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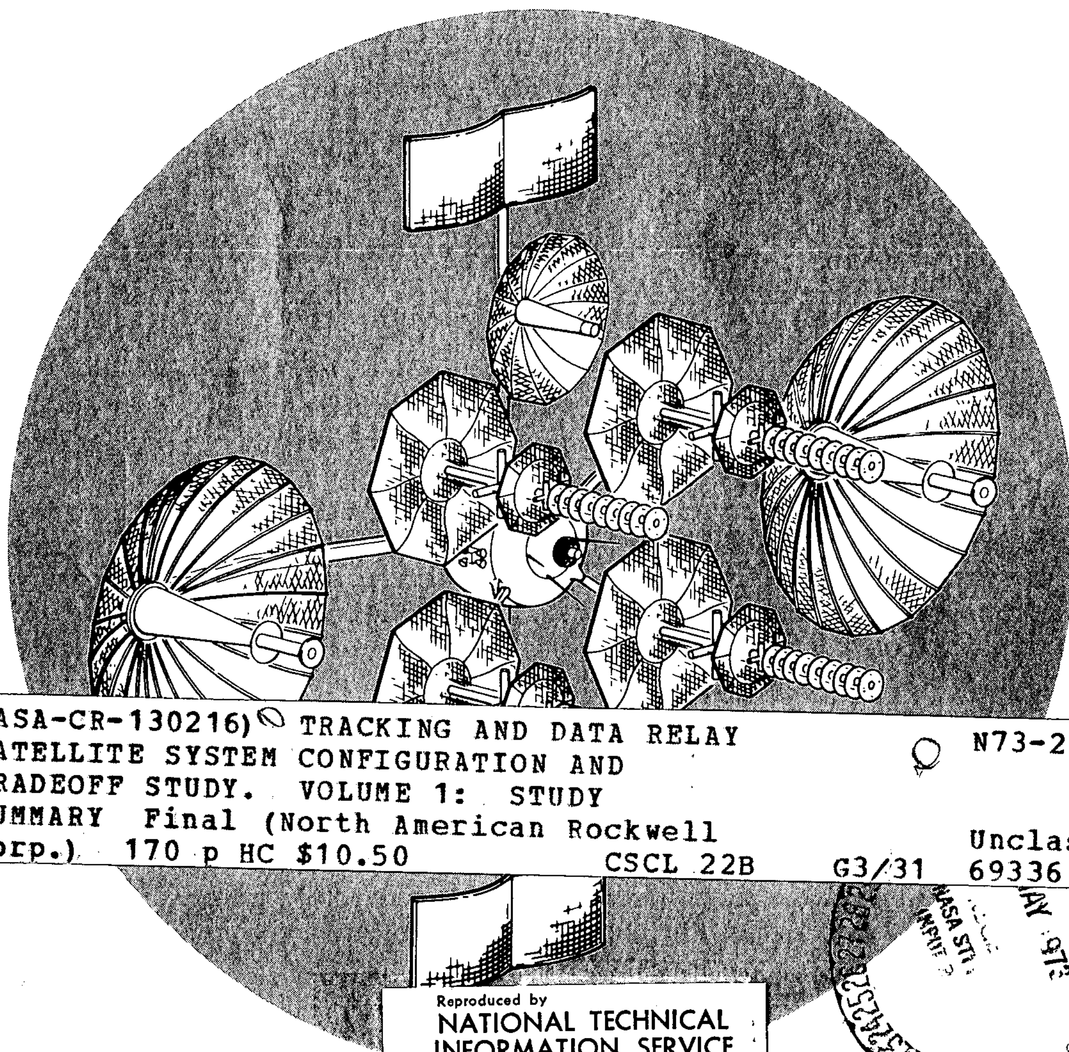
PART II FINAL REPORT

TRACKING & DATA RELAY SATELLITE SYSTEM CONFIGURATION & TRADEOFF STUDY

VOLUME I STUDY SUMMARY



APRIL 1973

SUBMITTED TO
GODDARD SPACE FLIGHT CENTER
NATIONAL AERONAUTICS & SPACE ADMINISTRATION

Space Division
Rockwell International

IN ACCORDANCE WITH
CONTRACT NAS5-21705

12214 Lakewood Boulevard, Downey, California 90241

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PART II FINAL REPORT

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VOLUME I
STUDY SUMMARY

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)	

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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
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W. F. Deutsch	Telecommunications Design
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A. Forster	Cost
A. F. Anderson	Integration

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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.

CONTENTS

ILLUSTRATIONS

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TABLES

1.0 INTRODUCTION

The study summarized in this volume was conducted in two parts. The intent of Part I was to provide the NASA with a definition of a Tracking and Data Relay Satellite System optimized for support of low and medium data rate user spacecraft and constrained to a Delta 2914 launch. This definition was accomplished to a preliminary design level of detail consistent with preparation of a final design specification, including cost estimates. Originally, Part II was to be concerned with: (1) considering and defining design variations and cost deltas associated with increasing the relay capacity of the Part I baseline design to include a high data rate service, and (2) synthesizing conceptual designs for a maximum capacity spacecraft constrained to Atlas and Space Shuttle launches. Emphasis was to be on the Part I activity, where not only was a practical low cost/low risk spacecraft and telecommunications design established, but also the major system and technology problems and trades were addressed and resolved.

Part I of the study was conducted essentially as proposed with some changes in technical guidelines and direction from the NASA/GSFC project office as technical problems were solved and tradeoff studies completed. Part II varied from the proposed general plan; not in total scope, but in emphasis on tasks and in design goals. These variations were the logical result of considering the system analyses and design conclusions of Part I. Basically, greater emphasis was placed on reconfiguring the Part I baseline spacecraft into a design that would support, low, medium, and high data rate users and in defining the spacecraft at a level of detail consistent with the Part I design. This did not eliminate consideration of Atlas- and Shuttle-launched configurations, but constrained design and analysis activities to a conceptual level of detail.

1.1 STUDY GUIDELINES

The key ground rules under which the study was initiated are:

- General
  1. Only 3-axis stabilized configurations will be considered.
  2. The telecommunications relay capacity will be maximized.
  3. The minimum spacecraft lifetime will be five years.
  4. Use of existing technology and equipment will be maximized.
  5. The flexibility of telecommunications relay service will be maximized.

- Part I

  1. The launch vehicle is a Delta 2914
  2. The minimum telecommunications relay capacity will support 20 LDR and 1 MDR users simultaneously.
  3. The MDR link will satisfy the requirements of a two-way voice channel.
  4. 1975 is the cut-off date for technology state-of-the-art.

- Part II

  1. The launch vehicles are a Delta 2914, an Atlas class, and a Space Shuttle.
  2. The minimum telecommunications relay capacity will be 20 LDR and 1 HDR users simultaneously.
  3. The HDR link will satisfy the requirements of a two-way voice channel.
  4. Communications characteristics for Shuttle launches was established by MSC.

As the spacecraft designs evolved and analyses were completed, particularly during Part I, some changes were made to existing ground rules and additional ground rules were added as follows.

- General

  1. TDRS forward link flux densities will be consistent with CCIR regulations.
  2. Frequency selections will be consistent with IRAC allocations.

- Part I

  1. The minimum telecommunications relay capacity was increased to support 20 LDR, and 2 MDR users simultaneously.
  2. The MDR link will provide service to and be compatible with the Shuttle.

- Part II

  1. Multiple launches of a Shuttle/TDRS will be emphasized during Part II/Phase II.
  2. The minimum telecommunications relay capacity will support 20 LDR, 1 MDR, and 1 HDR users and will be launched on a Delta 2914.
  3. The Atlas class vehicle is an Atlas-Centaur and the Shuttle luanch configuration is a Shuttle-Agena.

These ground rules provided the bases for establishing tradeoff study and design decision criteria, and along with directions provided the NASA/GSFC TDRS Project Office at the progress meetings, guided the design and analyses activities.

Throughout the following discussions, the spacecraft and telecommunications design developed during Part I will be referred to as the baseline configuration; the configuration developed during Phase I of Part II will be referred to as the uprated baseline; and the configurations synthesized during Phase II of Part II will be referred to as Atlas-Centaur Shuttle-Agena concepts or the maximum capability TDRS.

1.2 RESULTS AND CONCLUSIONS

In the following two sections the major conclusions that resulted from the study are summarized according to Part I and Part II. Before reviewing those conclusions it is important to point out the most basic recommendation set forth by the study in general. This concerns the selection of the TDRS spacecraft and telecommunications system considered optimum for tracking and data acquisition services in the post 1977 time period.

Two configurations are recommended as being optimum for the first generation system, and selection between the two is dependent on the telecommunications characteristics of the user spacecraft mission model yet to be determined. These recommended design approaches are illustrated in Figures 1-3 and 1-5.

The design shown in Figure 1-3 is referred to as the Delta Launched Uprated Baseline and that shown in Figure 1-5 is referred to as alternative 2 to the Delta Launched Uprated Baseline. The former is considered to be optimum if the majority of user spacecraft are in the Low Data Rate category or in the VHF/UHF frequency spectrum. The latter configuration is considered the optimum approach if the majority of the user spacecraft are planned to be in the Low Data Rate/Medium Data Rate category or in the S-band frequency spectrum.

Both of these designs were developed during the first phase of Part II of the study. However, the basic satellite designs including subsystems are nearlly identical to the Part I baseline configuration. The major variations to the part I baseline are in the design of the deployable antennas and in the TDRS to ground portion of the telecommunications subsystem. Considerably more time was spent in the design, analysis, and definition of the Delta Launched Uprated Baseline (Figure 1-3) than on alternative 2 (Figure 1-5), however the alternative received sufficient attention to insure that it is a practical variation to the uprated baseline.

1.2.1 Results and Conclusions--Part I

During Part I, study efforts were directed at defining all TDRS system elements with emphasis on synthesss of a space segment design optimized to support low and medium data rate user spacecraft and launched with a Delta 2914. In addition to developing a preliminary design of the satellite, conceptual designs of the user spacecraft terminal and TDRS ground station were developed.

It was determined that a three-axis stabilized tracking and data relay satellite launched on a Delta 2914 provides telecommunications services considerably in excess of that required by the study statement of work. Further, the design concept supported the needs of the Space Shuttle as defined at that time, and had sufficient growth potential and flexibility to provide telecommunications services to high data rate users.

The spacecraft and telecommunications system uses state-of-the-art technology and equipment in all instances. This, combined with a great deal of attention to reliability, produced a very low-risk design. The major operational attributes of the

1. Provides multiple access to both low and medium data rate users.

2. Provides two frequency options (S- or Ku-band) to medium data rate users.

3. Provides the basis for eventual support to high data rate users through the incorporation of Ku-band.

4. Provides the optimum approach to combating interference in the low data rate (VHF and UHF frequencies) channels.

5. Meets the Shuttle support requirements.

6. In the event of spacecraft subsystem or telecommunications equipment failures, satellite performance degrades gracefully.

From a spacecraft design standpoint some of the major attributes, in addition to very high reliability, are:

1. A weight contingency of 37.2 kg (82 lb).

2. A power contingency at end of life of approximately 40 watts during normal telecommunications operations and a power contingency of 6 watts during emergency UHF voice service to the Space Shuttle. The requirement for UHF voice was later found to be nonexistant.

3. A logical configuration for considering growth to higher power levels and larger antennas.

The telecommunications service modes provided by the 3-axis stabilized baseline satellite design are as summarized in Table 1-1.

Figure 1-1 is an illustration of the Delta 2914-launched baseline TDRS which provides the preceding modes of service. This configuration and its telecommunications capabilities will be discussed further in Section 2.0 of this volume.

Table 1-1. Modes of Service - Part I Baseline

Links	Modes of Services	Remarks
1. LDR · Return	· AGIPA Mode	Adaptive multi-beams to simultaneously service 20 LDR users with optimum signal-to interference ratio: G/T_s = -14.4dB/°K*
	· F-FOV Mode (backup)	A broad fixed beam that views all 20 LDR users: G/T_s = -18.8dB/°K*
· Forward	· Steerable Beam Mode	Provides 2 ground controlled high gain satellite steered beams for data at EIRP of +30 dBw, or data and/or voice at EIRP of +36 dBw (25% duty cycle)
	· F-FOV Mode	One of the above steerable beams can be replaced with a broad fixed beam to illuminate all 20 users for coherent ranging. The EIRP of 24 dBw can also be used for reduced data or voice.
2. MDR #1 · Return	· Dual Frequency S-band Mode	To support current S-band users; G/T_s + 4.7 dB°K
	Ku-band Mode	To support future high performance Ku-band users; G/T_s = 20.4dB°K
· Forward	· Data (Specification) Mode	Transmit EIRP of +41 dBw and 45.6dBw at S-and Ku-bands, respectively, at scan limits.
	· Shuttle Mode	Transmit EIRP of +47 dBw at scan limit for data at 2 kbps and 2 voice at 19.2 kbps.
· Return & Forward	· TDRS/GS Backup Mode	Antenna and/or transceiver can provide 100% functional redundancy for TDRS/GS antenna or transceiver at S- or Ku-bands
3. MDR #2 · Return	· Same as MDR #1 but does not have TDRS/GS backup capability	MDR #1 plus MDR #2 provides simultaneous support to: two S-band users; or two Ku-band users; or one S-band and one Ku-band user
4. TDRS/GS · Return	· Primary Mode	Primary mode after TDRS is "on" station; G/T_s = 6.0dB/°K
	· MDR #1 Backup Mode	Provides functional backup to primary mode at S- or Ku-band; G/T_s = 4.7 and 20.4dB/°K at S- and Ku-bands, respectively.
	· VHF Mode	TT & C subsystem provides VHF backup "on" station; becomes prime during inflight transit phases; G/T_s = -28.8dB/°K
· Forward	· Primary Mode	EIRP = 44.6 dBw
	· MDR #1 Backup Mode	EIRP = 41 and 45.6 dBw at S- and Ku-bands, respectively
	· VHF Mode	EIRP = 3 dBw
5. Tracking/ Order Wire · Return	· TDRS Tracking Mode	Provides trilateration ranging signal for TDRS spacecraft tracking and position location function when used in conjunction with 2 remote GS and main TDRS GS; G/T_s = -15.5 dB/°K
	· S-Band Order Line Mode	Provide order wire to establish priority access to MDR transponder; G/T_s = -15.5 dB°K
· Forward	· S-band Beacon Mode	Provides acquisition and tracking source for S-band MDR users with steerable antennas; EIRP = 14.5 dBw
	· TDRS Tracking Mode	EIRP = 14.5 dBw
6. Ku-band Beacon	· Beacon Mode	Provides acquisition and tracking source for Ku-band MDR users with steerable antennas; EIRP = 13 dBw
7. Frequency Source	· Slave Mode	Coherently locks on to pilot signal from GS
	· Prime Reference Mode	Becomes prime reference for telecommunication system and GS coherently locks on to Ku-band beacon

* Assumes antenna temperature of approximately 800°K; however, T in this frequency is a variable.

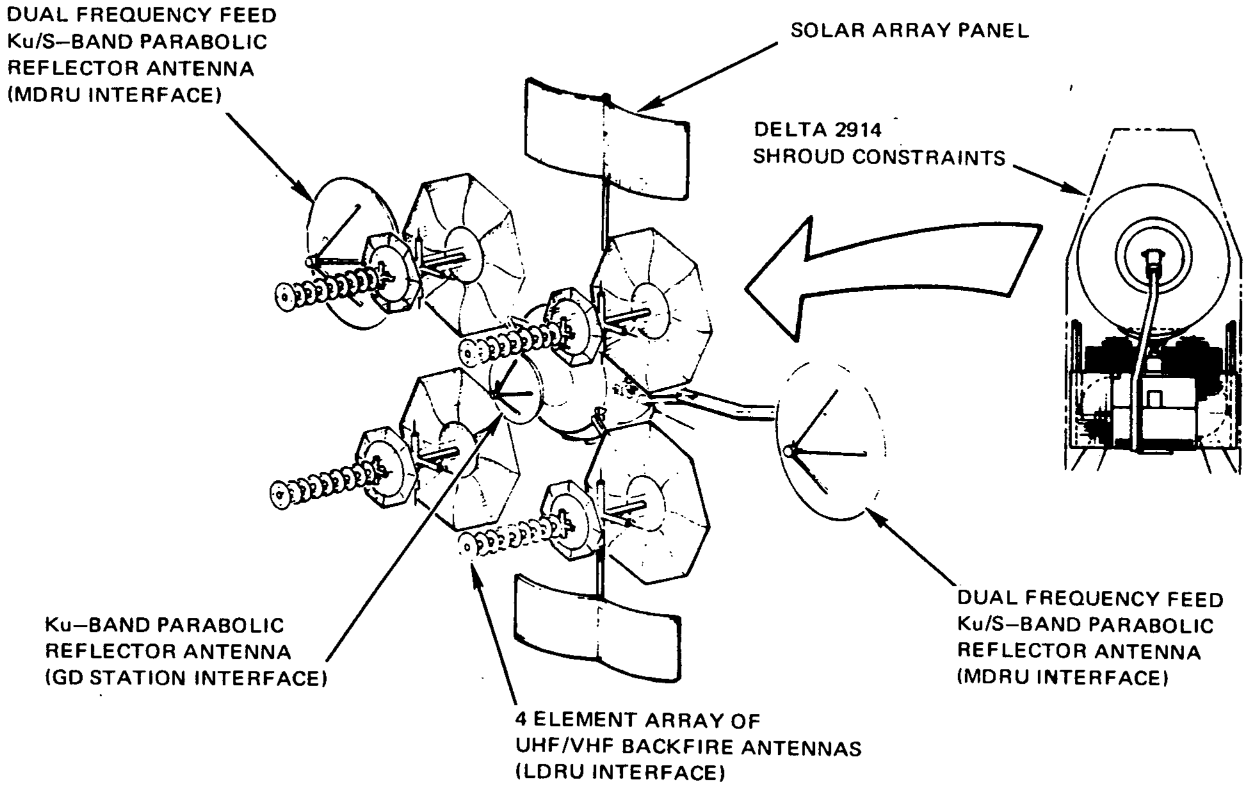

Figure 1-1. Delta 2914 Baseline TDRS

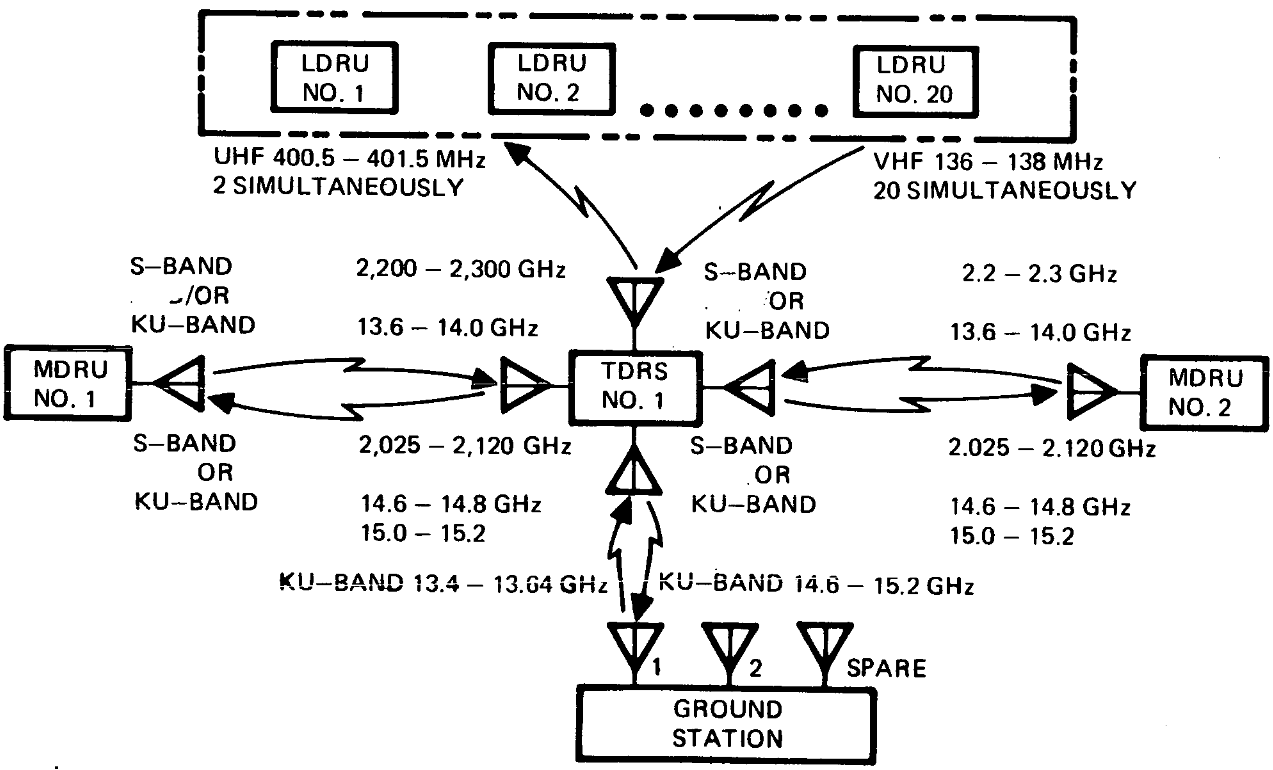

Figure 1-2. General RF Interface

The general RF interfaces between the baseline TDRS and the LDR and MDR user spacecraft and ground station are shown in Figure 1-2, along with the selected frequency bands.

1.2.2 Results and Conclusions--Part II

During Phase I of Part II, study efforts were directed at synthesizing design concepts of a Delta-launched TDRS that conceivably could provide the telecommunications relay service to low, medium, and high data rate users, and analysis of these concepts to determine the maximum performance available and any constraints on telecommunications operations. It was determined that this level of service was not only feasible but practical and that the operational constraints are minimal. As a result of these findings it was concluded that this configuration should be considered an uprated baseline design and defined to the same level of detail as the Part I baseline.

As in Part I, low risk was considered to be a very important design criterion. In view of this, only state-of-the-art technology and equipment were used, and mechanical complexity was minimized. The major attributes of the uprated baseline design are that it:

1. Provides multiple access to 20 LDR's and 2 MDR's or 20 LDR's, 1 MDR, and 1 HDR with S- and Ku-band frequencies available to the MDR and HDR users.

2. Provides the optimum approach to combating interference in the low data rate (VHF and UHF frequencies) channels.

3. Meets the increased requirements of the Shuttle.

4. In the event of spacecraft or telecommunications equipment failure, satellite performance degrades gracefully.

5. Block redundancy (cross-strapping of transceivers) ensures that each telecommunications mode has a backup.

6. A spacecraft weight contingency of 26.3 kg (58 lb).

A very high reliability goal was achieved, with the only increase in mechanical complexity being the use of deployable parabolic reflector antennas for the MDR/HDR interfaces. The telecommunications modes of service provided by the uprated baseline satellite design are summarized in Table 1-2.

The major changes in the modes of service from the Part I baseline are:

1. The addition of the high data rate capability necessitating design changes in the space-to-space microwave antennas and the TDRS-to-ground transmitter and antenna.

2. The use of a fixed field-of-view transmitter/antenna approach for the LDR forward link, to minimize operations work load at the ground station.

TABLE 1-2. MODES OF SERVICE--PART II, UPRATED BASELINE

Links	Modes of Services	Remarks
LDR:		
·Return	·AGIPA Mode	·Adaptive multi-beams to simultaneously service 20 LDR users with optimum signal-to-interference ratio: G/T_s = -13.7 db/K
	·F-FOV Mode (backup)	·A broad fixed beam that views all 20 LDR users: G/T_s = 16.3 db/K
	·F-FOV Mode	·The EIRP during daylight normal operation is 30 dbw, and during eclipse it is normally +27 dbw or +30 dbw (with a 50% duty cycle constraint). There is one forward link channel and users are commanded sequentially.
·Forward	·Steerable Beam Mode	·Provides ground controlled high gain satellite steered beam for data at EIRP of +36 dbw, or 39 dbw, or 42 dbw burst)
MDR/HDR #1 and #2:		
·Return	·Dual Frequency: S-band Mode	·To support manned and unmanned S-band users: G/T_s + 10.0 db/K
	Ku-band Mode	·To support future high performance manned or unmanned Ku-band users: G/T_s = 25.9 db/K
·Forward	·S-band Mode	·Transmit EIRP of +41 dbw and 47 dbw for unmanned and manned, respectively
	·Ku-band Mode	·Transmit EIRP of 23.6 dbw and 53.6 dbw for unmanned and manned, respectively
·Return & Forward	·TDRS/GS Backup Mode	·Antenna and/or transceiver can provide 100% functional redundancy for TDRS/GS antenna or transceiver at S- or or Ku-bands with approximately 6 db increase in link margin
TDRS/GS:		
·Forward	·Primary Mode	·Primary mode after TDRS is "on" station: G/T_s = 12.8 db/K
	·Backup Mode	·Either MDR/HDR provides functional backup to primary mode at S- or Ku-band: G/T_s = 10.0 and 25.9 db/K at S- and Ku-bands, respectively
	·VHF Mode	·TT&C subsystem provides VHF backup "on" station; becomes prime during inflight transit phases: G/T_s = -28.8 db/K
·Return	·Primary Mode (FDM/FM/FDM)	·EIRP = 56.7 dbw high power mode, 49 dbw low power mode
	·Backup Mode	·EIRP = 47 and 53.6 dbw at S- and Ku-band, respectively
	·VHF Mode	·EIRP = 3 dbw
Tracking/Order Wire:		
·Return	·TDRS Tracking Mode	·Provides trilateration ranging signal for TDRS spacecraft tracking and position location function when used in conjunction with two remote GS and main TDRS GS: G/T_s = -15.5 db/K
	·S-band Order Line Mode	·Provide order wire to establish priority access to MDR transponder: G/T_s = -15.5 db/K
·Forward	·S-band Beacon Mode	·Provides acquisition and tracking source for S-band MDR users with steerable antennas: EIRP = 14.5 dbw
	·TDRS Tracking Mode	·EIRP = 14.5 dbw
Ku-band Beacon	·Beacon Mode	·Provides acquisition and tracking source for Ku-band MDR users with steerable antennas: EIRP = 13 dbw
Frequency Source	·Slave Mode	·Coherently locks on to pilot signal from GS
	·Prime Reference Mode	·Becomes prime reference for telecommunication system and GS coherently locks on to Ku-band

3. The reduction from two to one in the number of LDR forward links, necessitated by the increased power requirements of the LDR Fixed FOV approach in the forward link and the increased bandwidth of the TDRS--GD link.

4. The elimination of the LDR voice requirement.

These service modes are provided by the TDRS configuration shown in Figure 1-3. The major differences between this configuration and the Part I baseline are in the telecommunications subsystem and the antenna designs. The spacecraft structure and its subsystems are basically the same as in the baseline configuration.

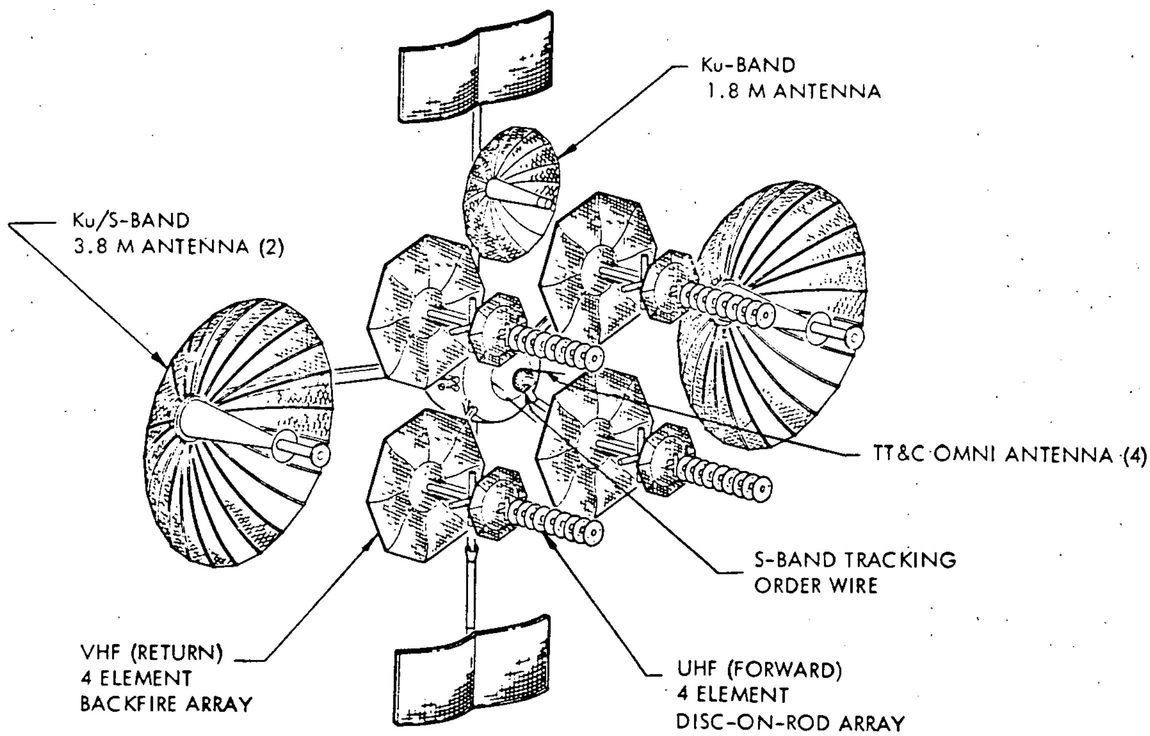

Figure 1-3. Delta 2914 Uprated Baseline TDRS

The general RF interfaces and frequency selections for this design are very similar to the Part I baseline. The only differences are that either one of the MDRU's can be an HDRU and the frequency associated with this interface is S-band and/or Ku-band rather than S-band or Ku-band.

Two variations to this uprated baseline configuration were considered late in the study as a result of analyzing more recent NASA mission models and in an effort to provide a design that required the deployment of fewer appendages while still providing the high data rate service. The first alternative approach shown in Figure 1-4 simply eliminated the LDR portion of the telecommunications subsystem and the UHF/VHF antennas with the weight and power savings being considered an increase in weight contingency.

The rationale that allowed consideration of this approach is derived from the fact that recent NASA earth orbital mission models indicate a potentially very small number of spacecraft in the low data rate category and an increase in the number of medium data rate user spacecraft or S-band frequency users. Thus, the power penalty associated with the use of S-band would be imposed on a relatively few spacecraft that would insist on using omni-directional antennas. Their alternative to incurring this penalty would be to constrain their communications to visibility times over the reduced number of ground stations.

The other variation to the uprated baseline (alternative 2),based on the same mission model reasoning, eliminated the UHF/VHF transceiver and replaced it with a multi-channel S-band phased array. The AGIPA concept was retained as an integral part of this link. This concept is illustrated in Figure 1-5.

The S-band array is 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 centerline because of the lowering of the solar shadow lines from the HDR/MDR antennas.

The TDRS/GS antenna diameter was increased from 1.8 M to 2.0 M to minimize link power. Eliminating the LDR UHF-VHF elements permitted this increased diameter.

The major attributes of these alternatives are:

1. Simpler mechanical design

2. Higher weight contingency

3. Avoids the potential RFI problem in the VHF spectrum

These attributes are gained at the expense of system flexibility since all user spacecraft would be forced to communicate in the microwave region. Alternative 2 is the more flexible of the two variations to the uprated baseline, in that it retains a multiple-semirandom access capability at S-band for up to 20 LDR/MDR users as well as the two high performance interfaces (12.5 ft parabolic reflectors) at S- or Ku-band.

In Phase II of Part II, attention was directed at synthesizing conceptual designs for an Atlas-Centaur launched TDRS and a Shuttle-Agena launched TDRS.

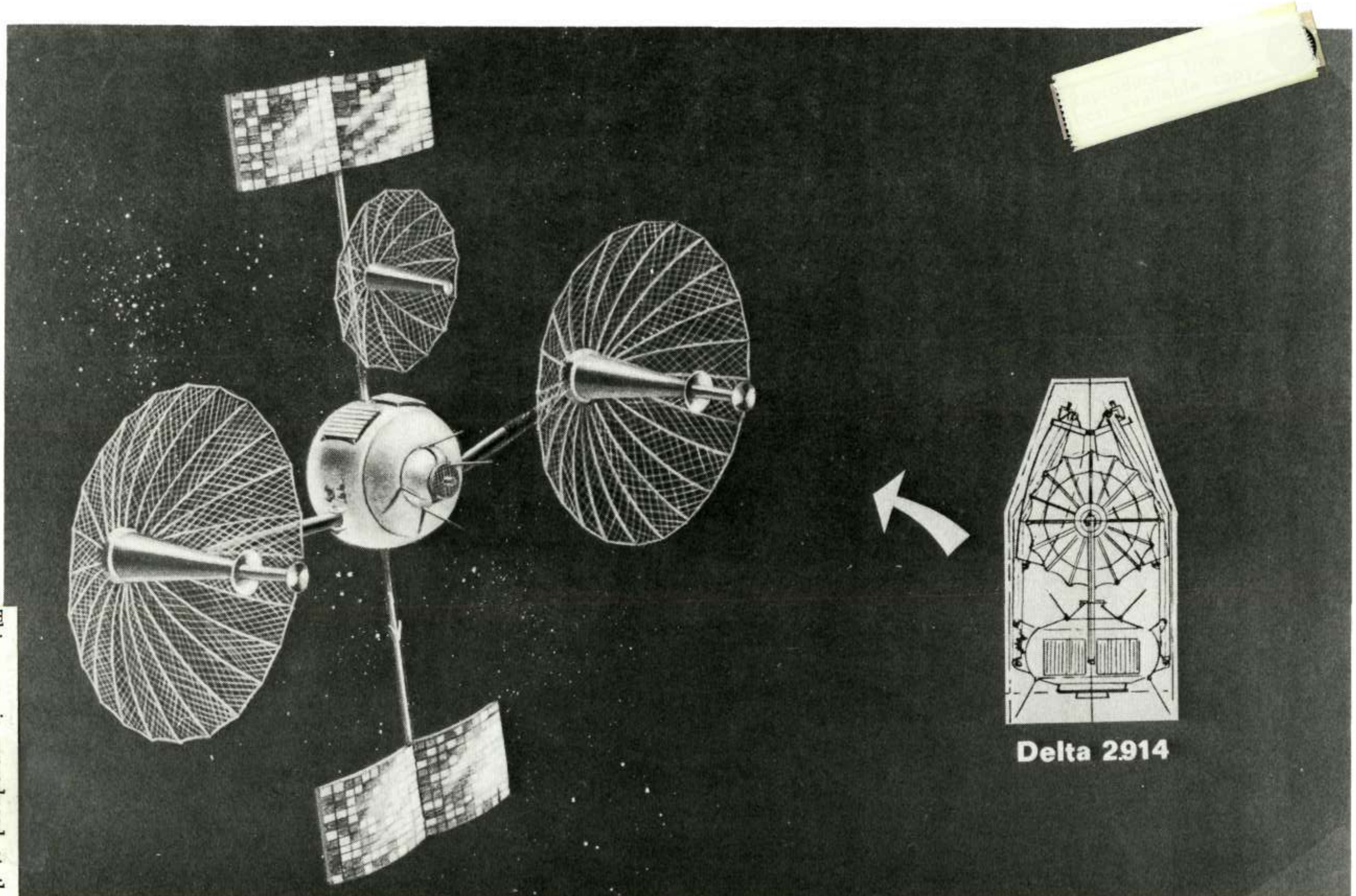

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Figure 1-4. Alternative 1 to Uprated Delta 2914 Baseline TDRS

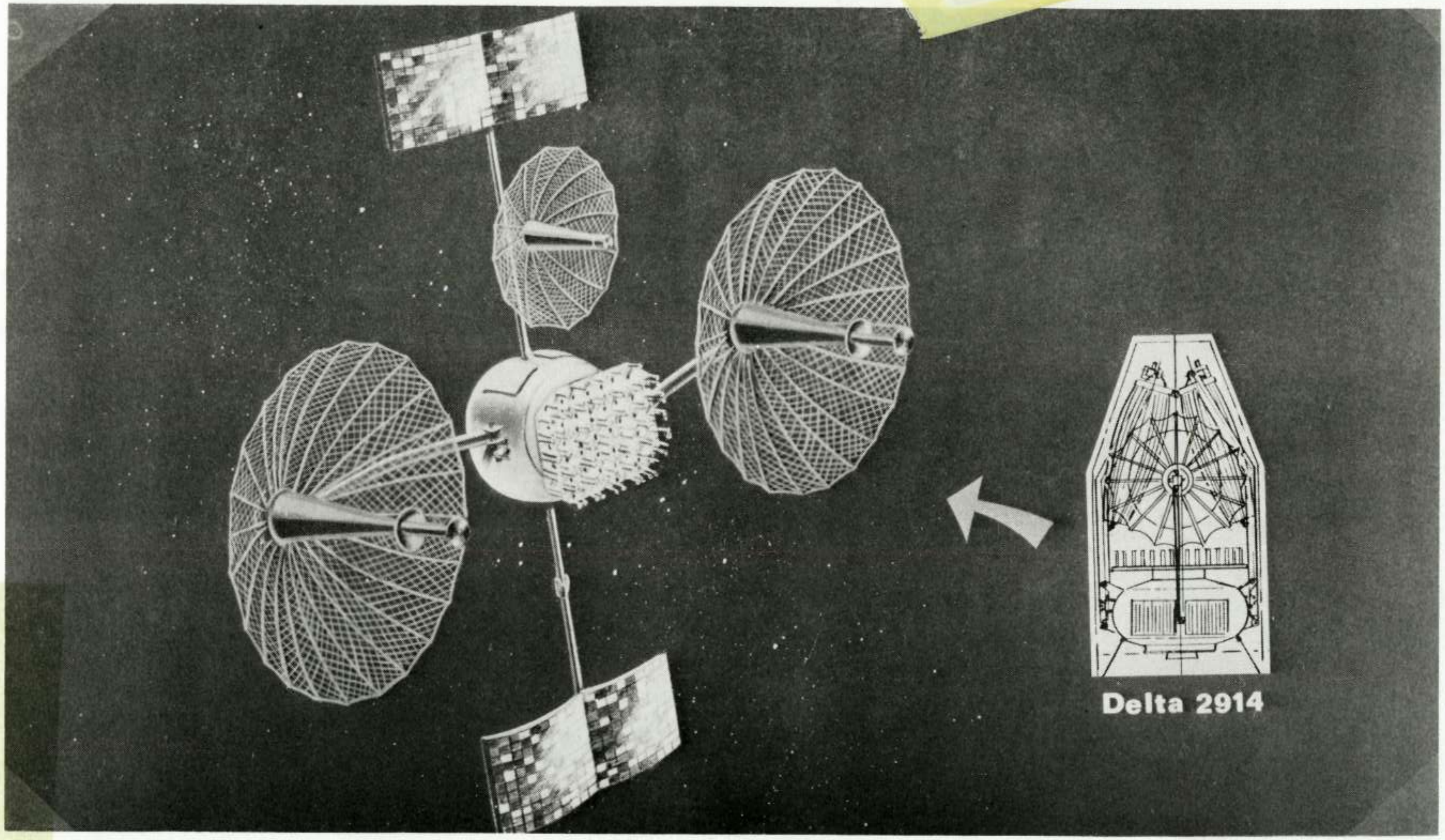

Figure 1-5. Alternative 2 to Uprated Delta 2914 Baseline TDRS

In both cases it was determined that a considerable increase in relay capacity could be obtained at an equally large increase in satellite cost. Further, analysis of projected support requirements does not indicate any requirements for relay capacities in excess of that provided by the Delta 2914 uprated baseline. In view of this, equal attention was given to the feasibility of using the Shuttle-Agena to launch multiple TDRS (three at a time). Multiple launches using the Atlas-Centaur also were considered but packaging constraints of the shroud make the mechanical and structural considerations unattractive. Consequently, only high-capacity single launches were considered for Atlas-Centaur.

The Atlas-Centaur/TDRS configuration is illustrated in Figure 1-6. It provides simultaneous support to four MDR/HDR users and 20 LDR users.

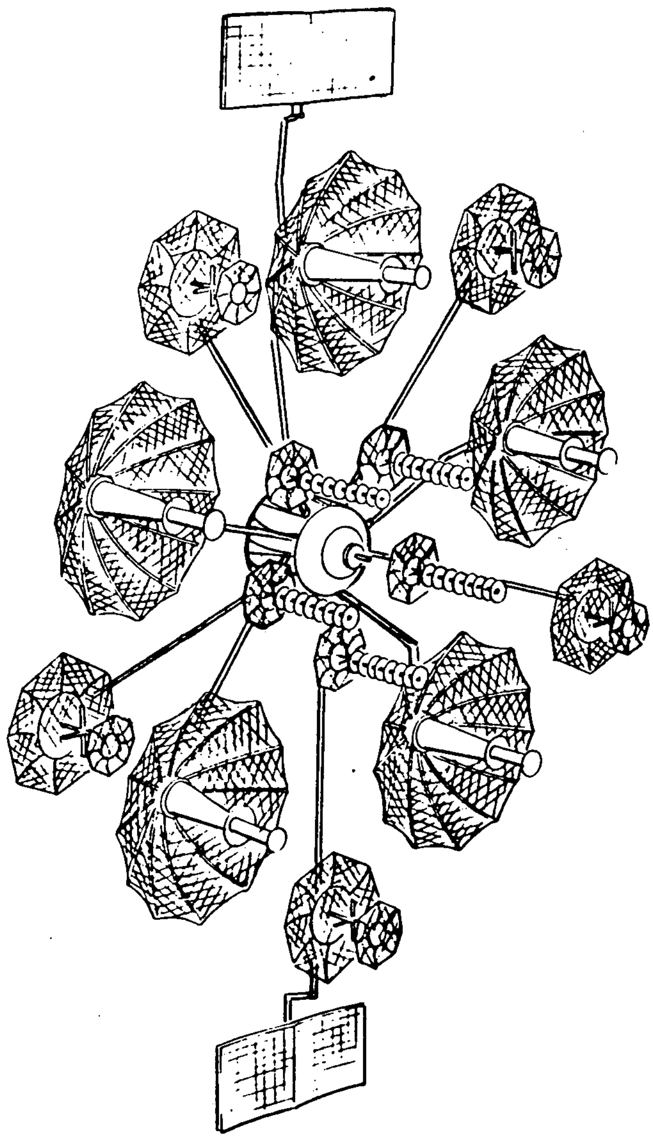

Figure 1-6. High Capacity Atlas-Centaur or Shuttle-Agena Launched TDRS

Two Shuttle-Agena/TDRS design concepts were developed. One is essentially the same as the uprated Delta 2914 baseline configuration when deployed and the other is the same as the high-capacity configuration designed for the Atlas-Centaur but packaged for launch three at a time.

2.0 PART I REVIEW (DELTA 2914/LDR AND MDR)

In this section the design and analysis activities conducted during the first part of the study are summarized. As indicated earlier, the reader is referred to the Part I Final Report for in-depth documentation of technical activities and findings.

2.1 SYSTEM CONCEPT

Although the present tracking and data acquisition network provides a sophisticated and comprehensive support service to the earth orbital space program, it does have practical limitations with respect to spacecraft access time and information bandwidth that impose both design and operational constraints on these spacecraft. To remove the mission-constraining time-bandwidth limitations of this supporting network and minimize the cost of the service, NASA is considering the implementation of the network concept entitled "Tracking and Data Relay Satellite System" that includes the use of geosynchronous communication satellites. These communication satellites will relay information to and from earth orbital spacecraft, using a single primary ground station in the United States, augmented by a reduced number of remotely located ground stations.

The programs that will require support from, and benefit by, the TDRS system include scientific satellites, earth application and observation satellites, and manned spacecraft such as Space Shuttle. The needs of these users include increased coverage, higher data transmittal rates, decreased reliance on low reliability components such as tape recorders, and real-time data acquisition and command capabilities, most of which are beyond the capabilities of the existing or planned ground tracking and data acquisition network.

The user spacecraft were categorized according to their range of support requirements as low data rate users (LDRU) whose data transmittal requirements are $\leq$ 10 kbs, medium data rate users (MDRU) whose data transmittal requirements are $\leq$ 1 mbs, and high data rate users (HDRU) whose data transmittal requirements are $>$ 1 mbs. The channels in the TDRS system over which the user spacecraft transmit are defined as return links, and the channels over which the user spacecraft receive are defined as forward links. Forward link requirements for LDRU's, MDRU's, and HDRU's are $\lesssim$ 1 kbs, $\lesssim$ 1 kbs, and $>$1 kbs, respectively. In addition, the Shuttle requirements were specified and for the purpose of this study have been categorized as an MDRU with unique requirements such as voice and a forward link data rate of 2 kbs. The Shuttle requirements changed in Part II, necessitating higher performance from the TDRS. During Part I, only LDRU's and MDRU's were considered. HDRU support was considered during Part II.

The system deployment concept synthesized during this study is illustrated in Figure 2-1 and consists of three satellites in synchronous near equatorial orbit. Two satellites are operational and are located at 11°W and 141°W longitude.

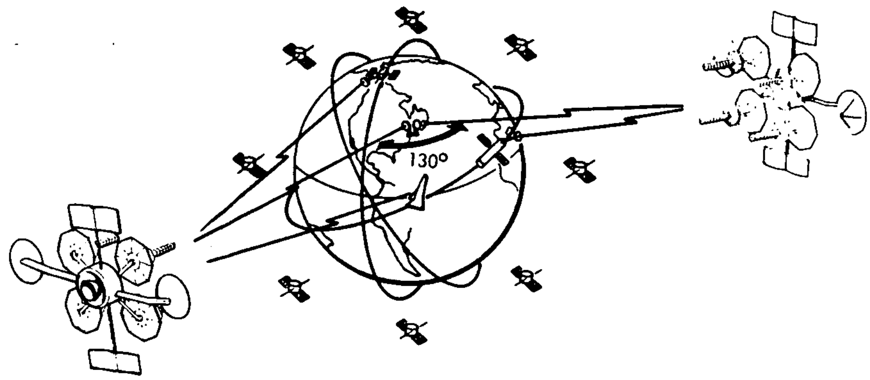

Figure 2-1. System Deployment Concept

These positions are consistent with continuous visibility to a ground station located at Rosman, N.C., with minimum elevation angles of 10 degrees. A third satellite (not shown in Figure 2-1) is an on-orbit spare located equidistant between the two operational satellites. Each vehicle is in an orbit with an initial inclination of 2.5 deg (.044 rad). This approach maximizes user spacecraft coverage within the constraint of operating the ground station at Rosman. As will be seen in the launch and deployment analyses, higher inclinations can be used, resulting in some decrease in user spacecraft coverage and an increase in useful payload capability or an increase in spacecraft weight contingency.

The satellite designed for this deployment scheme, illustrated earlier in Figure 1-1, provides telecommunications relay capability considerably greater than requirement by the statement-of-work spacification. Each TDR satellite can simultaneously maintain:

- Two LDRU forward links; each capable of carrying voice and data (spec requirement is one LDR forward link with voice and data). Power limitations permit voice on only one link at a time with 25% duty cycle.
- Twenty LDRU return links.
- Two MDRU forward links at both S- or Ku-band (spec requirement is one MDR forward link at either S- or Ku-band).
- Two MDRU return links at S- and/or Ku-band (spec requirement is one MDR return at S- or Ku-band).

The primary telecommunications relay interfaces are two mechanically steerable 2 m (6.5 ft) parabolic reflector antennas, each of which can transmit and receive at S- or Ku-band to support the MDRU's; a four-element array of backfire antennas that transmit at UHF and receive at VHF to support the LDRU's, and a 0.9 m (3 ft) Ku-band parabolic reflector to provide the transceiver interface with the ground station. An alternative configuration was synthesized replacing one 2 m (6.5 ft) parabolic reflector with an S-band phased array. If it is determined that multiple access (greater than two simultaneously) for MDRU's is more important than high link performance, this design can be used. The resulting alternative configuration provides one high performance MDRU interface at S- and Ku-band and one multiple access interface at S-band.

In addition to the forward and return link functions previously described, the system can track all 20 LDRU's simultaneously and an S-band order wire MDR interface is provided to allow reconfiguring of the operational mode by a manned spacecraft if necessary.

It was assumed that every user satellite could fall into the low-data-rate category ($\leq 10^4$ bps) since this link will service unsophisticated satellites, and provide emergency backup for primary links in both manned and unmanned systems. To fulfill this role it is assumed that no directivity exists on the user's antenna. The law of physics, the current use of the radio spectrum, and the evolutionary nature of the TDRS program provide strong motivation to operate this return link in the VHF band.

Another characteristics of the LDR user is that a particular user competes with other users through his and their multipaths, as well as with almost a whole hemisphere of ground-generated RFI at the TDRS receiver. This problem could be alleviated if a clear channel could be guaranteed each user; however, this solution does not appear to be available. The approach incorporated into the baseline system to combat these problems is a combination of spectrum spreading, data encoding, and adaptive beam forming.

The link that transfers the LDR users' data to the ground operates at Ku-band. The necessary transmitting power is small even in the face of the 17.5-dB margin required in the Washington-Rosman area. At the ground, the data from all users, their multipaths, and the RFI are received in a low-noise, uncooled parametric amplifier-receiver, amplified and divided so that each element of the signal may be recovered for despreading and decoding.

In the forward link from the ground to the LDR user, a somewhat different problem exists. Data rates are low ($\leq 10^3$ bits per second), but the RFI competition at the user's receiver will vary tremendously with his location over the earth. The analysis shows the UHF band to be best when considering RFI, allocation problems, and the use of VHF for the LDR return link.

Spread-spectrum coding is used in the LDR forward link, both to combat interference and to be consistent with CCIR flux density constraints.

The MDR user is characterized by higher data rates ($\leq 10^6$ bits per second on the return link), greater directivity and higher RF output. The MDR users may desire to operate at either S-band or higher frequencies. To minimize constraints on user spacecraft, the TDRS design provides both forms of service. This approach also anticipates the eventual need to support high data rate users who will undoubtedly use higher frequencies such as Ku-band.

The use of an orbital relay to provide the tracking and data acquisition service, while greatly enhancing the service and minimizing design and operational constraints on user spacecraft, does impose slightly different design requirements on the user spacecraft. Basically, the user terminal configuration will remain unchanged. However, some additional equipment is required so that the terminals can use PN modulation and be tunable to one of four carrier frequencies. PN coding is required because of the need for:

1. Distribution of the signal energy emanating from the TDRS to the user over a bandwidth such that the signal flux density at the earth will conform to the CCIR requirements.

2. Discrimination against the multipath signal which will exist in the LDR case.

3. Code division multiplexing from up to 20 users on the return link per TDRS.

The tunable receiver is necessary because of simultaneous visibility to two TDR satellites and compatibility with TDRS which transmits at UHF and STDN which transmits at VHF. If the STDN can be reconfigured to transmit at UHF, the user spacecraft need only be tunable to one of two frequencies.

A preliminary estimate of the size and prime power requirements for the user are:

		Size in^3	Size cm^3	Power (watts)
LDR	Transmitter	225	3690	16
	Receiver	195	3200	12
MDR	Transmitter	240	3930	33
	Receiver	205	3360	12

The TDRS ground station will require one 18.3 m (60 ft) antenna for each operational TDRS satellite. The receivers will use uncooled parametric amplifiers and the transmitters will use Klystron power amplifiers. Additional signal processing equipment will be required in the form of a mini-computer to analyze the LDR return link RF signals and determine the proper commands to adaptively steer the spacecraft VHF beam for maximum signal to interference (RFI) ratio.

2.2 SATELLITE LAUNCH AND DEPLOYMENT

The selection of the TDRS location and orbit inclination was based on three basic performance factors: payload weight, visibility of the ground station and user satellite coverage. Increased satellite spacing increases user spacecraft coverage. To increase satellite spacing, a low final TDRS orbit inclination is desirable. However, payload weight increases with increased orbit inclination, allowing for possible increases in relay capacity. Figure 2-2 shows the relationship of these basic factors. The payload capability is based on a due east launch by a Delta 2914 from KSC, with a modified Thiokol TE-M-616 apogee motor. The ground station visibility curve is based upon a minimum elevation angle at the ground station of 10 degrees (.17 rad), and a ground station latitude of 35.2 degrees.

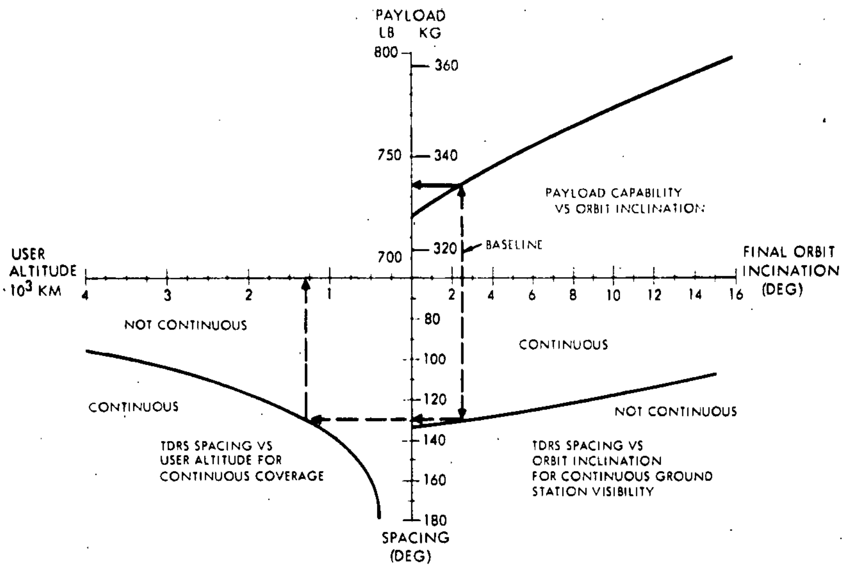

Figure 2-2. Inclination Tradeoffs

2.2.1 Deployment Analysis

As mentioned previously, the telecommunications and spacecraft designs developed in this study provide capabilities considerably greater than required by the SOW specification for Part I. The increase in relay capabilities, provided by increasing the orbit inclination, was weighed against the increase in size of the user spacecraft cone of exclusion, and in light of the impressive relay capacity provided by the baseline design, the decision was reached to maximize user spacecraft coverage by placing the TDRS in a low-inclination orbit.

As can be seen from Figure 2-2, for a final orbit inclination of 2.5 deg (.044 rad), which was selected for the baseline, the TDRS can weigh 333 kg (734 lb) plus the weight of the empty apogee motor. Continuous ground station visibility can be maintained with a TDRS spacing of 130 degrees (2.27 rad). This provides continuous visibility of user satellites above 1275 km (688 n mi). The minimum user altitude for continuous coverage increases rapidly as spacing is reduced below 130 degrees (2.27 rad). The region where user satellites are invisible to either TDRS satellite for the baseline locations is shown in Figure 2-3 for satellite spacings of 125 degrees (2.18 rad) and 130 degrees (2.27 rad).

The satellite final orbit inclination is perturbed by the lunar and solar gravitational fields at a rate of approximately 0.75 degree/year. Judicious selection of launch time will cause the orbit inclination to start decreasing, pass through zero, and then increase with an initial 2.5 deg (.044 rad). Therefore, no north-south station keeping is required.

2.2.2 Launch Analysis

The TDRS weight that can be delivered to synchronous orbit by a given booster is a function of the inclinations of the transfer orbit and of the final TDRS orbit. The apogee motor propellant and payload weights corresponding to a 2.5 degrees (.044 rad) final inclination orbit are plotted in Figure 2-4. The maximum payload is 333 kg (734 lb) plus the 22.7 kg (50 lb) empty motor case and 3.6 kg (8 lb) burned-out insulation, and the optimum transfer orbit inclination is 27 degrees (.47 rad). The corresponding propellant loading for the apogee motor is 314 kg (692 lb). The values shown are the capability to synchronous orbit. The TDRS baseline operational mode injects the spacecraft into a subsynchronous drift-bias orbit and permits an increase in payload, as described later.

The TDRS is launched eastward from KSC by a Delta 2914. It leaves the transfer orbit at apogee, using a solid apogee motor. There are several tradeoffs involved in the selection of which apogee to use as the departure points. Principal factors are time and the location of each apogee relative to the final TDRS station. The TDRS is injected into the transfer orbit at the first descending node of the parking orbit. If Tananarive, Rosman, and Orroral are used to track and command the TDRS during the transfer orbits through the third apogee, visibility is maintained for the entire time except for 165 minutes at the second perigee and 38 minutes at the third perigee. If Guam is used instead of Orroral, the 165-minute period is reduced to 118 minutes. These times are considered satisfactory.

The baseline transfer orbit has an apogee at synchronous altitude and the apogee motor burns out at a velocity slightly below geosynchronous to provide an eastward drift of 5 degrees (.087 rad)/day. The transfer orbit inclination is 27 degrees (.47 rad). Plane changes of 1.3 degrees (.023 rad) and 24.5 degrees (.426 rad) are made at perigee and apogee, respectively to provide a nominal 2.5 degrees (.044 rad) final orbit inclination. Synchronous orbit injection occurs at the second or third apogee of the transfer orbit, depending on the desired final location. This provides ample time for tracking, orbit determination, and vehicle precession; and places each TDRS in a position to the west of its assigned station. Figure 2-5 shows the ground trace of the TDRS during parking and transfer orbits, and the locations of the first five apogees, the tracking stations, and final TDRS locations.

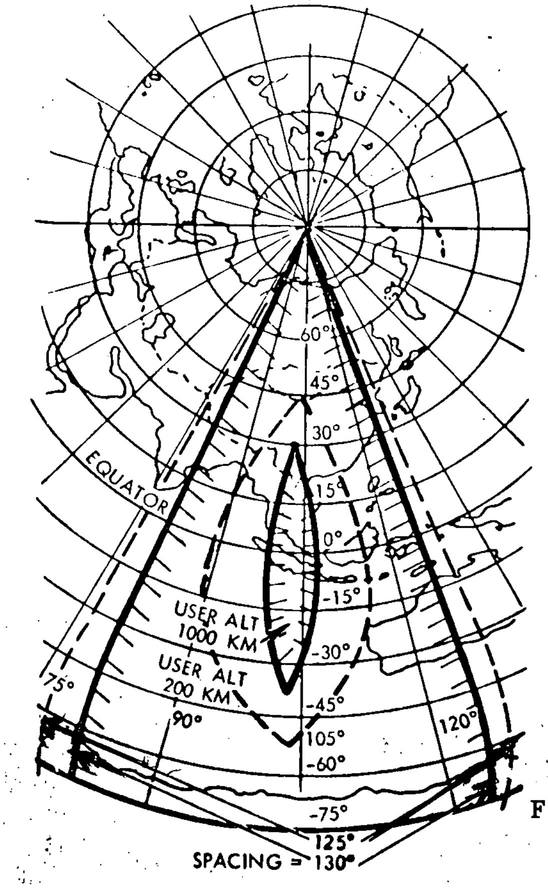

Figure 2-3. User Cone of Exclusion

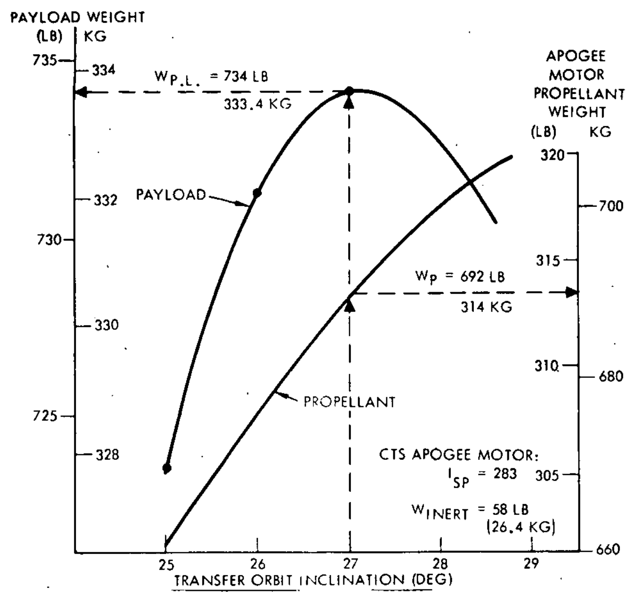

Figure 2-4. Delta 2914 Synchronous Orbit Payload

The impulse provided at the transfer orbit apogee injects the TDRS into a near-geosynchronous orbit at the final orbit inclination and has both an in-plane and an out-of-plane component. The in-plane component places the vehicle in a drift orbit with an easterly motion. If the TDRS is injected at exactly synchronous velocity (drift rate equal to zero), the on-board propulsion system (monopropellant hydrazine) must initiate and stop the drift to station. If the TDRS is injected with a velocity biased slightly below synchronous, it will have a "built-in" drift, and the on-board propulsion system must only stop the drift at the appropriate station, increasing the useful TDRS payload weight.

Figure 2-6 shows the effect of drift rate on (1) time-to-station for the three TDRS satellites, and (2) net "payload loss," apogee motor propellant reduction, and increase in on-board hydrazine. The net "payload loss" is defined as the loss in dry weight of the TDRS and is the difference between the required hydrazine and the reduced apogee propellant after the "drift bias" mode has been selected. For the drift range considered, the effect on payload is negligible. In using the drift bias mode, an actual gain in dry payload of approximately 1.8 kg (4 lb) results.

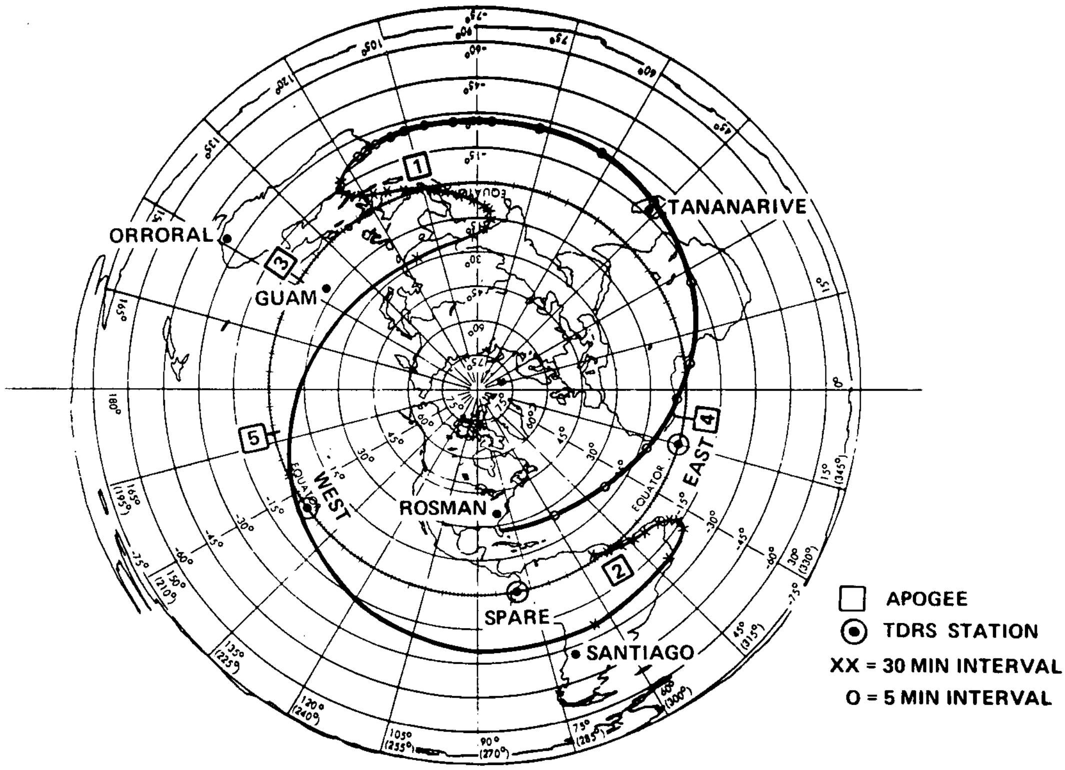

Figure 2-5. Ground Trace and Final Stations

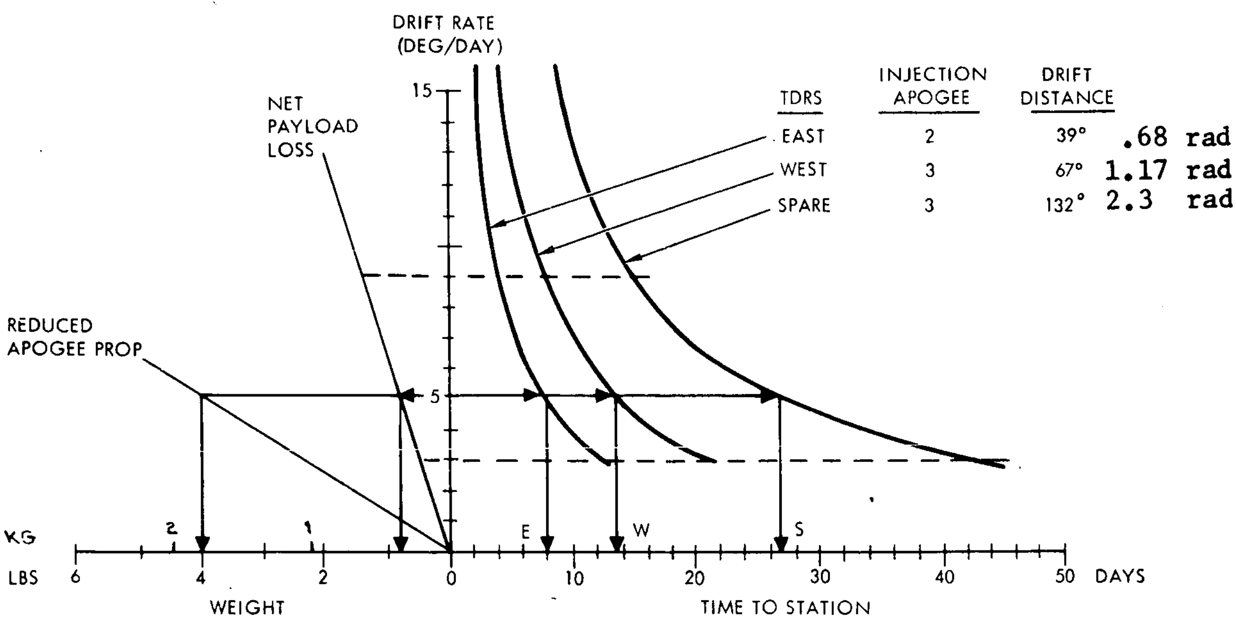

Figure 2-6. Drift Rate Effect on Payload and Drift Time

The east TDRS enters its drift orbit on the second apogee, 39 degrees (.68 rad) west of its destination, with an eastward drift of 5 degrees (.087 rad)/day. It arrives at its station in about 8 days. The west TDRS enters the drift orbit on the third apogee, 67 degrees (1.17 rad) west of its destination, with the same drift rate. It arrives at its destination in 13 days. The spare enters its drift orbit on the third apogee, 132 degrees (2.3 rad) west of its storage location, which is midway between the other two satellites. It drifts for 27 days.

An additional factor that must be considered in establishing the on-orbit payload weight is the reduction in apogee motor propellant due to on-board propellant (hydrazine) consumed during transfer orbit that need not be injected into synchronous orbit. This reduction in apogee propellant can be partially converted into payload.

The Delta 2914 injects 678 kg (1490 lb) into a 27-degree inclination transfer orbit. Six pounds (2.7 kg) of propellant are used for precession and nutation damping during transfer, leaving 673 kg (1484 lb) at synchronous orbit injection. This requires 314 kg (692 lb) of propellant and results in 359 kg (792 lb) of payload. Included in the payload is 22.7 kg (50 lb) of empty motor case and 3.6 kg (8 lb) of burned-out insulation. The resulting 333 kg (734 lb) of useful payload is injected into synchronous orbit. However, the "drift bias" mode allows a reduction in apogee propellant of 1.8 kg (4 lb) and an increase in payload of 1.8 kg (4 lb). This results in the following propellant and payload values into the 5-degree/day drift-bias orbit.

Payload = 333 + 1.8 = 334.8 kg (738 lb)
Apogee propellant = 314 - 4 = 312.2 kg (688 lb)

2.2.3 Launch and Deployment Profile

Based on the TDRS deployment philosophy previously discussed, a baseline flight profile was established. Each operational satellite is injected at the apogee most convenient for eastward drift to its station. Locations of the first three apogees in such case are at 104 degrees, 306 degrees, and 148 degrees longitude. The spacecraft/launch vehicle combination injects into a 100 n mi circular inclined orbit and at the first descending node is injected into a Hohmann ellipse to synchronous altitude. At some apogee passage (second or third) of the transfer orbit, the spacecraft is injected into a near-circular equatorial orbit, i.e., the thrust simultaneously removes the eccentricity and inclination of the transfer orbit, leaving slight residuals from non-perfect systems performance. These residuals are removed by a vernier propulsion correction system.

Figure 2-7 illustrates the total launch profile for launch and deployment into operational status. The total mission is divided into three phases: (1) boost, (2) transfer orbit, and (3) preoperational synchronous orbit phase.

Each TDRS is launched from ETR by a Delta 2914 with a TE-364-4 third-stage at a launch azimuth of 90 degrees (1.54 rad). The vehicle lifts into a parking orbit at a nominal altitude of 100 n mi (185 km) with an inclination of approximately 28.3 degrees (.5 rad). The fairing is jettisoned about 36 seconds after Stage II ignition and 4 minutes before the first Stage II cutoff command and start of the parking orbit coast phase. Coast lasts 16.22 minutes and concludes when the vehicle reaches the first descending node (first perigee) at 3 degrees east longitude. At the node, the second stage restarts for 28 seconds. After its burnout, the third stage and the TDRS are spun up to 90 rpm, Stage III ignites and burns for 24 seconds to complete transfer orbit insertion. Payload separation occurs two minutes later and the TDRS remains spinning until after insertion into synchronous orbit.

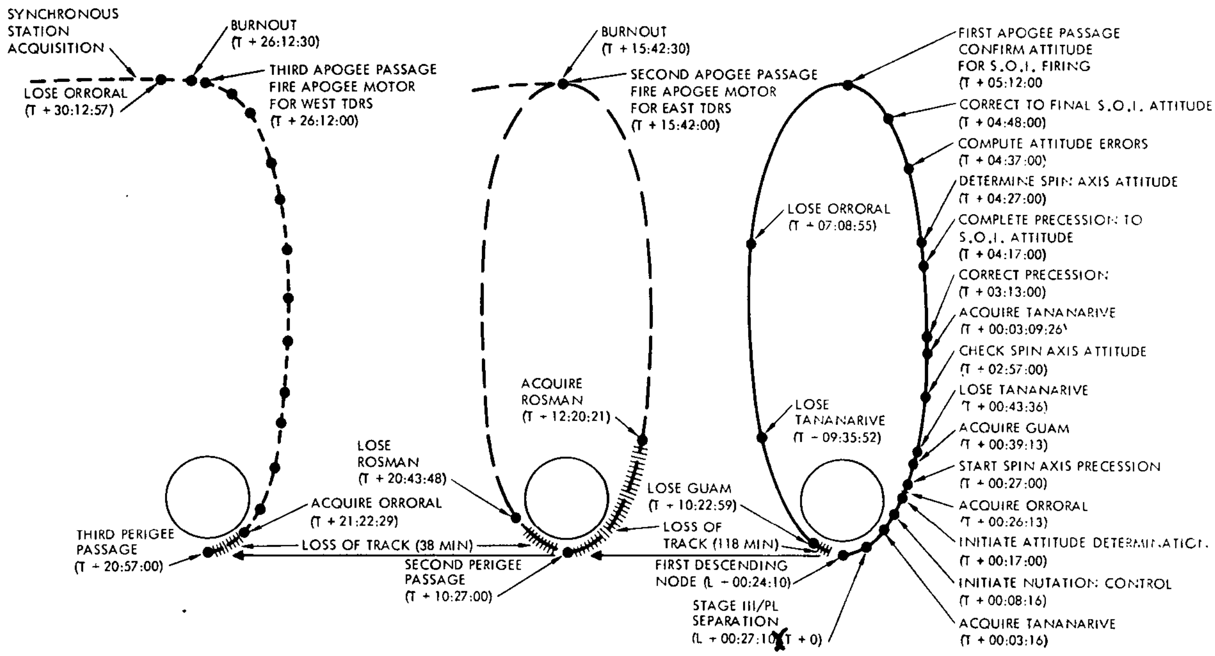

Figure 2-7. Overall Launch and Deployment Profile

After payload separation [~ 204 km (110 n mi)] the spacecraft coasts to synchronous altitude in an elliptical transfer orbit. The long transit allows time for smoothing and processing of tracking data and for reorienting the spacecraft for the apogee motor burn. The transfer time from injection (perigee) to first apogee (one-half orbit) is 5.25 hours. During the entire transfer orbit, the spacecraft will be spinning and will maneuver into appropriate attitudes for attitude determination and measurement, and nutation will be damped out.

At the given apogee, the apogee motor fires to change plane and circularize the orbit at synchronous altitude for approach to operational station. Complete transfer orbit time is approximately 15.75 hours to the second apogee and approximately 26.25 hours to the third apogee, sufficient for all required operations and economic fuel consumption.

After apogee motor burnout, the spacecraft is despun and stabilized (momentum wheels spun) in an essentially equatorial orbit. The solar panels are deployed 1.5 hours after spacecraft despin and the antennas are deployed 20 minutes later. The spacecraft then acquires the sun and earth and achieves near-continuous sunlight for the mission at synchronous altitude. The spacecraft drifts to its assigned station. Appropriate post-apogee delta-V maneuvers are performed to correct the spacecraft injection errors and to acquire the proper drift orbit (about 24 hours after apogee motor burnout).

2.3 TELECOMMUNICATIONS DESIGN

The major constraints that impacted the TDRS telecommunication design are the capabilities of the Delta 2914, the radio frequency interference (RFI) and multipath environment, and inclement weather at the ground station. The booster payload capability imposes a weight, power, and volume limit on the spacecraft which correspondingly limit antenna size, system redundancy, etc. From the outset the RFI and multipath phenomena presented the single most important technical problem area. The use of pseudonoise (PN) modulation concepts and the Adaptive Ground Implemented Phased Array (AGIPA) offers a viable solution to these problems associated with supporting LDRU's.

The basic design goal was to provide a viable cost-effective telecommunication service to a variety of user spacecraft. This requires a system that can provide a multiple semirandom access service to LDRU and MDRU spacecraft.

Telecommunication support required by the users as per the SOW is presented in Table 2-1. The table is divided into three basic categories; namely, LDR user, MDR user, and a manned user which is the Space Shuttle. In addition to those presented in the table, the ground station/TDRS link is required to operate at Ku-band, with a rain margin of +17.5 dB. Where there are multiple frequencies shown in the table, separated by the word/logic "OR," the option was given to the contractor to select one or more of these frequencies to optimally support that channel's performance requirements.

Each TDRS satellite must simultaneously support 20 LDR users and at least one MDR user in the return link (user to TDRS to GS) and one LDR user and at least one MDR user in the command link (GS to TDRS to user). However, throughout this study it has been a goal, both in the design of the telecommunications subsystem and of the spacecraft to (1) provide simultaneous support capabilities greater than those required, (2) to maximize system flexibility and adaptability to user needs, and (3) to minimize risk.

2.3.1 Telecommunications System Analysis

The low data rate service uses a UHF command link with two TDRS electronically steered beams and a relatively high-gain antenna. The LDR transponder provides an alternate capability for a F-FOV approach where all users are simultaneously illuminated at an EIRP of +24 dBw. This mode can be used to coherently lock all users simultaneously or to transmit command data or voice at reduced capability. System EIRP at UHF is +30 dBw/beam. The return link employs the adaptive ground implemented phased array (AGIPA) concept which provides 20 independent beams (one for each user). The user spacecraft are

Table 2-1. Telecommunications Service Requirements (as per SOW)

Description	LDR User	MDR User	Manned User (Shuttle)
Number of users	Forward: 1 (minimum) Return: 20	Minimum of one	Minimum of one (Replaces requirement for one MDR)
Frequency:	Forward: VHF or UHF or S-band Return: VHF	S- or X- or Ku-band	S-band VHF-band
Communications requirement	Forward: 100 to 1000 bps Return: 1 to 10 kbps	Forward: 100 to 1000 bps Return: 10 to 1000 kbps	Forward: 2 kbps 1 or 2 voice at 19.2 kbps Return: 76.8 kbps 1 or 2 voice at 19.2 kbps
Constraints	Linear transponder in return link High RFI Flux density (IRAC) VHF $\leq$ -144 dBw/m^2/4 kHz UHF $\leq$ -150 dBw/m^2/4 kHz S-band $\leq$-154 dBw/m^2/4 kHz EIRP = +30 dBw/channel for UHF & VHF +41 dBw/channel for S-band BER = 10^{-5}	Linear TDRS transponder in return link Variable frequency Flux density (IRAC) S-band $\leq$-154 dBw/m^2/4 kHz X-band $\leq$-150 dBw/m^2/4 kHz Ku-band $\leq$-152 dBw/m^2/4 kHz BER = 10^{-5}	User antenna gain = +3 dB BER Voice: 10^{-3} Data: 10^{-4} User transmit power = 16 dBw

discriminated by use of unique PN codes in the return link. The AGIPA concept provides an adaptive spatial filtering of RFI, and an optimization of the signal-to-interference ratio. The system is designed such that in the event of failures it will degrade gracefully to a fixed field-of-view configuration. The key design features of the low data rate services are shown in Table 2-2.

Table 2-2. Key Design Features of the Low Data Rate Service

Forward link	Frequency band Polarization EIRP [at 31 deg (0.54 rad) FOV]	400.5 to 401.5 MHz Circular Steered beam 30 dBw data/36 dBw voice at 25-percent duty cycle F-FOV 24 dBw
Return link	Frequency band Polarization G/T_S - AGIPA - FFOV	136 to 138 MHz Linear, 2 planes -14.4 dB/°K* -18.8 dB/°K*
Antenna configuration		Backfire antenna, 4-element array
Transceiver configuration		Frequency translating
*Assumes nominal antenna temperature of 800 K; however, T in this frequency is a variable.		

The MDR user telecommunications service has a dual frequency feature; namely, an S-band link to support current MDR users and a Ku-band link to support future MDR users with higher performance. Two MDR users can obtain simultaneous support via two 2 m (6.5 ft) dishes on the TDRS. In addition, the MDR service can support manned users such as the Space Shuttle. The antenna tracking consists of open-loop tracking at S-band and autotrack at Ku-band. If open-loop tracking is considered to result in significant increase in ground station operation, autotrack can be implemented on the S-band antenna. Furthermore, the MDR antenna system and/or transceiver serves as backup for the GS/TDRS link at either S- or Ku-band. The MDR service is essentially a "bent pipe" on the return link, providing support for a variety of MDR users regardless of the signal formats employed. The MDR service features are shown in Table 2-3.

2.3.1.1 The Interference Problem

The TDRS and low data rate user spacecraft will be confronted with four basic types of interference:

1. Unintentional, upward directed, radio frequency interference (RFI) originating from communications equipment located on earth.

Table 2-3. Key Design Features of the Medium Data Rate Service

Forward link	Frequency EIRP Polarization	S-/Ku-band S-band data = 41 dBw; voice = 47 dBw at 25-percent duty cycle Ku-band at 45.6 dBw Circular
Return link	Frequency G/T_S Polarization	S-/Ku-band 3.9/20.4 dB Circular
Antenna configuration		Two S- and Ku-band Parabolic reflectors
Transceiver configuration		Frequency translating

2. Background interference (i.e., "trash noise") originating primarily in urban areas and is the composite of such things as ignition noise, switching transients, corona, and other forms of man-made noise.

3. Multipath interference between the direct path signal in the USER/TDRS link and a replica of that signal reflected off the earth.

4. Co-channel interference between one user and all the other users in the same band. This is peculiar to the return link where all users are simultaneously accessing TDRS through the same channel.

Unintentional, upward directed RFI from emitters located on the earth and in view of both the TDRS and user spacecraft can be a deleterious source of interference. Frequency bands considered in evaluating the impact of RFI on the TDRS system are:

Forward Link	Return Link
117.5 MHz (±25 kHz) 127.7 to 127.85 MHz 148 to 150 MHz 400.5 to 401.5 MHz	136 to 138 MHz

An estimate of the worst case RFI distribution at the TDRS and user was derived. These distributions were based on analysis of FCC licenses and associated power levels. Multipath of 7 dB was used in all calculations. Figure 2-8 presents a graphical indication of the RFI power density at the TDRS located at all degrees west longitude over the frequency band 117 to 155 MHz. The band 144 to 148 MHz is the amateur band and has not been modeled.

The impact of RFI on the user spacecraft was also assessed. Two user locations were used, 50°N 30°E and 38°N 85°W and the RFI density levels at the user spacecraft are shown in Figures 2-9 and 2-10, respectively.

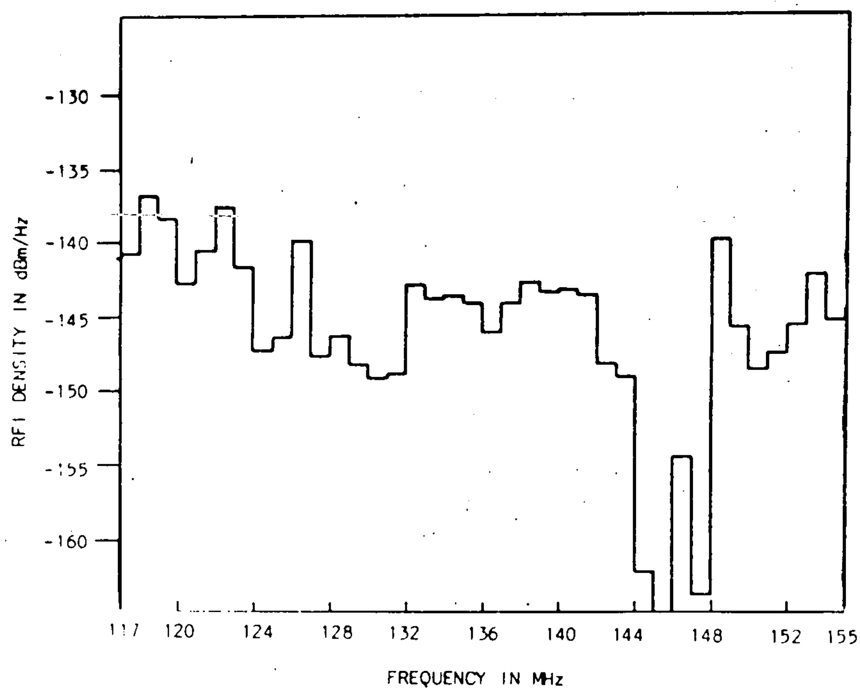

Figure 2-8. RFI Power Density for TDRS Located at 11° W

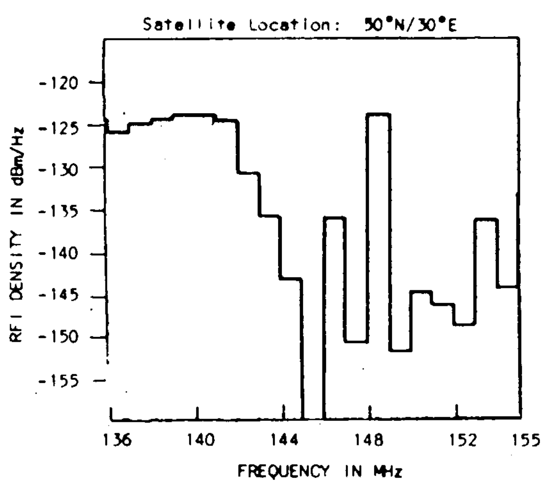

Figure 2-9. RFI Power Density at User Spacecraft (1000-km Altitude and Omnidirectional Antenna - Satellite Location 50°N/30°E)

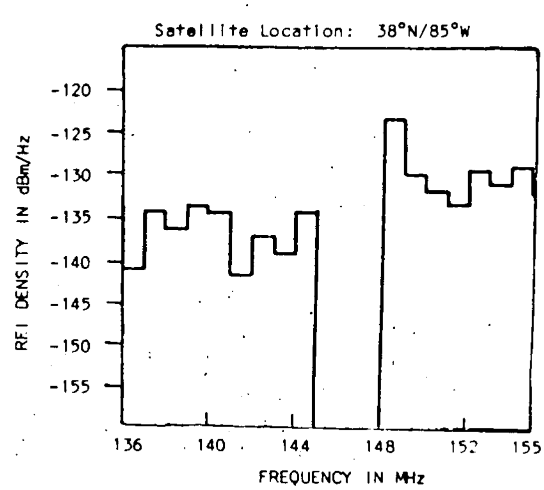

Figure 2-10. RFI Power Density at User Spacecraft (1000-km Altitude and Omnidirectional Antenna - Satellite Location 38°N/85°W)

In each case the user spacecraft is equipped with an omni-directional antenna and is at an altitude of 1000 kilometers. As an indication of the coverage of this user spacecraft, a satellite at 1000 km centered at 38°N and 88°W will, with an omni-directional antenna, be in view of all of CONUS and parts of Mexico and Canada.

As can be seen, the estimated RFI levels are extremely high throughout this portion of the spectrum. However, it must be remembered that these are worst case projections based on maximum transmitted power levels allowed by licences. The analysis does not consider station antenna directivity or duty cycles. A recent experiment conducted by the Space Division of North American Rockwell for the NASA/GSFC indicates that the duty cycles are very low and the actual RFI power levels received will be quite low. An example of the reduced data from this experiment is shown in Figure 2-11.

The remaining band of interest is in the 400.5 MHz to 401.5 MHz range. An estimate was compiled, through the International Frequency List, of the number of potential RFI sources in this band and is presented in Table 2-4 for the various ITU regions. These data show that RFI in this portion of the spectrum can be expected to be considerably less severe than in the VHF area.

Table 2-4. RFI Sources in the Band 400.5 to 401.5 MHz

ITU	Emitter Power on the Ground (watts)				
Region Number	0 - 0.9	1 - 9	10 - 99	100 - 999	≥1000
I	59	4	1	1	1
II	61	1	-	-	-
III	62	12		10	

Another important source of noise was considered and evaluated for its impact on TDRS/user channel. This interference is the sum total of the "trash" --a more or less continuous background noise which originates in an urban environment and is due primarily to such things as ignition noise, switching transients, corona, etc. This continuous low-level interference can have a deleterious effect on the antenna temperature at the user spacecraft.

Figure 2-12 depicts the "trash noise" density as a function of urban radius for various frequency bands and user spacecraft altitudes. Due to the large field of view of even low orbiting satellites, the urban radius can be large. This is the result of the appearance of many discrete point sources acting as one large area distributed source. Radius values can range from 200 to 400 km for the New York area and 100 km for the Chicago area, to negligible for water or desert areas.

A signal transmitted from either TDRS or user satellite arrives at the other satellite by direct and indirect (reflection off a non-smooth earth) paths. This multipath aspect of interference problem was included in the prediction of the RFI power density distributions.

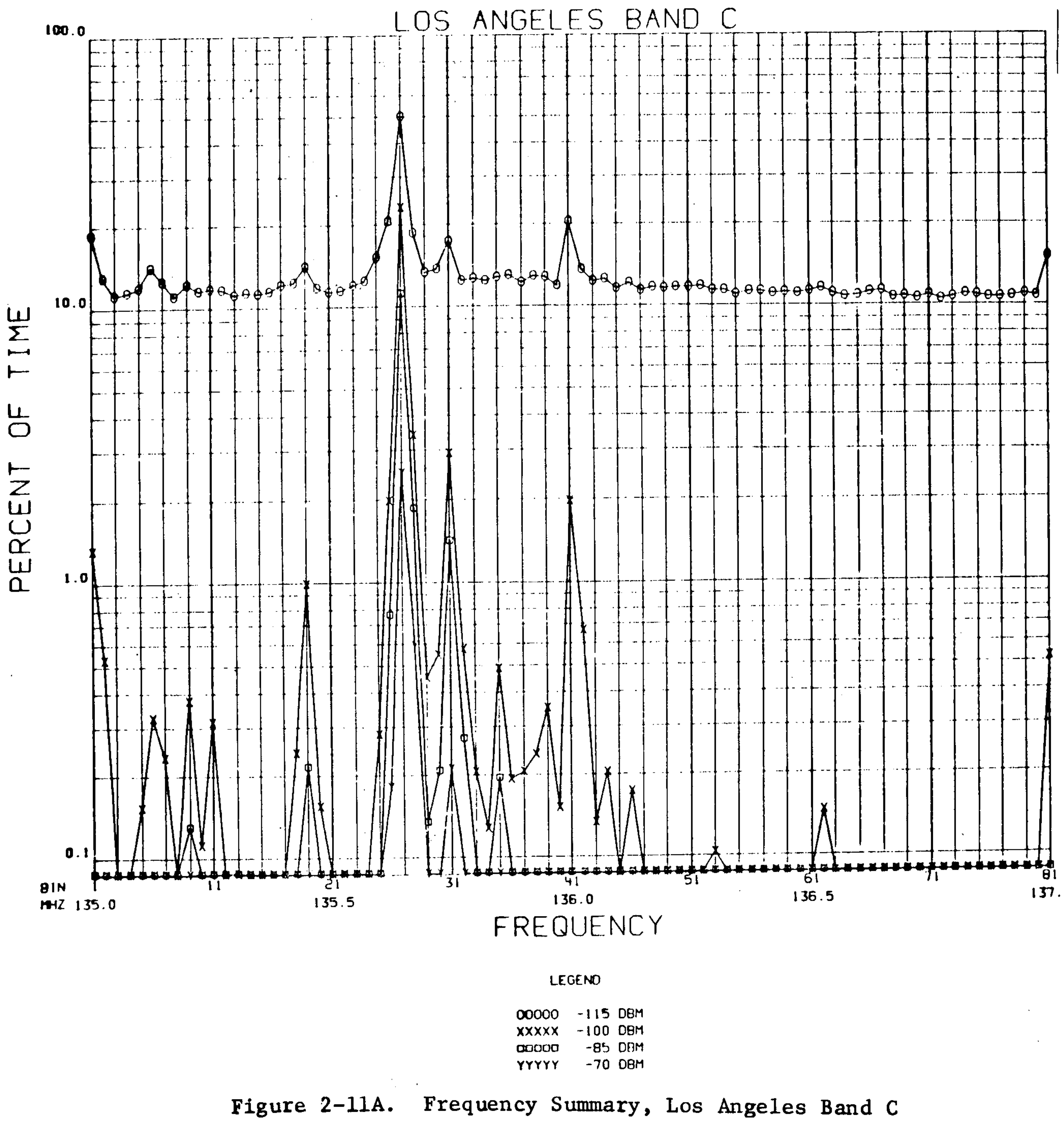

Figure 2-11A. Frequency Summary, Los Angeles Band C

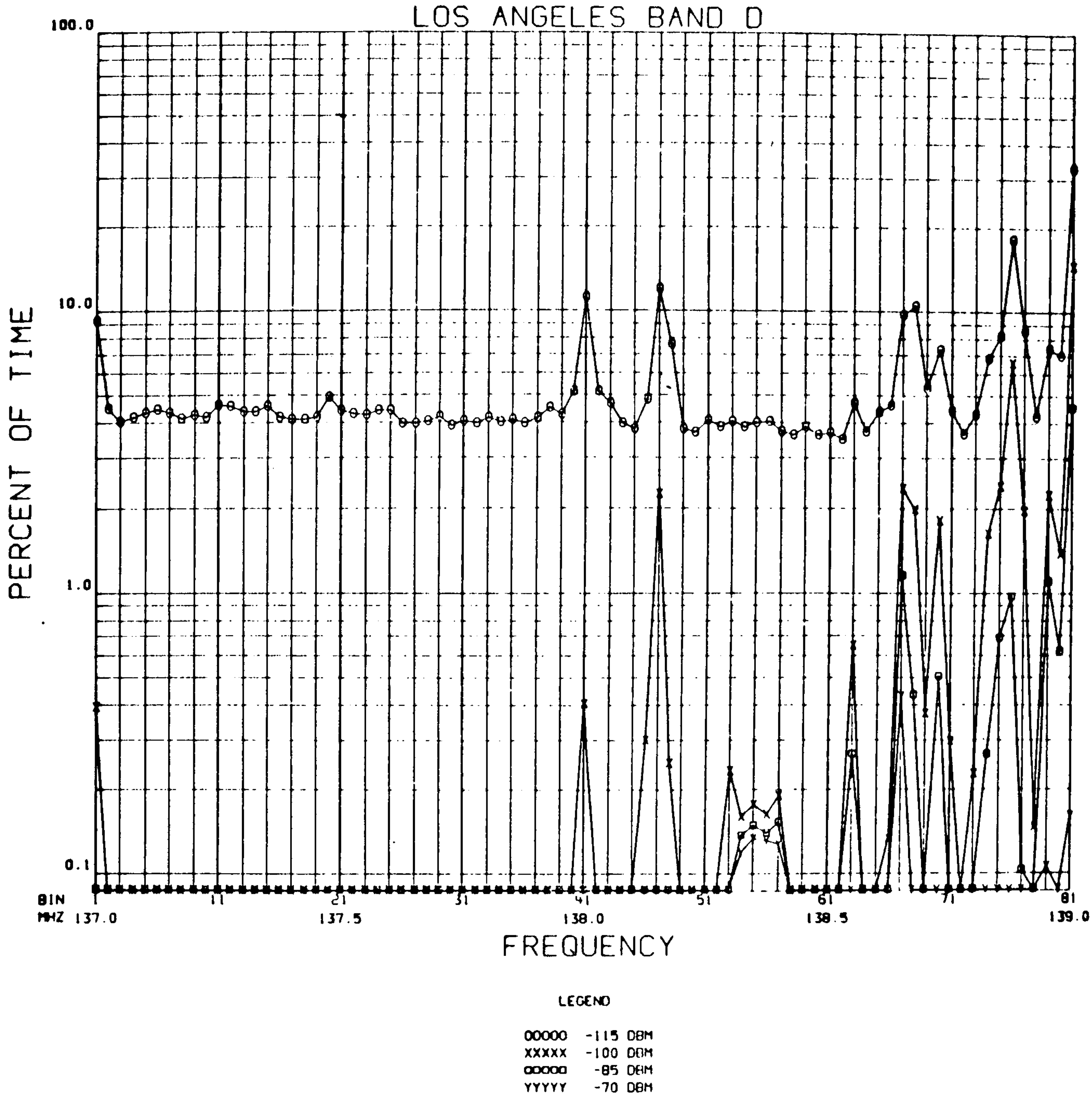

Figure 2-11B. Frequency Summary, Los Angeles Band D

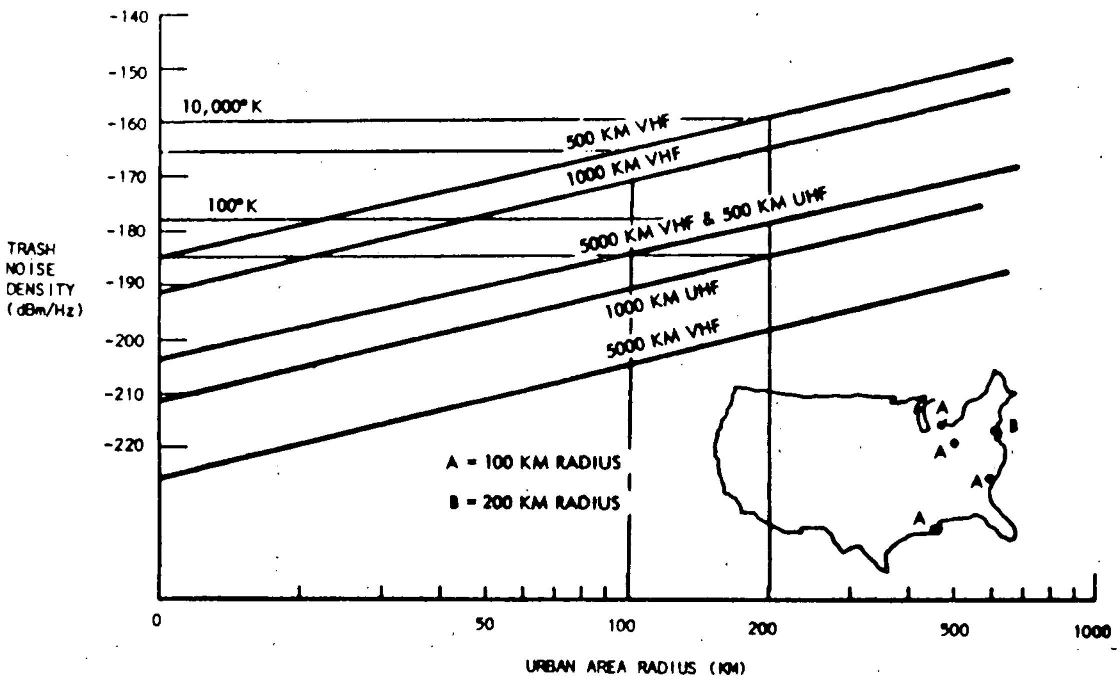

Figure 2-12. Trash Noise at User Spacecraft

The reflected power can consist of both specular and diffuse components. The specular component is essentially a delayed replica of the transmitted signal, whereas the diffuse component is noise-like. For reasonable roughness factors and correlation lengths, the primary source of reflected power is diffuse for grazing angles in excess of 20 degrees at VHF or UHF. Figure 2-13 indicates the multipath signal level as a function of orbital altitude of the user satellite.

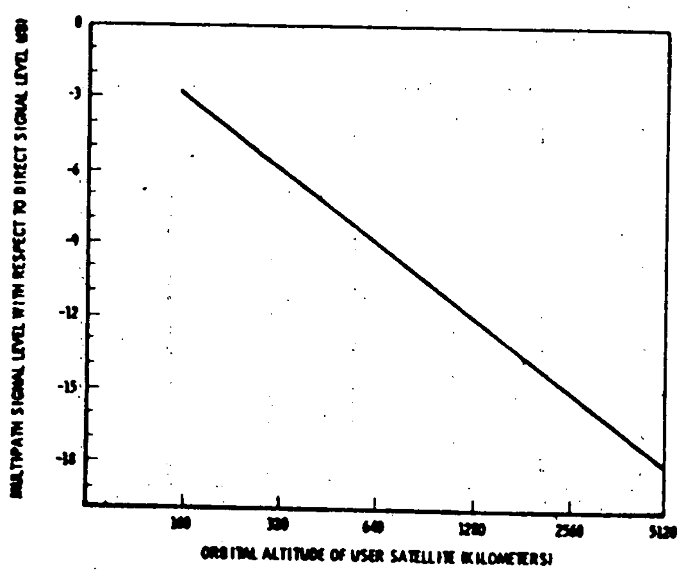

Figure 2-13. Multipath/Direct Signal Ratio as a Function of Orbital Altitude

In the forward link, the plot of Figure 2-13 can be used directly to determine the multipath impact on the user spacecraft; however, for the return VHF link, 20 user signals are accessing a TDRS simultaneously. Obviously, the total multipath signal in the field of view of a TDRS will depend upon the user altitudes. Table 2-5 lists the projected satellite populations for the low-altitude user spacecraft from 1976 through 1980.

Table 2-5. Low-Altitude Spacecraft Population Projections, 1976-1980

Altitudes (km)	Year				
	1976	1977	1978	1979	1980
150 - 276	1	1			
275 - 300	4		1	1	1
300 - 400	7	7	1	2	2
400 - 600	9	9	4	2	
700 - 800	10				
800 - 900	9	8	3		
1000 - 5000	10	9	2	3	3
5000	8		6	4	4

Figure 2-14 shows the expected total multipath contributed by the total satellite population by year. Summing up the multipath signal contributed by 20 simultaneous users in 1976, the total multipath relative to one user is +7 dB. This is considerably less than the worst case condition where all 20 users are assumed to be in low orbits, e.g., 300 km.

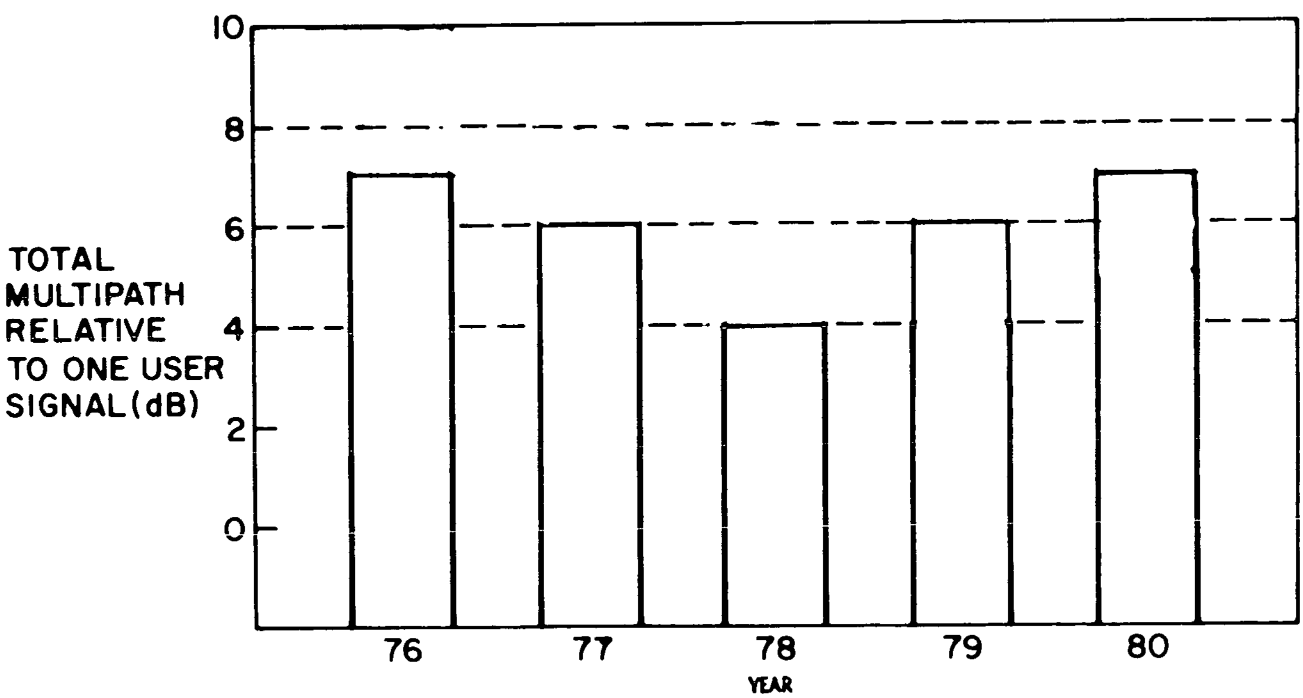

Figure 2-14. Projected Multipath Level

In addition to RFI and multipath signals, a return link will be subjected to interference from signals originating from the 19 other user spacecraft and propagating along the direct path from user to TDRS. Their effect will be much more pronounced than that of the return link multipath signals with an interference/signal ratio on the order of 12 dB.

As implied by the foregoing discussion, the major concern in the design of the LDRU communications link was interference and the development of electromagnetic and encoding techniques to meet performance requirements in the face of these problems.

RF links to the medium data rate users, although significantly influencing the size and configuration of the TDRS, need not contend with the enormous complications of RFI or multipath present in the LDRU links. The major design problem in these links was to configure antennas that will provide multiple access.

2.3.1.2 Frequency Selection

Frequency bands for both ground/space and space/space communications links are shown in Tables 2-6 and 2-7.

Table 2-6. TDRSS Ground/Space Link Frequency Band Selection

Requirement	Ground-to-Space Link	Space-to-Ground Link
Primary frequency Ku-band	13.4 to 14.0 GHz	14.6 to 15.2 GHz
Backup frequency S-band	2200 to 2290 MHz	2025 to 2110 MHz
Launch phase Stationkeeping VHF	148 to 150 MHz	136 to 138 MHz

Table 2-7. TDRSS Space/Space Link Frequency Band Selection

Link	LDR User Link	MDR User Link
Telemetry	VHF: 136 to 138 MHz	S-band: 2200 to 2300 MHz
Command	VHF: 108 to 132 MHz (2 - 100 kHz channels)	
	UHF: 400.5 to 401.5 MHz (2 - 500 kHz channels)	S-band: 2025 to 2120 MHz

During the study three LDR forward link frequency bands were investigated:

VHF $\begin{cases} \text{117.50 MHz (BW = 50 kHz} \\ \text{127.75 MHz (BW = 100 kHz)} \end{cases}$

UHF 401 MHz (BW = 1 MHz)

S-band 2025 - 2120 MHz (BW = 10 MHz)

The 117.5 MHz band is considered impractical because the narrow channel bandwidth (50 kHz) is inadequate to meet range error requirements. Furthermore, it does not provide enough processing gain to offset the effects of co-channel multipath. A similar objection is presented for an assignment at 127.7 MHz. In this case the 100 kHz band appears adequate; however, if it is split into two 50 kHz bands (one per TDRS) the range accuracies specified in the work statement (namely, 15 meters rms) cannot be achieved.

There appears to be less RFI at UHF than at VHF, making it more attractive for the LDR forward link. Moreover, evaluation of the "trash noise" indicates this interference is 20 dB less at UHF than at VHF (e.g., a "trash noice" temperature of 10,000 K at VHF would be $\sim$100 K at UHF. S-band was rejected because it had a larger implementation impact on than the user spacecraft than UHF.

On the LDR return link the 136 to 138 MHz band, while potentially possessing considerable RFI, appears to have the lowest level of activity. Within this band the 136 to 137 MHz segmen is preferable.

The forward and return MDR links appear to offer no dramatic tradeoffs. The links are relatively RFI free and selection is a function of obtaining appropriate frequency assignments such that adjacent bands introduce minimum interference, and sufficient bandwidth is available for growth to support high data rate users.

The space/ground link at Ku-band has adequate bandwidth for both uplink and downlink signals in an FDM/FM mode.

The final item which impacts frequency trades was the recommendations by IRAC regarding the flux density at the earth's surface from a satellite signal. For the bands of interest the IRAC specifications and the minimum spread bandwidth required to reduce the flux density to an acceptable level are presented in Table 2-8.

Table 2-8. Bandwidth Spreading Required to meet IRAC Specifications

Frequency Band	IRAC Required Flux Density in 4 kHz	EIRP	Minimum Spread Bandwidth
VHF	-144 dBw/m^2	30 dBw	63 kHz
UHF	-150 dBw/m^2	30 dBw	250 kHz
S-band	-154 dBw/m^2	41 dBw	8 MHz
X-band	-150 dBw/m^2	48 dBw	16 MHz
Ku-band	-152 dBw/m^2	52 dBw	63 MHz

From the candidate frequencies and their associated channel bandwidths, the overall frequency plan shown in Table 2-9 was recommended for the Part I baseline configuration.

Table 2-9. System Frequency Plan

Links	Frequency	Channel Bandwidth
Forward Link		
LDR	400.5 to 401.5 MHz	1 MHZ 4 - 250 kHz channels
MDR		
S-band	2025 to 2120 MHz	95 MHz channel 4 - 100 MHz channels
TDRS/GS		
Ku-band	13.4 to 13.64 GHz	240 MHz
VHF	148.26 MHz	
S-band	2200 to 2290 MHz	90 MHz
Return Link		
LDR	136 to 138 MHz	2 MHz (20 users multiple accessed/TDRS)
MDR		
S-band	2200 to 2300 MHz	20 - 10 MHz slots in 5 MHz steps or 100 MHz wide open
Ku-band	13.4 to 14.0 GHz	4 - 100 MHz channels
TDRS/GS		
Ku-band	14.6 to 15.2 GHz	200 or 600 MHz channels
VHF	136.11 MHz	--
S-band	2025 to 2110 MHz	85 MHz

2.3.2 Telecommunications Subsystem Design

To meet the design requirements and goals, the telecommunication subsystem analysis and design effort emphasized and stressed an overall design goal to maximize the simultaneous support of user spacecraft with the flexibility to adapt to anticipated changes in users' needs. In addition, minimum size, weight, and power for the TDRS spacecraft as well as minimum impact on the user spacecraft terminals were emphasized. High reliability is achieved through full block redundancies of all critical components/subassemblies, functional alternative redundancies employing backup via entire functional chain, and/or multiple operating channels which also provides for graceful degradation. The recommended design reflects state-of-the-art technology.

Figure 2-15 is a simplified block diagram of the telecommunications subsystem indicating the major elements and identifying the transceivers associated with supporting the 20 LDRU's and 2 MDRU's and maintaining the TDRS/GS interface.

The alternative telecommunication subsystem operating modes are summarized in Table 2-10 along with the dc power requirements of each mode. The powers shown in the table do not include the effect of power conditioning.

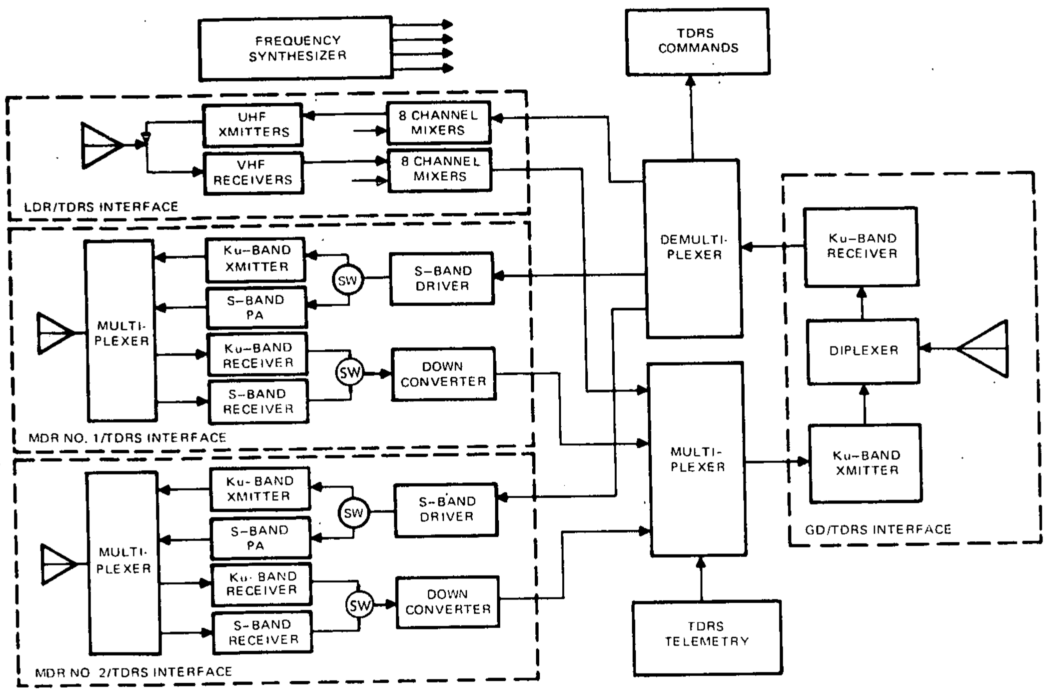

Figure 2-15. Telecommunications Subsystem Block Diagram

Table 2-10. Telecommunications Power Requirements

		Operational Mode	dc Power Requirements
1	Return Link	20 LDRU (VHF) 2 MDRU (S-band)	256.0
	Forward Link	2 LDRU (UHF) 2 MDRU (S-band)	
2	Return Link	20 LDRU (VHF) 2 MDRU (1 S-band, 1 Ku-band)	224.1
	Forward Link	2 LDRU (UHF) 2 MDRU (1 S-band, 1 Ku-band)	
3	Return Link	20 LDRU (VHF) 2 MDRU (Ku-band)	191.1
	Forward Link	2 LDRU (UHF) 2 MDRU (Ku-band)	

2.3.2.1 TDRS/LDR Transponder on TDRS

The TDRS satellite must provide a multiple access relay space-to-space interface with the low-altitude users; and a space-to-ground interface with the ground station.

As previously indicated, the potential interference levels could totally disrupt the entire LDR link if adequate precaution is not taken during the early design phases.

In the LDR return link, the TDRS transponder sees nearly an entire hemisphere of RFI emitters, its level being potentially 20 to 40 dB greater than the desired user signal. To combat the high RFI environment the TDRS telecommunications system uses an adaptive ground implemented phased array (AGIPA) which adaptively employs spatial filtering and polarization discrimination of RFI emitters to maximize the signal-to-interference ratio (SIR). As the name AGIPA implies, the beam steering of the phased array beam shaping and other signal processing functions are conducted at the ground station (GS). All the advantages of the phased array are achieved without placing the complexities of the multi-beam processing functions aboard the TDRS. AGIPA provides an independent beam and signal processor for each user, employing a mini-computer at the GS to provide the iterative SIR optimization.

A four-element phased array is used whose individual element characteristics are the same as a fixed field-of-view (F-FOV) approach which sees the entire 31 degrees (.54 rad) FOV.

The performance of AGIPA was compared with a F-FOV approach for the LDR return link. Since AGIPA is an adaptive system, a computer program and typical realistic RFI models were created to compare and assess the performances of the two approaches for typical user spacecraft orbits. For Atlantic and Pacific scenarios representing TDRS satellites at 11 degrees west and 141 degrees west longitude, AGIPA nominally provides 5 to 15 dB improvement over a F-FOV approach. It also achieves the specification requirement of 10 kbps in relatively high RFI density. In contrast, a F-FOV approach is limited to less than 1000 bps in the same RFI environment and in the presence of a large jammer (e.g., 20 kw emitter), the F-FOV is totally inoperative. AGIPA discriminates the large emitter along with the other RFI emitters, thus optimizing the SIR.

For the LDR return link, eight wideband (2.0 MHz) linear channels are used to relay the vertically and horizontally polarized components from the four AGIPA antenna elements. This is necessary since the LDR users are anticipated to have linear polarization and have random orientation relative to the TDRS. The loss of any channel or channels results in graceful degradation and merely represents reduced performance, thereby enhancing reliability.

The LDR forward link is also subject to RFI, but contrary to the return link, the user spacecraft receiver is confronted with the interference signals. Since user spacecraft can be more than 200 times closer to the RFI emitters than the TDRS, the TDRS transponder must overcome this large differential space loss to provide an adequate signal-to-noise ratio (SNR) at the user. This link was evaluated for operation at VHF, UHF, and S-band. The link analysis and

TDRS hardware implementation impact support the recommendation to operate this link at UHF. The relatively large implementation impact at S-band eliminates it for this Part I phase. In comparing VHF and UHF in this RFI limited link, the link performances of both bands are equivalent. The 10 dB additional space loss at UHF becomes immaterial since the desired as well as the RFI signals suffer the same space loss. Although a conclusive definition of the RFI power level cannot be provided at VHF and UHF, it appears that the RFI power density could exceed the desired signal by approximately 30 to 40 dB; and the existing data indicate that the density is less at UHF. In addition, the effective isotroptic radiated power (EIRP) is limited in both VHF and UHF to +30 dBw to maintain the flux density below the IRAC-established guidelines. As a consequence, the relative performance of these links can be evaluated for the same EIRP. By employing high gain backfire antenna elements colinearly stacked on each of the four VHF array elements, the hardware implementation tradeoff shows the UHF band can provide the maximum link performance for minimum size, weight, and power impact at the TDRS. The TDRS will form two beams on the satellite which will simultaneously support two command links to one or two independent LDR users, each with voice or data. Alternatively, one beam can operate F-FOV to simultaneously illuminate all users at a reduced EIRP for coherent ranging or reduced data or voice capability. Network operations analysis of the forward link conducted after the conclusion of Part I resulted in a decision to revert back to the fixed FOV approach as presented in Section 3.0.

2.3.2.2 TDRS/MDRU Transponder on TDRS

In the MDR space-to-space link, considerable forethought and analysis were given to (1) ensure flexibility for the transition between current and future needs, (2) provide operational flexibility to meet the frequency format of potential users, and (3) encompass the requirements of the TDRS specification and the Space Shuttle. Tradeoff analysis showed this link can meet the above needs with a dual frequency S-/Ku-band capability; S-band to meet the needs of current low performance ($H = 10^4 - 10^5$ bps) users, and Ku-band to meet the needs of the high performance ($H \geq 10^5$ bps) user including future TV requirement for the Space Shuttle. The TDRS provides two dual frequency MDR transponders to support simultaneously two independent users; both operating at S-band, both at Ku-band, or one at each frequency band. Link analyses show that the TDRS specifications as well as Space Shuttle requirements can both be supported with 2 m (6.5 ft) parabolic dishes. In the return link, both receivers were designed as a 10-MHz channelized receiver tunable in 20 discrete steps over the entire 100 MHz bandwidth; however, one of the receivers has the option to operate wide open with a 100 MHz bandwidth at S- and Ku-band to provide a true bent pipe repeater at the TDRS with growth capacity to handle TV. In addition, at Ku-band each receiver is tunable in four discrete steps over a 400 MHz bandwidth. This combination of a wide-open/channelized receiver provides operational flexibility for the user signal format and frequency consistent with the weight and power limitations of the Delta 2914 launch vehicle.

In the forward link, each transmit channel is wideband with wide-open 100 MHz channels at S-band, and 95 MHz channels tunable in four discrete seps over a 400 MHz bandwidth at Ku-band. One of the MDR antennas and/or trasceivers can functionally provide a backup for the ground link at S-band or Ku-band.

2.3.2.3 TDRS/GS Transponder on TDRS

The Ku-band ground link provides the interface not only between the space-to-space links and the ground station, but also the ancillary functions that are necessary to support and service the TDRS satellite itself; viz., the coherent reference for the on-board frequency source, and tracking, telemetry, and command functions.

Tradeoff analysis of multiplexing approaches shows that an FDM/FM technique, trading power and weight for increased bandwidth, meets the goals of frequency flexibility, maximum link performance, and minimum size, weight, and power. To achieve the wide-open operating flexibility in the MDR links, the Ku-band frequency spectrum for the space-to-space and ground links overlap; however, because of the spread spectrum (FM) used in the space-to-ground link, this signal will be below the user receiver noise level and will not degrade its performance. This link operates with a 0.9 m (3 ft) dish on the TDRS and a 18.3 m (60 ft dish) at the ground station. The baseline design provides a rain margin of 17.5 dB. In addition, one of the two 2 m (6.5 ft) dishes can be used for this ground link to provide an additional 6.5 dB of gain, or a total of 24.0 dB of margin for operation in emergency conditions.

The on-board ground link transmitter output is adaptive, such that a 10 dB final RF power amplifier is inserted only for operation of this link in inclement weather. Consequently, the average power requirement can be maintained at a low 7.6 watts (dc) level for normal operation.

In the TDRS/GS forward link the ground station has additional flexibility to generate high RF power; consequently, the on-board receiver is designed with a mixer front end. The RF front end is wide open (500 MHz wide) and subsequently down-converted and demultiplexed to provide the individual channelized data for relay to the LDR and MDR user spacecraft or for on-board coherent reference, tracking, telemetry, and command functions.

2.3.2.4 Frequency Synthesizer

A common frequency source locked to a pilot tone transmitted from the ground station generates the coherent signal references on the TDRS. A central frequency generator initially generates key reference signals which are subsequently used to lock in remote voltage controlled oscillators (VCO). This approach minimizes the transmission of numerous RF reference signals throughout the on-board telecommunication system, allows each VCO to operate incoherently as a backup mode, and minimizes the generation and filtering of spurious signals. In addition, the frequency source is fully redundant, and has additional backup capability to become the prime reference source so that the ground source can lock in to the spaceborne reference.

2.3.2.5 TDRS TT&C and MDR Order Wire

An S-band transponder is used to provide trilateration on ranging, an MDR acquisition beacon, and an order wire to allow S-band manned spacecraft to reconfigure the TDRS when they require support and are not in view of a ground station.

For telemetry and command (T&C) functions, a digital approach with a central multiplexer/processor was selected. All command functions will be preprogrammed into a read-only memory device which can be commanded or scheduled to operate as required.

2.3.3 Telecommunications Relay Performance

The integrated attributes of the Part I baseline TDRS design and the telecommunications system design provide a very flexible high-capacity communication tracking and data acquisition support service. The design presented provides simultaneous support to 20 LDRU's and 2 MDRU's in the return link, 2 LDRU's 2 MDRU's in the forward link, and simultaneous tracking of 20 users through each and TDRS. However, in the case of the LDRU's, even with a system that is optimized to combat interference, the service quality will be a function of the RFI levels; and in the case of the MDRU's the service quality will be a function of the characteristics of the user terminals. The quality of service (performance) is presented in the following sections in parametric form for each category of user spacecraft.

2.3.3.1 LDRU Forward and Return Link Performance

The forward link signal for LDR service must contend with multipath, RFI, and ambient noise at the user spacecraft. The impact of these effects can be minimized by obtaining the maximum processing gain through a specific bandwidth allocation. The processing gain (PG) obtainable within the system RF bandwidth constraints is proportional to the ratio of PN chip rate to the data rate. Forward link achievable data rates and rms range and range rate errors are shown in Figure 2-16.

The return link signal through the TDRS must be capable of simultaneously supporting a maximum of 20 users, and the choice of PN rates, sequence lengths, and user data rates are interdependent.

The return link performance curves shown in Figure 2-17 are based on 5-watt (37 dBm) user transmit power (P_T) levels. There are two distinct regions of the curves: (1) a lower region where RFI limits performance, and (2) an upper re region where performance is limited by noise, multipath, and other user signals. The curves are conservative in that all known potential losses have been included.

Figure 2-17 includes forward error control (FEC) coding gain of 4.7 dB. Moreover, application of the AGIPA processing enhances system performance in the presence of RFI by an additional 5 to 15 dB as shown by the shaded area.

2.3.3.2 MDRU Forward and Return Link Performance

The links designed to support the MDRU's provide service at both S- and Ku-band. The capacity of these links depends on the characteristics of the user spacecraft terminal. Figure 2-18 shows the achievable data rate as a function of user EIRP for the return link (for both S- and Ku-bands) and in Figure 2-19 for the forward link at S- and Ku-band as a function of user antenna gain. The curves include both specification and Shuttle user requirements.

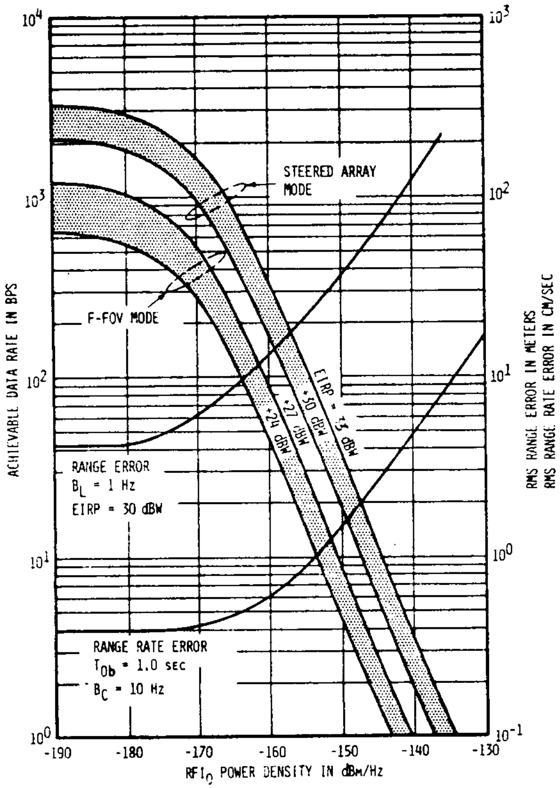

Figure 2-16. LDR Forward Link Performance Achievable Data Rate, Range Error, and Range Rate Error

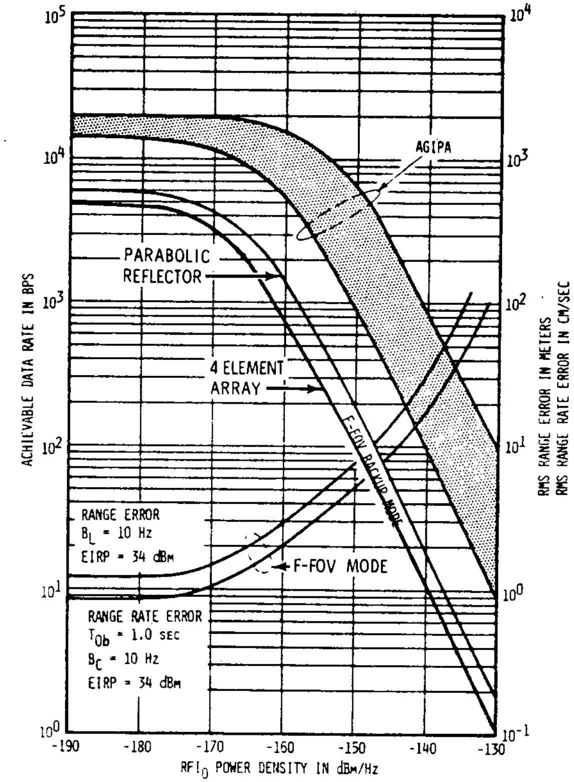

Figure 2-17. LDR Return Link Performance Achievable Data Rate, Range Error and Range Rate Error

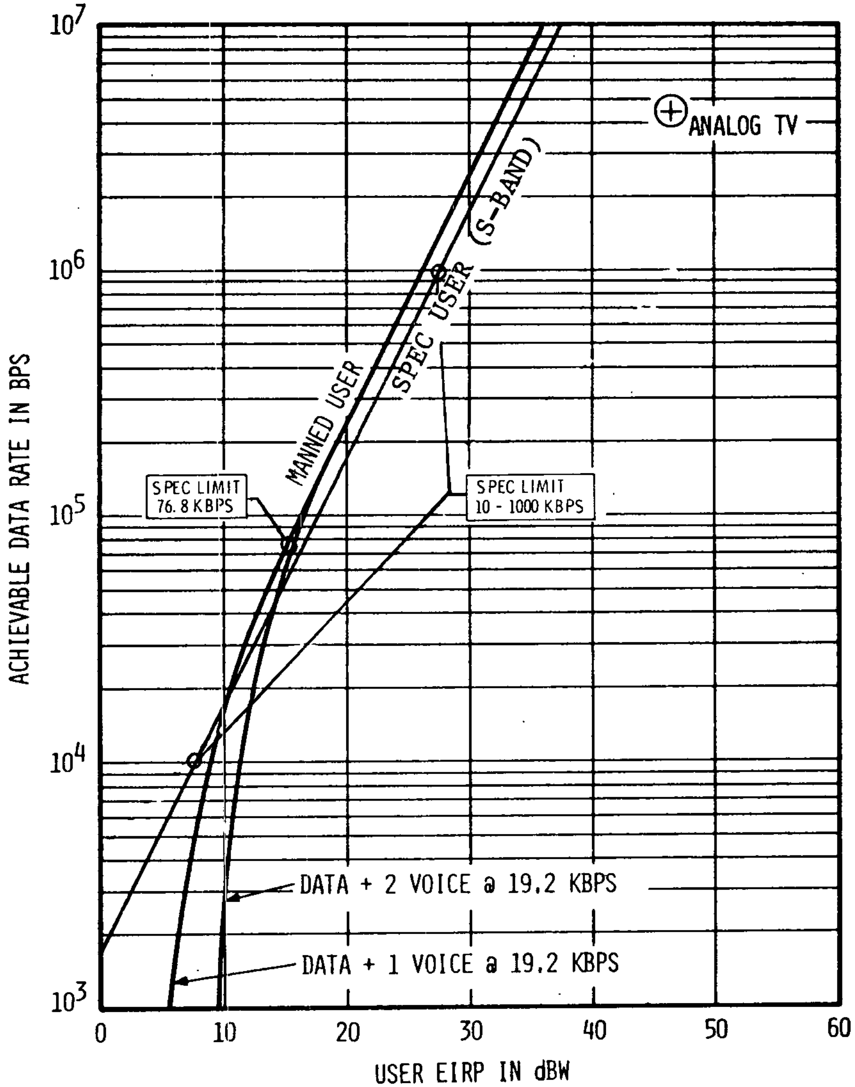

Figure 2-18. MDR Return Link Performance

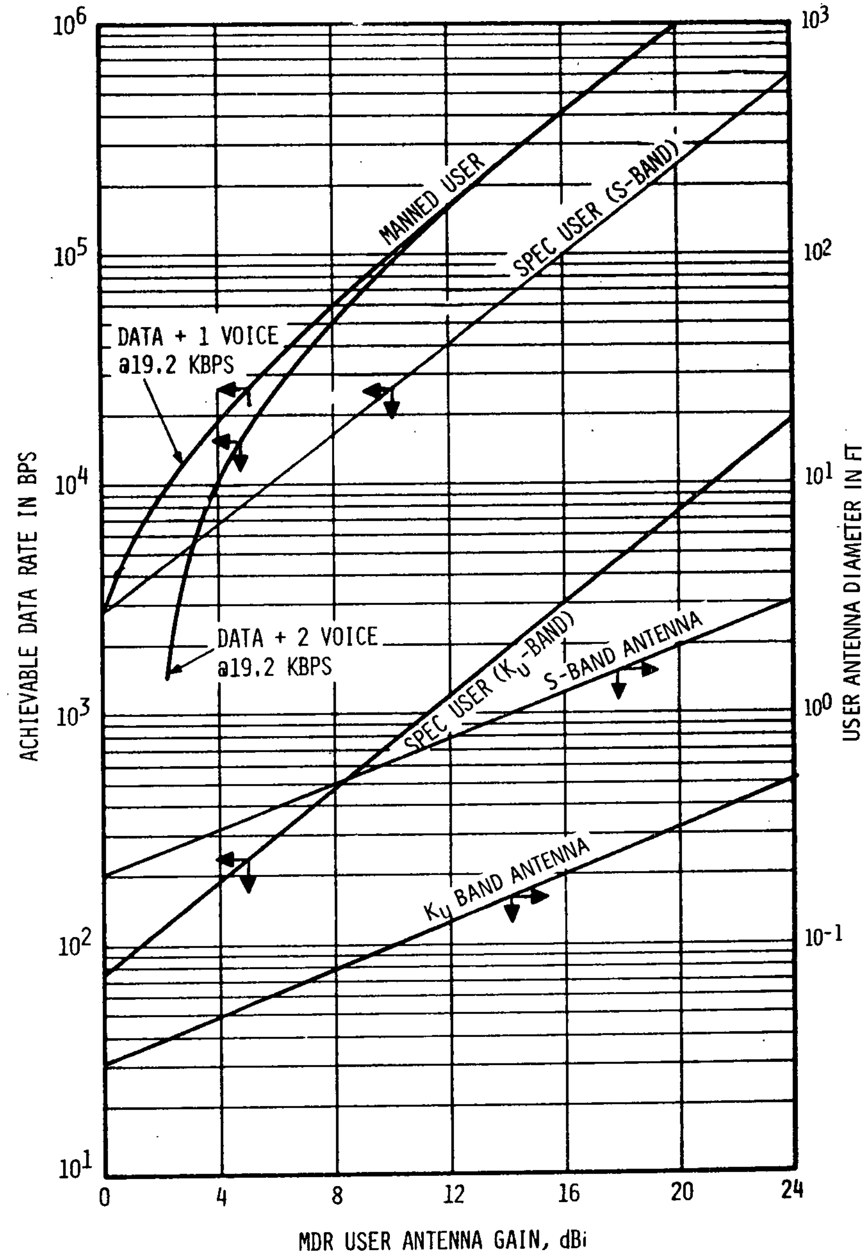

Figure 2-19. MDR Forward Link Performance

Tables 2-11 and 2-12 define the link characteristics and margins for specific channel capacities and user spacecraft characteristics for both the specified MDR requirements and for Shuttle requirements at S-band.

Table 2-11. MDR Forward Link Performance

Item		MDR S-Band User		Manned User (Space Shuttle)		MDR Ku-Band User	
Modulation		ΔPSK		ΔPSK		ΔPSK	
Transmitter power	dBw	12.5		12.5/18.5		-2.0	
TDRS antenna gain	dB	29.8		29.8		48.1	
Transmit line losses	dB	1.3		1.3		1.0	
EIRP	dBw	41.0		41.0/47.0[7]		45.1	
Losses							
Space	dB	-192.0		-192.0		-208.1	
Pointing	dB	-0.1		-0.1		-0.1	
Polarization	dB	-0.5		-0.5		-0.5	
User antenna gain	dB	G_u		3.0		G_u	
Received power	dBw	$-151.6 - G_u$		-148.6 -142.6		$-163.6 - G_u$	
System noise temperature	dB	29.1		27.3[1]		33.5[5]	
Thermal noise density	dBw/Hz	-199.5		-201.3		-195.1	
TDRS ΔCNR degradation	dB	-0.25		-0.25		-0.25	
Available C/N_0	dB-Hz	$47.65 - G_u$		52.45/58.45		$31.25 - G_u$[6]	
Support Requirements		100 bps[2]	1000 bps	Data[3]	Data - 1 Voice[4]	Data - 2 Voice[4]	BER = 10^{-5} (ΔPSK)
C/N_0 required	dB-Hz	29.9	39.9	41.7	49.6	52.4	--
Data rate achievable	bps	--	--	--	--	--	$68.4 \times G_u$[6]
Design margin	dB	$17.75 - G_u$	$7.75 - G_u$	10.75	8.85	6.05	3.0

NOTES
1 Noise temperature with an uncooled paramp.
2 For ΔPSK with a BER = 10^{-5} (E_b/N_o = 9.9 dB)
3 Data = 2 kbps ΔPSK with BER = 10^{-4} (E_b/N_o = 8.7 dB)
4 Voice is delta modulated at 19.2 kbps; carrier modulation is ΔPSK (BER = 10^{-3}; (C/N_o = 48.8 dB-Hz)
5 Tunnel diode amplifier
6 A 1.0 ft (.3 m) antenna at Ku-band provides about 28.5 dB gain (i.e., a factor of about 800) providing an achievable bit rate of about 55 kbps
7 Increased EIRP used for voice mode only

Table 2-12. MDR Return Link Budget

(a) S-Band

Item	MDR User			Manned User (Space Shuttle)			
	10 kbps[1]	1000 kbps[1]	Analog TV[2]	Data[3] (76.8 kbps)	Data + 1 Voice[4]	Data + 2 Voice	Analog TV
C/N_0 required, dB-Hz	49.9	69.9	87.54	57.59	58.17[5]	58.50[6]	87.54
FEC coding gain, dB	4.70	4.70		4.70	4.70	4.70	
Effective C/N_0, dB-Hz	45.2	65.2	87.54	52.89	53.47	53.80	87.54
Path loss[7], dB	191.1	191.1	191.1	191.1	191.1	191.1	191.1
System temperature[7], dB	26.2	26.2	26.2	26.2	26.2	26.2	26.2
Boltzmann's const, dBW/°K-Hz	-228.6	-228.6	-228.6	-288.6	-228.6	-228.6	-228.6
Pointing loss, dB	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Polarization loss, dB	0.5	0.5	0.5	0.5	0.5	0.5	0.5
G_{user}, dBi	G_u	G_u	G_u	3.0	3.0	3.0	3.0
G_{TDRS}, dBi	30.9	30.9	30.9	30.9	30.9	30.9	30.9
ΔCNR degradation, dB	1.0	1.0	1.0	0.5	0.5	0.5	0.5
System margin, dB	3.0	3.0	3.0	3.0	3.0	3.0	3.0
Required EIRP, dBw	7.6	27.6	49.94	14.79	15.37	15.70	49.44
Transmit power[8], dBw	$7.6 - G_u$	$27.6 - G_u$	$49.94 - G_u$	11.79	12.37	12.7	46.44

NOTES
1 ΔPSK; BER = 10^{-5} (E_b/N_0 = 9.9 dB)
2 Commercial quality $(S/N)_o$ = 40 dB; BW = 4.5 MHz
3 ΔPSK; BER = 10^{-4} (E_b/N_0 = 8.7 dB)
4 Voice is delta modulated at 19.2 kbps; carrier modulation is ΔPSK (BER = 10^{-3}; C/N_0 = 48.8 dB-Hz)
5 Combine value of data (C/N_0 = 57.59) and one voice (C/N_0 = 48.8)
6 Combine value of data (C/N_0 = 57.59) and two voice (C/N_0 = 51.8)
7 This value is selected at the point where the product of the path loss and system temperature is maximum
8 Transmit power into antenna

(b) Ku-Band

Item	10 kbps[1]	1000 kbps[1]	Analog TV[2]
C/N_0 required, dB - Hz	49.9	69.9	87.54
FEC coding gain, dB	4.7	4.7	-
Effective C/N_0, dB - Hz	45.2	65.2	87.54
Path loss[3], dB	208.1	208.1	208.1
System temperature[3], dB	26.7	26.7	26.7
Boltzmann's const., dBw/°K - Hz	-228.6	-228.6	-228.6
Pointing loss, dB	0.1	0.1	0.1
G_{user}, dB	G_u	G_u	G_u
G_{TDRS}, dB	48.3	48.3	48.3
ΔCNR degradation, dB	0.5	0.5	0.5
System margin	3.0	3.0	3.0
Required EIRP, dBw	6.7	26.7	49.04

1 ΔPSK; BER = 10^{-5} (E_b/N_0 = 9.9 dB)
2 Commercial quality $(S/N)_0$ = 40 dB; BW = 4.5 MHz
3 This value is selected at the point where the product of path loss and system temperature is maximum.

2.4 SPACECRAFT MECHANICAL AND STRUCTURAL DESIGN

The guidelines during the synthesis of design concepts and the criteria for decision making, for both design variations and for selection between alternative configurations, was maximum relay support capability and maximum telecommunications flexibility within the constraint of providing a low-cost, low-risk design, launched on a Delta 2914. Although the problems addressed during Part I of this study were extremely challenging, the resulting spacecraft concept is simple and straightforward. The major constraints within which the spacecraft mechanical design evolved are summarized in Table 2-13.

Table 2-13. Design Constraints

Constraint	Derived From
Maximum payload: 788 lb (357.5 kg) (Including burned-out apogee motor case)	Delta 2914 capability
Volume	Delta 2.4 m (8-foot) shroud
No shadowing on solar cells	Power loss cannot be tolerated
RCS system temperature 4.4 C (>40 F)	Hydrazine freezes
Electronics temperature < 40 C	Reliability
RCS jets operable in a stowed configuration	Stability and pointing requirements in transfer orbit
Unfurlable antenna dishes not desired	Reliability, surface control, and minimum cost
Clear sensor FOV before and after deployment	Stability and pointing requirements

The spacecraft provides the support for, integrates and protects the communications, electrical power, attitude control, and propulsion subsystems in both the launch and deployed environments. The deployable elements are packaged to withstand the structural and vibrational loads experienced during launch on the Delta 2914 booster and the spinning accelerations imposed during third-stage burn and transfer orbit. After apogee motor burnout, despin and stabilization maneuvers, the antennas and solar panels are deployed to their extended positions.

Areas of primary concern in the development of the Part I baseline design were packaging and deployment mechanization, antenna design and sizing, weight control, and optimization of the configuration for general-purpose support of multiple user spacecraft.

Figure 2-20 illustrates the arrangement of antennas and solar array panels symmetrically grouped around the central spacecraft body in both the stowed and deployed conditions. The two MDR parabolic reflector antennas are supported on struts on each side of the LDR UHF-VHF four-element array of backfire and dish-on-rod antennas with the body-mounted TDRS/GS antenna in the center. The one-degree-of-freedom solar panels are deployed above and below the spacecraft beyond the solar shadow limits of the antennas. Backup TT&C omni whip antennas located around the rear of the spacecraft are utilized during launch and spacecraft orbital maneuvers prior to the deployment of the primary antennas.

The spacecraft is packaged for launch within the 2.4 m (8 ft) shroud of the Delta 2914. The MDR antennas are folded forward in a face-to-face position with slight mutual rotation to allow the antenna feeds and supports to clear each other. The 0.9 m (3 ft) diameter TDRS/GS dish antenna is located at the front of the spacecraft between the rear rims of the angled MDR dishes. The LDR elements are packaged into four cylindrical shapes and positioned around the TDRS/GS antenna behind the MDR dishes forward of the structural body of the spacecraft. Both solar array panels are folded down around the sides of the spacecraft, allowing room for clearance with the side-mounted booms of the MDR antennas and the ACS thrusters.

Deployment of appendages to the operational configuration is achieved by extending the solar arrays and the two MDR antennas aft to their positions on each side of the spacecraft, and deploying the four elements of the LDR UHF-VHF array laterally and then extending them forward. The LDR antenna is deployed laterally by spring-loaded linkage joints which lock into the extended position. Release of the deployable elements is initiated by solenoid activation of packaging and restraint latches by ground commands. After the lateral deployment the LDR antenna elements are extended forward by motor-driven STEM units mounted at the rear of the elements and the extended STEM's become the center rod supports for the disc elements.

The spacecraft body (Figure 2-21) consists of inner aluminum tapered cones around the apogee motor, a transverse equipment shelf of aluminum honeycomb and the outer body shells of aluminum honeycomb that close off and protect the internal equipment and house the thermal louvers.

The apogee motor is installed on the spacecraft centerline by an accurately machined titanium ring attached to the apogee motor flange and is bolted to the inner structural cone. The attachment ring is located along the Z axis of the spacecraft to accurately position the apogee motor. The motor nozzle extends 6.4 cm (2.5 in.) beyond the spacecraft separation plane. The apogee motor is not jettisoned after burnout and remains in the spacecraft body.

The equipment shelf is an aluminum honeycomb bulkhead that provides the primary mounting surface for the equipment. It is fabricated of 3.8 cm (1.50 in.) thick aluminum honeycomb core with .025 cm (0.010 in.) aluminum face sheets. The insert panels are of similar construction and bolt to the main bulkhead. Equipment is mounted on both the bulkhead and the insert panels, allowing for simultaneous and sequential subsystem installation and checkout during manufacture.

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The outer body shells are bonded 1.3 cm (0.50 in.) thick aluminum honeycomb core with .025 cm (0.010 in.) reinforced fiberglass face sheets, and the shell halves are bolted to the inner shell structure and the outer rim of the equipment shelf using angle clips.

Both MDR antennas are dual Ku- and S-band frequency, 2 m (6.5 ft) diameter, solid-face, parabolic reflectors with two-axis gimbal drives. They are mounted on tubular booms and deployed 3.52 m (138.50 in.) outboard on each side of the spacecraft. As shown in Figure 2-20, the packaging restraints of the 2.4 m (8 ft) shroud in conjunction with the volume required for the spacecraft body, solar arrays, and packaged UHF-VHF array elements, limit the two solid-face antennas to a maximum of 2 m (6.5 ft) diameter. Solid-face reflectors were chosen over furlable antennas because they meet all Part I baseline ERP and G/T requirements, and for simplicity of design, accuracy of surface tolerances suitable for Ku-band antenna frequencies, weight, rigidity, and reliability.

The LDR antenna is an array of dual frequency backfire and disc-on-rod elements spaced uniformly around the spacecraft centerline. Each element consists of a series of discs-on-rod, a UHF dipole and mesh ground plane, and a VHF dipole and mesh ground plane. When deployed, each element is supported and positioned by a swing arm support link that places the element centerline at 1.24 m (48.75 in.) from the X and Y axes.

To package the UHF-VHF array elements in the available space with the other antennas and spacecraft body, each array element must be packaged in a volume 76 cm (30 in.) in diameter by 40.6 cm (16 in.)long, and must be deployed with a simple, lightweight, reliable arrangement with no loose packaging parts. The mechanical design of these elements is illustrated in Figure 2-22. The two large-diameter VHF ground planes fold down on spring-loaded arms which are equally spaced around a central mesh-frame disc 76 cm (30 in.)diameter. The VHF dipole assembly is reduced to 76 cm (30 in.) diameter by compressing the spring-loaded dipole extensions. The discs, the UHF dipole, and ground plane, and VHF dipole are closely stacked against the front of the STEM actuator. The mesh rims of the two ground planes are folded into the space between the two ground plane discs in their stacked position. The ground plane arms are spring-loaded and restrained by lock pin shafts that extend forward from a locking disc behind the VHF ground plane. To release the mesh arms and allow the ground plane meshes to extend to their full diameters, this locking disc is rotated on the case of the STEM by a cable system activated by the lateral deployment of the support struts. This small rotation releases the mesh arms from the lock pin shafts and extends the mesh surfaces.

The ground link antenna is a .9 m (3 ft) diameter Ku-band parabolic reflector located on the centerline of the spacecraft, forward of the spacecraft body. To provide clearance without blockage or reflective problems for the antenna beam between the LDR array elements, the antenna is positioned fore and aft as shown in Figure 2-20.

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The tracking, telemetry and command backup antennas are VHF omni whips. One set of four whips located radially around the rear of the spacecraft is utilized during launch when the primary antennas are in their stowed position. These TT&C antennas are behind the stowed solar panels and after the shroud is jettisoned they have a clear field of view to the ground tracking stations. After three-axis stabilization and deployment of the primary antennas, a TT&C backup to the TDRS/GS Ku-band link is supplied by another set of VHF omni-whip antennas mounted around the rim of the TDRS/GS Ku-band antenna.

Ku- and S-band tracking beacon antennas are mounted on the forward side of the spacecraft body adjacent to and offset from the TDRS/GS antenna. The Ku-band acquisition beacon antenna consists of a section of waveguide which terminates into a conical horn with a peak gain of nominally 12 dB and a 31-degree (.54 rad) FOV. The S-band acquisition beacon/order wire antenna is a 5 cm (2 in.) diameter helix mounted on a 28 cm (11 in.) ground plane to provide performance similar to the Ku-band beacon antenna.

The solar array panels are deployed to the positions shown in Figures 2-20 and 2-23, and rotate at one revolution in 24 hours to maintain solar illumination on the front of the array.

To permit packaging the solar array in the launch configuration, the panels are constructed on a radius that allows the panels to fold around the top and bottom of the body. The gaps between the panels permit clearance for the MDR antenna booms and operation of the ACS thrusters in the launch configuration. The solar panel support links fold forward and under the solar panels and are restrained during launch loading by solenoid-operated latch-release mechanisms. In this position the solar panels are active and provide electrical power during the spinning mode.

After stabilization of the spacecraft at synchronous orbit, ground commands activate solenoids to release the linkage. Spring-loaded fittings extend the solar panels and lock the joints. As the panels extend, spring-loaded hinges, between the panel halves and the support link, which are also released by the solenoid mechanism, extend the solar panel halves forward from their circular stowed configuration to provide a flatter shape for increased efficiency. Further flattening of the panels would result in undue complexity since satisfactory performance is provided by this design. After full panel deployment, the panel drive actuators at the base of the linkage system orient the solar panel sun sensors with the sun and begin the one revolution per day to maintain solar alignment.

The drive system, consisting of two actuators driven by electric stepper motors mounted on the rear surface of the equipment shelf, supports and rotates the deployed panels. Section A-A on Figure 2-23 illustrates the details of the Spar Aerospace Products actuator.

Most of the satellite equipment was installed on the front and back faces of the equipment shelf to provide proper center-of-gravity location, eliminate center-of-gravity travel during use of expendables, and provide ease of assembly, servicing, and checkout. Figures 2-24 and 2-25 are front and rear views of the equipment bulkhead with the subsystems installed. During Part II of the study,

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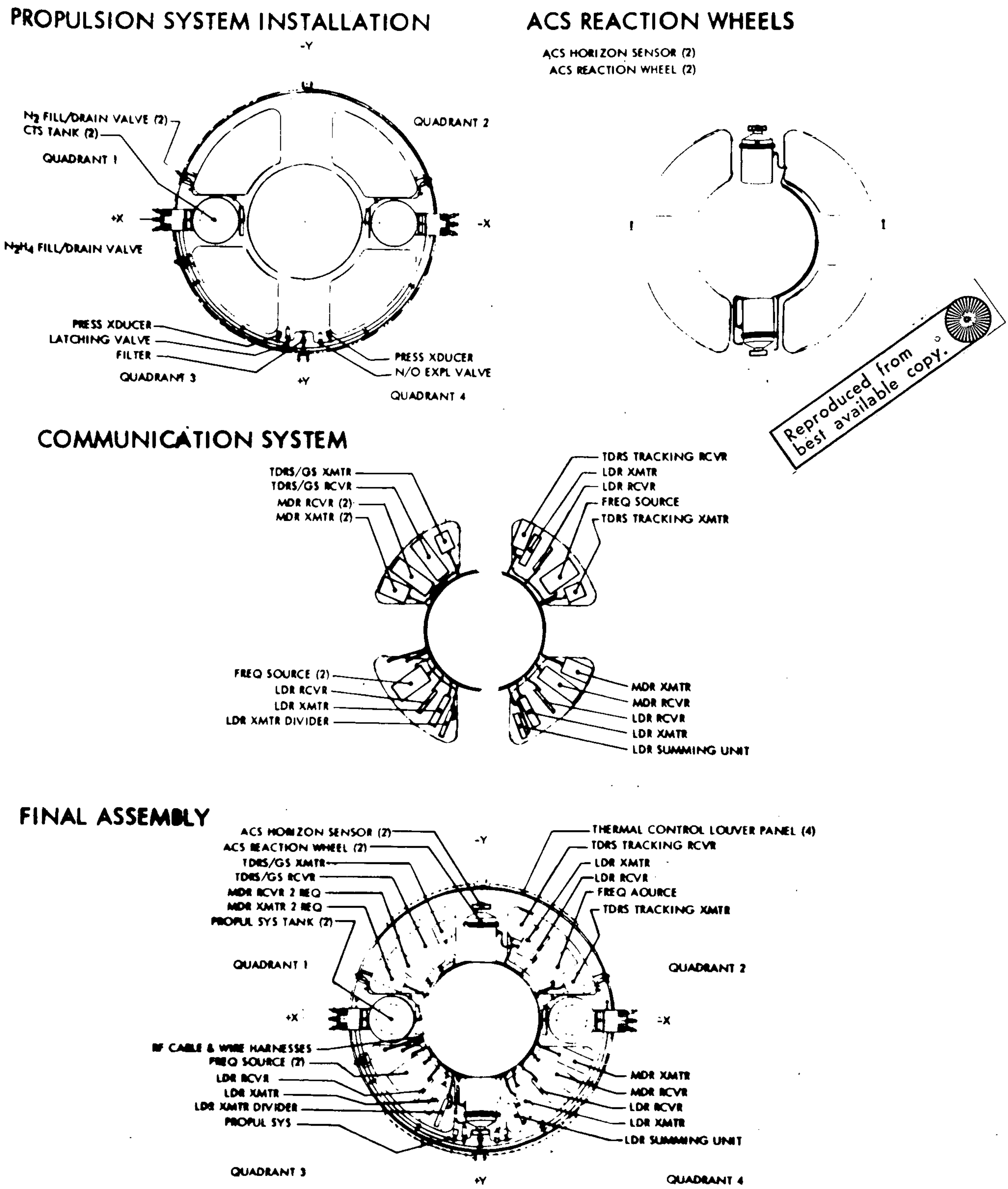


Figure 2-24. Part I--Equipment Shelf--Front View

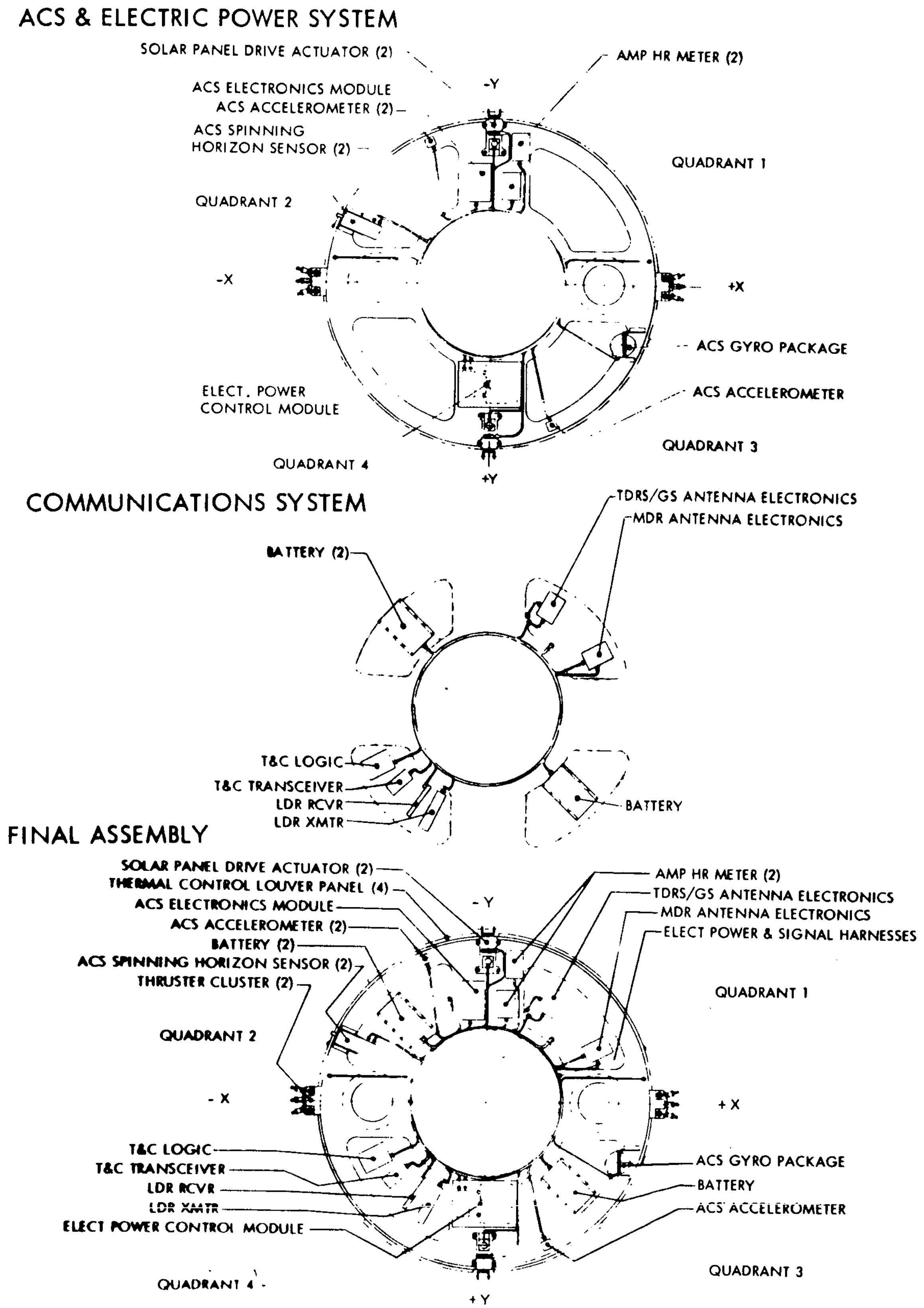


Figure 2-25. Part I--Equipment Shelf--Rear View

a more detailed evaluation of the heat transfer characteristics of this packaging approach led to a decision to move the high-power components from the positions shown to locations adjacent to the space radiators. This redesign is covered in Section 3.0.

After assembly and checkout of the propulsion system on the shelf, the attitude control system is installed and checked out. The communication system components are mounted on honeycomb panel inserts which match the equipment shelf cutouts, and are interconnected for bench checkout and testing as a subsystem unit. The communications panel inserts are then attached to the equipment shelf and the electrical power subsystem is installed on the rear face of the shelf and all cabling installed and tested.

The spacecraft equipment located on the equipment shelf provides convenient servicing through access panels on the front and rear outer body shells. At initial installation, the subsystems are mounted on the shelf prior to the assembly of the outer body shells and are fully exposed for checkout. Propulsion system fill and drain valves are accessible and convenient with the spacecraft in the stowed configuration. All launch restraints and solenoid release mechanisms are visible for prelaunch checkout and inspection. Prelaunch electrical check points are adjacent to the MDR antenna attachment fittings and are readily accessible. Access to the Delta third-stage motor through the existing access holes in the attach fitting remains clear and convenient.

Access to and visual inspection of the mating Delta clamp securing the TDRS to the attach fitting is clear of structure and convenient for mating prior to launch.

The weight summary for the Part I baseline TDRS is shown in Table 2-14. Center-of-gravity location and moments of inertia for both the launch and deployed configurations are summarized in Table 2-15.

2.5 ELECTRICAL POWER SUBSYSTEM

Design of the electrical power subsystem was accomplished with consideration for five-year lifetime, eclipse operations, sun angle, battery redundancy, excess power dissipation, and energy transfer mechanisms. A primary design goal was to use only flight-proven technology and hardware. As a result, nickel-cadmium batteries were selected for the energy storage assembly. The selection of two batteries for the baseline was made after examining the impact of a battery failure and considering system weight. The loss of one of the two batteries still permits an eclipse load of 195 watts for the full 72 minutes which is sufficient to operate partially one forward LDR and one forward MDR link. A direct energy transfer concept was selected since this permits power transfer directly from the solar array to the loads without any in-line power conditioning. The majority of time (all but 80.2 hours per year) is spent in direct sunlight with spacecraft loads supported directly from the solar array, thereby minimizing power conditioning losses.

During transfer orbit an average power requirement of 44 watts must be provided. The baseline design utilizes the solar arrays in the stowed configuration. This is done by curving the panels and exposing the cells to simulate

Table 2-14. Part I--TDRS Weight Summary

	Weight (lb)	Weight (kg)
Communications		
Electronics	122.2	55.5
Antennas	117.9	53.5
Attitude stabilization and control	57.7	26.2
Electric power	97.0	44.0
Solar array	58.6	26.6
Structure	91.0	41.3
Thermal control	23.9	10.8
Auxiliary propulsion hardware	38.4	17.4
	606.7	275.0
Propellant + N_2 (2-65° - 15-day station changes)	49.3	22.4
Total spacecraft	656.0	297.6
Contingency	82.0	37.2
Allowable P/L (Delta 2914 + CTS apogee motor)	738.0*	334.8*
Empty apogee motor case	50.0	22.7
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 separation weight (27° transfer orbit)	1490.0	675.9

*5 deg/day drift orbit

Table 2-15. Part I--Moments of Inertia

Configuration	Weight		Z Center-of-Gravity (from S.P.)		Inertia					
					I_{z-z}		I_{x-x}		I_{y-y}	
	lb	kg	in.	cm	slug-ft^2	kg-m^2	slug-ft^2	kg-m^2	slug-ft^2	kg-m^2
Launch										
Full apogee motor	1484	673.2	25.7	65.3	107	145	132	179	122	165
Burnout apogee motor	788	357.5	26.1	66.3	94	127	119	161	108	146
Deployed										
Full propellant	788	357.5	21.6	54.7	465	630	234	317	308	418
Fuel expelled	738.7	335.0	21.6	54.7	458	621	234	317	301	408

*x and y center of gravity, 0.00 and 0.00

a body-mounted panel. The dependence on batteries is minimized by providing a projected area equivalent to 44 watts output for the minimum allowable sunline/spacecraft orientations.

The EOL power requirements of 300 watts (daylight average) and 213 watts (eclipse average) are shown in Table 2-16, and the functional block diagram in Figure 2-26. Power is supplied to the loads from a central regulated 28 ± 1 volt dc bus. Voltage is regulated by a shunt regulator operating as a variable load across lower sections of the solar array panels. Each panel is approximately 2.09 m^2 (22.5 ft^2) in projected area for a total area of 4.18 m^2 (45 ft^2). This area provides a beginning-of-life power of 466 watts, and an end-of-life power of 400 watts.

Table 2-16. Electrical Power Requirements

SUBSYSTEM	DAYLIGHT AVE	DAYLIGHT PEAK	ECLIPSE AVE	ECLIPSE PEAK	TRANSFER ORBIT
ATTITUDE STAB. & CONTROL	16.5	100.5	13.5	85.5	5.2
HEATERS	2.0	25.2	1.0	25.2	10.4
TT&C	10.5	15.0	10.5	15.0	14.5
TELECOMM. SERVICE	249.3	344.3	179.4	274.5	4.8
EPS (ARRAY DRIVE AND CONTROLS)	15.7	33.2	8.5	33.2	5.2
CONTINGENCY	6.0		0		
SUBTOTAL	300.0		213.0		40.2
BATTERY CHARGE	48.0				
POWER CONDIT. LINE LOSSES	52.0		17.0		3.9
ARRAY OUTPUT E.O.L.	400.0				
DEGRAD ALLOW (5 YRS)	66.0				
ARRAY OUTPUT B.O.L. (45 FT2)	466.0				
BATTERY LOAD			230.0		44.0

BASELINE: NORMAL DAYLIGHT 2 LDR (ONE VOICE 25% DUTY CYCLE)
2 MDR (1S + 1 KU BANDS)
ECLIPSE 1 LDR (VOICE 25% DUTY CYCLE). 1 MDR (S-BAND)

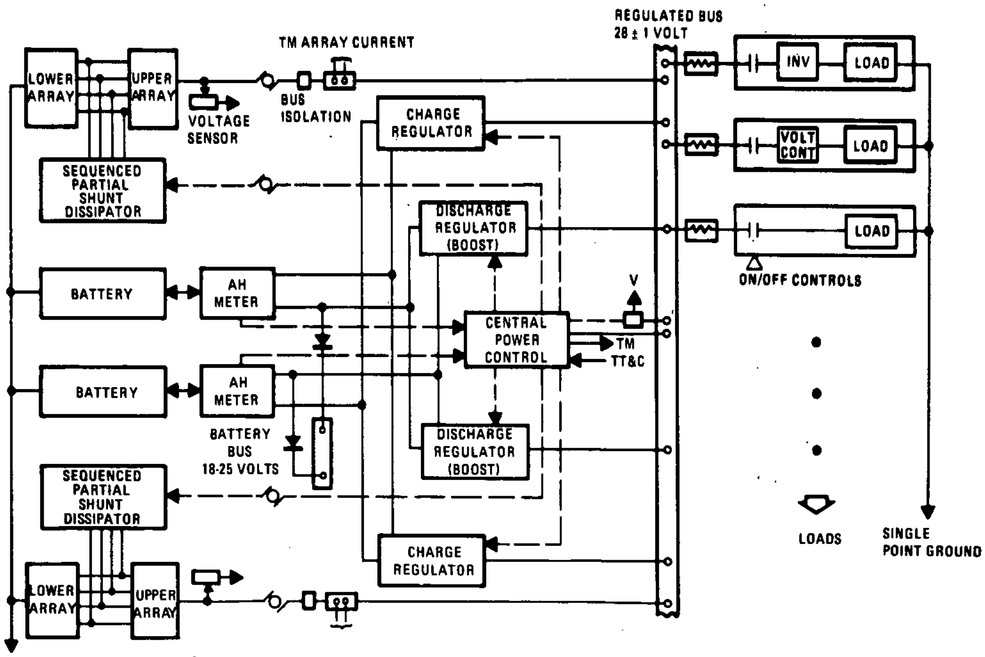


Figure 2-26. Electrical Power Subsystem Block Diagram

The central power control unit controls the various EPS operational modes. It detects the differences between the main bus and reference voltage levels. Since the solar array is a constant power source, to supply larger amounts of power to subsystem loads, battery charging must be inhibited and/or the boost regulator activated to supply power from the batteries. To provide for sun eclipse energy demands, a battery charging allowance of 48 watts (net to the batteries) is included in the sizing model to permit parallel battery charging.

Figure 2-27 shows a typical battery charging time for the two NiCd batteries. Several charge characteristics will be available to permit more rapid battery charging depending upon availability of solar array power. The voltage converter is required to provide 18 volts to the LDR for data transmission. This converter operates from the 28-volt regulated bus. As a general utility service the EPS delivers regulated 28-volt dc power with the exception of 18 volts dc for the LDR transmitters; all other nonstandard power conditioning will be part of the user equipment.

The beginning-of-life design is capable of supporting the higher requirements for two S-band links by utilizing the contingency (6 watts) and the solar array degradation allowance (66 watts). Dependence on S-band is expected to decline with mission time. The end-of-life design will support one S-band and one Ku-band or two S-band links without emergency voice.

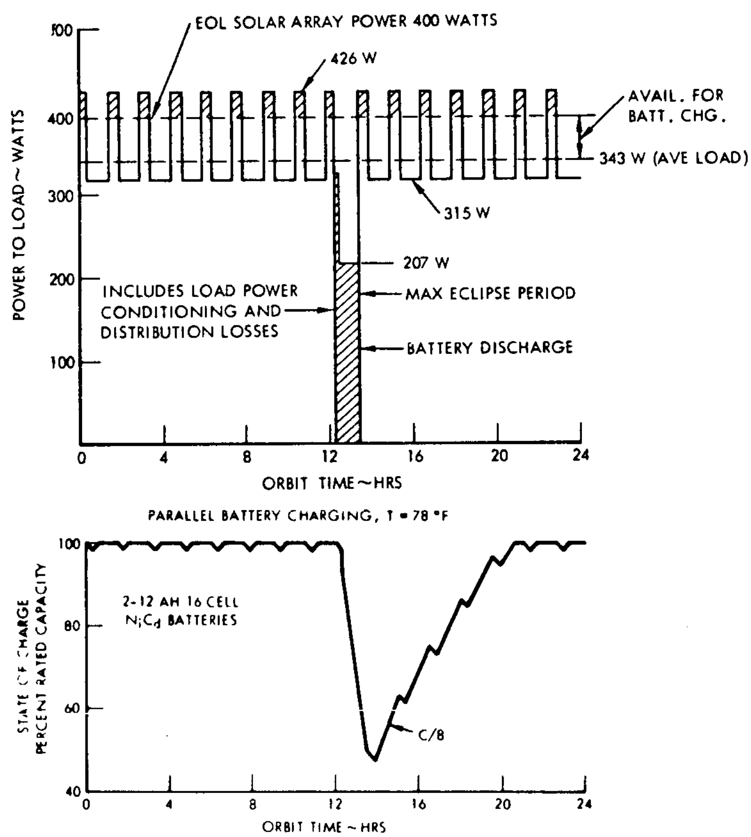


Figure 2-27. Time Required to Charge Batteries

A weight summary of the EPS is shown in Table 2-17.

Table 2-17. Electrical Power Subsystem Weights

Components/Assemblies	Weight lb	Weight kg	Potential Supplier
SOLAR ARRAY	(58.6)	(26.6)	
Panels (2)	38.6	17.5	EOS, Ferranti
Drive mechanism (2)	15.0	6.8	BBRC, Spar, G.E.
Linkage and fitting (2)	5.0	2.3	NR
POWER COND. & DISTRIBUTION	(52.7)	(23.8)	
Charge and discharge	11.3	5.1	G.E.
Central control and logic	5.1	2.3	G.E.
Packaging	4.9	2.2	G.E.
Shunt dissipators	2.4	1.1	G.F.
Amp-hour meters	4.0	1.8	Engr. Magnetics
Power conditioning	5.0	2.3	
Cabling	20.0	9.1	NR
ENERGY STORAGE			
Batteries (2)	44.3	20.1	G.E.
Total	155.6	70.5	

2.6 ATTITUDE STABILIZATION AND CONTROL SUBSYSTEM (ASCS)

The decision to use three-axis stabilization as the basic mode of operation allowed a great deal of creativity in the design of the spacecraft which ultimately was translated into the flexible high-capacity telecommunications relay system discussed previously. The design of the system and the sizing of its elements were governed primarily by the characteristics of the disturbance torques and system reliability goals. System analysis activities indicated pointing accuracy requirements imposed by the telecommunications function were not severe, since all antenna beams are steerable, and therefore had minimum influence in the design activity. Of greater importance is the need for accurate knowledge of spacecraft attitude to establish a reference for pointing the antennas for S-band.

The baseline ASCS has two functionally independent modes of operation. The first is the transfer orbit mode which uses spin stabilization. The second mode is the operational mode on orbit which uses momentum bias/momentum transfer three-axis stabilization. The momentum bias counters the cyclical components of the solar pressure torques and the reaction torques from the gimbaled antennas with the minimum expenditure of propellant.

A block diagram of the complete ASCS is presented in Figure 2-28, and a summary of key performance requirements and system parameters is presented in Table 2-18. All system components are existing flight-qualified hardware with the exception of minor changes in the horizon scanners. In addition to being a high performance design approach, the main attribute is high reliability and the absence of any single point failures.

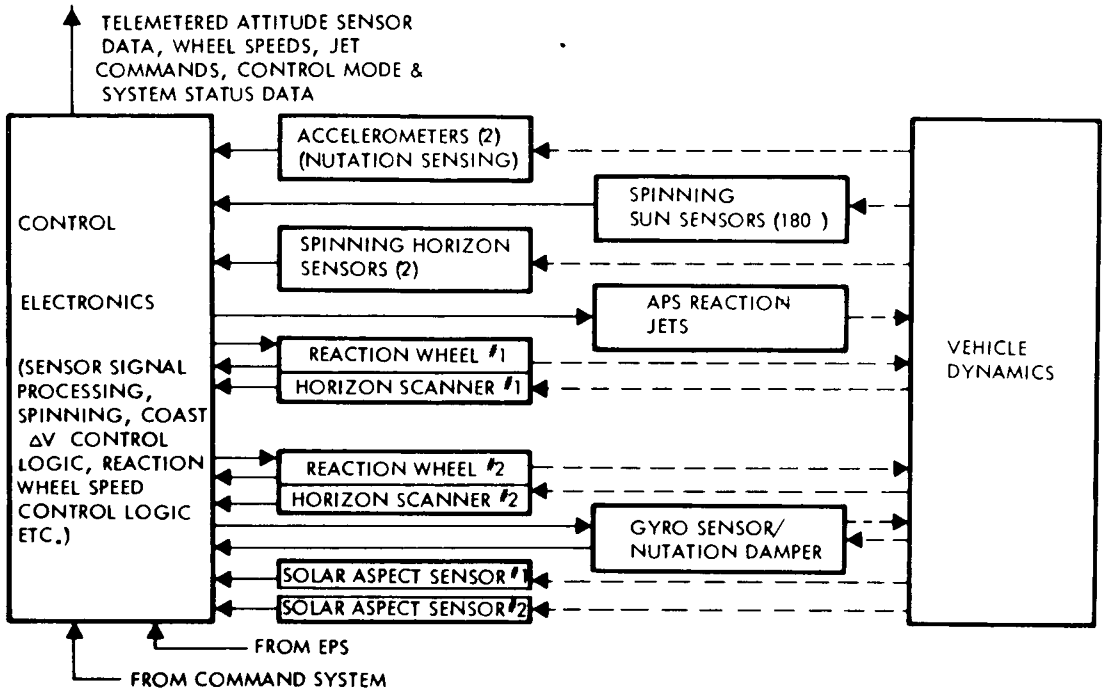


Figure 2-28. Attitude Stabilization and Control System

Table 2-18. ASCS Key Performance Requirements and System Parameter Summary

Attitude Determination Accuracy (Knowledge) Roll: 0.25 degree (.0044 rad) Pitch: 0.25 degree (.0044 rad) Yaw: 0.25 degree (.0044 rad)
Spacecraft Attitude Pointing Accuracy: ±0.58 deg (.0101 rad)/axis
ASCS System Weight: 26.2 kg (57.7 lb) total Spin control: 3.0 kg (6.6 lb) (0.85 percent of SOI weight) 3-axis stabilized: 23.2 kg (51.1 lb) (6.58 percent of SOI weight)
Propellant Requirements: 25.1 kg (55.3 lb) Spinning attitude control: 3.7 kg (8.2 lb) 3-axis attitude control: 0.7 kg (1.5 lb) Orbit change delta-V maneuvers: 20.7 kg (45.6 lb)
Stationkeeping Accuracy: ±0.125 deg (.0022 rad) (corrections approximately every 17 days)
Momentum Storage and Subsystem Requirements Momentum bias: 17.1 kg-m^2/sec (12.5 ft-lb-sec) Maximum cross axis momentum transfer: .27 kg-m^2/sec (±0.2 ft-lb-sec)
Momentum Dumping Maneuvers: No more than once per day

Spin stabilization (90 rpm) is employed from booster separation until after apogee motor firing. Established flight control techniques were selected for this phase. Active nutation control utilizing accelerometers to sense nutation and reaction jets to provide correction torques are used for this minor axis spinner. The spin vector precession maneuver is controlled using sun sensor pulses for proper phasing of reaction jet torque commands. Attitude is determined using body-mounted spin scanned horizon sensor and digital sun sensor data. The effect of propellant motion on nutation stability has been examined and is not expected to be a problem.

The baseline three-axis-stabilization system employs two momentum wheels with integral earth horizon scanners in a shallow "V" configuration and a gas bearing gyro/nutation damper normal to the V-plane (see Figure 2-29). This configuration was selected over the other approaches primarily because its freedom from single point failures gives an appreciable reliability advantage over the other candidates. The momentum wheels provide a nominal momentum bias of 17.1 kg-m^2/sec (12.5 ft-lb-sec) along the Y-axis. The momentum bias and its natural quarter orbit coupling permit passive yaw control during coasting flight. Pitch control is obtained by driving the wheels in unison. Differential wheel speed control is used to transfer momentum into the Z-axis to obtain active nutation damping and roll control. The two-degree-of-freedom gas-bearing gyro provides active yaw control during delta-V maneuvers. The variable speed gyro can also serve as a backup nutation damper by transferring a small amount of momentum into the spacecraft Z-axis. The gyro data also may be used in a backup gyro-compassing mode to aid in attitude determination in the event of failures in the primary attitude determination sensors. This gyro is normally left with power off except when performing delta-V maneuvers or a contingency function.

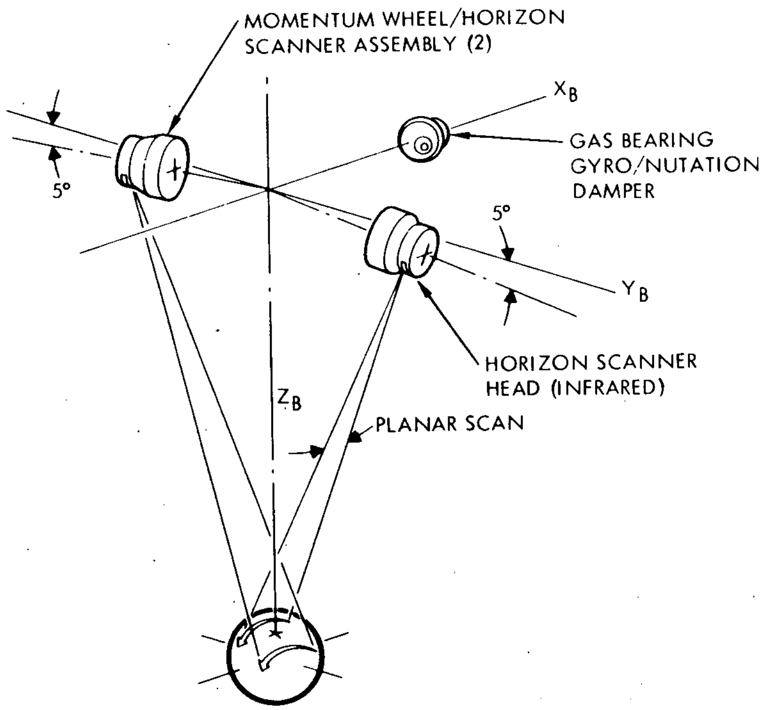


Figure 2-29. Momentum Storage Subsystem Arrangement

This basic concept is being developed at NR for use on a military satellite that will be flown in early 1974. The stabilization and control system design and spacecraft stability have been thoroughly analyzed using digital simulation programs and through use of an air-bearing table on which a similar configuration was tested. Rigid body dynamics analyses were used to determine the effects of antenna slewing on spacecraft attitude.

Three-axis attitude determination is performed at the ground station utilizing horizon sensor and solar aspect sensor data. The selected momentum wheel/infrared earth scanner assemblies use the wheel motion to scan the earth. This approach provides horizon sensing to better than 0.1 degree (.0017 rad) accuracy at minimum weight and cost. Digital solar aspect sensors mounted on the solar arrays provide the remaining data necessary for three-axis attitude determination to better than 0.25 degree (.0044 rad) in each axis.

The dominant disturbance torques acting on the TDRS are solar pressure (long term) and antenna motion (short term). Digital simulation programs were used to study these effects and to develop sizing requirements for the momentum storage subsystem. Analysis of the TDRS pointing accuracy requirements indicate that they are less severe than for most contemporary communication satellites. Attitude determination of 0.25 degree (.0044 rad) per axis and vehicle pointing accuracies of 0.58 degree (.0101 rad) per axis are found to be adequate. The baseline system has substantial growth potential in that much better sensing and costrol accuracies are possible without significant changes in the system components.

The reaction jet thruster arrangement is shown in Figure 2-30. The two-quad 16-jet configuration with initial and final thrust levels of 1.2 N (0.27 lb) and .4 N (0.09 lb) satisfies all mission requirements. The system can accommodate the worst combination of two jet failures (with an increase in propellant consumption in some cases). The configuration employs more thrusters than the minimum number possible to prevent physical interference with other spacecraft components.

State-of-the-art technology was used and in all instances flight-proven equipment was selected, needing no flight qualification testing. Figure 2-31 is a schematic of the RCS whose key features are:

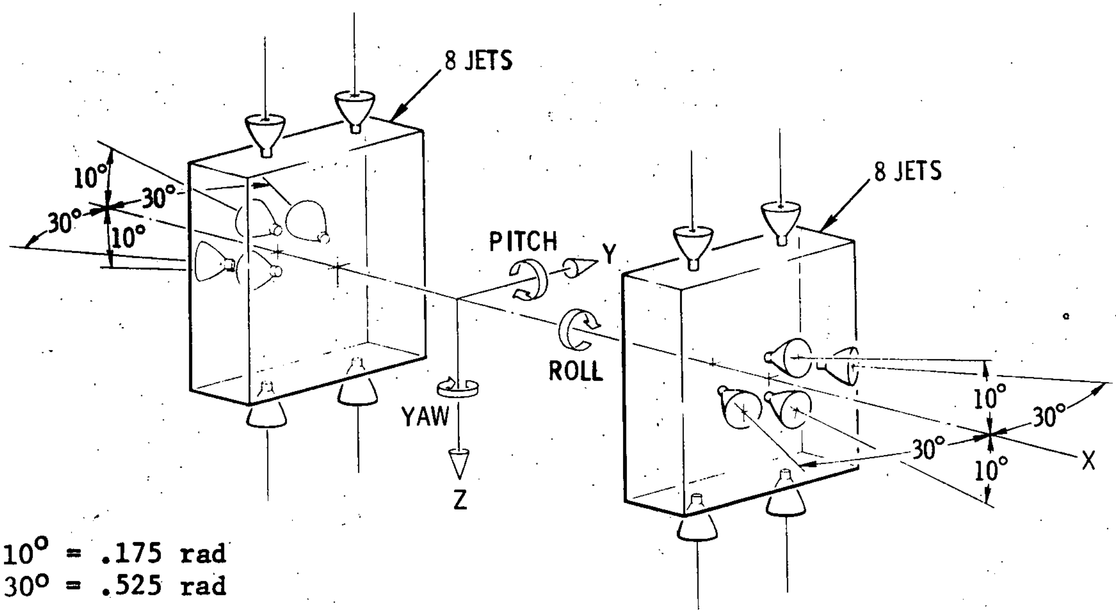
1. It operates in a blowdown mode (3.35 to 1).

2. Separate gaseous nitrogen fill/drain valves avoid loss of total pressurant if a leak occurs in one tank.

3. Either propellant tank can be isolated should a leak occur.

4. Each thruster is equipped with a redundant (two seats in series and two coils in parallel) propellant flow valve.

5. Any individual thruster can be isolated in the event of a "runaway" thruster if the redundant valve fails open.



10° = .175 rad
30° = .525 rad

Figure 2-30. APS Engine Arrangement

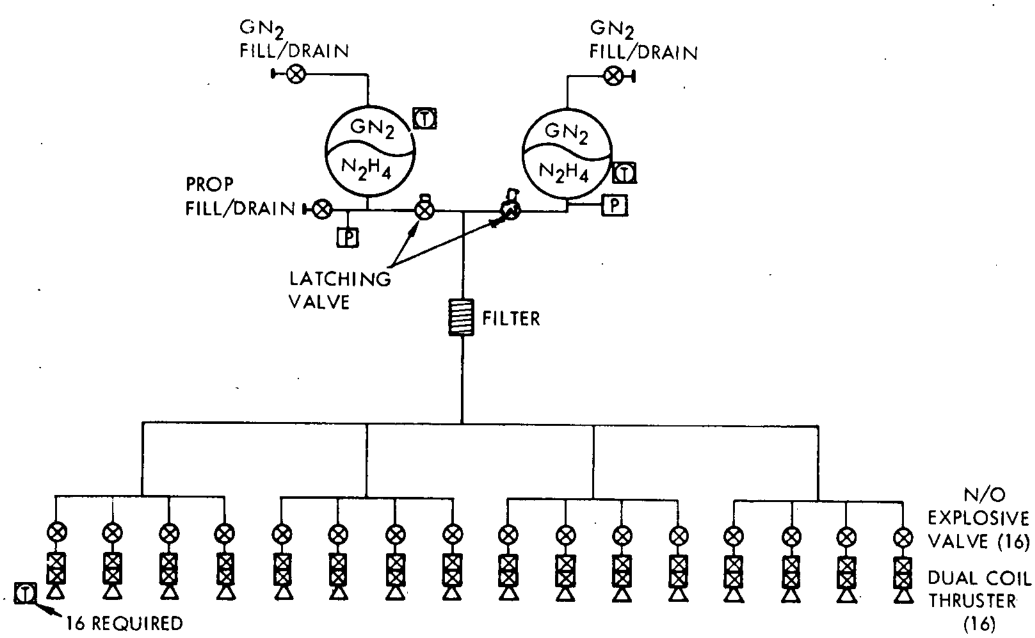


Figure 2-31. Auxiliary Propulsion System

6. The entire subsystem is made up of existing and proven components.

7. The 16 thrusters are housed in two identical and compact modules. All maneuvers can be accomplished after any two thrusters fail.

To satisfy the five-year operational requirement, an improved elastomer composite, ethylene propylene terpolymer (EPT-10) was chosen as the expulsion diaphragm material. This material is now in its third year of long-life testing by Pressure Systems, Inc. All evidence indicates that a five-year-life EPT-10 diaphragm will be demonstrated before the TDRS enters the implementation phase. Current programs using or planning to use EPT-10 include Pioneer F&G, ATS F&G, MVM, CTS, AERO-ERNO, P-95 (military), Viking Lander and the 4.5-year Jupiter/Saturn mission.

Another long-life consideration is the thruster catalyst. A large number of "cold" starts could impose serious damage to the catalyst as a result of the startup pressure overshoot. The solution to this problem incorporates active thermal control directly on the catalyst bed. Data from thruster manufacturers indicates adequate protection is provided if the bed is heated to 149 C (300 F) prior to each start.

The weight of the propulsion system hardware and trapped propellant is 17.4 kg (38.4 lb). Useful propellant weight is 25.1 kg (55.3 lb). Table 2-19 shows the RCS propulsion requirements established by the stabilization and control analysis and the resulting propellant requirements.

Table 2-19. Propellant Requirements

	Impulse		Propellant	
	N-sec	(lb-sec)	kg	(lb)
Precess to SOI attitude (150 deg)	5400	(1212)	2.5	(5.5)
Nutation control-transfer orbit (30 hr)	540	(121)	0.3	(0.7)
Despin (from 90 rpm)	1960	(442)	0.9	(2.0
Acquire local vertical (5 times)	835	(188)	0.4	(0.9)
Correct apogee injection error (3)	19300	(4350)	9.0	(19.8)
Stop drift at final station (5 deg/day)	4890	(1100)	2.3	(5.0)
Longitudinal stationkeeping (5 years)	3630	(818)	1.7	(3.7)
Momentum dumping (5 years)	520	(117)	0.3	(0.6)
Longitudinal station change 2 to 65 degrees - 15 days	16750	(3770)	7.7	(17.0)
Total			25.1	(55.3)

2.7 APOGEE MOTOR

A modified version of the Thiokol TE-M-616 solid motor was selected as the apogee injection motor. It is presently in development for the Canadian Communication Technology Satellite (CTS). Modifications of the motor include a 15.2 cm (6 in.) reduction in nozzle length which reduces inert weight by 3.2 kg (7 lb) and off-loading 20.8 kg (46 lb) of propellant. Both changes are minor and no requalification is required. A summary of the modified CTS apogee motor characteristics is given in Figure 2-32, and a summary of its performance in conjunction with the Delta 2914 was presented previously.

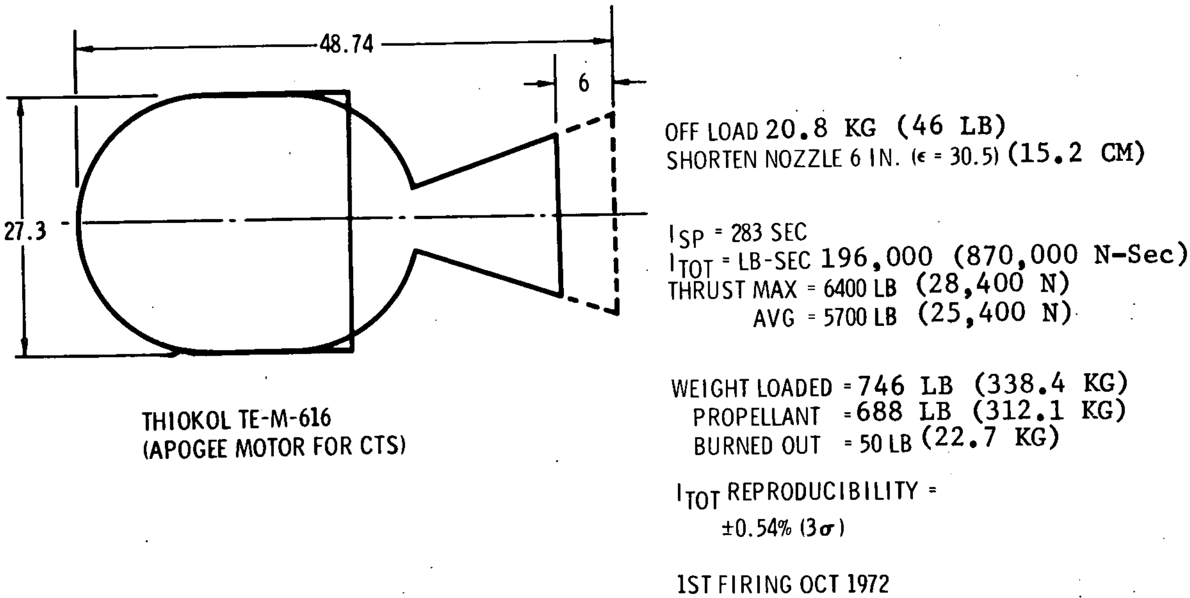


Figure 2-32. Apogee Motor

2.8 THERMAL CONTROL

The thermal control system accommodates induced heat loading from the apogee burn and motor case heat soakback and from the broad range of equipment power dissipation. The system is sized for a combination of operational and standby housekeeping modes, and the seasonal changes of the long-duration mission. The design employs standard techniques similar to those used on other satellites, and considers seasonal solar load variations on louvered radiator panels. During transfer orbit and eclipse standby operation when power dissipation is minimal, makeup heating maintains the RCS at operational temperatures. The system performs during all mission phases without constraining attitudes, maneuvers, or durations.

The mission thermal requirements are summarized for each mission phase in Table 2-20. The TDRS upper temperature is derived from the communications equipment needed to provide high reliability and efficiency without weight penalty. Therefore, the operational temperature limit of the communications subsystem electronics was established at 40 C (104 F). The TDRS lower design temperature limit of 4.4 C (40 F) is derived from the freezing point of hydrazine. The limit applies to all RCS components to provide continuous operational

Table 2-20. Mission Thermal Requirements

Phase	Duration	Environment	Considerations	Requirements
Prelaunch checkout	Hours, as required	Hot or cold day Full power load	Air flow provisions	Provide duct ports
Launch and ascent	4 minutes	Delta fairing heat soak-back, depressurization	Fairing rises to +400 F (204 C)	Jettison fairing, provide vent
Parking orbit	20 minutes	100 n mi (185 km) earth orbit; Min power loads	Fixed attitude, non-uniform heating Stored panels and antennas	
Transfer orbit	24 hours	Solar, with earth emission and albedo near perigee Minimum power loads	90 rpm spin-up, thrust axis attitude orientation maneuver for SOI; stowed configuration	Limit cooldown of TDRS and apogee motor
Insertion and pre-operations	1 to 2 hr	Solar, power-up	Deployment of panels and antennas	Apogee motor case heat soakback into spacecraft
Synchronous orbit	5 years	Solar, operational nominal and reduced power loading	Fixed earth orientation. Once a day roll to sun. 26° (.45 rad) seasonal out of orbit plane sunline change. Up to 72-minute eclipse.	Maintain subsystems design temperature. Reject heat loads. Limit eclipse cooldown

capability. The battery temperature limits are established from stringent requirements to limit energy storage degradation.

The thermal control design was constrained by the following:

1. The TDRS toroidal body offers limited surfaces for efficient heat rejection due to the daily solar incidence.

2. The TDRS equipment is separated from the heat rejection surfaces.

3. The transfer orbit and the spare TDRS operating modes are powered down.

The TDRS body thermal control design consists of (1) 1.3 m^2 (14.4 ft^2) of louvered panel area divided equally between the four equipment quads and located on the north and south surface, (2) multi-ply insulation, and (3) solar reflector thermal control coatings on the louver base and on the insulation blanket. The performance of the system is shown in Figure 2-33 for limiting insulation performance values. The design performs within the control range for normal operation, but the spare TDRS power load is insufficient to balance the heat leakage at the required limit temperature. Cooldown also occurs in transfer orbit for a combination of reduced power load and solar heating. Without constraining the TDRS sunline angle or powering up beyond the demand load, temperatures below allowable occur. Makeup heating is required for the RCS. The additional RCS power requirement is available because of the reduced TDRS demand during these two phases; makeup amounts to 18 watts and 73 watts for transfer orbit and for the spare TDRS in eclipse, respectively.

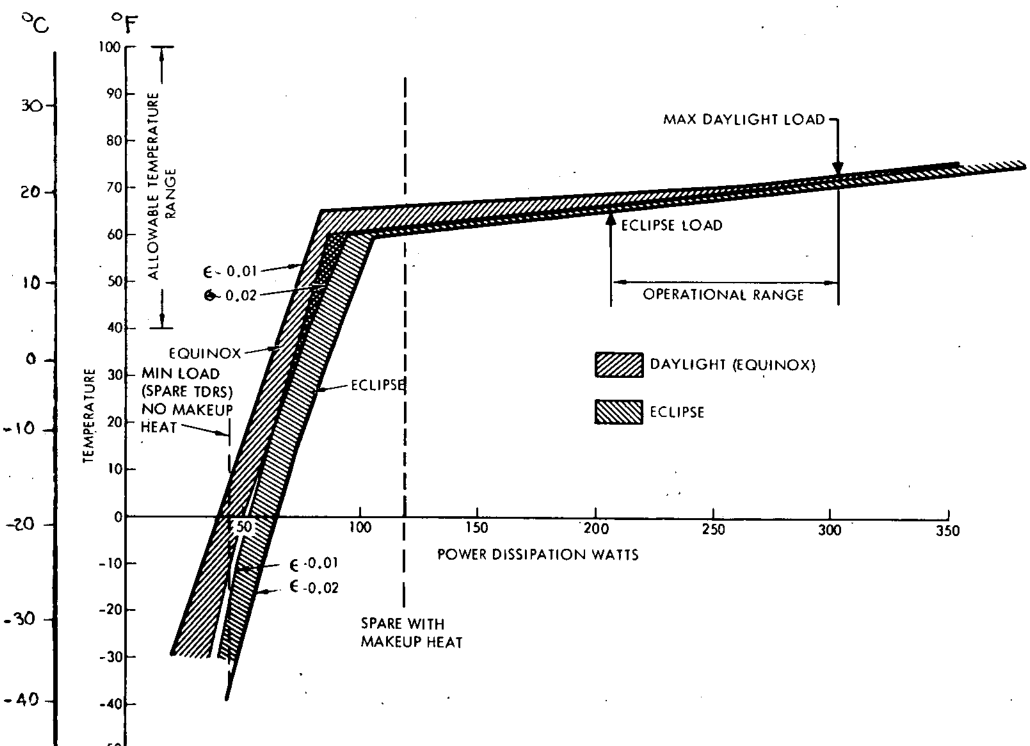


Figure 2-33. TDRS Mean Temperature

Exposed structure, arrays, antennas, etc., are passively controlled within design limits and gradients. Insulation is applied to masts and feeds to maintain alignment. The RCS thruster catalyst chambers are heated to 149 C (300 F) before operation for easy starts. Heat soakback from the expended apogee motor case is attenuated by high-temperature insulation lining the tunnel.

A thermal math model was developed to aid in establishing equipment locations. The results of the simulations provide thermal design criteria and preliminary values of equipment temperature.

2.9 RELIABILITY

The major reliability goal set for the TDRS design effort was to provide a satellite reliability of 0.8. This goal was established in conjunction with the GSFC Project Office after preliminary system reliability analyses showed the relationship between satellite reliability and the probability of mission success, which was defined as having one or two satellites remaining at the end of five years. This relationship is shown in Figure 2-34. In developing the curves a booster reliability of 0.95 and an apogee motor reliability of 0.98 were used.

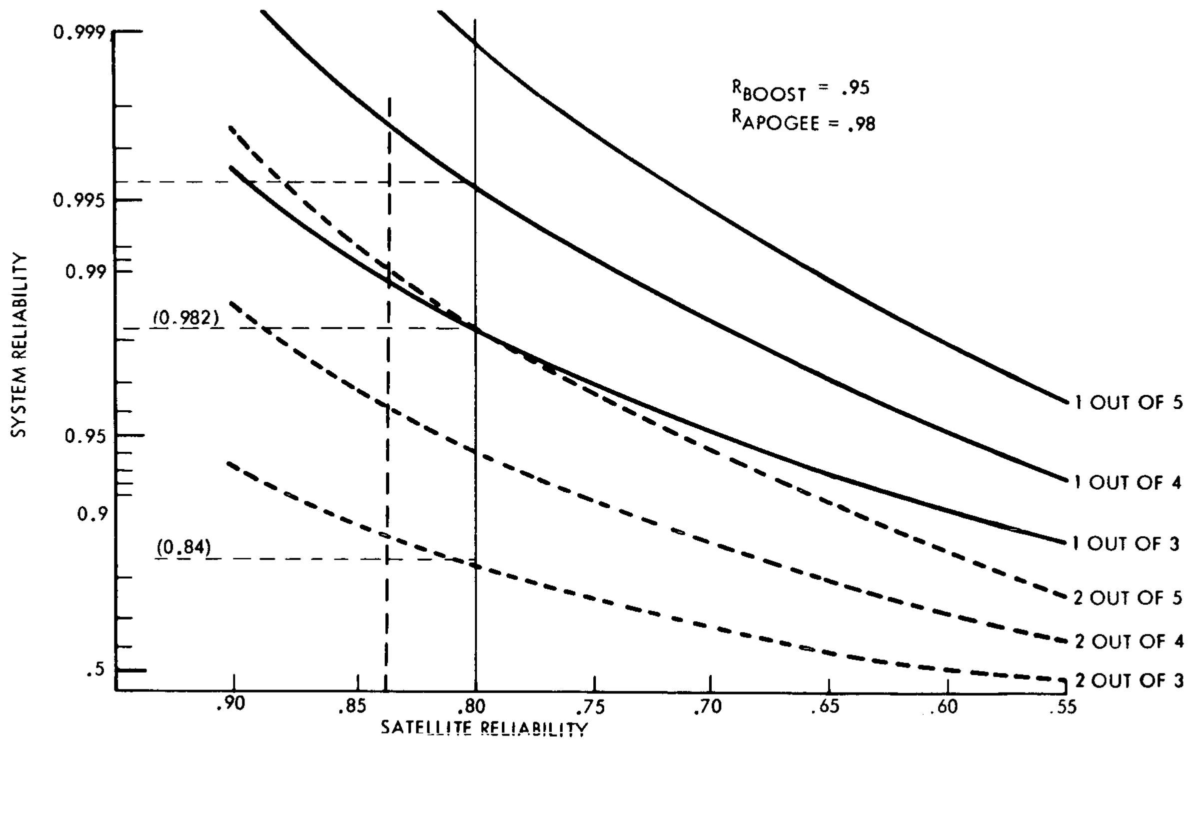


Figure 2-34. System Reliability Versus Satellite Reliability

The curves show the probability of mission success where mission success is defined as the ability of each TDRS to service 20 LDR users and 2 MDR users on the return link and 2 LDR and 2 MDR users simultaneously on the forward link. Reduced forward link capability is permitted during eclipse. This capability exceeds that required in the statement of work and a higher probability of success is obtained using the SOW capability requirement of one MDR user on forward and return and one LDR user on forward link. An even higher reliability will occur for reduced operations below the SOW capability since the satellite in nearly all cases degrades gracefully, allowing mission continuation.

Subsystem reliability analyses show that for the excess capability case, the satellite reliability is 0.804 and for the SOW capability the satellite reliability is 0.844.

Table 2-21 shows the system reliabilities taken from Figure 2-34 of one or two spacecraft remaining in full operation at the end of five years for these satellite reliabilities.

Table 2-21. Probability of Mission Success

Satellite Capability	Probability of Success			
	Excess Capability Case		SOW Capability Case	
Number of Spacecraft in Full Operation (Initially)	1	2	1	2
3	.983	.840	.990	.880
4	.996	.950	.998	.965
5	.999	.983	.9995	.991

These high goals were achieved by adhering to a design philosophy throughout the spacecraft of minimizing single point failures, using high reliability components, and using redundancy whenever necessary. The weight margins provided by the selected design approaches permitted the use of redundancy in all critical areas.

A reliability predictive analyses defined potential problem areas and indicated where redundancy or a modified design approach was worthwhile. Logic diagrams were prepared for each subsystem at the component level and the reliability values obtained are shown in Table 2-22.

Table 2-22. Reliability Prediction

Subsystem	Predicted Reliability
TT&C	.966
Communication	.915
Structures and mechanical	.999
Attitude	.962
Auxiliary propulsion	.997
Electrical power	.962
Thermal control	.999
Total satellite	.804

The failure modes and effects analyses and the resultant design changes eliminated all critical path failures that would seriously impact spacecraft reliability. In those few instances where single point failures were retained, the probability of occurrence was minimal and in addition the cost, weight, and complexity of elimination made their removal impractical.

2.10 USER TRANSPONDER DESIGN

The communications system analysis activity summarized in Section 2.3 resulted in the definition of requirements for the TDR satellite, the user spacecraft transponders, and the ground station telecommunications equipment. The requirements were established to ensure that the telecommunication goals could be met when confronted with the technical problems of multiple access, multipath and, potentially, high RFI environment. The following summarizes the characteristics of the transponder design concepts synthesized for the low and medium data rate user spacecraft. The design of the HDR transponder is covered in Section 3.5.

2.10.1 LDR Transponder

A simplified schematic block diagram of the LDR transponder is shown in Figure 2-35. The receiver is designed to tune to any one of four UHF carrier frequencies used by the two TDRS's. The carrier in the forward link is phase-modulated by a 167 K chip/sec pseudo noise (PN) sequence to discriminate against multipath and to distribute the signal energy radiated from the TDRS to conform to the IRAC requirements. A 1 M chip/sec PN sequence is used in the return link to permit code division multiple access of 20 LDR users through a common channel in the TDRS with sufficient process gain to achieve a specific level of performance for any one user in the face of 19 interfering users. The use of PN sequences in both the forward and return links provides a vehicle for deriving range information.

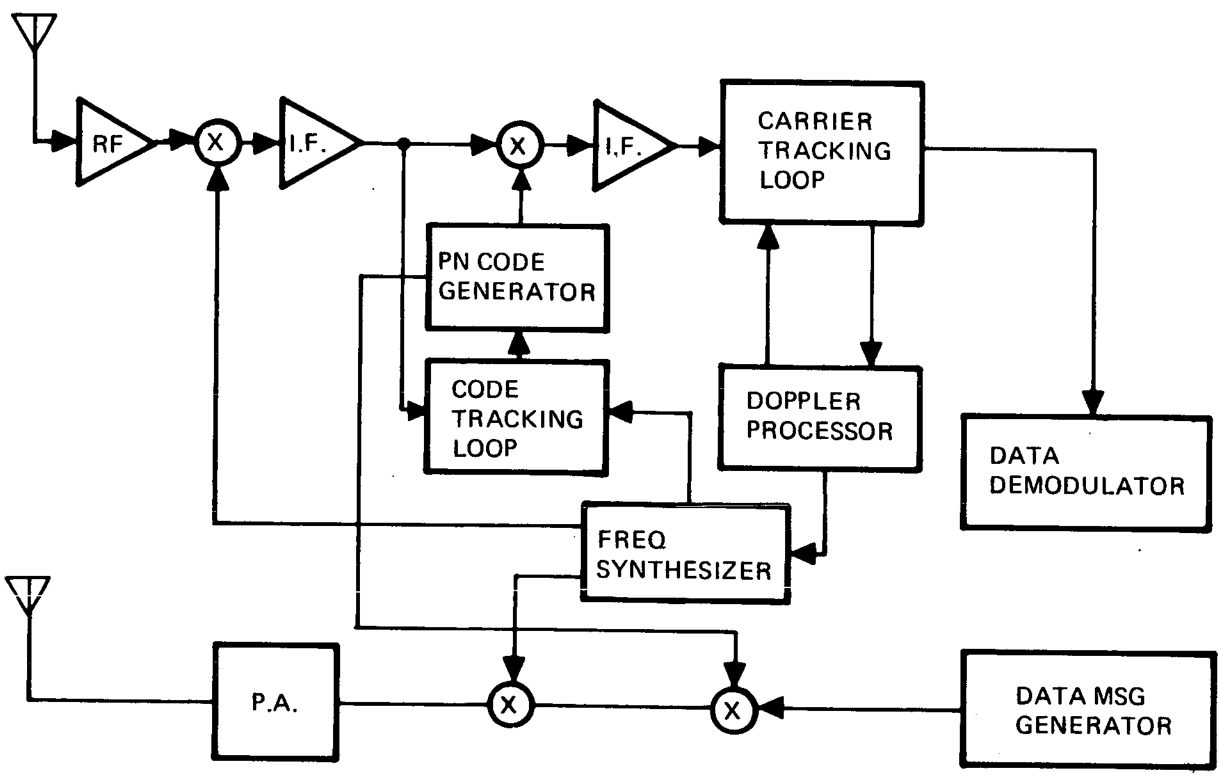


Figure 2-35. LDR Transponder

The data are delta PSK modulated with convolutional encoding to save power through error control. The demodulation process in the receiver is rather straightforward. The transmitter, except for the PN modulation, is a standard transmitter configuration.

A preliminary estimate of the size and prime power requirements of a typical LDR transmitter and receiver is: transmitter--power, 16 watts dc power; size, 3687 cc (225 in^3); and receiver--power, 12 watts; size, 3195 cc (195 in^3).

2.10.2 MDR Transponder

The MDR transponder has a single channel S-band receiver. The carrier in the forward link is modulated by a single 5 Mchip/sec PN sequence during the code acquisition phase to distribute the signal energy radiated from the TDRS to conform to IRAC requirements. During the ranging phase a second PN sequence of 500 Kchip/sec rate is modulo-2 added to the first. A 500-Kchip/sec PN code generator synchronized to the uplink 500-Kchip/sec code is used to modulate the downlink carrier.

In both manned and unmanned users the data are delta PSK with convolutional encoding for error control. In the manned user case, the data and delta modulated voice are time division multiplexed to form a serial data stream. A simplified schematic diagram of the MDR transponder is shown in Figure 2-36.

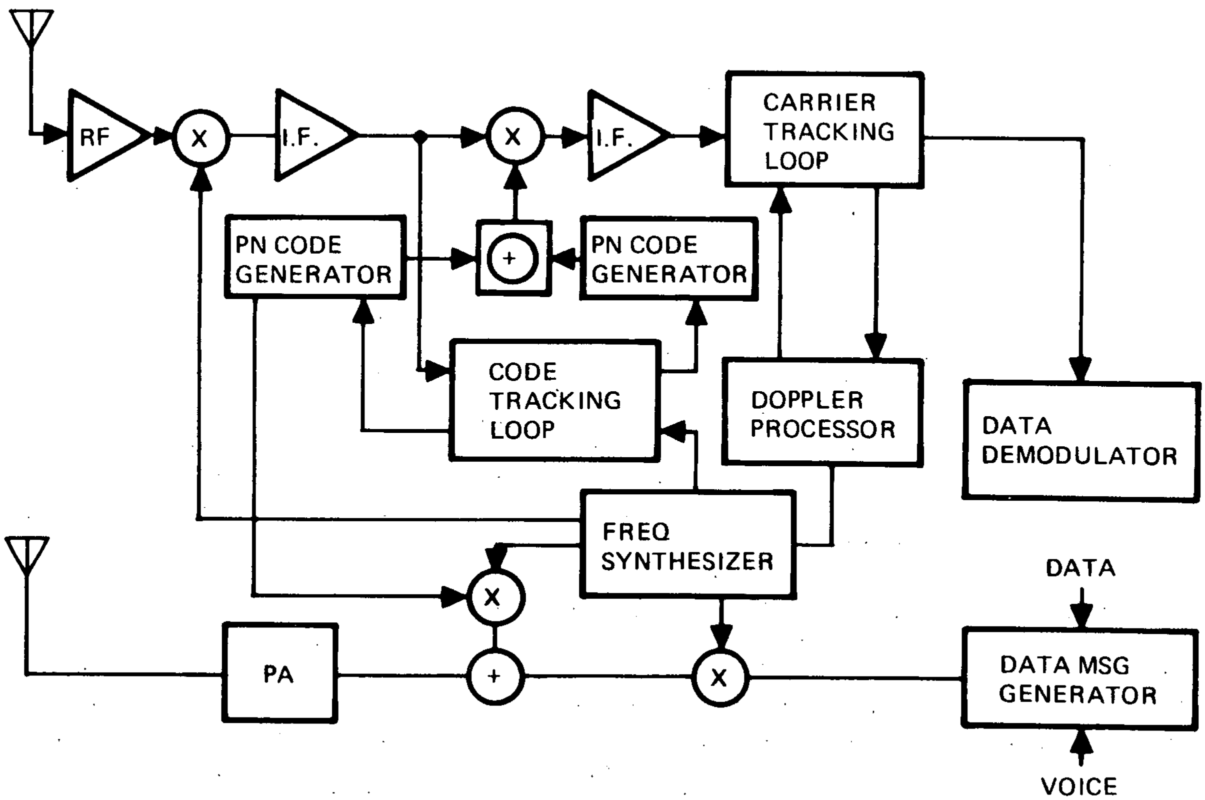


Figure 2-36. MDR Transponder

A preliminary estimate of the size and prime power requirements for the MDR transmitter and receiver are: transmitter--power, 33 watts dc power; size, 3933 cc (240 in^3), and receiver--power, 12 watts; size 3359 cc (205 in^3).

2.11 NETWORK OPERATIONS AND CONTROL

Primary emphasis in the network operations and control studies was to achieve a real-time operational system. At the same time, practical considerations required maximum use of existing and planned facilities and organizations for cost-effectiveness. The operational studies consisted of:

1. Development and description of a TDRS system concept.

2. Description of the operational and functional interfaces between primary TDRS system elements for network operations.

3. Functional analysis of TDRSS operations and development of functional flows for all important functions and operations to a third/fourth level of detail.

4. Development and description of a sequence of events and operations performed by the TDRS system elements in a representative operational phase mission of two operational TDRS spacecraft providing service to LDR and MDR user spacecraft.

Figure 2-37 illustrates the TDRS system concept. The figure shows three groups of elements: GSFC-located ground elements, remote-located ground elements, and space elements. The space group consists of the two operational TDRS spacecraft at geosynchronous altitude with two-way communications links to ground station, and the low data rate and medium data rate user spacecraft, including the Space Shuttle.

The ground system provides and maintains real-time communications between all elements. It also implies that long land lines and the number of required ground stations can be minimized, reducing cost and complexity. Larger users who now control their own spacecraft will continue to do so with TDRS and can now be given real-time capability. Small users who now operate through the system elements for spacecraft control and who do not require real-time control may continue either with the other elements or with TDRS to maintain routine operations but with provisions for TDRS to take care of urgent requirements.

The ground system will have GSFC-located and remote-located elements. The TDRS RF ground station very likely will be remote-located because of the large 18.3 m (60 ft) antennas and facilities involved. It is shown to be at Rosman, N.C., although it can be at any other desired location. Other remote elements are MSC at Houston, some user control centers, Launch Control Center at KSC, and STDN tracking stations. All these remote elements are linked directly to the Network Operations Control Facility at GSFC by NASCOM communication lines.

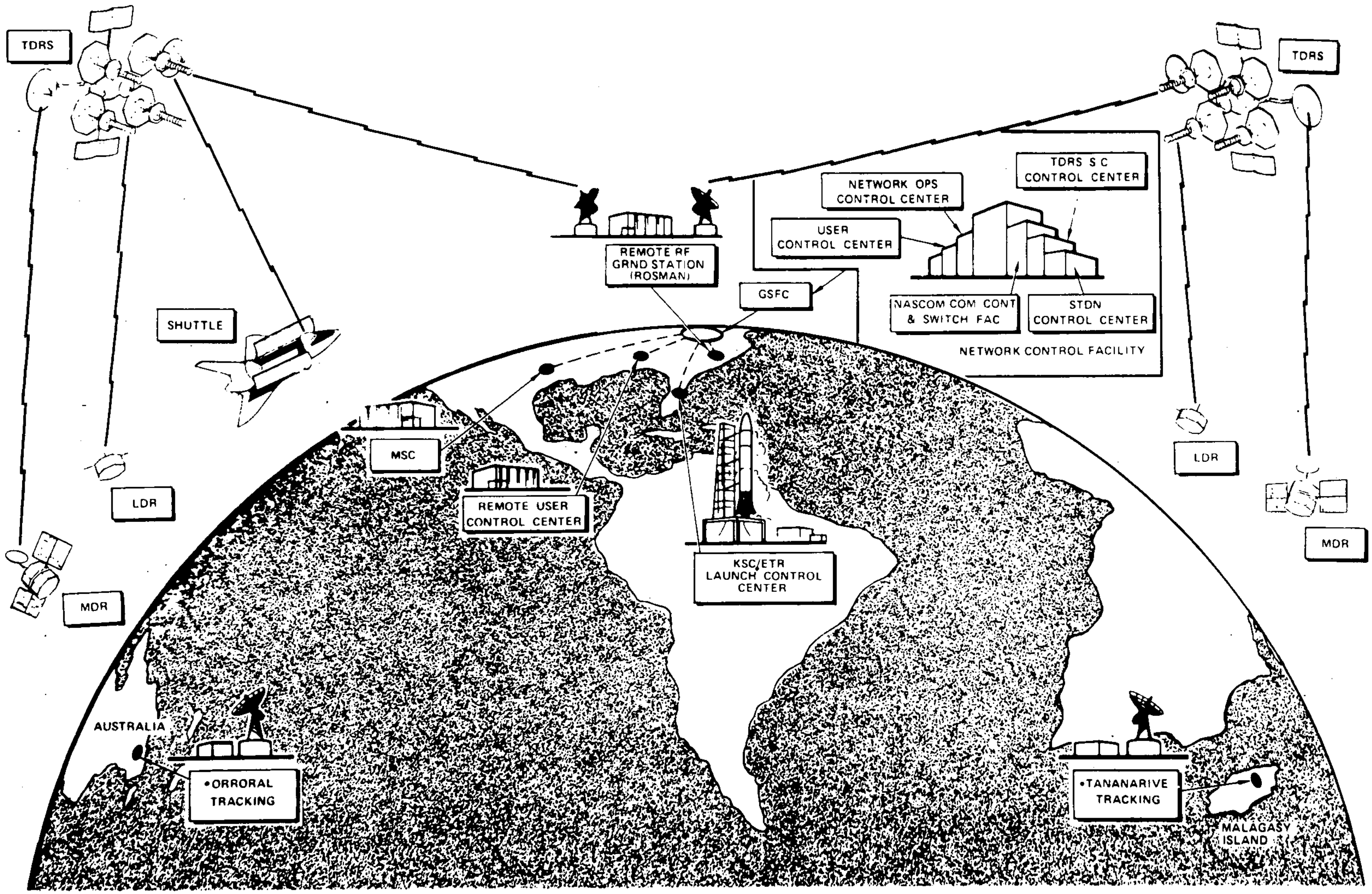


Figure 2-37. TDRSS Operational Concept

The Network Operations Control Facility at GSFC is a centralized operating organization with hard line communications links connecting the internal elements. It capitalizes upon the existing location at GSFC of many user control centers and of substantial facilities already in the GSFC complex suitable for TDRSS operation. GSFC elements include: (1) a network operations control center to interconnect and control the network and schedule its operation; (2) a TDRS spacecraft control center to control the TDRS; (3) user control centers to control individual user spacecraft; (4) a NASCOM communications control and switching facility to handle, switch, and direct the flow of communications to and from the various internal elements of the GSFC facility and to and from the remote elements; and (5) an STDN control center to direct and monitor the STDN support functions.

The large users at GSFC will have an operations control unit to manage their own service requirements, allowing them to proceed directly through a communications processor for command multiplexing, priority routing, data encoding, and high-speed modem operation to the ground station. Scheduling will be rapidly developed and provided to effect real-time implementation. Routine operations can be maintained by small users and others by using mail, TTY, and telephone facilities and the implementation by other operating centers.

Figure 2-38 illustrates a typical TDRSS network with ground station remotely located at Rosman, and a central network control facility at GSFC. The primary system elements and the operational and functional interfaces between them are shown. Automated and hard line interface links exist between all elements of the control ground network facility to facilitate fast response time and real-time operations. Ground communications are provided by the NASCOM network which furnishes all NASA mission control, technical control, and computation centers with access to remote tracking, data acquisition, and command stations. A highly automated ground facility is used, including computer control and processing as well as associated software, although the degree of centralization and automation of the data handling system has not been determined.

Ground link interface definition depends significantly on the diversity of the ground elements. The degree of ground station separation depends largely on the data handling requirements of broadband users, the need for data processing facilities, and the costs and real-time limitations of long line transfer and processing of data. Commands will be generated at the user control centers and forwarded through NASCOM to the TDRS ground station and appropriately formatted for transmission to the spacecraft. The primary elements involved in the operational process are shown in the figure with the flows between them and the principal functions of each element.

The NASCOM communications control and switching facility manages, controls, switches, codes and distributes all communications between elements in a format appropriate to the destination element. Service is provided by TTY, voice, high-speed data, and wideband data. Information may be carried through the transmission system in its original form at baseband, or signal converters may change it to a form more suitable for transmission. Modems (modulators/demodulators) may be required at each NASCOM terminal. Messages are routed by identifying addresses at the beginning of each message. The system is computer controlled, employing NASCOM digital computers at the switching centers for routing.

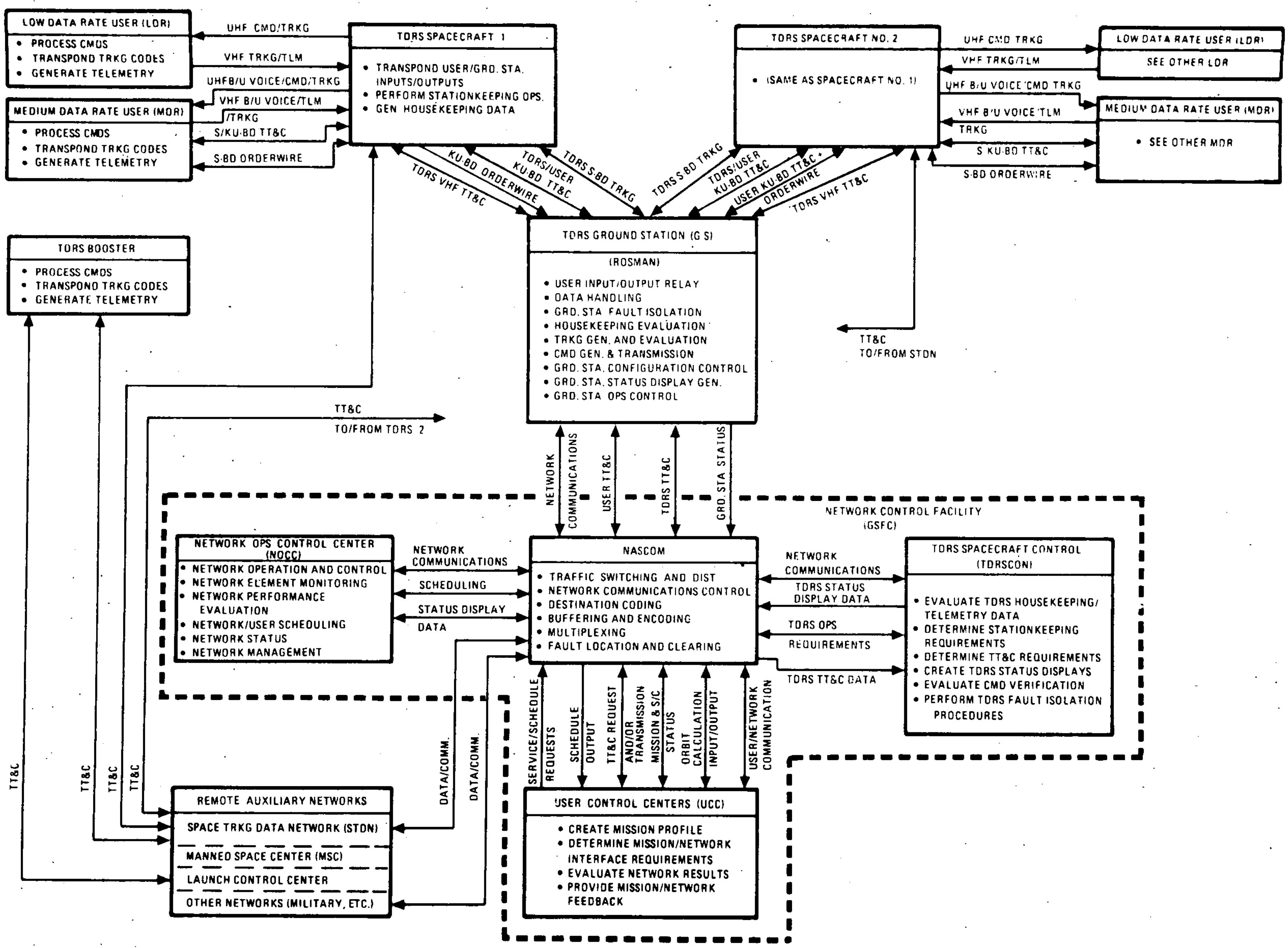


Figure 2-38. Primary System Elements and Their Operational and Functional Interfaces

Interfaces exist between all the network elements via NASCOM. The network operations control center (NOCC) manages and coordinates all network activities. Its interfaces via NASCOM are for network communications and network operations control, scheduling, and status display. The TDRS spacecraft control center (TDRSCON) manages and controls the TDRS spacecraft and relays communications service to user spacecraft. Its interfaces via NASCOM are for network communications, TDRS status and display, TDRS operations, and TDRS telemetry, tracking and command.

The user control center (UCC) manages and controls its user spacecraft and receives and evaluates its mission results. Its interfaces via NASCOM are for network communications, user spacecraft status and display, user spacecraft mission operations and control, and user spacecraft telemetry, tracking, and command. These require interface relationships for the flow of service and schedule requests, status information, tracking and orbit data, and real-time communications.

The TDRS ground station and its station control manage, control and operate an RF system to generate, transmit, receive, and handle RF commands and data to and from space and to and from the ground elements. It interfaces with the ground network elements through NASCOM for network communications, user and TDRS telemetry, tracking and command, and for station control. In turn, the TDRS ground station interfaces between the user and TDRS control centers and the user and TDRS spacecraft for the RF operations required for the transmission of commands and the reception of telemetry data. Its functions include: command and tracking generation and transmission; encoding, decoding, and high-speed modem operations; command data buffering and routing; command status and command verification multiplexing; data handling, processing, and conditioning; and RF antenna operations and control; station control and status display; station monitoring and checkout.

The interfaces between the user control center and the remote TDRS ground station were based on having user spacecraft outputs signal-conditioned at the ground station before transmission to GSFC. A message switching ground link interface configuration was used in which several data management functions as well as signal conditioning are performed at the ground station to assure time correlation and quality of all user data and to reduce data rate limitations, time delays, telemetry scheduling requirements, and equipment duplications. A computer at GSFC automatically distributes data messages according to destination codes.

The TDRS interfaces directly with the ground station and user spacecraft. It performs the relay operations to user spacecraft operations to the user via the ground station and NASCOM. The interfaces with the ground station provide two-way communications of TDRS S-band tracking, user Ku-band TT&C and orderwire, and TDRS VHF TT&C information. Interface links with the LDR user spacecraft are a UHF forward link and a VHF return link of tracking, telemetry, and command data. The interface links with the MDR spacecraft provide a two-way S-band or Ku-band link for transmission and receiving of TT&C information, a two-way S-band orderwire, and a backup UHF forward and VHF return link for voice, telemetry, tracking and command.

Additional interfaces exist between the remote ground elements and the space and GSFC-located elements for tracking, telemetry, and command. The interfaces between these remote elements and NASCOM are essentially operational and functional components of the overall system.

With the system concept and element interfaces as described, a detailed functional analysis of TDRSS functions and operations was conducted to apply the real-time system philosophy to implement the functional objectives. Detailed functional flow diagrams were developed to a third/fourth level of operations.

The functional procedures that permit real-time operation are, for example, (1) commands generated in the user and TDRS control centers; (2) commands transmitted directly to a communications processor (at GSFC) for command multiplexing and priority routing; (3) high-speed modem and decoding performed; (4) transmitted via NASCOM to the RF ground station (e.g., at Rosman); (5) high-speed modem and decoding at the ground station; and (6) ground station buffering and routing for transmission to the TDRS spacecraft. Similar operations are performed in reverse for return data, housekeeping, tracking, verifications, etc. The interfaces and functional relationships are compatible with these desired real-time functional procedures.

The procedures defined by the functional flow provide a detailed set of operational procedures for implementing a real-time TDRS system concept, while allowing routine procedures to be carried out where time is not a sensitive factor. They also exert minimum impact on existing or planned organizations and facilities, particularly in the early years of the system operation. Nevertheless, there is inherent flexibility in the system concept and operation to permit growth and modifications to allow for varying degrees of automation, centralization, and sophistication.

2.12 TDRS GROUND STATION

The TDRS ground station is the only link between the ground and the two TDRS satellites. The ground station RF equipment provides continuous and simultaneous communications with the two active satellites. One of the system requirements is a return link bandwidth of 600 MHz. This is easier to achieve at higher transmission frequencies so that a frequency of 14.6 to 15.2 GHz was chosen.

The communication link is mostly affected by water vapor attenuation, antenna size, receiver sensitivity, and transmitting power. The various effects of these parameters were traded off in selecting a baseline station design. Briefly, the conclusions reached are:

- 17.5 dB rain margin
- 18.3 m (60 ft) parabolic dish on the ground (one for each TDRS interface)
- Receiver is an uncooled parametric amplifier
- Three each, 25-watt Klystrons to transmit
- A minicomputer for each AGIPA processor

In addition, since both space-to-earth and earth-to-space links to the TDRS satellites operate at the same frequency, sufficient antenna isolation must be obtained to eliminate cross coupling between the two links. Techniques to provide the required isolation are described in Part I, Final Report (Section 12).

2.13 ACQUISITION AND HANDOVER OF USER SPACECRAFT

To pass control of the user spacecraft from one TDRS to the other or between the TDRS and an STDN site several operations are required both at the TDRS ground station and on board the user spacecraft. In the following sections a discussion of the various acquisition and handover procedures will be discussed.

2.13.1 Initial Acquisition by TDRS No. 1

As the user spacecraft comes into view of the TDRS it acquires the TDRS via the fixed-field of view ($\pm 13^{o}$) forward link antenna which is transmitting a PN code (one code common to all users). Typical code and doppler acquisition time for a user that requires 100 bps command support is on the order of 40 sec. The user tracks the code and doppler and activates the return link code generator and the transmitter. The return link code repeat rate is in sync with the forward link code (the code signaling rates, however, may be different). Each user spacecraft is equipped with a unique PN sequence for return link operation. The Ground Station supporting TDRS No. 1 acquires the return link PN code and the doppler and tracks same. Acquisition times can be decreased by parallel tracking of the PN code. Having acquired the downlink code the link is now fully synchronized to handle command/telemetry data.

The user spacecraft will track the forward link code for the entire transit time, while the return link transmitter can be commanded on/off at will. Therefore, after initial acquisition of the PN code command acquisition time is virtually instantaneous.

2.13.2 Handover From TDRS No. 1 to TDRS No. 2

As the user spacecraft being supported by TDRS No. 1 proceeds through its orbit a transition phase may be required to pass control of the user spacecraft to TDRS No. 2. To perform this operation a set of auxiliary demodulation/tracking units (ADTU) dedicated solely for handover are required at the Ground Station. As an example, the Ground Station supporting TDRS No 2 would activate an ADTU and load the PN code consistent with the downlink code of the user of interest. When the user reaches the point where it is in view of both TDRSs the ADTU of TDRS No, 2 acquires the downlink code and doppler and tracks them. The next step has several options depending on the desires of the user mission director, and consist of the following:

1. The user spacecraft can be commanded via TDRS No. 1 to switch its receive frequency to that of TDRS No. 2 and acquire its signal.

2. The user spacecraft can have dual receivers (one for each of the TDRS frequencies) each of which is followed by an address decoder. The outputs are then combined and applied to the command decoder. To hand

over the user spacecraft the user address is merely eliminated from the TDRS No. 1 link and applied to that of TDRS No. 2.

3. The third option (and probably the most costly) is to have two fully independent receivers (including command decoders) one each dedicated to a TDRS operating frequency.

Having established the connectivity between the user spacecraft and TDRS No. 2, the new link can now support command/telemetry transmission.

If the user spacecraft reaches a point where it will be out of view of both TDRSs for a brief period of time, the TDRS which must acquire the user spacecraft can command it (via the appropriate technique) to switch to the appropriate configuration for reacquisition.

2.13.3 Handover Between TDRS and STDN

To hand over a user spacecraft from one TDRS to the STDN (or vice versa), as it exists now the PN code generators on board the user would be disabled upon command and the user spacecraft transceiver would be configured to acquire the STDN signal formats. To hand over from STDN to a TDRS the user re-activates the two PN code generators and the spacecraft is configured to acquire TDRS signal formats.

2.13.4 Compatibility of the User Transceiver With TDRS and STDN

The user spacecraft transceiver required to be compatible (in frequency and signal formats) with both STDN and TDRS can be designed so that the reconfiguration from one type of support to the other is relatively simple. Estimates of a current design for an STDN transceiver put the size at approximately 100 in^3. To modify this transceiver so that it will support both the STDN and TDRS formats would increase the unit size to approximately 200 in^3.

3.0 PART II/PHASE I SUMMARY (DELTA 2914/LDR, MDR, AND HDR)

As described in the introduction to this volume, Part II of the study had two phases. The first was to analyze the telecommunications service requirements of high data rate user spacecraft and determine the design and operational impacts on the Part I baseline configuration. The second phase was to synthesize for Atlas-Centaur and Shuttle launches.

This section summarizes the results of Part II/Phase I (a Delta 2914 launched TDRS that supports LDR's, MDR's, and HDR's). A more detailed description of the Part II/Phase I activities and results are provided in Volume II, Telecommunications Design, and Volume III, Spacecraft Design.

It was determined that it was not only feasible but practical to provide simultaneous support to LDR's, MDR's, and HDR's, and that the operational constraints are minimal.

The basic support requirements were changed to eliminate LDR voice, to include S-band voice for Shuttle, to incorporate support of HDR users, and to provide a fixed field-of-view approach for the LDR forward link. To stay within the power limits of the spacecraft it is necessary to reduce the number of simultaneous LDR forward links from two to one as was originally specified.

These changes required modifications to the telecommunications subsystem including the antennas, the electrical power requirements, and the spacecraft weights. The satellite configuration was previously illustrated in Figure 1-3. The basic spacecraft structure and its subsystems are the same as in the Part I baseline. Also, the general RF interfaces and frequencies are the same as for the Part I baseline indicated in Figure 1-2. The only interface differences are that either one of the MDRU's can be an HDRU, resulting in the following modes of service:

Forward links

1 LDR and 2 MDR's

or

1 LDR; 1 MDR, and 1 HDR

Return links

20 LDR's and 2 MDR's

or

20 LDR's, 1 MDR, and 1 HDR

where the Space Shuttle is considered to be an MDR only because it operates at S-band.

3.1 TELECOMMUNICATIONS DESIGN

The most dramatic changes to the telecommunications system were the inclusion of a requirement to support a new category of spacecraft users identified as the high data rate user and the use of a fixed field-of-view antenna for the LDR forward link. As the name HDR implies, a much higher return data rate (H = 100 Mbps) is required than its Part I predecessors (the low data rate (1 to 10 kbps) and medium data rate (10 kbps to 1 Mbps) users). This additional requirement plus the LDR forward link change necessitated changes to the basic design established for the Part I baseline. These changes resulted in considerably improved link performance, service, and support capabilities.

The uprated Delta configuration is functionally similar to the Part I baseline design, but employs larger MDR/HDR antennas (3.8 meters) and a larger TDRS/GS antenna (1.8 meters); considerably increasing the support capabilities to the MDR and HDR users. In addition, since the forward voice requirement was eliminated for the LDR space-to-space link, this prime power can be used to enhance the MDR and HDR user support capabilities.

Figure 3-1 is a simplified block diagram of the telecom subsystem indicating the major elements and their groupings according to function.

In the LDR forward link, the prime mode is F-FOV which simultaneously illuminates the entire 31-degree FOV with a fixed beam. In this mode, a single transmit channel excites a high gain disc-on-rod antenna to provide an EIRP of +30, +27, or +24 dBw. The reduced EIRP's are used to conserve the prime power demands during eclipse and other periods when prime power must be reduced.

This F-FOV beam is used to sequentially transmit commands to the 20 LDR users and also to provide a continuous signal to all LDR users within the FOV to lock-up and synchronoize their on-board PN code generator and frequency source. This lock-up is required to enable range and range rate measurements to be made to each LDR user. Four identical transmit channels provide quadruple functional redundancy in the F-FOV mode or the four channels can be used as a phased array to provide an emergency steered beam mode with EIRP of +36, +39, or +42 dBw. This higher EIRP can be effectively used to aid command transmissions to a user in emergency condition (e.g., tumbling state) or when the user is in a high RFI environment.

In the LDR return link the adaptive signal processing design technique (adaptive ground implemented phased array) developed during Part I is unchanged. It employs adaptive spatial and polarization discrimination to reject interference signals, including unintentional RFI emitters as well as intentional in-band signals from other users. As the name AGIPA implies, the usual complexities associated with a multiple access phase array are located on the ground; minimizing the size, weight, and prime power demands as well as the system complexities on the TDRS. RFI model synthesis conducted in Part I showed the AGIPA concept provides upwards of 5 to 18 dB improvement over a F-FOV design, in the presence of interference environment. The LDR return link can also operate in the F-FOV backup mode by employing two orthogonally

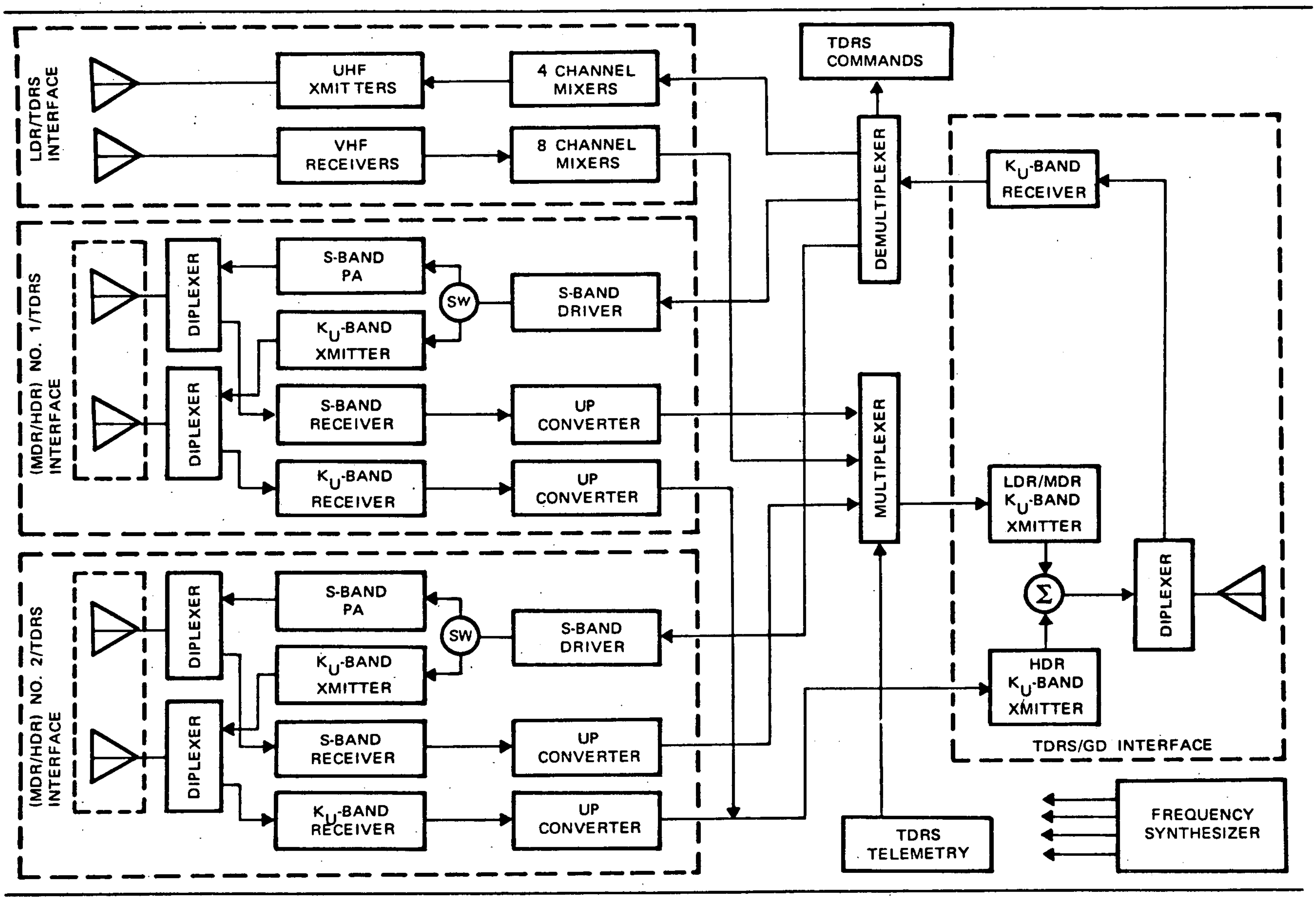


Figure 3-1. Telecommunication Block Diagram, Uprated Baseline

polarized receiver channels, or by using the 8 receive channels in an unadapted mode; viz. with the beam pointed along the spacecraft local vertical. In this backup mode however, the link performance is limited, since all RFI and in-band intentional interference signals are received without discrimination. The performance of the LDR forward and return links are shown as a function of RFI level in Figures 2-16 and 2-17. Note that in the new design the fixed field-of-view mode, the normal EIRP is 30 dBw.

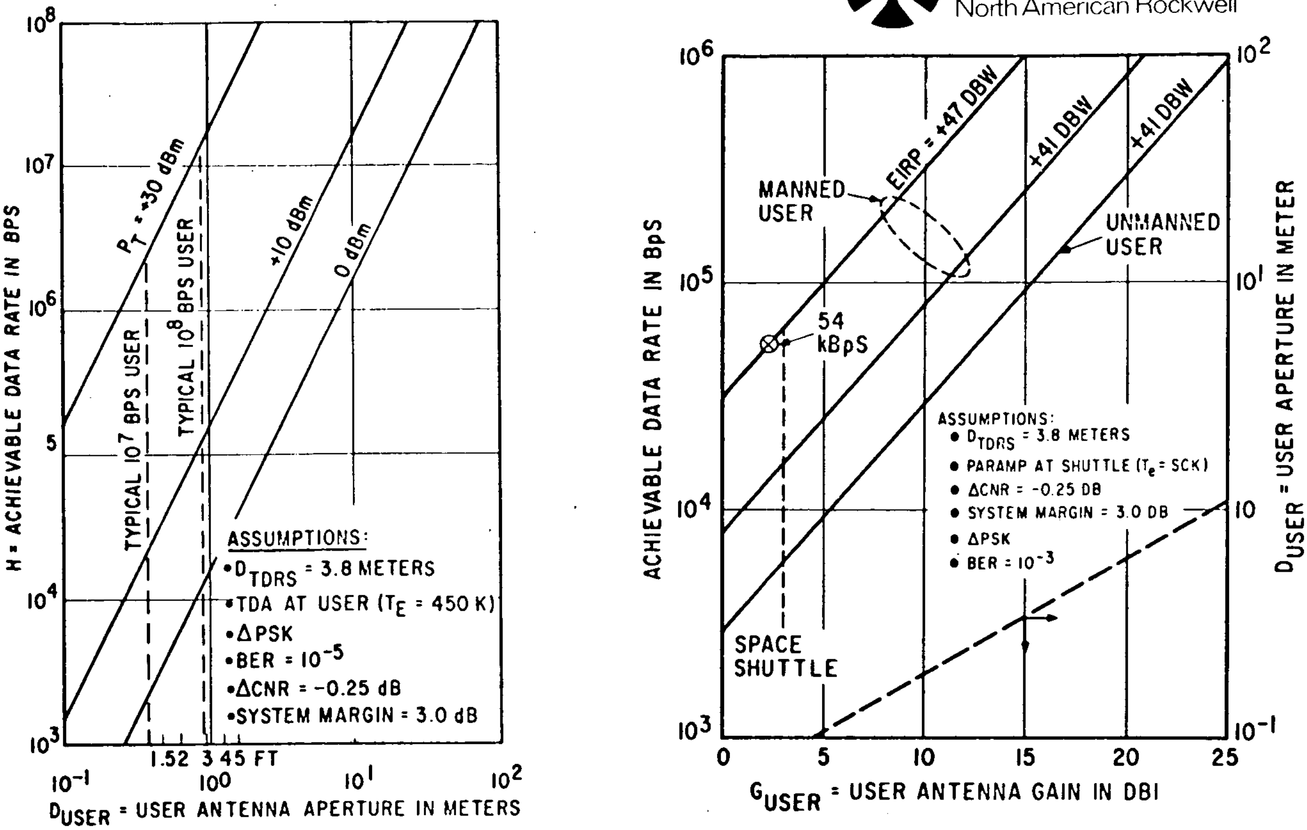
The MDR and HDR functions are combined into one dual frequency transponder as in Part I. There are two MDR/HDR transponders each employing a 3.8-meter deployable reflector antenna. The performance of the MDR and HDR links are presented in Figures 3-2 and 3-3 as a function of user spacecraft antennna gain.

Each MDR/HDR transponder also provides a functional backup redundancy to the TDRS/GS transponder. In addition, one of the 3.8-meter antennas can be used in lieu of the 1.8-meter TDRS/GS antenna, in the event that this antenna should fail or, to provide an additional 6.5 dB rain margin for the space-to-ground link to provide a total of 24 dB system margin for operation in rain.

The MDR/HDR antenna provides an HPBW of approximately 0.4 degree at Ku-band and 2.6 degrees at S-band; consequently a closed loop pseudo-monopulse system is employed for acquisition and track of the HDR user spacecraft; and open loop tracking via ground commands is used for acquisition and track of MDR users. However, if it is determined that open loop tracking of MDR's imposes too great a workload on ground station operations, closed loop tracking can be implemented for the S-band links also.

The TDRS/GS transponder uses a 1.8-meter fixed parabolic reflector antenna with a HPBW of approximately 0.8 degree. This antenna is designed with a closed loop auto-track system with a 4-hour pseudo-monopulse Ku-band feed. The TDRS/GS transmitter is an FDM/FM/FDM channel to transmit the remaining 8 LDR + 2 MDR + TDRS tracking + order wire + telemetry data. Both channels were designed with a 17.5 dB margin for operation in rain; however, the output power from the HDR channel can be reduced by 10 dB to conserve prime power during normal operation in clear weather, and can also be turned off to further conserve prime power demands.

The HDR channel employs dual mode TWT amplifiers in its final power output stage, providing RF power outputs of approximately 10 and 1 watts to meet the 17.5 dB and 7.5 dB system margin, respectively. Since the established reliability failure rates for TWT amplifiers are quite high compared to solid state, each TWT amplifier has two parallel TWT's hard wired to a common high voltage power supply system (the appropriate heater will be energized to excite the desired TWT). In addition, the HDR channel has full block redundancy (two TWT amplifiers) such that quadruple redundancy exist for the TWT which constitutes the major failure component in the amplifier. The FDM/FM (MDR and LDR) channel is a solid state Ku-band amplifier operating in the saturated mode. Solid state amplifiers were retained for this portion of the TDRS to ground link because of their high reliability and the MDR and LDR channels require relatively low power, thus making acceptable the lower DC to RF efficiency of these devices.



a) HDR b) MDR

Figure 3-2 Forward Link:Data Rate Versus D_{user}

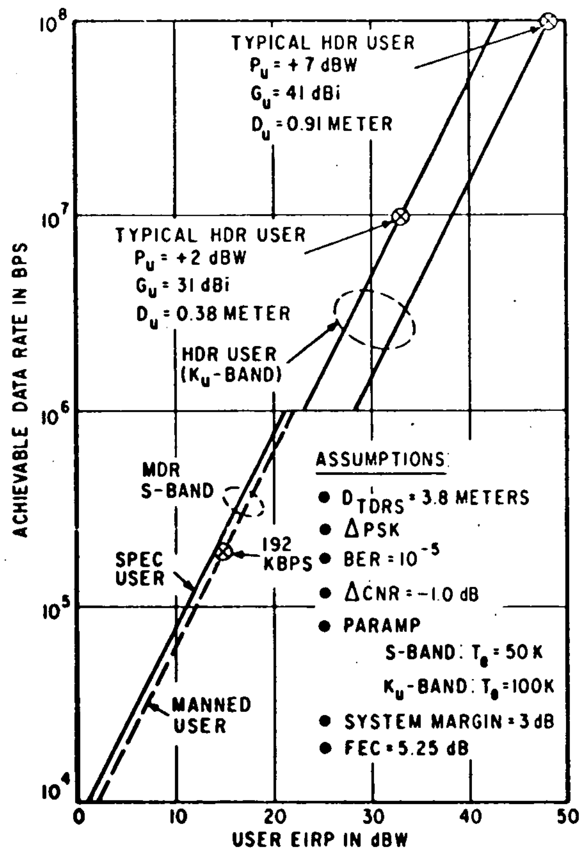


Figure 3-3 MDR/HDR Return Link:EIRP Versus Data Rate

The other elements of the uprated baseline telecommunications subsystem are: TDRS tracking and order wire transponder, TT&C transponder, and frequency source which are identical to the Part I baseline design.

3.2 SPACECRAFT DESIGN

The basic spacecraft design including all subsystems remained unchanged from the Part I baseline configuration. However, the various appendages and their deployment mechanisms were redesigned to permit the use of larger parabolic reflector antennas for both the space-to-space microwave links and for the TDRS-to-ground link. This was necessary to support high data rate users in addition to the medium and low data rate users which were serviced by the Part I baseline configuration. With the larger antennas, the solar shadow lines from the reflectors 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 F-FOV from steered beam which necessitated the electronic and electrical power changes.

The uprated baseline satellite configuration is shown in its deployed mode in Figure 1-3. Figure 3-4 is the engineering layout illustrating the satellite in its stowed and deployed modes, and mechanics of deployment.

The overall changes from the Part I baseline to the uprated configuration are indicated in Figure 3-5.

To incorporate the support requirements of the HDR users, the size of the parabolic reflector antennas was increased from 1.98 meters to 3.8 meters diameter.

The change in antenna diameter had the greatest impact on the spacecraft configuration. The 1.98-meter diameter 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 necessitated using furlable or folding antenna reflectors. The antenna selected is a furlable rib-mesh design developed by the Radiation Systems Division of Radiation, Inc.

For packaging in the Delta 2914 shroud 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 packaged configuration, and their support struts are swung forward around and in front of the LDR elements.

The TDRS/GS fixed surface mesh antenna, which was moved from the satellite body to a position on the solar array strut to allow for the increased diameter, 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.

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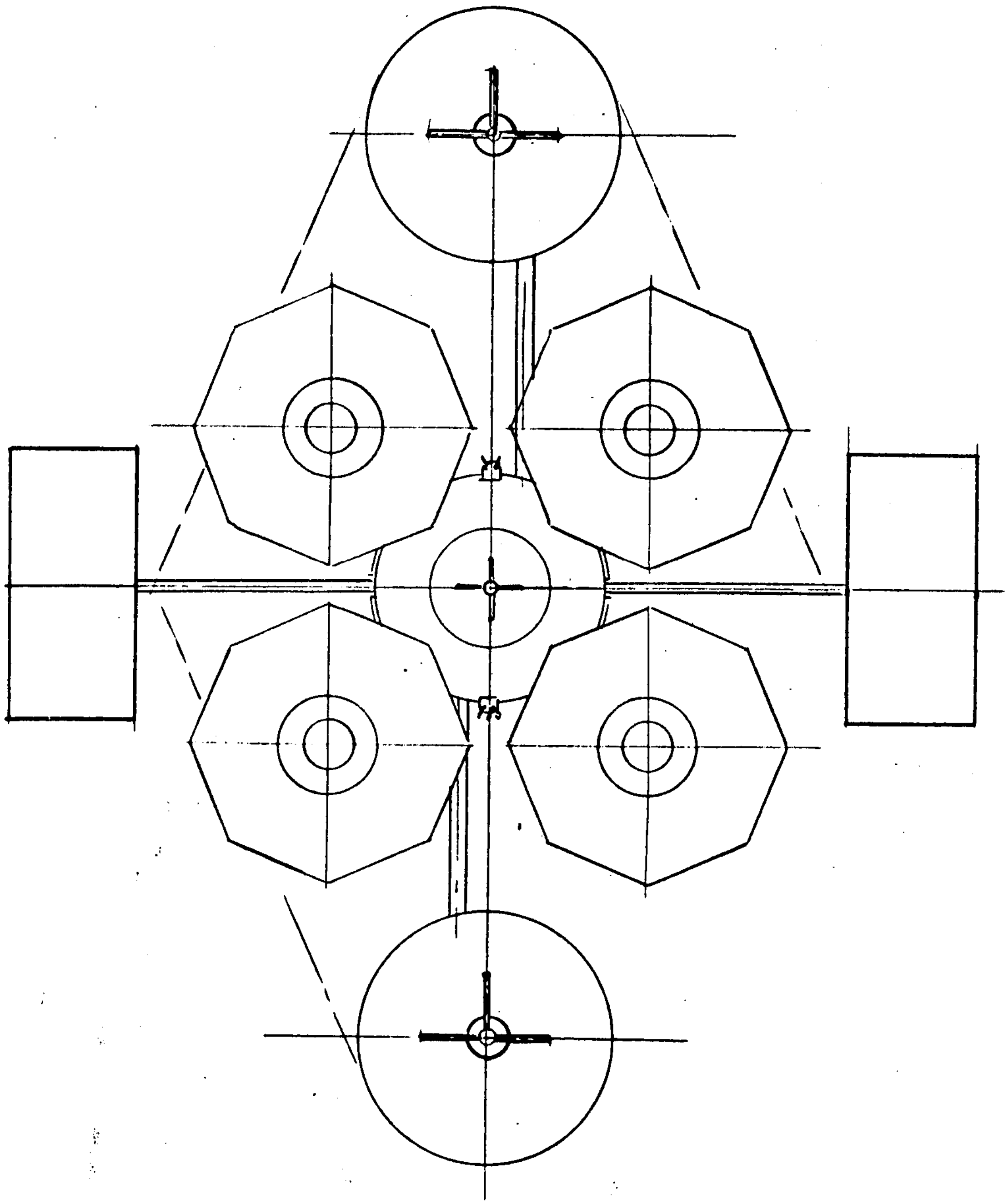

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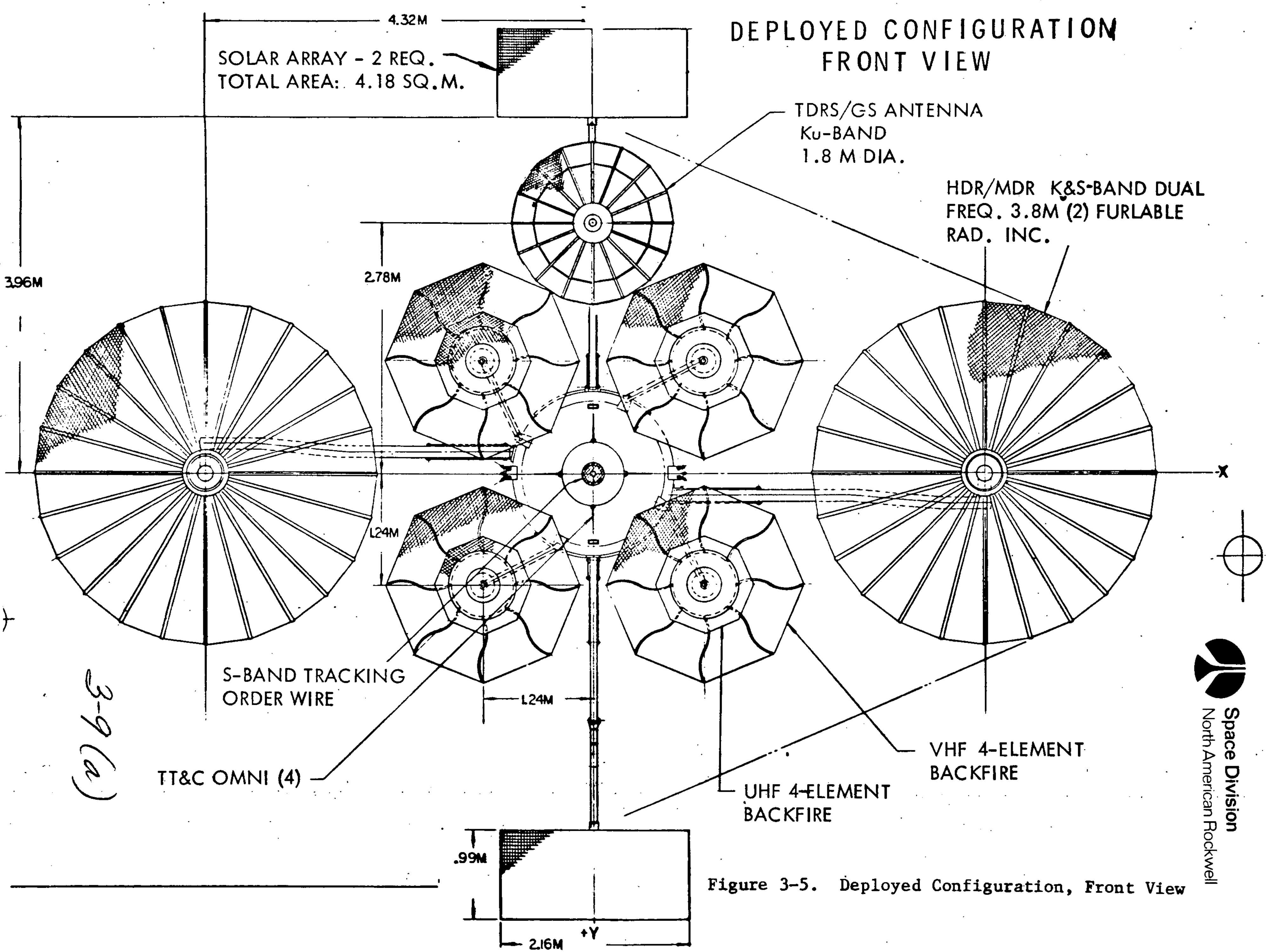

Figure 3-5. Deployed Configuration, Front View

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.

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 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.

The LDR antenna elements are deployed in the same manner as the Part I baseline.

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 rotate the panels to acquire the sun and 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 reduce the rate of closure while still assuring positive lock at all joints. Proper choice of lubrication and bearing design assures 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.

Incorporating the HDR capability resulted in an increase in the 0.91 meter TDRS/GS antenna to 1.8-meter diameter and the TDRS/GS transmitter changed from solid state to a redundant two-channel TWT design. The larger 1.8-meter diameter antenna could not be mounted in the same position as the smaller Part I antenna on the front of the spacecraft since in this location it interfered with the 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 antennas 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.

It was found during detailed operations analysis of the combined telecommunications and electrical power subsystems, that essentially all constraints on telecommunications service operations could be eliminated by increasing the solar array area by 0.18 m^2. To minimize the increase in weight the substrate design was changed to a light-weight construction.

Table 3-1 is a summary of the weight of the uprated baseline TDRS by subsystem. As can be seen, this high capacity version of a Delta 2914 launched TDRS has a weight contingency of approximately 10 percent the dry weight of the spacecraft.

The electrical power loads (total satellite requirement including telecommunications subsystem) for each mode of operation are listed in Table 3-2. These loads are plotted on Figure 3-6 which shows power available as a function of lifetime. There is sufficient power at end of life for all reasonable modes of service with no operational constraints.

One feasible mode that could be desired is a case where 17.5 dB rain margin is required for the HDR TDRS/GD link, the LDR forward link is on high power (30 dBw), S-band voice is required on one microwave antenna forward link, and Ku video is required on the other microwave antenna. 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 for 12 hours with battery augmentation, at which time certain operations such as Ku video transmission and voice must be reduced to permit battery recharging. This assumes continuous heavy rain at the ground station and continuous HDR transmission for this period.

A detailed analysis of daylight and eclipse electrical power operations is provided in Volume III, Spacecraft Design, of this report.

Table 3-1. Uprated TDRS Baseline 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		
Initial on orbit	788.0	357.5		
Burned-out insulation	8.0	3.6		
Apogee motor propellant	688.0*	312.1	No Change	
Synchronous orbit injection	1484.0	673.2		
Transfer orbit propellant	6.0	2.7		
Delta sep. weight (27° transfer orbit)	1490.0	675.9		

Table 3-2. 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
Power Margin** 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

* 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, high power LDR) must be reduced until batteries are fully charged.

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

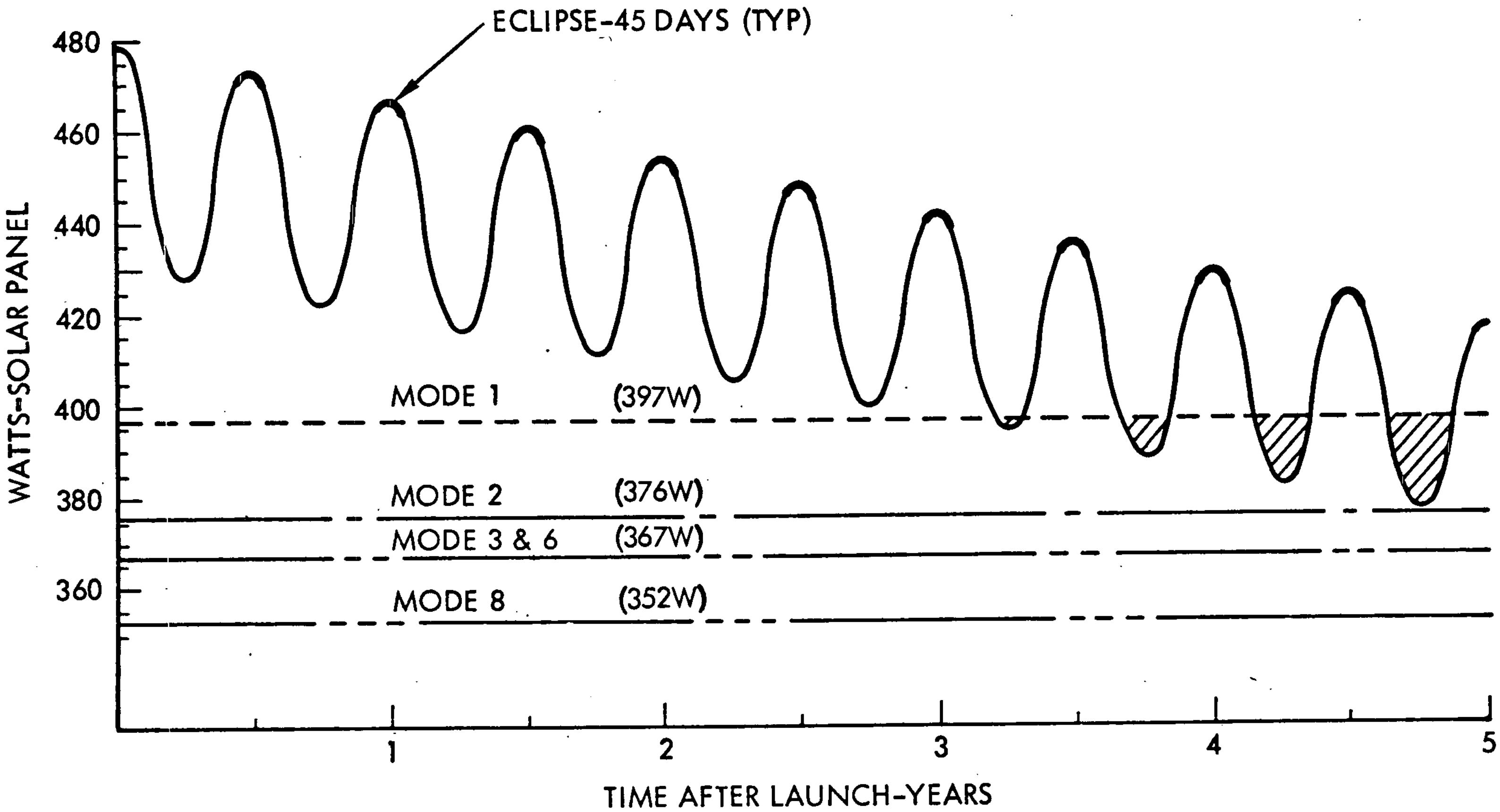

Figure 3-6. Electrical Power Versus Life Time Curve

3.3 RELIABILITY

The reliability of the uprated baseline TDRS configuration varies from the predicted values for the Part I baseline only by the design changes of the increased capacity telecommunications subsystem. The reader is referred to the Part I final report for a detailed review of this analysis and to Volume III, Spacecraft Design, of this report for analysis of the design changes.

In predicting the probability of mission success the following definitions were used:

- 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 five years; reduced forward link capability is permitted during eclipse

Based on a predicted reliability of 0.898 for the uprated communication subsystem, the satellite reliability is calculated as 0.800. This compares with the predicted reliability for the Part I baseline of 0.805.

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

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 of 4 (present definition of success)	= 0.99992
2 of 4 required	= 0.99586

3 of 4 required = 0.92907

4 of 4 required = 0.50407

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 of the LDR receivers. This would result in an increase in weight of 2 Kg. 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 0.92907 to 0.99787 and from 0.50407 to 0.92577, respectively.

3.4 ALTERNATIVE CONCEPTS

The uprated TDRS baseline configuration described in Sections 3.1 through 3.3 is considered to be the optimum satellite concept for the telecommunications objectives and requirements agreed to for this study. However, review of the most recent NASA mission models for earth orbital spacecraft indicate a trend toward decreasing numbers of small data rate users and increasing numbers of medium data rate users at S-band. Although the uprated TDRS baseline provides two S-band interfaces, the design was optimized for semi-random multiple access of low data rate users in the VHF/UHF frequency spectrum.

In light of this anlaysis and in an attempt to establish alternative designs with less complexity, two variations to the uprated baseline TDRS were considered late in the study.

3.4.1 Alternative 1 (No LDR Multiple Access)

The first alternative is a very simple design approach that provides relay service to one MDR plus one HDR or two MDR's simultaneously. This concept is illustrated in Figure 1-4, and an engineering layout drawing is provided in Figure 3-7. The design philosophy was simply to eliminate the LDR portion of the telecommunications subsystem and associated UHF/VHF antennas. Thus the assumption is made that any LDR's will be supported sequentially with the MDR's and HDR's through one of the two microwave transponders.

This design approach resulted in a reduction in weight of approximately 21 kg (46 lb), and in power requirements of approximately 125 watts, and corresponding increases in contingency.

3.4.2 Alternative 2 (LDR Multiple Access at S-Band)

The second alternative to the uprated baseline, previously identified as one of the two optimum approaches, adopted as a design philosophy the necessity to provide some form of semi-random multiple access capability. To implement this philosophy, in context with the design objective of mechanical simplicity and the mission analysis conclusion that most earth orbital user spacecraft may be S-band users, the UHF/VHF antennas and transceiver were replaced with a multichannel S-band phased array. This concept provides simultaneous relay service to 20 (S-band) LDR/MDR users via

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and 2 MDR, or 1 HDR and IMDR via the high gain S/Ku-band antenna and transceiver.

The concept is illustrated in Figure 1-5 and the engineering layout drawing is provided in Figure 3-8.

To maintain an adequate wieght margin in alternative 2 it was necessary to reduce the satellite repositioning capacity to one 65 degree trip in 20 days.

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. 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 requires 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.

A summary weight breakdown comparing each design concept is provided in Table 3-3. Not included in the weight contingency of alternative 1 to the uprated baseline is the possible wieght reduction due to the lower power requirements of the telecommunications subsystem. The major attributes of the alternatives are:

- Simpler mechanical design
- Larger weight contingency of alternative 1
- Larger power contingency of alternative 1
- Avoids the potential RFI problem in the VHF spectrum
- Alternative two is a more suitable design for multiple access of MDR's

Although the alternatives avoid the terrestrially generated RFI problem at VHF, alternative two must contend with in-band interference from the simultaneously accessing S-band users. To combat this problem it may be necessary to use the same AGIPA multichannel design approach synthesized to combat RFI for the LDR's in the baseline design. This was included in the weight estimate for the multichannel LDR/MDR transponder.

There are no daylight operation limitations imposed by this concept for any support demands that can reasonably be anticipated.

3.5 USER TRANSPONDER AND GROUND STATION DESIGN

The LDR and MDR user transponder designs developed during part I were covered in section 2.10. There were a minimum number of design changes in the LDR and MDR User Terminal during part II. However, the HDR terminal is a new desigh.

The LDR User transponder characteristics for the Delta launched uprated TDRS configuration are as follows:

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Table 3-3. TDRS Weight Summary

	Part I Baseline		Uprated TDRS		Uprated TDRS without LDR UHF-VHF		Uprated TDRS & LDR S-Band Array		Max Capability TDRS	
	Weight		Weight		Weight		Weight		Weight	
	lb	kg	lb	kg	lb	kg	lb	kg	lb	kg
Communications										
Electronics	122.2	55.5	141.9	64.4	98.7	44.8	187.39	85.0	187.39	85.0
Antennas	117.9	53.5	130.3	59.1	127.4	57.8	99.23	45.01	323.1	146.6
Attitude stabilization and control	57.7	26.2	57.7	26.2	57.7	26.2	57.7	26.2	51.1	23.21
Electric power	97.0	44.0	93.0	42.2	93.0	42.2	93.0	42.2	128.0	58.1
Solar array	58.6	26.6	61.1	27.7	61.1	27.7	57.54	26.1	67.1	30.4
Structure	91.0	41.3	91.0	41.2	91.0	41.2	83.88	38.05	101.0	45.8
Thermal control	23.9	10.8	23.9	10.8	23.9	10.8	19.25	8.73	23.9	10.8
Auxiliary propulsion hardware	38.4	17.4	32.0	14.5	32.0	14.5	32.0	14.5	32.0	14.5
	606.7	275.0	630.9	286.1	584.8	265.2	630.46	385.97	913.6	414.4
Propellant + N_2 (2-65°--15-day staion changes)	49.3	22.4	49.3	22.4	49.3	22.4	39.3**	17.83**	56.6	25.7
Total spacecraft	656.0	297.6	680.2	308.5	634.1	287.6	669.76	303.8	970.2	440.1
DELTA 2914 VEHICLE										
Total spacecraft	656.0	297.6	680.2	308.5	634.1	287.6	669.76	303.8	970.2	440.1
Contingency	82.0	37.2	57.8	26.3	103.9	47.2	68.24	30.95	Negative	Negative
Allowable PL (Delta 2914 + CTS apogee motor)	738.0*	334.8*	738.0	334.8	738.0	334.8	738.0	334.8	738.0	334.8
Empty apogee motor case	50.0	22.7								
Initial on orbit	788.0	357.5								
Burned-out insulation	8.0	3.6								
Apogee motor propellant	688.0*	312.1	No Change		No Change		No Change		Not Applicable	
Synchronous orbit injection	1484.0	673.2								
Transfer orbit propellant	6.0	2.7								
Delta sep. weight (27° transfer orbit)	1490.0	675.9								
ATLAS/CENTAUR VEHICLE										
Total spacecraft weight			680.2	308.5					970.2	440.1
Contingency			1339.8	607.7					1049.8	476.2
Allow. PL (Atlas/Centaur + TE364-4 apogee motor)			2020.0	916.2					2020.0	916.3
SHUTTLE/AGENA-TUG										
Total spacecraft weight			680.2	308.5			669.76	303.8	970.2	440.1
Contingency			392.8	178.3			403.24	182.9	102.8	46.7
Triple launch--allowable payload (Shuttle/Agena-Tug) each			1073.0	486.8			1073.0	486.8	1073.0	486.8

* 5 deg/day drift orbit

** 1-65° - 20 day station change

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(1) A 668 Kchip/sec PN chip rate in the forward link is to serve a twofold purpose, namely: (1) to provide sufficient spectrum spreading to meet the IRAC flux density requirements, and (2) to maximize the forward link processing gain (=PN chip Rate/Data Rate) by utilizing entire bandwidth allocation.

(2) A short PN code is used to provide a maximum code acquisition time of approximately 40 seconds.

(3) A 1 Mchip/sec code rate in the return link is necessary to permit the multiple accessing of 20 users with sufficient process gain to meet the required level of performance.

(4) Unambiguous ranging is achieved by transmitting a coded word over the Command/Telemetry link whose duration is equivalent to the two-way range uncertainty of 40,000 kilometers (approx. 133 Msec).

MDR User Terminal characteristics have changed only slightly from the baseline configuration, namely:

(1) A 5 Mchip/sec and 20 Mchip/sec PN chip rate for the MDR and Shuttle, respectively, in the forward link is necessary to distribute the sign energy to conform to the IRAC requirements.

(2) The short code used during the code acquisition phase limits the maximum acquisition time to 40 seconds or less.

(3) A coded word is switched in after code acquisition to provide unambiguous ranging over a two-way range uncertainty of 40,000 Km.

(4) The PN modulation in the return link is a 5 Mchip/sec PN code which employed in both the spec. MDR user and the space shuttle.

The HDR user transponder design is functionally similar to the spec MDR user transponder design. The major change is in the doppler processor design. Doppler rates at Ku-band are approximately seven times that at S-band. The doppler processor therefore must be designed to search over a greater range of frequency uncertainty. The transponder is shown functionally in Figure 3-9.

The PN rate in the HDR forward link consists of a 60 Kchip/sec code of length 2047 in the normal mode. To meet the IRAC in the high power mode (i.e., 53.6 dBw to support forward link video) a chip rate of 60 Mchips/sec is required.

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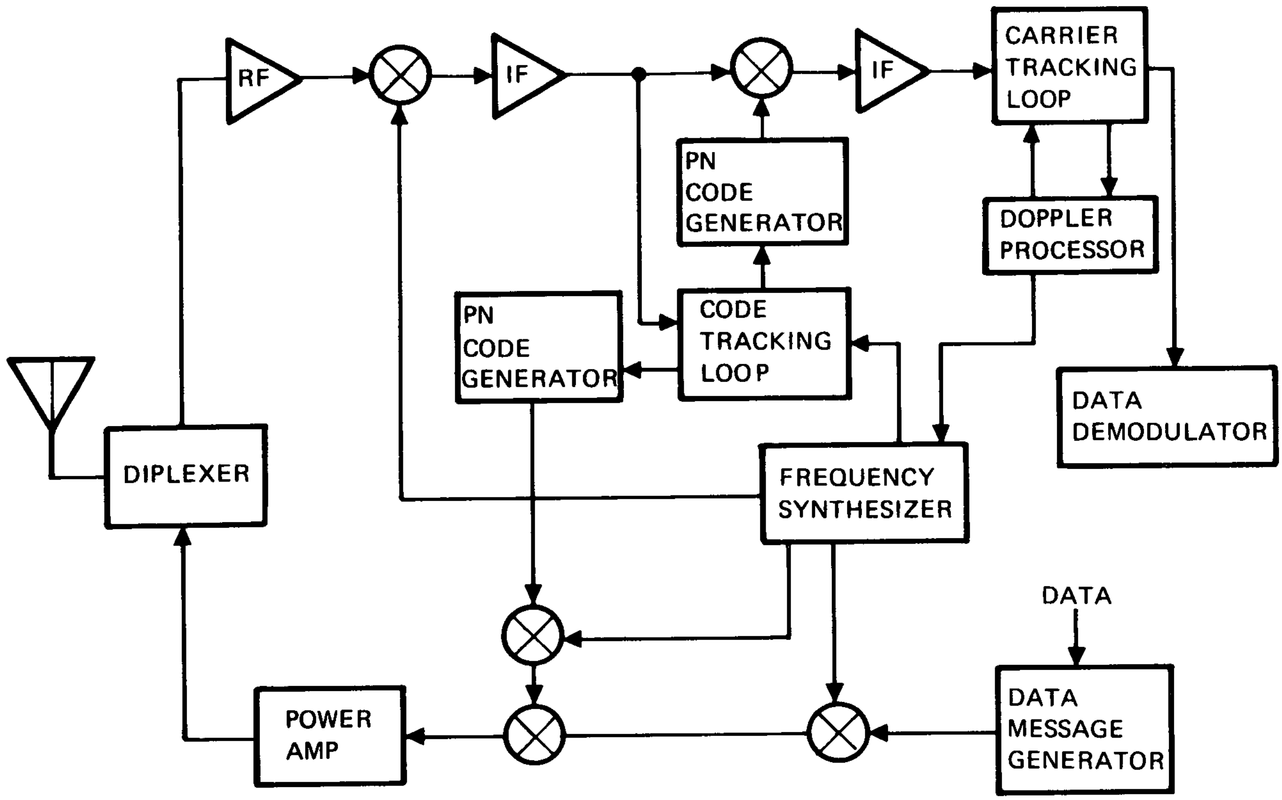

Figure 3-9. HDRU Transponder

Range ambiguity resolution is achieved by transmitting a unique data word of 133 msec duration. This is equivalent in length to approximately 8,000 PN chips on the forward link and 665,000 chips on the return link.

In addition to normal support of greater than IMBps, the return link may be required to support a 100 MBps HDR user. Performance of this link can be enhanced by approximately 6 dB with application of forward error control.

The TDRS ground station design for the Delta launched up-rated TDRS configuration differs from the baseline in the method of demodulation of the TDRS/GS signal, as follows:

The RDRS/GS link for the up-rated Delta configuration employs FDM/FM for all the serious except the the HDR return link which is in a separate channel.

Correspondingly the Ground Station frequency selectively separates the HDR channel and all others are demodulated by the Phase Lock Loop FM Discriminator.

The other essential differences between the two ground stations is in the number of MDR users the system can support simultaneously (four as opposed to two previously).

4.0 PART II/PHASE 2 SUMMARY (ATLAS-CENTAUR & SHUTTLE-AGENA CONCEPTS)

This section summarizes Phase 2 of Part II which was concerned with the synthesis of TDRS concepts launched with an Atlas-Centaur or a Shuttle-Agena.

A high-capacity TDRS design was developed to a conceptual design level of detail that is compatible with the launch characteristics of both the Atlas-Centaur and the Shuttle-Agena. In addition, since mission analyses indicated this level of relay support may not be required in the foreseeable future, an alternate approach was developed to launch the uprated baseline TDRS configuration, three-at-a-time, on the Shuttle-Agena.

4.1 ATLAS-CENTAUR TDRS DESIGN CONCEPT

The launch analysis of the Atlas-Centaur established the payload capability to 2.5^{o} inclined synchronous orbit at 915 kg (2020 lb). This assumed a TE364-4 apogee motor for synchronous orbit insertion.

4.1.1 Maximum Capacity Telecommunications Design

The high performance Atlas-Centaur TDRS telecommunication system is no longer constrained by the payload capacity as in the Delta 2914 Configuration and provides increased performance and service. This configuration as compared with the Delta 2914 Uprated TDRS Configuration is enhanced by the use of:

- Sr. AGIPA in the LDR VHF return link to improve its performance in high interference signal environment.

- 4 - MDR/HDR dual frequency Transponders to better service and support postulated MDR and HDR user populations.

In the LDR return link, Sr. AGIPA uses a 5 element (short backfire design) ring array with a diameter of 5λ to provide an effective half power beamwidth (HPBW) of approximately 8 degrees as compared to 24 degrees for AGIPA. RFI model analysis conducted in Part 1 showed Sr. AGIPA can more effectively provide interference discrimination, with improvements in signal-to-interference ratio (Δ SIR) of 9 to 15 dB as compared to a F-FOV approach. This Δ SIR provides considerable improvement to meet the required data rates even in the presence of extremely high RFI environment.

Since AGIPA (both Jr. and Sr.) is an adaptive signal processing system that continuously computes and adjusts the amplitude and phase weighting factors (at the ground station), the system is immune to the normal structural flexure and antenna pointing variations experienced with conventional phased array. Consequently, considerable flexibility can be exercised in the transponder design and spacecraft implementation of the AGIPA systems.

The LDR return link can also be used in the F-FOV backup mode by using any two orthogonally polarized antenna channels.

In the LDR forward link the operation and design is identical to the uprated TDRS Baseline, developed during Part II/Phase 1, except there is an extra element and transponder which increases reliability during normal operation and increases the gain by 1 db in the emergency steered beam mode.

The addition of two MDR/HDR channels provides multiple access to 4 MDR users and to 1 HDR user simultaneously. Since all 4 MDR/HDR Transponders are identical, quadruple functional redundancy exists for the support of MDR as well as HDR user. In addition the MDR/HDR channels are each designed to backup the TDRS/GS Transponder, providing triple redundance to the TDRS/GS Transponder.

The TDRS/GS Transponder is an all FDM system in both return and forward direction. In the return link, two channels are used; viz.an HDR channel and an FDM channel (which combines the 10 LDR + 4 MDR + TDRS Tracking + Order Wire + Telemetry Data - this channel replaces the FDM/FM channel used in the Uprated TDRS. The HDR channel is 150 MHz wide to support the 100 Mbps return data and uses a solid state Ku-band amplifier operating in a saturated mode. A 6 dB larger antenna is used in this link as compared to the Delta Uprated Configuration. Therefore a TWT amplifier is no longer required, which reduces the size and weight and increases the overall system reliability. The FDM channel occupies a spectrum of 88 MHz and uses a solid state Ku-band amplifier operating in the linear mode with approximately a 10 dB backoff from saturation. Both channels are designed to operate at all times with a system margin of 17.5 dB.

The TDRS/GS forward link data is an FDM signal, occupying a spectrum of approximately 390 MHz. Each MDR/HDR channel is designed broadband (75 MHz wide) such that no tuning is required at the TDRS relay spacecraft. - all tuning (frequency selection) is determined at the ground station, as in the Part I and Delta Uprated Configuration.

The TDRS/GS 3.8 meter Ku-band antenna provides a HPBW of approximately 0.4 degree; therefore closed loop tracking using a 4 horn pseudo-monopulse feed is employed on the TDRS relay spacecraft.

The remaining elements of the telecommunications subsystem are the same as the Part I baseline.

4.1.2 Spacecraft Design

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.

The configuration in Figure 4-1 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/MDS users and one is the TDRS/GS antenna.

As shown in the sectional views on Figure 4-1, all 3.8 M antennas, (which are of the furlable rib-mesh design) 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 basic design of AGIPA elements is the same as the Part I baseline except larger to obtain more gain in the return link. The UHF disk on-rod elements are identical to those used in the baseline.

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 can easily be adapted by an increased gimbal travel to track and communicate with user spacecraft up to and including synchronous orbit altitude, with a third TDRS on the other side of the world, or with a second ground station, with the remaining HDR/MDR antennas still furnishing full communications capability equivalent to the uprated TDRS concept. However, the telecommunications system would require modification for these alternatives.

The spacecraft is shown stowed in the Atlas/Centaur in Figure 4-2. 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.

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 sunline. 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. A weight summary of this maximum capacity TDRS is provided in Table 4-1.

Table 4-1. 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

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.

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.

4.2 SHUTTLE-AGENA TDRS DESIGN

In the case of the Atlas-Centaur, the shroud dimensions constrained design considerations to single payload concepts. However, when launching in the Shuttle the packaging constraints are relaxed and multiple launches are practical. With the payload capacity of the Shuttle-Agena combination either the uprated TDRS baseline configuration or the maximum capacity design can be launched three at a time.

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4.2.1 Telecommunications Design

The telecommunications designs for the Shuttle configurations are the same as described in sections 3.1 and 4.1.2.

4.2.2 Spacecraft Design

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 an apogee kick motor.

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.

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). The Agena and TDRS drift to the assigned stations where each TDRS is released and stops its own drift.

The total payload capability 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).

The packaging philosophy of both the maximum capacity TDRS (as designed for the Atlas-Centaur and the uprated Baseline TDRS (as designed for the Delta 2914) are the same. 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.

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.

Figure 4-3 illustrates the packaging of the uprated baseline version. The maximum capacity version is packaged in a similar manner and is shown in Volume III.

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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.

The weight of each maximum capacity 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 (see Table 3-3).

With the allowable payload of the Shuttle-Agena Tug of 1529 kg, the weight of the adapter of 68 kg is subtracted to give 1461 kg 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 (see Table 3-3).

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5.0 RECOMMENDATIONS

The basic conclusion of this study is that it is practical to implement a Tracking and Data Relay Satellite System using a Delta 2914 launch vehicle that meets all the service requirements postulated for LDR, MDR, Shuttle, and HDR user spacecraft. The satellite design developed in greatest detail is optimum for providing relay service to an earth orbital spacecraft population consisting mostly of LDRs with only a few MDRs and/or HDRs. This was appropriate based on projected mission models available during the study and from an historical point of view. However, cursory review of the most recent mission models indicates that future populations may be predominately medium data rate users with only a few LDRs and HDRs. This change in the makeup of the mission set would lead to a change in one of the mission optimization criteria. Specifically, instead of emphasizing multiple access to LDRs, which resulted in the design of a VHF/UHF antenna array and transceiver, multiple access to MDRs would be emphasized. This would undoubtedly lead to the selection of a configuration similar to alternative 2 of the uprated baseline design, when the UHF/VHF array and transceiver are replaced with an S-band array and transceiver.

Consequently, it is recommended that two studies be conducted, three technology areas demonstrated, and one experiment performed.

The two studies are as follows:

1. A TDRS Mission/System Analysis Study - to determine the advisability and desirability of designing a TDRS that emphasizes support of MDRs and would have only microwave interfaces.

2. A TDRS Design Optimization Study - to analyze the telecommunications requirements and develop the satellite designed for MDR multiple access to a preliminary design level of detail.

Although the TDRS design developed in this study uses predominately existing equipment and, in all instances uses state-of-the-art technology, predicted capabilities in the following three areas should be demonstrated.

1. The telecommunications system. Many of the elements of the system have been or are being developed. However, in order to establish the confidence needed by the owners or operators of the user spacecraft to accept the change from the ground oriented T&DA support to the space oriented, it will be necessary to demonstrate that performance will be satisfactory in the face of the conceivable problems postulated for the operational TDRS. This can be accomplished through system simulation using the actual components for each of the major elements; user transponder, TDRS telecom subsystem, ground station transceiver and signal processing equipment.

2. Demonstration of the performance of the TDRS stabilization and control subsystem. This subsystem, although using existing equipment, configured that equipment in a unique way that enhances performance and reliability but has not been demonstrated. The ability of the system to counter the major disturbing torques such as slewing of the large antennas should be shown and the actual dynamics of the spacecraft shown to be within acceptable attitude limits and rates.

3. Development and demonstration of the elements of the UHF/VHF array. If the RFI problem does in fact exist, then the relay satellite must be capable of supporting the AGIPA concept. The accomplishment of this hinges on the success of the deployment mechanism of the mechanically integrated VHF backfire and UHF disk-on-rod antenna elements.

The recommended experiment is closely related with the telecommunications simulation and antenna development and is concerned with determining the degree or severity of one of the major problem areas addressed during this study. The actual levels of terrestrially generated RFI are unknown. Worst-case literature surveys concluded that the RFI will be so high as to preclude operation. However, this analysis did not allow for duty cycles or the electromagnetic far field patterns of the antennas. In the recent airborne RFI survey conducted by NR, the reduced data gives every reason to suspect that the RFI levels as seen by earth orbiting spacecraft is not nearly as high as previously imagined. This survey was quite limited as to geographical areas and time.

To adequately define the RFI level for evaluation of the proposed telecommunications links it is necessary that a much more extensive airborne survey be conducted to establish confidence that there is no possibility of RFI completely disrupting the communications links.

THE FOLLOWING PAGES ARE DUPLICATES OF ILLUSTRATIONS APPEARING ELSEWHERE IN THIS REPORT. THEY HAVE BEEN REPRODUCED HERE BY A DIFFERENT METHOD TO PROVIDE BETTER DETAIL