRONT VIEW
1/20 SCALE
SHUTTLE STA
X 592



SHUTTLE/AGENA-TUG LAUNCH CONFIGURATION
3 TDRS S/C IN SINGLE LAUNCH
SIDE VIEW
1/20 SCALE


SEPARATION PLANE
V-BLOCK CLAMP (DOUGLAS)
SEPARATION SPRING SYSTEM
ADAPTER REF
X, 592
X, 1305
VIEW OF SHUTTLE
NO SCALE
AGENA TUG
STA.246.50
P/L INTERFACE
MENT
SEPARATION
PLANE
.787M
AGENA TUG
STA.411.86
AGENA TUG
STA.462.50
AGENA TUG
STA.495.02
AGENA TUG
(REF)
1.156M
TYP 3 PLACES
120° APART
S/C ADAPTER
REAR VIEW
S/C ADAPTER ONLY
SHUTTLE STA
X, 1056.48
SHUTTLE STA
X, 1305





NOTES
FOR TDRS DEPLOYED CONFIGURATION
REFER TO DRWG [illegible]

Figure 4-3. TDRS Shuttle/Agena-Tug Launch, Five 3.8 M Antennas

4.2.2 Uprated TDRS Concept

The uprated TDRS concept can be packaged in the Shuttle-Agena Tug triple launch configuration in a similar fashion to the maximum capability concept. As shown in Figure 4-4, the overall arrangement is quite similar except for the internal packaging of a smaller number of antennas on each TDRS.

The solar panel design and linkage were changed from the uprated TDRS-Delta launch concept to move the panels forward away from over the spacecraft body to a position forward of the furled antennas to reduce the diameter of the solar panel stowed configuration. This provided clearance between solar panels on adjacent spacecraft and with the sides of the Shuttle bay. It also eliminated one folding joint in the solar panel support strut linkage, thereby simplifying the solar panel deployment.

Except for the solar panel change, each TDRS is packaged and stowed in an identical manner around its own centerline as the Delta launch configuration described in Section 2.2.1.1. The three spacecraft are mounted side-by-side on an adapter of the same geometry as the maximum capability TDRS Agena adapter. The orientation of the X-X axis of each spacecraft along the 120-degree radial lines from the center of the adapter is maintained to provide the necessary clearances between the spacecraft and the Shuttle bay wall.

The separation from the Agena Tug at synchronous orbit of each spacecraft at its proper position is accomplished in a fashion similar to that described in Section 4.2.1.

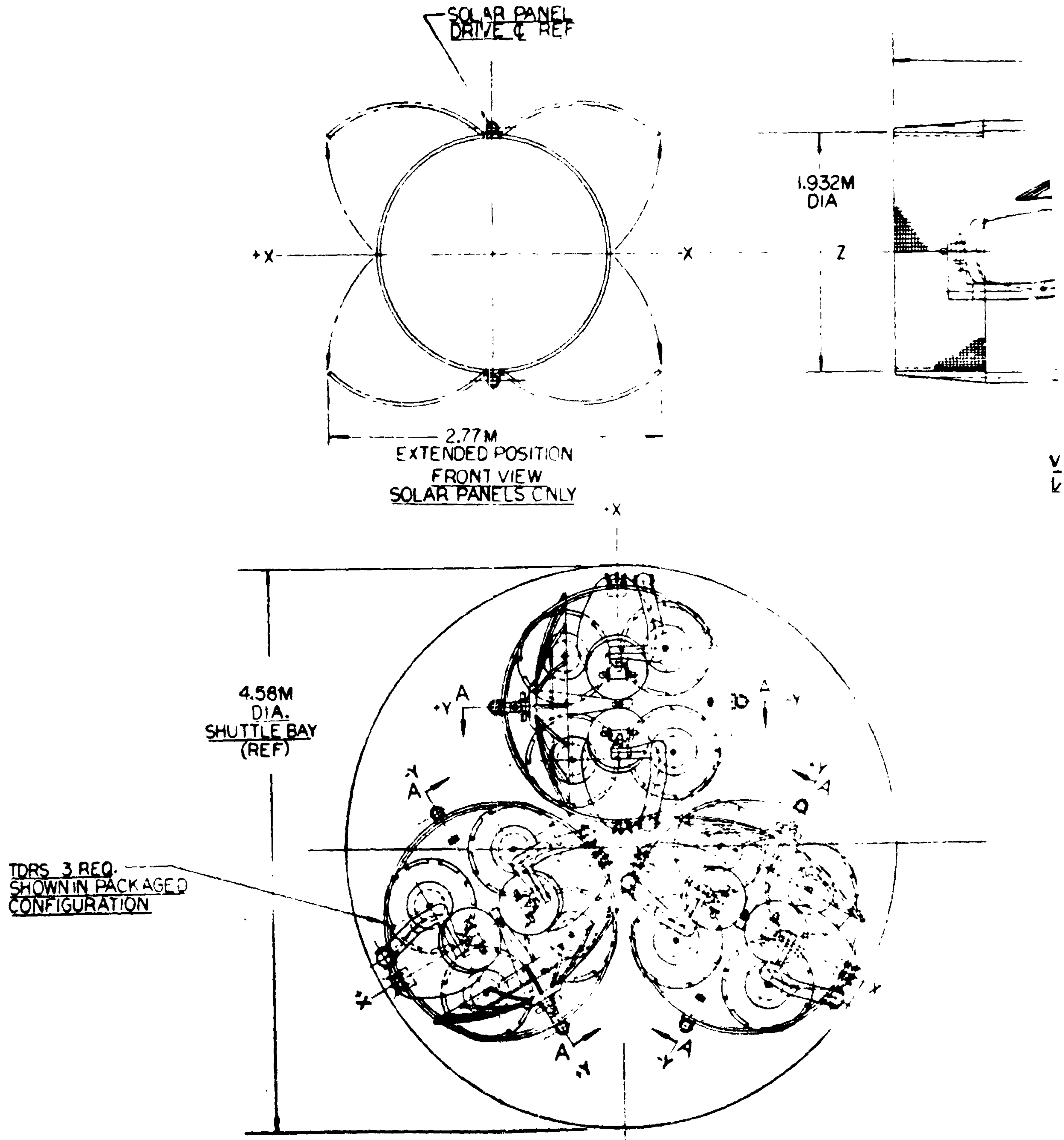
The deployment of each TDRS is achieved as described for the Delta-launched uprated TDRS in Section 2.2.1.1.

This uprated TDRS has the same 308.5 kg weight as the Delta-launched uprated TDRS. With the Shuttle/Agena Tug allowable weight of 487 kg per spacecraft for a triple launch, the contingency per spacecraft is 178.5 kg or approximately 60 percent of the spacecraft dry weight.

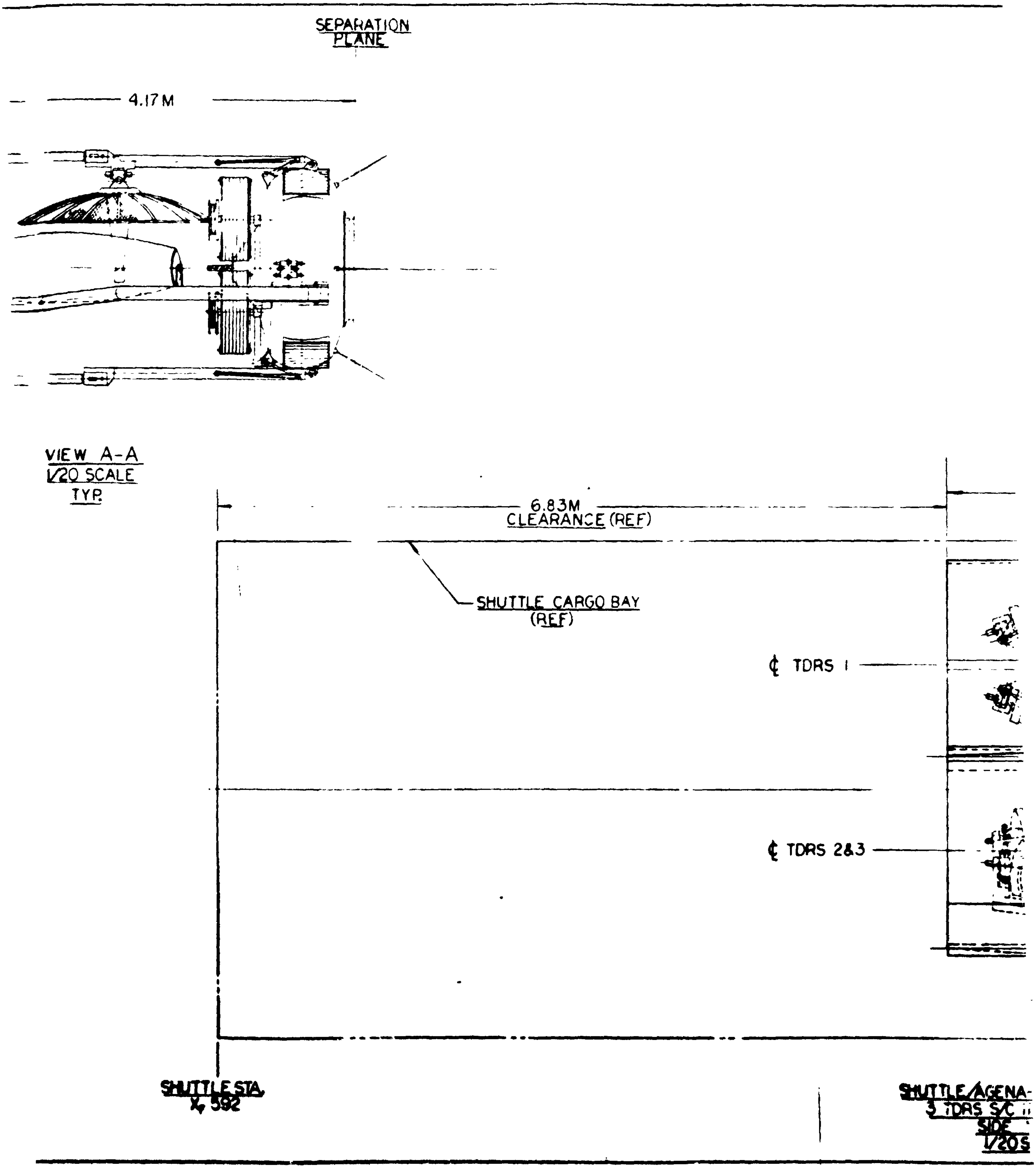
4.3 SHUTTLE/AGENA RELIABILITY CONSIDERATIONS

There is no change in spacecraft reliability when launching three TDRS's with an Agena Tug from the Shuttle. There is, however, a change in overall system reliability due to the differences in booster type and quantity.

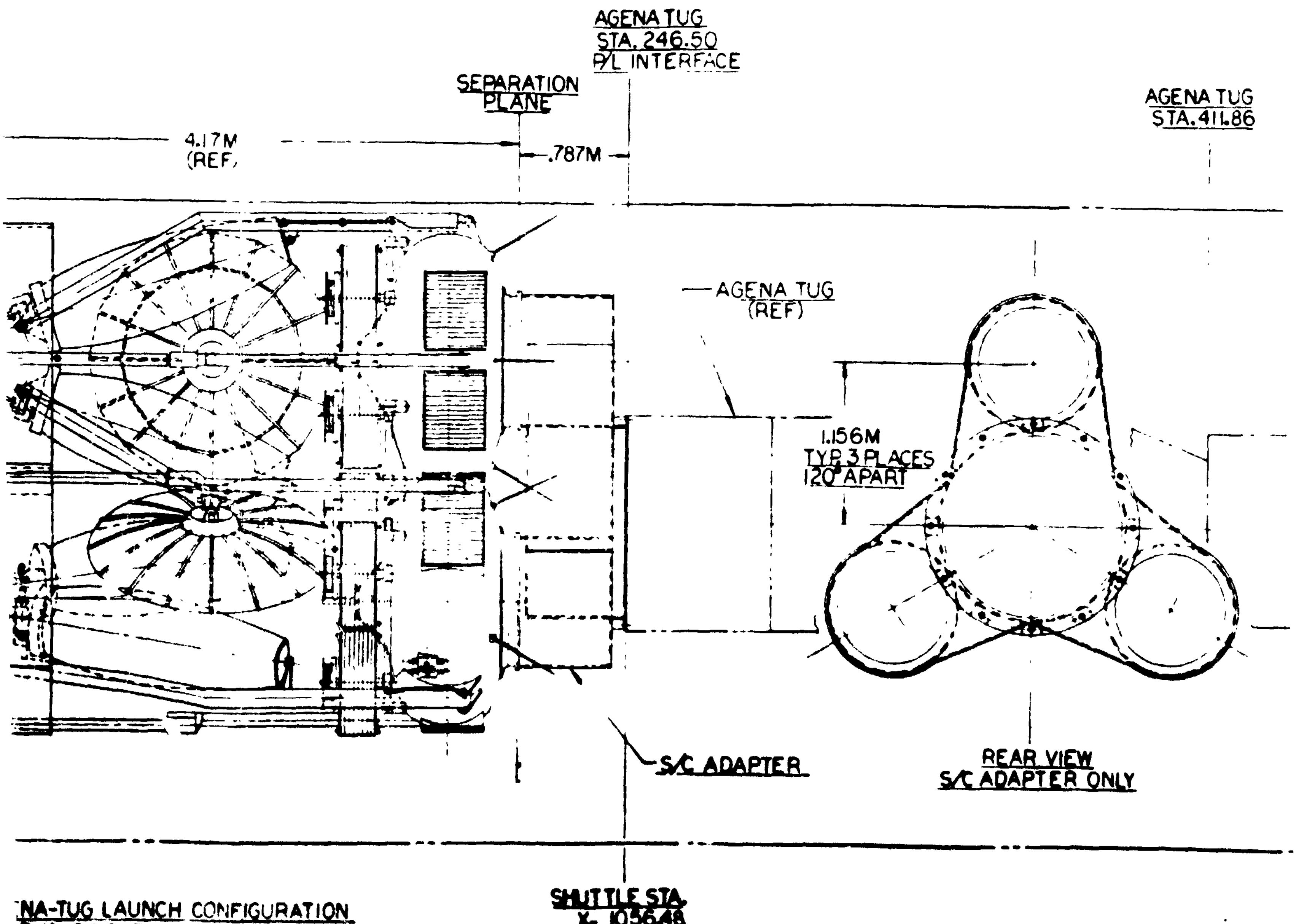
The solid lines of Figure 4-5 show the system reliability of either one or two of three satellites servicing for five years when each is launched with a Delta 2914 booster. The dashed lines represent the corresponding system reliabilities with a cluster launch with an Agena from the Shuttle. In the latter case the overall system reliability is limited by the 0.96 reliability of the Agena booster, whereas triple redundancy exists in launch in the individual launch case. With triple redundancy, two launches can fail and still attain mission success. The overall system reliabilities with individual launches are higher than a cluster launch, even though the reliabilities of the Delta 2914 and the AKS combined are lower than that of the Agena Tug. By



FRONT VIEW
1/20 SCALE

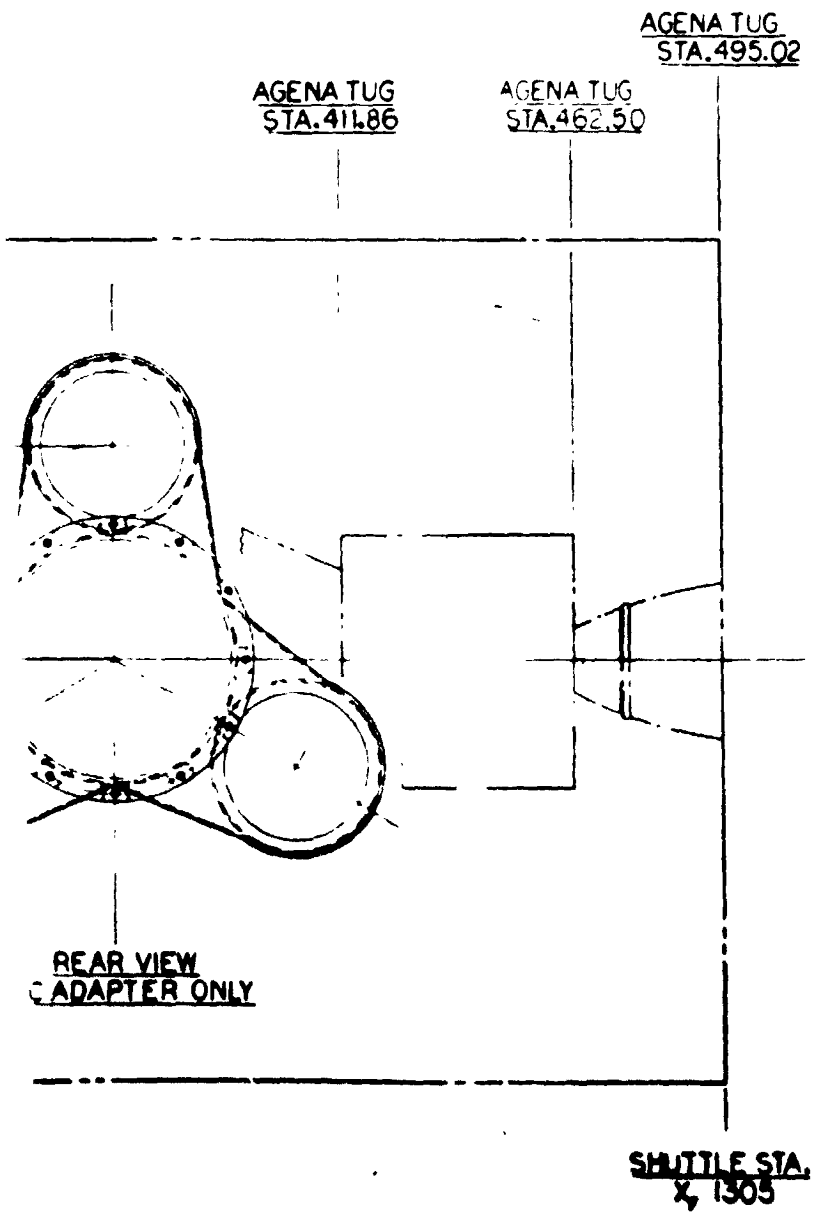


FOLDOUT FRAME 2



NA-TUG LAUNCH CONFIGURATION
IN SINGLE LAUNCH
VIEW
SCALE

PRECEDING PAGE BLANK NOT FILMED



NOTES
FOR TDRS DEPLOYED CONFIGURATION
REFER TO DRWG 1198-54

Figure 4-4. TDRS Shuttle/Agena-Tug Launch, Two 3.8 M Antennas increasing the reliability requirements to where two out of three satellites must survive, the two curves come closer together. The advantage of the higher Agena reliability is now more evenly balanced by the need for two out of three individual launches to succeed. If nonredundant payloads were used, i.e., if three out of three satellites are required to function at the end of the mission, a cluster launch would be more advantageous.

The model utilized in the analyses for the Shuttle/Tug was simplified and the numbers quoted are slightly optimistic since it is assumed that the reliability of the Shuttle launch equals unity and that the quiescent reliabilities of each satellite during Shuttle and Agena boost also is equal to one.

There are, however, indications that the reliability quoted for the Agena Tug is conservative. Lockheed stated that the Agena has a 98-percent injection success rate over the last five years. In that case, the dashed lines of Figure 4-5 will approach a system reliability of 0.98.

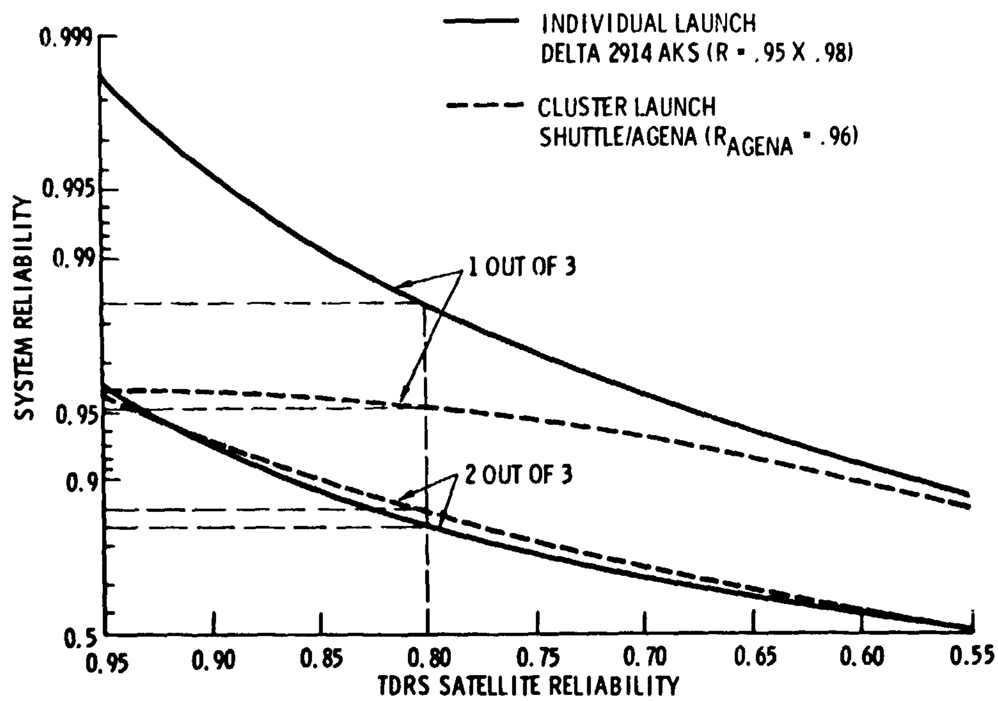


Figure 4-5. System Reliability Versus Satellite Reliability for Three Satellites in Orbit