NASA CR-66377

SYSTEM ANALYSIS AND INTEGRATION STUDIES FOR A 15-MICRON HORIZON RADIANCE MEASUREMENT EXPERIMENT

Horizon Definition Study

May 1967

Prepared under Contract No. NAS 1-6010 by

HONEYWELL INC.
Systems & Research Division
Minneapolis, Minn.

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA CR- 66377

SYSTEM ANALYSIS AND INTEGRATION STUDIES FOR A 15-MICRON HORIZON RADIANCE MEASUREMENT EXPERIMENT

By: William E. Eckstrom
Henry W. Berry

HORIZON DEFINITION STUDY

May 1967

Prepared under Contract No. NAS-1-6010

by

Honeywell Inc.
Systems and Research Division
Minneapolis, Minnesota

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

This report documents Phase A, Part II of An Analytical and Conceptual Design Study for an Earth Coverage Infrared Horizon Definition Study performed under National Aeronautics and Space Administration Contract NAS 1-6010 for Langley Research Center.

The study was performed in two parts. Part I provided for the delineation of the experimental data required to define the earth's infrared horizon on a global basis for all time and space. The capabilities of a number of flight techniques to collect the experimental data were then evaluated; a rolling-wheel spacecraft in a nominal 500 km, near-polar orbit was selected as the baseline technique.

The Part II portion of the study, which this report documents, provides a more extensive analysis of the sampling requirements and operational methodology for the measurement program, including the evaluation of various system constraints. In addition, design requirements and conceptual designs are established for the overall system and its associated subsystems, including radiometer, attitude determination, data handling, communications, attitude control, electric power, structures and integration, flight vehicle operations, and launch support.

Honeywell Inc., Systems and Research Division, performed this study program under the technical direction of Mr. L. G. Larson. The program was conducted from 28 March 1966 to 10 October 1966 (Part I) and from 10 October 1966 to 29 May 1967 (Part II).

Gratitude is extended to NASA Langley Research Center for their technical guidance, under the program technical direction of Messrs. L. S. Keafer and J. A. Dodgen with direct assistance from Messrs. W. C. Dixon, Jr., E. C. Foudriat, H. J. Curtman, Jr., and P. Zaepfel, as well as the many people within their organization.

PRECEDING PAGE BLANK NOT FILMED.

CONTENTS

PRECEDING PAGE BLANK NOT FILMED.

ILLUSTRATIONS

TABLES

SYSTEMS ANALYSIS AND INTEGRATION STUDIES FOR A 15-MICRON HORIZON RADIANCE MEASUREMENT EXPERIMENT

By William E. Eckstrom
Henry W. Berry

SUMMARY

The systems analysis and integration report establishes the basic conceptual design requirements, identifies the Horizon Definition Study (HDS) program functions, describes the system studies leading to the description of the recommended system, and establishes the HDS operational plan.

The basis requirements are discussed in terms of radiometric, spacecraft, compatibility, and error analysis. Certain of these requirements were subject to modification during the course of Part II, notably the error analysis and the radiometric requirements. This modification was due to generalization of the date requirements to incorporate a larger array of potential horizon sensing techniques. The final requirements delineated herein are compatible with the recommended system.

The functional analysis contains the operations required for the prelaunch, orbital, and ground phases of the program. The prelaunch phase is principally concerned with the launch vehicle and launch site and the development of the constraints and requirements within these areas. The orbital operations are correlated with the spacecraft subsystems. The ground operations identify the telemetry, tracking, and ground data processing required during the collection period and the data processing required to yield the radiance profile suitably identified for subsequent analysis.

The systems studies discuss evolution from the multiple concepts considered for the various spacecraft subsystems to a single concept and, finally, to recommend a system mechanization following effectiveness consideration including reliability.

The merger of numerous subsystem operational plans into a single program level operational plan was an integration task reported herein as the final program operational plan.

The recommended flight technique provides a means for obtaining all of the required data. Through the use of multiple flights, a high level of statistical data expectation can be achieved.

The appendices discuss the effects of radiometric errors and the potential of utilizing the Scout booster on this program.

INTRODUCTION

The systems analysis and integration studies documented herein are a portion of the Horizon Definition Study (HDS) conducted for NASA Langley Research Center, Contract NAS 1-6010, Phase A, Part II. The purpose of the HDS is to develop a complete horizon radiance profile measurement program to provide data which can be used to determine the earth's atmospheric state, especially at high altitudes. These data can then be effectively used in the many atmospheric sciences studies and in the design of instruments and measurement systems which use the earth's horizon as a reference.

Part I of the HDS resulted in the following significant contributions to the definition of the earth's radiance in the infrared spectrum:

- The accumulation of a significant body of meteorological data covering a major portion of the Northern Hemisphere.
- A large body of synthesized horizon radiance profiles computed from actual temperature profiles obtained by rocket soundings.
- A very accurate analytical model and computer program for converting the temperature profiles to infrared horizon profiles (as a function of altitude).
- An initial definition of the quantity, quality, and sampling methodology required to define the earth's infrared horizon in the CO_2 absorption band for all temporal and spatial conditions.
- An evaluation of the cost and mission success probabilities of a series of flight techniques which could be used to gather the radiance data. A rolling-wheel spacecraft was selected in a nominal 500 km polar orbit.

The Part II study effort was directed toward the development of a conceptually feasible measurement system, which includes a spacecraft, to accomplish the measurement program developed in Part I. In the Part II HDS, a number of scientific and engineering disciplines were exercised simultaneously to conceptually design the required system:

- The scientific experimenter refined the sampling methodology that must be accomplished by the measurement system. This portion of the study recommends the accumulation of approximately 380 000 radiance profiles taken with a sampling rate that varies with the spacecraft's latitudinal position.
- A conceptual design was defined for a radiometer capable of resolving the earth's radiance in the 15-micron spectrum to 0.01 watt/meter^2-steradian with an upper level of response of 7.0 watt/meter^2-steradian.

- A starmapper and attitude determination technique were defined capable of determining the pointing direction of the spacecraft radiometer to an accuracy of 0. 25 km in tangent height at the earth's horizon. The combination of the radiometer and starmapper instruments is defined as the mission experiment package.

- A solar cell-battery electrical power subsystem conceptual design was defined which is completely compatible with the orbital and experimental constraints. This system is capable of delivering 70 watts of continuous electrical power for one year in the sun-synchronous, 3 o'clock nodal crossing, 50 km orbit.

- A data handling subsystem conceptual design was defined which is capable of processing in digital form all the scientific and status data from the spacecraft. The subsystem is completely solid state and is designed to store the 515 455 bits of digital information obtained in one orbit of the earth. This subsystem also includes command verification and execute logic.

- A communications subsystem conceptual design was developed to interface between the data handling system of the spacecraft and the STADAN network. The 136 MHz band is used for primary data transmission and S band is used for the range and range-rate transponder.

- A spacecraft structural concept was evolved to contain, align, and control the thermal environment for the spaceborne subsystems. The spacecraft is compatible with the Thor-Delta launch vehicle.

- An open-loop, ground-commanded attitude control subsystem conceptual design was defined utilizing primarily magnetic torquing which interacts with the earth's field as the force for correcting attitude and spin rates.

- A selection of the Thor-Delta as booster was made from the 1972 NASA "stable" which provides low cost and adequate capability.

- Western Test Range was selected as the launch site due to polar orbit requirements. This site has adequate facilities except for minor modifications to handle the program and is compatible with the polar orbital requirements.

This report on the system analysis and integration studies conducted during Part II had the following objectives:

1. The distribution of the error budgets resulting from the basic requirements between the various subsystems to maximize the program feasibility by minimizing the sub-system design problems.

2. The identification of a complete set of program functions to achieve, by consideration of all these functional require-ments in the definition of a system concept, solid feasibility of the measurement program.

3. The recommendation of a single system concept which imple-ments the basic requirements and is based upon consideration of system effectiveness.

4. The development of a complete operational plan demonstrating the feasibility of conducting the measurement program with the recommended system.

The detailed studies are contained in four major sections treating the basic systems requirements, the functional analysis, the system development from concept definition through mission effectiveness and reliability, and the operational plan.

SYSTEM REQUIREMENTS

The system requirements represent the basic set of requirements which serve to define the goal of the measurement program. The initial systems requirements set were made up of the conclusions reached as a result of the Part I study, which evaluated the profile characteristics and their accuracies together with the global distribution and quantities to allow a useful extension of the knowledge of the earth's radiance profile. Amendments to this initial set were the result of additional data application, mission profile, and radiometer instrument specification requirements analysis.

This section establishes that initial set of basic requirements for which a feasible system design was required as a product of Part II and the recommended set of requirements which would yield a program improvement. The requirements are presented according to their classification into the following categories.

- Radiometric
- Spacecraft
- Compatibility
- Error Analysis

RADIOMETRIC REQUIREMENTS

The radiometric requirements define the characteristics of the earth's phenomenon which are to be observed. The accuracy of each observation and the quantity and global distribution is established. These requirements are subdivided into the spectral interval, profile accuracy, and data requirements.

Spectral Interval

The spectral interval is selected as the 615 to 715 cm^{-1} (14.0 to 16.28μ) carbon dioxide absorption band. This band has been established as being capable of providing the most stable horizon profiles. Part I profile synthesis and numerous programs have contributed to the establishment of this requirement. This requirement has remained firm throughout the Part II studies.

Profile Accuracy

The accuracy of the radiance profile refers to the variation from the absolute radiation. The instrument must be designed to accommodate all these requirements expressed in terms of radiance characteristics and resolution. The original requirements are as follows:

Peak radiance:	Maximum	$= 7\ W/m^2 - sr$
	Minimum	$= 3\ W/m^2 - sr$
Minimum radiance		$= 0.01\ W/m^2 - sr$
Slope resolution:	Maximum	$= 0.6\ W/m^2 - sr - km$
	Minimum	$= 0.02\ W/m^2 - sr - km$
	Maximum change	$= 0.15\ W/m^2 - sr - km^2$
Radiance magnitude resolution		$= 0.01\ W/m^2 - sr$

The above requirements were translated into instrument specifications which were expressed in terms of scale and bias quantities. An error analysis (see Appendix A) was conducted to establish the radiometer specifications of the maximum error that can be tolerated by the measurement program. The results of the analysis are reflected in these requirements:

Scale calibration	3 percent
Scale drift	0.72 percent
Scale noise	0.27 percent
Bias calibration	$0.01\ W/m^2 - sr$
Bias drift	$0.01\ W/m^2 - sr$
Bias noise	$0.01\ W/m^2 - sr$

These errors are to be interpreted as one-standard deviation errors.

Data Requirements

The data requirements establish the profile quantities and distribution required. Part I determined the expected variations of the earth's radiance profile and from this variation, the model time and space distribution requirements were defined. Statistical confidence considerations to yield a 95 percent level of confidence established the quantities of profiles for a particular space-time cell. This resulted in the original set of data requirements defined as follows:

"Uniform" time sampling in each space cell over each time cell, i.e., no more than two samples/space cell/day

One year continuous coverage

13 time cells (28 days/cell)

408 space cells

Latitude (60°S to 60°N)	320 cells
Latitude (60°N to 90°N)	44 cells
Latitude (60°S to 90°S)	44 cells

Samples per cell

Latitude (0° to 60°)	16 samples
Latitude (60° to 90°)	38 samples

Total samples for one year- 110 032

The data requirements were refined during the Part II study so as to incorporate a much larger selection of locators in arriving at the sampling needs. In addition, introduction of variable scan rate (as a function of latitude) made it possible to bring the actual geographic sampling much closer to the required sampling. The scan rates resulting are:

Profile rate/min	Latitude interval
0.419	0 - 30°
0.750	30 - 60°
1.00	60 - 82.6°

The result of collecting data to the above requirement results in 14 558 profiles in each hemisphere for each time cell of 28 days. The global total for the one-year mission becomes 378 508 profiles/year.

SPACECRAFT REQUIREMENTS

This classification contains the set of the original requirements which affect spacecraft subsystems physical relationships, mechanization constraints, quantities, and operation. They are further subdivided into requirements on the total spacecraft, the experiment package, and the mission profile.

Spacecraft

The following requirements affect the complete spacecraft with regard to the spacecraft-type classification, viewing relationships, weight, and constraints on the mechanization.

- Type - The spacecraft shall be of a rolling-wheel configuration. This classification provides the spacecraft requirement which is prerequisite to other basic requirements, i. e., that requirement for passive mechanizations.

- Spin rate - The spin rate was not specified in the original set of basic requirements. Considerations included the vehicle stability versus the detector time constants and spin-axis drift versus residual torques. The selection of a cooled detector eliminated the time constant consideration. Magnetic torques causing drift of the spin axis are lower with higher spin rates whereas eddy current drag serving to slow the vehicles spin is lower at lower spin rates. The best design value for the vehicle spin rate has been determined as three rpm.

- Attitude - The spacecraft spin-axis attitude must be within five degrees of the normal to the orbit plane. This basic requirement is related to the radiometer line of sight and the passive scanning requirements. This 10-degree band is the allowable envelope of spin-axis drift resulting from torques acting on the spacecraft and will establish a correction interval for a non-continuous control system.

- Weight - The spacecraft shall be in the 800-pound class. The original requirement further stipulated an examination of the changes to yield a compatibility with a 270-pound payload limit. The former weight implied a Thor-Delta booster, whereas the latter implied a Scout booster. The requirements to consider the Scout booster was implemented by a Scout Compatibility Study completed midway in Part II. Following this study, the final weight requirement was defined as above, i. e., below 800 pounds.

- Passive system - The basic requirement for passive measurements by the experimental package was extended during Part II to all portions of the spacecraft. This is reflected in the concept tradeoff considerations, however, excepted from this requirement are one-shot devices, e. g., initial deployment of solar panels.

Experiment Package

The following requirements affect the experiment package, i. e., the radiometer and attitude determination instruments. The type of measurements, line of sight, and redundancy requirements are specified.

- Passive measurements - Both the radiometer and attitude determination measurements are to be made with a passive scanning mechanism. The feasibility for this requirement is founded on the spacecraft type previously established, i. e., rolling wheel. The requirement is limited to the scanning function only as two subfunctions, chopping and calibration require moving elements. These mechanizations are necessary in order to fulfill the more fundamental measurement accuracy requirements.

- Scanning line of sight - The radiometer and starmapper lines of sight shall be positioned normal to the spin axis of the spacecraft.

- Redundancy - The measurement program end product of properly labeled horizon profiles which result from combining the experiment package measurements from the radiometer and attitude determination instruments places these devices in a special category wherein redundancy is a basic requirement. Redundancy of critical subsystems is to be considered as necessary in view of high reliability demands of the mission. Other redundancies are to be based upon mission effectiveness and reliability requirements discussed under the compatibility requirements section following.

Mission Profile

The following requirements are included under the classification vehicle requirements because of the effect on vehicle thermal design, drag, available solar energy, magnetic fields, etc.

- Orbit - The basic requirement established by Part I was for the vehicle to be in a near-polar orbit. This single specification produced the necessary vehicle ephemeris to produce the data requirements. Refinement during Part II established the additional requirement for a sun-synchronous orbit, inclined at 97.38 degrees for an altitude of 500 km, with nodal crossing at 3:00 a.m./3:00 p.m. These refinements were essential to the definition of numerous subsystems including the radiometer.

COMPATIBILITY REQUIREMENTS

The compatibility requirements consist of that set of basic requirements which are not directly related to either the radiometric or vehicle areas. They are tracking and data acquisition, state of the art, and mission effectiveness and reliability requirements.

Tracking and Data Acquisition

This basic requirement has remained unchanged throughout Part II stating that the existing Satellite Tracking and Data Acquisition Network (STADAN) shall be utilized with minimum modification. The accuracy of tracking, data rate capability, and command technique represent specific functions, where at least two choices are available within the basic requirement.

State of the Art

This basic requirement primarily represents a tradeoff consideration to be weighed in relation to all other basic requirements. To implement this requirement, proven subsystems were used where possible.

Mission Effectiveness and Reliability

The basic approach to these areas has remained unchanged throughout Part II. Reliability was to be approached from the standpoint of designing for a spacecraft lifetime of one year; however, no reliability goals or budgets were to be established. Relative reliability has been considered throughout the study from the concept considerations to the mechanization details. The upgrading potential of dual role equipment and redundant systems (in addition to the experimental package basic requirement redundancy) was considered in the light of the overall mission effectiveness. This combined consideration, therefore, resulted in continued tradeoffs in subfunctional areas (e.g., profile selection, tracking aids) against established mission effectiveness criteria.

The final recommended system shall be analyzed to produce a probable mean time between failure. Failure shall be functional failure, i.e., the loss of sufficient number of subfunctions to preclude data production by the spacecraft. Finally, an analysis shall be made of the use of multiple launch approaches to increase the probability of achieving the basic requirement for one-year data. Strong consideration shall be given to the use of reserve spacecraft as a back up rather than as a continuously ready standby.

ERROR ANALYSIS

The basic requirements for horizon profile positional accuracy relates to both the horizontal position and the altitude which radiometric readings are taken. The basic requirement for horizontal resolution is ≤ 25 km whereas the altitude or tangent height resolution requirement is ± 0.25 km, 3-sigma, over the complete altitude interval of +80 to -30 km. These requirements, unlike the radiometric measurement accuracies must be budgeted among several error sources. The tangent-height resolutions of ± 0.25 km represents the most severe requirement and was subjected to the following error distribution.

The distribution of tangent height error allowance followed an evolution from a uniform allocation among identified error sources, to a grouping into three levels of relative error and finally into a revised error allocation with the exception of a single error, alignment, which is treated parametrically.

Eight error sources were identified as follows:

1. Attitude determination - The uncertainty of the line of sight (LOS) relative to a physical alignment reference of the instrument or instruments (starmapper/sun sensor).

2. Unknown spacecraft dynamics - The uncertainty in predicting spacecraft motion between attitude determination intervals. In particular, the predictions over certain periods wherein attitude determination instruments are not functional.

3. Orbit determination - The uncertainty of the tracking function to determine the orbit and the spacecraft's position in the orbit. Errors in in-track and altitude measurements map into a tangent-height uncertainty.

4. Radiance to attitude determination instrument alignment - The total uncertainty of the radiometer LOS to the physical alignment reference of the attitude determination instruments. This would include both optical and physical uncertainties the latter being particularly difficult to evaluate due to launch stress shifts, long-term thermal change, etc.

5. Data processing on spacecraft - Any uncertainties resulting from data processing operations such as the analog to digital conversion, half-pulse errors in the location of star pulse centers.

6. Time correlation - The residual error following correlation of spacecraft time labels from the spacecraft and individual STADAN station clock references to a single time standard.

7. Data reduction on ground - The uncertainties associated with ground data not budgeted previously.

8. Star ephemeris - The uncertainties in the available star catalog positional information.

The original error allocation provided a uniform distribution of the ±0.25 km tangent height error allowance to the eight sources identified above on a root-sum-square basis, assuming independent errors. This yielded a budget of ± 0.88 km (7 arc sec). The second error distribution classified the eight error sources into three groups according to their potential error magnitude. Three errors were classified into Group 1 (largest errors) with the remaining errors assumed to be the equivalent to a single Group 1 error. The Group 2 individual error sources plus the entire Group 3 sources received an apportioned budget equal to a single Group 1 error. Continuing in this cascaded error distribution process, the number of Group 3 sources were apportioned from an error equal to a single Group 2 source. The result of this distribution yielded three Group 1 errors, three Group 2 errors and two Group 3 errors of ±0. 125 km, ±0. 062 km, and ±0. 044 km, respectively. Table 1 reflects this distribution.

TABLE 1. -TANGENT HEIGHT ERROR DISTRIBUTION

Error no.	Group	Description	3-sigma error budget, ± arc seconds	
			Original	Final
1	1	Attiude determination	10	10
2	1	Unknown vehicle dynamics	10	10
3	1	Orbit determination	10	10
4	2	Attitude determination alignment	5	~ 8.6
5	2	Spacecraft data processing	5	3
6	2	Time correlation	5	3
7	3	Ground data reduction	3+	3
8	3	Star ephemeris	3+	
		Total error (rss)	20	20

Subsequent evaluation identified the radiance to attitude determination instrument alignment as a critical parameter which would be extremely difficult to achieve whereas certain other errors could tolerate some degree of reduction. As a result, the final error allocation was made combining errors 7 and 8 and budgeting 3 arc sec for this combination and 3 arc sec to each of errors 5 and 6. This resulted in an allowance of ±8.6 arc sec for error alignment; however, because of the potential difficulty of evaluating the alignment for the post-launch condition and for long-term changes, this error has been evaluated up to 20 arc sec to determine the total tangent-height error resulting from this single variable parameter. The last column in Table 1 represents the final error budget yielding the ±0.250 km (20 arc sec) for the seven error sources and Figure 1 illustrates the total tangent-height error resulting from varying the alignment budget from 0 to 0.250 km.

CONCLUSIONS

The final basic requirements recommended for the subsequent program phases are as follows:

Spectral interval 615 to 715 cm^{-1} (14.0 to 16.28μ)

Radiometric

Peak radiance:	Maximum	= 7 W/m^2 - sr
	Minimum	= 3 W/m^2 - sr
Minimum radiance		= 0.01 W/m^2 - sr
Slope resulution	Maximum	= 0.6 W/m^2 -sr-km
	Minimum	= 0.02 W/m^2-sr-km
	Maximum change	= 0.15 W/m^2-sr-km^2

Radiometer, 1-sigma accuracies

Scale calibration	3 percent
Scale drift	0.72 percent
Scale noise	0.27 percent
Bias calibration	0.01 W/m^2-sr
Bias drift	0.01 W/m^2-sr
Bias noise	0.01 W/m^2-sr

Data

Profile rate/min	Latitude interval
0.419	0 - 30°
0.750	30 - 60°
1.00	60 - 82.6°

Total data samples = 378 508 profiles

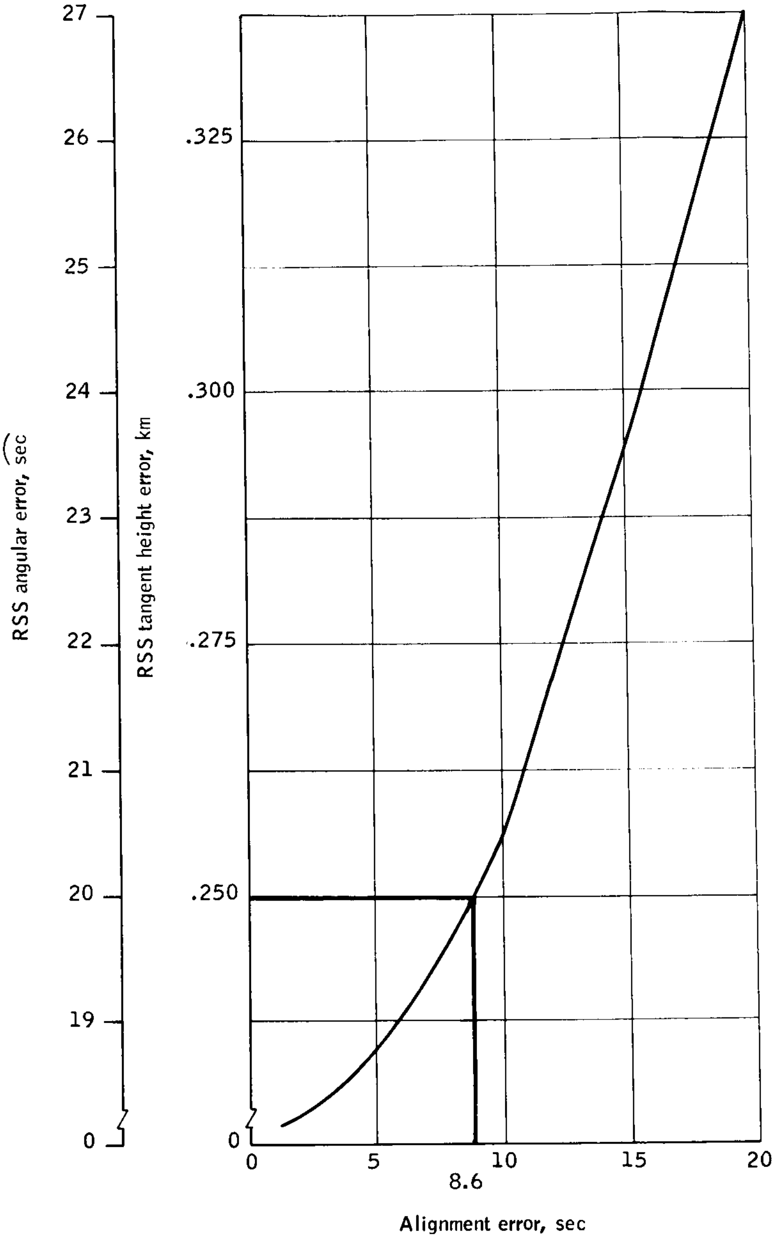


Figure. 1. Effect of Radiometer-to-Starmapper Alignment Errors

Vehicle

Rolling-wheel spacecraft

Spin rate = 3 rpm ± 5 percent

Spin axis in orbit plane within 5 degrees

Weight less than 800 pounds

Passive scanning

Radiometer line of sight normal to spin axis

Mission profile

Sun-synchronous; 3:00 a.m./3:00 p.m. nodal crossing; altitude ≈ 500 km.

Compatibility

STADAN to be used for tracking and data acquisition

Proven subsystems to be used wherever possible

One-year spacecraft design life

Multiple launches for continuous coverage

FUNCTIONAL ANALYSIS

To satisfy the study requirement to define the complete Horizon Definition Study (HDS) measurement program and to establish the program's feasibility, an analysis was conducted which defined all required program functions. A complete system concept would include appropriate coverage of each function. The functions were classified according to three basic program phases.

- Prelaunch operations
- Orbital operations
- Ground operations

Figure 2, HDS system functional diagram, identifies the HDS major functions for the above phases and, in addition, continues with a subfunctional breakdown and number identification. Each of these subfunctions is described to establish clearly the scope of the subfunction. This analysis and the number identification served to facilitate communication of tradeoff study information and other functional related data during the course of the study.

PRELAUNCH OPERATIONS

The prelaunch operations (Figure 2) involve the tasks associated with the launch site and the launch vehicle. The subfunction descriptions are as follows:

1.0 Launch Support

1.1 Launch site constraints. -- Constraints affecting the measurement program are to be identified in sufficient detail to evaluate system and/or concept feasibility as it pertains to launch site constraints. Consideration must include but not be limited to numbers of launch sites, range safety provisions, and competing launch schedules.

1.1.1 Launch site standby constraints. -- Specific variable factors as a function of the requirements for standby or backup launches shall be identified such as response time versus cost. These considerations will serve as system/mission effectiveness analysis inputs.

1.2 Spacecraft test philosophy. -- The philosophy of testing for both the initial and any subsequent (backup) launches must be identified in sufficient detail to establish feasibility and to provide for future program phase development.

1.3 Special test equipment requirements. -- For the purpose of future program phase development and cost projections, the special test equipment required to support the measurement program will be identified.

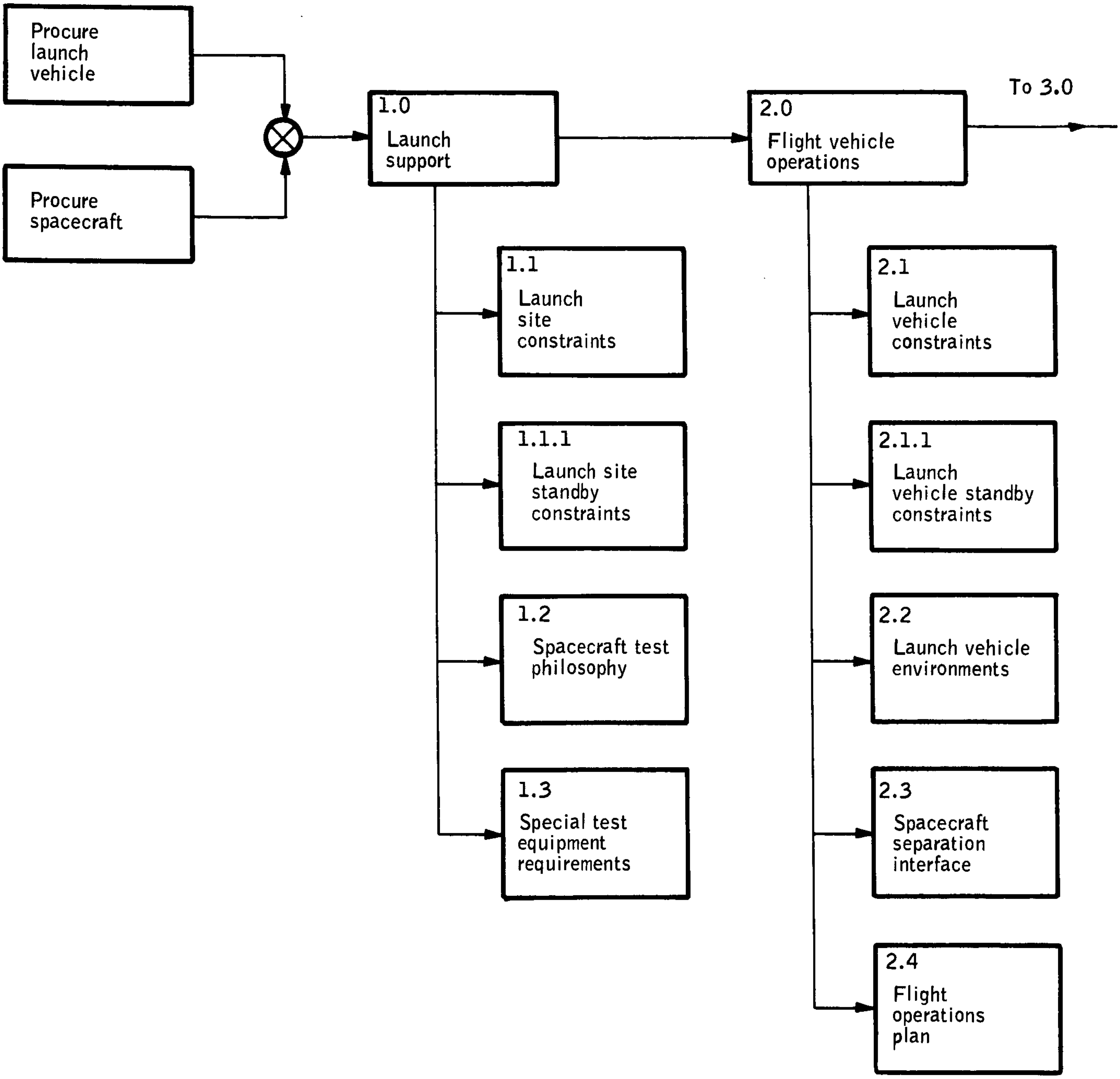


Figure 2. HDS System Functional Diagram

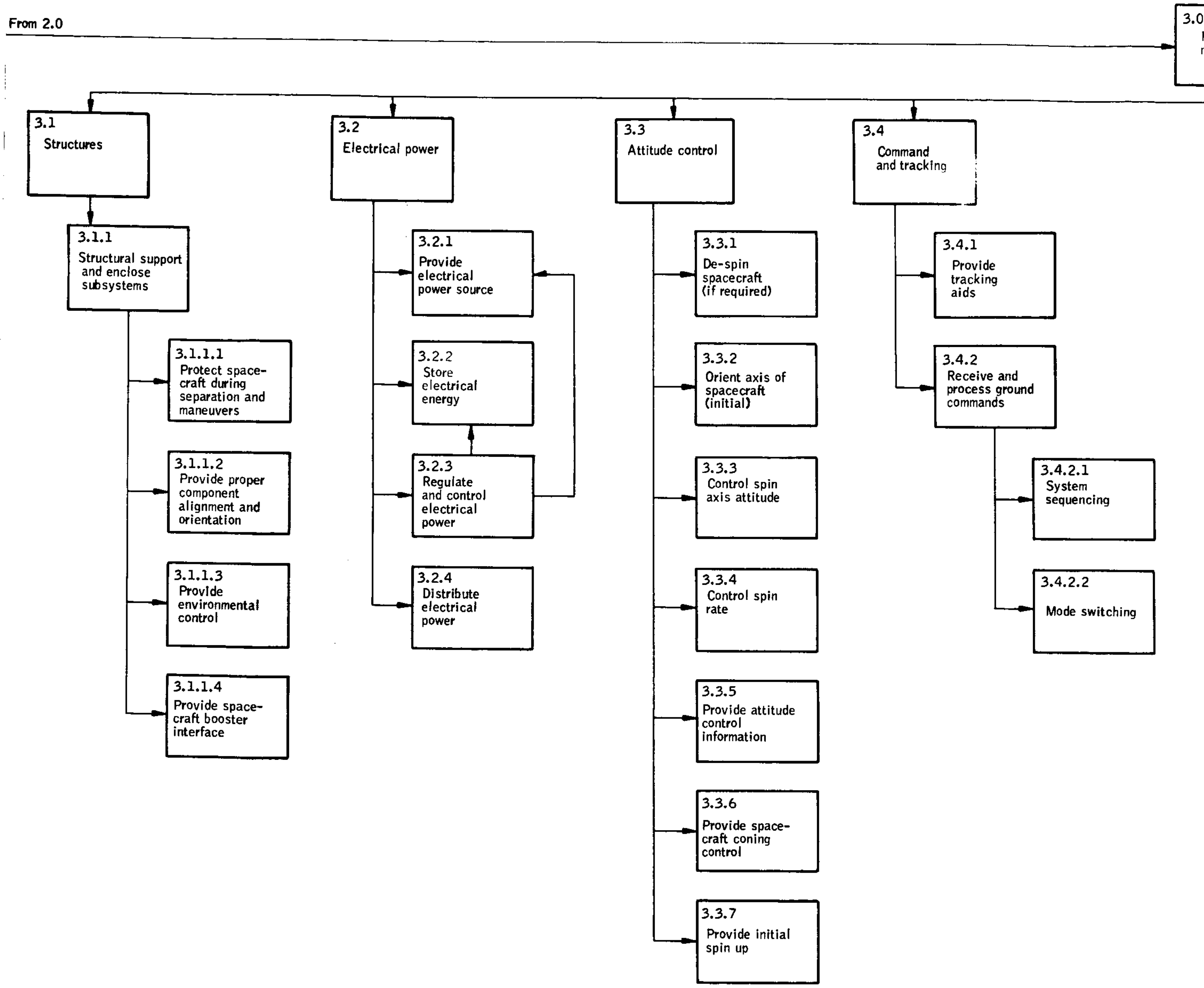
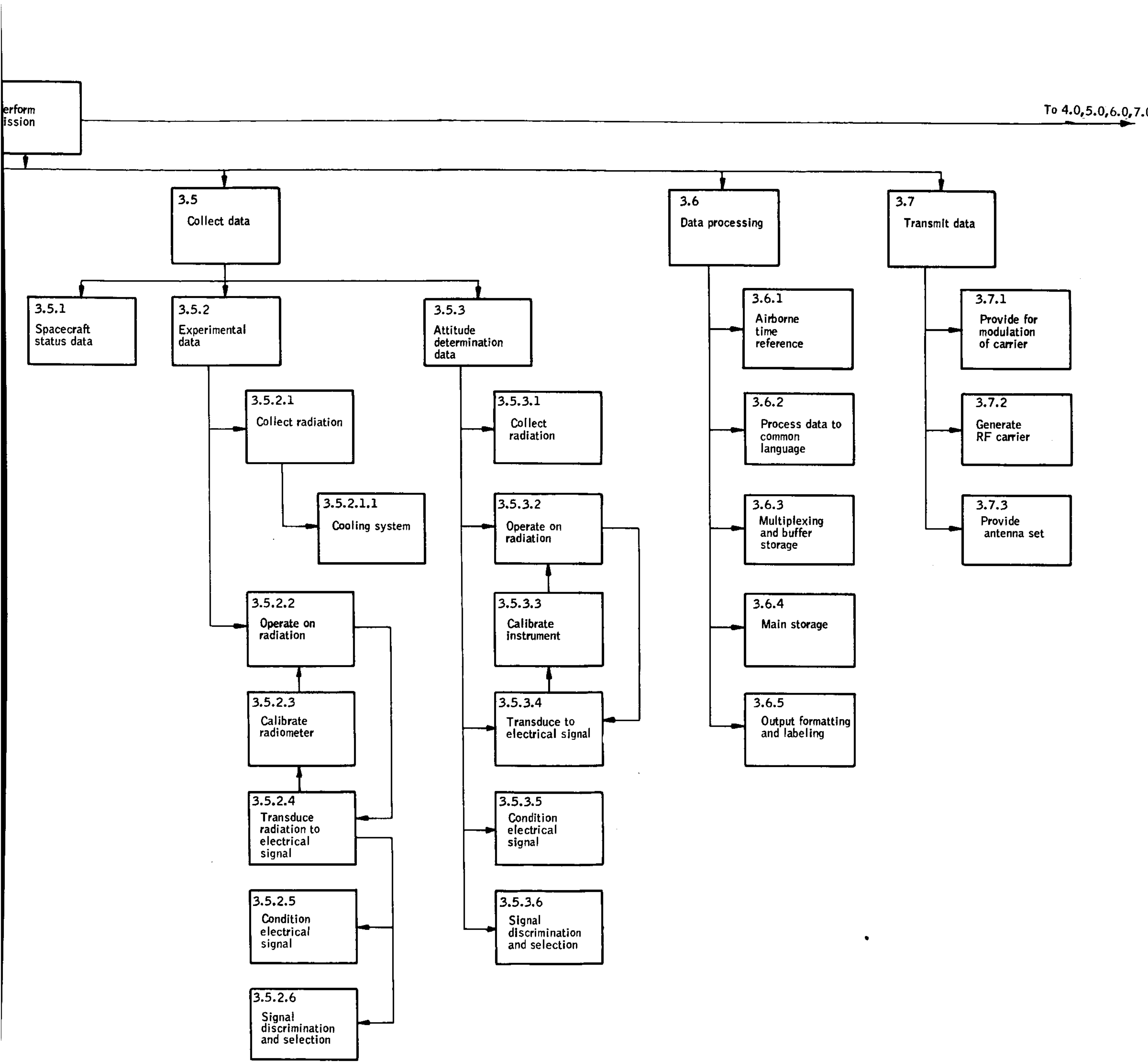


Figure 2. HDS System Fun



ctional Diagram - Continued

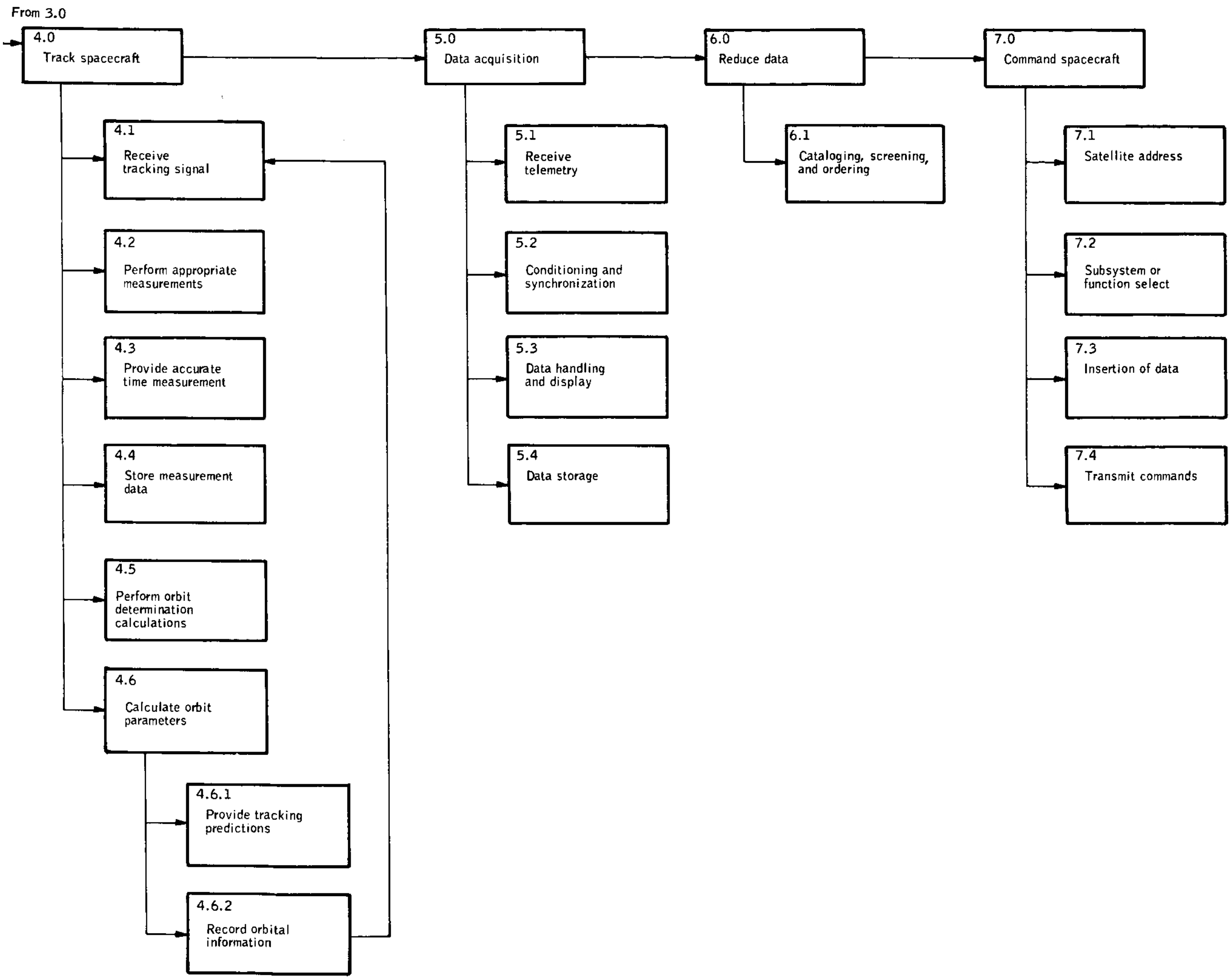


Figure 2. HDS System Functional Diagram - Concluded

2.0 Flight Vehicle Operations

This subsection will consider candidate flight vehicles which satisfy the basic requirements for payload capability.

2.1 Launch vehicle constraints. -- Detail information including but not limited to payload/orbit capability, injection accuracies, and spacecraft envelope geometry will be determined as required by spacecraft designers. Complete information relative to increased capability under development, which may be suitable to expanding the launch vehicle functional capability, shall be delineated, together with expected introduction schedules.

2.1.1 Launch vehicle standby constraints. -- Specific variable factors which affect the functions or requirements for standby backup launches shall be identified, i.e., response time versus cost, equipment shelf life, verification test expiration. This information is required for system/mission effectiveness analysis and phased program development inputs.

2.2 Launch vehicle environments. -- The environmental conditions for the booster vehicle from time of spacecraft mating through injection will be provided as spacecraft design requirements. The support equipment will be identified from these environments, wherein special conditioning is required for spacecraft subsystem needs.

2.3 Spacecraft separation interface. -- The details of the separation interface, e.g., physical mounting, separation method and disturbance forces, spin capability, etc., will be determined in sufficient detail for feasibility verification and to satisfy subsystem design inputs.

2.4 Flight operations plan. -- A flight operations plan consisting of all program events and their sequential relationship shall be provided. This plan shall, in chronological order, identify the elements of the program from the arrival at the launch site of the launch vehicle or the spacecraft through the data reduction tasks following the combination of the various telemetry data collected. This plan will establish the mutual compatibility and, therefore, concept feasibility of such considerations as spacecraft command cycles involving spacecraft measurements, data telemetry, communication to GSFC for ground processing and command selection, command transmittal to the appropriate station and finally the transmittal to, and execution by, the spacecraft.

ORBITAL OPERATIONS

The orbital operations are defined as those functions which follow the booster separation and, therefore, relate to the operation of the spacecraft subsystems. Figure 2 reflects these orbital operations in terms of the spacecraft subsystems. The delineation of this phase is the major task conducted during Part II, and, therefore, significantly greater depth is reflected in the expansion which follows.

The subsystems have been classified into three levels according to the evolution of the complete satellite concept description to firmly establish feasibility of the measurement program. These levels consider the order in which concept definitions must progress, i.e., the first level implements the basic requirements established for the program and/or are essentially involved with state-of-the-art solutions. The experimental package (radiometer and attitude determination instruments) is the only first level subsystem.

The development of this subsystem will produce requirements according to the concepts employed which must be coordinated with the appropriate subsystems (second or third level) to establish the compatibility and feasibility of a total system.

The second level subsystems are unique to the program, must satisfy basic requirements, derived requirements and, in addition, probably require state-of-the-art concepts. Second level subsystems are the attitude control, electrical power, and data handling subsystems.

The third level subsystems are considered to be capable of sufficient latitude of quasi on-the-shelf solutions and/or must satisfy compatibility requirements. Third level subsystems are the command and tracking, data transmission, and the structures subsystems. The first two are constrained by the compatibility constraint to employ STADAN, whereas the latter has sufficient design latitude to accommodate the requirements imposed by the subsystems.

The subsystem subfunctions will be delineated herein in the numerical order found in Figure 2.

3.0 Orbital Operations

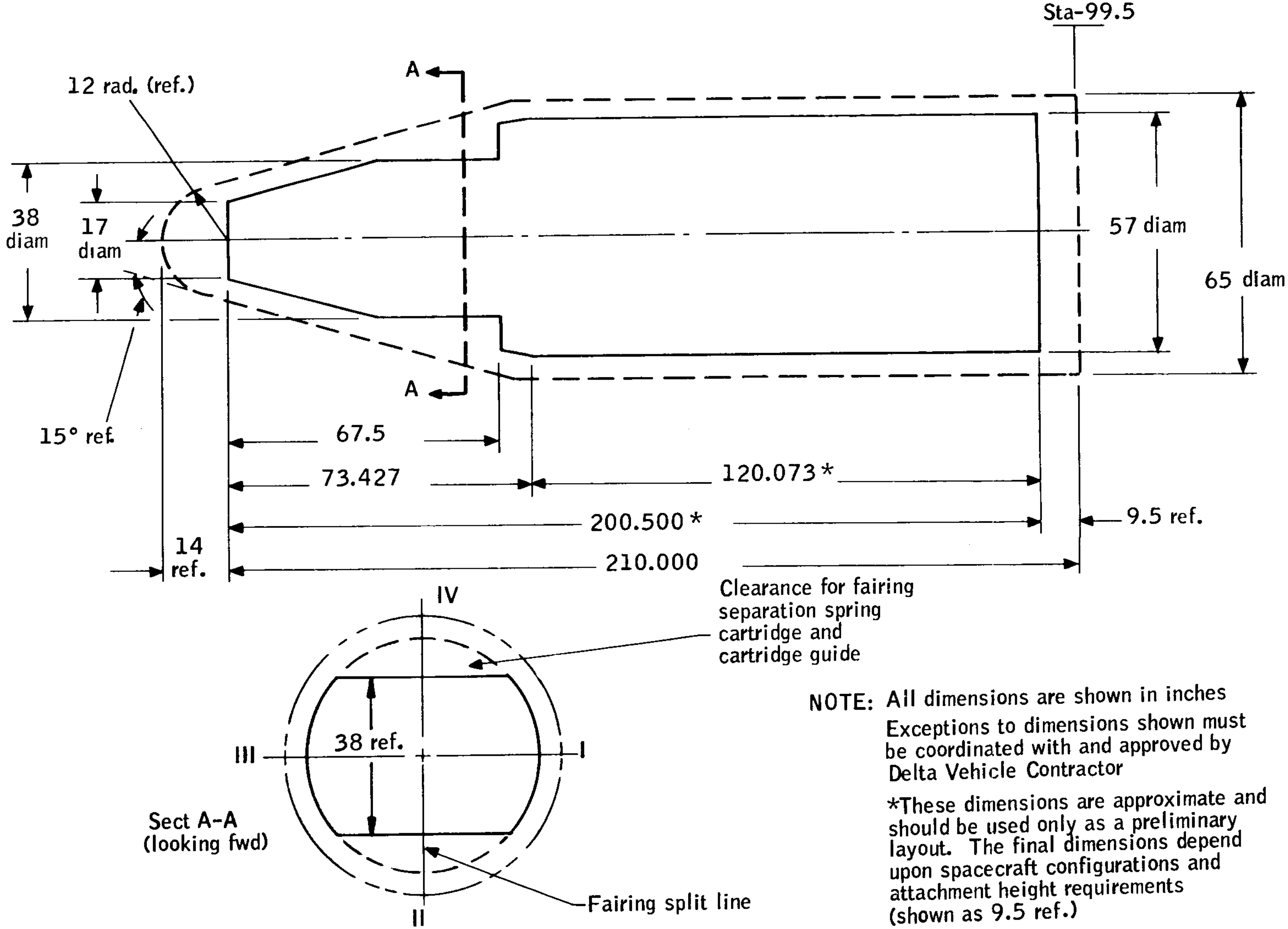
These are the operations from orbit injection by the launch vehicle to the completion of the satellite operation. As a result, all spacecraft subsystems are included and mechanizations providing these functions would establish the complete orbiting hardware package.

3.1 Structures. -- The structures subsystem must provide the physical interface for all other subsystems, satisfying all requirements for alignment, environmental control, and protection.

3.1.1 Structural support and enclose subsystem. -- The spacecraft outer envelope shall be compatible with protective shrouds available with the candidate launch vehicles, e.g., the fairing available for use with the Thor-Delta launch vehicle is the Improved Delta fairing (Figure 3).

The spinning spacecraft requires a symmetrical moment of inertia about the spin axis. For stability and to establish a preferred spin axis, the inertia ratio of the spin axis to the transverse axis must be greater than one.

The weight of the spacecraft shall be less than 800 pounds.



NOTE: All dimensions are shown in inches

Exceptions to dimensions shown must be coordinated with and approved by Delta Vehicle Contractor

*These dimensions are approximate and should be used only as a preliminary layout. The final dimensions depend upon spacecraft configurations and attachment height requirements (shown as 9.5 ref.)

Figure 3. Improved Delta Fairing/Spacecraft Envelope

Within the above constraints, the physical spacecraft configuration shall be delineated by a complete layout, moment of inertia determination, and weight estimates.

3.1.1.1 Protect subsystems during spacecraft separation and maneuvers. -- This subfunction provides for the identification of all one-shot functions. These functions will be detailed in sufficient depth to establish concept feasibility. One-shot functions include but are not limited to removal of the radiometer and the attitude determination instrument covers, and solar panel deployment.

3.1.1.2 Provide proper component alignment and orientation. -- Examples of the function within this category include instrument viewing relationships, antenna orientations, and solar panel deployment sequences. These shall be described in sufficient detail to establish compatibility of requirements and constraints in these areas. The precise alignment between the experimental package instruments is not included beyond the provision of the mounting requirements to the standards imposed by the experimental package for maintenance of distortion limits, thermal interface, etc.

The total mission duration of one year resulting in a range of environmental conditions, including all variations resulting from injection inaccuracies, shall be considered in the solution of this subfunction.

3.1.1.3 Provide environment control. -- Consideration of thermal coatings required for the control of spacecraft temperature, evaluation of the subsystem's collective thermal interchange, the provision for the separation (thermal) required by the experimental package to maintain temperature levels below the maximum design limits of that subsystem constitute the structures environmental control requirements. Thermal control shall provide for heat balance considering solar albedo and internal heating and protection during launch aerodynamic heating.

3.1.1.4 Provide spacecraft booster interface. -- The complete interface including hardpoint or mounting definition, the electrical interface, sequencing associated with the separation of the spacecraft, etc., shall be established. This includes the provision for the initial spin axis orientation and spin up. These latter functions are included under the launch vehicle; however, the interface with the spacecraft is described within this section.

3.2 Electrical power system. -- The electrical power system subfunctions include the power source, storage, regulation and control, and the distribution.

3.2.1 Provide spacecraft electrical power source. -- The electrical power source shall be established to be compatible with the power requirements. The power requirement shall indicate subsystem loading and growth requirements. For concepts using solar energy, the power load per orbit shall be deliverable under the worst sun conditions of the baseline orbit which exhibits a sun angle of 64° relative to the orbit normal and a shadow fraction of 0.364.

3.2.2 Store electrical power. -- Storage capability shall be provided as required to supply power during peak loads and when the solar array is in the earth's shadow.

3.2.3 Regulate and control electrical power. -- Electrical power regulation and control shall be provided so as to result in minimum system complexity and weight.

3.2.4 Distribute electrical power. -- The system must distribute electrical power to all using spacecraft subsystems. Fault protection and isolation should be provided. The magnetic moment of the spacecraft shall be controlled with limits required by the attitude control system. Radiated and conducted electromagnetic interference (EMI) shall be minimized.

3.3 Attitude control. -- The attitude control subfunctions produce the dynamic conditions required for the precise determination of spacecraft attitude. These requirements must be evolved in coordination with the attitude determination tasks.

3.3.1 Despin spacecraft. -- This subfunction may require a separate mechanization for launch vehicles using spin stabilization. The overspin magnitudes could be beyond the capability of the spin-rate control subfunction mechanization which, although it includes a despin program, it will be scaled for small corrections. Spin stabilization of the final stage is employed by the Scout booster.

3.3.2 Orient axis of spacecraft (initial). -- This attitude control function is a one-shot requirement which could require a special mechanization or be provided by the Thor-Delta launch vehicle. Prior to spacecraft operation, the appropriate maneuver must be performed to place the spin axis normal to the orbit plane within the attitude deadband.

3.3.3 Control spin-axis attitude. -- The disturbance torques acting on the vehicle shall be cataloged and the significant torques established to determine control torque requirements. Concepts applicable to the requirement for control torque shall be evaluated and the selected concept analyzed to establish the predictibility of spacecraft motion which must be compatible with the capability of the attitude determination process. The drift must be controlled to the limits of $\pm 5°$.

The moments of inertia shall be such that the body is symmetrical about the spin axes. For stability, the ratio of the spin axis to transverse axis inertia shall be greater than one. The sensitivities of the vehicle to inertia ratio variation shall be evaluated to establish the requirements for the structure subsystem limit requirements.

3. 3. 4 Control spin rate. -- The design spin rate will not remain stable for the complete mission. The torques acting on the vehicle which affect spin rate shall be evaluated and predictions made of their effect in terms of the correction requirements. Suitable mechanization of spin-up/spin-down functions shall be accomplished.

3. 3. 5 Provide attitude control information. -- The attitude control system shall provide for attitude error measurement to determine the necessity for corrections. An accuracy of measurement of 0. 5° is required to allow control to the deadband of ±5°.

3. 3. 6 Provide spacecraft coning control. -- Without damping, the spin vector would remain at some angle to the body principal axis because the spin-up of spacecraft is not likely to align the spin vector with the principal axis. To minimize the angle, damping is required. The attitude control subsystem shall provide a means to damp the spacecraft coning. The damping is required to assure that the spacecraft is operating in a region of stability.

3. 3. 7 Provide initial spin up. -- This function may require a special mechanism; however, the design function 3. 3. 1 and this function are related to the booster capability and constraints. For example, the Thor-Delta launch vehicle may provide this function with a modification of the spin table used for spin stabilizing final stages.

3. 4 Command and tracking. -- This subsystem provides for tracking aids and the receipt and processing of ground commands.

3. 4. 1 Provide tracking aids. -- Accurate orbit determination is required to achieve the measurement program accuracy requirements. Tracking will be used to re-establish and constantly update the orbit information. To aid the STADAN in acquiring and accurately tracking the spacecraft, on-board tracking equipment such as beacons or transponders will be required on the spacecraft.

3. 4. 2 Receive and process ground commands. -- Because a command subsystem will be required on the spacecraft to provide for ground command of at least one function (destruct), the elimination of a command system will not be considered. Ground commands will, therefore, be utilized in all concepts. The command requirements shall be delineated to establish operational sequence and the number of commands. The capability of STADAN shall be established and the command subsystem defined within the basic requirement to be compatible with STADAN with minimum modification.

3. 4. 2. 1 System sequencing. -- Monitoring the subsystem requirements to develop the total sequencing requirements will precede the configuration of the hardware.

3. 4. 2. 2 Mode switching. -- A capability is to be provided for alternate modes of operation of the spacecraft. Mode requirements shall be established by subsystem monitoring and the implementing mechanism described in association with the command system of 3. 4. 2. 1.

3.5 Collect data. -- This function includes data for the spacecraft status and the scientific requirements which is further divided into categories of experimental data and starmapper data. These are described separately below.

3.5.1 Spacecraft status data. -- The necessary spacecraft diagnostic, performance, and environmental data requirements shall be determined. Representative quantities to be measured are temperatures, voltages, currents, command responses, status, spacecraft nominal attitude, tank pressures, strain, etc. These shall be classified into separate categories according to the necessity to store the data and the capability to monitor during telemetry contact thereby reducing storage requirements.

3.5.2 Experimental data. -- The horizon radiance must be collected and operated on to provide the primary data to build horizon profiles. The following breakdown identifies the hardware oriented functions which must be defined.

3.5.2.1 Collect radiation. -- This subfunction is the selection of the collecting optics considering the basic requirements of radiance quality and alignment.

3.5.2.1.1 Cooling system. -- The detector operating temperature requirement establishes the need for a cooling system. This system shall be defined in concert with the optics system to establish the necessary concept feasibility and satisfaction of all basic requirements. Secondary cooling of optics and structure must be established to define the thermal requirements to be delivered by the structure subsystem (3.1.1.3).

3.5.2.2 Operate on radiation. -- These functions including focusing, filtering, and recollection and shall be defined as required to fulfill the basic requirements.

3.5.2.3 Calibrate radiometer. -- The calibration requirements both on the ground and in flight must be defined including detail error analysis.

3.5.2.4 Transduce radiation to electrical signal. -- This subfunction is the specification of the detector. This selection shall be supported with detail considerations arising from the basic requirements, the radiometer design constraints, vehicle stability, etc.

3.5.2.5 Condition electrical signal. -- The electronic function of amplification shall be defined associated with the detector choice, mounting interface, environments, drift stability noise, etc., in sufficient detail to establish feasibility and, in addition, potential development requirements.

3.5.2.6 Radiometer signal discrimination and selection. -- The analog to digital conversion and profile selection logic for radiance data collection shall be defined. These definitions will be consistent with the basic HDS requirements and will be stabilized to develop the spacecraft main storage requirements. Requirements include the selection of forward or rearward scans, and selection ratio (rate of profile collection). Implementation of this function may be either in the radiometer subsystem or the data handling subsystem, subject to the detailing of the relative merits of each.

3.5.3 Attitude determination data. -- The attitude determination must provide the necessary information to establish the position of the radiometer line of sight within the error allowance specified by the tangent-height error budget. The type of instruments suitable for this function considering the basic requirements for passive scanning can be limited to starmappers and sun sensors. The following functions shall be delineated for each of the instruments required to produce the radiometer line-of-sight attitude information.

3.5.3.1 Collect radiation. -- The optical elements shall be established for each instrument in sufficient detail to produce error analysis capability including thermal parameter.

3.5.3.2 Operate on radiation. -- The focusing and filtering requirements for each instrument type shall be determined, including the definition of various reticle patterns.

3.5.3.3 Calibrate instruments. -- The long term operation will require a calibration function. The instrument concepts shall completely describe the stability of operation demonstrating sufficient stability or shall determine the frequency and method of calibration and the resulting error contribution.

3.5.3.4 Transduce to electrical signal. -- A photomultiplier tube will be utilized. The characteristics of the available unit shall be established considering overload capability, stability, reliability, thermal constraints, etc.

3.5.3.5 Condition electrical signal. -- The electronics (preamplifier and amplifier) shall be specified in sufficient detail to provide interface information development requirements, error contribution, and feasibility.

3.5.3.6 Signal discrimination and selection. -- The analog to digital conversion and the selection logic for observation collection shall be compatible with basic accuracy requirements of HDS system. The logic necessary to minimize accumulation of extraneous data, but produce the attitude pulses in sufficient number and proper distribution, will be defined. The data quantities versus spacecraft mode shall be established. Implementation of the function may be either in the radiometer subsystem or the data handling subsystem, subject to the detailing of the relative merits of each.

3.6 Data processing. -- The data processing subsystem includes the subfunction of the spacecraft master time reference, data processing to a common language format, data multiplexing and buffer storage, main storage, and the output formatting and labeling.

3.6.1 Airborne time reference. -- The relative time on the spacecraft must be provided for labeling data and controlling functions. The error allowance provided for in the error budget shall be evaluated to establish compatibility and feasibility.

3.6.2 Process data to a common language. -- The data from the various experimental and spacecraft status sources must be collected, digitized, and processed for storage and transmission.

3.6.3 Multiplexing and buffer storage. -- The spacecraft status data will, in general, be collected in a time sharing method (multiplexed). These data and others will be fed to a buffer storage for organization. Usage of the main storage for spacecraft status data will be minimized.

3.6.4 Main storage. -- A memory system must be provided on the spacecraft for retaining data for a sufficient period based upon basic data requirements and compatibility with STADAN network, the coverage of which precludes real time data transmission. Applicable concepts shall be evaluated. The required capacity shall be determined considering selection complexities, data requirements, development, lead time, compatibility with basic requirements, and reliability.

3.6.5 Output labeling and formatting. -- The output of the data storage bank will be labeled and formatted before transmission. Error detection codes and synchronizing information will be added to the data from storage prior to being telemetered.

3.7 Transmit data. -- The compatibility with STADAN is the major consideration for this subfunction. For the numerous frequency links available, selection must be made and the following definitions completed.

3.7.1 Provide for modulation of carrier. -- Provide hardware for modulation of carrier to be compatible with STADAN.

3.7.2 Generate rf carrier. -- Provide the hardware for generation of rf carrier compatible with STADAN telemetry and tracking facilities.

3.7.3 Provide antenna set. -- Provide the antenna set on the HDS spacecraft compatible with airborne transmitters and receivers and STADAN capabilities. Establish the mounting requirements for structure subfunction 3.1.1.2.

GROUND OPERATIONS

4.0 Track Spacecraft

Spacecraft tracking for orbit determination and telemetry reception which is compatible with STADAN shall be provided. The goal is to describe a method of implementing the tracking requirement which demonstrates full compliance with the basic requirements. Any changes to the STADAN shall be identified.

4.1 Receive tracking signal. -- The frequency or frequencies required for the tracking function will be identified together with the STADAN station ephemerides.

4.2 Perform appropriate measurements. -- The specification of the measurements and measurement accuracies necessary to achieve compliance to the basic requirements and error allowances shall be made.

4.3 Provide accurate time information. -- The tracking station time and on-board spacecraft time errors must be within the allocation.

4.4 Store measurement data. -- Provisions will be made to record orbital tracking signals for transmission to the orbit determination facility.

4.5 Perform orbit determination calculations. -- The tracking signal will be utilized in the appropriate ground computer program to determine the spacecraft orbit.

4.6 Calculate orbit parameters

4.6.1 Provide tracking predictions. -- Spacecraft position versus time will be determined for use by the tracking and data acquisition stations. This function will be used periodically to create a detail tracking plan according to the general cycles operation over the mission duration.

4.6.2 Record orbital information. -- Precise spacecraft position versus time will be determined for use in the tangent-height calculations. The data will be identified for further development of the data reduction task 6.1.

5.0 Data Acquisition

The data acquisition subfunctions similar to the tracking subfunction, in terms of the STADAN constraints solid feasibility of concept, must be established including any potential changes to the STADAN.

5.1 Receive telemetry. -- The telemetry frequency or frequencies for the primary experimental and attitude determination data will be specified. The STADAN station ephemerides will be evaluated to determine the requirement for storage on board. Backup capability and status data transmission requirements will be evaluated and a final configuration selected.

5.2 Conditioning and synchronization of signal. -- Needs of signal conditioning and synchronization of received satellite signals will be evaluated.

5.3 Data handling and display. -- Identification of the data to be processed and displayed at the station to monitor the spacecraft performance will be provided.

5.4 Data storage. -- The experimental, attitude determination, and spacecraft status data storage requirements shall be delineated for the subsequent processing under subfunction "6.0 Reduce Data".

6.0 Reduce Data

6.1 Cataloging, screening, and ordering. -- The product of the measurement program will be individual profiles properly labeled as to tangent height, location, time, etc., comprising the raw data to subsequent analyses. The conversion of the various portions of the spacecraft data into this end produce must be established to describe a complete measurement program. These data merge from the various sources of raw spacecraft data, attitude determination programs, tracking programs, and time correlations to yield the final radiance profile data. Included are the requirements for the final attitude determination programming.

Special attention will be given to the initial solution of the spacecraft attitude following the spacecraft injection and magnetic moment compensation. The processing of the coarse attitude control subsystem attitude and spin errors to select the appropriate commands for changes in vehicle attitude and spin will be delineated.

7.0 Command Spacecraft

This subfunction must be considered in conjunction with the command, tracking, and data processing subsystems. The basic requirement is that of STADAN compatibility.

7.1 Satellite address. -- The accepted method for coding satellites will be discussed to define spacecraft subsystem interface requirements.

7.2 Subsystem or function select. -- The method of constructing a command for transmission to the spacecraft will be determined to define spacecraft decoding requirements

7.3 Insertion of data. -- An evaluation of data insertion requirements over and above discrete commands will be made.

7.4 Transmit command. -- The constraints for commanding the spacecraft will be determined. Commands for mode change, redundancy selection, etc., will be considered separately from tracking and telemetry commands due to a single STADAN station of the former command types.

CONCLUSION

This functional analysis identifies the elements which constitute a complete measurement program. To achieve the study goals of establishing the feasibility of the total program, each of these subfunctions must be evaluated and/or mechanized in sufficient detail to assure the complete compatibility of a system configuration.

SYSTEMS STUDIES

This section reports the Part II systems studies resulting in a description of the recommended system concepts, redundancies, and flight technique. The section has been organized into four major subsections:

- Background
- Initial concept definition
- Systems effectiveness
- Recommended system

The background subsection reviews the total set of program requirements placing them into the context of priorities which served as a primary influence on the systems studies during the course of Part II. The initial concept definition represents a major milestone of Part II. The study evolution leading to this milestone is discussed in terms of the interrelationships between the data analysis, mission analysis, and concept definition tasks. The systems effectiveness subsection discusses the approach to systems effectiveness, contains in-depth discussions of the tradeoff studies including those leading to the initial concept definition, summarizes the system reliability, and, finally, reports the analysis conducted on redundant flight techniques leading to a recommended technique. The fourth subsection summarizes the recommended system in terms of final subsystem descriptions and the flight technique to be used in conducting the HDS program.

BACKGROUND

The basic systems requirements in the areas of radiometric, spacecraft, and compatibility collectively form the total definition of the study goals. The key elements of these requirements will serve herein to provide an appropriate introduction to the systems studies discussions.

The fundamental data requirement for a full continuous year's coverage effected every element of the measurement program being mirrored in the specific requirements for redundancy, spacecraft configuration, passive systems, and the basic approach to mission effectiveness and reliability. This requirement served as a primary tradeoff criteria which forced the subsystem analyses to consider the fundamental capability of any concept to deliver the function required with an adequate inherent reliability. An example would be the resistance to introduce complete flexibility to program data collection in the interest of minimizing storage requirements due to the reliability degradation.

Accuracy in terms of the tangent height resolution of ± 0.25 km was a second significant requirement being distributed among potential error sources both on the spacecraft and the ground. The previous discussion detailing the specific error goals adequately indicates the impact of this requirement. Basic questions of the feasibility of achieving these specific errors, representing in some cases entire subsystem feasibility demonstrations, is the question involved.

The final requirement deserving introductory emphasis is that of the mission profile. The effect on subsystems design, data characteristics, and the ground functions of tracking and data gathering result in an acute program sensitivity to the mission profile.

INITIAL CONCEPT DEFINITION

The identification of the subsystem concept candidates and the analyses yielding an initial system concept definition followed the establishment of the functional requirements. Prior to concept screening, the identification of mission profile functional sensitivities and the definition of a design mission profile was required. Against, this mission profile requirement, the rationale for the selection of sybsystem concepts, was developed.

This program milestone will be discussed by describing the interrelationship of the major study elements of Phase A, Part II (Figure 4) in the development of the key parameters leading to the system concept definition. The basic requirements resulting from Phase A, Part I represents the common point of departure for the three major study elements which are:

- Data analysis
- Mission profile
- Concept definition

The data analysis represents the continuing activity of monitoring the final product of the program, i.e., horizon profiles, to assure the quantity, quality, and applicability of the data.

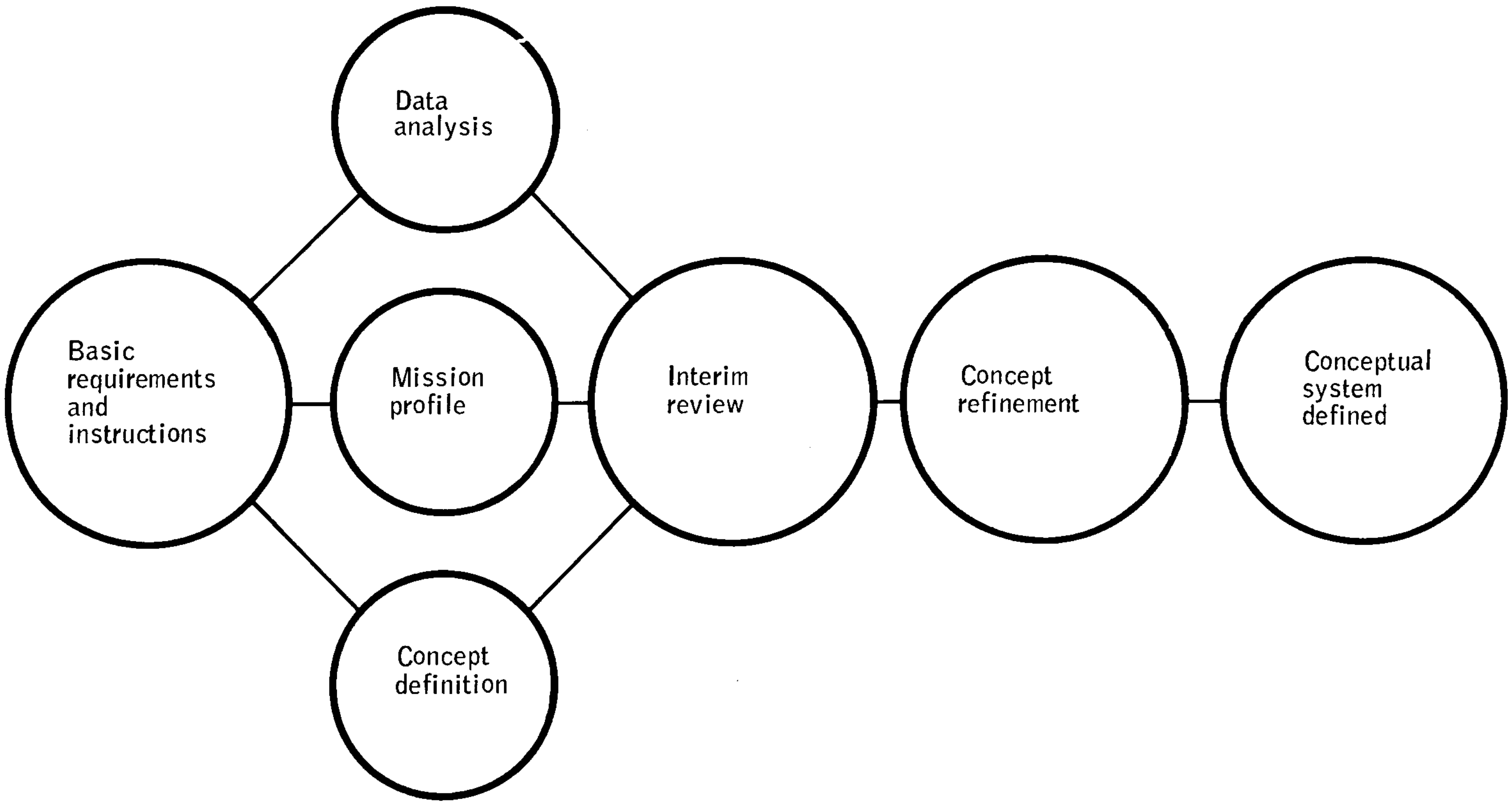


Figure 4. Part II Study

The mission profile task must yield a single design mission and the supporting design information required by the subsystem designers. The basic requirements defined the class of orbits, i.e., near polar, any of which will provide the data quantity properly distributed to analyze the significant horizon profiles factors. However, the range of design constraints resulting from this family of profiles is too great for adequate concept analysis; therefore, it was necessary to establish a single mission profile against which concept feasibility could be assessed.

The concept definition task was the major task of Part II. This task goal was the solid establishment of the spacecraft feasibility, the flight project feasibility, and the delineation of the complete measurement program.

The first step in the conceptual definition was the identification of functional sensitivities to mission profile which affect the orbital operations.

The major sensitivities were identified as:

- Radiometer - Extreme sensitivity to sun-line relationship, giving rise to baffling and/or shuttering requirements.
- Starmapper - Sun, moon, and earth sensitive to a greater degree than the radiometer.
- Electrical Power - The solar cell concept feasibility is intimately related to sun-line and earth shadow parameters.
- Structure - Passive thermal control feasibility is sensitive to the particular orbit.

The sun-synchronous twilight orbit produces the most favorable sun-line relationships which reduce the solar power and thermal design problems to a minimum. The angle between the sun line and the radiometer line of sight, viewing in the orbit plane, is maintained at a maximum thereby resulting in minimum baffling problems. An additional advantage was the ability to collect polar data without introducing any additional requirements on the vehicle. Based on the above advantages, the twilight sun-synchronous orbit was selected as the design mission profile early in Part II.

Examination of the resulting radiance profiles identified an almost total lack of typical daily, or diurnal, variation. The nominal local time for the profiles would be 6:00 a.m. to 6:00 p.m. periods of near equal temperatures; therefore, because of the high correlation of the expected energy in the 14- to 16-micron band with temperature, the variations could be predicted to be at a minimum. Due to the significance of possible diurnal variations, the two additional orbits were considered. These were:

- Sun-synchronous noon orbit (12:00 a.m. / 12:00 p.m. nodal crossing)
- Maximum precession orbit (70° inclination)

The sensitivities previously identified were evaluated for each of these orbits.

The sun-synchronous noon orbit represents the worst case sun-synchronous orbit from the standpoint of sun-line design problems and was employed as a test of the limit for sun-synchronous orbits. The maximum precession orbit represented a limit of the near-polar basic requirement. This orbit was examined due to its capability of yielding data with a maximum diurnal content. The advantages and disadvantages of each are summarized below.

Sun-synchronous noon orbit. --

- Advantages
  - Insensitive to injection inaccuracies
  - Polar coverage available without penalty
  - Diurnal variation is near a maximum
- Disadvantages
  - Solar power concepts are less efficient
  - Thermal control (passive) complexity is greatly increased
  - Instrument shutters and baffles are required

Maximum precession orbit. --

- Advantages
  - Insensitive to injection inaccuracies
  - Precession of 15 hours in 28 days increases diurnal data content
- Disadvantages
  - Solar and thermal problems are maximum
  - Polar coverage requires profiles to be taken ± 20° from the orbit plane

The conclusion was to examine the diurnal content and sun line related design problem of sun-synchronous orbits which lie between the two limits established by the noon and twilight orbits to achieve a satisfactory level of diurnal content and an acceptable sun-line set of constraints.

To evaluate the diurnal content, an analysis was conducted which yielded a typical curve establishing the expected diurnal temperature variation (Figure 5). Considering this relationship and the above conclusion, the baseline mission profile was established as a sun-synchronous orbit with a local time (nodal crossing) of 3:00 a.m. and 3:00 p.m. This selection provides a significant diurnal variation. Further shifting toward the twilight case (4:00 a.m./4:00 p.m. nodal crossing) would relieve design problems related to marginal solar sun-line conditions and represents a design relief valve. These shifts would be contemplated only in the event detail subsystem definition disclosed a solution for the established mission model to be marginal.

Candidate subsystem concepts were then evaluated against the subfunctions required by the functional analysis and in compliance with the functional sensitivities. Figure 6 identifies the concepts which were considered for each subsystem area of the spacecraft.

The three subsystem levels previously identified are restated to serve as an introduction to the concept selection discussion.

Level	Definition	Subsystem
First	Systems which implement the basic requirements and/or represent advanced state-of-the-art solutions.	Radiometer Attitude determination
Second	Systems which must satisfy the requirements imposed by the experimental package, are unique to HDS, but are not considered to require state-of-the-art advances.	Attitude control Electrical power
Third	Systems having sufficient design latitude to satisfy all imposed requirements.	Structures Command and tracking Transmit data

- Radiometer -- The first consideration was the choice of the detector, the basic choice being between a thermistor bolometer and a cooled detector. Bolometers do not have adequate response characteristics to satisfy the basic requirements and, therefore, were ruled out. Selection of Ge:Cd as the cooled detector was based upon its availability and high detectivity. The second consideration was the type of optics with the basic decision being between refractive and reflective optics. Refractive optics which would satisfy the resolution requirements would be larger than currently available giving rise to questions concerning the ability to extend the refractive optics state of the art, the weight of the optics, and the stability of structural alignment. Reflective optics, on the other hand, are not bound by this restriction and were, therefore, selected as the preferred approach.

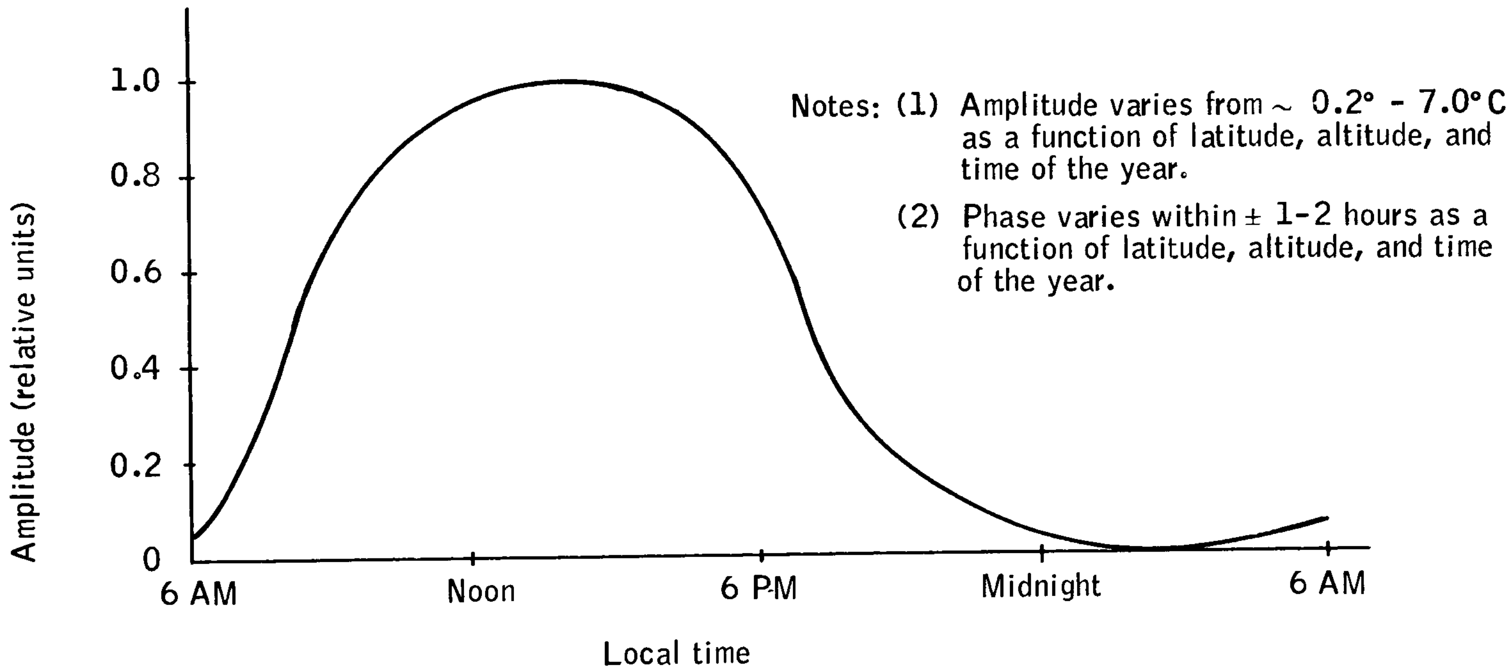


Figure 5. Typical Curve of Diurnal Temperature Variation

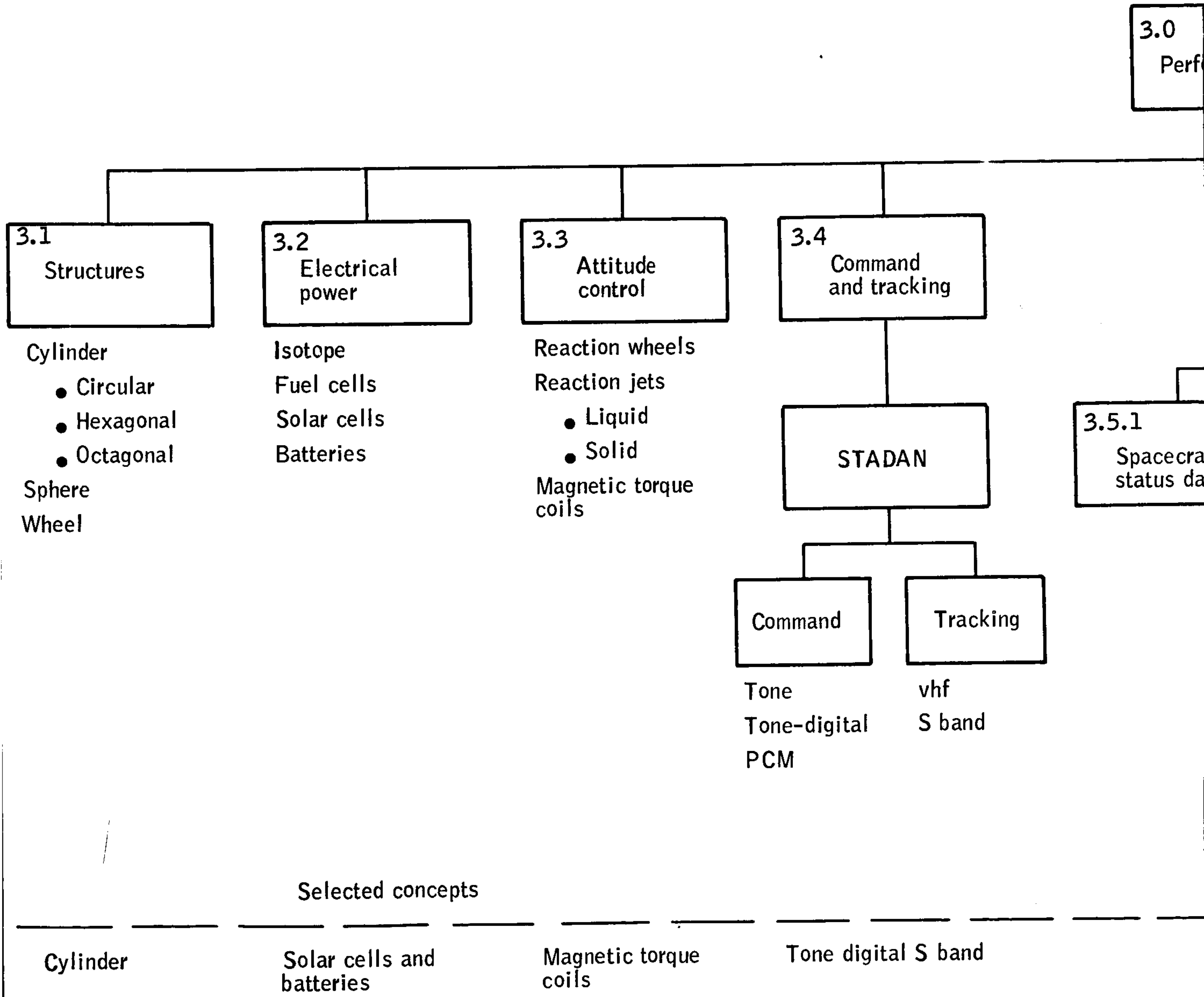
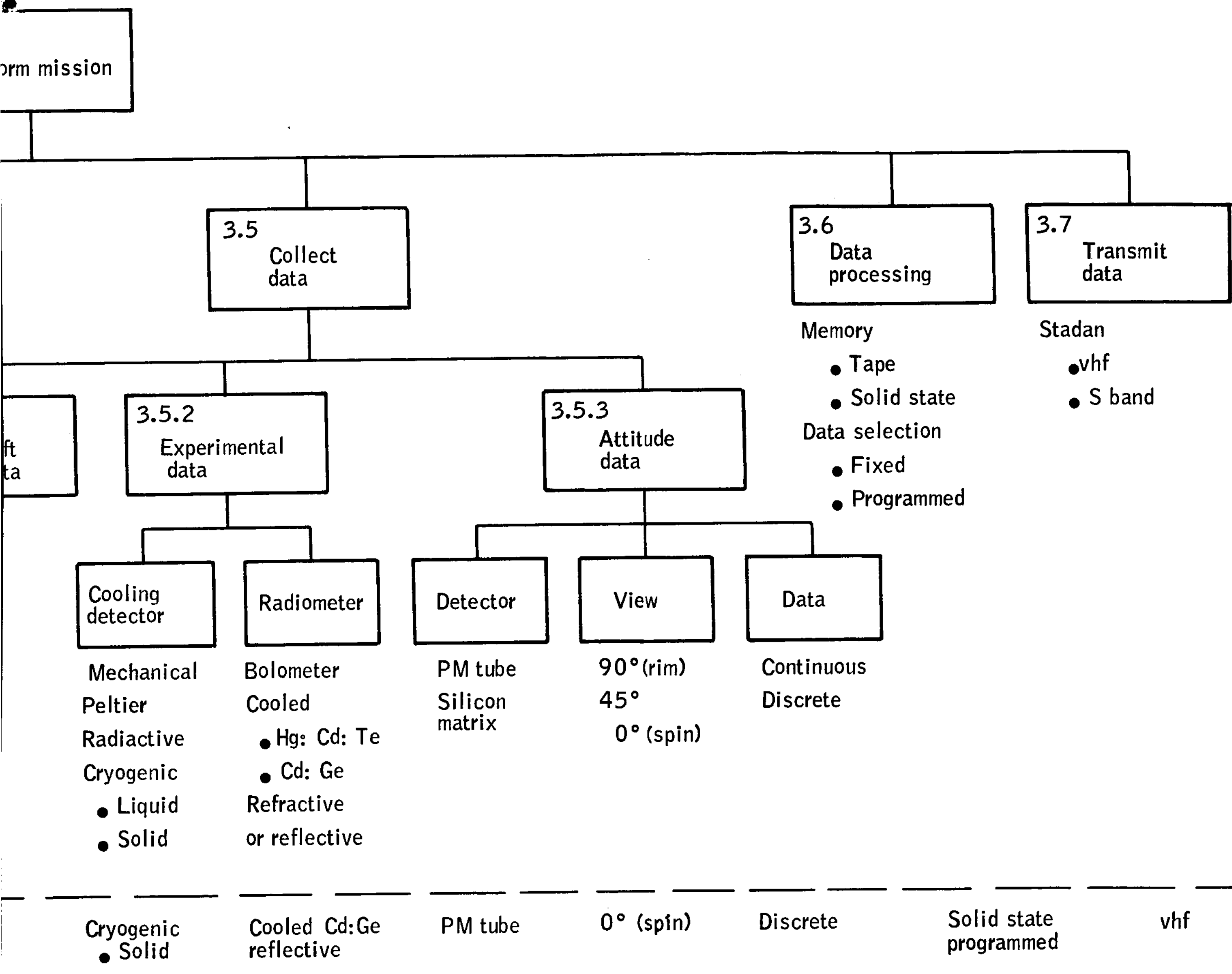

3.0
Perf
3.1
Structures
3.2
Electrical power
3.3
Attitude control
3.4
Command and tracking
Cylinder
Circular
Hexagonal
Octagonal
Sphere
Wheel
Isotope
Fuel cells
Solar cells
Batteries
Reaction wheels
Reaction jets
Liquid
Solid
Magnetic torque coils
STADAN
Command
Tracking
Tone
Tone-digital
PCM
vhf
S band
3.5.1
Spacecra
status da
Selected concepts
Cylinder
Solar cells and batteries
Magnetic torque coils
Tone digital S band

Figure 6. HD



)S Spacecraft Concepts

The cooling function is a derivitive of the detector choice. Mechanical refrigeration systems cannot produce the required temperatures and have limited life. Peltier or electric cooling cannot produce the required temperatures and are heavy power consumers due to their inefficiencies. Radiative cooling cannot produce the necessary temperatures. Cryogenic systems can produce the low temperatures required. Liquid cryogenic systems weigh more than solid systems for a given cooling capacity and have unpredictable mass shifts during their operation. Feasibility of solid-cryogenic cooling systems has been demonstrated and has been selected.

- Attitude Determiniation -- Both photomultipliers and matrices of silicon cells have been considered for the starmapper detector. A photomultiplier tube (PM) was selected in preference to silicon matrix due to state-of-the-art detection capabilities. PM tubes, with greater overload capability than presently available, are under investigation to minimize the potential shuttering problems.

  The viewing aspect relates to the direction of pointing relative to the vehicle spin axis (normal to the orbit plane). The 0° case is actually a range from 0° to 30° from spin axis and represents the least design problems associated with sun angle. However, in this position and operating full time, the sensor cannot provide sufficient star data due to the limited portion of the celestial sphere which is viewed, the occultation due to the earth, and earth shine. The 45° case (30° - 60° range) was potentially a full time system subject to adequate sun baffling solutions and did provide sufficient star sightings. The 90° or rim case is limited to operation during shadow (night) periods and would require an auxiliary sun sensor. Alignment problems between the radiometer and star mapper are minimized with the rim mounting as the optical systems can be parallel and are always in the same plane.

  The rim concept has been selected with auxiliary sun sensor providing data during daylight operation. The considerations in selecting the rim case include the previous experience with instruments capable of operation with this viewing attitude and the ability to provide an adequate solution over the entire orbit with the combined instruments. The ability to collect data at intervals (discrete) versus continuous is based on the predictability of spacecraft motion.

- Attitude Control -- Reaction wheels (and control moment gyros) were considered even though they are not passive systems; however, a more basic consideration is the torques which couple into the spinning spacecraft creating large disturbance force resulting in an inability to predict spacecraft motion to the accuracies required by the attitude determination subsystem.

Reaction jets are active systems, suffer from mass shifts and leakage induced disturbances forces and, therefore, were eliminated.

Magnetic toruqing is a feasible concept, is passive, requires little power, and has proven capability for one year's operation in orbit. Therefore, magnetic torquing has been selected as the control concept. Augmentation of the long term control by magnetic torquing coils with a reaction jet system for the initial spin up or despin functions is required only if the selection of the launch vehicle produces an appropriate requirement.

- Electrical Power -- Three basic types of power generation subsystems have been considered; isotope, fuel cells, and solar cells with batteries. Isotope systems are considered feasible; however, the development of a suitable low power system would be required. Several logistic problems exist, including a recovery requirement. Fuel cells are too heavy for the HDS mission since fuel alone is estimated to weigh 800 pounds. Solar cell systems are feasible for the projected power requirements and the mission profile. The resulting choice was solar cells with battery storage to supply power during shadow periods and peak load demands.

- Data Processing -- Solid-state and magnetic tape memory systems were examined for the onboard data and storage function. Considering only digital data processing, the estimate of memory requirements to satisfy the data quantity and distribution resulted in a bulk data storage level within the range of feasibility for solid-state memories. Tape storage requires an active subsystem which produces disturbance torques and requires complex buffering interfaces. Probable difficulty in obtaining reliable operation over a one-year period is an additional disadvantage of tape recorders. The selected concept is a solid-state memory system.

  The selection of data to be stored as a fixed ratio of the raw data minimizes the complexity; however, it results in a larger bulk memory than a programmed selection. The term "fixed" refers to a constant ratio of stored to total available profiles, whereas the programming function could be designed to have some degree of ground commanded and automatic profile selection complexity. The selected concept is to have a programmer to implement a set of profile selection commands. The degree of complexity required of the programmer will be a balance between the complexity penalty versus the utility of the resultant data and the versatility of the spacecraft.

- Structures -- The basic cylinder offers adequate design freedom to control inertia ratios, minimize mass change effects, and provide protection and thermal control; and, therefore has been selected as the recommended concept.

- Command and Tracking -- The basic requirement was that the subsystem be compatible with STADAN with minimum modification. The command capability of STADAN is:

  Tone: Seven discrete commands

  Tone Digital: Seventy discrete commands

  Pulse Code Modulation (PCM): Unlimited word commands

  It is presently estimated that more than seven commands will be required, thereby eliminating the basic tone command system. Capability to handle up to seventy commands and its simplicity when compared to the PCM system, resulted in the selecting of the tone digital system following an estimate of a command total less than seventy.

  The tracking choice was between vhf and S band. The accuracy required for orbit determination resulted in the selection of the S-band Range and Range-Rate system based on present knowledge of tracking accuracies.

- Transmit Data -- The telemetry function must also be implemented within the STADAN. The choice of vhf over S band was a direct result of the greater coverage available with vhf which allows on-board storage to be kept to a minimum.

The selected concepts are summarized by Table 2. These concepts were refined through systems effectiveness considerations to yield a total program definition demonstrating both solid feasibility and adherence to all basic requirements.

TABLE 2. - CONCEPT SUMMARY

Function	Concept
Radiometer	Cooled detector Reflective optics
Detector cooler	Solid cryogenic
Attitude determination	Starmapper - rim plus sun sensor
Electrical power	Solar cells and batteries
Attitude control	Magnetic torque coils
Data processing	Solid-state memory Programmed data Collection

TABLE 2. - CONCEPT SUMMARY - Concluded

Function	Concept
Structure	Cylinder
Command	STADAN tone-digital
Tracking	STADAN S band R & RR
Telemetry	STADAN vhf

SYSTEMS EFFECTIVENESS/RELIABILITY

The effectiveness effort during Part II emphasized the total system concept. This approach ensured that judgements made in each subsystem area would result in the most effective overall system. Reliability goals were not established. Instead, effort was concentrated on obtaining achievable results. Consequently, redundancy configurations were based largely on systems engineering judgments and analysis of sensitivities to failures rather than on conventional, component-count reliability predictions. Following these tradeoffs and decisions, reliability estimates of the resultant system were made. The concluding task in the system effectiveness area was to estimate the number of standby systems required to achieve a one-year mission.

Generally, the study efforts proceeded through the following sequence:

- Analyze and document completed tradeoffs in each subfunction area
- Determine the need for, and monitor additional tradeoffs
- Estimate subfunction failure rates
- Incorporate appropriate redundancy
- Estimate system reliability

The system effectiveness section of this report is divided into three subsections. The first summarizes the tradeoffs performed in each subfunction area; the second presents a summary discussion of the reliability estimates; and the third subsection describes an analysis of redundant flight techniques capable of increasing the probability of achieving the mission goals.

Tradeoff Summary

This subsection contains summaries of the tradeoffs which collectively lead to the defined system concept.

For convenience, the presented tradeoffs are subdivided into subsystem areas. However, because of the many interrelationships between the various subfunctions, these tradeoffs were not performed independently. Since it was found that the decisions related to the radiance measurement and attitude determination subfunctions (research package) had a dominant effect on the decisions made in other subfunction areas, it was necessary to make many iterations of the individual subfunction studies to achieve the most effective overall system configuration. The summaries which follow do not attempt to describe the many iterations which actually occurred but instead identify each significant tradeoff that was considered and state the resultant decisions. The details of the tradeoffs involved are documented in separate reports.

Radiometer. -- The radiometer is described functionally in Figure 7 as an electro-optical instrument which accepts radiance inputs and converts them into electrical outputs. The significant elements are seen to be an optical system and baffle, a modulator (or chopper), an ir filter, an ir detector (radiance-to-electrical transducer), and associated electronics. Cooling of the ir detector and/or the optics may be required. In addition, it is necessary to initially calibrate the instrument before flight and to check its calibration during flight.

Figure 8 shows the general flow of the radiometer studies starting at the left and moving to the right. Figure 8 also shows, from left to right, the relative significance of each trade as far as the radiometer and the total system configuration are concerned. Of these areas, the most significant in their impace on the system configuration were:

- IR detector
- Cooling
- Optics

The single, most-significant decision area in the radiometer study was the selection of the ir detector, since its characteristics strongly influenced the remainder of the system. An extensive investigation of potentially suitable ir detectors, showed that the use of photon detectors was necessary to be compatible with the required ir sensitivity, with scanning of the horizon by a rotating-orbiting spacecraft, and with practical optics dimensions.

Since all photon detectors for the 15-μ region require cooling to very low (cryogenic) temperatures for effective operation, it was necessary to decide on the most practical way to provide the cell cooling. This resulted in the selection of a sublimating, solidified gas as the cryogen which was compatible with both the duration of the mission and the payload size and weight limitations. To minimize the weight of cryogen and the "noise" generated by radiation from the optical elements, it is necessary to keep the radiometer's structure and optical elements at a relatively low temperature by allowing continuous radiation to cold space. This decision strongly influenced the design of the spacecraft structure as well as the configuration of the radiometer.

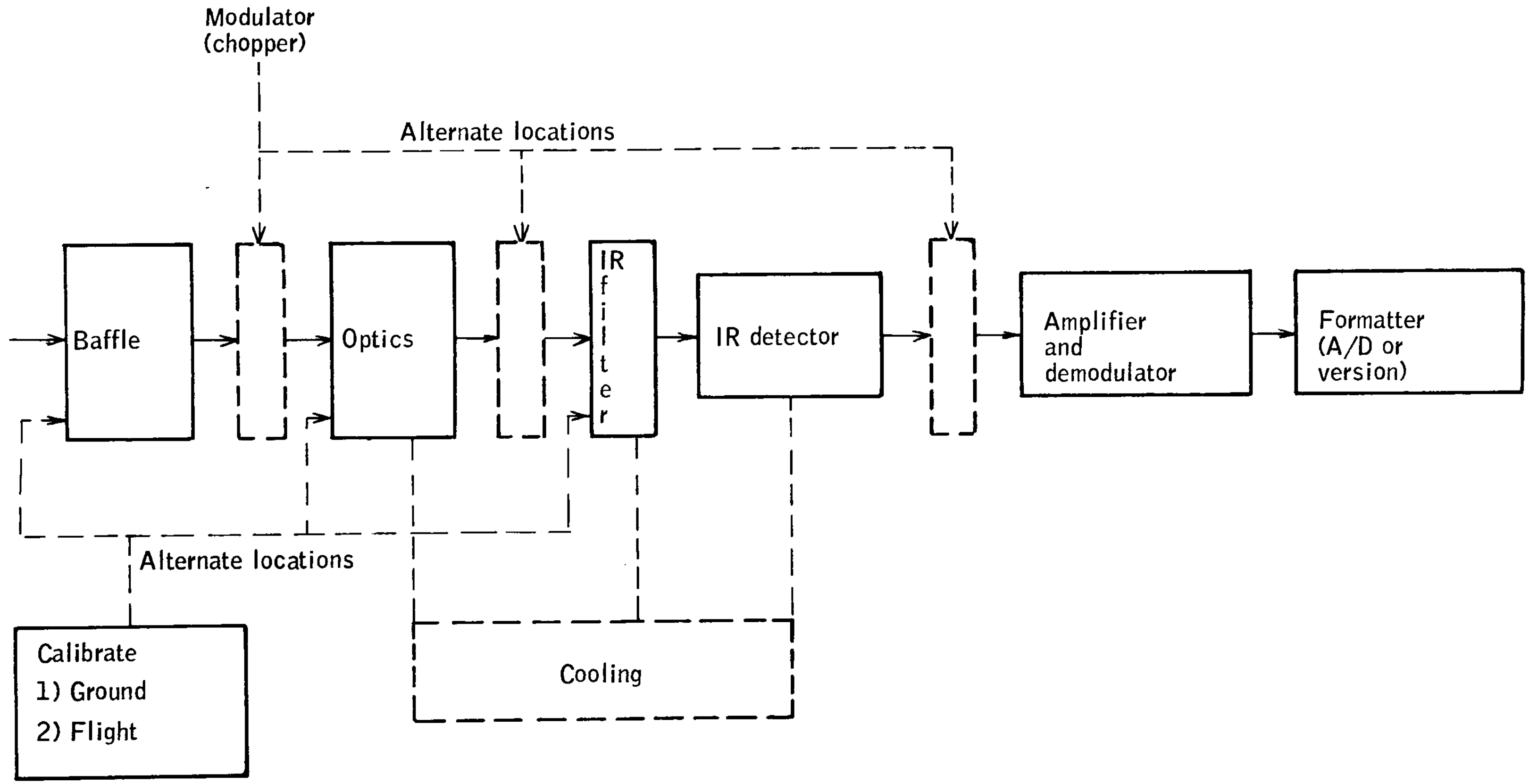


Figure 7. Radiometer Functional Diagram

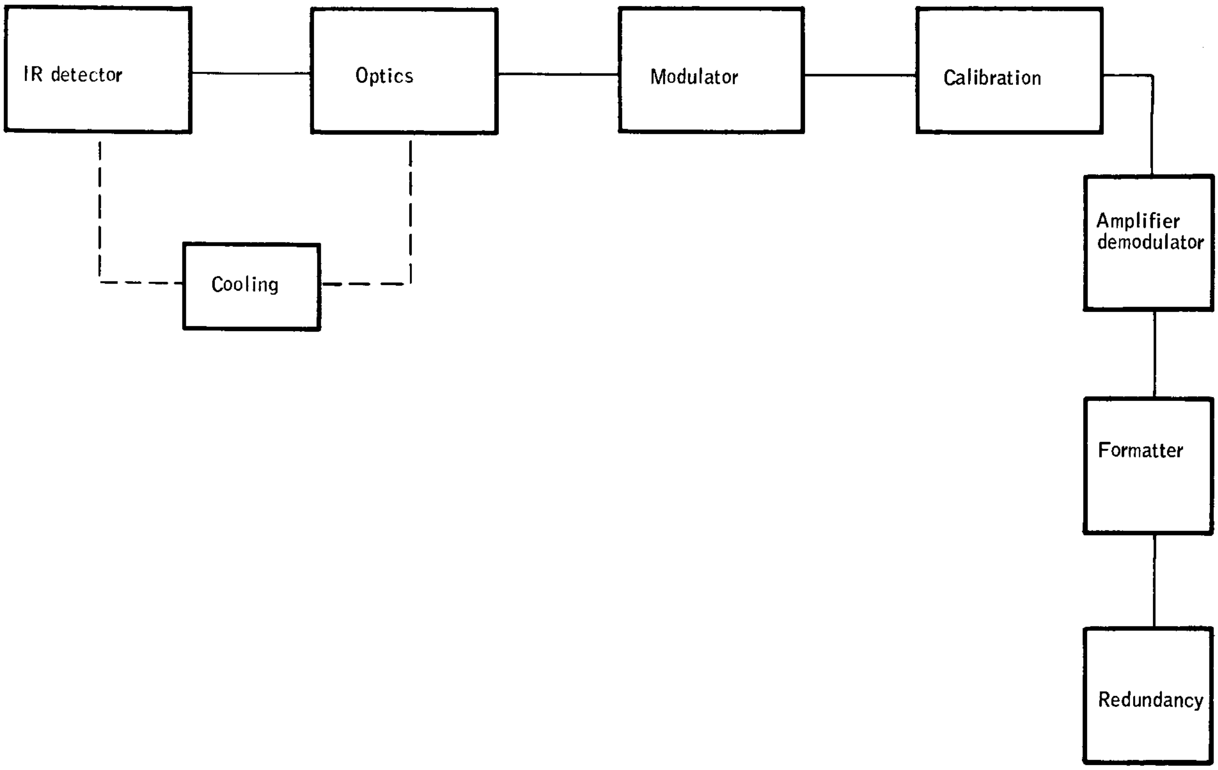


Figure 8. Radiometer Tradeoff Flow

While the basic aperture of the optics was set by (and entered into) the choice of ir detector, the choice of the particular optical configuration to be used had a significant impact on the spacecraft structure. Conversely, the decisions relating to choppers, calibration,and electronics, while of considerable importance related to the effectiveness of the radiometer, had only minor effects on the design of the rest of the system.

The radiometer tradeoffs are summarized in Tables 3 through 13.

TABLE 3. - RADIOMETER TRADEOFFS, INFRARED DETECTOR

Detectors considered	Disadvantages	Advantages
Bolometer	• Large apertures > 1-m diameter required for sensitivity of 0.01 W/m^2-sr • Time constant requires very slow rotational speeds ► Smearing of data ► Vehicle not spin-stabilized • Chopping must be slow	Cooling not needed
Thermopile	Similar to bolometer	Cooling not needed but "cold" junction temperature must be monitored
Photon detectors	Requires cooling and development of cooling system	• High sensitivity results in reasonable aperture size, ($\approx$30-to 40-cm diameter) • Short-time constant ► Removed spin-rate restriction ► Permits wide latitud in chopping frequenc

Photon detectors chosen - performance outweighs disadvantages of cooling.

TABLE 4. - RADIOMETER TRADEOFFS, PHOTO DETECTOR

Detectors considered	Disadvantages	Advantages
Copper-Doped Germanium	• Requires cooling below 10°K; critical D* versus temperature relationship above 10°K	• High D* (highest of those considered) • Off the shelf
Cadmium-Doped Germanium	• Requires cooling to below 20°K; critical D* versus temperature relationship above 20°K	• Second-highest D* • Off the shelf
Mercury-Cadmium-Telluride	• Not presently available in 16-μ design • Lowest D* (however, D* is adequate)	• 40°K to 70°K operating temperature, (smaller amount of coolant) • D* insensitive to temperature

Cadmium-Doped Germanium - Adequate D* and requires smallest amount of coolant of off-the-shelf cells. May later be displaced by Hg:Cd:Te.

TABLE 5. - RADIOMETER TRADEOFFS, COOLING OF DETECTOR

Approaches considered	Disadvantages	Advantages
Radiative cooling	• Apparently cannot achieve sufficiently low temperature in orbit being considered	• Minimum moving parts • Minimum consumption of power
Peltier cooling	• Efficiency too low for practical purposes ► Power consumption excessive ► May not be possible to reach required low temperature in spacecraft environment	• No moving parts
Sterling cycle cooling	• Moving parts • Short life due to pump wearout especially at the lower cell temperatures	• Weight and power requirements compatible with application
Cryogenic cooling	• New technology • Refrigerant boil-off before launch requires special control • Weight of refrigerant and container may be greater than some other approaches	• Unquestionably can produce the range of temperatures needed • No moving parts

Cryogenic cooling chosen - Certainty of reaching the desired low temperatures and probable longer life offsets the new technology risks.

TABLE 6. - RADIOMETER TRADEOFFS, CRYOGEN AND COOLER

Approach considered	Disadvantages	Advantages
Hydrogen (no buffer)	• Requires several months development	• Single material (simpler cryostat) • No 15-μ absorption • Reasonable weight
Neon (no buffer)	• Requires long development • Weight probably high	• Single material • No 15-μ absorption
Neon and CO_2 buffer	• Strong 15-μ absorption by CO_2	• Reasonable development time
Neon and CH_4 buffer	• Slightly greater weight than Neon and CO_2 buffer	• Reasonable development time • Buffer reduces weight • Insignificant 15-μ absorption

Neon with CH_4 buffer chosen as best compromise between cooling efficiency (size and weight), development time, and freedom from 15-μ absorption.

TABLE 7. - RADIOMETER TRADEOFFS, OPTICS

Configurations considered	Disadvantages	Advantages
Refractive	• Difficult to obtain homogeneous material of sufficiently large size • Weight is high for objective lens of 30-to 40-cm diameter	Visible light not passed, reduces heat load on cell and filter
Reflective		Readily fabricated from variety of materials
Hybrid (reflective and refractive)	• Same as refractive	

Reflective optics chosen - lightest, easiest fabricated, and no serious material problems.

TABLE 8. - RADIOMETER TRADEOFFS, REFLECTIVE OPTICAL SYSTEM

Concepts considered	Disadvantages	Advantages
Cassegrainian	• Limited length available if cooler is mounted on spin axis ► Less room for baffling	• Low abberrations • Low observation by secondary
On-axis Newtonian	• Greater aberrations than Cassegrainian • Limited length available for baffle if cooler is mounted on spin axis	• Only one curved surface
Classical Newtonian	• Greater aberrations than Cassegrainian (adequate, however)	• Only one curved surface (only two mirrors total) • Nearly full spacecraft diameter can be used for baffle length
Off-axis parabaloid	• Large aberrations	• Single surface • Nearly full spacecraft diameter available for baffle length

Classical Newtonian chosen as the best compromise between resolution, number of elements, baffling, and location of detector and cooler on spacecraft spin axis.

TABLE 9. - RADIOMETER TRADEOFFS, MODULATOR, CHOPPER

Configurations considered	Disadvantages	Advantages
Optical modulator	• Requires moving parts • May require cooling of chopper	• Includes part or all of optical noise and offset in "zero" level; ∴ minimum correction required for these effects • Includes all cell and amplifier noise and offset in "zero" • Has been qualified in space applications
Detector modulator	• Does not include offset and noise of optics in "zero" level; requires cooling of optics to very low level • May introduce "chopper" noise which will require careful design of amplifier and demodulator	• Simple, mechanically - no moving parts
Post-detector modulator	• Only amplifier noise and offset appear in zero • 1/f noise appears	• No moving parts • Chopping involves only low-voltage switching

Optical modulation (chopping) chosen - advantages of including most or all of noise and offset sources in "zero" level outweighs mechanical problems.

TABLE 10. - RADIOMETER TRADEOFFS, OPTICAL CHOPPER

Concept	Disadvantages	Advantages
Wedge (deflection to "space")	• Difficult design • Chops radiation of optical surfaces	• Continuous "zero" calibration with perfect baffle
Mirror (cell sees itself)	• Chops radiation of optical surfaces	• Small size • "Zero" not related to baffle effectiveness

Mirror approach chosen because design is simpler.

TABLE 11. - RADIOMETER TRADEOFFS, CHOPPER DRIVE

Concept	Disadvantages	Advantages
Motor-driven disc	• Bearing wearout • Gyroscopic torques	• Old technology
Torsional pendulum	• Magnetic moment	• No journal or ball bearings
Push-pull electromagnetic	• 2. 5 kc max.rate	

Torsional pendulum chosen since elimination of bearings offsets need to determine magnetic moment and to include its effect in computing vehicle motion.

TABLE 12. - RADIOMETER TRADEOFFS, FORMAT, RADIOMETER OUTPUT

Formats considered	Disadvantages	Advantages
Analog	• Requires careful control of noise, offset, linearity and bandwidth in spacecraft storage, data link, ground storage, and ground usage of the data	• Minimum amount of electronics in radiometer
Digital	• Requires onboard analog-to-digital converter	• Once converted to digital format, signal is insensitive to noise and other disturbances • Storage and transmission is accomplished simply and with low errors

Digital format chosen - advantages of digital system outweighs addition of A/D converter located in data handling subsystem - minimum power dissipation into "cold plate" of spacecraft and permits added redundancy in data handling system.

TABLE 13. - RADIOMETER TRADEOFFS, RADIOMETER REDUNDANCY

Approach considered	Disadvantages	Advantages
Two radiometers	• Large size and weight • Difficult mounting problems	• All elements are redundant - maximum reliability
Duplicate detectors, electronics, calibrators, and choppers only	• Optics not redundant	• Reduced size and weight • Greater length and volume available for baffling • Smaller cryogen requirement

Duplicate detectors, electronics, calibrators, and choppers chosen. Failure of optics is very unlikely.

The radiometer decisions, which served to define the radiometer were then as follows:

Area	Decision
Detector type	Photon detector
Photon detector type	Cadmium-Doped Germanium
Cooling approach	Cryogenic
Cryogen and buffer	Neon cryogen-CH_4 buffer
Optics type	Reflective
Optical configuration	Classical Newtonian
Modulation type	Optical chopper
Chopper configuration	Moving mirror torsion pendulum
Redundancy	Duplicate detectors, electronics, calibrators, and choppers

Attitude control and attitude determination. -- In arriving at the concepts of the attitude determination and of the attitude control function, it was necessary to perform many of the studies jointly. Consequently, these studies are combined in this summary.

The function of the attitude determination "system" is to determine the attitude of the vehicle and, consequently, of the radiometer during the time when the fied of view of the radiometer is traversing the earth's horizon. This function is accomplished by measuring the relationships between the spacecraft and the stars and sun, reversing these relationships, and later using these data to compute the attitude of the vehicle with respect to time.

The attitude control function is described diagramatically in Figure 9 where it is apparent that this function includes initial orientation of the spacecraft attitude thereafter throughout the useful life of the spacecraft in orbit. This latter function includes all control of the motions of the spacecraft with respect to the orbit plane including coning, spin rate, nutation and precession.

Background: To provide a suitable context for understanding the tradeoffs that were performed, the following background discussion is provided.

The desired vehicle motions are:

- Spin at constant rate about an axis perpendicular to the orbit plane
- Translation of the spin axis along the orbit

The actual motions however will consist of the following components:

- Spin about an axis stable in inertial space
- Nutation and coning
- Precession of the spin axis due to disturbing torques
- Translation along the orbit
- Slow down (decay) of the spin rate

The attitude control consequently is needed to provide the following functions:

- Damp or limit the nutation and coning
- Overcome the torque-induced precessions
- Add precession to make the spin axis perpendicular to the orbit plane
- Overcome the spin rate decay

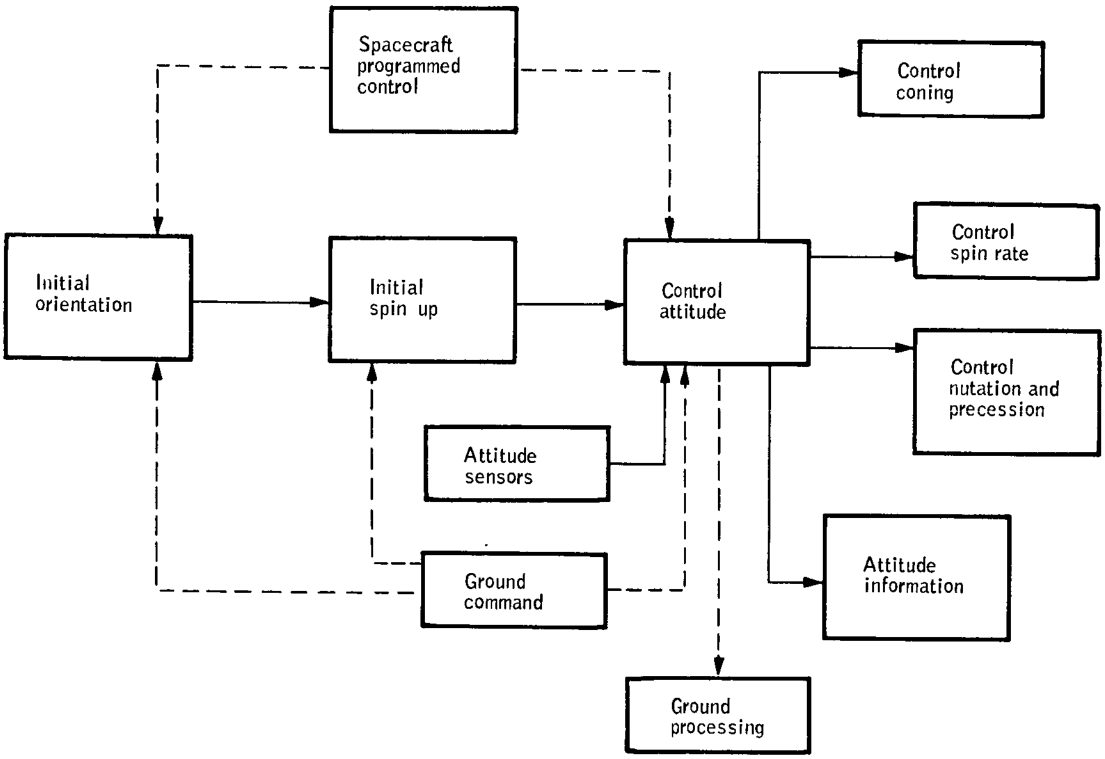


Figure 9. Attitude Control Functional Diagram

Conversely the attitude determination function is concerned with:

- Determining the orientation of the spacecraft with respect to the earth's coordinate system

Tradeoff studies: The studies were performed following the flow diagram of Figure 10. The tradeoff studies moved generally from left to right in this diagram with numerous iterations. To provide a basis for determining the preferred approaches to attitude control and to attitude determination, the simulation study described by the four blocks on the left side of Figure 10 were performed. That is, the dynamic characteristics of candidate spacecraft configurations were first determined. The disturbing torques which would affect the spacecraft motions were then estimated. Finally a computer simulation was performed to determine the spacecraft motions under both "open loop" (no attitude control) and "closed loop" conditions. The "open loop" simulations showed that the spacecraft motions were highly predictable and that the most significant departures from ideal spacecraft motions were caused by:

- The magnetic moments of the spacecraft cause spin axis precession due to coupling with the earth's magnetic field
- The earth's magnetic field induces eddy currents into the spacecraft causing spin-rate decay
- Precession of the orbit

Under "closed loop" conditions it is apparent that the motions would be highly dependent upon the characteristics of the various attitude control loops and probably would not be as readily predictable as under "open loop" conditions. Using the information from these simulations, the attitude control and attitude determination concepts were traded off essentially simultaneously.

The most significant tradeoffs in the attitude control area were concerned with:

- Intermittent versus continuous control
- Computation location
- Respin and reorientation concept

The most significant decision in connection with the attitude control function was the selection of a combination of continuous and intermittent control of spin rate, yaw, and roll. The simulations of spacecraft motions showed that in the "open loop" configuration, the motions of the spacecraft were highly predictable. Moreover, if the spacecraft magnetic moments were approximately trimmed out by suitably excited "torquer" coils, the motions of the spacecraft would be sufficiently bounded so that precession of the orbit would be the dominant motion of the spacecraft with respect to the orbit. Since it

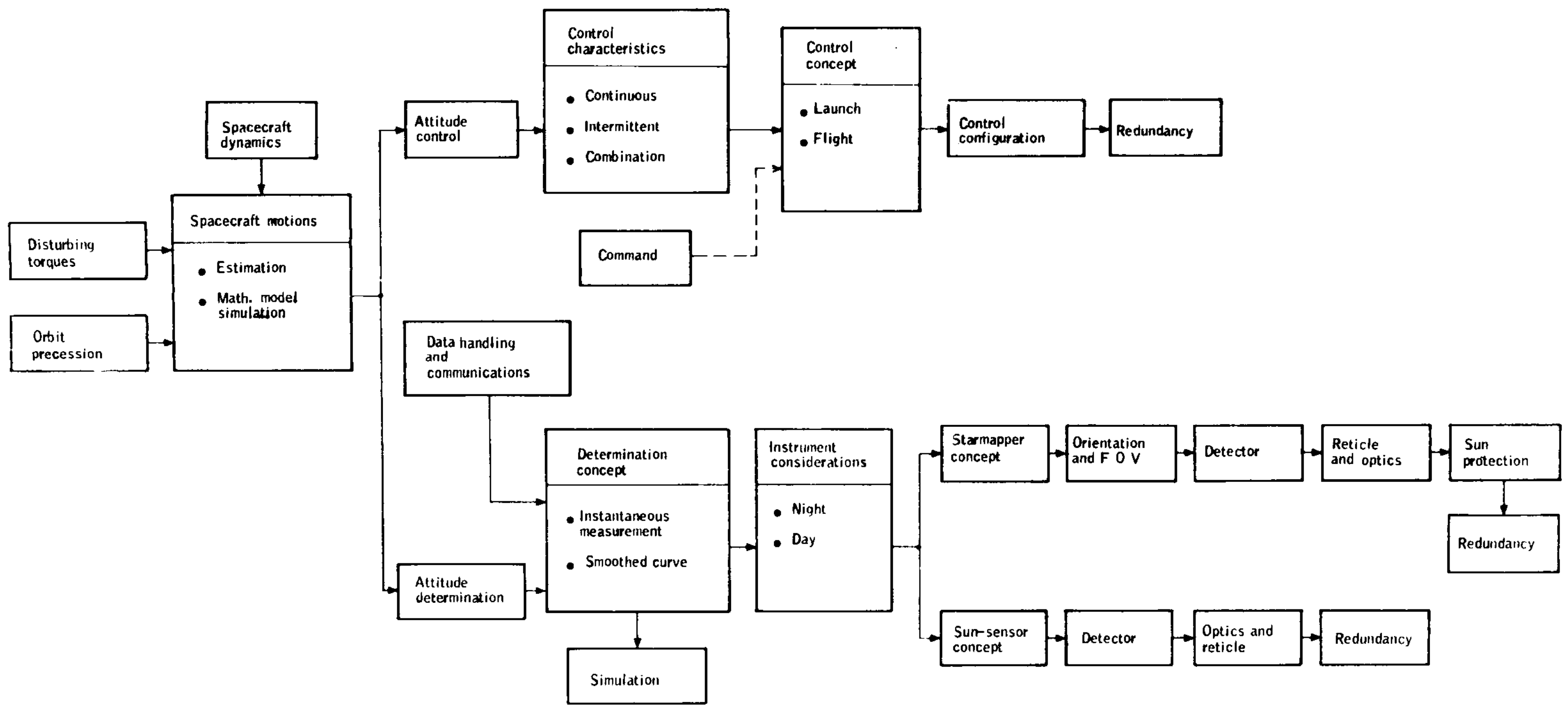


Figure 10. Attitude Determination and Control Study Flow Diagram

would require several days before orbit precession would make reorientation of the spacecraft (with respect to the orbit) necessary, and this reorientation could be accomplished in less than one orbit period, the spacecraft would operate nearly all of the time in the highly predictable "open loop" configuration. This, in turn, resulted in significant simplifications in the attitude determination and data handling functions.

Since it was not necessary to perform spacecraft reorientation and respin frequently, it was then possible to greatly simplify the on-board portion of the attitude control system by performing the necessary computations on the ground. The simplification of the spacecraft systems more than offset the very small additional communications load.

The selection of magnetic torquing for respin and reorientation of the spacecraft with respect to the orbit caused this function to be accomplished in a highly reliable manner uncomplicated by moving parts or other "wearout" type problems.

The most significant tradeoffs related to attitude determination were:

- Instantaneous versus smoothed continuous attitude
- Starmapper orientation
- Inclusion of a sun sensor

The most significant result of the attitude determination function tradeoffs was the decision to generate the spacecraft attitude as a smoothed function of time. This approach could only be considered after it was determined that the "open loop" motions of the spacecraft were highly predictable and that the spacecraft was actually "open loop" most of the time. The results of this decision were increased freedom in the design of the attitude sensors, and reduction of the complexity of the on-board data handling system.

The decision to orient the axis of the starmapper perpendicular to the spin axis (made practical by the predictable nature of the vehicle motions) simplified the starmapper design by reducing the complexity of the starmapper baffle design. Moreover, since the star sightings were always taken from the portion of the celestial sphere very close to the orbit plane, the effect of star sighting errors upon the determination of spacecraft attitude is minimized. In addition, the alignment of starmapper axis with respect to the radiometer axis could be more accurately determined and maintained since, in this configuration, the axes of these two instruments are parallel.

Since spacecraft motions are "well-behaved", it is not necessary to obtain starmapper data throughout the orbit if a sun sensor is available. Consequently, the addition of a sun sensor can be used to reduce the stringency of the starmapper design requirements by removing the necessity for obtaining star data during the "sunny" portion of the orbit. Not only is the difficulty of the starmapper design reduced, but the volume is also minimized.

The attitude control and attitude determination tradeoff studies are summarized in Tables 14 through 27.

TABLE 14.- ATTITUDE CONTROL TRADEOFFS, SEPARATE, SPIN UP, AND ALIGN TO ORBIT

Concepts	Disadvantages	Advantages
Separate, (unaligned), then spin up and align	• Most complex spacecraft • Battery power used during spin-up and long alignment - will require one or more battery charging orbits before precise alignment and data taking can be accomplished	• Simplest booster
Separate (aligned), then spin up	• Tip-off angular rates require either: ► Rapid spin up ► or, large post-spin-up attitude corrections	• Shorter time between injection and beginning of data taking • Probably no special battery recharge orbits required
Align, spin up, then separate	• Booster must include ability to align to orbit and hold alignment during spin up • Thor-Delta presently does not have spin-up capability, (being planned for time period of interest)	• Minimum delay between injection and data taking • Simplest space craft • No special battery recharge orbits

Align, spin up and then separate chosen as preferred concept. Booster capability is expected to be available, spacecraft is simplest and mission success is enhanced. Other modes could be used as alternate back-ups using normal torquing modes for spin up and alignment to orbit.

TABLE 15.-ATTITUDE CONTROL TRADEOFFS, CONING DAMPER

Concepts	Disadvantages	Advantages
Mercury Ring	• Difficult to clamp (remove damping)	• Simple • Proven principle
Pendulum	• Location on spin axis is preferred	• Easily clamped • Proven principle
Ball in gas-filled tube	• Clamping difficult but possible	• Simple • Proven principle

Selection not critical - anyone can be selected depending on space and location available on the spacecraft.

TABLE 16. - ATTITUDE CONTROL TRADEOFFS, ROLL, YAW, AND SPIN CONTROL

Concepts	Disadvantages	Advantages
Continuous	• Spacecraft motions dependent upon characteristics of control system as well as past and present disturbances. Motions are difficult to predict • Continuous control difficult within practical limitations of power, weight and orbit precession over one-year period	• Attitude error is continuously forced to desired attitude, within errors of sensors and control loops
Intermittent	• Attitude departs from "nominal" attitude during open-loop periods	• Motions are highly predictable during period when control is not applied (open loop) • Compatible with use of magnetic torquing (coupling with earth's magnetic field)
Combination • Magnetic moment trim - continuous • Re-erection, realign to orbit (precess), and respin - ntermittent	• Continuous power drain for trim • Requires controls for both the continuous and the intermittent loops	• Permits "exact" correction of most significant disturbance torques - the residual magnetic moment • Minimizes "drifts" of the spin axis • Maximizes time required between re-erections

Combination of continuous magnetic moment trim and intermittent torquing to accomplish re-erection, realign to orbit and re-spin. Added complexity is offset by improved stability of spacecraft motions in "open loop" periods which simplified attitude determination

TABLE 17.-ATTITUDE CONTROL TRADEOFFS, ROLL, YAW, TORQUING

Concepts	Disadvantages	Advantages
Momentum interchange • Inertia wheels • Control moment gyros	• Continuous power consumption (or wheel rundown) • Complicated spacecraft motions due to cross-couplings • Bearing wear-out • Requires (practically) additional torquing source to take care of long-time effects (such as orbit precession)	• Continuous control possible • Can provide damping of undesired motions
Mass explusion • Reaction jets	• Moving parts-wear out • Gas leakage generates unknown torques • Possible mass shifts upon depletion of gas supply	• Quick correction of attitude errors • Weight compatible with one-year life
Magnetic	• Continuous control impossible • Requires commutation to match earth's field • Limited practical torque capability	• No moving parts • Can compensate for residual magnetic moment • No rundown or similar limitation to life • Spacecraft motions highly predictable during no-torque period

Magnetic torquing chosen because:

- Simplest - no moving parts
- Most reliable
- Highly predictable spacecraft motions
- Compensation for residual magnetic moments possible

TABLE 18.- ATTITUDE CONTROL TRADEOFFS, SPIN CONTROL, RE-SPIN

Concepts	Disadvantages	Advantages

Reaction jet	• Includes moving parts:
	  • Wear out
	  • Freeze up
	• Leaking gas may cause:
	  • Over-spin
	  • Erratic spin rates
	  • Shortened mission
	• Mass changes during mission
		• Fast re-spin
		• No restriction on portion of orbit used for re-spin

Magnetic torquer	• Commutation to match earth's field • Limited re-spin-up rate	• No moving parts • Compatible with magnetic attitude control

Magnetic torquer chosen because:

- No moving parts
- Spin-up rate compatible with attitude control period
- Orbit restriction not serious

TABLE 19. -ATTITUDE CONTROL TRADEOFFS, ROLL-YAW SENSORS

Concepts	Disadvantages	Advantages
Gyro	• Drift rate unacceptable	• Can be used as damping sensor • Can sense motions having frequencies higher than spin rate
Star tracker	• Design difficult for rotating vehicle • Moving parts - short life • Star reacquisition several times per year	• High accuracy

Sun sensor	• Does not measure attitude directly. Must be corrected for:
	  • Seasonal sun angle variations
	  • Departure of orbit from sun-synchronous
	• Requires accurate knowledge of spin rate
	• Occulted during part of orbit
		• Simple - no moving parts

Earth's horizon sensor	• Requires knowledge of spin rate	• Simple - no moving parts
V- head	• Measures attitude error in only one axis at a time (Restrictions on tolerable magnitudes and periods of attitude errors)	• Most direct method of attitude measurement • Effective throughout orbit • Measures local vertical directly

V - head horizon sensor chosen because:

- Simple - no moving parts
- Effective throughout orbit
- Most direct method of measurement

(Restrictions on spacecraft motions and required knowledge compatible with other system functions)

TABLE 20. - ATTITUDE CONTROL TRADEOFFS, ATTITUDE AND SPIN CORRECTION, COMPUTATION LOCATION

Concepts	Disadvantages	Advantages
On-board computation	• Complexity of on-board computation facility.	• Self-contained (does not require command link)
	• Limited flexibility to adapt to change of flight plans	• Continuous control may be possible
On-ground computation	• Requires command link	• Simplest spacecraft
	• Continuous control not possible	• Greatest flexibility for flight mode changes
	• Requires ground link to computer periodically	
	• Requires generation of torquing commands on ground	

On-ground computation chosen because:

- Spacecraft is simpler and more reliable
- Greater flexibility to adapt to needed changes in flight

(Command link required for other reasons, ground link, ground computer and data link needed only at intervals of several days.)

TABLE 21. - ATTITUDE DETERMINATION TRADEOFFS, CONCEPTS

Concepts	Disadvantages	Advantages
Instantaneous measurement* * (Full set of attitude data is obtained for each radiance profile measured.)	• Requires largest amount of data storage (memory) on board the spacecraft • Instruments must be capable of seeing a sufficiently large number of separated celestial objects during each measurement ► Difficult instrument design for daylight measurement	• Spacecraft attitude measurement does not require knowledge of vehicle motions (provided the spin rate is "constant enough" and the other angular motions are "small enough")
Smoothed curve ** ** (A number of sets of data, but not all possible sets, are obtained each orbit. A curve is fitted to these sets and the attitude for the individual radiance profiles is obtained from this curve.)	• Smoothed curve must be sufficiently good fit to actual vehicle motions ► All significant motions must be accounted for in model ► Sufficient data, properly distributed	• Relatively large motions can be accommodated provided that the significant motion components can be considered "time stationary" for the period to which the curve is fitted • Minimizes the required spacecraft data storage • Less difficult daylight instrument design

Smoothed curve chosen because:

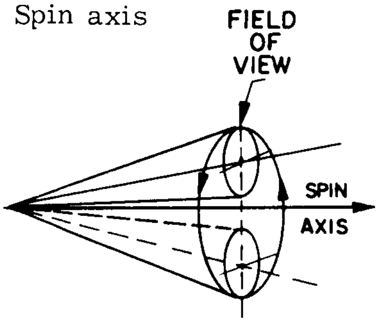
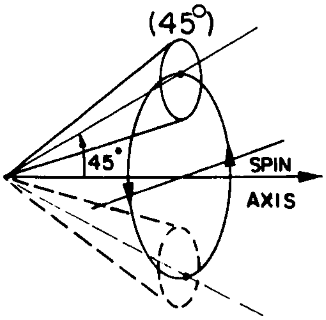
- Simpler instrument design
- Required memory size is smaller

TABLE 22. - ATTITUDE DETERMINATION TRADEOFFS, INSTRUMENT CONSIDERATIONS

Concepts	Disadvantages	Advantages
Star tracker	• Requires at least two trackers (or equivalent)	• Continuous attitude information in real time
	• Requires moving parts ► Wear out ► Torque reactions on vehicle if used	• Daylight design is feasible
	• Photomultiplier must be protected from overload	
Starmapper	• Baffling is difficult during daylight portion of orbit	• Attitude can be obtained from one sensor
	• Photomultiplier must be protected from overload	• No moving parts
	• Complex data reduction	
	• Attitude not available in real time	
Sun sensor	• Insensitive to spacecraft motions about sun-line because of single reference point	• No sunlight baffle problems
	• Functions only in daylight portion of orbit	• No moving parts

Combination of sun sensor and starmapper chosen as best compromise which eliminates moving parts, provides adequate day and night sensing, and is compatible with the present state of the baffle design art. (Real time attitude information is not required.)

TABLE 23. - ATTITUDE DETERMINATION TRADEOFFS, STARMAPPER ORIENTATION AND FIELD OF VIEW

Orientation & field of view	Disadvantages	Advantages
Rim (FOV perpendicular to spin axis.)	• Daylight baffling problem is very difficult • Photomultiplier must be protected from illumination by the daylit earth (also probably the moon) and the sun	• Smallest F.O.V. required because maximum portion of celestial sphere is swept • Greatest tolerance to instrument angular errors • Simple boresighting alignment with radiometer
Spin axis 	• FOV requirement is greatest • Large F OV requires careful baffling against earthshine • Moon will be within FOV for a portion of 2-3 days/month (Data taking impossible.) • Most critical to instrument angular errors (including alignment errors.)	• Does not intersect daylight earth
Canted axis 	• Intercepts daylight earth ► Photomultiplier must be protected during intercept	• Sun baffling probably possible within present state of the baffle art (Some daylight capability) • Tolerance to angular instrument errors (including alignment) within 40% of that of "rim" orientation

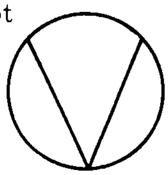
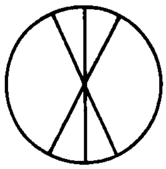
Rim configuration chosen due to greater tolerance to inter and intra-instrument angular errors - daylight capability not needed.

TABLE 24. - ATTITUDE DETERMINATION TRADEOFFS, STARMAPPER DETECTORS

Detectors	Disadvantages	Advantages
Silicon photo diode	• Only available in small sizes - consequently a mosaic must be used - added electronic complication • Lower sensitivity than photomultiplier - large optics required • No required previous usage in star mappers	• Insensitive to light input overload • Photovoltaic - no bias required
Cadmium sulfide	• Extremely low detectivity. (long time constant) • No previous usage in starmappers	• Insensitive to light input overloads • Uses low voltage bias supply
Photomultiplier	• Sensitive to light input overload • Requires high-voltage power supply for multipliers • Variations in responsivity over face of photo cathode • Physically larger than semiconductor detectors	• Extremely sensitive ► Modest optics sizes ► Large magnitude star capability ► Single-channel electronics • Extensive previous usage in starmapper applications

Photomultiplier chosen because minimum optics size and extensive history of past usage offsets overload sensitivity.

TABLE 25. - ATTITUDE DETERMINATION TRADEOFFS, STARMAPPER RETICLE

Reticle configuration	Disadvantages	Advantages
V-slot 	• Difficult associated optical design because of radial dissymetry • Difficult to separate stellar ambiguities	• Simplest slit configuration
Triple-X slot 	• Requires either coding of slots to separate upper and lower set of slit crossings or use of double detectors	• Simpler associated optical system since radial symmetry is provided • Considerably more effective than V-slot in separating stellar ambiguities

Triple-X slot selected because advantages of simple optical design and effectiveness in separating stellar ambiguities offset slightly greater difficulty in fabricating reticle

TABLE 26. - ATTITUDE DETERMINATION TRADEOFFS, SUN SENSOR

Concept	Disadvantages	Advantages
Shadow ► Measure ► Mean direction of sun rays	• Requires moving parts for best accuracy • Requires very good balance of detector - amplifier combination for passive configuration • Requires collimator for best accuracy	• Simple • Potentially very accurate
V-slit ► Measure crossings of sun's edge.	• Accuracy depends upon sharpness of sun's edge and knowledge of sun's shape • Requires focussing of sun's image on slit	• Simple • Potentially very accurate • Does not require moving parts

V-slit chosen because:

- No moving parts required
- Sun's edge and shape is apparently sufficiently well defined over a narrow spectral band.

TABLE 27. - ATTITUDE DETERMINATION TRADEOFFS, SUN SENSOR DETECTOR CONFIGURATION

Concept	Disadvantages	Advantages
Detector adjacent to slit	• Does not permit detector redundancy • Requires uniform sensitivity over cell surface	• Simple configuration
Detector displaced from slit	• Requires optical elements between slit and cell(s) • Most complex	• Permits detector redundancy • Less sensitive to detector surface uniformity
Integrating sphere	• Complexity intermediate between adjacent and displaced locations	• Insensitive to detector surface uniformity • Permits detector redundancy

Integrating sphere chosen because it is the simplest scheme which permits detector redundancy and is insensitive to detector surface uniformity.

The major attitude control and attitude determination decisions were then as follows:

Attitude control

Area	Decision
Initial sequence	Align, spin up, and then separate
Coning damper	Any of several concepts suitable
Roll, yaw and spin control	Continuous magnetic trim, intermittent magnetic torquing
Roll/yaw torquing	Magnetic
Re-spin	Magnetic
Roll/yaw sensors	V-head horizon sensor
Computation location	On ground

Attitude Determination

Area	Decision
Determination concept	Smoothed curve
Basic sensor	Starmapper and sun sensor
Starmapper orientation	Rim (perpendicular to spin axis)
Starmapper detector	Photo multiplier
Starmapper reticle	Triple-X slot
Sun sensor concept	V-slit
Sun sensor detector configuration	Integrating sphere

Data handling. -- As shown in Figure 11, the data handling subsystem collects data generated by the various devices on board the spacecraft, transforms the data into the proper format for temporary storage in the memory, stores the data, and delivers the stored data with the proper format to the communications subsystem. In addition, the data handling subfunction generates the necessary timing and internal control signals, and verifies and decodes commands delivered to it from the command receiver portion of the communications system.

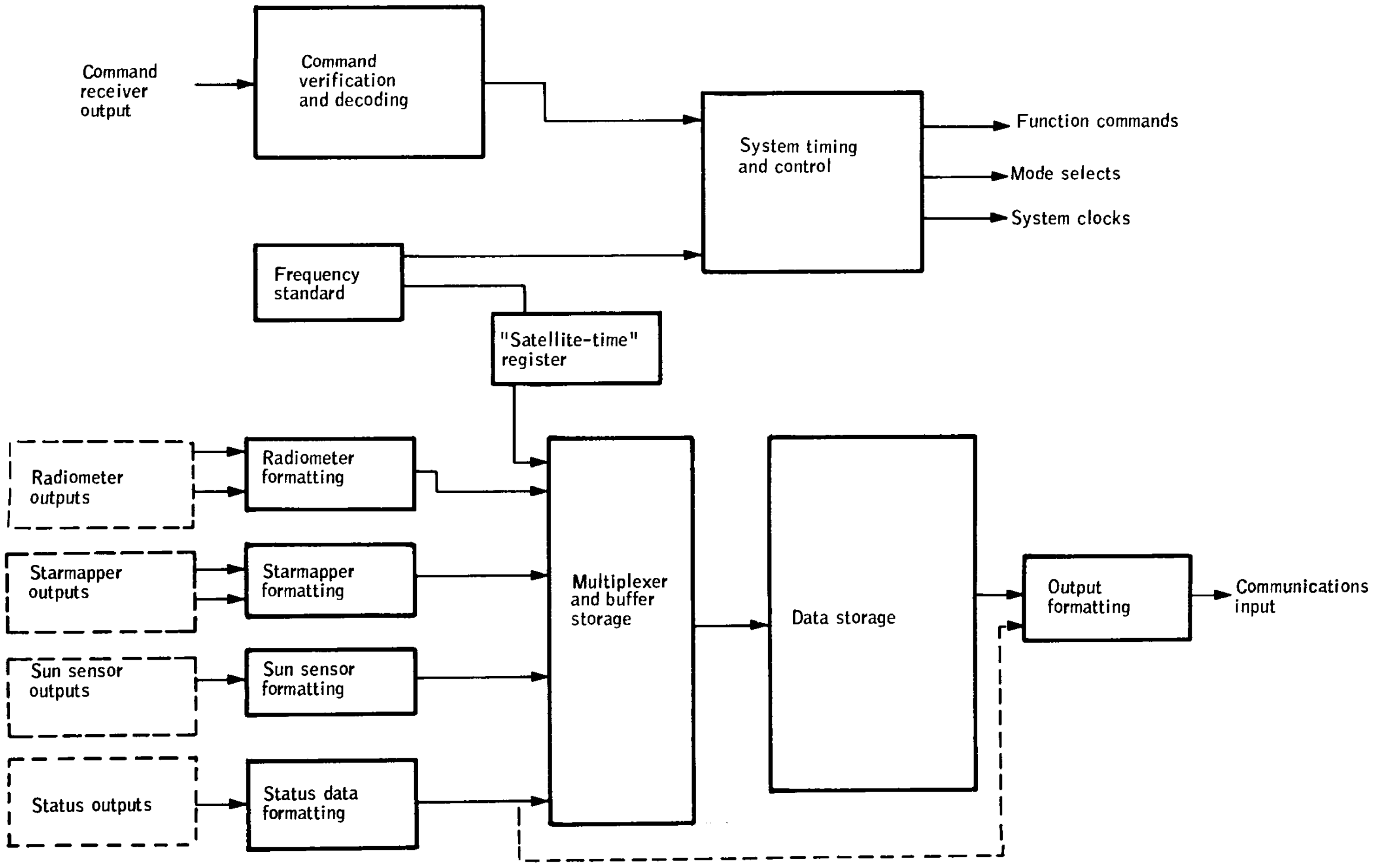


Figure 11. Data Handling Functional Diagram

The tradeoff studies in this subfunction area followed the general pattern outlined in Figure 12. Many of the data handling tradeoffs were strongly influenced by other parts of the system, especially by the research package (radiometer, starmapper and sun sensor). There were, however, several areas in which the bases for decision were largely within the data handling area alone. The most significant of these are listed in the order of their significance to the total system concept:

- Memory type
- Time reference correlation
- Sampling control

Careful investigation of the amount of data required to be stored between "dumps" to the telemetry system disclosed that the memory requirements were compatible with existing "solid-state" memory technology. Consequently, this memory type was chosen in preference to magnetic tape or magnetic drum storage techniques, thereby eliminating any memory "wearout" problems or torque reactions upon the spacecraft due to rotating components.

A frequency monitoring technique was chosen to provide the necessary time reference correlation between the spacecraft and ground stations. This relatively simple technique involves transmitting a "time pulse" to the ground periodically during data dump. Simultaneously with transmission, a "time count" is temporarily stored and then transmitted to the ground at a convenient interval during the data dump. Upon receipt of the "time pulse", the ground station similarly picks and stores a "time count" from the ground station's time register. The airborne-derived "time count" is then later compared with the ground-derived "time count" to determine an incremental time correction factor. This concept is considerably simpler than other concepts that potentially could be used for this purpose.

Analysis of the problem of providing on-board sampling of data to be stored from the research package instruments disclosed that addition of the necessary controls could permit the memory size to be kept within practical bounds. Moreover, by making the sampling control variable (i.e., having various modes which could be selected by ground command), many advantages in the data collection could be realized such as: obtaining an optimum data distribution over all space/time cells, and providing ability to accomodate special situations during the life of the mission when it would be desirable to alter the distribution of samples.

The data handling system tradeoffs are summarized in Tables 28 through 36.

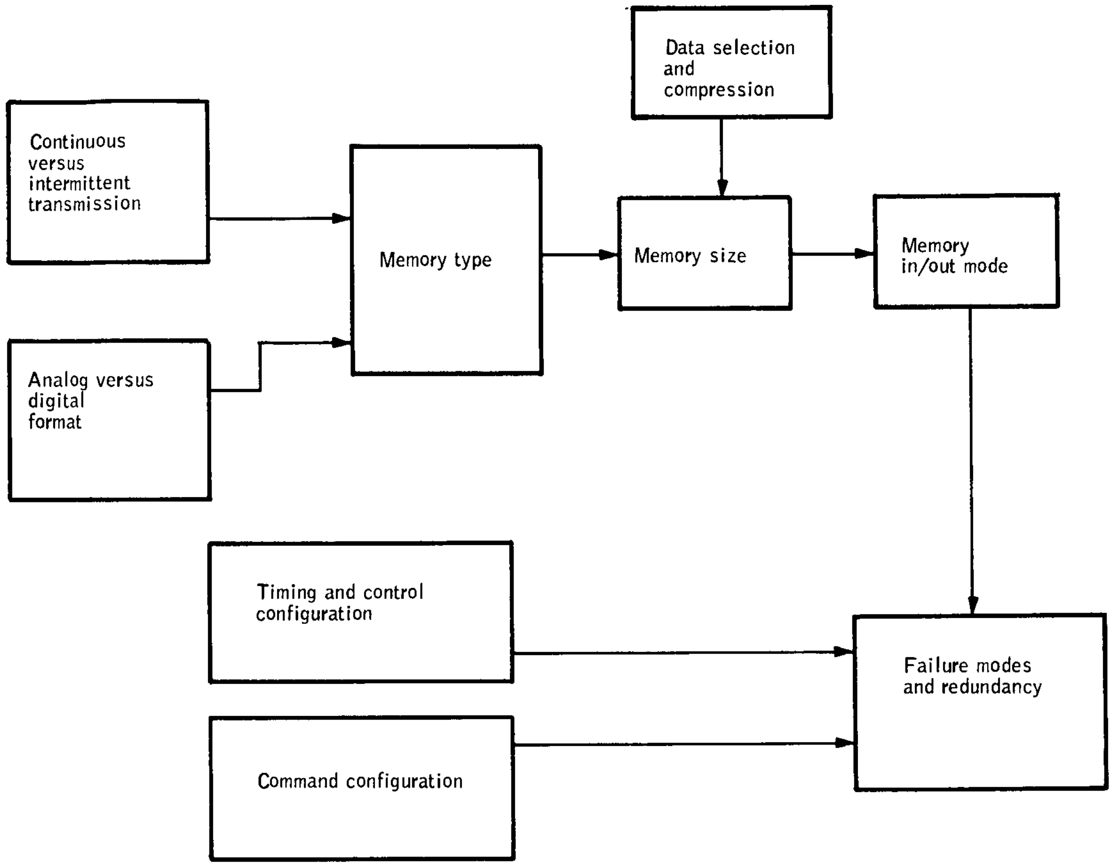


Figure 12. Data Handling Flow Diagram

TABLE 28. - DATA HANDLING TRADEOFFS, DATA TRANSMISSION

Concepts Considered	Disadvantages	Advantages
Continuous transmission	• Requires world-wide, continuous telemetry coverage, (unavailable now) • Continuous transmitter operation ► Greater power consumption ► Ties up rf spectrum	• "Real time" data transfer • No memory on board, (less complex)
Intermittent transmission	• Requires onboard memory ► Greater complexity	• Compatible with STADAN system • Minimizes tie-up of rf spectrum • Does not require continuous, world-wide telemetry coverage

Intermittent transmission selected because compatibility with STADAN and maximum sharing of available rf spectrum offsets added spacecraft complexity.

TABLE 29. - DATA HANDLING TRADEOFFS, DATA FORMAT

Concepts	Disadvantages	Advantages
Analog	• Requires low-distortion and low noise in all elements in data chain including: ► Storage ► Telemetry transmitter ► Ground telemetry receiver ► Ground recording	• Eliminates onboard analog-to-digital converters
Digital	• Requires onboard analog-to-digital converters	• Once digitized, data is insensitive to linearity or noise in subsequent handling • Minimizes bandwidth requirements due to noncontinuous data output characteristics

Digital format selected because insensitivity to disturbances simplifies design of the entire data collecting chain and more than offsets the addition of the necessary analog-to-digital converters.

TABLE 30. - DATA HANDLING TRADEOFFS, MEMORY TYPE

Concepts	Disadvantages	Advantages
Magnetic tape or drum	• Significant reaction torques if start-stop operation is used • Wearout probable during one year	• Required capacity is readily achieved • Short development time • Low cost
Solid state (no moving parts)	• Present high-capacity devices are expensive • Longer development time required for largest capacities	• No moving parts; eliminates wear-out problem • Low-power consumption • Minimum input buffering required

Solid-state memory chosen because longer life, absence of reaction torques, simpler buffering, and low power offset higher cost and development time.

TABLE 31. - DATA HANDLING TRADEOFFS, SOLID-STATE MEMORY TYPE

Concepts	Disadvantages	Advantages
Laminated ferrite	• Presently in early development stage	• Batch process (potentially low cost) • Small size and weight • Potential high reliability
Plated wire	• Presently in development	• Batch process (potentially low cost) • Potential high reliability
Thin films	• Difficult noise problems in larger sizes	• Batch process (potentially low cost) • Potential high reliability
Ferrite cores	• Discrete elements tend to keep cost high	• Many systems in existence • Minimum development time

Ferrite core memory selected because minimum development time required, program risk is smallest, and present costs are equal or less than other approaches.

TABLE 32. - DATA HANDLING TRADEOFFS, MEMORY SIZE, DATA COMPRESSION

Concepts	Disadvantages	Advantages
Full data (no compression)	• Maximum memory size	• Minimum errors (individual errors do not propagate)
Radiometer data increments	• Individual errors in compression process propagate into all subsequent increments	• Minimizes memory size
Radiometer profile averaging	• Requires complete onboard attitude determination • May introduce errors due to averaging process	• Minimizes memory size • Filters some of noise in raw profile data

Full data selected since potential memory size reduction does not offset potential data errors.

TABLE 33. - DATA HANDLING TRADEOFFS, MEMORY IN/OUT MODE

Concepts	Disadvantages	Advantages
Sequential (interrupt storing while dumping data to telemeter)	• Data lost during dump, (loss is concentrated) at specific locations)	• Simpler buffering and timing of memory • Simpler control circuitry
Simultaneous (continue data collection during dumping)	• Memory and control circuitry more complicated	• No limitation on data collection; improved completeness of data

Simultaneous configuration preferred, but not critical to data requirements. It should be included if further design shows a simple mechanization.

TABLE 34. - DATA HANDLING TRADEOFFS, COMMAND CONFIGURATION

Concepts	Disadvantages	Advantages
Tone-command	• Limited number of on-off commands available • More complicated on-board control required to extend capacity to size needed	• Simplest system - inherent reliability is greatest for basic system (not necessarily most reliable when capacity is extended by onboard controls
Tone-digital command	• More complicated than basic tone-command system	• Command capacity adequate for program • Capability available at all command stations
PCM	• More complicated than other systems • Not available at all command stations	• Very large capacity • Very flexible

The tone-digital system was chosen because command capacity is: adequate (including growth), available at all command stations, less complex than that of PCM system or extended tone-command system.

TABLE 35. - DATA HANDLING TRADEOFFS, TIME REFERENCE CORRELATION

Concepts	Disadvantages	Advantages
Precise frequency standard	• Size, weight and power is large compared to other approaches • Most expensive	• All data obtained against true time; no correction routines or clock reset required if performance is maintained throughout flight
Synchronized oscillator	• Requires continuous comparison with a ground standard ► Coverage not presently available ► Disturbed by doppler and other propagation effects	• Can tolerate oscillator instability if synchronization is continuous
Frequency monitor	• Requires periodic transmission of spacecraft time for comparison with ground • Single readings affected by doppler and other propagation effects	• Can use conventional crystal-controlled oscillator • Simplest • Lowest cost

Frequency monitor approach chosen since it is simplest and lowest cost approach and propagation effects are tolerable.

TABLE 36. - DATA HANDLING TRADEOFFS, SAMPLING CONTROL

Concepts	Disadvantages	Advantages
Store all samples	• Excessively-large memory required; major development item • Excess data collected does not represent equal redundancy in all space/time cells	• All possible data collected
Store constant percentage of samples	• Data collected does not represent equal redundancy in all space/time cells	• Practical memory size
Constant latitude (store percentage of samples according to a constant function of latitude)	• Requires knowledge of orbit period and polar crossing times ► Ground processing and command to spacecraft • Lacks flexibility for seasonal effects, concentrated data taking, etc.	• Practical memory size • Optimum redundancy distribution over space/time cells
Variable (adjustable by ground command, normal mode of operation is above listed constant latitude)	• Requires determining orbit period and polar crossing times on ground • Requires logic necessary to provide the variable timing	• Practical memory size • Optimum data redundancy over all time/space cells • Can accommodate special situations: ► Seasonal effects ► Lost transmissions ► Concentrated data taking (such as polar warming)

Variable approach selected because maximum capability is provided for obtaining useful data with a practical memory size. Experiment flexibility gained offsets the added logic circuitry.

The data handling subsystem decision, which served to define the salient features are then as follows:

Area	Decision
Data transmission concept	Intermittent transmission
Data format	Digital
Memory type	Solid-state
Solid-state type	Ferrite core
Data compression	Full data
Memory in/out mode	Simultaneous (preferred)
Command configuration	Tone-digital
Time reference correlation	Frequency monitor
Sampling control	Variable (adjustable by ground command)

Communications. -- As shown in Figure 13, the communications subfunction provides the necessary two-way radio communication between the spacecraft and ground stations. Specifically the telemetry transmission, command reception, position determination transponder, and acquisition functions are provided by the communications subfunction.

The tradeoff studies in this subfunction area followed the general pattern in Figure 14. The area available for tradeoffs was very much prescribed by the requirements to interface with the STADAN system as it is envisioned to be in the mission time period. There were, however, a number of decisions made within these constraints, which were significant with respect to the total system configuration. The most outstanding tradeoffs made in this context were:

- S-band tracking beacon
- Antenna locations
- Redundancy and alternate modes

A careful review of the possibility of using vhf for the range and range-rate transponder function was conducted with NASA Goddard. Vhf was especially attractive since the telemetry and command functions could be accomplished in this region of the rf spectrum, and use of vhf would simplify the spacecraft, reduce the number of antennas, and make possible additional alternate "redundant" configurations. However, it was determined that the vhf system

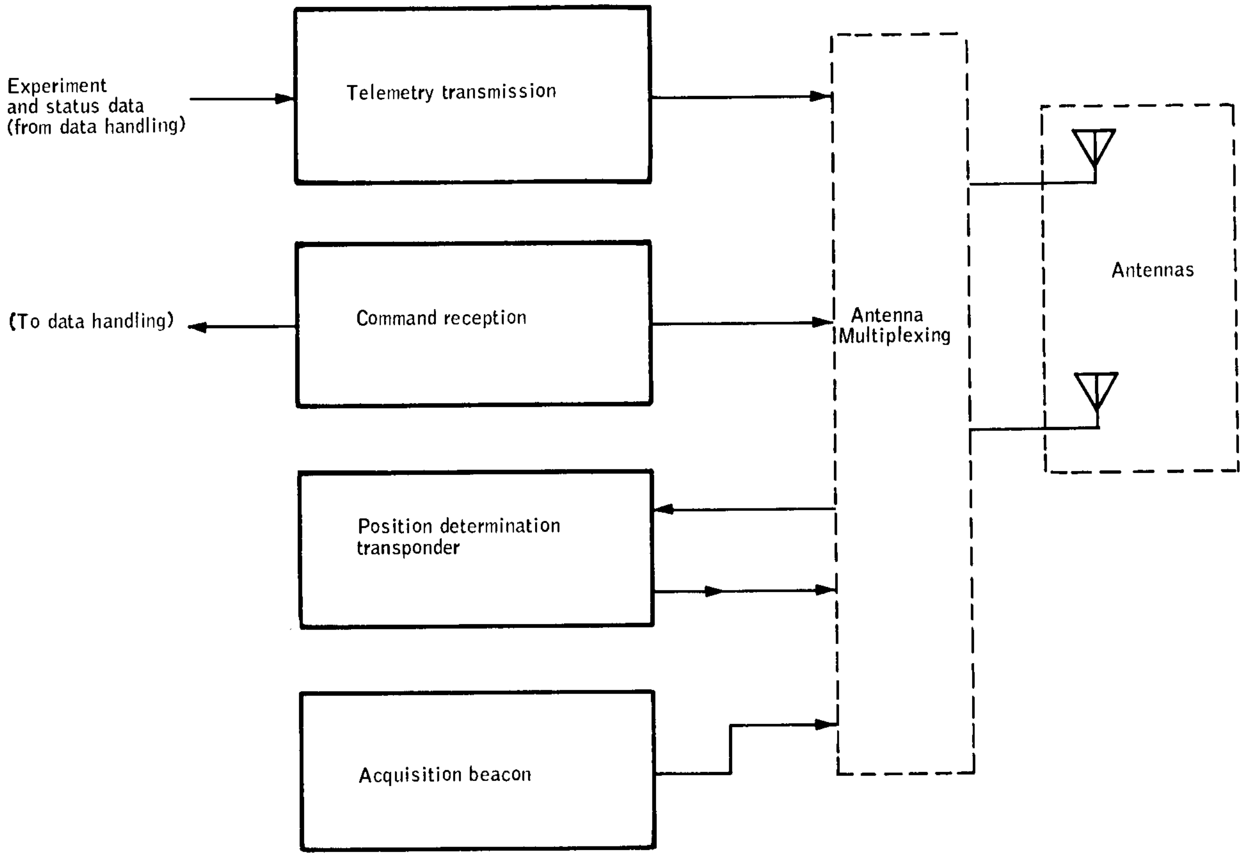


Figure 13. Communication Functional Diagram

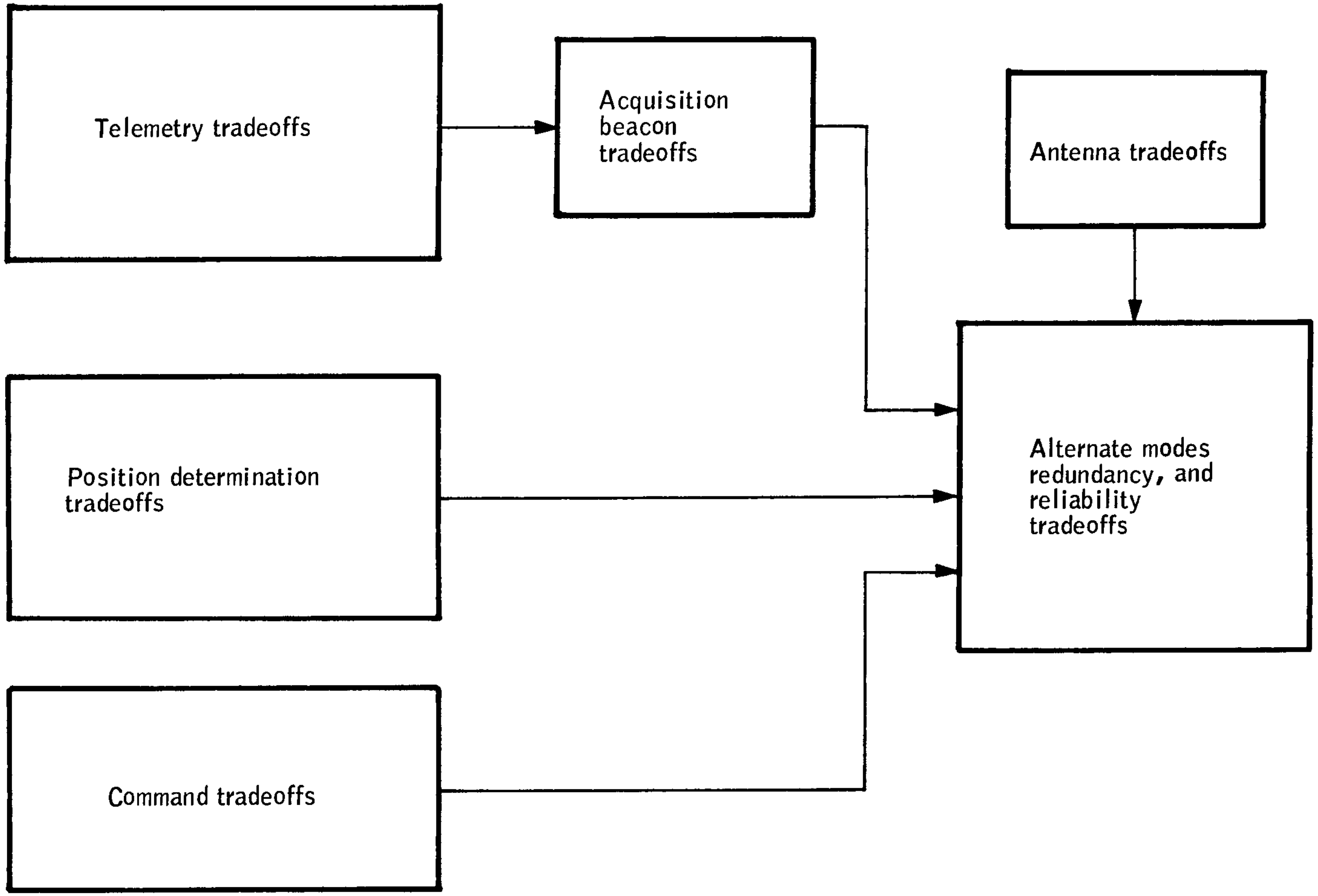


Figure 14. Communication Tradeoffs Flow Diagram

probably is not capable of achieving the required accuracy in measuring range rate. Consequently, it was necessary to utilize S band for this transponder.

The location of antennas required reconciling the requirements for adequate antenna patterns, maintaining minimum coupling with the radiometers and starmappers, and a practical mechanical design. This was accomplished by locating the S-band antennas (slots) on the tips of the power supply solar cell panels, by locating the vhf antennas on the front and back faces of the spacecraft, and by devising multiplexing filter circuits so that all vhf antennas could be shared by all vhf transmitters and receivers.

Considerable attention was given to securing adequate redundancy and providing for alternate modes of operation that could be utilized in the event of failure of one or more of the elements of the communications subsystem. A configuration was devised whereby at least two subsystem element failures must occur before the communications function is impaired. As a consequence, there is a very high probability that this function will not be impaired during the normal duration of the mission.

The communications subsystem tradeoffs are summarized in Tables 37 through 42.

Redundancy: The final configuration shown in Figure 15 in functional form, provides effective functional redundancy by making provision for numerous alternate modes of operation in the event of failure of one or more parts of the communications subsystem. The alternate modes of operation are summarized in Table 43. The ground rules used in determining the degree of redundancy and the alternate modes are:

- S-band range, range-rate transponder
  Two transponders, since complete redundancy cannot be supplied in any other way.

- Telemetry transmitter
  In event of failures:
  - Transfer telemetry load to beacon transmitter, or
  - Transfer telemetry load to operating S-band range, range-rate transponder transmitter.

- Beacon transmitter
  In the event of failure:
  - Transfer function to telemetry transmitter, or
  - Transfer function to operating S-band range, range-rate transponder transmitter.

- Command receiver
  In the event of failure:
  - Transfer function to the receiver of the operating S-band range, range-rate transponder.

TABLE 37. - COMMUNICATIONS TRADEOFFS, TELEMETRY FREQUENCY

Concepts	Disadvantages	Advantages
S band	• Limited number of S-band telemetry stations at present	• Wide data bandwidth available
	• Higher power consumption	• Could be superimposed on S-band range, range-rate transponder function
VHF band	• Some parts of vhf are being reallocated	• Adequate bandwidth available
		• Large number of ground stations

VHF-band chosen because of more extensive ground stations. (S-band can be used as alternate if S-band range, range-rate transponders are used.)

TABLE 38. - COMMUNICATIONS TRADEOFFS, POSITION DETERMINATION, TRANSPONDER, FREQUENCY

Concepts	Disadvantages	Advantages
S band	• Less efficient than vhf • Adds S-band antennas, etc. if vhf used for telemetry, command & tracking beacon	• Accuracy of measuring range and range-rate compatible with requirements
VHF band	• Not accurate enough	• Compatible with vhf for telemetry, command and tracking beacon

S band chosen since vhf-band does not provide needed accuracy.

TABLE 39. - COMMUNICATIONS TRADEOFFS, COMMAND RECEIVER FREQUENCY

Concepts	Disadvantages	Advantages
S band	• Lower efficiency than vhf band	• Compatible with S-band range & range-rate system
VHF band	• Some parts of vhf are being located	• Standard STADAN configuration

VHF band selected as most compatible with STADAN. (S-band could be used as an alternate if S-band transponders are used - would require modification of some STADAN facilities.)

TABLE 40. - COMMUNICATIONS TRADEOFFS, ACQUISITION BEACON MODULATION

Concepts	Disadvantages	Advantages
Beacon code only	• Requires a standby for desired reliability	• Normal configuration • Simplest beacon
Beacon code & status data	• Requires added modulation circuitry	• Provides redundancy ► Status data • Provides alternate experiment data capability • Redundancy can be provided by vhf telemetry transmitter

Selected beacon code and status data configuration because of enhanced redundancy features.

TABLE 41. - COMMUNICATIONS TRADEOFFS, S-BAND ANTENNA LOCATION

Locations	Disadvantages	Advantages
Spacecraft rim	• Location may be difficult and pattern may be distorted due to openings for optical instruments • Excessive energy may be coupled into radiometers or starmappers	• Simplest plumbing
Solar panel tips	• Plumbing more complex. Includes joint at solar panel hinges	• Minimum pattern distortion • Minimum interference with optical instruments

Solar panel tips chosen because low pattern distortion and minimum interference offset the added plumbing complexity.

TABLE 42. - COMMUNICATIONS TRADEOFFS, VHF ANTENNA LOCATION

Locations	Disadvantages	Advantages
Rear (sun) face only	• Distorted pattern	• Simplest
Rear &front (dark) faces	• Added complexity • Starmapper interference (if starmappers are on front face)	• Best pattern

Rear and front face locations chosen in order to obtain best pattern. (Starmappers are not located on faces.)

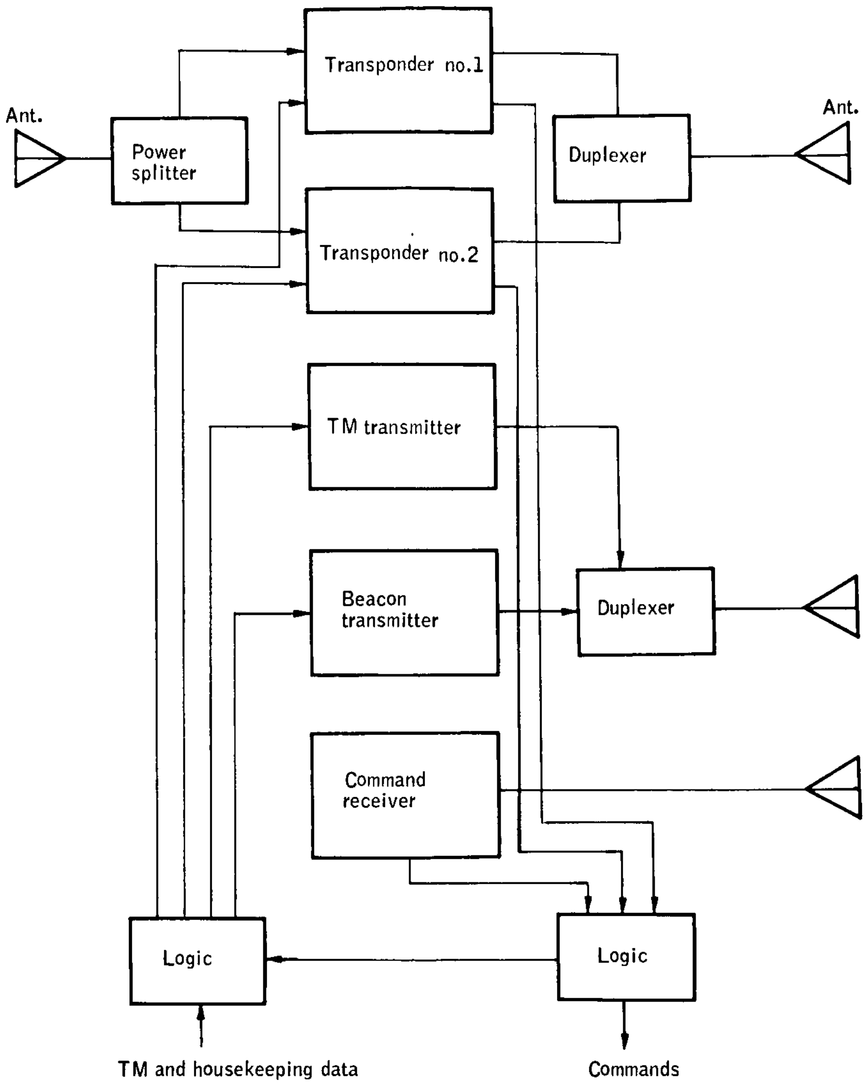


Figure 15. Communications System Signal Paths

The communications subsystem decisions, which served to define its salient features, are then as follows:

Area	Decision
Telemetry frequency	VHF band
Position determination transponder frequency	S band
Command receiver frequency	VHF band
Acquisition beacon	Add status data modulation
S-band antenna location	Solar panel tips
VHF antenna location	Rear and front faces
Redundancy	Minimum failure susceptibility by use of alternate modes

Power supply subsystem. -- As shown in the functional diagram of Figure 16, the power supply subsystem includes the prime power source, power storage, regulation and battery charge rate control, and power distribution functions for the spacecraft.

The tradeoff studies in this subsystem area followed the general sequence shown in the flow diagram of Figure 17. The most significant tradeoff decisions in the power supply subsystem were:

- Prime power source
- Solar cell configuration
- Storage battery type

Numerous prime power sources were initially considered as potential candidates for this function. However, the choice was soon narrowed to two types: (1) radioisotope-thermoelectric generator; or (2) silicon solar cells. The first of these has several attractive characteristics, the most significant being that its power-generating capacity is not a function of the orbit. However, the many restrictions that must be observed in handling, launch, and recovery together with the high cost markedly reduce the attractiveness of radioisotope-thermoelectric generators. Consequently silicon solar cells were selected as the most practical, available approach.

Careful study was made of the possible locations of the silicon solar cells. Body mounted cells minimize the mechanical complexity of the spacecraft by eliminating the need for deploying solar panels. However, the body-mounted configuration is inefficient. The most efficient concept considered was articulated (moveable) solar panels. Moveable panels, however, can be expected to generate undesirable reactions on the spacecraft motions. Consequently, fixed solar panels that are folded against the spacecraft until orbit injection and then permanently deployed were chosen as the most effective compromise.

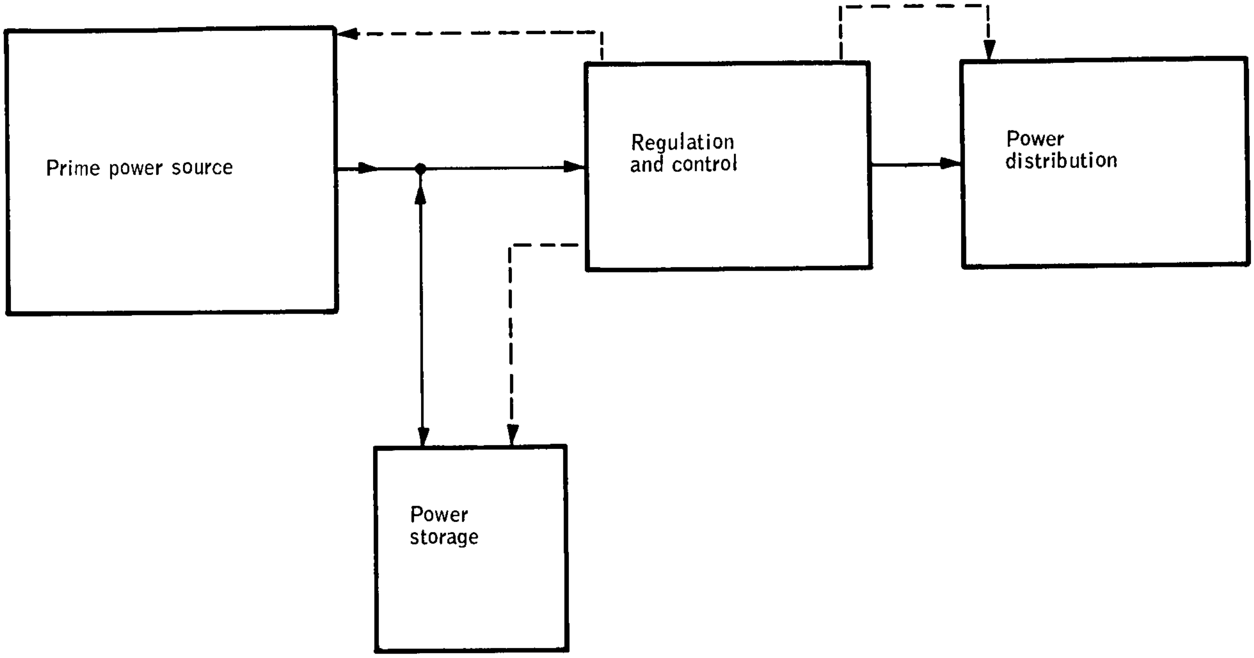


Figure 16. Power Subsystem Functional Diagram

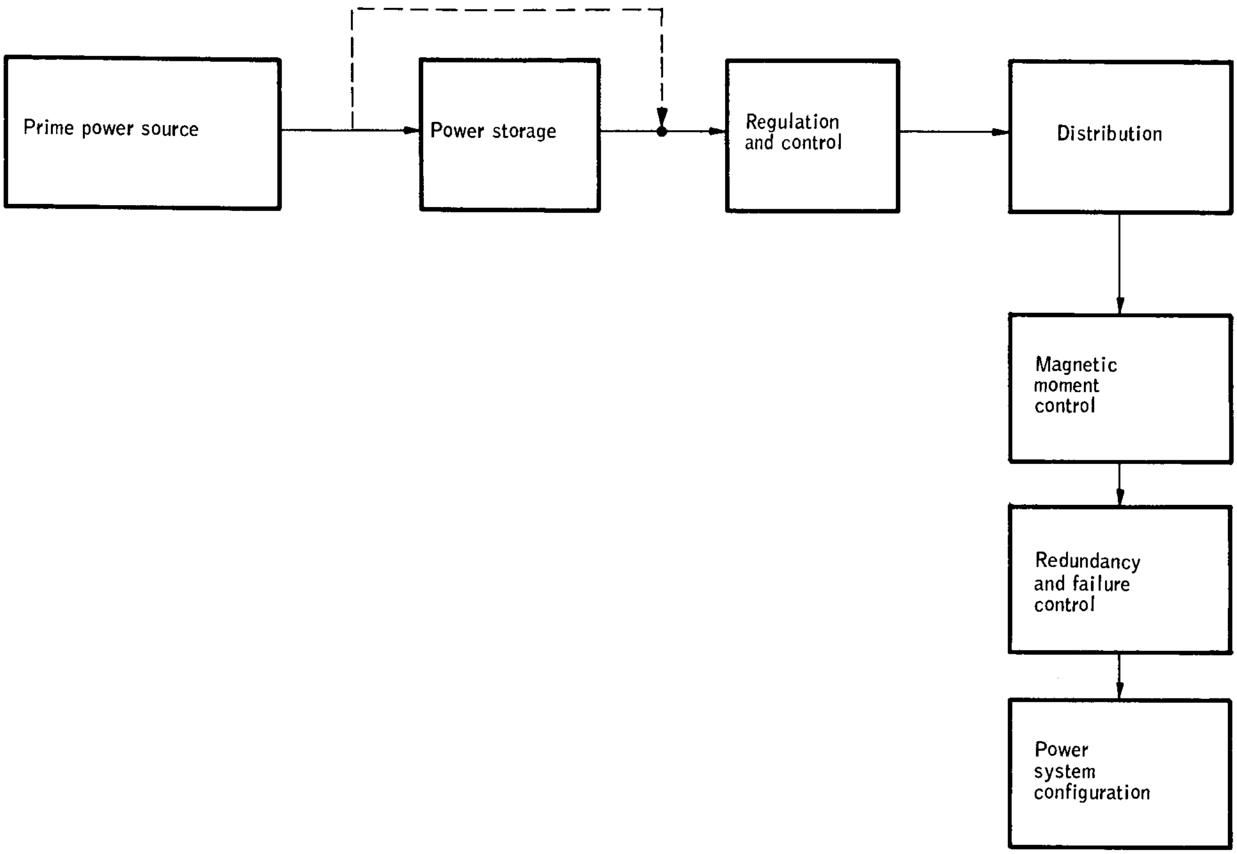


Figure 17. Power System Tradeoff Flow

TABLE 43. COMMUNICATIONS SYSTEM ALTERNATE FUNCTION COMBINATIONS

Failure	Function	TM transmitter	Command receiver	Beacon transmitter	Transponder #1	Transponder #2
	Telemetry	X				
	Command		X			
	Beacon			X		
	Tracking				X	
*	Telemetry					
	Command		X			
	Beacon			XX		
	Tracking				X	
	Telemetry	X				
*	Command					
	Beacon			X		
	Tracking				XX	
	Telemetry	XX				
	Command		X			
*	Beacon					
	Tracking				X	
	Telemetry	X				
	Command		X			
	Beacon			X		
*	Tracking					X
*	Telemetry					
	Command		X			
	Beacon			X		
	Tracking				XX	
	Telemetry	X				
*	Command					
	Beacon			X		
	Tracking				XX	

The nickel-cadmium type battery was chosen as the storage source based on charge-discharge cycle life. The only potential disadvantage of this battery type is the possible residual magnetic moments due to the magnetic characteristics of the nickel electrode. Although this is a potential disadvantage, weight and reliability advantages are gained.

The power supply subsystem tradeoffs are summarized in Tables 44 through 48.

The significant decisions which effectively defined the power supply subsystem configuration are as follows:

Area	Decision
Prime power source	Silicon solar cells
Solar cell configuration	Fixed panels
Storage battery type	Nickel-cadmium
Charge control	Conventional two-terminal battery
Regulation and control	Non-dissipative regulators

TABLE 44. - POWER SUBSYSTEM TRADEOFFS, PRIME POWER SOURCE

Concepts	Disadvantages	Advantages
Batteries	• Weight is prohibitive for one year orbit	• Simple • Insensitive to orbit
Fuel cell	• Weight is prohibitive for one-year orbit; 840 pounds for estimated load • One year life not available in existing systems (500 - 1000 hours now) • Exhaust of combustion products ► Can affect spacecraft motions ► Radiation and absorption affects radiometer	• Insensitive to orbit • Does not require power storage
Radioisotope-thermoelectric	• Launch path restricted due to safety requirements • Ground handling and launch restrictions because of radioactive material • Accountability and recovery restrictions on radioactive material • Internal high temperatures must be well insulated from radiometer cryogenic cooling • Considerable design effort required • High fuel cost ($ 2 to 3 million)	• Insensitive to orbit • Does not require power storage • Weight probably compatible with program
Silicon solar cells	• Sensitive to orbit • Requires storage of power for launch and during "night" • Requires careful design to minimize magnetic moments	• Lightest weight system (including storage batteries) of the concepts considered • Greatest space experience of any of the concepts considered

Silicon solar cells (with battery storage) chosen because light weight and proven space experience offsets the need for storage (batteries).

Sensitivity to orbit orientation can be tolerated in comparison with the disadvantages of the only other useable concept, radioisotope - thermoelectric generator.

TABLE 45. - POWER SUBSYSTEM TRADEOFFS, SOLAR CELL CONFIGURATION

Configurations	Disadvantages	Advantages
Body-mounted cells	• Inefficient geometrically • Cell dissipation must be radiated from vehicle body • Power capability limited by body area	• Does not require deployment of panels - eliminates all moving parts
Moveable solar panels	• Moving panels affect vehicle motions • Actuator life is limited	• Most efficient geometrically, lightest weight approach
Fixed solar panels	• Radiation from panels may enter aperture of radiometers • Deployment mechanism may fail	• More efficient geometrically than body-mounted cells (not as efficient as moveable panels except in "twilight" orbit)
Fixed solar panels and body-mounted cells	• Includes all the problem of fixed solar panels and body mounted cells.	Can deliver more power than body (mounted cells only. - minimum solar panel size for given power capacity)

Fixed solar-cell panel chosen as most efficient and reliable compromise for delivering estimated power. (Back-panel radiation into the radiometer is controlled to a suitably low level by shielding radiometer aperture.)

TABLE 46. - POWER SUBSYSTEM TRADEOFFS, POWER STORAGE BATTERIES

Battery types	Disadvantages	Advantages
Nickel-cadmium	• Nickel being magnetic can have remanent magnetism causing a magnetic moment to be present even when current is not flowing	• Maximum useable storage of types considered. (4 watt-hr/pound utilizing 40% discharge - 7000 cycles or greater than one-year operating life.) • Longest shelf life of types considered
Silver-cadmium	• Usable storage is only 1.75 watt-hr/pound for 7000 cycles (only 10% discharge is practical for 7000 cycles) • More expensive than nickel-cadmium for equivalent useable capacity and life	• Uses no magnetic material
Silver-zinc	• Operational life limited to less than 1000 charge-discharge cycles	• Uses no magnetic material

Nickel-cadmium selected because of weight, charge-discharge cycle capability, and cost advantages.

TABLE 47. - POWER SUBSYSTEM TRADEOFFS, BATTERY CHARGE CONTROL

Concept	Disadvantages	Advantages
Three-terminal adhydrode battery	• No flight history	• Close charge control • Minimum battery size • Minimum prime power requirement
Two-terminal battery - conventional charge control	• Extra battery size to allow for undercharging • Added power dissipation	• Standard technique with long experience record

Conventional two-terminal battery chosen because proven capability offsets the small increase in battery size and in power dissipation.

TABLE 48. - POWER SUBSYSTEM TRADEOFFS, REGULATION AND CONTROL

Concepts	Disadvantages	Advantages
Unregulated solar-cell/battery voltage	• Wide variations in voltage will require individual regulators in most other subsystems - overall efficiency lowest	• Simplest power supply
Dissipative regulators	• Less efficient than non-dissipative regulators	• Minimum rfi problems
Non-dissipative (pulse-width regulators)	• Potentially high rfi	• Most efficient

Non-dissipative voltage regulator chosen because overall system efficiency is highest. (RFI can be controlled by proper design and shielding of the regulator.)

Reliability Summary

In this section, the general method employed in the reliability analysis is presented, the estimated reliabilities of the subsystems are identified, and these reliabilities are combined to present an estimate of the reliability of the total system. The subsystem reliability analyses were made following the completion of the tradeoff decisions. The subsystem reliability predictions have been based upon estimates of the individual subsystem complexities and upon experience data.

Reliability failure rate sources.-- Table 49 presents the failure rates used in determining the reliability assessment of equipment based on an estimated component parts list. These failure rates are based on high-reliability procurement employing 100 percent screening for known weaknesses, approved derating policies, and approved fabrication techniques. For parts that probably would not be procured and handled in this way, MIL Handbook 217A failure rates were used.

In specific areas where field operating experience data was available, such data has been used where a significant similarity between equipment exists and where the equipment complexity is considered to be approximately equivalent. In the case of integrated circuits, it is reasonable that a somewhat lower failure rate could be used for digital circuit applications than for analog applications since a failure due to parameter drift in an analog circuit may not be a failure in a digital circuit (also the duty cycle is somewhat less in a digital application).

Although historically reliability improvements continue to appear in successive generations of equipment, no attempt has been made to adjust the observed experience data or the predicted failure rates to allow for any predicted improvements in reliability. However, it is reasonable to expect that some reliability improvements may appear during succeeding phases of the HDS program. Certainly it can be expected that during the design phases the reliability effort will include consideration of worst case analysis, significant piece-part derating, and an extensive part application review program. A parts reliability improvement program will consider burn-in of piece parts and reliability testing as required to ensure high equipment reliability.

Reliability prediction. -- A reliability analysis has been made of the individual subsystems of the system and then combined using the success diagram in Figure 18. The details of the internal redundancies of the individual subsystems are not shown in this diagram. A summary of the estimated individual subsystem reliabilities, including their internal redundancies, is presented in Table 50. The estimated reliability of the total system for one year of operation is shown to be 0.71.

Redundancy and critical failure modes. -- Only a minimum effort was spent in attempting to obtain a precise reliability prediction. Instead, effort was concentrated on ensuring the major subsystem failure modes were considered, that alternate modes of operation were provided for in case of a subsystem failure, and that the design concepts were chosen that conceptually

TABLE 49. -FAILURE RATES FOR HIGH RELIABILITY PROCUREMENT

Item	Failure rate (a)
Solder joint	$.0001 \times 10^{-6}$
Integrated circuit	$.10 \times 10^{-6}$
Transistors (silicon planar)	$.02 \times 10^{-6}$
Diode (spring type)	$.04 \times 10^{-6}$
Diode (solid-glass type)	$.01 \times 10^{-6}$
Resistor (carbon)	$.001 \times 10^{-6}$
Resistor (metal film)	$.001 \times 10^{-6}$
Capacitor (mica)	$.002 \times 10^{-6}$
Capacitor (ceramic)	$.002 \times 10^{-6}$
Capacitor (tantalum)	$.02 \times 10^{-6}$

[a] Failure rates correspond to:

65°C - (50°C max. ambient and 15°C temperature for part)
50-percent rated stress (conservative stress estimate for high reliability design)

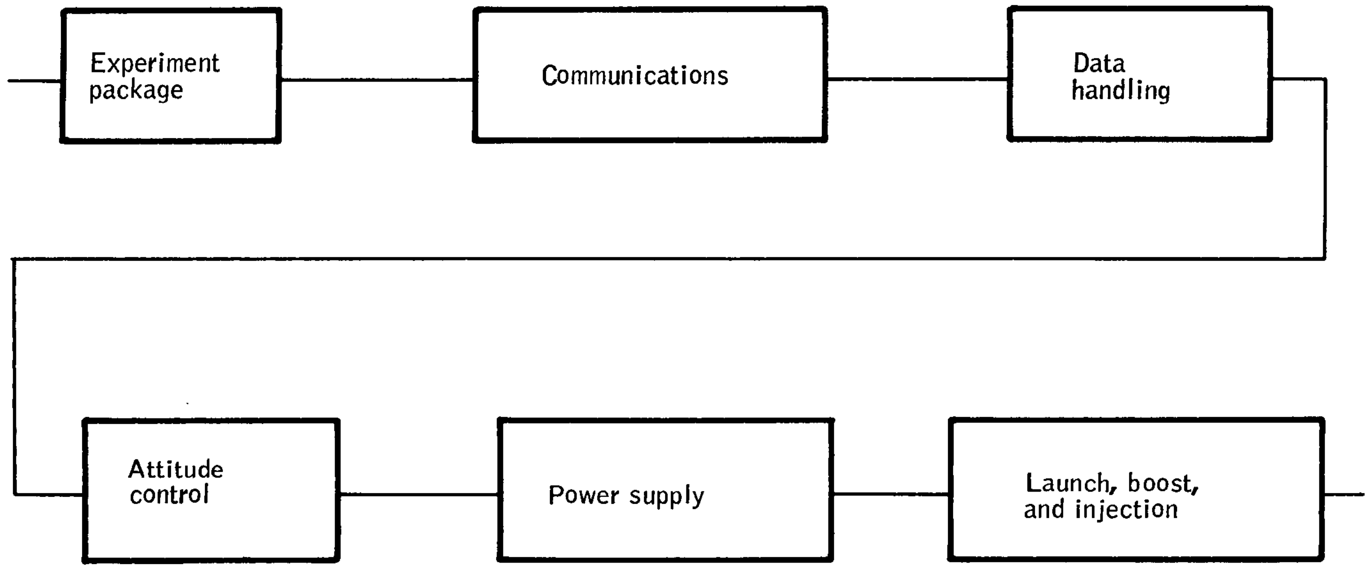


Figure 18. Reliability Success Diagram

TABLE 50. - RELIABILITY PREDICTION, ONE-YEAR MISSION

Subsystem	Reliability
Experiment package	.948
Communications	.977
Data handling	.909
Attitude control	.994
Power supply	.944
Launch, boost, and injection	.900
System total	.71

have a high probability of resulting in reliable designs. The key redundancy features incorporated as a result of these analyses are as follows:

Experiment package: Critical failure modes were minimized in the radiometer by using redundant choppers, calibration sources, IR detectors, detector power supplies, and electronics. Completely redundant starmappers and sun sensors were included as a practical redundancy approach in these areas. The remaining elements of these subsystems are non-reliability critical elements.

Communications: In the event of a failure on one of the major items in the communications subsystem, there are several alternate models of operation. As long as the S-band Range and Range-Rate system is operating and either the TM, transmitter or the 135 MHz beacon is operating, the communications subsystem could still perform its functions. To increase the subsystem probability of success, a redundant S-band Range and Range-Rate system has been added.

Data handling: An analysis of the data handling subsystem resulted in recommending redundancy in the command verifier and decoder, the timing oscillator and register, the programmer, the radiometer data collection unit, the attitude determination data collection unit, and in critical circuit areas of the memory.

Attitude control: The attitude control subsystem includes redundant V-head sensors and redundant control logic.

Power supply: The power supply subsystem includes two battery sets, two charge regulators, and internal redundancy within the regulators. The concept of standby redundancy has been used for the battery and charger combination.

Multiple Flight Techniques

An analysis was conducted to determine the potential improvements in achieving the HDS mission goals that might be obtained by using more than one spacecraft. This was accomplished by assuming conservative values for "random" failures and "wearout" failures for the spacecraft and then calculating the probability of survival of the system for a number of flight techniques.

It was assumed that the "random" system failures could be approximated by a Poisson distribution such that an individual spacecraft has a probability of operating for a year equal to 0.55. (This number is much smaller than the number presented in Table 50, to be conservative.) However, since "wearout" effects do not obey the Poisson distribution, they must be handled in a different manner. The dominant wearout effect will be the using up of the cryogen in the radiometer detector's cooler. This effect was approximated by assuming that the usage rate had a standard deviation about a mean rate, and than the quantity of cryogen was sized for 14-months life at the mean usage rate.

The "Poisson" and "wearout" probabilities of operation are shown in Table 51 and in Figure 19. The terminology used is defined as follows:

P_{RS} = probability of operation of spacecraft less cooler

$P_{RS} = \epsilon^{-\lambda t}$

t = time in months

$\epsilon^{-12\lambda}$ = 0.55 probability of survival of spacecraft less cooler for one year

$\lambda = 0.60$

P_{CS} = probability of operation of cooler

P_S = probability of operation of spacecraft

$= P_{RS} \cdot P_{CS}$

Examination of Figure 19 shows a 55 percent probability that the system will last 12 months and only a one percent probability that it will last 16 months. If it is desired to increase the probability of lasting one year, two possibilities exist:

- Increase the probability of survival of some or all of the subsystems (including more redundancy within the spacecraft)
- Use redundant spacecraft

Certainly the probability of operation of a single system (spacecraft) should be made as high as practical by suitable care in the design, construction and assembly of the spacecraft and its subsystems. However, at this point in the program, it is interesting to examine the possible advantages of using multiple spacecraft.

Simultaneous multiple satellites. -- One obvious but not necessarily practical approach would involve simultaneously launching more than one spacecraft. The probability of operation of N simultaneously launched spacecraft can be computed as follows:

$$P_{nS}(t) = 1 - \left[P_F(t)\right]^n$$

where:

$P_F = 1 - P_S$ = probability of one or more failures in a single satellite.

TABLE 51. - PROBABILITY OF OPERATION OF A SINGLE SYSTEM, P_S

t, months	$P_{RS} = \epsilon^{-\lambda t}$	P_{CS}	P_S	$P_F = 1 - P_S$
1	0.95	1.00	0.95	0.05
2	0.90	1.00	0.90	0.10
3	0.86	1.00	0.86	0.14
4	0.82	1.00	0.82	0.18
5	0.78	1.00	0.78	0.22
6	0.74	1.00	0.74	0.26
7	0.70	1.00	0.70	0.30
8	0.67	1.00	0.67	0.33
9	0.64	1.00	0.64	0.36
10	0.61	1.00	0.61	0.39
11	0.58	0.999	0.58	0.42
12	0.55	0.977	0.54	0.46
13	0.52	0.841	0.44	0.56
14	0.50	0.500	0.25	0.75
15	0.47	0.159	0.07	0.93
16	0.45	0.023	0.01	0.99
17	0.43	0.001	0.0004	0.9996

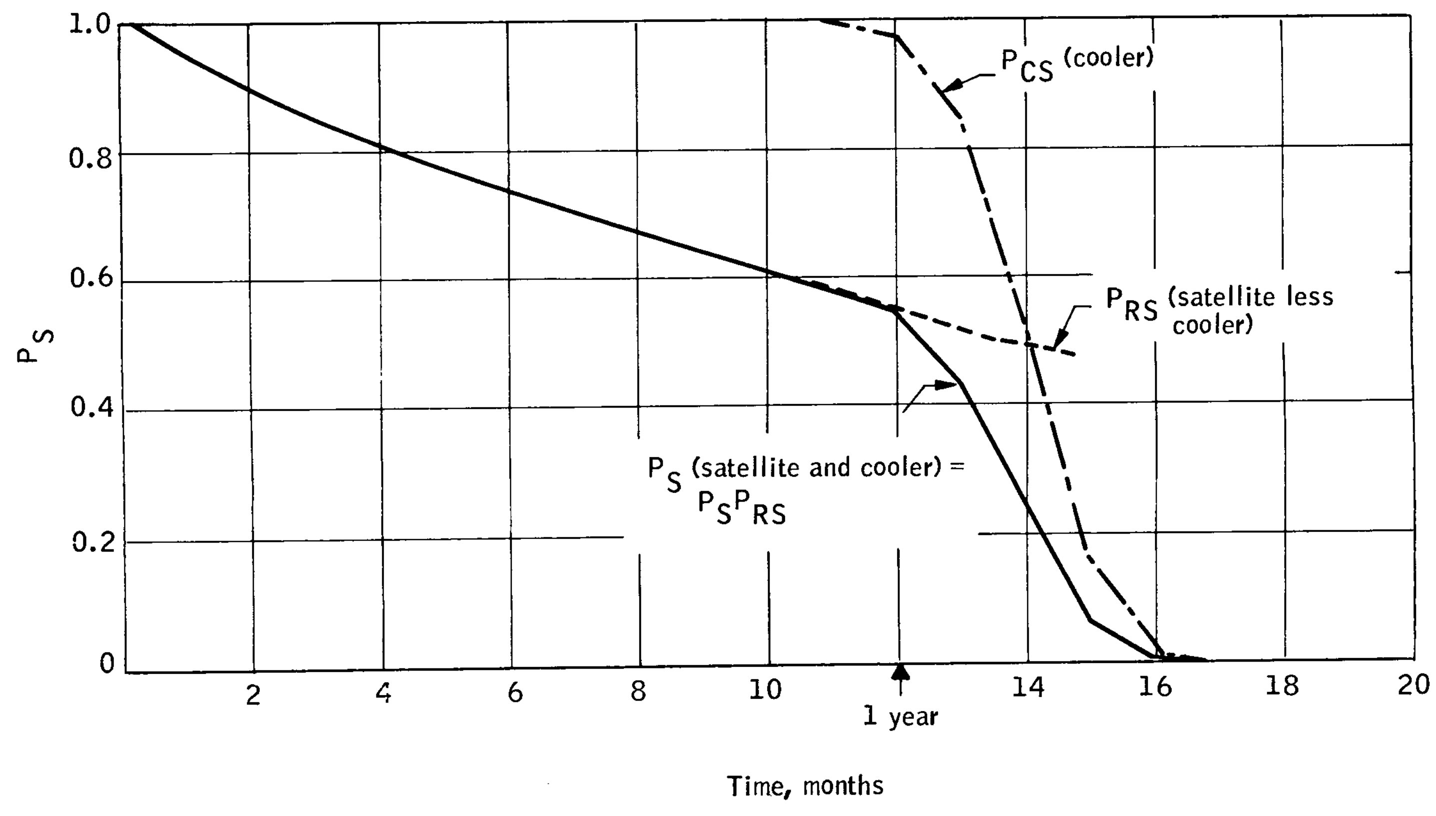


Figure 19. Single Satellite

Curves are plotted in Figure 20 for 1, 2, and 3 of the spacecraft launched simultaneously. The curve for a single satellite is that of Figure 19. It is apparent in Figure 20 that simultaneous launch of two spacecraft will improve the probability that at least one of them has not failed at the end of a year from 54 percent to 78 percent. Similarly, three simultaneously launched satellites show a 90 percent probability that at least one of them will be working at the end of a year. However, there is substantially no improvement in the probability of operating beyond 15 months due to the limited life of the cryogen.

There are numerous practical disadvantages of this method of obtaining spacecraft redundancy. One of the most significant being the necessity of simultaneously preparing three spacecrafts and three boosters for launch. This difficulty might be minimized somewhat if a sufficiently large booster were available to carry three spacecrafts simultaneously. This simplification would be offset however to the need to inject the three spacecrafts into orbit with adequate separation along the orbit path. An even greater disadvantage from an experimental point of view would be the high cost of a single booster failure since each booster failure would cause the loss of all the satellites it carried. Moreover, this approach commits all spacecraft simultaneously eliminating the possibility of spacecraft configuration or orbit changes in later launches to take advantage of knowledge gained during early flights.

Sequential multiple launches: To eliminate some of the objections to simultaneous launches, and to allow a month or more between launches for launch-pad refurbishing, sequential launches can be considered. The fact that the spacecraft design can be essentially checked out operationally early in the mission is an added advantage. Subsequent launches could utilize modified spacecraft if necessary. Since the P_S of a single satellite operating is 90 percent at the end of two months, a second satellite could be launched then the pair reaches to 90 percent. Such a situation is plotted in Figure 21 where the second satellite is launched at the end of two months and a third satellite is launched at the end of eight months. It is apparent that the probability of at least one satellite working is greater than 90 percent for one year. Primary consideration in these analyses was that of obtaining experimental data over a continuous year's period. These analyses yield assurance that at least one spacecraft will be operating during all time for the one-year period.

Curve A of Figure 22 illustrates a set of launch times which would yield a 95 percent probability that at least one satellite would be working at the end of one year.

Operational alternatives: Since the random component failures dominate the P_S of the satellite for the first year of flight, it can be argued mathematically (and practically also to a large degree) that if the satellite is still operating at the end of any given number of months, the probability of it operating one more month is the same as its probability of operating the first month after launch. From this one could conclude that the cheapest way to obtain a complete set of data for one year would be to launch the first satellite and then wait to launch the second satellite until the first one fails. If the second satellite could be launched immediately no data would be lost. Practically, however, the minimum time required to prepare and launch a spacecraft is about one month. Consequently, this approach will cause at least one month of data to be lost unless one of the satellites manages to survive for an entire

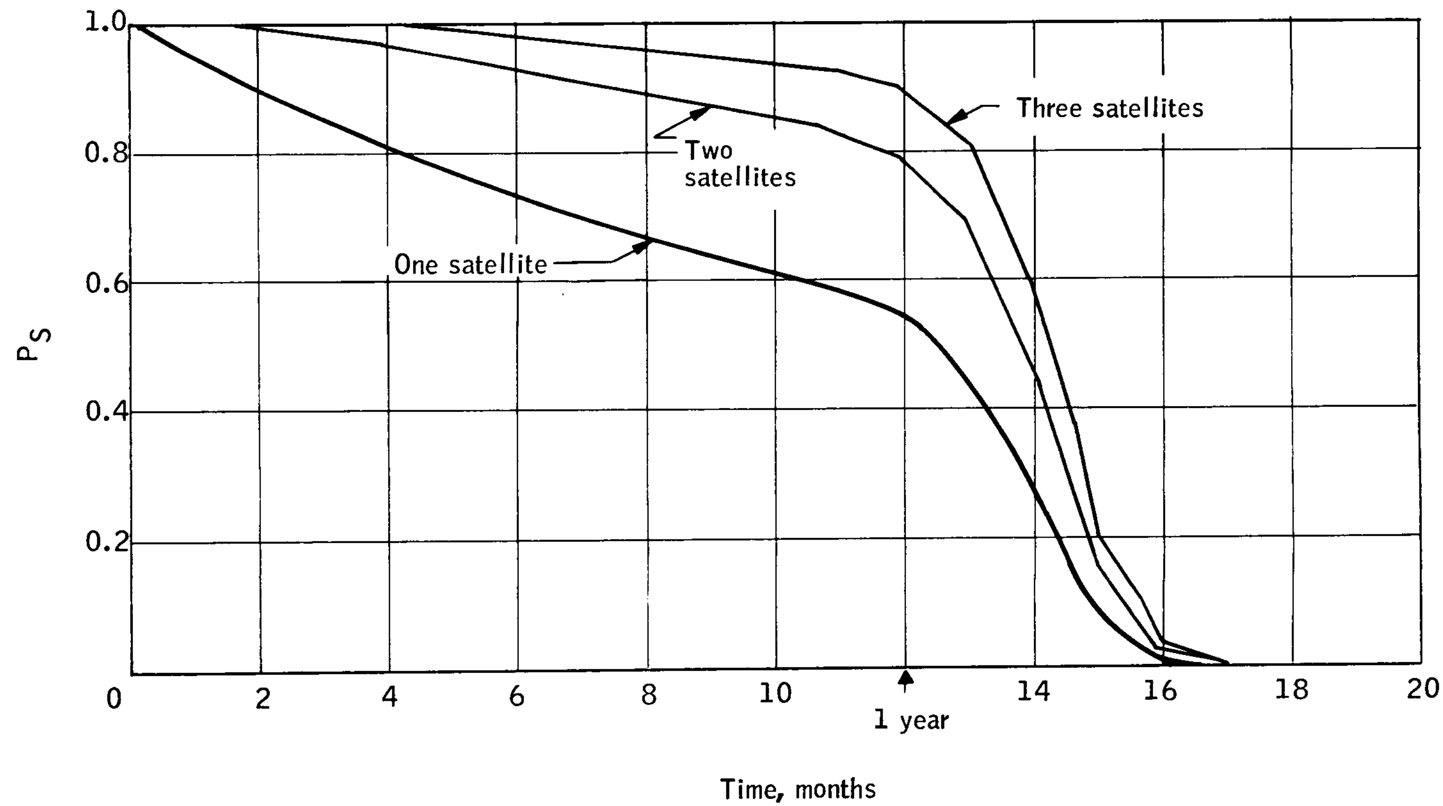


Figure 20. Simultaneous Multiple Satellites

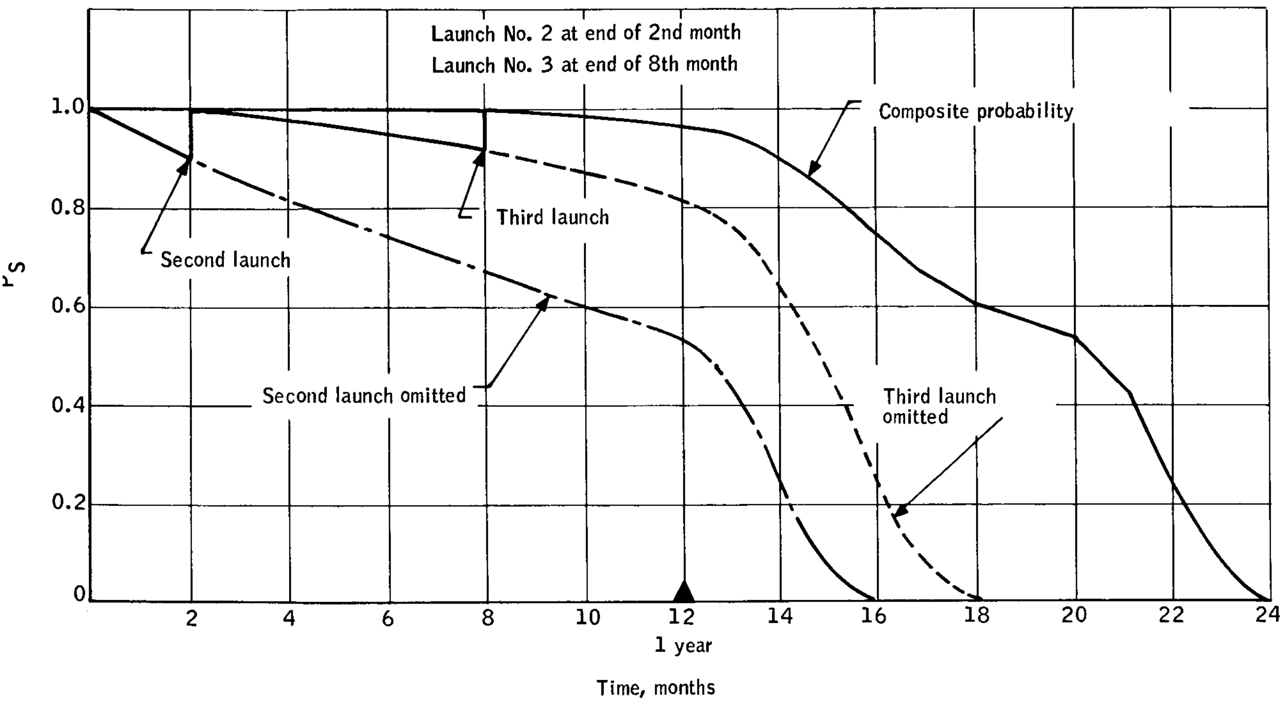


Figure 21. Multiple Launches, 2 - 8

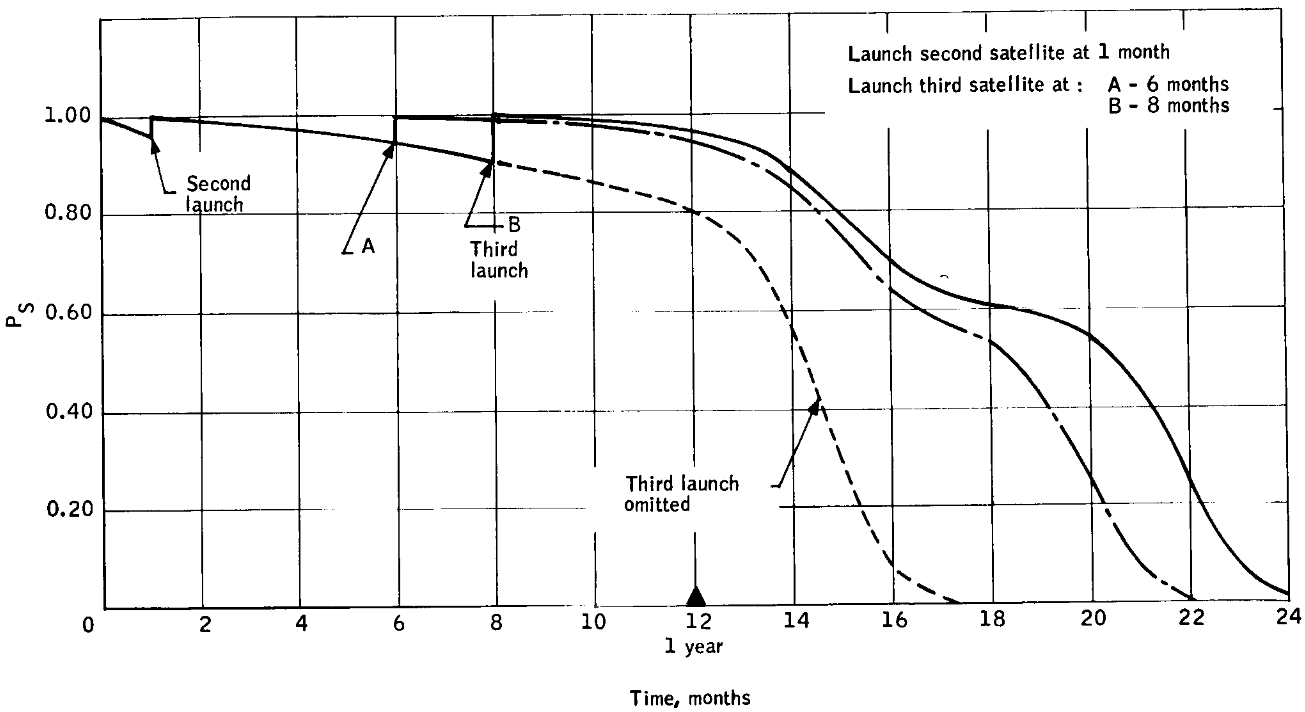


Figure 22. Multiple Launches, 1 - 6 and 1 - 8

year. However, there is a high probability that none of the satellites will last a year. Consequently, no matter how many satellites are launched, there is a definite possibility that a continuous, one-year set of data will not be obtained using this approach. Although the resulting set of data may be useful, even with one or more one-month gaps in it, it is desirable to examine approaches which have a higher probability of providing a continuous, one-year set of data.

Examination of Figures 21 and 22 shows that if a second vehicle is launched at the end of the first or second month, the probability that the first one is still operating (and hence causing no interruption of data) will be greater than 90 percent. Moreover, after both are launched, the probability that at least one is functioning does not fall below 90 percent for several months. Following the second launch then, a good strategy might be to withhold launching a third satellite until one of the two orbiting satellites has failed. If then launch preparations are immediately begun and the third satellite is launched within one month, there is at most a five percent probability that an interruption will occur (since there is a 95-percent probability that the surviving satellite will continue to function for a month).

This strategy can be utilized in many ways to:

- Minimize program cost
- Achieve added program benefits

For instance, if both satellites continue to perform well and the data appears to be satisfactory as the 12-month point is approached, a decision could be made not to launch any additional satellites. In this way the cost of the booster and the cost of storing and processing the additional data can be eliminated.

However, it may be decided to launch a third satellite in order to obtain data for a period longer than a year. Among the useful results of such a lengthened data-taking period would be acquisition of data useful for determining year-to-year effects. Figures 21 and 22 show that the data-taking period could be extended to 20 or more months while preserving great flexibility in the launch dates of each satellite. If it was decided to maximize the data-collection time,a full two-year program could be accomplished by suitable adjustment of launch times.

Alternatively, one could choose to launch one or more of the satellites in an orbit(s) other than the preferred 3:00 p.m. - 3:00 a.m. orbit. In this way added points could be obtained for extracting diurnal information. For instance (as shown in Figure 23, if the flight plan of Curve A, Figure 22, was used with each orbit at a separate hour, there would be over 50-percent probability that a 4-point fit to a diurnal model could be made for 12 months. However, the probability that a 6-point fit could be made is only greater than 50 percent for one month.

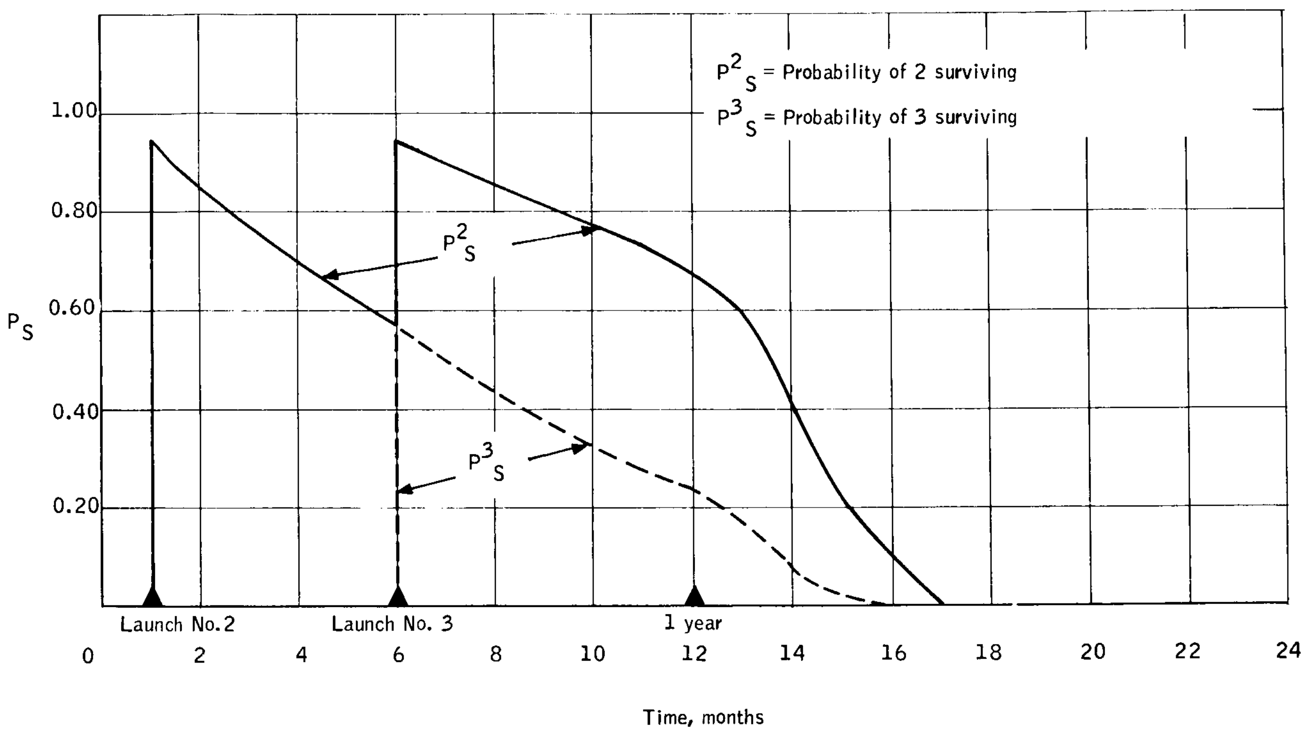


Figure 23. Probability of More Than One Satellite Being in Operation, Sequence of Figure 22

If it turns out that the wearout time of the cooler is considerably greater than estimated at this time, then these periods would be extended further. Greater probability of time overlap could, of course, be obtained by launching the third satellite earlier with a corresponding reduction in the probability of obtaining data beyond the first year.

Recommended flight technique: Based on the present estimate of the probability of operation of a single satellite, it is recommended that a second satellite be launched within one to three months after the successful orbiting of the first satellite in a sun-synchronous orbit one to two hours later than the initial satellite. If a failure of one of the first two satellites occurs within the first four to five months, a third satellite should be launched into the same orbit as the failed satellite. This approach maximizes the probability of obtaining a continuous one-year set of data, provides a 4-point diurnal data fit and provides a high probability of collecting a significant amount of data in the second year.

RECOMMENDED SYSTEM

The recommended system is summarized in terms of the subsystem descriptions including redundancy requirements, system reliability, and flight technique. This represents the achievement of goals which satisfy the Part II objective to establish solidly HDS mission feasibility.

Subsystem Descriptions

Each of the spacecraft subsystems definitions are established in terms of the final results of the systems effectiveness studies. The organization is identical to that used in the preceding section.

Radiometer. -- The radiometer is designed to employ a photon detector of cadmium-doped germanium. The operating temperature of the detector will be produced using a solid cryogenic cooling system containing a prime refrigerant of neon and a buffer refrigerant of CH_4. Reflective optics utilizing a classical Newtonian configuration will be employed. Optical modulation of the radiant energy will be performed utilizing a mirror driven by a torsional pendulum. Redundancy of detectors, electronics, and calibration and modulation elements will be provided.

Attitude control. -- Magnetic torquing will be designed to provide for spin and attitude corrections, compensation for the residual magnetic moment of the spacecraft, and precession of the spin axis at the sun-synchronous orbit plane rate. Torquing periods will be intermittent for the spin and attitude corrections, continuous for the compensation functions. Damping of coning motions will be provided to be activated upon ground command. Computation of torquing levels and sequences will be ground computed and implemented on the vehicle through stored programs.

A V-head horizon sensor will be required as a roll-yaw sensor to produce the necessary information for ground computation of commands. The functions of initial alignment and spin up will not be provided in the attitude control subsystem because of the launch vehicle capability to supply to these functions.

Attitude determination. -- The on-board attitude determination instruments will consist of a starmapper and a sun sensor. The starmapper will view in the orbit plane restricting its use to the shadow periods. Star signals will be obtained using a photomultiplier detector under a "triple-X" slit-reticle configuration. To provide separation of top and bottom of the "triple-X" slit, two photomultipliers are used in each starmapper. The sun sensor will be a V-slit configuration measuring the solar disc using an integrating sphere detector. An intermittent data process will be implemented which utilizes a number of sets of data, but not all sets producing a smoothed curve yielding the required attitude information for individual radiance profiles.

Data Handling. -- The data handling will enable intermittent transmission of digital spacecraft data by employing a solid-state ferrite-core memory. Data from the radiometer will not be subjected to any averaging or compression techniques. Simultaneous storing into memory and dumping of data on telemetry is provided. The spacecraft will have the capability for seventy commands through the use of the tone-digital capability of STADAN. Spacecraft time reference will use a standard frequency monitor concept. Programming of the radiometer data into memory will be required to produce experiment flexibility. Approximately 500 000-bit memory capacity is provided to store all data between once per orbit telemetry contacts.

Communications and tracking. -- The communications subsystems shall utilize the vhf frequency band for primary telemetry and have an alternate capability for telemetry using either of the dual S-band range and range-rate transponders of the tracking system. The command receiver frequency shall be the vhf band. An acquisition beacon will be required with a capability for modulation with spacecraft status data or the primary telemetry data. Beacon modulation can be switched to either the vhf or S-band antennas. S-band antennas will be located at the tips of the solar panels. VHF antennas will be on both faces (sun and shadow) of the spacecraft. The tone-digital vhf command capability will be provided with switching capability to the S-band transponder provided.

Power subsystem. -- Silicon solar cells mounted on fixed (fold-out) panels and conventional 2-terminal nickel-cadmium batteries will provide continuous power. Regulation will be with a non-dissipative voltage regulator. Two voltage levels will be provided.

System reliability. -- The subsystem reliability including the recommended redundancy is shown in Table 52.

TABLE 52. - RELIABILITY PREDICTION

Subsystem	Reliability
Experiment package	.948
Communications	.977
Data handling	.909
Attitude control	.994
Power supply	.944
Launch, boost, and inspection	.900
System total	.71

Flight technique. -- Based on the requirement for one year of continuous radiance data, the probability of success estimates, and the launch constraints, a flight technique requiring scheduled launches prior to a spacecraft functional failure is recommended. The orbits and frequency of launches to be recommended are contingent upon final spacecraft failure estimates, the success history of previous spacecraft, and the priority of introducing different diurnal content into the data.

OPERATIONAL PLAN

An operational plan containing the required tasks from the prelaunch phases through a portion of the orbital data-collecting period, sufficient to establish a repetitive cycle, has been developed to establish the feasibility of the spacecraft concept and the procedures for its utilization. The operational phases will be identified and described followed by a task/time sequence yielding an operational plan.

OPERATIONAL PHASES

The plan is divided into the following phases:

- Prelaunch
- Launch

- Orbital
  - ► Attitude trim and magnetic moment compensation
  - ► Initial attitude determination and thermal stabilization
  - ► Profile collection
  - ► Attitude and spin correction
  - ► Data reduction, analysis, and applications

Prelaunch

The prelaunch phase includes the operations required by the launch vehicle and the spacecraft following their delivery to the launch site (Western Test Range). The launch vehicle operations are an established sequence of events based upon a history of successful Delta launches. Into this sequence the necessary spacecraft preparations must be fitted. Spacecraft development and qualification proceeds in parallel with launch vehicle production.

The spacecraft is delivered to the launch site in a flight-qualified status approximately 90 days prior to launch. All spacecraft systems are thereafter checked and functionally verified. In preparation for shipment, the cryogenic cooler will be emptied and purged and the solar panels, batteries, and antennas removed. The spacecraft will then be placed in a sealed, transportable container provided with an inert, dry atmosphere for shipment. Spacecraft testing at the launch site will be primarily directed at the system level to verify that the spacecraft is functioning properly prior to launch. The spacecraft test phases at the launch site are as follows:

1. Receive and inspect spacecraft (S/C) and subassemblies
2. Assemble S/C and check mechanical operation
3. Conduct S/C electrical checkout
4. Conduct S/C subsystem functional checkout
5. Conduct S/C integrated systems test
6. Weigh S/C and conduct dynamic balance
7. Mate S/C to second stage
8. Conduct S/C integrated system test
9. Conduct simulated countdown
10. Conduct final countdown operations

The cryogenic cooler will be operational following the mechanical operational checks. The operation of the cooler is described to establish the resulting constraints on the above spacecraft test phases.

A basic feature of the cooler is that is uses two different solid cryogens: the inner chamber contains neon which acts as the detector coolant; the outer shell contains methane which shields the inner container from heat penetrating the system insulation. An evaporation vent, which also serves as a filler port, is provided for each cryogen. During operation in space, the solidified cryogens are slowly sublimed by absorbing heat from the detector and the surroundings.

The cryogens will be loaded into the container in gaseous form and solidified by circulating liquid helium through heat-exchanging coils. Maintenance of the cooler under earth atmosphere conditions (with some depletion of the coolant) may be accommodated for short time periods (< 7 days) by super-cooling the cryogens. Normal maintenance for longer periods will require a continuous circulation of the liquid helium coolant supply. The initial servicing requirements of the container are small, requiring approximately two liters of neon and 2.5 liters of methane. Solidifying of the cryogens will require a supply of liquid helium and a method for circulating it through the heat-exchanging coils,which are a part of the spacecraft cryogenic container. Maintenance of the cooler prior to launch will require a vacuum vent system. In general, support requirements of the cooler appear to pose no unique design problems and can be met without significant modification to existing WTR launch site facilities.

The launch vehicle is delivered to the launch site approximately 30 days prior to launch. The first and second stage boosters are checked out individually, and the first stage is erected on the launch pad. The second stage is then mated to the first stage , then a prototype rf model of the spacecraft is mounted on the erected second stage vehicle, and rf system tests are conducted. Upon completion of spacecraft checkout and approximately four to seven days prior to launch, the flight spacecraft is mated to the launch vehicle (spacecraft test phase 7). Spacecraft/vehicle compatibility and spacecraft systems checks are conducted from this time until approximately two days prior to launch. The tasks from this time until launch are accomplished in a pre-established sequence,documented in a mission countdown manual. The significant events from the spacecraft standpoint include:

Time before launch	Event
2 days	Begin countdown tasks
1 day	Install payload fairing
3 hours	Remove gantry
1-1/2 hours	Spacecraft final checks
20 minutes	Final countdown

Launch

The launch phase includes all the launch vehicle flight operations which are boost, injection, orientation and spin up. The boost and injection are functions common to all launches, whereas the orientation and spin-up operations are peculiar to the spacecraft requirements. Orientation results in a 90° maneuver which positions the spacecraft spin axis normal to the orbit plane. The spin-up function requires a spin table to produce the nominal rate of 3 rpm required for the HDS spacecraft. Solar-panel erection will precede the spin-up function. Separation follows, and the spacecraft is in a condition within the limits for spin magnitude and spin-axis attitude.

Orbital

Orbital operations begin with the attitude trim and magnetic moment compensation phase. This phase requires telemetry of attitude information, transmission to a computing facility, computation of a correction command, and implementation from the STADAN command station. One magnetic torquing period of one-half orbit will provide for the correction of the spacecraft attitude to the nominal condition.

Following the initial torquing period, the spacecraft attitude will be measured by both the attitude control system, using a V-head horizon sensor, and the attitude determination instruments, using the starmapper and sun sensor. These measurements are telemetered to STADAN stations possessing direct communication with Goddard in order to evaluate the spacecraft's actual magnetic moment. The prediction accuracy of the spacecraft's magnetic moment is not considered to be sufficiently accurate; however, a single command to correct the magnetic compensation coil will produce the required level of residual spacecraft torque to allow the next phase to begin. During this phase, the radiometer data would not be valid because fine attitude information is not available and the long-term thermal transients in the radiometer have not had time to stabilize. This phase is expected to be completed during the first day of orbital operation.

The initial attitude determination and thermal stabilization period duration is expected to be completed during the first seven days of orbital operation. During this period, the spacecraft memory will essentially be devoted to the storage of attitude data, with a minimum amount for radiometer calibration and checkout and spacecraft status. Memory readouts will occur every orbit, principally to the two STADAN stations (College and Rosman), which have direct data communication capability with Goddard. This enables the initial solution of the attitude determination ground program to be implemented in a minimum of time.

During periods wherein these two stations are not in contact with the spacecraft at orbital intervals, other stations will provide the telemetry coverage. This data will not be required for the initial solution and, therefore, will not require expedited delivery to Goddard. The coverage is required in the event early results disclose unexpected anomalies. This period will be complete when the spacecraft attitude uncertainties are within the required limits and the radiometer thermal gradients have stabilized.

The profile-collection period is the typical data taking sequence with telemetry contact at intervals of one orbit or less. Spacecraft status data provides the information to predict the frequency of the attitude correction phase. The intervals between these corrections are estimated to be as short as five days. Telemetry contacts will be scheduled according to existing STADAN priorities; however, a set of four primary and six secondary stations are expected to yield the necessary coverage flexibility to prevent losses of telemetry data. The STADAN system does not have the capability to provide for contact on every orbit during the year. At intervals of approximately four days the contact intervals extend to periods of approximately 160 minutes or nearly two orbits. To accommodate these periods, profile selection program changes may be commanded. Several approaches to extend the data coverage are time delays, elimination of stable latitude intervals, and reduced selection ratio.

A process for selecting the data profile to accommodate the memory limitation, achieve the proper profile distribution and to accommodate the periods in the orbit during which sun-line relationships render the forward profiles unacceptable will be discussed. This discussion serves to establish a potential programming requirement in the event solar baffling of the radiometer firmly establishes the requirement to switch to the rearward scans. A data analysis would also be required to determine the equality of the scan types.

The normal mode will be to collect space/earth (forward) profiles to provide for the ratios of 1/7, 1/4 and 1/3 for 0 to 30°, 70 to 60°, and 60 to 90° latitude bands, respectively. In addition, however, switching to accommodate changing to earth/space (rearward) profiles may be required during the orbital period wherein the sun line is below the minimum angular position relative to the radiometer line of sight. At this time, forward profiles will not be collected. To provide full coverage, rearward profiles can be taken following a time delay of approximately 11.5 minutes. This time delay is determined by considering the forward horizon to be 22° away from the spacecraft local vertical. The spacecraft must, therefore, travel through twice this angular distance to result in the last forward horizon position to be located 22° behind the spacecraft's local vertical. At this time, rearward scans are selected at appropriate latitude interval ratios, thus, eliminating overlapping of data.

Figure 24 displays graphically, for a typical orbit having an ascending node at 3:00 a.m. during the equinox, the point at which the radiometer scan selection (forward or rearward) must be changed to accommodate a requirement for a one-quarter orbit of collecting rearward scans.

The relationships are based on the 22° angle to the horizon for the design mission profile altitude of 500 km. The result is four data selection phases consisting of forward only, none, rearward only, and both forward and rearward.

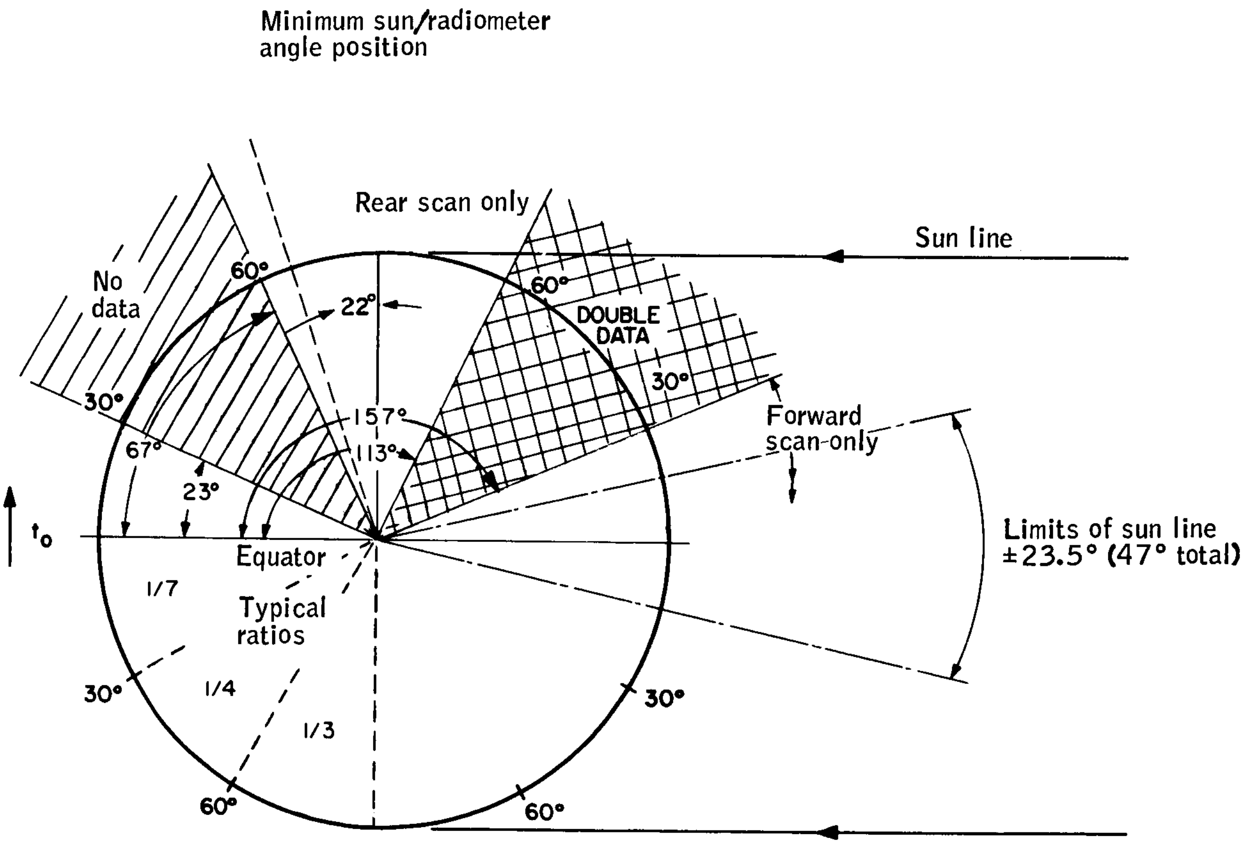


Figure 24. Single Orbit, Data Selection

Additional switching is required to provide for the proper radiance profile quantities stored in memory. Figure 25 illustrates the combined scan selection and ratio discretes required for each orbit. The total data into storage is shown as a composite ratio. The switching of forward and rearward scans will shift ±23.5° about the equinox resulting in a total profile selection ratio of 0.575 for the period approximately 10° either side of the summer solstice.

It should be noted that ratio changes for the rearward scans are not changed over the Southern Hemisphere because there is no established requirement for a balance of the profile types. Also of importance is the capability to shift the discretes commanding rearward scans to the Southern Hemisphere by changing the launch hour, i.e., the example is for a 3:00 p.m. launch from WTR resulting in an ascending node at 3:00 a.m. Changing to a launch time of 3:00 a.m. would shift the "no data" and "double data" intervals of Figure 25 to the Southern Hemisphere.

The cyclic profile collection phases will alternate with the attitude and spin correction phase. Attitude correction will be required at intervals of five days or longer, while spin correction to maintain 3 rpm ±5 percent are expected to be at longer intervals. Each of these corrections will be implemented by commanding a stored program. Although the periods of expected correction are not equal, spin correction will be phased with an attitude correction. The necessity to re-establish the spacecraft attitude following each correction and the invalidating of radiance profile data during either of these magnetic torquing programs identify the reasons for minimizing the total torquing intervals. One-half orbit is adequate to accomplish both corrections.

Data reduction analysis and applications refers to the operations necessary to obtain horizon radiance profiles properly tagged with auxiliary data after the GSFC Data Processing Center has cataloged, screened, ordered, inserted time, and formatted the raw data. The primary data reduction tasks are those associated with quality checks, determination of attitude, calculation of tangent heights, calibration of radiometer outputs, generation of horizon radiance profiles, and finally the tagging of auxiliary data such as meteorological, atmospheric, etc. Figure 24 illustrates these operations. These operations are expected to be performed concurrent with the cyclic orbit operations of profile collection; however, because there is no requirement to use the outputs to modify the orbital operations, concurrency is not mandatory. Further discussion of these areas follows:

Attitude determination. -- The identification, from star catalogs, of the star crossings will be procured to yield the spacecraft attitude as a function of time. Sun sensor data consisting of solar disc crossings versus time will be utilized to maintain the attitude determination predictions over the non-shadow orbit periods within the error allocations.

Tangent-height calculator. -- The tangent-height calculator is a mathematical tool for evaluating the minimum earth to scan-line distance when the following data are given:

1. The transformation from spacecraft axes to celestial axes (starmapper data reduction output);

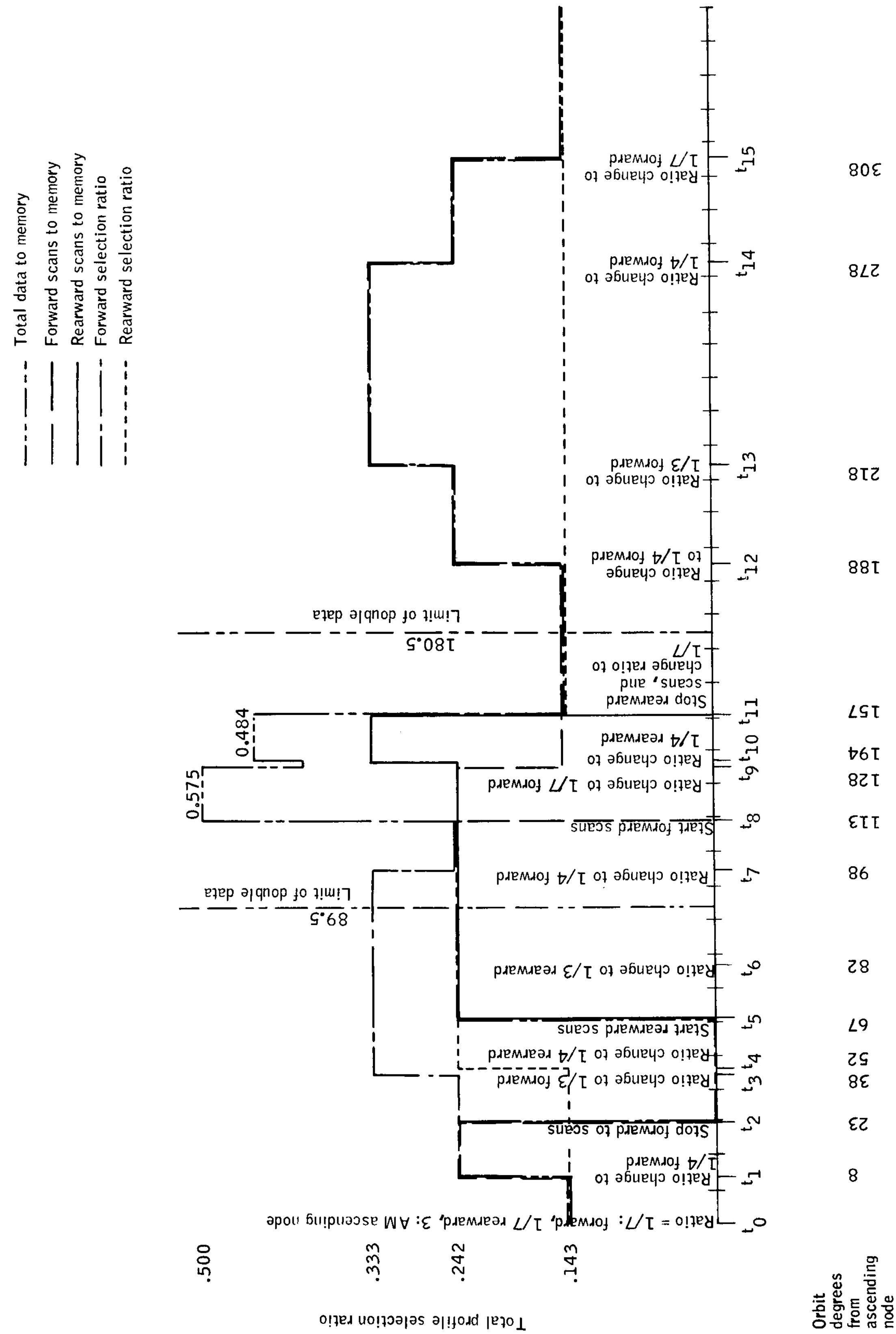


Figure 25. Profile Selection Timing Discretes

2. The radiometer scan vector in spacecraft axis components; and

3. The geocentric spacecraft vector.

The tangent-height calculation is therefore basically a sequence of coordinate transformations where the output consists of tangent height and latitude, longitude of the sub-tangent point tagged with time, and any useful auxiliary data as may be required. The accuracies associated with this approach have not been evaluated; however, it is expected that a first-order earth oblateness correction will be necessary. Depending on the accuracy required, second and higher order iterations of an oblateness correction procedure may be necessary. The efficiency of this approach is important to the cost and time involved in this particular data reduction function.

Radiometer calibration. -- The in-flight calibration data is used to produce a time-functioned calibration which is used to compensate the radiometer profile data producing uniform calibration during the profile generation phase immediately following.

Generation of radiance profiles. -- This operation is basically what is referred to as "merging of the data" to obtain the radiance profiles. The radiometer data is input as a function of time and the corresponding tangent heights are input as a function of time. These two sets of data are merged to provide radiance levels or a function of tangent height which results in a horizon radiance profile for each two sets of input data (i. e., data associated with one scan of the horizon). Although not mentioned, it is assumed that any profile identifiers (latitude, longitude, time, etc.) are carried along through the process. If certain errors have been identified (i. e., calibration errors, tangent-height errors, etc.), these too should be identified and recorded with the resulting profile.

Profile tagging. -- This is not an actual data reduction function but rather a merging process of profiles and auxiliary data which will be useful in subsequent data analysis. Various types of auxiliary data have been identified in Figure 26. It is expected that this type of data will be available in abundance by the early 1970's due to increased activity in meteorological measurement programs. The merged data, horizon profiles and auxiliary data, will then be stored on magnetic tapes and await data analysis and/or utilization by industry and various investigators.

Task/time sequence. -- A chronological task sequence of the previously identified operational phases will be discussed herein to establish the feasibility of the measurement program by demonstrating task compatibility.

Figure 27 illustrates the prelaunch phase from spacecraft delivery to WTR through two days prior to launch. Figure 28 details the 48-hour countdown terminating at lift-off.

Figure 29, launch phase, establishes the timing interval of 12 minutes as the expected duration, during which time all booster functions will be accomplished through orientation and spin up of the spacecraft.

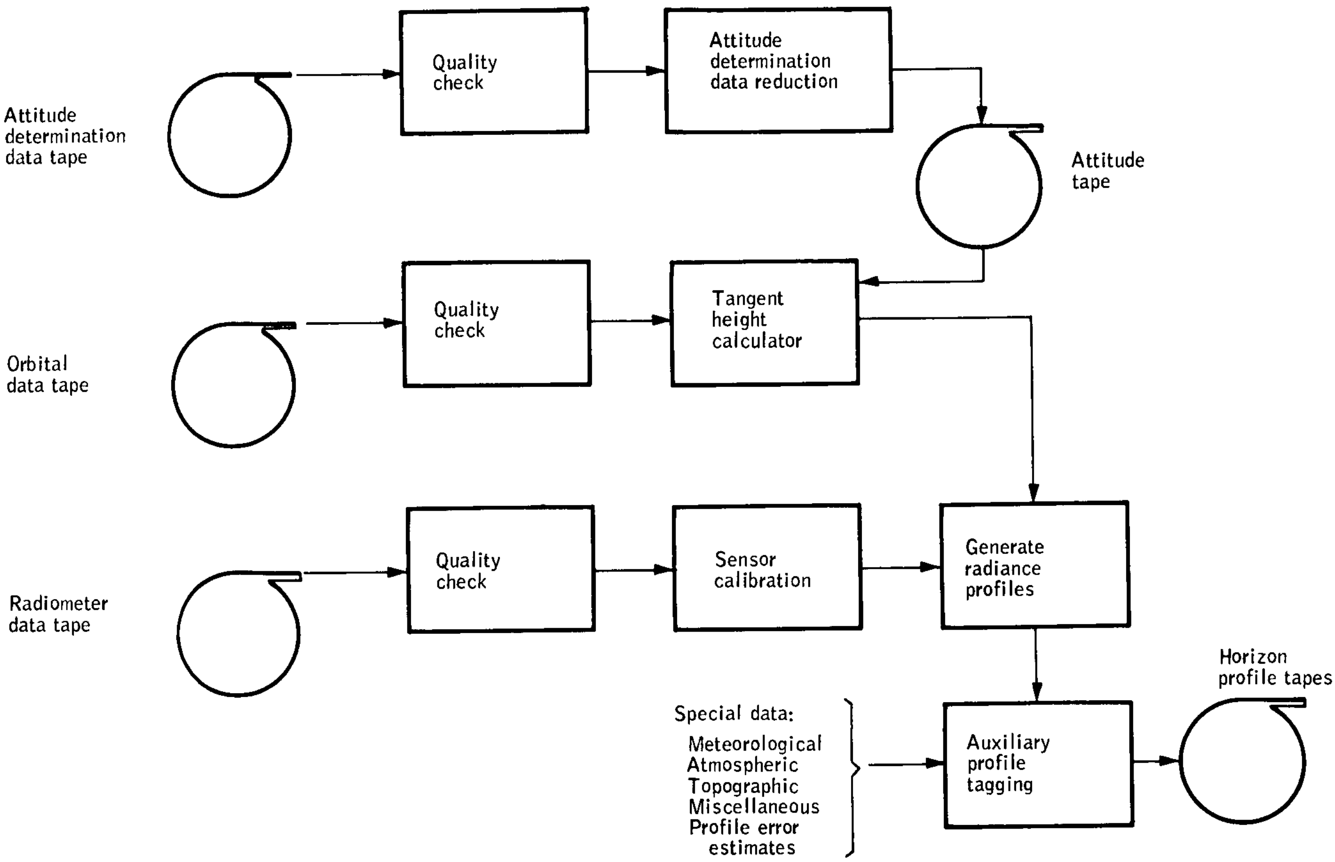


Figure 26. Data Reduction

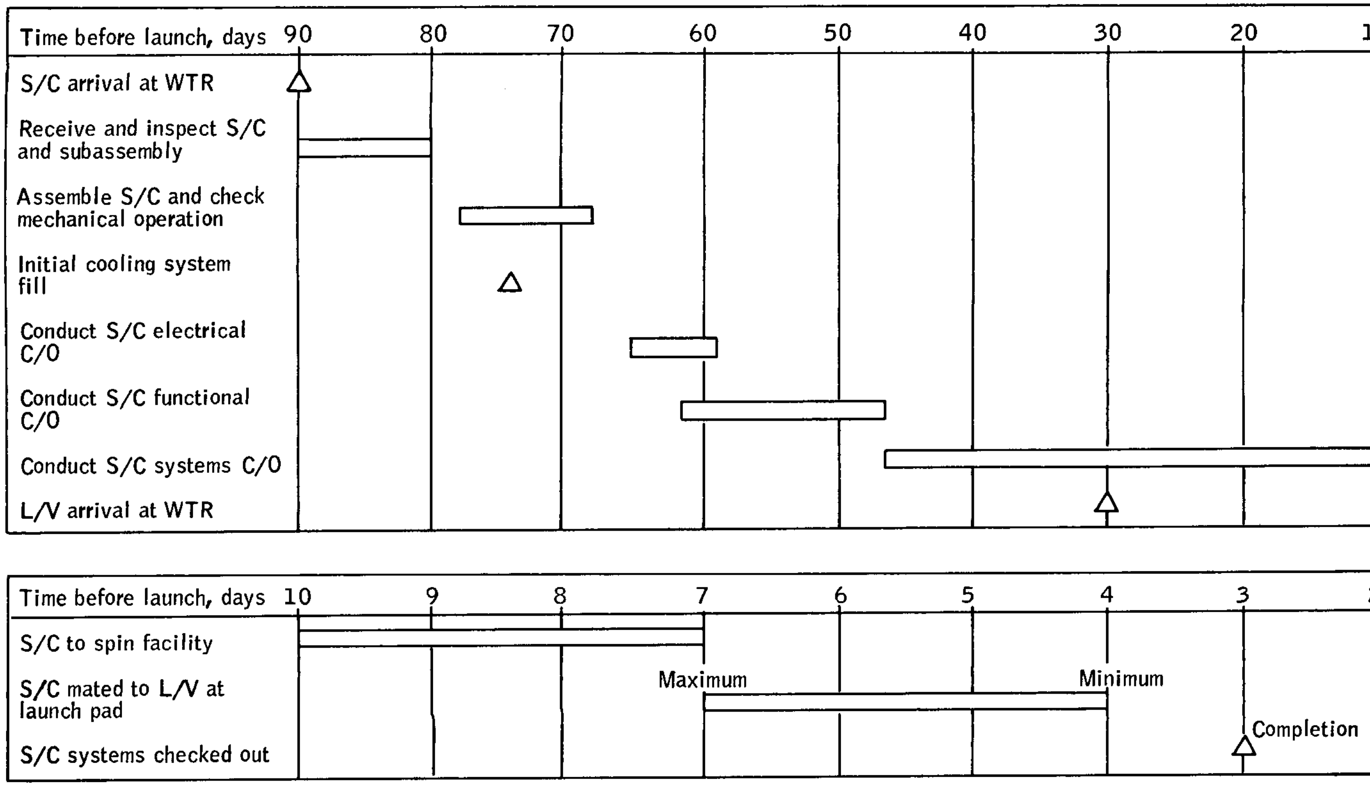


Figure 27. Operations Plan, Prelaunch Phase

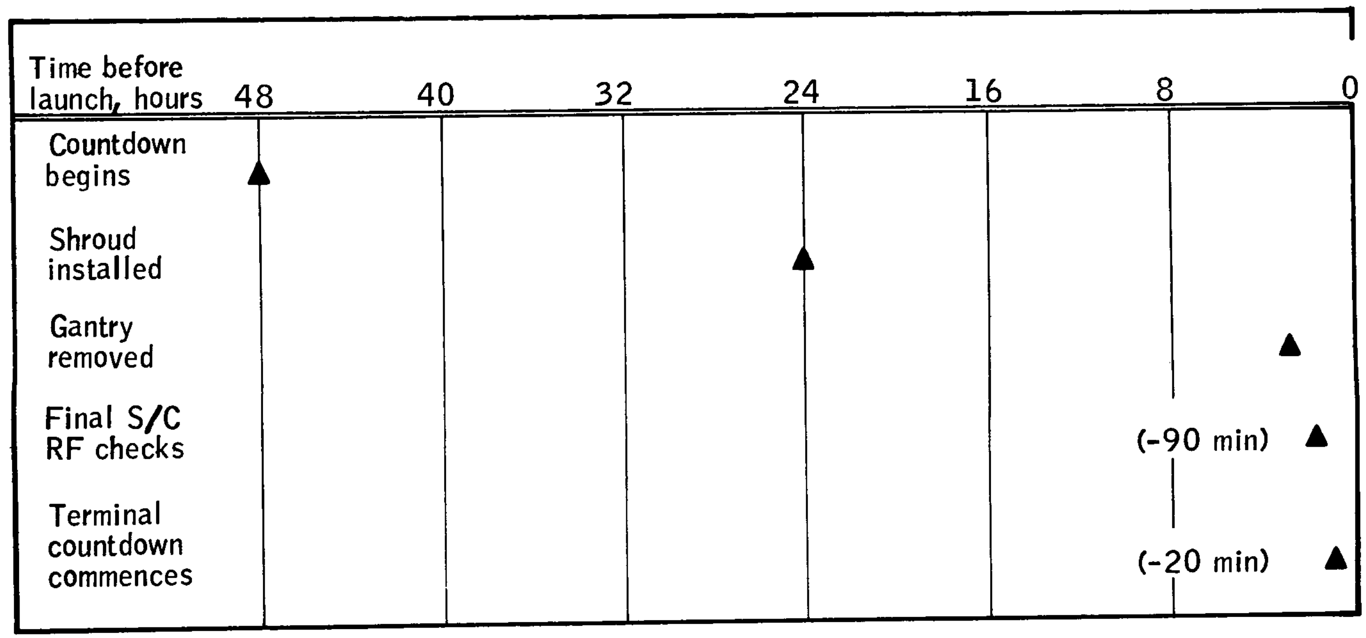


Figure 28. Operations Plan, Prelaunch Phase, Final 48 Hours

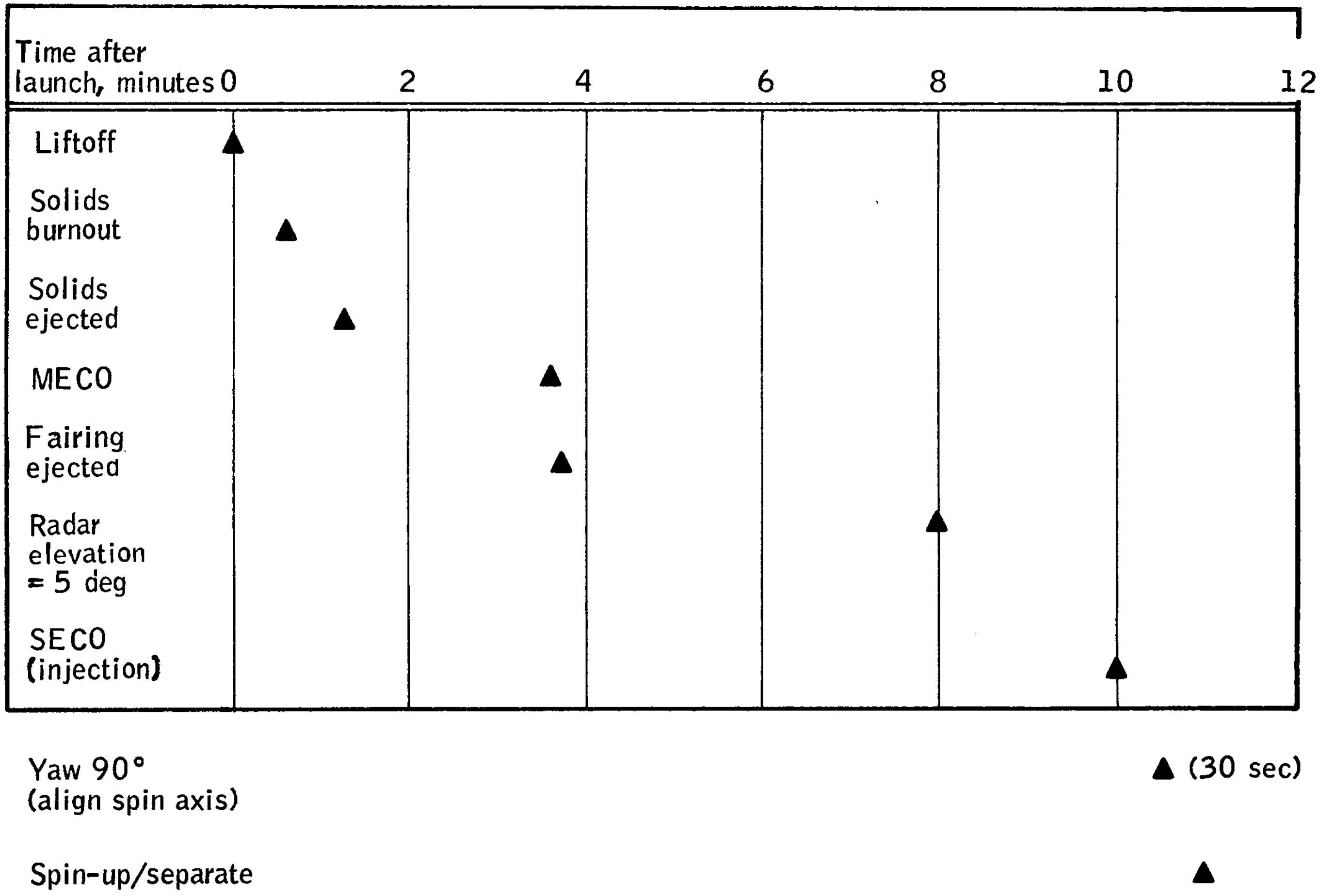


Figure 29. Operations Plan, Launch Phase

The third phase, attitude trim and magnetic moment compensation, will be accomplished during the first day in orbit. The telemetry contacts for a typical day are shown in Figure 30 for stations classified as primary and secondary. The primary stations can accomplish the entire program; however, due to expected conflicts and priority considerations secondary stations are identified to provide a complete range of STADAN alternatives. These will be of greater significance during the repetitive profile collection phases extending over the entire mission duration of one year.

The event sequence for the first day is contained in Figure 31, attitude trim and magnetic moment compensation. Telemetry and tracking contacts are identified in numerical sequence and station location.

Figure 32, mission control inputs and tasks - Goddard, identifies the events required of the mission control function located at Goddard. Data delivery from the tracking and telemetry sites, computation functions, and command transmission to the sending STADAN station are typical functions defined.

The telemetry information received at stations not possessing direct communication to mission control is not required for the magnetic moment computation; however, its collection will provide full coverage in the event anomolies or failures require diagnosis of this contingent data.

The period required for the initial attitude determination and thermal stabilization is estimated to require approximately six days (seven days in orbit). The sequence will be similar to the previous phase,in that direct communication with the mission control location is required for near real-time input of data from the attitude determination instruments to produce coverage to the solution as rapidly as possible; therefore, the telemetry contacts from the primary stations of College and Rosman, possessing direct communication with Goddard, will be utilized. The other data contacts, completing the record, will not be required for the initial solution because usable profiles are not being measured. The collection of this data will provide for contingency analysis in the event of failure or anomolies.

Figure 30 reflects the available coverage with the two stations, which is very nearly complete.

Command requirements could be required during this phase in the event the spacecraft attitude exceeds the ± 5° limits.

The profile collection phase will be repetitive at intervals of at least five days, the minimum period between attitude corrections. Figure 33 establishes a typical period of the profile collection showing the available contacts with both the STADAN tracking range and range-rate stations and the telemetry vhf stations.

The attitude and spin-correction phase has the following sequence:

1. Evaluation of status data from College and Rosman stations and prediction of time to reach the attitude limit.

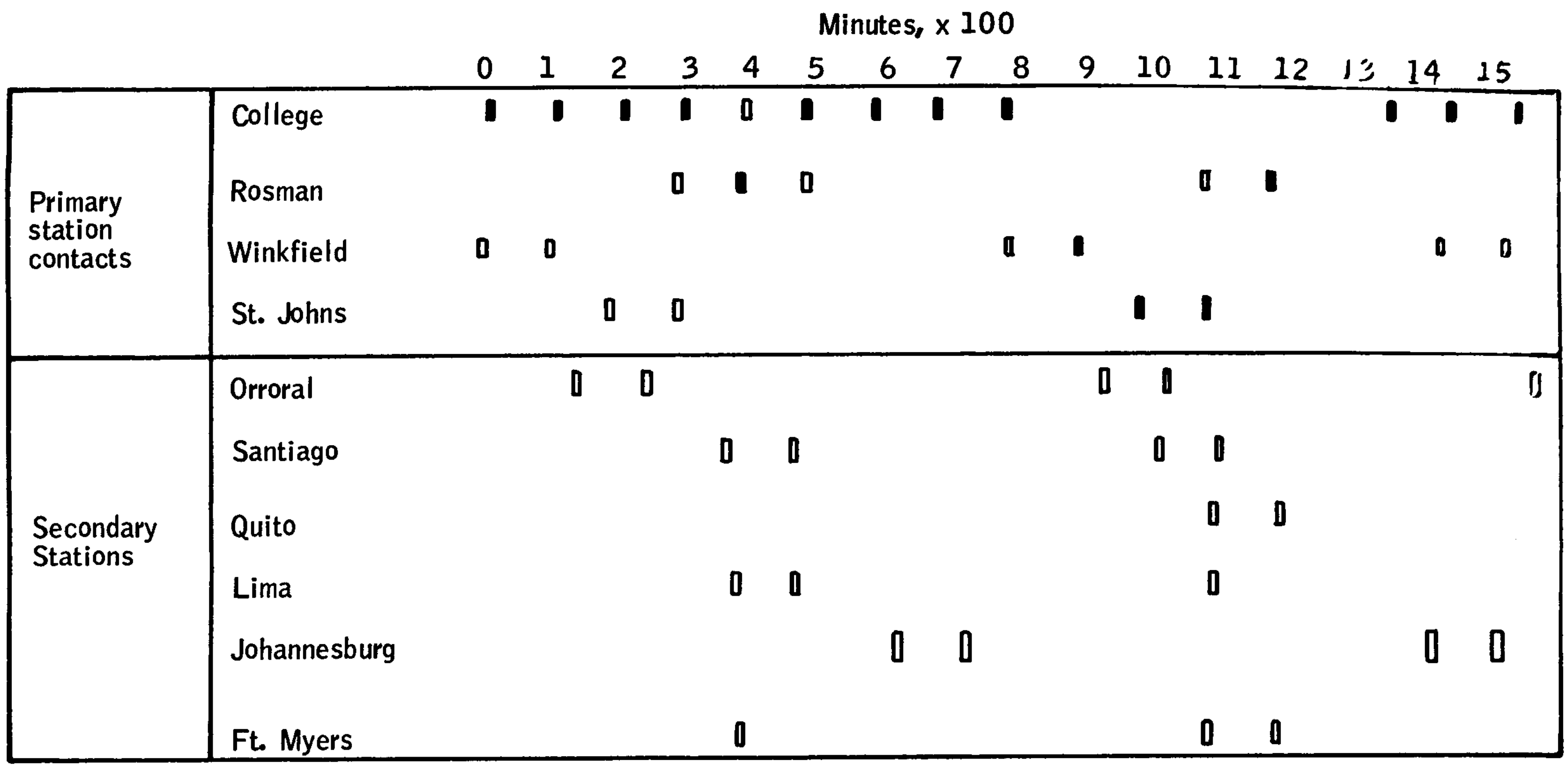


Figure 30. Telemetry Contacts for a Typical Day of Operation

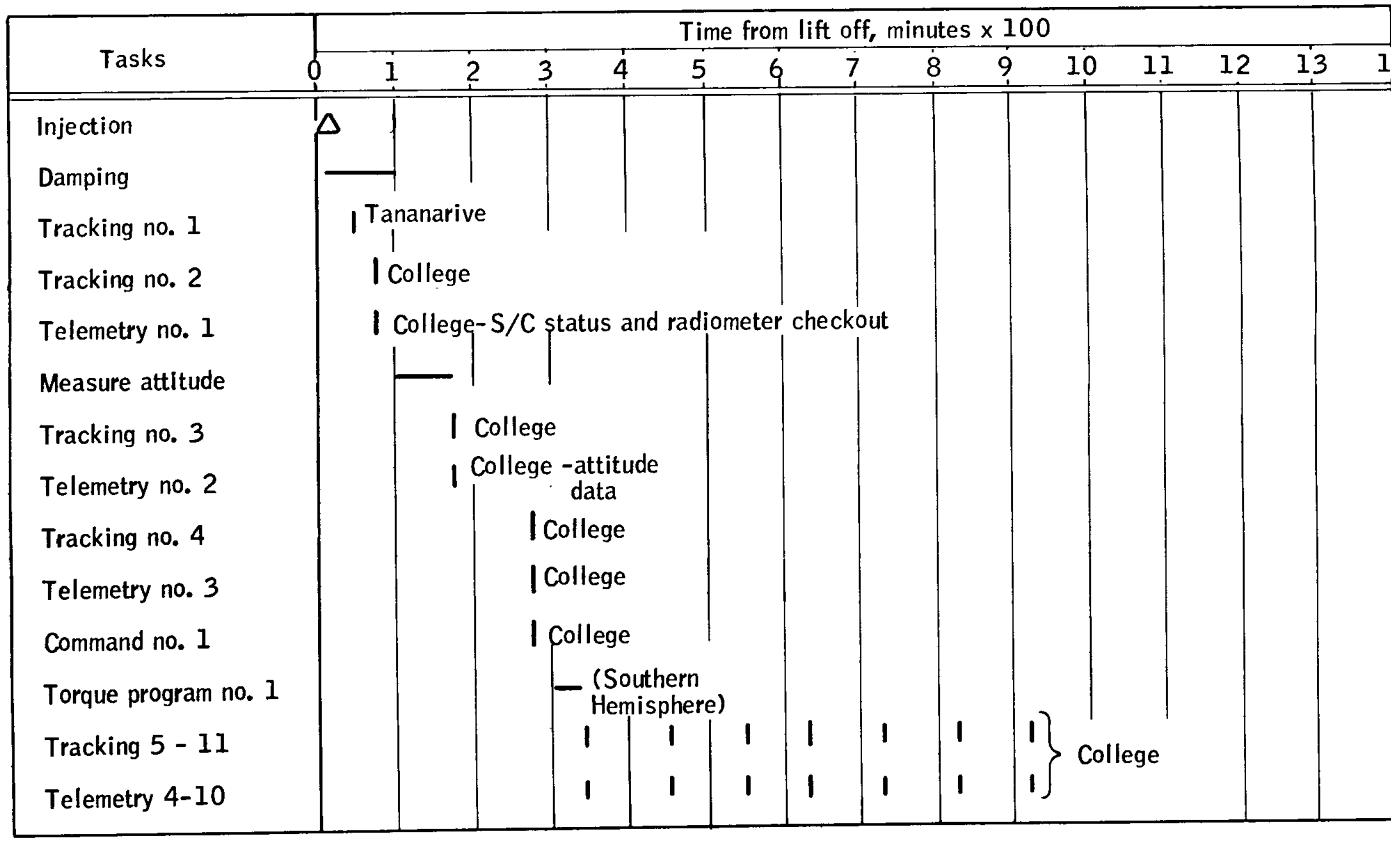


Figure 31. Attitude Trim and Magnetic Moment

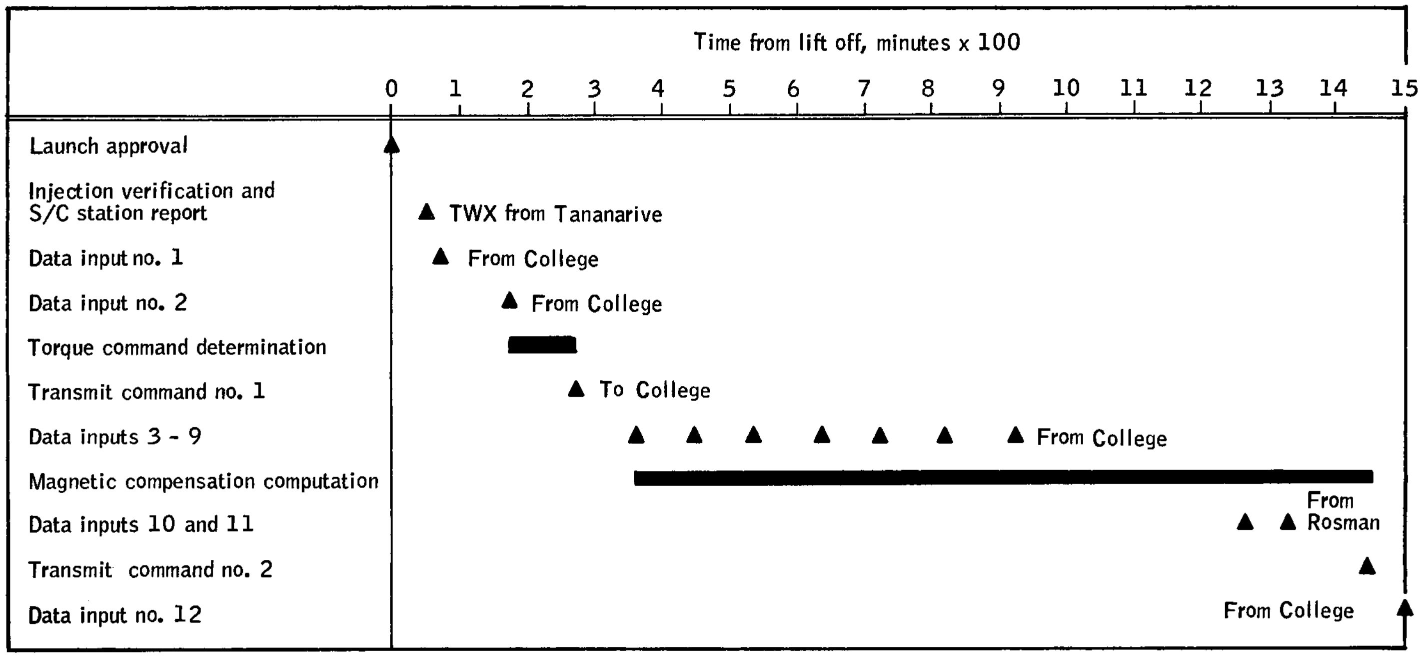


Figure 32. Mission Control Inputs and Tasks, Goddard

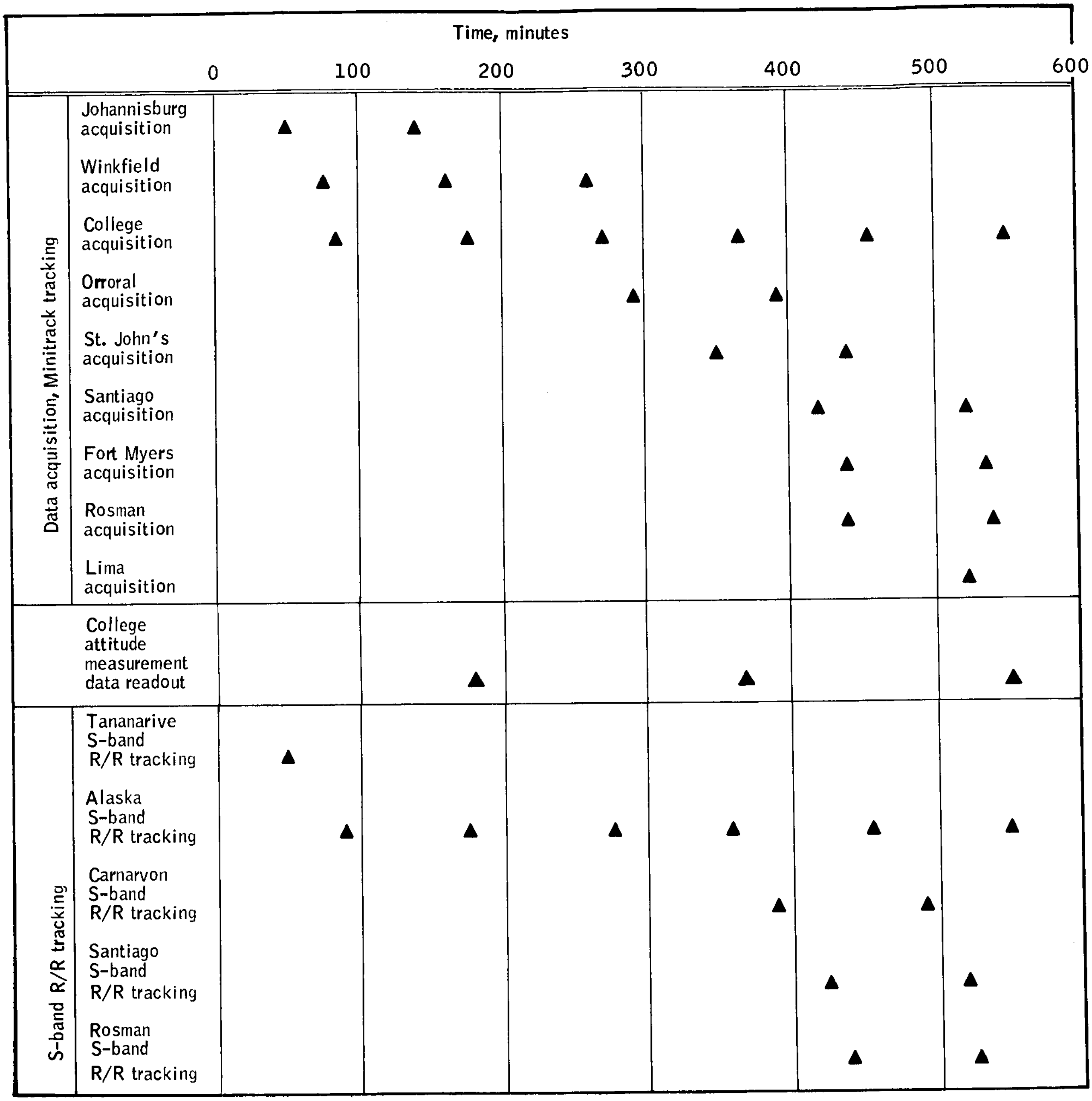


Figure 33. Operations Plan, Stations Availability, Typical Profile Collection Period, STADAN

2. Selection of the appropriate command program for both attitude change and spin correction.

3. Determine College contact immediately prior to predicted time to reach attitude limit.

4. Transmit command requirement to College.

5. Transmit command to spacecraft.

6. Command implementation by spacecraft. One-half orbit required.

7. Resume normal operation (profile collection).

The operational plan presented above does provide for the elements of the measurement program to be completed within all program requirements. The actual plan for the one-year mission will be one which evolves from spacecraft performance data. The actual STADAN coverage sequences can only be predicted for periods of several days in order to maintain adequate timing accuracies. A mission control function, therefore, will be the determination at periodic intervals (one week) of the coverage sequence and priorities.

Conclusion

The HDS mission functions can be conducted in a compatible,chronological manner as established by the existing operational plan.

The actual tracking telemetry and command functions will be established at weekly intervals, according to the priorities in existence at the time.

CONCLUSION

The systems analysis and integration conclusions are as follows:

1. The concept feasibility of the complete measurement system for the collection of one-year continuous data fulfilling the stringent radiometric and line-of-sight accuracy requirements and the need of the experiment goals has been established. The concept represents a complete integral solution of the requirements for long life and high accuracy through the use of the spacecraft's fundamental motion to satisfy the sensor's dynamic requirements resulting in passive, long-life systems.

2. The reliability of the most effective spacecraft for a one-year mission is 0.71 within all program constraints, e.g., the weight goal of less than 800 pounds. This achievement is realized only through the full exploitation of fundamentally passive systems judicous redundancy choices.

3. The multiple-flight concept will produce continuous data for one year with a high confidence and, furthermore, is expected to result in 18 months of data for a three-flight program. This period would be even longer if the cryogenic cooling system was not subject to limited life.

4. Compatibility of operational tasks has been established through the development of an operational plan covering launch operations through post-flight data analysis.

5. Scout as a potential booster for the measurement program would require significant alteration of the basic requirements and, therefore, was not carried beyond a compatibility study.

APPENDIX A
RADIOMETER ERROR ANALYSIS

APPENDIX A

RADIOMETER ERROR ANALYSIS

RADIOMETER ERROR REQUIREMENTS

Accurate analyses of the experimental data over all time and global space requires knowledge of the errors induced in the data by the measuring instruments. Correlations of these data, with the data reflecting only horizon variations, then yield acceptable accuracy requirements for the radiometers.

Effects of the radiometer errors on the data-measurement requirements were determined by perturbing a set of 120 synthesized radiance profiles with the expected types and magnitudes of radiometer errors and calculating the resultant error in the located horizon found by various horizon detection techniques.

The locations of the radiance profiles used are given in Figure A1 which shows geographical coverage from the equator to the North Pole and longitudes from West 90° to West 165°. For each of the 30 locations shown, profiles for times in April, August, October, and January were used to include seasonal effects.

Each of these profiles was perturbed by four values of six different types of radiometer error. Radiometer errors were divided into two basic errors, scale and bias. Scale errors are percentage or fractional errors; bias errors are expressed in terms of radiance magnitude, e.g., radiometer noise equivalent radiance of X.XXX W/m^2 -sr, rather than percentage. These two kinds of errors affect radiance measurements in three different ways, embodying all previously discussed concepts of accuracy and precision.

The first method of application of error is that the errors are constant in time, such as calibration errors. When a given radiometer is calibrated, all of the actual errors (sometimes called nonlinearity, offset, etc.) are calibrated out, but a residual calibration uncertainty remains (sometimes referred to in discussions of accuracy).

By analysis, test, or otherwise, the maximum value of the uncertainty is known, but within this ± range the actual value of the uncertainty is uncertain. However, the value that existed at the time of calibration remains fixed for a given instrument. That is, if we know the calibration radiance to within ±1 percent, then we know that for a given calibration the input radiance is somewhere between 0.99 N_o and 1.01 N_o. Assume that it was 0.995 $N_o p$, then a fixed calibration error of 0.5 percent did exist, which exists for that instrument for all time. These fixed errors would seem to have no effect on local vertical because they affect all profiles in the same direction and subtract out when differences are taken to find local vertical. However, they do affect located horizons since they propagate into located horizon errors as a function of profile shape and magnitude. Then although the radiometer error is a constant, the resulting located horizon error is not and, local vertical is affected.

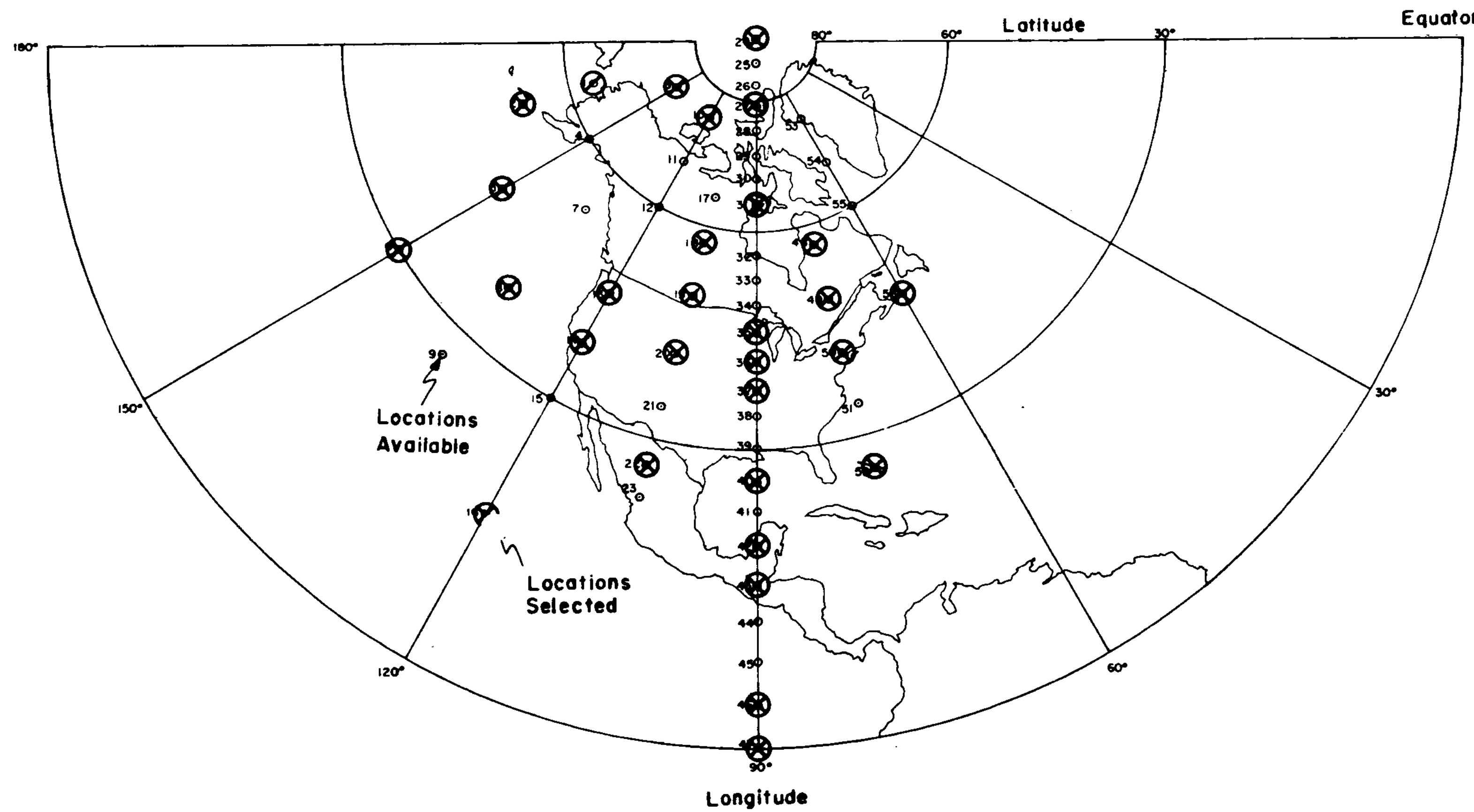


Figure A1. Synoptic Data Map

The magnitudes of thie error investigated were selected to bracket the expected value of error to be associated with the HDS radiometer calibration. Thus scale errors of 0.4 percent, 1 percent, 2 percent, and 5 percent and bias errors (all W/m^2-sr) of 0.001, 0.004, 0.01, and 0.04 were used.

The kinds of errors that vary over the operation of the devices have been referred to in discussions of precision, but since in-flight calibration will be done for the radiometer, this class of errors coming under precision or accuracy could be discussed further. Since repeated calibration is to be done, the stability of the calibration is what counts; this stability includes that of the calibration source and that of the radiometer response.

Instabilities are of two kinds, those that fluctuate rapidly like detector or electronics noise, and those that fluctuate slowly like temperature-caused drift. Without discussing the specifics of frequency distributions, it is assumed for the HDS radiometer application, that a slowly-varying error is one which has a constant magnitude over a single profile measurement but which is different for some other profile measurement. A rapidly-varying error is one which changes in magnitude from point to point on each profile being measured. Both of these are assumed to have gaussian amplitude distributions. The magnitudes chosen are the same as for calibration error.

For a given value of calibration error, all profiles were perturbed by a constant (e.g., multiplied by 1.05 for a 5-percent scale error). For a given value of drift, each profile was perturbed by a different error value selected from a gaussian distribution, e.g., for 5-percent drift, each profile was multiplied by $(1 + X)$ where X was selected from a gaussian distribution with a δ of 0.05. Similarly for noise, except that each radiance value on each profile was perturbed by a different error value selected from the applicable distribution.

The body of data operated on by different horizon detection techniques to determine horizon error thus consists of 24 different sets of 120 radiance profiles in each set. Each of these sets of profiles was operated on by four different threshold techniques with two threshold values used for each technique, as summarized in Table A1.

These results are presented in Table A2 which gives the located horizon error sensitivity coefficients in km/percent for scale error, and in km/0.1 W/m^2-sr for bias errors. Both error standard deviation, which is the contribution to horizon noise caused by the radiometer, and error mean value, which is an absolute accuracy indication, are given.

These horizon errors caused by instrument errors cause a degradation in the measurement program performance. A measure of program performance is the confidence level in obtaining a given confidence interval on the estimate of horizon noise obtained from the measured profiles. In the Part 1 study, data sampling requirements were determined which would produce a 95 percent confidence level of estimating the horizon noise to within ±0.5 km. Because of the added effect of instrument-caused horizon noise, this confidence interval must be reduced, or alternatively, the number of samples must be increased.

TABLE A1.- HORIZON DETECTION TECHNIQUES

Threshold technique	Threshold values
Radiance magnitude	2.0, 3.0 W/m^2-sr
Normalized radiance magnitude (% of peak radiance)	.15, .90 (15%, 90%)
Integral of radiance	4.5, 20.0 $W\text{-}km/m^2$-sr
Integral of normalized radiance	2.5, 10.0 km

The radiometer error analysis leads to the following equation relating the parameters of interest.

$$I_c = \left[\frac{2\sigma_2^2 \left(1 + \frac{\sigma_1^2}{\sigma_2^2}\right)^2 + m\,(N-1)\,k^2 \frac{\sigma_1^2}{\sigma_2^2}}{m\,(N-1)\,(1-P/100)} \right]^{1/2}$$

where

I_c = confidence interval on the estimate of natural horizon noise

σ_2 = horizon noise in nature

σ_1 = instrument-caused horizon noise

m = number of time cells

N = number of samples per time-space cell

k = fractional uncertainty in the estimate of σ_1.

The approach taken here is to allow the confidence level to decrease to 90% from 95% rather than to increase the sample size. The resultant allowable instrument-caused horizon error is determined from Figure A2 which shows the confidence level P versus instrument-caused error σ_1 for various values

TABLE A2. - SENSITIVITY COEFFICIENTS

Locator	Scale, km/%			Bias, km/0.01 W/m^2-sr		
	Cal	Drift	Noise	Cal	Drift	Noise
Std dev						
L1 (2.0)	.02	.13	.10	.01	.06	.05
L1 (3.0)	.04	.18	.13	.015	.05	.05
L2 (0.15)	0	0	.08	.02	.09	.08
L2 (0.9)	0	0	.48	.002	.008	.10
L3 (4.5)	.006	.07	.026	.06	.04	.15
L3 (20.0)	.008	.10	.023	.27	.19	.053
L4 (2.5)	0	0	.06	.04	.22	.07
L4 (10.0)	0	0	.08	.02	.10	.03
Mean error						
L1 (2.0)	.12	-.02	.004	.06	-.006	-.01
L1 (3.0)	.15	-.03	-.004	.05	-.006	-.008
L2 (0.15)	0	0	-.116	.09	-.01	-.029
L2 (0.9)	0	0	-.625	.008	-.001	-.077
L3 (4.5)	.06	-.007	-.004	.41	-.038	-.006
L3 (20.0)	.09	-.01	0	.18	-.018	-.003
L4 (2.5)	0	0	-.12	.21	-.02	-.02
L4 (10.0)	0	0	-.19	.10	-.01	-.03

of horizon noise σ_2 and k, the fractional uncertainty in σ_1. At the 90% confidence level, the following instrument-caused errors are allowable:

σ_1 = 0.3 km ±31.6%; 0.34 km ±20%; 0.38 km ±10%; 4 km ±0%

To simplify the radiometer design problem, both the allowable error and its tolerance should be made as large as possible. However, as the allowable error increases, the allowable tolerance decreases necessitating a compromise. The combination selected as being a reasonable compromise is an instrument caused error σ_1 of 0.34 km, known to within ±20¢.

For this instrument-caused error, the allowable values of radiometer scale and bias errors are determined from Table A2.

Using the sensitivity coefficients of Table A2, the locator exhibiting the maximum sensitivity for each error was constrained to produce a horizon error of 0.34 $\sqrt{6}$ km in order to guarantee that the rss of the six errors did not exceed 0.34 km. For each error, this approach led to a maximum allowable instrument error.

Since a given locator exhibits high sensitivity to one kind of error and low sensitivity to another, the approach described above led to a root-sum-square horizon error considerably lower than the maximum allowable 0.34 km. Certain adjustments were made in the errors known to cause difficulty in the radiometer design process (bias, calibration, and drift) to relieve the requirements and remain within the allowable 0.34 km error.

The resulting allowable radiometer errors and the resultant root-sum-square horizon standard deviation and overall mean error are shown in Table A3.

These radiometer errors become radiometer specifications, defining the maximum allowable scale and bias errors for calibration drift and noise. For clarity they are restated here:

Scale calibration	.3 percent
Scale drift	.72 percent
Scale noise	.27 percent
Bias calibration	.01 W/m^2-sr
Bias drift	.01 W/m^2-sr
Bias noise	.01 W/m^2-sr

TABLE A3. - ALLOWABLE INSTRUMENT-CAUSED HORIZON ERROR

Locator	Sum	Scale			Bias		
		Cal 3%	Drift 0.72%	Noise 0.27%	Cal	Drift	Noise
Std dev							
L1 (2.0)	.139	.06	.094	.027	.010	.060	.050
L1 (3.0)	.197	.12	.130	.035	.015	.050	.050
L2 (0.15)	.126	0	0	.022	.020	.090	.080
L2 (0.9)	.164	0	0	.130	.002	.008	.100
L3 (4.5)	.175	.018	.050	.007	.060	.040	.150
L3 (20.0)	.34	.024	.070	.006	.270	.190	.053
L4 (2.5)	.235	0	0	.016	.040	.220	.070
L4 (10.0)	.109	0	0	.022	.020	.100	.030
Mean							
L1 (2.0)	.393	.36	-.014	.001	.06	-.006	-.01
L1 (3.0)	.463	.45	-.022	-.001	.05	-.006	-.008
L2 (0.15)	.019	0	0	-.032	.09	-.01	-.029
L2 (0.9)	-.239	0	0	-.169	.008	-.001	-.077
L3 (4.5)	.54	.18	-.005	-.001	.41	-.038	-.006
L3 (20.0)	.422	.27	-.007	0	.18	-.018	-.003
L4 (2.5)	.137	0	0	-.003	.21	-.02	-.020
L4 (10.0)	-.009	0	0	-.051	.10	-.01	-.030

APPENDIX B
SCOUT COMPATIBILITY STUDIES

APPENDIX B
SCOUT COMPATIBILITY STUDIES

The basic systems requirements allow a satellite weight of 800 pounds. This weight can be placed into the required near-polar orbit by a number of the family of Thor-Delta launch vehicles. To reduce significantly the cost of the measurement program, it was required to investigate the compatibility requirements, i.e., program modification to utilize the next smaller launch vehicle, namely the Scout. This investigation is reported herein, first by discussing the effect of selected changes to basic requirements and then by describing the subsystems resulting from extrapolation of the "baseline" fully responsive system as a result of the requirements changes. Potential Scout growth capabilities are presented and finally a discussion of the major feature of a Scout program completes the Scout compatibility study.

It should be noted that this study was completed approximately midway in Part II; therefore, final subsystem definitions are not reflected.

SCOUT BASIC REQUIREMENTS

The basic requirements changes represent relaxations necessary which would not affect either the quality or quantity of the data resulting from a measurement program. These changes are in the areas of instrument redundancy, spacecraft design life, and orbit profile. Each of these is discussed below.

Instrument Redundancy

The requirement for redundant experimental package instruments represents a significant weight penalty. To determine the minimum spacecraft weight, it is necessary to consider only a "single thread" system, i.e., each required function implemented once.

Spacecraft Design Life

The compatibility guidelines establish a design life of a single spacecraft of one year. The implementation of this requirement, in the absence of a reliability goal, does not directly introduce weight increases except for the case of subsystems which employ mass. The single subfunction in this category is the cryogenic cooler which increases nearly linearly with required life over a basic minimum system weight. Therefore, the most significant result of a reduction in design lifetime is reduced cryogenic cooler weight; however, a more intangible but potential benefit could be subsystem redundancy to achieve a level of reliability and associated confidence to warrant a spacecraft launch.

Orbit Profile

The design mission profile is a sun-synchronous orbit with nodal crossings at 3:00 a. m. and 3:00 p. m. This choice satisfied the requirement for near-polar orbit and was selected over a polar orbit due to the reduction of spacecraft design problems associated with sun relationship, i. e., a true polar orbit results in all conditions of angles on all spacecraft surfaces during a year's mission. The nodal-crossing choice resulted from consideration of the diurnal content of the profile, being selected to reflect the maxima and minima radiance levels. However, the most favorable sun-synchronous orbit from the standpoint of subsystem design is one having nodal crossings at 6:00 a. m. /6:00 p. m., i. e., a "twilight" orbit. The twilight sun synchronous allows a more efficient solar electrical power system, minimizes any required instrument baffling and has a secondary cryogenic-cooler weight reduction effect due to greater efficiencies of the radiative cooling employed in the spacecraft. The penalty accompanying the twilight orbit is the loss of any significant diurnal variation.

Given the above changes in requirements for elimination of redundancy, reduced spacecraft design life, and selection of the orbit most favorable to subsystem design problems, an extrapolation from an existing fully responsive baseline system was made.

SCOUT SYSTEM DESCRIPTION

The subsystems were evaluated for the potential optimistic weight reduction resulting from the requirements as modified. The following summary includes the significant rationale for the new weight estimate. Comparative weights are those for the baseline system.

	Weight, lbs
Experiment package	146

- No redundant devices utilized
- Baffling considered to be minimum
- Six-month cooling system nominal
- Radiometer/Starmapper, 40 pounds versus 120 pounds
- Cooling,100 pounds versus 150 pounds
- Structures,6 pounds versus 10 pounds

Note: not considered was cooled radiometer starmapper optics

	Weight, lbs
Power supply	40

- Body-mounted solar cells
- Output 35 watts with 24-percent maximum shadow
- 40 pounds versus 66 pounds

Note: A weight of 26 pounds may be feasible; however, growth requirements in power and shadow fraction and injection errors contribute to a larger weight budget. Also a high power data handling system is being considered.

Attitude control — 29

- Magnetic torque coils for control of attitude and spin (smaller due to lighter satellite)
- Despin jets required (LTV studying guided final stage)
- Solid versus liquid spin-up jets
- Redundancy (dual electronics) eliminated
- Support and coil weight cut 40 percent
- 29 pounds versus 60 pounds

Note: Fifty pounds weight considered only reduction in coil weight and elimination of spin-up jets.

Communications — 25

- No change

Note: Nineteen pounds weight was an earlier weight estimate

Data handling — 45

- Minimum power configuration with weight penalty
- 45 pounds versus 32 pounds

Note: Twenty-five pounds weight was derived by reducing memory size but not type. However profile data loss is very high due to potential requirement for continuous attitude determination (starmapper) data. The requirement to consider data down to the minimum level has been dropped.

	Weight, lbs
Structures	70

- 70 pounds versus 120 pounds

  Note: Forty pounds weight was based upon a 15-percent structure allowance representing a difficult design requirement. Raising to a more feasible 20 percent results in new figure.

Total	355

The key problems are the radiometer cryogenic cooler weight and the subsystem redundancy required to achieve an acceptable total failure rate (e.g., attitude control type redundancy deleted in this configuration). Based upon this evaluation the existing Scout, with a payload capacity of approximately 270 pounds in a sun-synchronous orbit, could not satisfy the requirements for the HDS launch vehicle. However, potential improvements were investigated and are reported in the following section.

POTENTIAL SCOUT GROWTH

Two growth possibilities were identified which would impact on the conclusions as to Scout compatibility. These are an up-rated first state and a guided final stage. These modifications would not affect the Scout payload envelope shown for reference in Figure B1.

Up-rated First Stage

Increases in payload capability up to 400 pounds in the HDS orbit are under consideration. This would be mandatory to further serious consideration of refining a Scout satellite and measurement program. Upon investigation, the up-rating program was not committed beyond a study.

Guided Final Stage

The change from a spin-stabilized final stage to a guided stage would improve the injection accuracies to the level required to prevent unwanted orbit precession to eliminate the advantages from reversing the nominal orbit nodal crossing to 6:00 a.m./6:00 p.m. (twilight) and could therefore eliminate subsystems whose operation is for limited sun-angle conditions. This is only a contractor (LTV) in-house study, however, and is not currently scheduled for implementation.

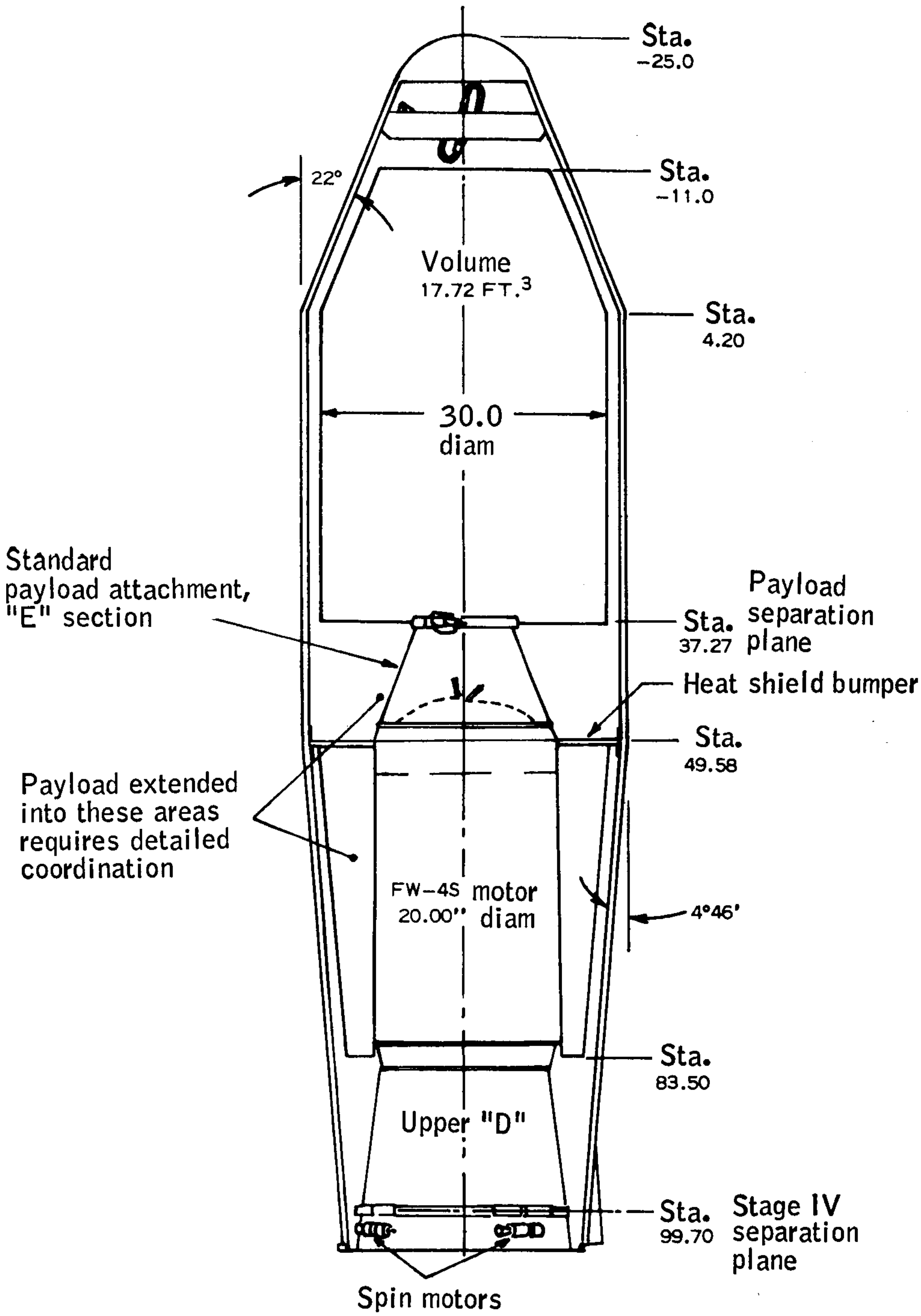


Figure B1. Scout Payload Envelope

PROGRAM CONSIDERATIONS

A program level analysis was made of the comparative cost effectiveness of an HDS spacecraft compatible with the Scout launch vehicle to provide a one-year-mission data profile. The parameters considered were:

- 1, 1/2, 1/3, and 1/4-year spacecraft operational lifetime
- Spacecraft failure rates from 10-30 percent per 1000 hours
- Booster reliabilities $0.85 \leq R \leq 0.95$
- Nominal injection accuracies, 2-stage Delta and 4-stage (spin-stabilized fourth stage) Scout
- Booster launch readiness - 5 to 90 days

The significant results are shown on Figure B2. This curve indicates that lifetime less than six months with Scout are not cost effective with one year lifetimes on Delta. This is due to the initial inventory and standby costs for hardware. This comparison does not include development costs, which should be approximately equal for either alternative.

The estimated comparative weight summary is given in Table B1. These changes reflect no redundancy in the experiment package, a six-month, nominal spacecraft operational life with reduction in cryogenic cooler weight, and a sun-synchronous twilight orbit for electrical power system efficiency.

TABLE B1. - ESTIMATED WEIGHT SUMMARY

Subsystem		Scout	Delta
Experiment package		146	280
Power supply		40	66
Attitude control		29	60
Communications		25	25
Data handling		45	32
Structures		70	120
	Total weight	355	583
Lifetime		6 months	12 months

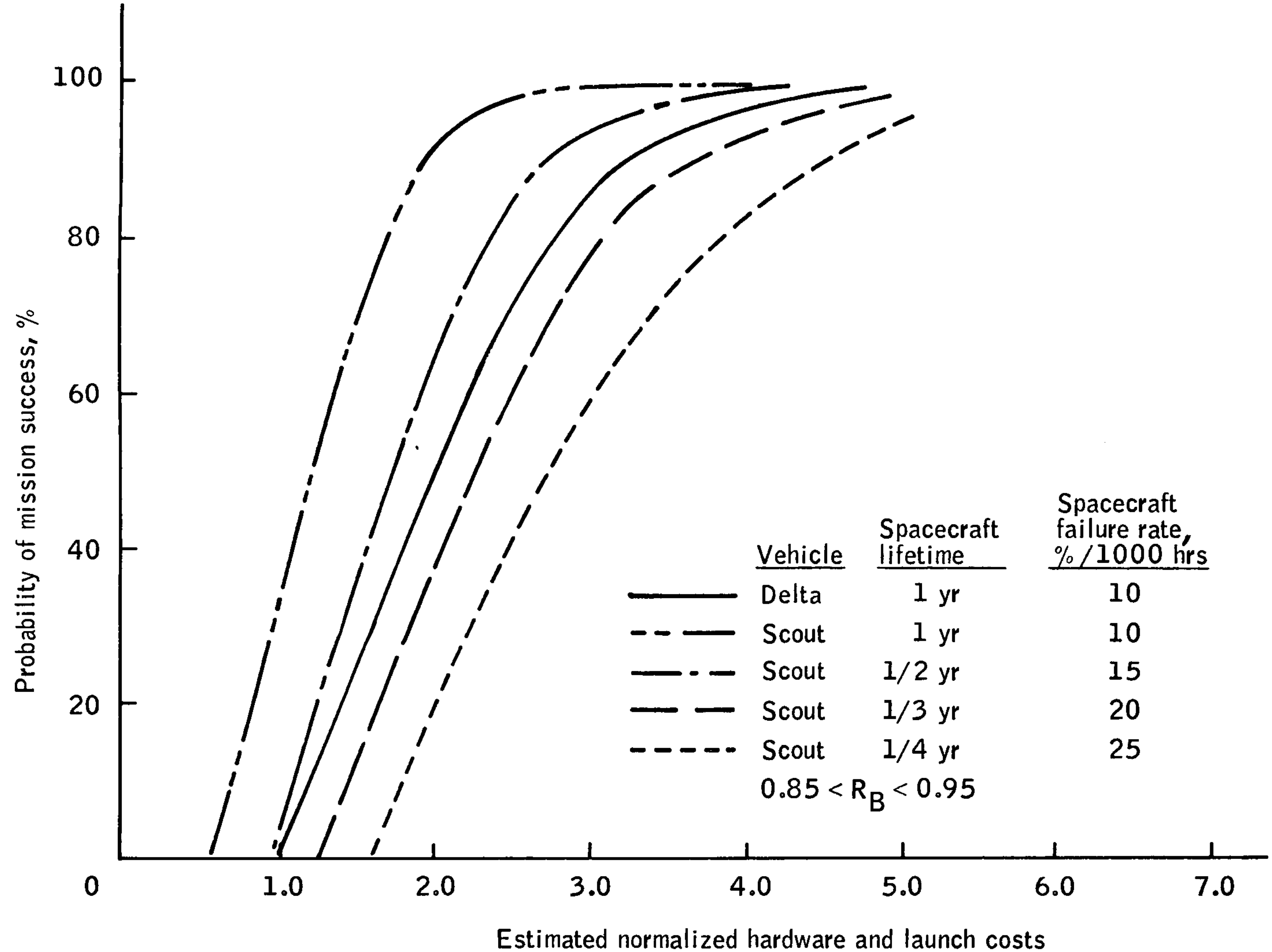


Figure B2. Comparative Launch Costs and Performance Envelope, Delta and Scout Vehicles

CONCLUSIONS

Using these changes of the HDS mission requirements, a spacecraft compatible with the Scout booster was configured. The 355-pound spacecraft could be compatible with an advanced Scout having a payload capability of 400 pounds.

The 355-pound conceptual design is not compatible with the total HDS mission requirements objectives or the present Scout booster and was, therefore, not considered as feasible.