

Systems Cost/Performance Analysis (Study 2.3) Final Report

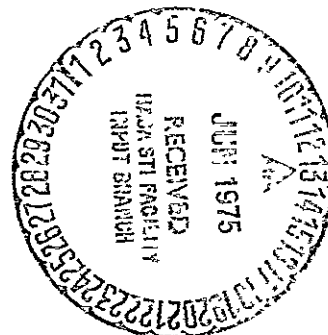
Volume II: Systems Cost/Performance Model

Prepared by

ADVANCED MISSION ANALYSIS DIRECTORATE
Advanced Orbital Systems Division

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Prepared for

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(STUDY 2.3) FINAL REPORT

Volume II: Systems Cost/Performance Model

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FOREWORD

This report documents The Aerospace Corporation effort on Study 2.3, Systems Cost/Performance Analysis, performed under NASA Contract NASW-2575 during Fiscal Year 1974. The effort was directed by Mr. B. H. Campbell. Mr. R. D. Kramer, Marshall Space Flight Center and Mr. R. R. Carley, NASA Headquarters were the NASA Study Directors for this study. Their efforts in providing technical direction throughout the duration of the study are greatly appreciated.

This volume is one of three volumes of the final report for Study 2.3. The three volumes are:

Volume I	Executive Summary
Volume II	Systems Cost/Performance Model
Appendix	Data Base
Volume III	Programmer's Manual and User's Guide

Volume I summarizes the overall report. It includes the relationship of this study to other NASA efforts, significant results, study limitations, and suggested additional effort.

Volume II provides a detailed description of the Systems Cost/Performance Model. It also includes the model checkout and the results for three payload test cases. The Data Base is provided in the Appendix to Volume II.

Volume III provides a detailed description of how the Systems Cost/Performance Computer Program is organized and operates. The program listing, detailed flow charts and user restrictions are included.

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1. INTRODUCTION

1.1 General

During the preceding year a new methodology for developing balanced designs of spacecraft subsystems which interrelates cost, performance, safety, and schedule considerations was developed as part of the FY 1973 NASA Study 2.3 (See Ref. 1-1). The methodology consists of a two-step process. The first step is one of selecting all hardware designs which satisfy the given performance and safety requirements. The second step is one of estimating the cost and schedule required to design, build, and operate each spacecraft design. Using this methodology to develop a systems cost/performance model allows the user of such a model to establish specific designs and the related costs and schedule. In addition, the user is able to determine the sensitivity of design, costs, and schedules to changes in requirements.

Previous cost modeling approaches fall into one of two basic categories: "bottom-up" or "top-down." The "bottom-up" approach depends on development of a specific system. Detailed estimates of tasks, material costs, manpower requirements, and schedules are made, and total estimates are obtained by summing individual costs and task durations.

"Top-down" models use CER (cost estimating relationship) approaches to estimate the cost of a specific system (See Ref. 1-2). In these models, the CERs are related to distinct parameters such as weight, power, and pointing accuracy. The deficiency of the CERs lies in that, although they identify what are cost drivers, CERs do not model why and how the costs are driven by the parameters.

Since CERs have not been completely successful in meeting the prime criterion of determining sensitivity of cost to changes in program requirements, top-down approaches were judged unacceptable for a cost/

performance model. Hence, it was thought that a model oriented from the bottom-up could lead to fulfillment of this criterion.

During the FY 1973 effort, a set of basic equations, termed "aggregate equations," were written to describe the performance, safety (reliability), cost, and schedule required for one type of Stabilization and Control Subsystem in terms of the equipment used. This year's effort was devoted to refining the methodology and applying it to unmanned, automated spacecraft subsystems.

1.2 Objectives

This year's study had three objectives. The first objective was to refine and improve the cost/performance methodology which was developed during the preceding fiscal year's study. The same two-step process of first establishing hardware designs and then estimating costs and schedules was retained. However, incomplete portions of the methodology such as the cost and schedule models were to be improved. A product of this year's effort is the Systems Cost/Performance Model shown in Figure 1-1.

The second objective was the application of the cost/performance methodology to the following vehicle subsystems:

- a. Stabilization and Control (S&C)
- b. Auxiliary Propulsion Subsystem (APS)
- c. Communications, Data Processing and Instrumentation (CDPI)
- d. Electrical Power (EP)
 1. Sources
 2. Conditioning and Distribution
- e. Thermal Control Subsystem (TCS)
- f. Structure

The third objective was to implement the Systems Cost/Performance Model as a digital computer program. The program would be used to perform initial program planning, cost/performance tradeoffs, and sensitivity analyses.

1-3

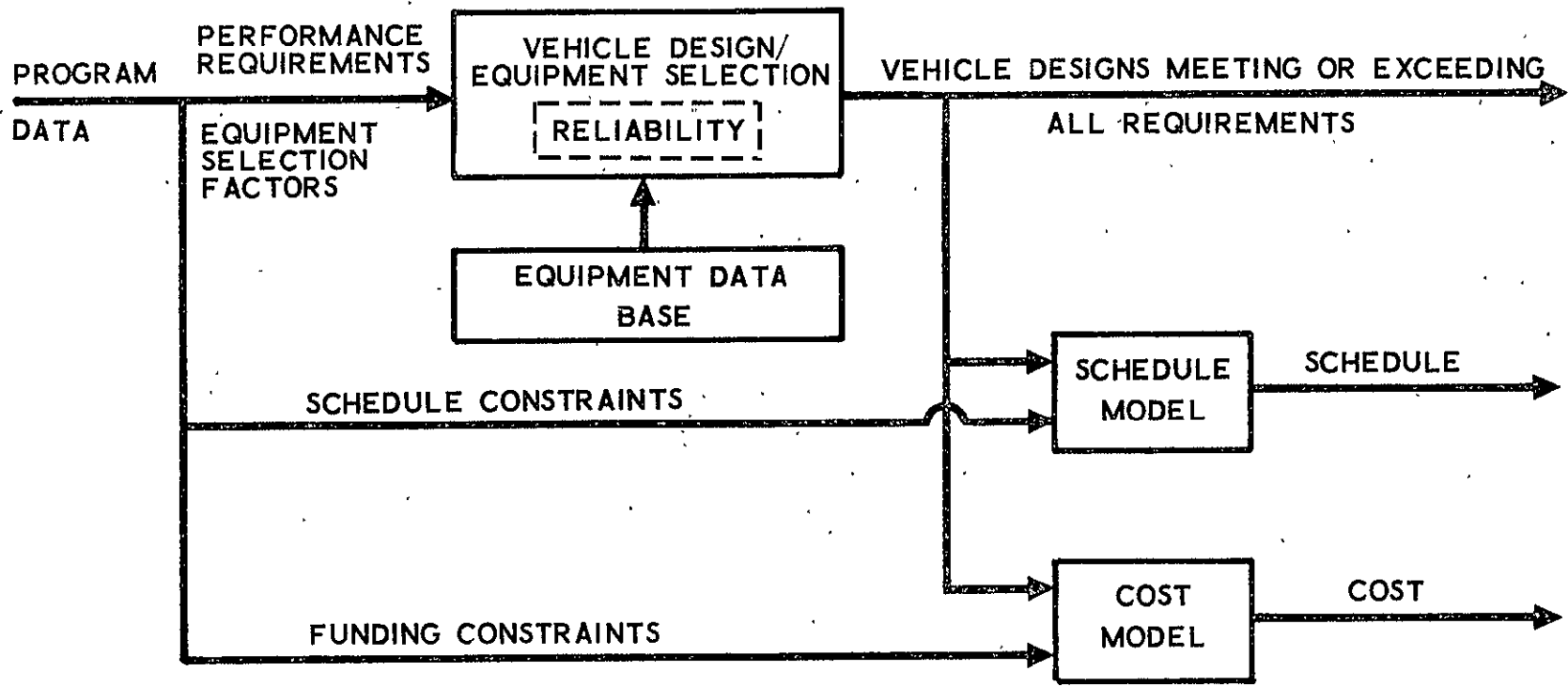


Figure 1-1. Systems Cost/Performance Model

1.3 Approach

The general format of the vehicle design/equipment selection model is presented in Figure 1-2. The subsystems to be modeled are represented by solid boxes. Mission equipment, which was not modeled as part of the FY 1974 effort, is represented by a box with broken lines. The modeling applies to unmanned, automated spacecraft including:

- a. Defense System Communication Satellite (DSCS II)
- b. Defense Support Program (DSP)
- c. Earth Observatory Satellite (EOS)
- d. High Energy Astronomy Observatory (HEAO)
- e. Large Space Telescope (LST)

The vehicle subsystems' models have four key parts:

- a. The input data required to establish a specific design and the necessary equipment.
- b. An algorithm which selects acceptable designs and the hardware required to implement the designs.
- c. The output data including a description of the design, the equipment list associated with the design, and any other data required to interface with other portions of the Cost/Performance Model.
- d. A data base consisting of off-the-shelf hardware from which the design algorithm can select.

The following tasks were performed in order to ensure that the models would be as complete as possible.

1.3.1 Functional

One of the first tasks was to determine the functions performed by each subsystem and the functions performed by specific hardware types within the subsystem. Obviously, interfaces between subsystems determined some of the functions to be performed. The outline of functions to be performed had to be complete since potential subsystem designs are, for the most part, related directly to the functions they are required to perform.

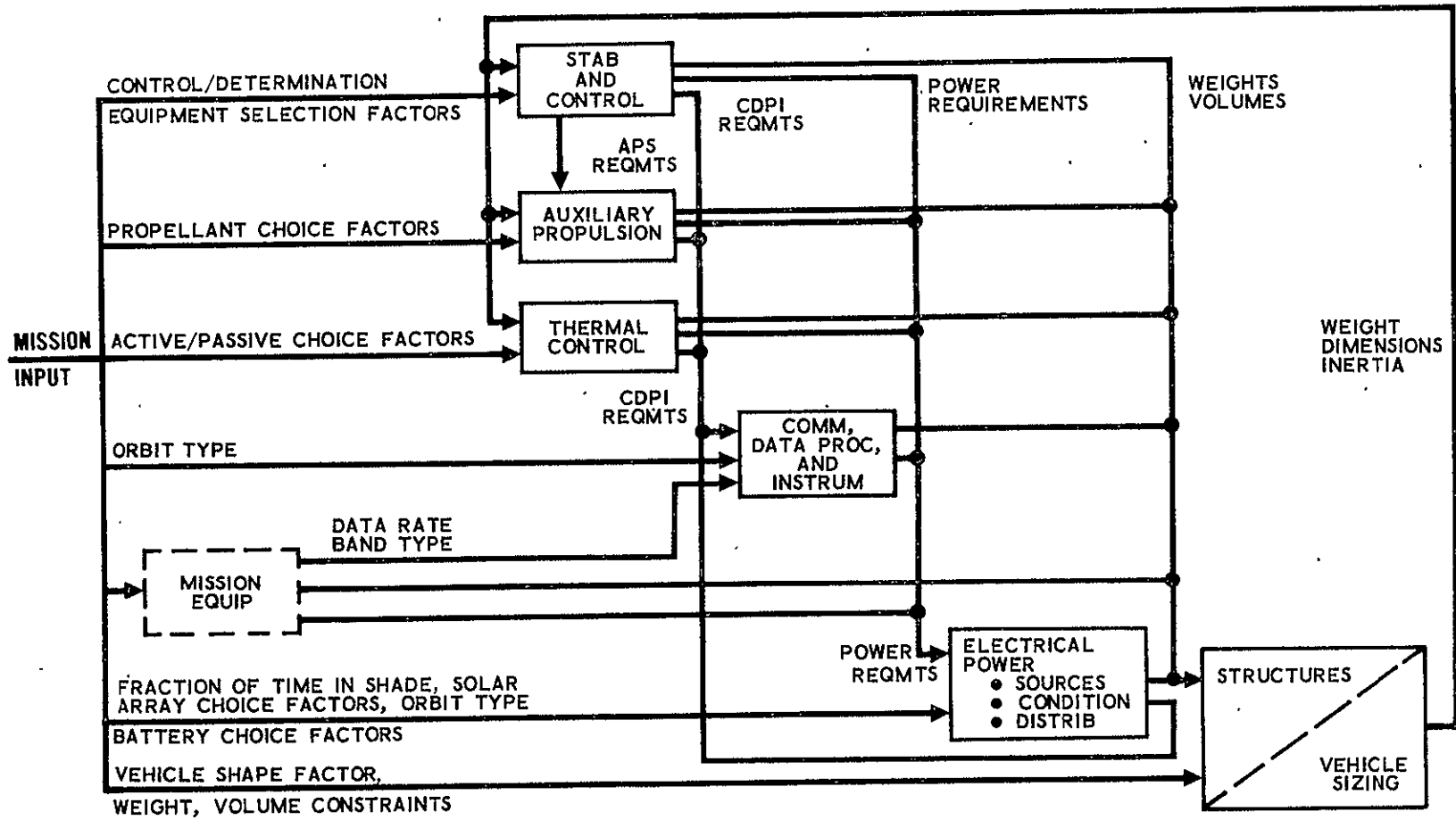


Figure 1-2. Vehicle Design/Equipment Selection

1.3.2 Block Diagrams

Block diagrams were developed for all generally used subsystem configurations. The block diagrams contained the equipment types used in each configuration and illustrated the functions performed by the equipment. Since there may be an infinite number of block diagram variations, the designer established certain general block diagrams that were valid for most designs.

1.3.3 Design Algorithm

The design algorithm performs the function of selecting pre-configured subsystem designs which will meet the input requirements. This implies that, as part of the vehicle design algorithm, a complete set of alternative designs has been established from which to choose.

The design algorithm consisted of a composite of logical and arithmetic operations. An example would be to determine whether to use gaseous nitrogen or bipropellant and then what quantity for the Auxiliary Propulsion Subsystem. A logical decision as to which propellant should be used may be made on the basis of the required total impulse. The amount of the propellant can be calculated by knowing the required total impulse and the propellant's specific impulse. This example incorporates a simple logical decision (based on certain assumptions and approximations) with an arithmetic calculation. In addition, the algorithm result may be overridden if the program input data specifies that a particular propellant (e.g., gaseous nitrogen) will be used.

1.3.4 Hardware Selection

Given any specific design meeting the input requirements, the hardware (and software) required to implement such a design must be selected and sized. The hardware is selected from available off-the-shelf hardware which is listed in the data base. Obviously, the model must be capable of differentiating between hardware components of the same type and of determining which hardware component has the characteristics to satisfy all of the requirements.

1.3.5 Input Data

In order to have a workable algorithm, the exact input data list required to select a design and size the necessary equipment has been established. The input data would normally include subsystem performance requirements, interface requirements, and any other data necessary to make design decisions.

1.3.6 Aggregate Equations

The aggregate equations are a set of basic equations describing the technical performance, safety (reliability), cost, and schedule in terms of the equipment used in the specific configuration. A list of the System Cost/Performance Model parameters described by aggregate equations is presented in Table 1-1.

As an example, the aggregate equation for the pointing accuracy (which is a technical characteristic) of a three-axis Stabilization and Control Subsystem (in this example, a specific design type) considers variables such as horizon sensor noise and misalignment, gyroscope drift and misalignment, amplifier noise and offset, and electronic deadzone. Each of these variables is multiplied by a computed sensitivity coefficient and combined in either a worst case or a root-sum-square fashion to form the aggregate equation for the S&C pointing accuracy.

Generally, the technical characteristics and safety aggregate equations were used to ascertain whether a specific design with specific hardware satisfied the input requirements. The remaining model parameters, including cost and schedule, were used as output variables describing the specific design's characteristics. However, any of the parameters could be specified as an input parameter. Examples would include weight, volume, cost, or schedule constraints.

Table 1-1. Model Parameters

Subsystems

Configurations

Equipment types

Equipment

1. Performance

1.1 Technical characteristics

- | | | |
|--------|---|---|
| 1.1.1 | } | Subsystem-peculiar; no more than ten items per subsystem; does not include items listed below |
| 1.1.10 | | |

1.2 Power

- | | |
|-------|-------------------------------|
| 1.2.1 | Average power |
| 1.2.2 | Peak power |
| 1.2.3 | Minimum power |
| 1.2.4 | Nominal voltage |
| 1.2.5 | Maximum voltage specification |
| 1.2.6 | Minimum voltage specification |

1.3 Weight

1.4 Volume

- | | |
|-------|--------------|
| 1.4.1 | Total volume |
| 1.4.2 | Length |
| 1.4.3 | Width |
| 1.4.4 | Height |

1.5 Inertia

- | | |
|-------|----------|
| 1.5.1 | I_{xx} |
| 1.5.2 | I_{yy} |
| 1.5.3 | I_{zz} |

1.6 Vibration specification

- | | |
|-------|-----------|
| 1.6.1 | Random |
| 1.6.2 | Nonrandom |

Table 1-1. Model Parameters (Continued)

- 1.7 Temperature specification
 - 1.7.1 Maximum temperature limit
 - 1.7.2 Minimum temperature limit
- 1.8 Ambient pressure specification
- 2. Safety
 - 2.1 Reliability assessment
 - 2.1.1 MMD
 - 2.1.2 Reliability
 - 2.1.3 Reliability truncation time
 - 2.2 Failure detection probability (fault isolation)
 - 2.3 False alarm probability
- 3. Cost
 - 3.1 DDT&E
 - 3.1.1 Design engineering
 - 3.1.2 Tooling and test equipment
 - 3.1.3 Qualification units
 - 3.1.4 Test and evaluation
 - 3.1.5 Quality control
 - 3.1.6 Systems engineering and integration
 - 3.1.7 Program management
 - 3.2 Investment (Recurring)
 - 3.2.1 Engineering
 - 3.2.2 Production
 - 3.2.3 Tooling and test equipment
 - 3.2.4 Quality control
 - 3.2.5 Systems engineering and integration
 - 3.2.6 Program management
 - 3.3 Operations

Table 1-1. Model Parameters (Continued)

- 4. Schedule
 - 4.1 Component design and development
 - 4.2 Component qualification
 - 4.3 Subsystem development
 - 4.4 Subsystem qualification
 - 4.5 System test, checkout and flight readiness

1.3.7 Data Base

A data base consisting of off-the-shelf hardware was established. The data content would revolve around each hardware component. The data for each component would consist of four types:

- a. Performance
- b. Reliability
- c. Cost
- d. Schedule

The four types of data would contain sufficient information to:

- a. Allow the equipment selection algorithm to select specific pieces of equipment
- b. Allow the aggregate equations to be computed
- c. Provide necessary output data

The data was collected from in-house, Air Force, and NASA sources. Selecting the equipment components to be incorporated in the data base was the responsibility of the technical specialists. Reliability, schedule, and cost data for each equipment component was obtained by the reliability, schedule and cost specialists.

1.3.8 Computer Program Model

The Systems Cost/Performance Model was implemented as a digital computer program. The program was written in the language of Fortran IV as adapted to the CDC 7600 computer and the Univac 1108 computer (for use at MSFC). The program included the Systems Cost/Performance Model (presented in Figure 1-1) and the related data base.

1.3.9 Model Checkout

Two forms of model checkout were performed. The first was a set of computer runs to ensure that both the logic and arithmetic models were accurate and complete and that all submodels were interfacing properly. The second set of computer runs was limited to a few special runs, selected for the purpose of comparing the Systems Cost/Performance Model against actual cost, performance, safety and schedule data and against other existing models.

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2. MODEL OVERVIEW

2.1 GENERAL

2.1.1 Systems Cost/Performance Model

The general concept of the Systems Cost/Performance Model was illustrated in Figure 1-1 and is repeated in Figure 2-1 for the convenience of the reader. The user of the Cost/Performance Model must supply certain program data which would normally include the payload performance requirements as well as general information necessary to select a payload design. The technical portion of the model consists of a two-step process: the first step is to select subsystem configurations which are acceptable to the user, and the second step is to select equipment from a data base to mechanize the subsystem configuration. The reliability portion of the model adds redundancy to the design such that the reliability requirements are met. The resulting output of the technical model is any number of payload designs which meet or exceed the input requirements. The acceptable designs are specified down to the subsystem component (assembly) level. The cost and schedule required to design, build, and operate each payload design is estimated by summing up the individual cost and schedule allocations based on each end item assembly specified as part of the particular design.

2.1.2 Subsystem Interaction

The technical portion of the Systems Cost/Performance Model was depicted in Figure 1-2 and is also repeated for the convenience of the reader in Figure 2-2. The expanded detail summarizes the inputs required by each subsystem.

Most importantly, the interaction between subsystems as a design problem is illustrated. In order to design the Stabilization and Control (S&C)

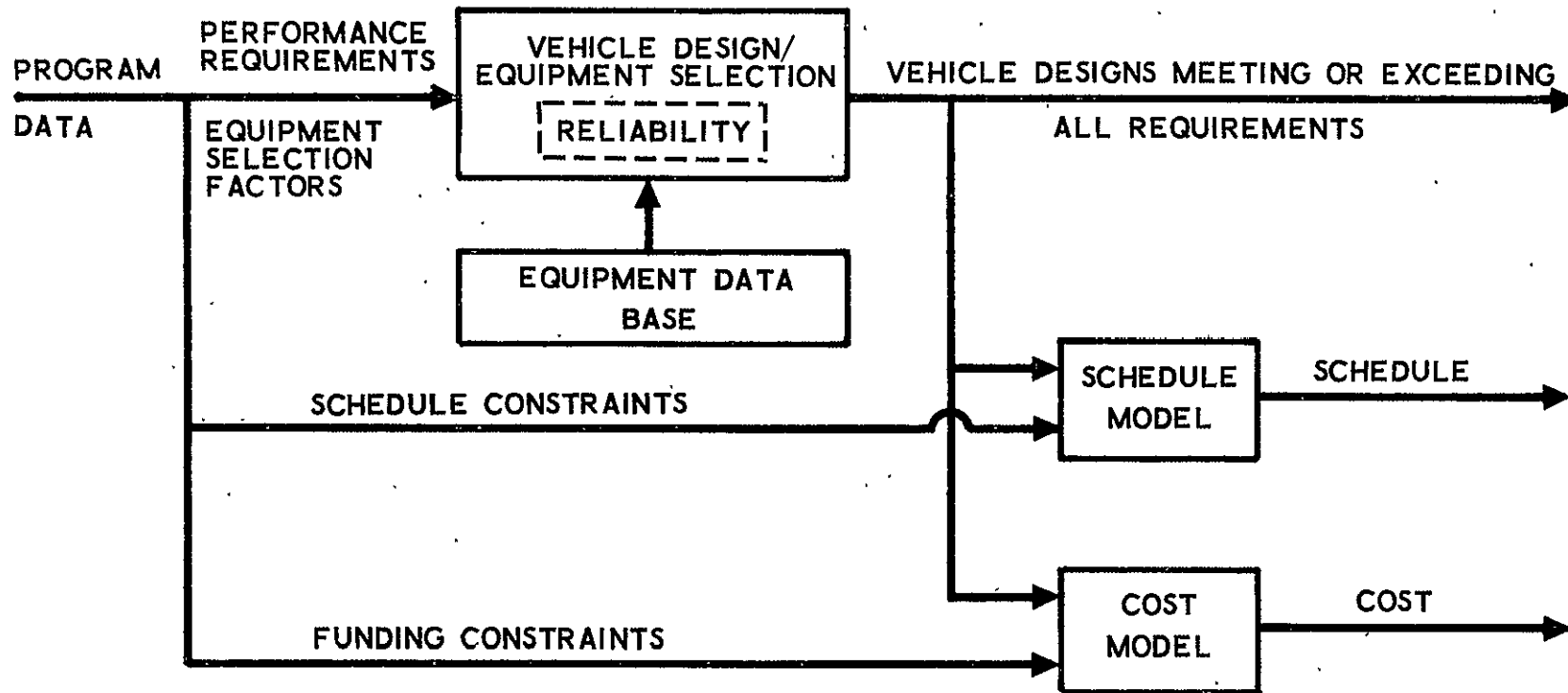


Figure 2-1. Systems Cost/Performance Model

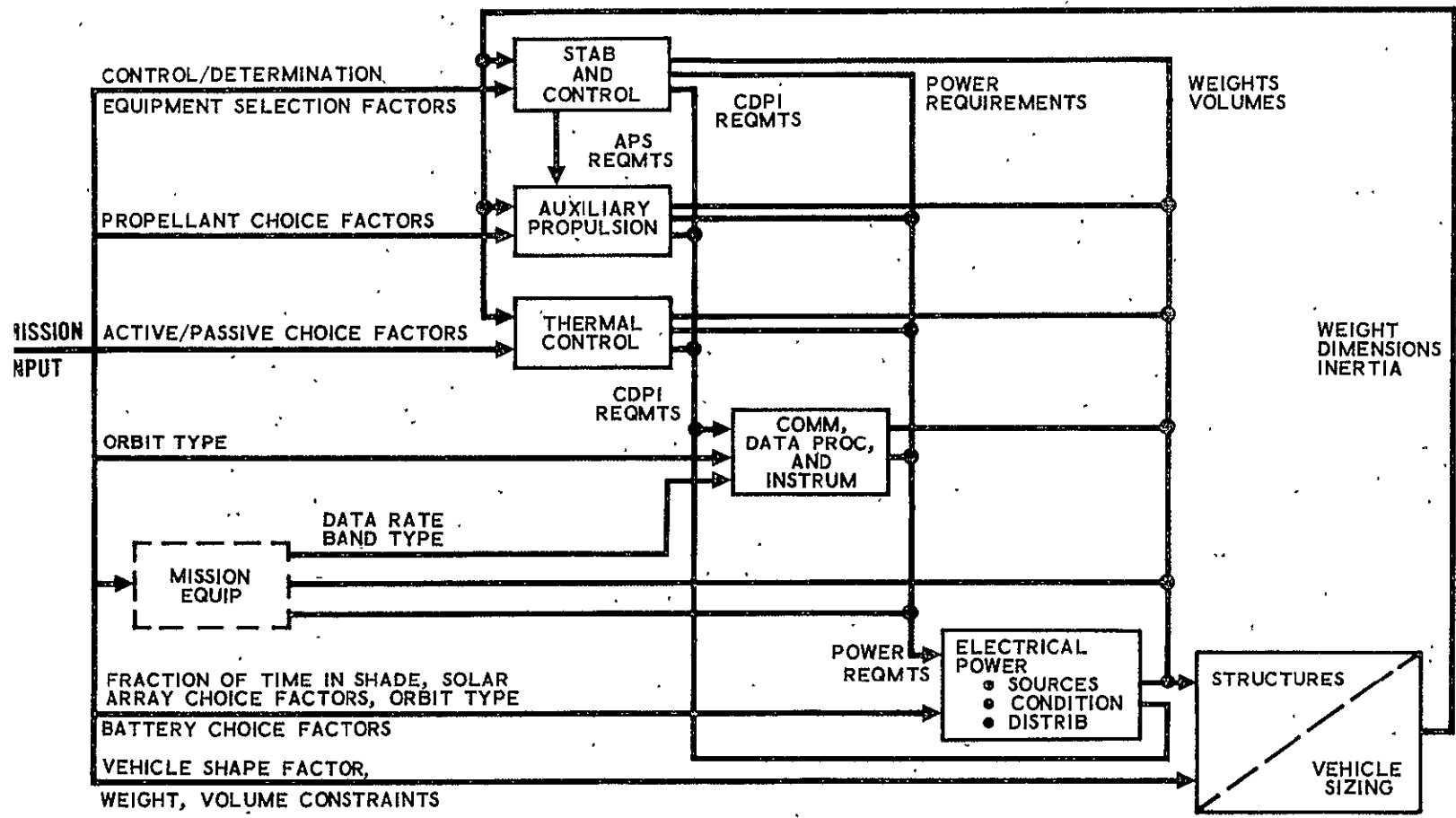


Figure 2-2. Vehicle Design/Equipment Selection

Subsystem, the vehicle weight, dimensions, and moments of inertia must be known. Design of the Auxiliary Propulsion Subsystem (APS) requires knowledge of the total impulse and thrust levels from S&C. Design of the Data Processing (DP) Subsystem requires knowledge of the telemetry and data processing requirements for each piece of equipment in the vehicle. Design of the Communication Subsystem requires knowledge of the command and communication requirements for the entire vehicle. Design of the Electrical Power (EP) Subsystem requires knowing the power requirements. Determining the structural makeup of the vehicle and the weight, dimensions, and inertias requires some insight into what is contained within the vehicle and what the environment is. The reliability requirements impact the design of every subsystem through the addition of redundancy. The major point to be made here is that by modeling the interaction of the subsystem design processes, the Systems-Cost/Performance Model is not only a subsystem design tool, but is also a system design tool.

2.1.3 Model Operation

The Systems Cost/Performance Model approach to designing payloads is illustrated in Figure 2-3. The starting point consists of a data base. The data base contains a large array of equipment (assemblies) which are to be candidates in designing the payload. Each piece of equipment in the data base has its attributes specified. The attributes include technical characteristics, power requirements, weight, volume, vibration, temperature and pressure specifications, command and telemetry requirements, a reliability description, and cost and schedule allocations.

The general approach to establishing a specific payload design is to select any combination of equipment from the data base. Next, the payload performance, safety, cost, and schedule can be estimated by using a set of aggregated equations (design algorithm) to process the equipment attributes. If the payload attributes, as determined by the aggregate equations, meet the user supplied requirements, then the specific design is

2-5

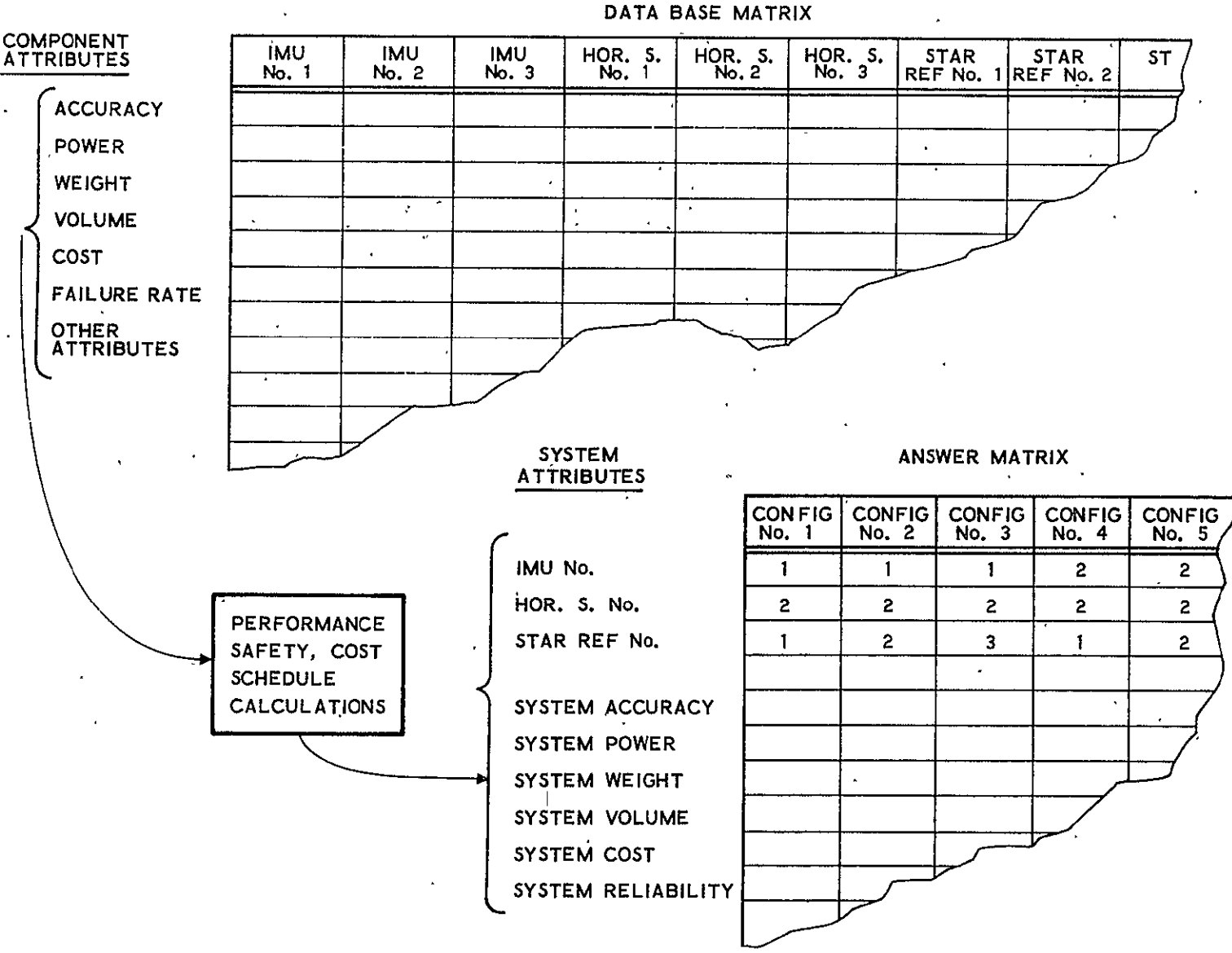


Figure 2-3. Model Operation

printed in an answer matrix as an acceptable design. If the payload attributes are not satisfactory, the design attempt is aborted. In either case, the Cost/Performance Model continues to try new combinations of equipment from the data base. The net result is a complete set of payload designs meeting or exceeding the input requirements.

2.2 SUBSYSTEM MODELS

2.2.1 Subsystem Configurations

A subsystem configuration is a general design type for which equipment listed in the data base will be searched out in order to mechanize the design. Configurations, then, are synonymous with subsystem types. The configurations incorporated in the Systems Cost/Performance Model are listed in Table 2-1.

Table 2-1. Configurations

Stabilization and Control

- a. Dual Spin
- b. Yaw Spin
- c. Three-Axis Mass Expulsion
- d. Mass Expulsion with Control Moment Gyros
- e. Mass Expulsion with Pitch Momentum Wheel

Auxiliary Propulsion

- a. Cold gas
- b. Monopropellant
- c. Bipropellant

Electrical Power Sources

- a. Body mounted solar arrays
- b. Oriented solar array paddles

Table 2-1. Configurations (Continued)

Electrical Power Conditioning

- a. Shunt regulation
- b. Shunt and discharge regulation
- c. Series load regulation

Communications

- a. Separate uplink and downlink
- b. Unified link, common antenna
- c. Unified link, separate antenna
- d. Unified link, common antenna, plus separate downlink
- e. Unified link, separate antenna, plus separate downlink

Data Processing

- a. General purpose processor
- b. Special purpose processors

Thermal Control

(Dependent on other subsystems and component requirements)

Vehicle Shapes

- a. Cylinder
- b. Box
- c. Sphere

Structure

- a. Semi-monocoque

Redundancy

- a. Single system
- b. Dual system

Each configuration has associated with it certain data including:

- a. Compatibility or incompatibility with other subsystems' configurations.
- b. Types of equipment required to mechanize the configuration.
- c. Duty cycle for each equipment type.
- d. Schedule data.

2.2.2 Equipment Description

The model selects equipment for a specific design in one of three ways:

- a. Most equipment is selected from the data base on the basis of technical performance.
- b. Some equipment which cannot be differentiated on the basis of technical performance is called up from the data base on a first called basis in order to provide a complete design description.*
- c. Certain equipment is not amenable to cataloging in the data base. This equipment is identified and specific parameters are determined. Examples include the wiring harness and the Thermal Control Subsystem components.

An example of an equipment description in the data base is provided in Table 2-2.

2.2.3 Design Algorithms

The design algorithms for all subsystems are summarized in the following paragraphs. The input data required by each subsystem is stated with the source of the data given in parentheses. A brief statement of what the subsystem design algorithm does is provided along with a summary of the output information.

2.2.3.1 Stabilization and Control Subsystem

- a. Input Data
 1. Vehicle orientation (User)
 2. Orbit altitude (User)
 3. Mission lifetime (User)
 4. Attitude control requirements (User)
 5. Vehicle physical description (Vehicle Sizing)

*It is proposed that this category be eliminated in future models by differentiation of all equipment as suggested in paragraph a.

Table 2-2. Data Base Example

Subsystem:	Auxiliary Propulsion (0808)	
Configurations:	Monopropellant	
Equipment Type:	Thruster (TRW 404620)	
Performance		
	Technical Characteristics	
(1)	Thrust level (N)	18
(2)	Pulse life (cycles)	93,000
(3)	Inlet pressure (N/m ²)	4.14 x 10 ⁶
(4)	Total impulse (N-sec)	6.49 x 10 ⁴
(5)	ISP (sec)	230
(6)		
(7)		
(8)		
(9)		
(10)		
	Power	
	Average Power (watts):	(near zero)
	Maximum Power (watts):	5.5
	Minimum Power (watts):	0.0
	Nominal Voltage (volts):	28.0
	Maximum Voltage (volts):	32.6
	Minimum Voltage (volts):	26.0
	Converter/Inverter Requirement (flag):	N. A.
	Weight (Kg):	0.3
	Volume (cc):	1700
	Vibration	
	Randon (g, rms):	19.5
	Non-Random (g):	10.5
	Temperature	
	Maximum (deg K):	322
	Minimum (deg K):	278
	Pressure (psia):	(Unknown)

Table 2-2. Data Base Example (Continued)

Performance (continued)	
CDPI	
Power Switching Commands (No.):	0
Time Tagged Commands (No.):	0
Other Commands (No.):	0
High Rate Telemetry	
Number of Analog Points (No.):	0
Number of Digital Points (No.):	0
Sample Rate (sec^{-1}):	0
Word Length (bits):	0
Low Rate Telemetry	
Number of Analog Points (No.):	2
Number of Digital Points (No.):	0
Sample Rate (sec^{-1}):	1
Word Length (bits):	8
Safety	
Failure Model (flag):	5
Failure Parameters	
Failure Rate or Mean ($\times 10^{\pm 9}$ hr):	1700
Standard Deviation ($\times 10^{\pm 9}$ hr):	N. A.
Dormancy Factor (N. D.):	0.1
Total Number of Redundant Elements (No.):	12
Cost	
Design Engineering (\$1000):	127
Test and Evaluation (\$1000):	150
Unit Production (\$1000):	9
Reference Quantity (No.):	4
Factor (N. D.):	1
Schedule	
Development Lead Time Constant (months):	3.0
Development Lead Time Variable (months):	1.0
Qualification Lead Time Constant (months):	1.5
Qualification Lead Time Variable (months):	0.1
State-of-Art Factor (N. D.):	1.0

- b. Model
 - 1. Selects attitude measurement equipment.
 - 2. Selects momentum exchange equipment.
 - 3. Computes attitude control thrust level.
 - 4. Computes total impulse required.
 - c. Output Data
 - 1. S&C equipment
 - 2. Attitude control thrust level
 - 3. Total impulse requirement
- 1.2.3.2 Auxiliary Propulsion Subsystem
- a. Input Data
 - 1. Powered flight thrust level (User)
 - 2. Attitude control thrust level (S&C)
 - 3. Total impulse requirement (S&C)
 - b. Model
 - 1. Selects thruster equipment.
 - 2. Selects propellant equipment.
 - 3. Selects pressurant equipment.
 - c. Output Data
 - 1. APS equipment
 - 2. Propellant description
- 1.2.3.3 Data Processing Subsystem
- a. Input Data
 - 1. Selected equipment (Subsystems)
 - 2. Equipment command requirements (Data Base)
 - 3. Equipment telemetry requirements (Data Base)
 - b. Model
 - 1. Selects computer or one digital telemetry unit per communication downlink.
 - 2. Selects command distribution equipment.

c. Output Data

1. DP equipment
2. Bit rate for each downlink

2.2.3.4 Communication Subsystem

a. Input Data

1. SGLS or USB compatibility requirement (User)
2. Range and range rate requirement (User)
3. Bit rates (DP)

b. Model

1. Selects communication equipment.

c. Output Data

1. Communication equipment

2.2.3.5 Electrical Power Subsystem

a. Input Data

1. Selected equipment (Subsystems)
2. Equipment power requirements (Data Base)
3. Voltage regulation requirements (Data Base)
4. Power conditioning requirements (Data Base)

b. Model

1. Sizes solar array.
2. Selects batteries and voltage regulation equipment.
3. Selects power conditioning equipment based on requirements of all other selected equipment.

c. Output Data

1. Solar array description
2. EP equipment

2.2.3.6 Thermal Control Subsystem

a. Input Data

1. Orbit description (User)
2. Attitude control description (S&C)
3. Selected equipment (Subsystems)
4. Equipment power requirements (Data Base)
5. Equipment temperature control requirements (Data Base)

b. Model

1. Sizes thermal mass, insulation, heaters, radiators, louvers, and heat pipes.

c. Output Data

1. Thermal control description

2.2.3.7 Vehicle Sizing

a. Input Data

1. Selected equipment (Subsystems)
2. Equipment weights and volumes (Data Base)
3. Structural description (Structures)
4. Mission equipment description (User)
5. Maximum diameter, length and weight (User)

b. Model

1. Estimates structural weight.
2. Estimates thermal control weight.
3. Estimates mechanism, booms, and electrical harness weight.
4. Estimates total vehicle weight.
5. Estimates adapter weight.
6. Estimates vehicle dimensions.
7. Estimates moments of inertia.

c. Output Data

1. Vehicle mass
2. Vehicle dimensions
3. Vehicle moments of inertia

2.2.3.8 Structural Subsystem

a. Input Data

1. Vehicle weight and dimensions (Vehicle Sizing)
2. Structural material description (User)
3. Launch loads environment (User)

b. Model

1. Sizes monocoque wall thickness.
2. Determines equivalent semi-monocoque wall thickness.
3. Determines actual wall thickness based on optimum weight design.
4. Determines stringer size and spacing.
5. Determines frame size and spacing.
6. Sizes end covers and center plate (if applicable).
7. Sizes mission bay and solar array extensions.

c. Output Data

1. Skin thickness
2. Stringer size, number, and locations
3. Frame size, number, and locations
4. End covers and center plate dimensions
5. Mission bay and solar array extension dimensions

2.3 RELIABILITY MODEL

As a result of satisfying the input performance requirements, a finite number of designs are established by the Cost/Performance Model. As the next step in processing these designs, the reliability aggregate equations are brought into play. These equations are categorized as reliability assessment, failure detection probability, and false alarm probability aggregate equations.

The first of these equations, the reliability assessment, is used to calculate the reliability of each configuration. This is done at an element level. Each identifiable subsystem component is considered as an element.

Thus, horizon sensors, inertial reference units, computers or control logic, thrusters, and propellant tanks would qualify as subsystem elements. Failure rate information stored in the equipment data base for each component is extracted as needed by the model. The failure rates are then combined by the reliability equations to calculate total reliability for a given mission duration. The calculated reliability of each particular design is evaluated against the specified level provided as the model input. However, the design is not discarded if it does not meet the specified reliability level. Instead, a search for the least reliable element is initiated. The criterion for least reliable is that element which, if made redundant, results in the largest increase in reliability or in mean mission duration per unit weight or cost increase. Upon identification, it is paralleled by an identical unit, and suitable aggregate equations are used to recalculate the system reliability. The evaluation and paralleling process continues until the redundancy exceeds a specified limit. If the system still does not meet the specified reliability, the system is deleted from consideration as a viable single-string system. However, should it at any time meet or surpass the required reliability level, aggregate equations are used to calculate system failure detection and false alarm probabilities. The process described above continues until each design stored as a result of meeting performance requirements has been processed.

The required input data includes:

- a. Mission life (User)
- b. System reliability (User)
- c. Basis for selecting redundancy (User)
- d. Selected equipment (Subsystems)
- e. Equipment reliability description (Data Base)
- f. Equipment weight or cost (Data Base)

The reliability aggregate equation procedure described above constitutes one-half of the total Reliability Model. Following completion of the basic scheme, the whole procedure is repeated with each design

mechanized as an active/standby (dual string) system. The term active/standby refers here to a completely separate system in addition to modular levels of redundancy.

The output information supplied by the Reliability Model includes the redundancy required for each component and the amount of expendables (propellant) required.

2.4 COST MODEL

The Cost Model consists of cost aggregate equations which process cost information associated with each subsystem component. This costing technique requires each component to have cost information for each of the five cost categories illustrated in Table 2-2.

The required input data includes:

- a. Selected equipment (Subsystems)
- b. Equipment costs (Data Base)
- c. Number of qualification vehicles (User)
- d. Number of production vehicles (User)

The Cost Model adds up the following cost information for every piece of equipment (up to 39 types) selected from the data base:

- a. Design engineering
- b. Test and evaluation
- c. Production engineering
- d. Unit production

Cost Estimating Relationships (CERs) are used to estimate the costs for components which are not amenable to cataloging, including:

- a. Structure
- b. Thermal control
- c. Wiring
- d. Power conditioning equipment
- e. Solar arrays
- f. Propellant tanks

The nonrecurring cost for each component takes into account redundancy, deviations or improvements in design, and inflation. The average recurring cost for each equipment component is adjusted to account for redundancy, labor, materials, deviations or improvements in design, and inflation. If more than one unit is to be built, a learning curve is used to account for reduced unit cost as additional quantities are built.

Remaining cost categories including:

- a. Tooling and test equipment
- b. Quality control
- c. System engineering and integration
- d. Program management

are estimated on the basis of predetermined percentages of the total of each of the four basic cost categories.

The total nonrecurring cost is then the summation of the non-recurring costs for all the system components. The total recurring cost is the summation of the products of the equipment quantities and the appropriate average recurring costs. The total spacecraft cost is obtained by summing the total recurring and nonrecurring costs and then adding in the mission equipment cost and contractor's profit.

2.5 SCHEDULE MODEL

Schedule aggregate equations estimate the amount of time required to develop an operational system. The aggregate equations estimate the following five schedule phases:

- a. Component design and development lead time
- b. Component qualification lead time
- c. Subsystem development lead time
- d. Subsystem qualification lead time
- e. System test, checkout, and flight readiness

In general, the estimates of the schedule lead times are functions of the hardware and software selected by the Cost/Performance Model. The

justification for such an approach lies in the fact that specific equipment provide an indication of the complexity of the system and, hence, a measure of the time required to complete the activities associated with the system.

The input data required by the Schedule Model includes:

- a. Selected equipment (Subsystems)
- b. Equipment lead times (Data Base)

The model performs the following operations using the appropriate aggregate equations:

- a. Computes the development and qualification lead times for each component.
- b. Computes the development and qualification lead times for each subsystem.
- c. Computes the system lead time.
- d. Determines the critical path.
- e. Computes the total program duration.

The Schedule Model output includes the various lead times, the total program duration, and the critical path.

2.6 COMPUTER PROGRAM

The Systems Cost/Performance Model has been implemented as a digital computer program. The program is written in the language of Fortran IV, as adapted to The Aerospace Corporation CDC 7600 computer and MSFC's Univac 1108 computer. The program includes the Cost/Performance Model and the related data base.

2.6.1 Program Techniques

The Systems Cost/Performance Computer Program incorporates four techniques to make the program as efficient as possible while retaining maximum versatility. The first technique is to pre-sort the equipment data base according to attributes specified by the program user. This

technique is desirable in order to allow the program to select equipment from the data base on the basis of the first piece identified which satisfies the requirements.

The second technique consists of having the program always do a "macro" search of combinations of major subsystem configurations. As an example, one combination of major subsystem configurations would be a three-axis stabilized payload using cold gas propellant, oriented solar array paddles, shunt power regulation, and so forth. The subsystem configurations have been specified in Paragraph 2.2.1.

The third technique is to mechanize the digital program to have the capability to try all combinations (micro-search) of equipment in any single subsystem if requested by the user. The user must specify the configuration types of the other subsystems in exercising this option. The program will select, design, and print out all acceptable combinations of equipment for the specified subsystem. This technique or option allows the subsystem specialist to perform detailed trade studies.

Because of the large number of design combinations that the program may identify which satisfy the input requirements, a post-sort routine (the fourth technique) is included which sorts the acceptable designs according to attributes as specified by the user. This technique performs the role of providing the computer program user with the designs listed in an organized fashion. Hence, the process of finding the "best" design out of all of the possible contenders is performed by the program.

2.6.2 Program Operation

The general sequence followed by the computer program is to read the input requirements, make one pass through the subsystem design algorithms, determine the required redundancy, and then make a second pass through the subsystem design algorithms with the data obtained from the first pass. Redundancy is not altered on the second pass primarily because the Reliability Model is extremely time-consuming. Cost and schedule are estimated for each acceptable design.

The computer program sequence is as follows:

- a. Read the input requirements supplied by the user,
 1. Subsystem requirements
 2. Safety, cost and schedule requirements
 3. Mission equipment description
 4. Pre-sort attributes
 5. Micro-search option
 6. Macro-search preference
 7. Post-sort attributes
- b. Pre-sort the data base according to the attributes specified by the user.
- c. Set up a new design attempt. This will be a new combination of configurations if the program is in the macro mode. If micro mode, this will be a new combination of equipment from the data base for the specified subsystem.
- d. Test to ensure that the subsystem configurations are compatible.
- e. Establish an initial estimate of the vehicle size based on the mission equipment description.
- f. Design the Stabilization and Control Subsystem.
- g. Design the Auxiliary Propulsion Subsystem.
- h. Design the Data Processing Subsystem.
- i. Design the Communications Subsystem.
- j. Design the Electrical Power Subsystem.
- k. If this is the first pass through the logic for the particular design, add the necessary redundancy to the components to meet the reliability requirements. If this is the second pass, the reliability model is not used.
- l. Design the Thermal Control Subsystem.
- m. Perform vehicle sizing.
- n. Design the Structural Subsystem.
- o. If this is the first pass through the logic for the particular design, make a second pass through the logic using the information collected during the first pass.
- p. Estimate the cost of the particular payload design.

- q. Estimate the schedule for the particular payload design.
- r. If the design is acceptable, print the output information describing the design.
- s. Return for a new design attempt, as necessary.
- t. When all design attempts have been completed, post-sort the acceptable designs according to the attribute supplied by the user.
- u. Print the design descriptions according to the post-sort ordering.

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3. STABILIZATION AND CONTROL SUBSYSTEM

3.1 GENERAL

3.1.1 Subsystem Functional Description

The Stabilization and Control (S&C) Subsystem stabilizes the spacecraft to a desired accuracy about a tracking line from a reference on the vehicle to an external reference. The external reference may be the local vertical of a planet, the sun, or a more distant star; an inertial reference; or the line of sight to a natural phenomenon like a gravity gradient or the lines of the earth's magnetic field. In many cases, a platform free to rotate with respect to the main structure of the vehicle must also be aligned with an external reference. On the Orbiting Solar Observatory (OSO) for example, the basic "wheel" of the satellite is lined up with the earth's local vertical, while the solar paddles are pointed at the sun.

The necessary accuracy of attitude stabilization depends, of course, on the mission of the vehicle. For most purposes, ± 0.017 radians (1 deg) is good enough; for special experiments, 48-97 mrad (10-20 arc sec) or better may be needed. Generally speaking, system performance is the result of design tradeoffs involving accuracy, average available power, the vehicle's moments of inertia, and the maximum disturbing torques.

3.1.2 Subsystem Configurations

Five S&C subsystem configuration models have been developed for the Systems Cost/Performance Model. The five configurations which are described briefly in the following sections, include:

- a. Dual Spin
- b. Yaw Spin
- c. Three-Axis Mass Expulsion
- d. Mass Expulsion with Control Moment Gyros
- e. Mass Expulsion with Pitch Momentum Wheel

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3.1.2.1 Dual Spin Configuration

The Dual Spin spacecraft, shown in Figure 3-1, consists of a spinning portion of the vehicle and a portion which is despun relative to the spinning portion. The spinning portion of the Dual Spin vehicle provides stiffness in roll and yaw (the vehicle spins about the pitch axis). The error in these two axes is very small and only needs correcting on the order of once a month. For this purpose, an open loop axial thruster system is employed which is actuated on command from the ground. An on-board system could be an alternative.

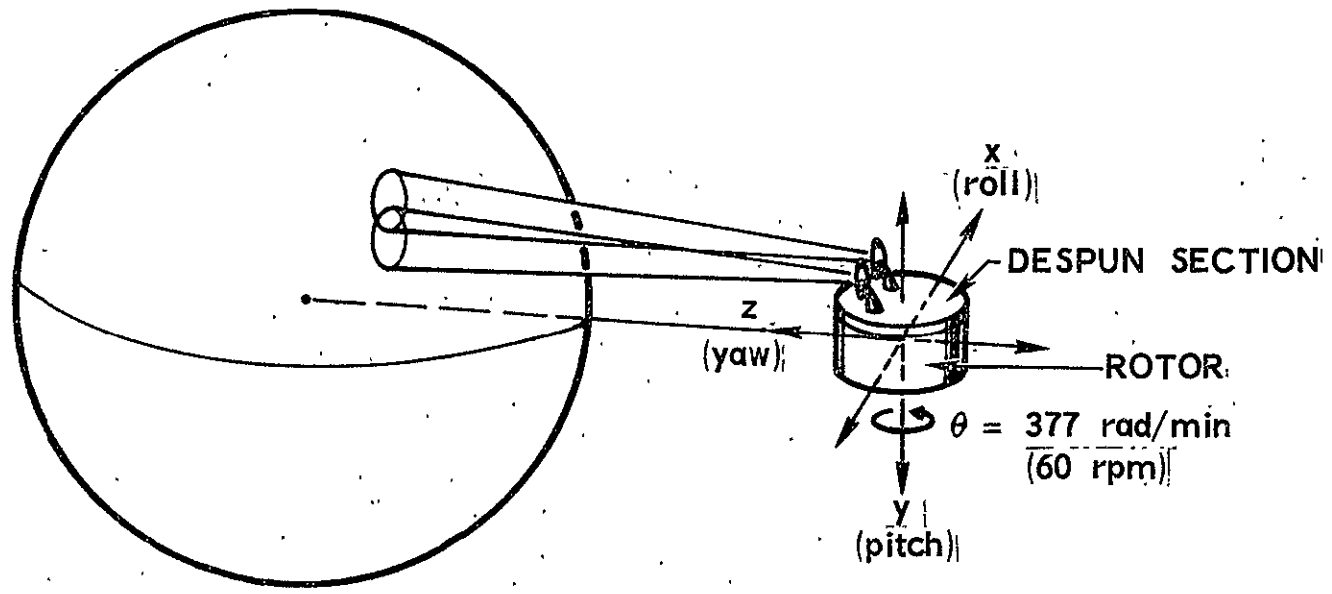
The despun control system provides the function of control about the pitch axis by keeping the despun section pointing at the center of the earth in the pitch direction. The despun section normally provides the stable platform required by certain mission equipment.

3.1.2.2 Yaw Spin Configuration

The Yaw Spin satellite, shown in Figure 3-2, obtains its name from the fact that it normally rotates about its yaw axis. The spinning of the satellite about its yaw axis, which is pointed at the earth's center, performs the function of providing a scan pattern for the payload. The payload is earth pointing and is mounted at a small angle to the spin axis. The vehicle is earth synchronous and is in an equatorial orbit.

Since (a) the spin axis must be kept pointing at the center of the earth and (b) the vehicle requirements call for long life, it is not feasible to rotate the momentum vector at orbital rate. Therefore, a counter-rotating reaction wheel keeps the spinning spacecraft's net momentum near zero.

The S&C subsystem performs the function of control about the pitch and roll axes to keep the spin (yaw) axis pointed at the center of the earth. The spinning vehicle, along with synchronous sampling of a horizon scanner which scans in one plane, provides the two-axis control utilizing a single sensor and thruster.



2
2

Figure 3-1. Dual Spin Configuration

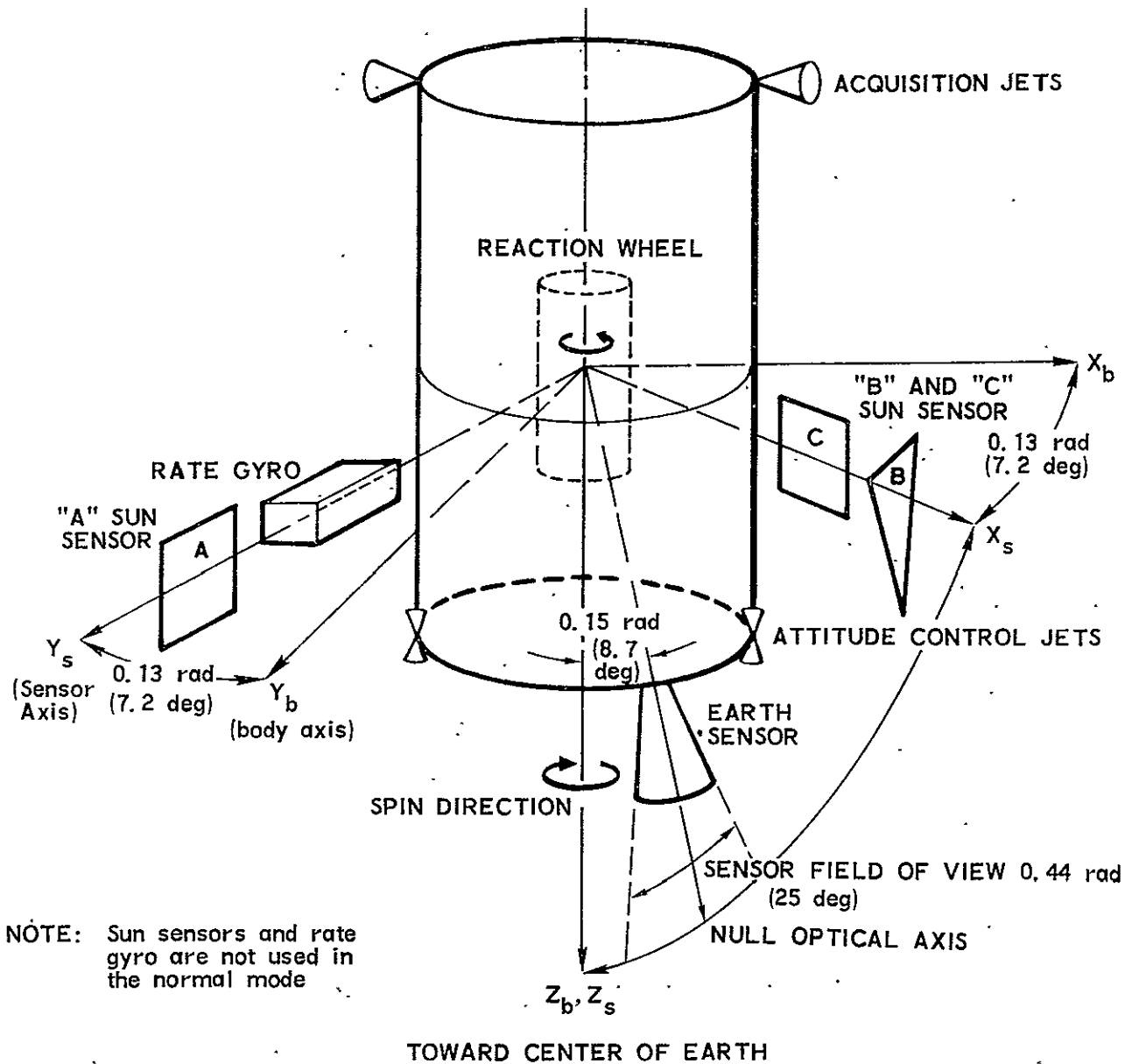


Figure 3-2. Yaw Spin Configuration

3.1.2.3 Three-Axis Mass Expulsion Configuration

The Three-Axis Mass Expulsion configuration, shown in Figure 3-3, performs the function of keeping the vehicle pointing at the local vertical in the orbit plane. The vehicle attitude is sensed by a three-axis body-mounted inertial reference unit containing three rate integrating gyros that are referenced to local vertical/orbit plane (LV/OP) coordinates by two horizon scanners and gyrocompassing. The vehicle is maintained in a fixed attitude with respect to the LV/OP with a pitch program, where the orbital pitchover rate is achieved by programming the appropriate signal into the pitch gyro. The horizon scanners bound the effect of gyro drift, thereby keeping the vertical axis of the vehicle aligned with the center of the earth. Control of the vehicle attitude is maintained by the appropriate signals to the Auxiliary Propulsion Subsystem to fire the attitude control jets.

3.1.2.4 Mass Expulsion Configuration with Control Moment Gyros

The Mass Expulsion configuration with Control Moment Gyros is derived from the Mass Expulsion configuration by incorporating Control Moment Gyros (CMGs). The CMGs provide the following improvements in vehicle control:

- a. More accurate rate and attitude control.
- b. Control torque and momentum storage to counteract disturbance torques.
- c. Control torque for vehicle angular acceleration and momentum storage for vehicle rotations.

3.1.2.5 Mass Expulsion Configuration with Pitch Momentum Wheel

The Mass Expulsion configuration with Pitch Momentum Wheel, shown in Figure 3-4, is an active, three-axis S&C control system incorporating a momentum wheel, attitude sensor, and mass expulsion jets. The unique feature of the one-wheel system is the use of an offset roll-actuated

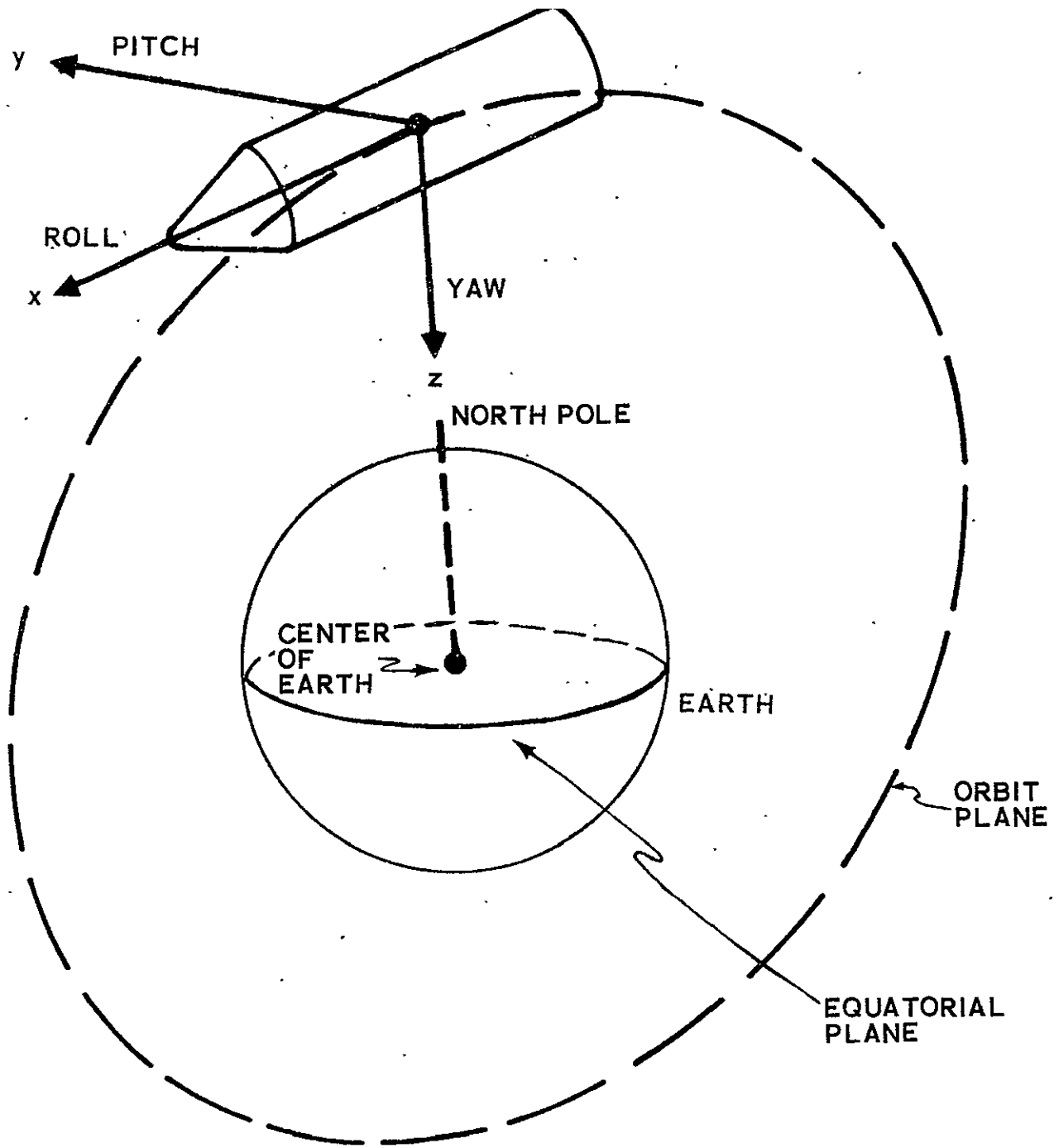


Figure 3-3. Three-Axis Mass Expulsion Configuration

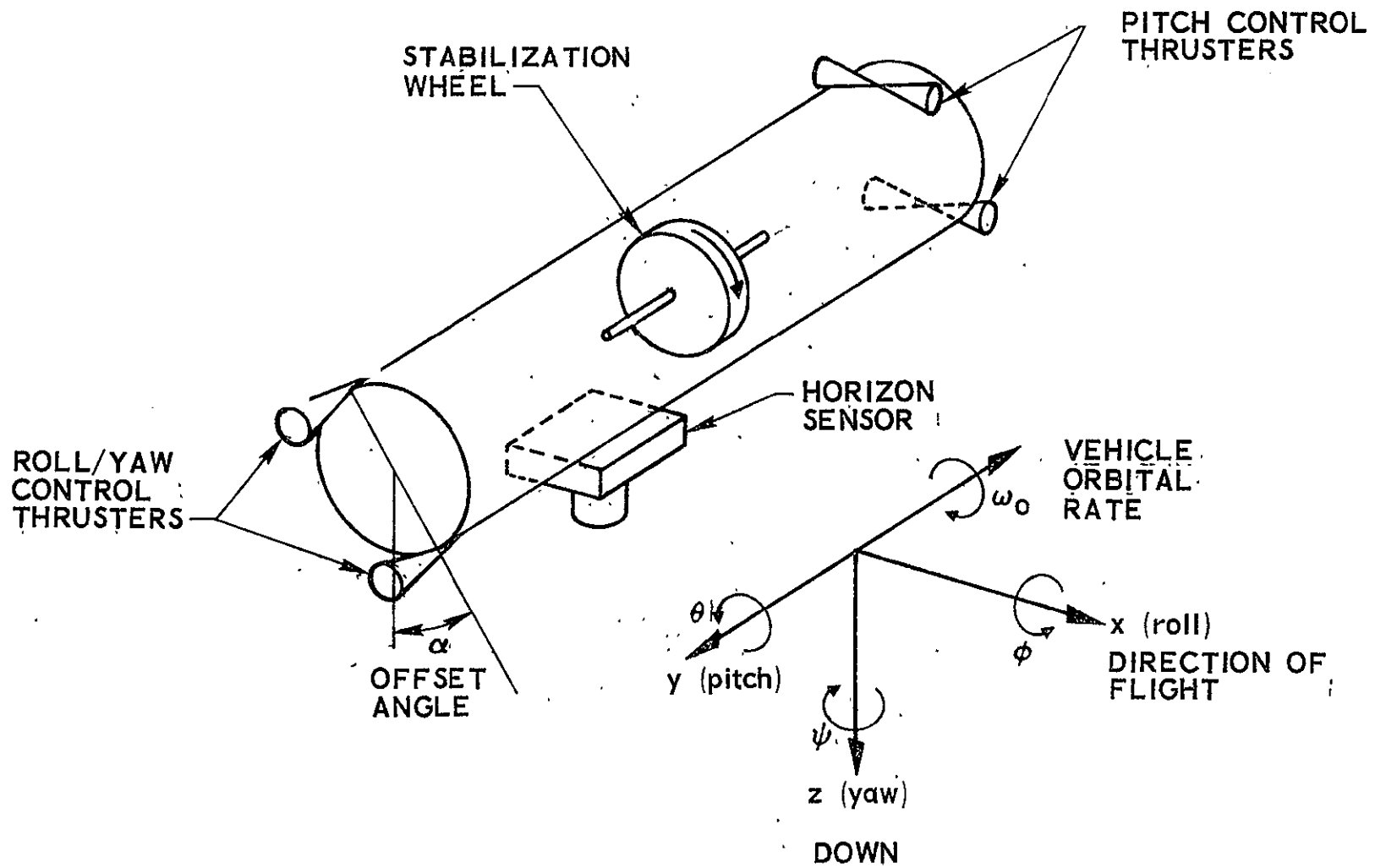


Figure 3-4. Mass Expulsion Configuration with Pitch Momentum Wheel

control torque and the momentum wheel to control the yaw axis without a direct yaw sensor. This system performs the following functions:

- a. Keeps the antenna beam within a required accuracy of the desired pointing position.
- b. Keeps the vehicle yaw angle relative to the local vertical within a required accuracy.

3.1.3 Configuration Compatibility

It is clear from both analysis and actual flight experience that limitations exist for the utilization and performance of each of the S&C configurations. Table 3-1 summarizes the limitations. Table 3-2 summarizes the requirement by the Mass Expulsion configuration with Control Moment Gyros for a General Purpose Processor as part of the Data Processing Subsystem.

3.1.4 Equipment Types

The complete list of equipment types from which the subsystem components will be selected is provided in Table 3-3 along with the technical characteristics which are used to select an acceptable subsystem design.

3.2 INPUT DATA

The information required to design the Stabilization and Control Subsystem is identified in Tables 3-4 and 3-5. The tables indicate what data are used by the design algorithm and whether the data are required or optional. If optional, the tables indicate what default value is used in the event the data are not supplied. Table 3-4 identifies the data required from the user. Table 3-5 identifies data to be supplied by the Vehicle Sizing model. An input not shown in the tables is the degree of redundancy required by the S&C subsystem components which must be specified by the Reliability model.

Table 3-1. Stabilization and Control Configuration Selection

Requirements	Dual Spin	Yaw Spin	Three-Axis Mass Expulsion	ME with CMGs	ME and Momentum Wheel
Payload yaw scan requirement	No	Yes	No	No	No
Orientation					
Inertial	Yes	No	Yes	Yes	Yes
Earth pointing	Yes	Yes	Yes	Yes	Yes
Sun pointing	Yes	No	Yes	Yes	Yes
Maneuverability requirements					
Powered flight control	Yes	Yes	Yes	Yes	Yes
Stationkeeping	Yes	Yes	Yes	Yes	Yes
Orbit correction control	Yes	Yes	Yes	Yes	Yes
Vehicle slewing	No	Yes	Yes	Yes	No
Altitude					
185-566 km (100-300 mi)	Yes	Yes	Yes	Yes	Yes
556-46,300 km (300-25,000 mi)	Yes	Yes	Yes	Yes	Yes
>46,300 km (25,000 mi)	Yes	Yes	Yes	Yes	Yes
Pointing accuracy					
35-170 mrad (2-10 deg)	Yes	Yes	Yes	Yes	Yes
3.5-35 mrad (0.2-2 deg)	Yes	Yes	Yes	Yes	Yes
0.17-3.5 mrad (0.01-0.2 deg)	Yes	No	No	Yes	No
<0.17 mrad (<0.01 deg)	No	No	No	Yes	No
Rate accuracy					
1.7-17 mrad/sec (0.1-1.0 deg/sec)	Yes	Yes	Yes	Yes	Yes
0.17-1.7 mrad/sec (0.01-0.1 deg/sec)	Yes	Yes	Yes	Yes	Yes
<0.17 mrad/sec (0.01 deg/sec)	No	No	No	Yes	No

Legend:

- Yes - Configuration can be used
- No - Configuration cannot be used

Table 3-2. Stabilization and Control Configuration Compatibility

Stabilization and Control Subsystem Configurations	Data Processing Subsystem	
	General Purpose Processors	Special Purpose Processors
Dual Spin	Yes	Yes
Yaw Spin	Yes	Yes
Three-Axis Mass Expulsion	Yes	Yes
Mass Expulsion with Control Moment Gyros	Yes	No
Mass Expulsion with Pitch Momentum Wheel	Yes	Yes

Legend:

- Yes - Compatible
- No - Incompatible

Table 3-3. Equipment Types and Their Technical Characteristics

Equipment Type	Configuration(s)	Technical Characteristic	Notation
Sensors:			
Sun Sensor Assembly	All		
Nonscanning Earth Sensor Assembly (with electronics)	Dual Spin	Sensor noise (deg, 3σ)	e_1
		Radiance irregularity (deg, 3σ)	e_2
		Quantization error (deg, 3σ)	e_3
		Sun interference (deg, 3σ)	e_4
		Moon interference (deg, 3σ)	e_5
		Threshold aging (deg, 3σ)	e_6
Horizon Sensor (with electronics)	All except Dual Spin	Sensor noise (deg, 3σ)	e_1
		Radiance irregularity (deg, 3σ)	e_2
		Quantization error (deg, 3σ)	e_3
		Sun interference (deg, 3σ)	e_4
		Moon interference (deg, 3σ)	e_5
		Threshold aging (deg, 3σ)	e_6
		Null or bias error (deg, 3σ)	e_7
		Maximum output frequency (rad/sec)	ω_H
Star Sensor Assembly (with electronics)	CMG	Type 1 for star mapper	s_1
		Type 2 for body fixed (electronic) star tracker	
		Type 3 for gimballed star tracker	
		Sensor accuracy (deg, 3σ)	s_2
		Mapper field of view (deg ²)	s_3
Mapper sensitivity (visual magnitude)	s_4		
Rate Integrating Gyro Assembly (with electronics)	All Mass Expulsion Configurations	G-insensitive gyro drift (deg/sec, 3σ) (24-hr stability)	g_1
		Total misalignment relative to vehicle (deg, 3σ)	g_2
		Gyro scale factor error (N.D.) (24-hr stability)	K_{sf}
Rate Gyro Assembly	Yaw Spin		

Table 3-3. Equipment Types and Their Technical Characteristics (Continued)

Equipment Type	Configuration(s)	Technical Characteristic	Notation
<u>Electronics:</u>			
Control Timing Assembly	Dual Spin	Programmer sine wave (deg, 3σ)	c_1
		Drive quantization and delay (deg, 3σ)	c_2
		Measurement compensation (deg, 3σ)	c_3
		Pipper drift (deg, 3σ)	c_4
		Quantization noise (deg, 3σ)	c_5
Center Electrical Assembly	Yaw Spin	Controller error (deg, 3σ)	c_6
Attitude Reference Electronics	Max Expulsion	Pitch horizon scanner gain (sec^{-1})	H_θ
		Roll horizon scanner gain to roll axis (sec^{-1})	H_ϕ
		Roll horizon scanner gain to yaw axis (sec^{-1})	H_ψ
		Pitch feedback gain (sec^{-1})	$H_{F\theta}$
		Roll feedback gain (sec^{-1})	$H_{F\phi}$
		Roll-to-yaw coupling gain (sec^{-1})	$H_{F\psi}$
Power Converter	All	Special requirement code (C__)	
Despin Electronics Assembly	Dual Spin		
Gimbal Electronics Assembly	Dual Spin	Resolver accuracy (deg, 3σ)	h_1
<u>Actuators:</u>			
Valve Driver Assembly	All	Number of valves	
Reaction Wheel Assembly (with electronics)	Yaw Spin and ME with Momentum Wheel	Nominal momentum (ft-lb-sec)	$H_W(\text{nom})$
		Maximum momentum (ft-lb-sec)	$H_W(\text{max})$
		Minimum momentum (ft-lb-sec)	$H_W(\text{min})$
		Nominal speed (rpm)	$\omega_W(\text{nom})$
		Maximum speed (rpm)	$\omega_W(\text{max})$
		Minimum speed (rpm)	$\omega_W(\text{min})$

Table 3-3. Equipment Types and Their Technical Characteristics (Continu

Equipment Type	Configuration(s)	Technical Characteristic	Notation
<u>Actuators (continued):</u>			
Single Gimbale Control Moment Gyro	Mass Expulsion with CMGs	CMG momentum (ft-lb-sec) Peak gimbal rate (rad/sec) Peak torquer torque (ft-lb)	$\frac{h}{\sigma}$ max $\frac{T}{\sigma}$ max
Despin Mechanical Assembly	Dual Spin	Bearing and motor friction (deg, 3σ) Bearing runout (deg, 3σ)	d_1 d_2
Biaxial (Gimbal) Drive Assembly (two required per antenna)	Dual Spin	Drive quantization (deg, 3σ) Gimbal drive error (deg, 3σ) Biax droop error (deg, 3σ)	b_1 b_2 b_3
Nutation Damper	Dual Spin		

Table 3-4. Input Data Supplied by User

Symbol	Name	Required(R) or Optional(O)	Default Value
	Configuration type	O	Not required
	Yaw spin requirement	O	
θ_{R_x}	Required system pointing accuracy about the roll, pitch and yaw axes (deg)	R	
θ_{R_y}		R	
θ_{R_z}		R	
$\dot{\theta}_{R_x}$	Required system rate accuracy about the roll, pitch and yaw axes (deg/sec)	R	
$\dot{\theta}_{R_y}$		R	
$\dot{\theta}_{R_z}$		R	
$\dot{\theta}_x(\text{max})$	Maximum maneuver rates about the roll, pitch and yaw axes, if applicable (deg/sec)	R	
$\dot{\theta}_y(\text{max})$		R	
$\dot{\theta}_z(\text{max})$		R	
N_x	Number of maneuvers about the roll, pitch, and yaw axes	R	
N_y		R	
N_z		R	
$\dot{\theta}_o$	Maximum initial rate (deg/sec) (assumed same on all axes)	R	
ω_o	Average orbital rate (computed elsewhere in the computer program) (rad/sec)	R	

Table 3-4. Input Data Supplied by User (Continued)

Symbol	Name	Required(R) or Optional(O)	Default Value
η	Control system efficiency	O	3 with 3-axis rate gyros, 12-15 with no rate sensing
M_{D_x}	External roll disturbance torque (ft-lb)	R	
M_{D_y}	External pitch disturbance torque (ft-lb)	R	
M_{D_z}	External yaw disturbance torque (ft-lb)	R	
F_e	Main engine thrust (lb)	R	
t	Main engine burn time (sec)	R	
$\Delta\theta$	Main engine alignment relative to vehicle thrust axis (deg)	O	0.25 deg
Δd	Lateral vehicle c.g. distance plus lateral thrust chamber c.g. distance from the reference axis (ft)	O	0.04 D
T	Mission lifetime (months)	R	
τ_x	Time that the roll, pitch, and yaw disturbance torques are in effect for a mission (sec)	R	
τ_y		R	
τ_z		R	
	<u>Specific Inputs for Dual Spin Configuration</u>		
K	$= \begin{cases} 0 & \text{if errors are desired for spin} \\ & \text{axis relative to nadir} \\ 1 & \text{if errors are desired for pay-} \\ & \text{load relative to nadir} \end{cases}$	R	

Table 3-4. Input Data Supplied by User (Continued)

Symbol	Name	Required(R) or Optional(O)	Default Value
	<u>Specific Inputs for Dual Spin Configuration (continued)</u>		
J_p	Platform spin axis inertia (slug-ft ²)	R	
ω_R	Spin rate of rotor (rpm)	R	
N	Number of days between corrections in the spin axis pointing of the vehicle	R	
	<u>Specific Inputs for Yaw Spin Configuration</u>		
ω_s	Spin rate about z-axis for yaw-spin vehicle (rad/sec)	R	
	<u>Specific Inputs for Three-Axis Mass Expulsion Configuration</u>		
e_p	Maximum programmed pitchover rate error = ω_o (max) - ω_o (function of orbit eccentricity) (deg/sec)	O	$0.1 \omega_o$
a_x, a_y, a_z	Misalignment errors in mounting inertial measurement units relative to vehicle x, y, z axes (deg)	O	0.05 deg (3 σ)
	<u>Specific Inputs for Mass Expulsion Configuration with Control Moment Gyros</u>		
n	Number of skewed single-gimbaled control moment gyros (4-6)	O	4

Table 3-4. Input Data Supplied by User (Continued)

Symbol	Name	Required(R) or Optional(O)	Default Value
	<u>Specific Inputs for Mass Expulsion Configuration with Control Moment Gyros (continued)</u>		
$\dot{\theta}_{avg}$	Average body rate for low orbit during period when high accuracy is required (deg/sec)	R	
$\dot{\theta}_{ST(max)}$	Maximum vehicle rate at which star information must be obtained (deg/sec)	R	
θ_{FOV}	Maximum range of attitude freedom (full FOV) required to track specific guide stars over wide range of vehicle motion (deg)	R	
t_{hold}	Time vehicle must be in inertial hold (min)	R	
t_{ℓ}	Time between unloading wheel momentum (days)	O	1 day
t_{accel}	Acceleration time for maneuvering (sec)	O	20 sec
	<u>Specific Inputs for Mass Expulsion Configuration with Pitch Momentum Wheel</u>		
e_a	Antenna misalignment (deg, 3σ)	R	
E	Antenna elevation angle (rad)	R	
α	Thruster offset angle in the roll-yaw plane (deg)	O	12 deg

Table 3-5. Input Data Supplied by Vehicle Sizing Model

Symbol	Name	Required(R) or Optional(O)	Default Value
D	Vehicle diameter (ft)	R	
d_T	Distance from c. g. to main engine (ft)	R	
d_x	Gas jet lever arm on roll axis (ft)	R	
d_y	Gas jet lever arm on pitch axis (ft)	R	
d_z	Gas jet lever arm on yaw axis (ft)	R	
J_x	Vehicle roll inertia (slug-ft ²)	R	
J_y	Vehicle pitch inertia (slug-ft ²)	R	
J_z	Vehicle yaw inertia (slug-ft ²)	R	
J_R	Rotor spin axis inertia (slug-ft ²) (for Dual Spin)	R	

3.3 DUAL SPIN CONFIGURATION

3.3.1 Functional Description

The Dual Spin configuration components perform the following functions:

- a. Attitude Sensors. The horizon sensors mounted in a "V" configuration on the spinning part detect when a point on the spinning section is pointing at the earth's center. This provides attitude errors in pitch and roll and also provides a measure of the spin rate in pitch.
- b. Control Electronics. There are four electronic assemblies involved: the Control Timing Assembly, the Despin Electronics Assembly, the Valve Timing Assembly, and the Gimbal Electronics Assembly. The Control Timing Assembly processes the difference between the horizon sensor output and a reference pip between the spinning and despun sections (once every revolution of the rotor) and converts this error to an analog signal. This error is sent to the Despin Electronics Assembly which commands the despin motor to correct for the error. The Control Timing Assembly also provides a signal to the Valve Timing Assembly to assist in synchronizing the firing of thrusters to correct for roll and yaw errors (once every 20 to 30 days). The Gimbal Electronics Assembly also receives a signal from the Control Timing Assembly which is used to command the gimbals which point the antennas on the despun section.
- c. Control Mechanisms. There are three mechanisms: the Despin Mechanical Assembly, the Gimbal Drive Assembly, and the Nutation Damper. The Despin Mechanical Assembly contains a despin motor which controls the spin speed of the rotor and the relative alignment of the despun platform with the earth. The Gimbal Drive Assembly contains gimbal drive motors to move the gimbals which point the antennas that are mounted on the despun section. The Nutation Damper is a passive controller which is mounted on the despun section to provide energy dissipation for stabilizing the angular momentum vector of the spinning vehicle. Deviations in the angular momentum vector occur due to spurious torques such as inadvertent mass expulsion, magnetic, gravity-gradient and solar radiation pressure. This damper helps to provide the stiffness in the roll and yaw axis.

A block diagram of the Dual Spin configuration is presented in Figure 3-5.

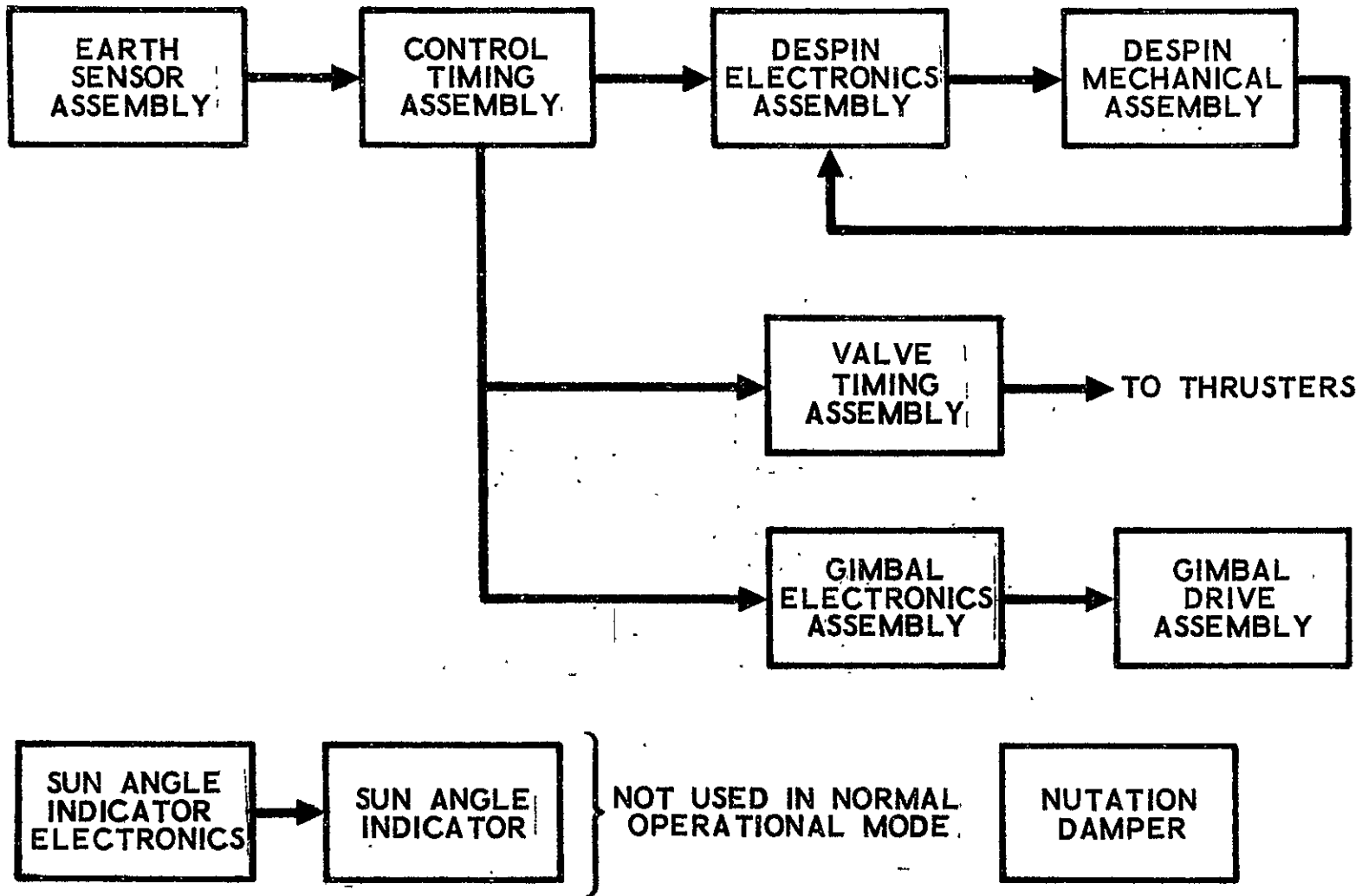


Figure 3-5. Dual Spin Configuration Block Diagram

3.3.2 Design Equations

The various contributors to the pointing error in each of the three vehicle axes are shown in Table 3-6 (see Ref. 3-1). The first two categories pertain to errors of the spin axis (i. e., platform base) relative to nadir. The next two categories pertain to the errors of the gimbals relative to the spin axis. The D term refers to errors which are deterministic in nature and have worst case values. These are not RSSed with the random (R, 3σ) errors but summed directly with them after the random errors are RSSed.

The earth sensor noise errors are scaled by 1/2.94 to account for processing by the control system.

$$E_1 = e_1/2.94 \quad (3-1)$$

$$E_3 = e_3/2.94 \quad (3-2)$$

The bearing design dependent numbers and the error due to nutation are scaled by $75/J_p$ where J_p is the platform spin axis inertia with a nominal value of 102 kg-m^2 (75 slug-ft^2).

$$D_1 = 75 d_1/J_p \quad (3-3)$$

$$N_u = 0.75/J_p \quad (3-4)$$

The spin axis errors relative to nadir are all scaled by $N/21$ where N is the number of days between corrections in the spin axis pointing of the vehicle with a nominal value of 21 days.

$$E_2 = N e_2/21 \quad (3-5)$$

$$E_6 = N e_6/21 \quad (3-6)$$

$$D_2 = N d_2/21 \quad (3-7)$$

The external torque errors are scaled by $320 \times 60/J_R \omega_R$ (in addition to $N/21$) where the rotor spin axis inertia (J_R) has a nominal value of 434 kg-m^2 (320 slug-ft^2) and the nominal value for the rotor spin rate is 377 rad/min (60 rpm.)

Table 3-6. Dual Spin Configuration Error Contributors

Error Category	Definition of Error Contributors	Symbol or Error Value If Assumed	Deterministic (D) or Random (R)
Despin errors (pitch axis relative to nadir)	Fixed scan earth sensor assembly - Earth sensor noise - Quantization noise - Sun interference - Moon interference Control timing assembly - Pipper drift - Quantization noise Despin motor assembly - Bearing and motor friction Nutation	e_1 e_3 e_4 e_5 c_4 c_5 d_1 0.01°	R R D D R R R R
Spin axis errors (roll and yaw errors relative to nadir)	Fixed scan earth sensor assembly - Radiance - Threshold Despin motor assembly - Bearing runout External torques - Mean solar torque - Solar torque variation	e_2 e_6 d_2 2.02 mrad (0.116 deg) 0.52 mrad (0.030 deg)	R R R D

Table 3-6. Dual Spin Configuration Error Contributors (Continued)

Error Category	Definition of Error Contributors	Symbol or Error Value If Assumed	Deterministic (D) or Random (R)
Azimuth gimbal errors (platform pitch error relative to the spin axis)	Gimbal drive assembly - Drive quantization - Gimbal drive error - Biax droop error Control timing assembly - Programmer - sine wave Gimbal electronics assembly - Gimbal angle determination	g_1 g_2 g_3 c_1 h_1	R R R R R
Elevation gimbal errors (platform roll error relative to the spin axis)	Fixed scan earth sensor assembly - Sun interference Control timing assembly - Programmer-sine wave - Drive quantization - Measurement quantization Gimbal electronics assembly - Gimbal angle determination Gimbal drive assembly - Drive quantization - Gimbal drive error - Biax droop error	e_4 c_1 c_2 c_3 h_1 g_1 g_2 g_3	D R R R R R R R

$$M_1 = 0.116 (320 \times 60 \text{ N}) / (21 J_R \omega_R) \quad (3-8)$$

$$M_2 = 0.030 (360 \times 60 \text{ N}) / (21 J_R \omega_R) \quad (3-9)$$

The errors due to solar torques are not scaled by input solar torque values as nominal values were not available. Future work should include this effect.

A Dual Spin vehicle utilizes a linear control law. As a result, all noises are random and can be RSSed to yield the final 3σ pointing error.

The user has a choice of either specifying the spin axis errors (i.e., platform base) relative to nadir:

$$K_1 = (E_1^2 + E_3^2 + c_4^2 + c_5^2 + D_1^2 + N_u^2)^{1/2} \quad (3-10)$$

$$E_y = (e_4^2 + e_5^2)^{1/2} + K_1, \text{ (pitch error)} \quad (3-11)$$

$$K_2 = (E_2^2 + E_6^2 + D_2^2 + M_2^2)^{1/2} \quad (3-12)$$

$$E_x = E_z = M_1 + K_2, \text{ (roll and yaw error)} \quad (3-13)$$

or the payload errors relative to nadir (i.e., the gimbals relative to nadir:

$$G = (g_1^2 + g_2^2 + g_3^2)^{1/2} \quad (3-14)$$

$$E_y = (e_4^2 + e_5^2)^{1/2} + (K_1^2 + G^2 + c_1^2 + h_1^2)^{1/2}, \text{ (pitch error)} \quad (3-15)$$

$$E_x = (M_1^2 + e_4^2)^{1/2} + (K_2^2 + c_1^2 + c_2^2 + c_3^2 + h_1^2 + G^2)^{1/2} \quad (3-16)$$

Once the errors (E_x , E_y , and E_z) are determined, they are compared with the user-specified system accuracies (θ_{R_x} , θ_{R_y} , and θ_{R_z}) to determine whether the design is acceptable.

3.3.3 Design Logic

The flow logic or sequence that the design algorithm must follow is:

- a. Input data
- b. Call up configuration-special hardware from the data base which are not differentiable:

1. Sun Sensor Assembly
 2. Despin Electronics Assembly
 3. Valve Timing Assembly
 4. Nutation Damper
- c. Select Control Timing Assembly
 - d. Select Despin Mechanical Assembly
 - e. Select Non-scanning Earth Sensor
 - f. Compute the spin axis (platform) errors relative to nadir
 - g. Select Gimbal Electronics Assembly
 - h. Select Gimbal Drive Assembly
 - i. Compute the errors for gimbals relative to nadir.
 - j. Compare the appropriate pointing errors with the maximum allowable pointing error. If acceptable, use the selected components. If unacceptable, select more accurate components and repeat the sequence.

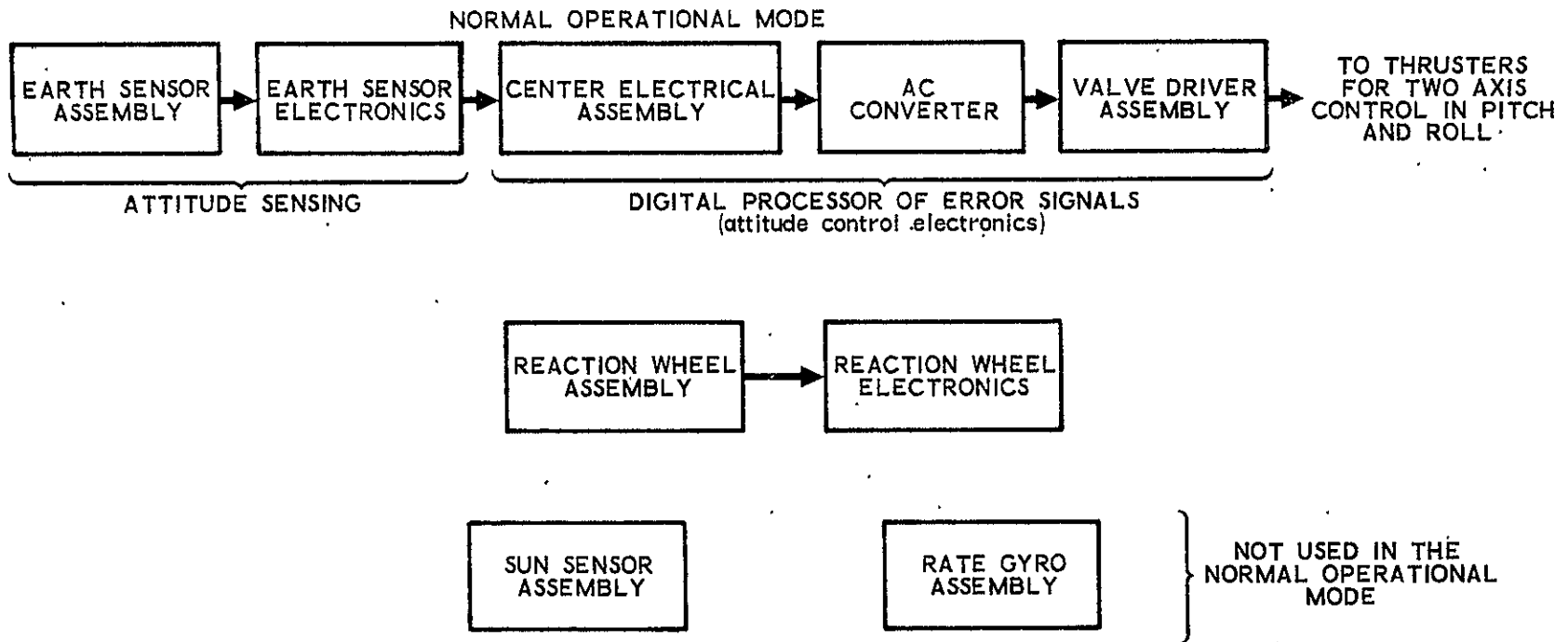
3.4 YAW SPIN CONFIGURATION

3.4.1 Functional Description

The Yaw Spin configuration components perform the following functions:

- a. Attitude Sensors. A single horizon scanner which scans in the vehicle x-z plane is mounted at an angle of 0.15 rad (8.7 deg) to the spin axis (scan distance = $\pm 0.223 \text{ rad}$ (12.8 deg) of the scanner null axis). This scanner scans with a period of 213 msec as compared with a vehicle spin period of 10 sec. This provides error signals in pitch and roll.
- b. Digital Processor of Error Signals. The Center Electrical Assembly processes the measured pitch and roll errors with a network consisting of a pulse-width modulator and a derived rate feedback (utilizing a digital high pass filter). The Valve Driver Assembly determines when the thrusters will fire to correct for errors.
- c. Controller. Consists of a pair of axial thrusters which are fired at appropriate times to correct for errors in either pitch or roll.
- d. Reaction Wheel Assembly. A counterrotating reaction wheel keeps the spinning spacecraft's net momentum near zero.

A block diagram of the Yaw Spin configuration is presented in Figure 3-6.



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Figure 3-6. Yaw Spin Configuration Block Diagram

3.4.2 Design Equations

3.4.2.1 Rate Error (Maximum Thrust Requirement) (Ref. 3-2)

$$\begin{aligned} \text{Rate Error} &= \frac{\text{Torque Impulse}}{\text{Transverse Inertia}} \\ \dot{\theta}_x &= \frac{2(\Delta t) \cdot F d_x}{J_x} \end{aligned} \quad (3-17)$$

where

$$\begin{aligned} \Delta t &= \text{minimum impulse bit (sec)} \\ d_x &= \text{moment arm (ft)} \\ F &= \text{thruster output (lb)} \\ J_x &= \text{transverse inertia (slug-ft}^2\text{)} \\ \dot{\theta}_x &= \text{rate error about the roll axis (deg/sec)} \end{aligned}$$

The factor of two in the torque impulse equation accounts for a maximum of four minimum impulse firings sequentially before the control logic stops the jets. It is conservative in that usually only two or three firings will occur at one time. If only one occurred, then the factor would be equal to one-half.

Given a required system rate error, θ_{R_x} , the maximum thrust level (for a particular minimum impulse bit) is obtained by rewriting Equation (3-17) as

$$F_{\max} = \frac{J_x \dot{\theta}_{R_x}}{2(\Delta t) d_x} \quad (3-18)$$

3.4.2.2 Pointing Error (Selection of Earth Sensor) (Ref. 3-2)

The pointing error of the spin axis in pitch and roll is usually specified, which creates a square error pattern as shown in Figure 3-7. The maximum spin axis pointing error lies at the corner of this square and is larger than either pointing error by itself. The equations following deal with the maximum 3σ pointing error.

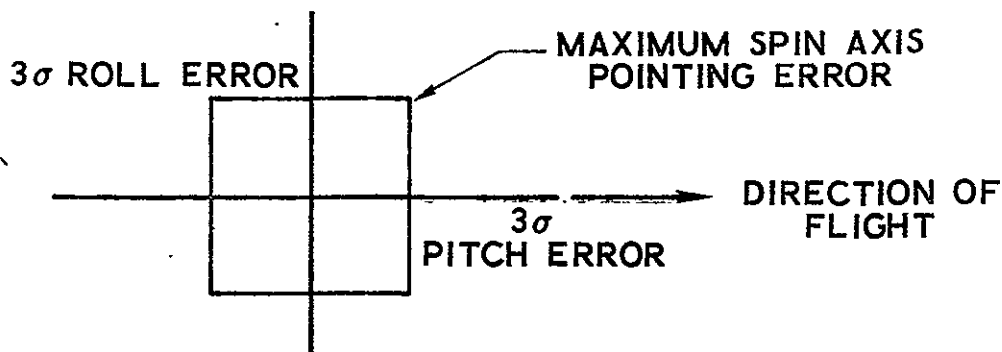


Figure 3-7. Yaw Spin Pointing Error Pattern

The spin axis pointing error is made up of two types of errors: the slow-varying or fixed sensor errors and the dynamic errors which have observable effects. These are detailed below.

a. Slow-varying or fixed sensor errors

1. Null offset*
2. Alignment error*
3. Thermal distortion*
4. Radiance irregularity (cold clouds, etc.) = e_2
5. Quantization = e_3

The errors with the * after them cancel out because of the method of averaging out the horizon sensor errors using four measurements with two taken on each side of the earth. The resulting equation for the slow-varying or fixed sensor errors (E_1) is the RSS of the noncanceling 3σ errors as shown below.

$$E_1 = \left[(e_2)^2 + (e_3)^2 \right]^{1/2} \quad (3-19)$$

b. Dynamic Observable Errors

1. Deadband error
2. Controller error
3. Disturbance torque error
4. Earth sensor noise error

The total spin axis pointing error due to dynamic observable errors is the sum of the above errors and is added to the RSS of the sensor errors to obtain the total spin axis pointing error. Each of the observable errors is discussed below.

1. Deadband Error. The error due to the control system deadband (E_2) is simply the value of the deadband.

$$E_2 = db \quad (3-20)$$

where

$$db = \text{deadband (deg)}$$

2. Controller Error. The controller error is a major component of the total spin axis error and varies with disturbance torque, measurement gain and control impulse. However, the peak controller error and percentage of time spent outside of the deadband are relatively insensitive to variations in these quantities (although the average value does vary). Since the 3σ value is equivalent to the peak error, the equation for 3σ spin axis error due to controller error (E_3) becomes

$$E_3 = c_6 = 1.789 \text{ mrad (0.1025 deg) } (3\sigma) \quad (3-21)$$

3. Disturbance Torque Error. The disturbance torques acting on the vehicle are due to solar winds, momentum bias, and control plane rotation. The error due to control plane rotation can be kept at zero. For zero control plane rotation, the error (E_4) is given as

$$E_4 = \begin{cases} \frac{0.3}{d_x F} (0.18 \times 10^4) \times M_{D_{\max}} + 0.12 - db, & \text{if } > 0 \\ 0 & \text{if the above quantity } \leq 0 \end{cases} \quad (3-22)$$

where

$$M_{D_{\max}} = \text{maximum disturbance torque} \\ = \text{the larger of } M_{D_x} \text{ or } M_{D_y}$$

4. Total Pointing Error. The total pointing error (θ_x) is the sum of all of the errors previously discussed as shown below.

$$\theta_x = E_1 + E_2 + E_3 + E_4 \quad (3-23)$$

Substituting in the equations for E_1 yields

$$\theta_x = (e_2^2 + e_3^2)^{1/2} + db + c_6 + E \quad (3-24)$$

where

$$E = E_4 = \begin{cases} \frac{0.54 \times 10^3 M_{D_{\max}}}{d_x F} + 0.12 - db \\ 0 \text{ if the above is } \leq 0 \end{cases} \quad (3-25)$$

3.4.2.3 Momentum Wheel Sizing

The momentum wheel is sized so that its momentum will be able to match that of the spinning vehicle and thus cancel it out. The momentum of the vehicle about the yaw axis is given by

$$H = J_z \omega_s \quad (3-26)$$

where

$$\omega_s = \text{vehicle spin speed (rad/sec)}$$

3.4.2.4 Minimum Thrust Requirement

The minimum thrust requirement is obtained by computing the thrust necessary to compensate for

- a. the translational thruster offset and misalignment during a powered flight phase
- b. the external disturbance torques acting on the vehicle.

These equations follow:

$$\left. \begin{aligned} F_1 &= \left(\frac{\Delta\theta}{57.3} d_T + \Delta d \right) \frac{F_e}{d_y} \\ F_2 &= \left(\frac{\Delta\theta}{57.3} d_T + \Delta d \right) \frac{F_e}{d_z} \\ F_3 &= \frac{2}{57.3} \Delta d \Delta\theta \frac{F_e}{d_x} \\ F_4 &= M_{D_x} / d_x \\ F_5 &= M_{D_y} / d_y \end{aligned} \right\} \quad (3-27)$$

The minimum thrust required is the maximum of F_1 through F_5 .

3.4.2.5 Total Impulse Requirement

The total impulse required is a function of the requirements for limit cycling, powered flight, recovery from initial rates, single-axis rotational maneuvers and overcoming external torques:

a. Limit Cycle Impulse

$$\left. \begin{aligned} I_{x_1} &= 37 \times 10^6 T d_x (F \Delta t)^2 / (J_x db_x) \\ I_{y_1} &= 37 \times 10^6 T d_y (F \Delta t)^2 / (J_y db_y) \\ I_{z_1} &= 0 \end{aligned} \right\} (3-28)$$

b. Powered Flight Impulse (Translational)

$$\left. \begin{aligned} I_{y_2} &= \left(\frac{\Delta \theta}{57.3} d_T + \Delta d \right) \frac{F_e t}{d_y} \\ I_{z_2} &= \left(\frac{\Delta \theta}{57.3} d_T + \Delta d \right) \frac{F_e t}{d_z} \\ I_{x_2} &= \frac{2}{57.3} \Delta d \Delta \theta \frac{F_e t}{d_x} \end{aligned} \right\} (3-29)$$

c. Impulse Due to Recovery From Initial Rates

$$\left. \begin{aligned} I_{x_3} &= \eta J_x \dot{\theta}_0 / (57.3 d_x) \\ I_{y_3} &= \eta J_y \dot{\theta}_0 / (57.3 d_y) \\ I_{z_3} &= \eta J_z \dot{\theta}_0 / (57.3 d_z) \end{aligned} \right\} (3-30)$$

d. Single Axis Rotation Impulse

$$\begin{aligned}
 I_{x_4} &= \frac{2 J_x \dot{\theta}_x (\max) N_x}{57.3 d_x} \\
 I_{y_4} &= \frac{2 J_y \dot{\theta}_y (\max) N_y}{57.3 d_y} \\
 I_{z_4} &= \frac{2 J_z \dot{\theta}_z (\max) N_z}{57.3 d_z}
 \end{aligned}
 \tag{3-31}$$

e. Impulse Due to Overcoming External Torques

$$\begin{aligned}
 I_{x_5} &= M_{D_x} \tau_x / d_x \\
 I_{y_5} &= M_{D_y} \tau_y / d_y \\
 I_{z_5} &= 0
 \end{aligned}
 \tag{3-32}$$

f. Total Impulse

$$\begin{aligned}
 I_x &= \sum_{i=1}^5 I_{x_i} \\
 I_y &= \sum_{i=1}^5 I_{y_i} \\
 I_z &= \sum_{i=1}^5 I_{z_i} \\
 I &= I_x + I_y + I_z
 \end{aligned}
 \tag{3-33}$$

3.4.3 Design Logic

The sequence followed in implementing the design algorithm is as follows:

- a. Input data.
- b. Compute minimum thrust requirement to overcome external torques.
- c. Compute the maximum thrust requirement such that the rate error equation will be satisfied.
- d. Use a thrust level which is between the computed max and min.
- e. Select the deadband based on the maximum allowable pointing error.
- f. Call up configuration-special hardware from the data base which are not differentiable:
 1. Valve Driver Assembly
 2. Sun Sensor Assembly
 3. Rate Gyro Assembly
 4. Power Converter
- g. Select an earth sensor with e_1 less than one half of the deadband with associated e_2 and e_3 . Select a Center Electrical Assembly with an error of c_6 .
- h. Compute the pointing error using Equation (3-24).
- i. Compare the pointing error with the maximum allowable pointing error. If acceptable, add this earth sensor and controller to acceptable list.
- j. Size and select a reaction wheel for data base.
- k. Compute the total impulse required for the mission.

3.5 THREE-AXIS MASS EXPULSION CONFIGURATION

3.5.1 Functional Description

The Three-Axis Mass Expulsion configuration components perform the following functions:

- a. Attitude Sensors. Two horizon scanners mounted in a "V" configuration detect when the vehicle's yaw axis is pointed at the earth's center. This provides attitude errors in pitch and roll, which provide the reference for the gyros and bound the effect of gyro drift .
- b. Gyros. Three rate-integrating gyros measure inertial rates about all three axes and integrate these rates to provide attitude information.
- c. Attitude Reference Electronics. The electronics process the outputs from the horizon scanners and gyros to produce an estimate of the vehicle's roll, pitch, and yaw attitude.
- d. Valve Drive Electronics. The electronics utilize the output of the attitude reference electronics to command the reaction control thrusters to correct for errors.

A block diagram of the Three-Axis Mass Expulsion configuration is presented in Figure 3-8.

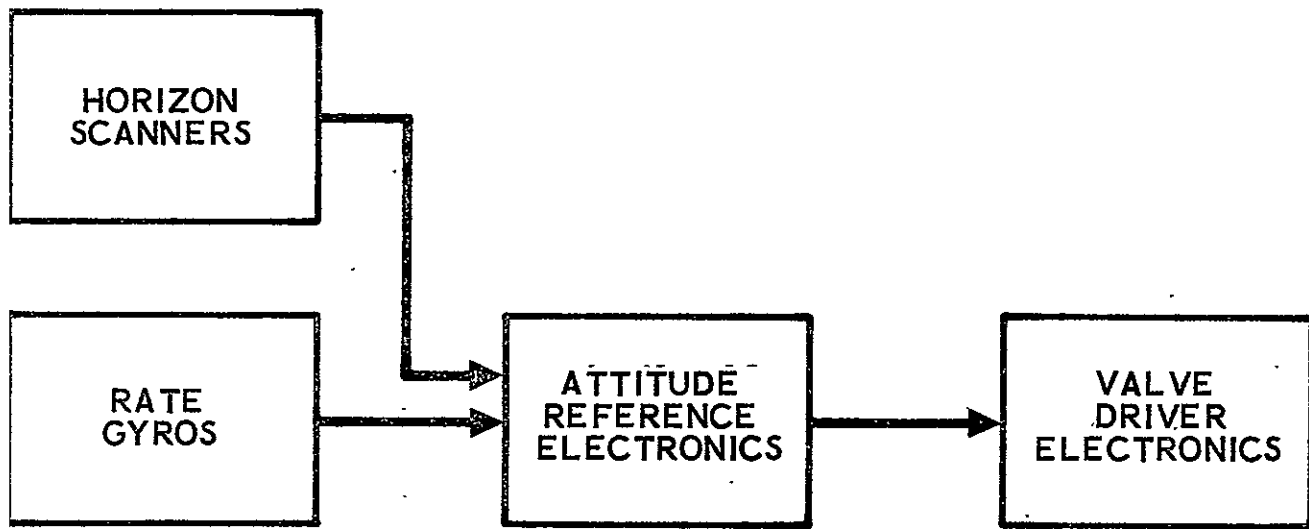


Figure 3-8. Three-Axis Mass Expulsion Configuration Block Diagram

3.5.2 Design Equations

The pointing equations for a typical Three-Axis Mass Expulsion S&C Subsystem have been developed in Reference 3-3. The control law used signals from three rate integrating gyros (one per vehicle axis) and

signals from two horizon scanners to estimate the vehicle roll, pitch and yaw attitude relative to the local vertical/orbit plane (LV/OP) coordinate axes.

A block diagram of the Attitude Reference Unit (taken from Ref. 3-3) is shown in Figure 3-9. The on-orbit accuracy equations are rewritten as follows:

$$E_y = db_y + \left\{ P_{\theta N} + (e_7 H_{\theta} / H_{F\theta})^2 + (e H_{\theta} / H_{F\theta})^2 + a_y^2 + (e_p / H_{F\theta})^2 + \Delta db_y^2 \right\}^{1/2} \quad (3-34)$$

$$E_x = db_x + \left\{ P_{\phi N} + (e_7 G_1)^2 + (e G_1)^2 + a_x^2 + \Delta db_x^2 \right\}^{1/2} \quad (3-35)$$

$$E_z = db_z + \left\{ (g_2 \omega_0 G_3)^2 + (g_1 G_3)^2 + (g_2)^2 + \left(\frac{g_1}{\omega_0} \right)^2 + a_z^2 + (e_7 G_2)^2 + (\Delta db_z)^2 \right\}^{1/2} \quad (3-36)$$

where

$$G_1 = H_{\psi} / (H_{F\psi} + \omega_0) \quad (3-37)$$

$$G_2 = \left[H_{\psi} H_{F\phi} - H_{\phi} (H_{F\psi} + \omega_0) \right] / \left[\omega_0 (H_{F\psi} + \omega_0) \right] \quad (3-38)$$

$$G_3 = H_{F\phi} / \left[\omega_0 (H_{F\psi} + \omega_0) \right] \quad (3-39)$$

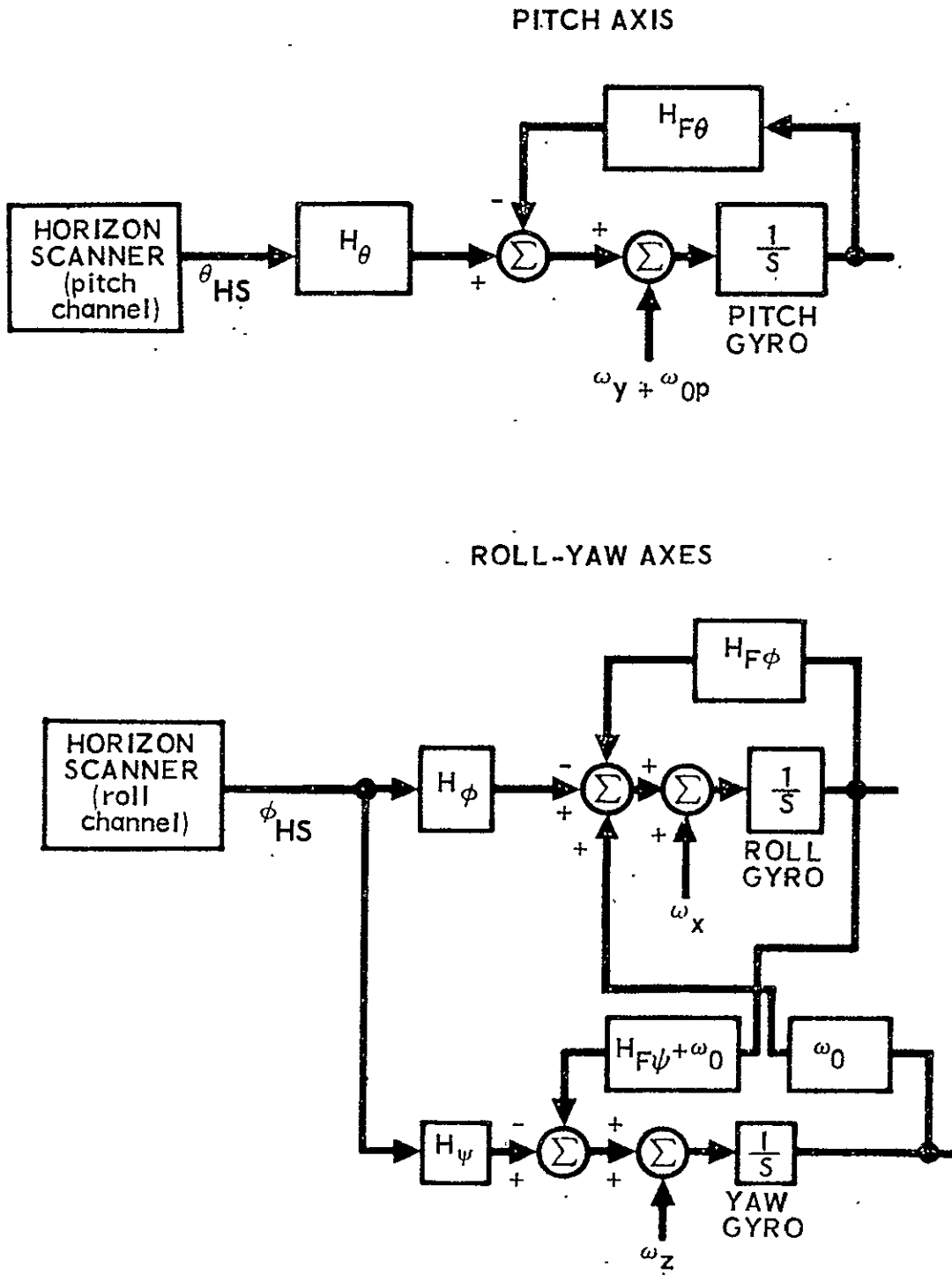


Figure 3-9. Attitude Reference Unit Block Diagram

The sensitivity terms are:

- H_{θ} = pitch horizon scanner gain
- H_{ϕ} = roll horizon scanner gain to roll axis
- H_{ψ} = roll horizon scanner gain to yaw axis
- $H_{F\theta}$ = pitch feedback gain
- $H_{F\phi}$ = roll feedback gain
- $(H_{F\psi} + \omega_0)$ = roll-to-yaw coupling gain
- ω_0 = true orbital pitchover rate

The error terms are:

a. Pitch Axis

- $P_{\theta N}$ = filtered pitch horizon scanner noise power
[defined in Eq. (3-45)]
- e_7 = pitch horizon scanner null error
- e = Horizon scanner anomalies (deg, 3σ)
 $\hat{=} (e_2^2 + e_6^2)^{1/2} + (e_4^2 + e_5^2)^{1/2}$
- e_2 = error due to radiance irregularity (3σ)
- e_4 = sun interference error (3σ)
- e_5 = moon interference error (3σ)
- e_6 = threshold aging error (3σ)
- e_p = programmed pitchover rate error
($\omega_{0p} - \omega_0$)
- a_y = misalignment error in mounting inertial measurement
units relative to vehicle pitch axis (deg, 3σ)
- db_y = pitch deadband setting
- Δdb_y = pitch deadband tolerance

b. Roll Axis

- $P_{\theta N}$ = filtered roll horizon scanner noise power
 [defined in Eq.(3-49)]
- e_7 = horizon scanner null error
- e = horizon scanner anomalies
- a_x = misalignment error in mounting inertial measurement
 units relative to vehicle roll axis (deg, -3σ)
- db_x = roll deadband setting
- Δdb_x = roll deadband tolerance

c. Yaw Axis

- g_2 = pitch misalignment of yaw gyro relative to sensor package
- g_1 = G-insensitive drift of yaw gyro
- g_2 = pitch misalignment of roll gyro relative to sensor package
- g_1 = G-insensitive drift of roll gyro
- e_7 = roll horizon scanner null error
- a_z = misalignment error in mounting inertial measurement
 units relative to vehicle yaw axis (deg, -3σ)
- db_z = yaw deadband setting
- Δdb_z = yaw deadband tolerance

The expressions for the filtered pitch and roll horizon scanner
 e power are given in Reference 3-3 as:

$$P_{\theta N} = \int_0^{\infty} \frac{S_{\theta}(\omega) H_{\theta}^2}{\omega^2 + H_{F\theta}^2} d\omega \quad (3-41)$$

$$P_{\phi N} = \int_0^{\infty} \frac{S_{\phi}(\omega) \left[\omega^2 H_{\phi}^2 + \omega_0^2 H_{\psi}^2 \right]}{\omega^4 + \omega^2 \left(H_{F\phi}^2 - 2H_{\phi}/G_3 \right) + H_{\phi}^2/G_3^2} d\omega \quad (3-42)$$

where S_θ and S_ϕ are the pitch and roll horizon scanner output noise power spectra, respectively.

The assumption for S_θ and S_ϕ is made that the horizon scanner output noise is white noise and is band limited at a frequency ω_H as shown in Figure 3-10.

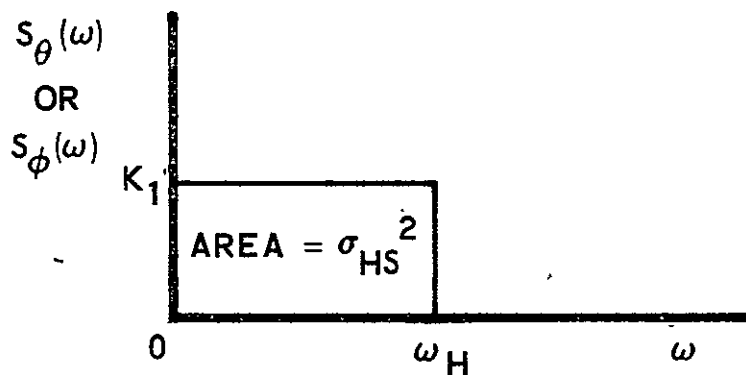


Figure 3-10. Band-Limited White Noise

The area is equal to:

$$\begin{aligned} \text{Area} &= \sigma_{HS}^2 \\ &= K_1 \omega_H \end{aligned} \quad (3-43)$$

where ω_H is the band limited frequency and σ_{HS} is the 1σ value of horizon scanner noise. Therefore,

$$S_\phi(\omega) = S_\theta(\omega) = \begin{cases} \sigma_{HS}^2 / \omega_H & \text{for } 0 \leq \omega \leq \omega_H \\ 0 & \text{for } \omega > \omega_H \end{cases} \quad (3-44)$$

Substituting Equation (3-44) into Equations (3-42) and (3-43) and using a 3σ noise error to get a 3σ value for noise power yields

$$P_{\theta N}(3\sigma) = \frac{c_1^2}{\omega_H} \int_0^{\omega_H} \frac{H_\theta^2}{(\omega^2 + H_{F\theta}^2)} d\omega$$

$$P_{\theta N}(3\sigma) = \frac{e_1^2 H_\theta^2}{\omega_H H_{F\theta}} \tan^{-1} \left(\frac{\omega_H}{H_{F\theta}} \right) \quad (3-45)$$

where

$$e_1 = \text{horizon scanner noise } (3\sigma)$$

Also:

$$P_{\varphi N} = \frac{e_1^2}{\omega_H} \int_0^{\omega_H} \frac{(\omega^2 H_\varphi^2 + \omega_0^2 H_\psi^2)}{[\omega^4 + \omega^2 (H_{F\varphi}^2 - 2H_\varphi/G_3) + (H_\varphi^2 + G_3^2)]} d\omega \quad (3-46)$$

Since the above equation cannot be solved analytically, it is desirable to find another solution to avoid integrating in the computer simulation.

In Reference 3-4, Equation (3-46) is solved numerically for "typical" values of the attitude reference parameters. Since these values do not vary a great deal between various attitude reference systems, the value of the numerical integration is used and scaled appropriately by e_1 and ω_H . From Reference 3-4:

$$\frac{e_1^2}{\omega_H} C = 0.0013 \quad (3-47)$$

or

$$C = \frac{0.0013 \omega_H}{e_1^2} = 0.026 \quad (3-48)$$

for this attitude reference system.

Thus

$$P_{\varphi N} \cong \frac{e_1^2}{\omega_H} (0.026) \quad (3-49)$$

The total impulse required is a function of the requirements for limit cycling, powered flight, recovery from initial rates, single axis rotational maneuvers, and overcoming external torques:

a. Limit Cycle Impulse (3-50)

$$I_{x_1} = 37 \times 10^6 T d_x (F \Delta t)^2 / (J_x db_x)$$

$$I_{y_1} = 37 \times 10^6 T d_y (F \Delta t)^2 / (J_y db_y)$$

$$I_{z_1} = 37 \times 10^6 T d_z (F \Delta t)^2 / (J_z db_z)$$

b. Powered Flight Impulse (Translational) (3-51)

$$I_{y_2} = \left(\frac{\Delta \theta}{57.3} d_T + \Delta d \right) \frac{F_e t}{d_y}$$

$$I_{z_2} = \left(\frac{\Delta \theta}{57.3} d_T + \Delta d \right) \frac{F_e t}{d_z}$$

$$I_{x_2} = \frac{2}{57.3} \Delta d \quad \Delta \theta \quad \frac{F_e t}{d_x}$$

c. Impulse due to Recovery from Initial Rates (3-52)

$$I_{x_3} = \eta J_x \dot{\theta}_0 / (57.3 d_x)$$

$$I_{y_3} = \eta J_y \dot{\theta}_0 / (57.3 d_y)$$

$$I_{z_3} = \eta J_z \dot{\theta}_0 / (57.3 d_z)$$

d. Single Axis Rotation Impulse (3-53)

$$I_{x4} = \frac{2 J_x \dot{\theta}_x(\max) N_x}{57.3 d_x}$$

$$I_{y4} = \frac{2 J_y \dot{\theta}_y(\max) N_y}{57.3 d_y}$$

$$I_{z4} = \frac{2 J_z \dot{\theta}_z(\max) N_z}{57.3 d_z}$$

e. Impulse Due to Overcoming External Torques (3-54)

$$I_{x5} = M_{D_x} \tau_x / d_x$$

$$I_{y5} = M_{D_y} \tau_y / d_y$$

$$I_{z5} = M_{D_z} \tau_z / d_z$$

f. Total Impulse (3-55)

$$I_x = \sum_{i=1}^5 I_{x_i}$$

$$I_y = \sum_{i=1}^5 I_{y_i}$$

$$I_z = \sum_{i=1}^5 I_{z_i}$$

$$I = I_x + I_y + I_z$$

3.5.3 Design Logic

The sequence followed in implementing the design algorithm is as follows:

- a. Input data.
- b. Compute minimum forces necessary to overcome main engine misalignments and external torques.
- c. Compute maximum allowable force such that rate error equation will be satisfied.
- d. Use a thrust level which is between the computed maximum and minimum.
- e. Select the deadband based on the maximum allowable pointing error.
- f. Call up configuration special hardware from the data base which are not differentiable:
 1. Attitude Reference Electronics
 2. Valve Driver Electronics
- g. Select rate-integrating gyros with rate sensing threshold g_1 less than 0.2 of the maximum allowable rate error and with associated g_2 .
- h. Select scanning horizon sensor from data base with associated $e_1, e_2, e_4, e_5, e_6, e_7$, and ω_H .
- i. Compute the pointing errors.
- j. Compare the pointing errors with the maximum allowable pointing error. If acceptable, use the selected components. If not acceptable, select more accurate components and repeat the sequence.
- k. Compute the total impulse required for the mission.

3.6 MASS EXPULSION CONFIGURATION WITH CONTROL MOMENT GYROS

3.6.1 Functional Description

The Mass Expulsion configuration with Control Moment Gyros, subsequently referred to as the Control Moment Gyro configuration, consists of components performing the following functions:

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- a. Attitude Reference Unit. The attitude reference unit provides accurate rate and attitude information for control by the momentum exchange assembly. The attitude reference unit should be of high accuracy when used with a momentum exchange assembly, thereby requiring star sensors and an accurate rate gyro assembly.
 1. The star sensor assembly provides accurate attitude information which is processed by the onboard computer. Three types of star sensors are considered: star mappers, body-fixed (electronic) star trackers, and gimbaled star trackers.
 2. The gyro reference assembly (GRA) provides rate information which is used in connection with the attitude information in the data processing assembly.
 3. The electronic processor assembly provides the function of processing the signals from the star sensor assembly and the GRA electronically for use in the onboard computer.
- b. Data Processing Assembly. The data processing assembly consists of the onboard computer and performs the following functions:
 1. Derivation of the attitude reference using the signals from the attitude reference unit.
 2. Processing of the commands for attitude, rate, and acceleration (if needed) to generate commands to the momentum exchange assembly for vehicle pointing.
 3. Processing of control laws governing how the individual momentum exchange devices are controlled and when to fire the jets to dump momentum (or for attitude control in place of the momentum exchange assembly).
- c. Momentum Exchange Assembly. Two basic types of three-axis momentum exchange assemblies are considered:
 1. Control moment gyros (CMGs)
 2. Three-axis reaction wheels

The CMGs may be either single-gimbaled or double-gimbaled. The momentum exchange configuration selected for this study is the one utilized by LST and HEAO which is a skewed arrangement of from four to six single-gimbaled CMGs. These are constant-speed CMGs with a total angular momentum vector varied by varying the gimbal angles of the individual CMGs.

3.6.2.1 Momentum Exchange Assembly

- a. CMGs Versus Three-Axis Reaction Wheels. The factors influencing the decision between CMGs and three-axis wheels are the momentum storage and torque requirements. The momentum storage requirement consists of maneuver and disturbance momentum requirements with the equation given as follows:

$$\begin{aligned} h_{\text{req}} &= h_{\text{man}} + h_{\text{dist}} \\ &= (J \dot{\theta}_{\text{max}} / 57.3) + 86400 t_{\ell} M_D(\text{max}) \end{aligned} \quad (3-56)$$

where

t_{ℓ} = time between unloading (days)

J = inertia about the maneuver axis (slug-ft²)

$\dot{\theta}_{\text{max}}$ = maximum maneuver rate about the maneuver axis (deg/sec)

$M_D(\text{max})$ = maximum disturbance torque

For sizing purposes,

$$h_{\text{man}} = \max \left\{ \frac{J_x \dot{\theta}_x(\text{max})}{57.3}, \frac{J_y \dot{\theta}_y(\text{max})}{57.3}, \frac{J_z \dot{\theta}_z(\text{max})}{57.3} \right\} \quad (3-57)$$

The torque requirement consists of the sum of the acceleration torque and the disturbance torque requirements.

$$\begin{aligned} T_{\text{req}} &= T_{\text{accel}} + T_{\text{dist}} \\ &= (J \dot{\theta}_{\text{max}} / (57.3 t_{\text{accel}})) + M_D(\text{max}) \end{aligned} \quad (3-58)$$

where

t_{accel} = acceleration time (sec)

This can be written as

$$T_{\text{req}} = h_{\text{man}} / t_{\text{accel}} + M_D(\text{max}) \quad (3-59)$$

If $T_{\text{req}} / h_{\text{req}} > 0.02$, then CMGs should be considered

If $0 > T_{\text{req}} / h_{\text{req}} > 0.1$, then three axis wheels should be considered.

- b. Single Versus Double-Gimbaled CMGs. The decision to select single or double-gimbaled CMGs is based on the gimbal rate requirement, $\dot{\sigma}_{\max}$.

If $\dot{\sigma}_{\max} < 0.1$ rad/sec, then both types should be considered.

If $\dot{\sigma}_{\max} > 0.1$ rad/sec, then only single-gimbaled CMGs should be considered.

Alternately, the gimbal rate can be related to the acceleration time requirement by assuming typical worst-case gimbal motion during acceleration or deceleration of approximately 2 rads. This yields:

If $t_{\text{accel}} > 20$ sec, consider both types.

If $t_{\text{accel}} < 20$ sec, consider only single-gimbaled CMGs.

- c. Select and Size Single-Gimbaled CMGs. Single-gimbaled CMGs are selected and sized based on the momentum storage requirement, peak gimbal rates, and peak torquer torques.

The momentum storage requirement is the sum of the maneuver momentum and disturbance momentum requirements. The momentum required per axis is shown below:

$$\begin{aligned} h_{\text{req}_x} &= [J_x \dot{\theta}_x (\text{max}) / 57.3] + M_D (\text{max}) t_\ell \\ h_{\text{req}_y} &= [J_y \dot{\theta}_y (\text{max}) / 57.3] + M_D (\text{max}) t_\ell \\ h_{\text{req}_z} &= [J_z \dot{\theta}_z (\text{max}) / 57.3] + M_D (\text{max}) t_\ell \end{aligned} \quad (3-60)$$

For this study we will assume a skewed configuration of from 4 to 6 CMGs. The smaller number is preferred unless more are needed to meet the required momentum with off-the-shelf CMGs. It is further assumed here that the hangup problem is not solved so that the maximum momentum capability of the set of CMGs is approximately:

$$h_{\text{cap}} = (n - 2) h \cos \gamma \quad (3-61)$$

where

n = number of CMGs in the skewed configuration
($4 \leq n \leq 6$)

h = momentum of each CMG.

γ = skew angle (default value of 25 deg)

$$= \sin^{-1} \left(\frac{h_{\text{min}}}{nh} \right) \quad (3-62)$$

Letting

$$h_{\text{cap}} = \max (h_{\text{req}_x}, h_{\text{req}_y}, h_{\text{req}_z}) \quad (3-63)$$

$$h_{\text{min}} = \min (h_{\text{req}_x}, h_{\text{req}_y}, h_{\text{req}_z}) \quad (3-64)$$

then

$$\gamma = \tan^{-1} \frac{h_{\text{min}} (n-2)}{h_{\text{cap}} n} \quad (3-65)$$

$$h = h_{\text{min}} / (n \sin \gamma) \quad (3-66)$$

The peak gimbale rate is determined by the acceleration time and the fact that a single CMG in a skewed set will typically move two rads at most while the vehicle accelerates from rest to $\dot{\theta}_{\text{max}}$ (for typical LST CMGs). The peak gimbale rate is approximately equal to the average rate since the gimbale acceleration is high. Thus:

$$\dot{\sigma}_{\text{max}} \cong \frac{2h}{\underline{h} t_{\text{accel}}} \quad (3-67)$$

where \underline{h} is the actual momentum of the individual CMGs selected from the hardware data base.

The peak torquer torque (which must be delivered at peak gimbale rate) is:

$$T_{\text{max}} = \underline{h} \dot{\theta}_{\text{max}} / 57.3 \quad (3-68)$$

where

$$\dot{\theta}_{\text{max}} = \max \left(\dot{\theta}_x (\text{max}), \dot{\theta}_y (\text{max}), \dot{\theta}_z (\text{max}) \right) \quad (3-69)$$

The CMGs are selected from the hardware data base by using \underline{h} , $\dot{\sigma}_{\text{max}}$, and T_{max} such that each value of the selected CMG is above the values of \underline{h} , $\dot{\sigma}_{\text{max}}$, and T_{max} . The actual values for the CMG selected from the data base will be denoted by \underline{h} and $\dot{\sigma}_{\text{max}}$ for use in the following sizing equations.

The CMG configuration is sized for weight, standby power, and volume utilizing empirical relationships. These relationships have been approximated by straight line equations to be programmed for this model. The equations are presented below. The weight and volume numbers include the CMG and its associated gimbale drive electronics. The power

numbers are standby power only. Peak power during acceleration of a maneuvering vehicle can be twenty times as large as the standby power. Some knowledge of the duty cycle is required to compute average power for any particular application.

$$K_2 = \frac{\dot{\theta}_{\max}}{\theta_{\max}} / 57.3 \quad (3-70)$$

$$W = n [-32 + (0.068 + 0.29K_2) (\underline{h} + 960)] \quad (3-71)$$

$$P = n (0.0103 + 0.0235 K_2) (\underline{h} + 1430) \quad (3-72)$$

$$V = n [7.45 + (0.00265 - 0.0062 K_2) (\underline{h} - 1720)] \quad (3-73)$$

where

W = weight of CMG configuration (lb)

P = standby power of CMG configuration (watt)

V = volume of CMG configuration (ft³)

1. CMG Rate and Pointing Errors. Assuming that a CMG control system, sized for a slew rate of $\dot{\theta}_{\max}$, is in an inertial hold mode. The state of the art in CMG control is such that 2σ low-frequency (within the control bandwidth) errors are approximately

$$\theta_{\text{CMG}} = 0.7 \dot{\theta}_{\max} (\widehat{\text{sec}}, 2\sigma) \quad (3-74)$$

$$\dot{\theta}_{\text{CMG}} = 0.2 \dot{\theta}_{\max} (\widehat{\text{sec/sec}}, 2\sigma) \quad (3-75)$$

These numbers are based on GE Phase 1 CMG development test results (Ref. 3-5) on a single-axis, air-bearing table. Estimated HEAO errors are somewhat larger than this (1 $\widehat{\text{sec}}$ for a system which could slew at approximately 0.6 deg/sec). CMGs also induce vibration errors at spin speed and bearing regainer speed (approximately 40 percent of the spin speed), but their effect depends on the specific structural design and is beyond the scope of this effort.

For this model, the errors must be in terms of 3σ numbers and the units in degrees and seconds. The above equations are shown below in these terms.

$$\theta_{\text{CMG}} = 0.0000833 \dot{\theta}_{\max} \quad (3-76)$$

$$\dot{\theta}_{\text{CMG}} = 0.0002916 \dot{\theta}_{\max} \quad (3-77)$$

- d. Valve Driver Electronics. The electronics utilizes the output of the data processing assembly to command the reaction control thrusters to dump the stored momentum in the momentum exchange devices or to control the vehicle in place of the momentum exchange assembly.
- e. Coarse Attitude Sensing. For acquisition purposes, a coarse attitude sensing assembly is required. This may consist of either a horizon sensor for earth-pointing vehicles or a sun sensor for inertially-pointing vehicles.

A block diagram of the Control Moment Gyro configuration is presented in Figure 3-11.

3.6.2 Design Equations

The Control Moment Gyro configuration will consist of the momentum exchange assembly coupled with an accurate attitude reference assembly. The attitude reference assembly will contain star sensors for accurate attitude information along with a gyro reference assembly (GRA) for rate information. The outputs from the star sensor and GRA will be processed by an error processor and evaluated by an onboard computer.

The momentum exchange assembly can consist of three-axis reaction wheels or control moment gyros. Equations for the three-axis wheel configuration will not be covered in this model, although the decision process concerning three-axis wheels is covered. Two types of CMGs are considered in the decision process; double-gimbaled CMGs (whose equations are not covered in this model) and single-gimbaled CMG systems.

The total error of the Control Moment Gyro configuration is the sum of the attitude reference error plus the control (CMG) error. The equations for the errors will be covered later.

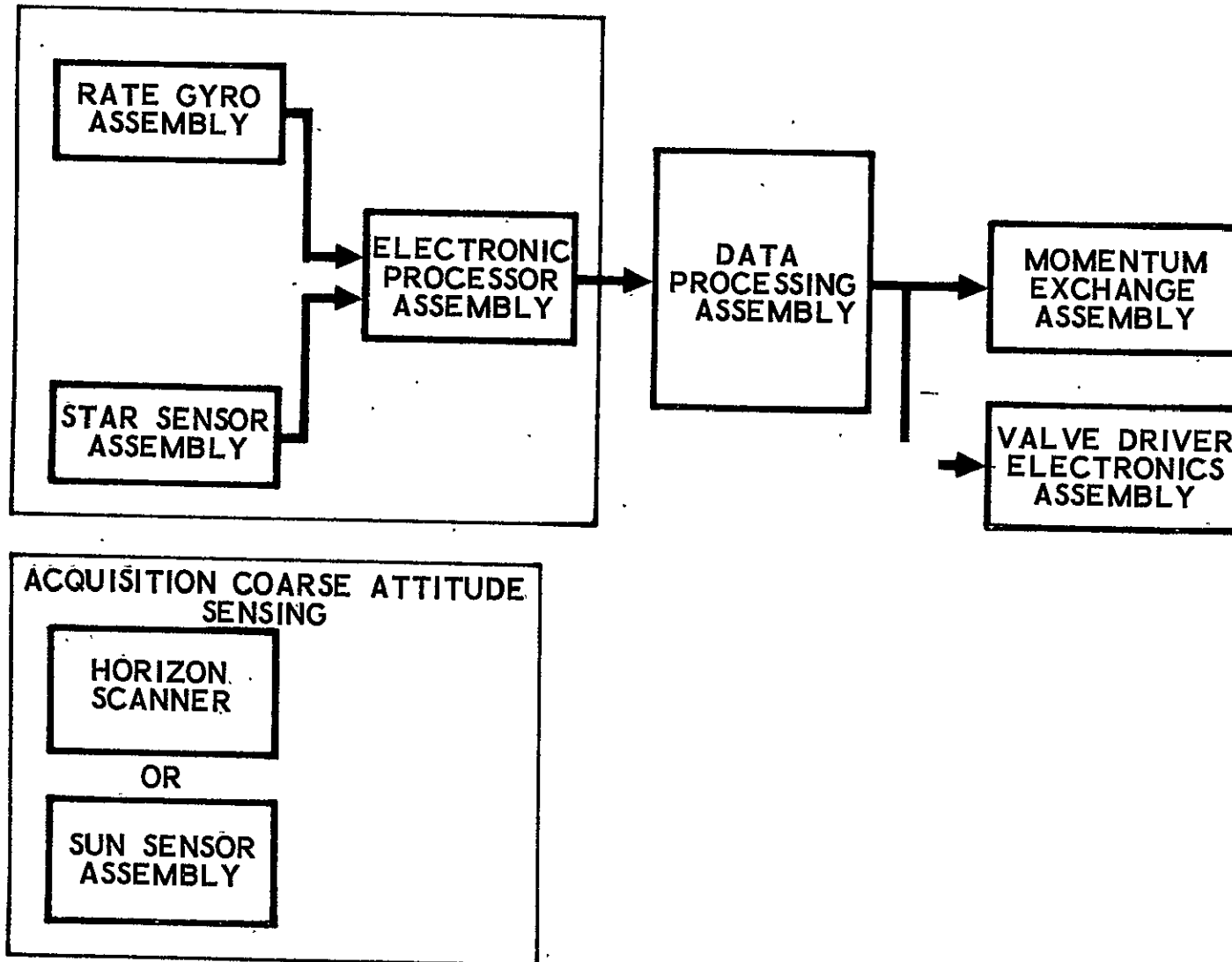


Figure 3-11. Mass Expulsion Configuration Block Diagram with Control Moment Gyros

3.6.2.2 Selection of Star Sensors

Several types of star sensors are available in an accurate attitude reference system. The three types considered in this study include star mappers, body-fixed (electronic) trackers, and gimbaled trackers.

a. Decisions Between Candidate Star Sensors. Several tests are listed in the following material to rule out star sensor types which will not meet specific requirements.

1. Test to determine whether star information must be obtained in inertial hold.

Assume that the gyro bias drift can be calibrated to ± 0.01 deg/hr. Allow one-half of the required pointing error for the bias drift during inertial hold. Thus

$$\frac{\theta_R}{2} = \dot{\theta}_D t_{\max} \quad (3-78)$$

or

$$\begin{aligned} t_{\max} &= \frac{\theta_R}{2 \dot{\theta}_D} \\ &= \frac{\theta_R (60)}{2 (0.01)} \\ &= 3000 \theta_R \text{ (minutes)} \end{aligned} \quad (3-79)$$

where

$$\theta_R = \min (\theta_{R_x}, \theta_{R_y}, \theta_{R_z}) \quad (3-80)$$

If the time the vehicle must be in inertial hold, t_{hold} , exceeds t_{\max} , then it is necessary to use a star tracker to obtain star information during inertial hold. Therefore, star mappers are rejected.

2. Test to determine whether attitude motion is sufficient to generate an adequate number of star crossings with a star mapper.

To make judgment on this question requires tradeoffs between mapper field of view (FOV), accuracy, sensitivity, number of mappers, mounting geometry, and

attitude motion profiles. As a rough test for whether a star mapper is a practical approach, determine whether the average body rate exceeds one-fourth of the orbital rate, ω_o , for a low orbit during the period when high accuracy is necessary. This corresponds to a rate of 1 deg/min. If the average body rate is greater than this, then star mappers should be considered.

3. Test to determine whether high body rates preclude star trackers.

A thorough resolution of this question requires extensive tradeoffs between sensitivity (required star magnitude), FOV, accuracy, tracking loop bandwidth and degree of sophistication, command sophistication (i. e., acceleration feed forward, vehicle dynamic model in computer, etc.) and detailed tracker mechanization. However, for preliminary component selection purposes, it is reasonable to exclude trackers from consideration for body rates exceeding some value based on existing trackers (assuming that star information must be obtained at the high body rates). Define two different thresholds, one for body-fixed (electronic) trackers and one for gimballed trackers. Input the maximum vehicle rate at which star information must be obtained and compare this to the two thresholds to see if trackers should be rejected. It may be possible for trackers to operate at higher rates with reduced accuracy. If so, this must be factored into the tracker selection algorithm.

4. Given that both types of trackers are candidates, test to determine whether large view angles necessitate gimbals.

A requirement for large view angles is generated when it is desired to track specific guide stars over wide ranges of vehicle motion. The maximum reasonable total FOV for electronic trackers is approximately 30 deg. Input whether the mission requirements dictate tracking specific guide stars and what the range of attitude freedom is to be (denoted by θ_{FOV}). Test this against 30 deg to see if body-fixed trackers should be rejected.

- b. Star Tracker. Given the surviving candidate types of star sensors from the preliminary selection, select specific sensors from the data base with accuracy as a criterion. Reject all sensors whose accuracy is not better than the required attitude accuracy. The reason for allowing a star sensor error to be equal to the entire system pointing error is that the effective star sensor error can be reduced below this value by averaging the random errors and calibration of systematic errors, thus allowing room for other system errors.

3.6.2.3 Momentum Exchange Subsystem Error Equations

- a. Rationale. The CMG attitude control system analyses and tests reported in Reference 3-5, along with related results, indicate that the errors in the control moment gyro configuration can be characterized as follows:
1. Pointing errors* are dominated by the attitude reference error, which consists primarily of the star sensor error plus any gyro drift that occurs between star sightings. These are low-frequency errors, and the control system follows them essentially perfectly. In addition, there are higher frequency "jitter" pointing errors which are caused primarily by CMG hardware characteristics such as torquer stiction, ripple, and backlash. CMG vibration can cause significant rate error, but generally causes negligible position error because of the high frequency of the vibration source (CMG spin speed, typically 100 Hz). The state of the art in CMGs, gyros, and star sensors is such that the pointing errors caused by gyro and star sensor noise are generally insignificant compared to those caused by the CMGs. Pointing errors produced by disturbance torques are generally insignificant but, if necessary, can be reduced to negligible values by computing the torques and feeding them forward as commands to the control system. It is assumed for present purposes that this is done.
 2. Rate errors are dominated by the system response to CMG jitter. The rates associated with attitude reference errors such as the uncompensated components of gyro drift are negligible. Response to gyro and star sensor noise is small compared to the jitter produced by the CMGs. High frequency vibration produced by the CMG wheel and spin bearings is an important and sometimes dominant source of rate error. It requires very careful attention to CMG balancing, structural design and tuning, and possibly vibration isolation mounting of the CMGs. However, its magnitude is so sensitive to the detailed system design that there is no reasonable way to quantify this error for present purposes.

* Pointing errors in this discussion are measured relative to the attitude reference system; misalignments of the payload relative to the attitude reference system are assumed to be considered separately.

b. Error Equations

1. System with Star Trackers. The total pointing error (θ_ϵ) is the RSS of the star tracker accuracy (s_2) and the CMG-produced jitter (θ_{CMG}). Although some of the star tracker errors will be noise and will be averaged by the system, this will be approximately compensated by the sum of the other smaller errors which have been ignored here, resulting in total system performance of the order indicated above. Thus

$$\theta_\epsilon = (s_2^2 + \theta_{\text{CMG}}^2)^{1/2} \quad (3-81)$$

The total rate error is the RSS of the CMG-produced jitter and the CMG-produced vibration, the latter of which is unknown at this time. Thus, the rate error equation is:

$$\dot{\theta}_\epsilon = \dot{\theta}_{\text{CMG}} \quad (3-81a)$$

2. System with Star Mappers. The system rate is the same as above. The position error is the same as above with the RSS addition of errors due to gyro drift. The gyro drift error is computed as follows.

The star mapper FOV, s_3 , and the sensitivity (star magnitude, m), s_4 , are obtained from the star mapper data base for the candidate mapper. The equation below for the average star density, N_m , is a function of s_4 .

$$N_m = \log^{-1} (-4 + 0.451 s_4) \quad (3-82)$$

where

$$s_4 = \text{mapper sensitivity (visual magnitude)} \\ \text{and ranges from 0 to 20}$$

The average time between star crossings is computed by

$$t_{\text{sc}} = \frac{57.3}{N_m s_3 \dot{\theta}_{\text{max}}} \quad (3-83)$$

where

$$s_3 = \text{star mapper field of view (deg}^2\text{)}$$

$$\dot{\theta}_{\text{max}} = \text{maneuver rate (deg/sec)}$$

The average angular motion between star crossings is

$$\Delta \theta_{avg} = \dot{\theta}_{max} t \quad (3-84)$$

The gyro bias drift error is computed from

$$\begin{aligned} \theta_{e_b} &= \dot{\theta}_b t_{sc} \\ &= 0.1 g_1 t_{sc} \end{aligned} \quad (3-85)$$

where $g_1 = G$ -insensitive gyro drift (deg/sec, 3σ) and the gyro bias drift, $\dot{\theta}_b$, has arbitrarily set to ten percent of the total drift. The gyro scale factor error is

$$\theta_{e_{sf}} = K_{sf} \Delta \theta_{avg} \quad (3-86)$$

where K_{sf} is the gyro scale factor multiplier (3σ), which is obtained from the hardware data base.

The total system error with star mappers thus becomes

$$\theta_e = \left(s_2^2 + \theta_{CMG}^2 + \theta_{e_b}^2 + \theta_{e_{sf}}^2 \right)^{1/2} \quad (3-87)$$

3.6.2.4 Size Thrusters

The only factor sizing the thrusters for the control moment gyro configuration is that they are able to control the vehicle's attitude during firing of the main engine with its associated misalignments. The equations for required thrust levels are

$$\left. \begin{aligned} F_1 &= \left(\frac{\Delta \theta}{57.3} d_T + \Delta d \right) \frac{F_e}{d_y} \\ F_2 &= \left(\frac{\Delta \theta}{57.3} d_T + \Delta d \right) \frac{F_e}{d_y} \\ F_3 &= \frac{2}{57.3} \Delta d \Delta \theta \frac{F_e}{d_x} \end{aligned} \right\} \quad (3-88)$$

$$F = \max \text{ of } (F_1, F_2, F_3) \quad (3-89)$$

where

d_x, d_y, d_z = gas jet level arms for the roll, pitch, and yaw gas jets (ft)

Δd = lateral vehicle c.g. distance plus lateral thrust chamber c.g. distance from the reference axis (ft). Default value is 4% of the vehicle diameter.

$\Delta \theta$ = main engine alignment relative to the vehicle thrust axis (deg)

F_e = main engine thrust (lb)

3.6.2.5 Size Impulse

Three factors influence the impulse required for the Control Moment Gyro configuration mission. These are the impulse due to controlling the vehicle during translational powered flight, the impulse required to dump the momentum in the CMG's accumulated due to the steady disturbance torques, and the impulse for recovery from initial rates. The equations for impulse due to these three factors are shown below:

a. Powered Flight Impulse

$$\left. \begin{aligned} I_{y_2} &= \left(\frac{\Delta \theta}{57.3} d_T + \Delta d \right) \frac{F_e t}{dy} \\ I_{z_2} &= \left(\frac{\Delta \theta}{57.3} d_T + \Delta d \right) \frac{F_e t}{dz} \\ I_{x_2} &= \frac{2}{57.3} \Delta d \Delta \theta \frac{F_e t}{dx} \end{aligned} \right\} \quad (3-90)$$

where t = main engine burn time (sec).

b. Momentum Dumping Impulse

$$\left. \begin{aligned} I_{x_1} &= 2.592 \times 10^6 M_{D_x} \text{ (steady) } T/d_x \\ I_{y_1} &= 2.592 \times 10^6 M_{D_y} \text{ (steady) } T/d_y \\ I_{z_1} &= 2.592 \times 10^6 M_{D_z} \text{ (steady) } T/d_z \end{aligned} \right\} (3-91)$$

where

$$\left. \begin{aligned} M_{D_x} \text{ (steady)} \\ M_{D_y} \text{ (steady)} \\ M_{D_z} \text{ (steady)} \end{aligned} \right\} = \text{steady disturbance torque level} \\ \text{about the vehicle roll, pitch,} \\ \text{and yaw axes (ft-lb)}$$

T = mission lifetime (months)

c. Impulse Due to Recovery From Initial Rates

$$\left. \begin{aligned} I_{x_3} &= \eta J_x \dot{\theta}_o / (57.3 dx) \\ I_{y_3} &= \eta J_y \dot{\theta}_o / (57.3 dy) \\ I_{z_3} &= \eta J_z \dot{\theta}_o / (57.3 dz) \end{aligned} \right\} (3-9)$$

where

- η = control system efficiency (3 with 3-axis rate gyros, 12-15 with no rate sensing)
- J_x, J_y, J_z = vehicle roll, pitch, and yaw inertias (slug-ft²)
- $\dot{\theta}_o$ = maximum initial rate (deg/sec) (assumed the same on all axes)

3.6.3 Design Logic

The sequence followed in implementing the design algorithm is as follows:

- a. Input data.
- b. Decide on CMGs versus 3-axis reaction wheels.
 1. Compute ratio of maximum momentum storage to maximum torque requirement.
 2. Compare ratio with prescribed values for decision.
- c. Select CMG configuration (3-axis wheels not modeled).
 1. Double versus single-gimbaled CMGs (double-gimbaled CMGs not modeled).
- d. Select and size single-gimbaled CMGs.
 1. Compute momentum storage requirement for each CMG.
 2. Select a CMG from the hardware data base such that the momentum h , provided by the individual CMG, is $\geq h$. Associated with this CMG will be a peak gimbal rate, $\dot{\sigma}_{\max}$, and a peak torquer torque, T_{\max} .
 3. Compute the peak gimbal rate and peak torquer torque which a CMG with a given h must provide.
 4. Compare data base $\dot{\sigma}_{\max}$ and T_{\max} with the computed values. If the data base variables are greater than the computed values, add this CMG configuration to the acceptable list.
 5. Compute the weight, standby power, and volume for the CMG configuration if the configuration is acceptable.
- e. Compute CMG 3σ rate and pointing errors.
 1. Compare the CMG rate error with the minimum required rate error. Abort if not small enough.
- f. Select star sensors.
 1. Test to determine whether star information must be obtained in inertial hold. If yes, reject star mappers.
 2. Test to determine whether attitude motion is sufficient to generate an adequate number of star crossings with a star mapper. If no, reject star mappers.
 3. Test to determine whether high body rates preclude trackers. If yes, reject either body-fixed or gimbaled star trackers, or both. If both types are acceptable, test to determine whether large body angles necessitate gimbals.
- g. Select a gyro reference assembly.
- h. Select a star sensor from the acceptable types whose error is less than the minimum required accuracy.

- i. Compute the configuration pointing accuracy which is a function of the accuracy of the star sensor, CMGs, and (for star mappers) the gyro reference assembly (GRA).
- j. Compare the configuration pointing error with system pointing accuracy requirement. If less, add this subsystem combination to the acceptable list. If more, iterate on star sensors and GRAs until the requirement is met.
- k. Call up configuration-special hardware from the data base which is not differentiable. Test whether horizon sensors or sun sensors are called for coarse attitude sensing.
 - 1. Electronic Processor Assembly
 - 2. Valve Driver Electronics Assembly
 - 3. Horizon Sensor Assembly
 - 4. Sun Sensor Assembly
- l. Size thrusters to overcome main engine misalignments during powered flight.
- m. Size total impulse required based on:
 - 1. Dumping stored momentum due to steady disturbance torque bias
 - 2. Fighting main engine misalignments during powered flight
 - 3. Recovery from initial rates.

3.7 MASS EXPULSION CONFIGURATION WITH PITCH MOMENTUM WHEEL

3.7.1 Functional Description

The Mass Expulsion configuration with Pitch Momentum Wheel, subsequently referred to as the Pitch Momentum Bias Configuration, consists of components performing the following functions:

- a. Horizon Sensor Assembly. The horizon sensor measures attitude errors in roll and pitch relative to the local vertical.
- b. Momentum Wheel. The momentum wheel is aligned normal to the orbit plane along with negative pitch axis. This serves to provide restraint in roll and yaw.
- c. Electronic Processor. Damping of the system is provided in the electronic processor by a lead controller which is best

implemented by a pseudo-rate circuit. In particular, the controller in the roll channel serves to damp the nutation frequency mode of the coupled roll-yaw dynamics.

- d. Valve Driver Electronics. The electronics utilize the output of the electronic processor to command the reaction control thrusters to correct for errors in pitch and roll.
- e. Reaction Control Thrusters. Normally, discussion of the thrusters themselves is not a concern of the S&C subsystem. However, the pitch momentum bias configuration utilizes an offset of the roll thrusters to provide a proportional amount of torque about yaw. The roll thruster offset into yaw is designed to damp the orbit frequency mode associated with the vehicle dynamics. This offset along with the restraint provided by the momentum wheel provides control in yaw without a direct yaw sensor.

Block diagram of the Pitch Momentum Bias configuration is presented in Figure 3-12.

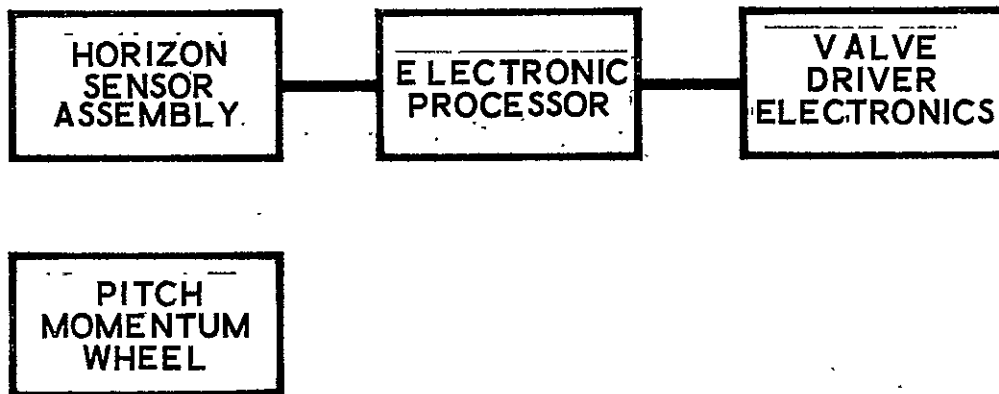


Figure 3-12. Mass Expulsion Configuration Block Diagram With Pitch Momentum Wheel

3.7.2 Design Equations

The pitch momentum bias configuration is an active, three-axis control system incorporating a momentum wheel, an attitude sensor, and mass expulsion jets. The momentum wheel is aligned normal to the orbit plane along the negative pitch axis. The restraint provided by the wheel in yaw along with offset roll-yaw thrusters provides control in yaw without a direct yaw sensor.

The pitch momentum bias configuration used in this study has a constant speed momentum wheel with pitch thrusters for control about the pitch axis. An alternate method which would be preferable for longer missions is to incorporate a variable speed reaction wheel to store the momentum created by disturbance torques and to use the pitch thrusters to unload the wheel. This alternate configuration is not covered in this model.

3.7.2.1 Horizon Scanner Selection from Beam Pointing Equation

The following equations were taken from Reference 3-6. The equation for beam pointing error is the RSS of the error due to antenna misalignment and the attitude reference error, e_{AR} . Solving for the attitude reference error gives:

$$e_{AR}^2 = e_{beam}^2 - e_a^2 \quad (3-93)$$

where

e_a = antenna misalignment error (deg, 3σ)

e_{beam} = antenna beam pointing error (deg, 3σ)

The beam pointing error (less the misalignment error) can be expressed as a function of the yaw error and pitch and roll sensor errors.

$$e_{AR}^2 = \varphi_s^2 + (E\psi)^2 + \theta_s^2 \quad (3-94)$$

where

φ_s and θ_s are the roll and pitch horizon scanner errors (deg, 3σ)

E = antenna elevation angle (rads)

ψ = yaw error

Assuming that the horizon scanner error is the same for both roll and pitch and setting the yaw error equal to the maximum allowable yaw error, one can solve for the maximum horizon scanner error as

$$e_{HS} = \left\{ \frac{e_{AR}^2 - (E\theta_{R_z})^2}{2} \right\}^{1/2} \quad (3-95)$$

where

$e_{HS} = \varphi_s = \theta_s =$ horizon scanner error (deg, 3σ)

$\theta_{R_z} =$ required pointing accuracy on the yaw axis (deg, 3σ)

The horizon scanner error is the RSS of the individual error contributors.

$$e = (e_1^2 \quad e_2^2 \quad e_3^2 \quad e_6^2)^{1/2} \quad (e_4^2 \quad e_5^2)^{1/2} \quad (3-96)$$

where

$e_1 =$ earth sensor noise (deg, 3σ)

$e_2 =$ radiance irregularity (deg, 3σ)

$e_3 =$ quantization error (deg, 3σ)

$e_4 =$ sun interference (deg, 3σ)

$e_5 =$ moon interference (deg, 3σ)

$e_6 =$ threshold aging (deg, 3σ)

} worst case deterministic errors (added directly with random errors)

The horizon sensor error, e , is computer for each scanner in the equipment data base and compared with the maximum allowable horizon sensor error,

e_{HS}^*

3.7.2.2 Momentum Wheel Selection

Estimates of wheel size depend on preliminary knowledge of the disturbance torque on the spacecraft. For synchronous orbits, the disturbance is due to solar pressure and the torque is dependent on the specific spacecraft configuration and size. For a wide class of missions, the peak solar torque is in the range of $10^{-6} - 10^{-5}$ kg-m (10^{-5} to 10^{-4} ft-lb), and typically, the cyclic torques dominate. Large unfurlable antennas, 6.1 m (20 ft) or more in diameter, and a particularly unbalanced configuration could lead to peak torques above 10^{-5} kg-m (10^{-4} ft-lb). The wheel size can be significantly reduced by designing a configuration which minimizes the

solar torque. In the case of systems using gyroscopic stiffness to limit yaw, a particular premium is placed on minimizing the yaw torque. The equation for sizing the wheel is:

$$h = 57.3 M_{D_z} / (\omega_o \psi) \quad (3-97)$$

where

h = required momentum (ft-lb-sec)

M_{D_z} = peak yaw disturbance torque (ft-lb)

ω_o = average orbital rate (rad/sec)

and ψ is the maximum yaw error which can be computed as follows:

$$\psi = \frac{1}{E} (e_{AR}^2 - 2e^2)^{1/2} \quad (3-98)$$

A momentum wheel is now selected from the equipment data base which has a momentum, \underline{h} , which is nearest to h and greater than h .

3.7.2.3 Thrusters Sizing

An upper bound on the minimum torque impulse bit is shown in the equation below (Ref. 3-7):

$$I_{\min} < 2 \text{ db } \underline{h} \sqrt{\frac{J_x}{J_z}} \quad (3-99)$$

$$I_{\min} = \frac{F \Delta t d_x}{x} \quad (3-100)$$

where

J_x = roll axis moment of inertia (slug-ft²)

J_z = yaw axis moment of inertia (slug-ft²)

db = roll deadband

I_{\min} = minimum torque impulse bit (ft-lb-sec)

F = thruster output (lb)

d_x = roll moment arm (ft)

Δt = minimum impulse time (sec)

The basic assumption in deriving Equation (3-99) was that the residual rates within the deadband were zero. The equation also applied for a zero

roll-yaw thruster offset angle [the actual angle will be approximately 0.21 rad (12 deg)]. Substituting Equation (3-100) into Equation (3-99) and solving for F yields:

$$F_{\max} = \frac{db \ h}{\Delta t \ d_x} \sqrt{\frac{J_x}{J_z}} \quad (3-101)$$

where the factor of 2 was removed to select a conservative value of thruster size. The thruster moment arms can be calculated roughly by:

$$\begin{aligned} d_x &= D \cos \alpha \\ d_z &= D \sin \alpha \\ d_y &= D/2 \end{aligned} \quad (3-102)$$

where

$$\begin{aligned} D &= \text{diameter of the vehicle in the roll-yaw plane (ft)} \\ \alpha &= \text{roll-yaw thruster offset angle (deg)} \end{aligned}$$

The minimum thruster size is the size which is required to control the vehicle while the main engine is operating.

$$\begin{aligned} F_1 &= (\Delta\theta \ d_T / 57.3 \ \Delta d) F_e / d_y \\ F_2 &= (\Delta\theta \ d_T / 57.3 \ \Delta d) F_e / d_z \end{aligned} \quad (3-103)$$

$$\begin{aligned} F_3 &= 2 \ \Delta d \ F_e / (57.3 \ d_x) \\ F_{\min} &= \max(F_1, F_2, F_3) \end{aligned} \quad (3-104)$$

where

$$\begin{aligned} \Delta\theta &= \text{main engine misalignment (deg)} \\ d_T &= \text{distance from c. g. to main engine (ft)} \\ \Delta d &= \text{lateral vehicle c. g. distance plus lateral thrust chamber c. g. distance from the reference axis (ft)} \\ F_e &= \text{main engine thrust (lb)} \end{aligned}$$

The actual thrust level should be between F_{\min} and F_{\max} . To select a level for computation of the impulse requirements, a value of $F_{\max}/2$ is chosen. If this value is smaller than F_{\min} , then the value of F_{\min} is selected.

3.7.2.4 Impulse Requirement Sizing

The impulse required for a pitch momentum bias configuration is calculated using the same equations as for the three-axis mass expulsion configuration except that rotational maneuvers are not included. It is unlikely that a pitch momentum bias vehicle would be maneuvered because of the wheel momentum.

a. Powered Flight Impulse (Translational)

$$\begin{aligned}
 I_{y_2} &= \left(\frac{\Delta\theta}{57.3} d_T + \Delta d \right) \frac{F_e t}{d_y} \\
 I_{z_2} &= \left(\frac{\Delta\theta}{57.3} d_T + \Delta d \right) \frac{F_e t}{d_z} \\
 I_{x_2} &= \frac{2}{57.3} d_T \Delta\theta \frac{F_e t}{d_x}
 \end{aligned} \tag{3-105}$$

b. Impulse due to Recovery from Initial Rates

$$\begin{aligned}
 I_{x_3} &= \eta J_x \dot{\theta}_0 / (57.3 d_x) \\
 I_{y_3} &= \eta J_y \dot{\theta}_0 / (57.3 d_y) \\
 I_{z_3} &= \eta J_z \dot{\theta}_0 / (57.3 d_z)
 \end{aligned} \tag{3-106}$$

c. Limit Cycle Impulse

$$\begin{aligned}
 I_{x_1} &= 37 \times 10^6 T d_x (F \Delta t)^2 / (J_x db_x) \\
 I_{y_2} &= 37 \times 10^6 T d_y (F \Delta t)^2 / (J_y db_y)
 \end{aligned} \tag{3-107}$$

d. Impulse Due to Overcoming External Torques

$$\begin{aligned}
 I_{x_4} &= M_{D_x} \tau_x / d_x \\
 I_{y_4} &= M_{D_y} \tau_y / d_y
 \end{aligned} \tag{3-108}$$

e. Total Impulse

$$I_x = \sum_{i=1}^4 I_{x_i}$$

$$I_y = \sum_{i=1}^4 I_{y_i} \quad (3-109)$$

$$I_z = I_{z_2} + I_{z_3}$$

$$I = I_x + I_y + I_z \quad (3-110)$$

3.7.3 Design Logic

The sequence followed in implementing the design algorithm is as follows:

- a. Input data.
- b. Select a deadband.
- c. Call up configuration special hardware from the data base which is not differentiable.
 1. Electronic Error Processor
 2. Valve Driver Electronics
- d. Select a horizon scanner based on the maximum antenna beam pointing error, antenna misalignments, antenna elevation angle, and horizon scanner error.
- e. Size the pitch momentum wheel based on the maximum yaw error which will occur with the selected horizon scanner, the orbital rate of the vehicle, and the maximum value of the cyclic disturbance torque relative to the vehicle.
- f. Size the thruster force based on a maximum thrust level to ensure that the residual rates within the deadband are zero and a minimum thrust level to ensure that the vehicle will remain in control while the main engine is firing.
- g. Compute the impulse requirements for the above configuration.

4. AUXILIARY PROPULSION SUBSYSTEM

4.1 GENERAL

4.1.1 Subsystem Functional Description

The two functions of the Auxiliary Propulsion Subsystem are to provide attitude control forces and stationkeeping or maneuvering forces. The attitude control function includes pitch and yaw torques and roll moments. The forces involved must be provided with a response rate compatible with the dynamic characteristics of the vehicle and the S&C Subsystem.

Stationkeeping or maneuvering velocity increments are provided by translational forces along a vehicle axis and through the center of gravity. Sufficient thrust and propellant must be provided to account for any requirements of orbit circularization, apogee/perigee correction, and plane change ΔV .

4.1.2 Component Functional Descriptions

4.1.2.1 Propellant Group

The propellant group stores the required impulse propellant and provides control and ΔV forces on command. The specific components, their functions, and operational considerations (where appropriate) are as follows:

- a. Thruster(s)
 1. Provide steady force: duration, thrust level.
 2. Provide pulse mode force: number of starts, minimum impulse bit.
- b. Tank(s)
 1. Contain propellant(s) sufficient to meet mission requirements plus a safety margin.
 2. Provide propellant/pressurant separation.
- c. Filter(s). Protect small flow passages and moving parts from contamination.

- d. Isolation Valve(s)
 - 1. Close off propulsion system from tank to minimize risk of leakage during boost and inoperative periods.
 - 2. Provide redundant on-off function in the event of system leakage.
- e. Fill and Drain Valve. Provide for loading system with propellant and draining after ground test or launch abort.
- f. Thruster and System Heaters/Insulation
 - 1. Maintain the components within feasible qualified thermal limits.
 - 2. Prevent propellant freezing (heaters and insulation) and vaporization (insulation only).
- g. Plumbing and Connectors
 - 1. Provide propellant flow between all components.
 - 2. Provide for component attachment and system integrity checkout.

4.1.2.2 Pneumatic Group

The pneumatic group stores sufficient pressurant gas (or gaseous propellant) to maintain the required propellant flow rate throughout the mission life. A constant regulated pressure is provided, if necessary. The specific component functions are:

- a. Regulator. Establish a constant system pressure within allowable tolerances.
- b. Tank. Contain pressurant to satisfy mission requirements plus a margin.
- c. Fill and Vent Valve. Provide for pressurizing/unloading pressurant.
- d. Filter. Protect the regulator and valves from contamination due to loading.
- e. Relief Valve. Automatically vent excessive system pressure in the event of a regulator failure or system malfunction.
- f. Isolation Valve
 - 1. Provide means of switching from the primary pressure regulator to a backup regulator.
 - 2. Provide on-off pressurization in the event of a system leak.

g. Plumbing and Connectors

1. Provide pressurant flow between all components.
2. Provide for component attachment and system integrity checkout.

4.1.2.3 Electrical Instrumentation Group

The electrical and instrumentation group provides and routes valve control signals, heater power, and telemetered data. The general components are:

- a. Valve Power Harness. Provide functioning power to thruster valves and isolation valves.
- b. Heater Harness. Provide power to various system resistance heaters.
- c. Propellant Group Instrumentation
 1. Monitor system pressure and temperature to establish residual for blowdown type operation and to indicate leakage.
 2. Monitor all valve signals for indication of system failures.
- d. Pneumatic Group Instrumentation. Monitor system pressure and temperature to establish residual and indicate failures which require switching to a redundant system.

4.1.3 Subsystem Configurations

Figure 4-1 diagrams a typical cold gas propulsion subsystem configuration. Both attitude control and translational thrusters are included. (P) and (T) represent pressure and temperature transducers. Isolation valves are included to allow for close-off of any propellant manifold which indicates leakage during inoperative periods. In general, no redundancy is included. Increased redundancy or fail-operational requirements may be fulfilled by adding more thrusters in separate manifolds by using thruster valves in a dual-sealing configuration and by adding other redundant backup components.

A typical hydrazine monopropellant subsystem configuration is diagrammed in Figure 4-2. A pressurization system is included; however, the subsystem can also be operated in a blowdown mode by locking up sufficient pressurant in the propellant tanks. A rubber or metal diaphragm

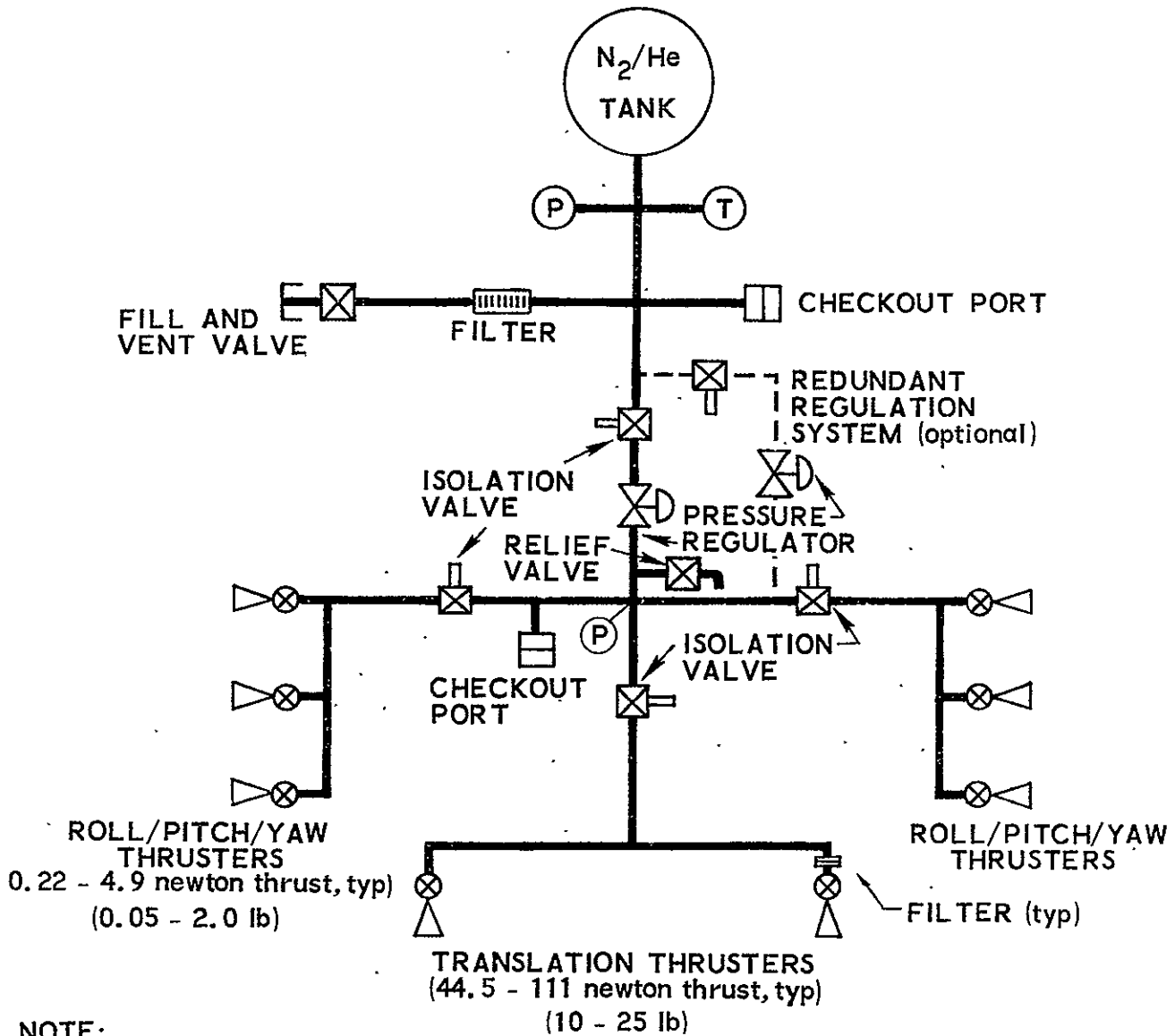


Figure 4-1. Cold Gas Configuration

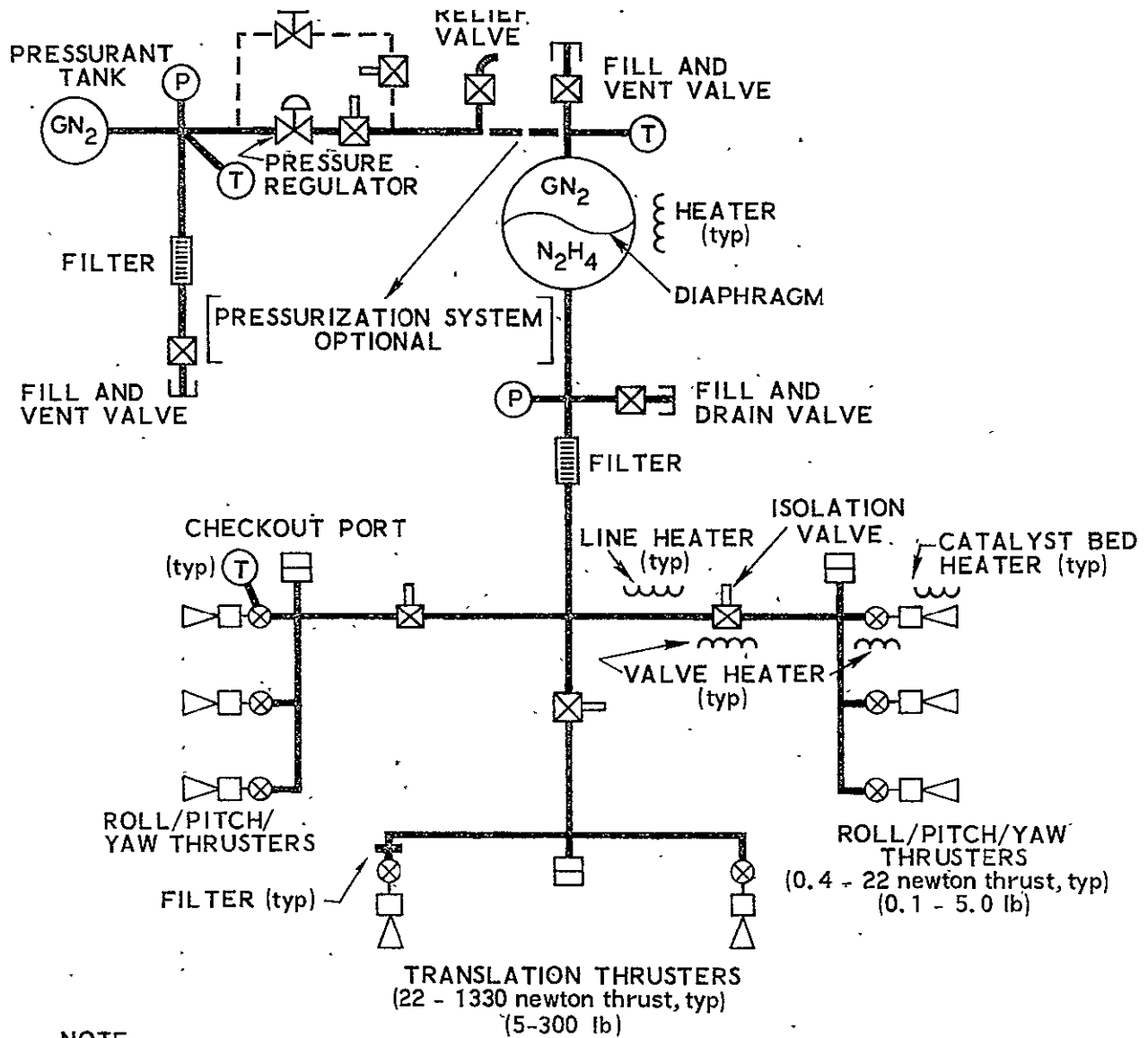


Figure 4-2. Hydrazine Monopropellant Configuration

separates the pressurant from the propellant. The use of isolation valves and optional redundant components is indicated. Heaters and/or insulation are generally necessary.

A typical bipropellant subsystem configuration is diagrammed in Figure 4-3. Isolation and redundancy concepts are the same as above. Insulation and elastomeric tank bladders are indicated.

4.2 INPUT DATA

Input data for the subsystem model is required from the user and several subsystems. As shown in Table 4-1, the mandatory inputs are function, mission life, thrust, total impulse, and pulse life requirements.

Table 4-1. Input Data Requirement:

User Input

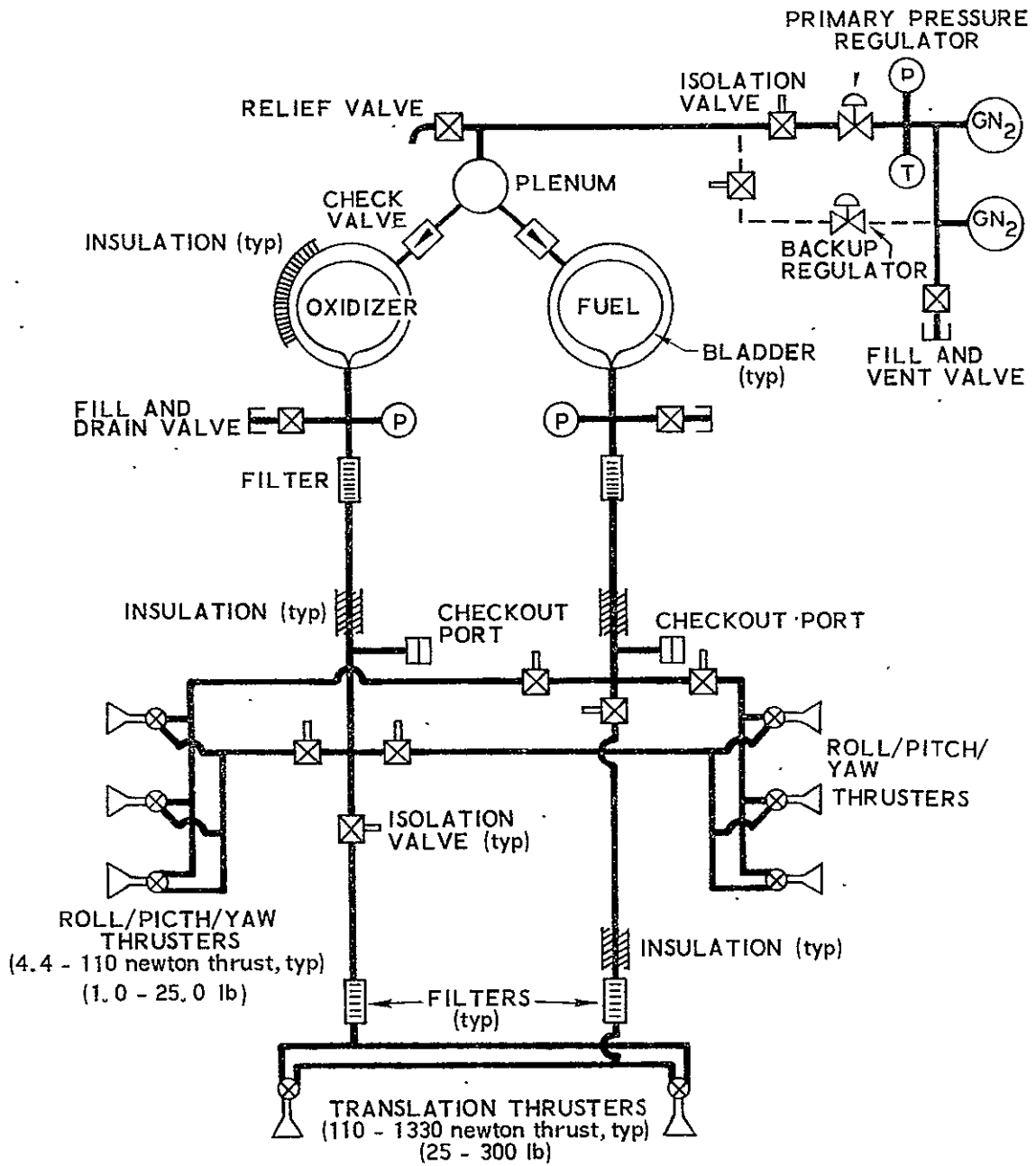
1. Function (attitude control and/or ΔV)
2. Mission life

Stabilization and Control Subsystem

1. Attitude control thrust level
2. Attitude control pulse life
3. Total impulse

Reliability

1. Degree of redundancy



NOTE:

Increased redundancy or fail-operational requirements met by using more thrusters in each of the two attitude control clusters, and by using thruster valves with dual-sealing configuration.

Figure 4-3. Bipropellant Configuration

4.3 SUBSYSTEM DESIGN ALGORITHM

There are three basic steps in the Auxiliary Propulsion Subsystem Design Algorithm. The first step is one of ascertaining whether a specific configuration is compatible with the input requirements. The second step is to calculate or otherwise determine the important parameters which describe the subsystem design. Using this information, the subsystem components may then be selected from the data base. These three steps are described in detail in the following sections.

4.3.1 Configuration Compatibility

The design algorithm will design all three subsystem configurations (cold gas, monopropellant, and bipropellant) and will provide the reliability, weight, cost, etc. of each to allow selection by the user through the process of elimination. If the input data so indicate, one or more of the configurations may be eliminated automatically by the design algorithm, as indicated in Table 4-2. This configuration test is intended to preclude designing those configurations which are clearly undesirable, according to engineering experience and analysis.

4.3.2 Design Equations

4.3.2.1 Liquid Propellant

The propellant group characteristics are defined first; then the pressurant group is sized to complement the propellant group size and pressure. Propellant and pressurant characteristics required for component sizing are presented in Table 4-3.

Propellant weight (W_p) is established from total impulse (I_t) requirements, with a 10-percent factor added for softness in mission requirements, subsystem analyses, and component operation variances.

$$W_p = 1.1 I_t / I_s \quad (4-1)$$

Table 4-2. Auxiliary Propulsion Configuration Selection

Input Requirements	Cold Gas	Monopropellant	Bipropellant
Thrust < 224 newtons (< 50 lb) 224-4450 newtons (50-1000 lb) > 4450 newtons (> 1000 lb)	Yes No No	Yes Yes No	Yes Yes Yes
Total Impulse < 4.4×10^4 newton-sec ($< 10^4$ lb-sec) $4.4 \times 10^4 - 2.2 \times 10^5$ newton-sec ($10^4 - 5 \times 10^4$ lb-sec) $2.2 \times 10^5 - 8.9 \times 10^5$ newton-sec ($5 \times 10^4 - 2 \times 10^5$ lb-sec) > 8.9×10^5 newton-sec (2×10^5 lb-sec)	Yes Yes No No	No Yes Yes No	No No Yes Yes

Legend:

- Yes - Acceptable
- No - Unacceptable

Table 4-3. Propellant and Pressurant Characteristics

Propellant/Pressurant	Density, ρ g/cm ³	Average Specific Impulse, I_s (sec)	Molecular Weight, M_{pr} (kg/mole)
Nitrogen	---	65	12.7
Hydrazine	1.0	200	---
Monomethyl Hydrazine	0.89	260 (with NTO)	---

The relative fuel (W_f) and oxidizer (W_o) weights are derived from the mixture ratio (MR).

$$W_f = \frac{W_p}{1 + MR} \quad (4-2)$$

$$W_o = W_f \times MR$$

Propellant volumes (V) are based on propellant density (ρ) at 294°K (70°F).

$$V_f = W_f / \rho_f \quad (4-3)$$

$$V_o = W_o / \rho_o$$

Propellant tank volumes for pressure-regulated configurations are 10 percent greater than the propellant volumes to allow for a 5 percent minimum ullage and, again, to account for possible requirement changes. In the case of a monopropellant blowdown configuration, the ullage requirement is 100 percent of the fuel volume.

$$V_{ft} = 1.1 V_f$$

$$V_{ot} = 1.1 V_o \quad (4-4)$$

$$V_{ft} \text{ (blowdown)} = 2V_f$$

Nominal propellant tank pressures at 294°K are based on the rated inlet pressure (P_{ti}) of the selected thruster plus the calculated pressure drop of the propellant group valves (ΔP_{iso}) plumbing (ΔP_l) and filters (ΔP_{filt}). For the regulated configuration:

$$P_{ft} \text{ and } P_{ot} = P_{ti} + \Delta P_{iso} + 2\Delta P_{filt} + \Delta P_1 \quad (4-5)$$

where

$$\Delta P_{iso} = \frac{1.29 \times 10^{-3}}{\rho} \left(\frac{\dot{w}}{C_d A} \right)^2 \quad (4-6)$$

where $C_d A$ = effective flow area

and

$$\Delta P_{filt} = R \dot{w}^2 \quad (4-7)$$

R = Flow resistance

and, since the plumbing length and configuration cannot be defined, the line drop (ΔP_1) is arbitrarily:

$$\Delta P_1 = \Delta P_{iso} \quad (4-8)$$

Equations (4-4) and (4-5) are used for propellant tank selection.

For the flowdown case, the initial pressure is the same as above and the final fuel tank pressure is:

$$\text{final } P_{ft} = P_{ft}/2 \quad (4-9)$$

The answer from Equation (4-9) is substituted back into Equation (4-5) to establish the minimum flow rate. Minimum blowdown thrust (F_{min}), to be compared with a minimum control requirement input, is then

$$F_{min} = \dot{w}_{min} I_s \quad (4-10)$$

The flow rate (\dot{w}) used in Equations (4-6) and (4-7) and for sizing valves and regulators is the maximum to be expected when a number (N) of thrusters are operating simultaneously for pitch or yaw, roll, and translation functions. The maximum propellant flow rates are:

$$\begin{aligned} \dot{w}_f &= NF/I_s (1 + MR) \\ \dot{w}_o &= \dot{w}_f \times MR \end{aligned} \quad (4-11)$$

The above flow rate, plus pressures from Equation (4-5) and the pressure drops from Equations (4-6) and (4-7), are used to select the propellant valves, based on a maximum allowable ΔP of 3.4×10^5 newton/m² (50 psi).

The corresponding pressurant flow rate (\dot{w}_{pr}) at 294°K is:

$$\begin{aligned} \dot{w}_{pr} &= (\dot{V}_f + \dot{V}_o) \rho_{pr} \\ &= (\dot{w}_f / \rho_f + \dot{w}_o / \rho_o) P_{reg} \times M_{pr} \times 1.02 \times 10^{-7} \end{aligned} \quad (4-12)$$

where the regulator set pressure (P_{reg}) is 5 percent greater than the highest tank pressure:

$$P_{reg} = 1.05 P_{ft} \text{ or } 1.05 P_{ot} \quad (4-13)$$

The flow rate from Equation (4-12) and the pressure from Equation (4-13) are used to select pneumatic valves and regulators. The maximum allowable regulator pressure drop (ΔP_{reg}) is 1.38×10^5 newton/m² (200 psi) at 294°K, calculated by:

$$\Delta P_{reg} = [1.27 \times 10^4 / P_{reg} M_{pr}] \left(\frac{\dot{w}}{C_d A} \right)^2 \quad (4-14)$$

The regulated pressurant volume (V_{pr}) at 294°K is based on the propellant group size and pressure after all propellant is expelled, plus an added factor for the pressurant tank which remains at approximately twice the propellant tank pressures.

$$V_{pr} = V_{ft} + V_{ot} + 2V_{prt} \quad (4-15)$$

For the blowdown configuration:

$$V_{pr}(\text{blowdown}) = V_{ft} \quad (4-16)$$

The pressurant tank volume (V_{prt}) for tank selection is based on an assumed tank capability of 2.07×10^7 newton/m² (3000 psia) and is substituted back into the above pressurant volume equation.

$$V_{prt} = V_{pr} \times P_{ft}/3000 \quad (4-17)$$

Finally, the pressurant weight (W_{pr}) at 294°K and 2.07×10^6 newton/m² (300 psia) is:

$$\begin{aligned} W_{pr} &= \rho_{pr} V_{prt} \\ &= 3.05 \times 10^{-4} M_{pr} V_{prt} \end{aligned} \quad (4-18)$$

The subsystem weight is compiled from the component data base. Since line lengths are unknown, the propellant and pneumatic plumbing weights are estimated as equal to 10 percent of the associated tank weights. The number of filters for each propellant is equal to the number of thrusters plus one for cold gas and monopropellant systems, and the number of thruster clusters plus one for the bipropellant system. The electrical harness weight is included in the total vehicle electrical harness weight estimate in the Vehicle Sizing Model.

4.3.2.2 Gaseous Propellant

The total gas weight (W_{pr}) requirement is:

$$W_{pr} = 1.1 I_t/I_s \quad (4-19)$$

The required tank volume (V_{prt}) at 294°K and 2.07×10^7 newton/m² including an allowance for residual gas at 6.9×10^5 newton/m² (100 psia), is:

$$\begin{aligned} V_{prt} &= 1.03 W_{pr}/\rho_{pr} \\ &= 3.4 \times 10^3 W_{pr}/M_{pr} \end{aligned} \quad (4-20)$$

The regulated pressure (P_{reg}) is based on thruster inlet pressure (P_{ti}) and system pressure drop:

$$P_{reg} = P_{ti} + \Delta P_{iso} + \Delta P_{filt} + \Delta P_l \quad (4-21)$$

where

$$\rho_{pr} = 1.02 \times 10^{-7} P_{ti} M_{pr} \quad (4-22)$$

Finally, the maximum flow rate (\dot{w}_{pr}) at 294°K is:

$$\dot{w}_{pr} = NF/I_s \quad (4-23)$$

The pneumatic tank is selected from Equation (4-20) and the pressure regulator and valves from Equations (4-14), (4-21), and (4-23), based on a maximum allowable pressure drop of 1.38×10^6 newton/m² (200 psi).

4.3.3 Design Logic

The sequence followed in implementing the Design Algorithm is as follows:

4.3.3.1 Liquid Propellant

- a. Select translational thrusters.
 1. Thrust level
 2. Pulse life
- b. Select attitude control thrusters.
 1. Thrust level
 2. Pulse life
- c. Compute propellant flow rate.
- d. Select filters.
 1. Flow resistance
 2. Maximum pressure rating
- e. Select isolation valves.
 1. Flow area
 2. Maximum pressure rating
- f. Estimate line pressure drop.
- g. Compute required propellant tank pressures.
- h. Compute propellant weight.
- i. Compute required propellant tank volumes.
- j. Select propellant tanks and number required.
 1. Volume
 2. Pressure

- k. Compute pressurant flow rate.
- l. Select isolation valve for each pressure regulator.
 - 1. Flow area
 - 2. Maximum pressure rating
- m. Select pressure regulator.
 - 1. Flow area
 - 2. Pressure set point
 - 3. Maximum pressure rating
- n. Compute pressurant volume.
- o. Compute pressurant weight.
- p. Select pressurant tank.
 - 1. Volume
 - 2. Maximum pressure rating
- q. Call up configuration special hardware from the data base which is not differentiable.
 - 1. Fill and drain valves

4.3.3.2 Gaseous Propellant

- a. Select translational thrusters.
 - 1. Thrust level
 - 2. Pulse life
- b. Select attitude control thrusters.
 - 1. Thrust level
 - 2. Pulse life
- c. Compute maximum flow rate.
- d. Select filters.
 - 1. Flow resistance
 - 2. Maximum pressure rating
- e. Select isolation valves.
 - 1. Flow area
 - 2. Maximum pressure rating
- f. Estimate line pressure drop.
- g. Compute required regulated pressure.

- h. Select pressure regulator.
 - 1. Flow area
 - 2. Pressure set point
 - 3. Maximum pressure rating
- i. Compute gas weight.
- j. Compute required tank volume.
- k. Select tank and number required.
 - 1. Volume
 - 2. Pressure
- l. Call up configuration special hardware from the data base which is not differentiable.
 - 1. Fill and drain valve

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5. DATA PROCESSING SUBSYSTEM

5.1 GENERAL

5.1.1 Subsystem Functional Description

The Data Processing Subsystem performs any or all of the following functions:

- a. Command Decoding - Command decoding generally takes the form of receiving data on the uplink and developing unique commands to control the mode of the other subsystems.
- b. Telemetry Data Processing - Internal status monitors of the spacecraft state-of-health provide signals for multiplexing and encoding. The output goes to the baseband assembly unit in the Communication Subsystem for transmission on the downlink.
- c. Attitude Processing - Attitude processing generally takes the form of processing sensor and rate data (pulse width, timing, coincidence, etc.) and developing error signals. This may require developing state vector or ephemeris from sun, earth, and star sensor data.
- d. Mission Equipment Data Processing - Mission equipment processing is unique to each payload but generally takes the form of data compression.

5.1.2 Subsystem Configurations

Two configuration models have been developed for the Data Processing Subsystem:

- a. Special Purpose Processors
- b. General Purpose Processors

The Special Purpose Processor configuration, depicted in Figure 5-1, has been subdivided into two categories:

- a. One (1) Digital Telemetry Unit
- b. Two (2) Digital Telemetry Units

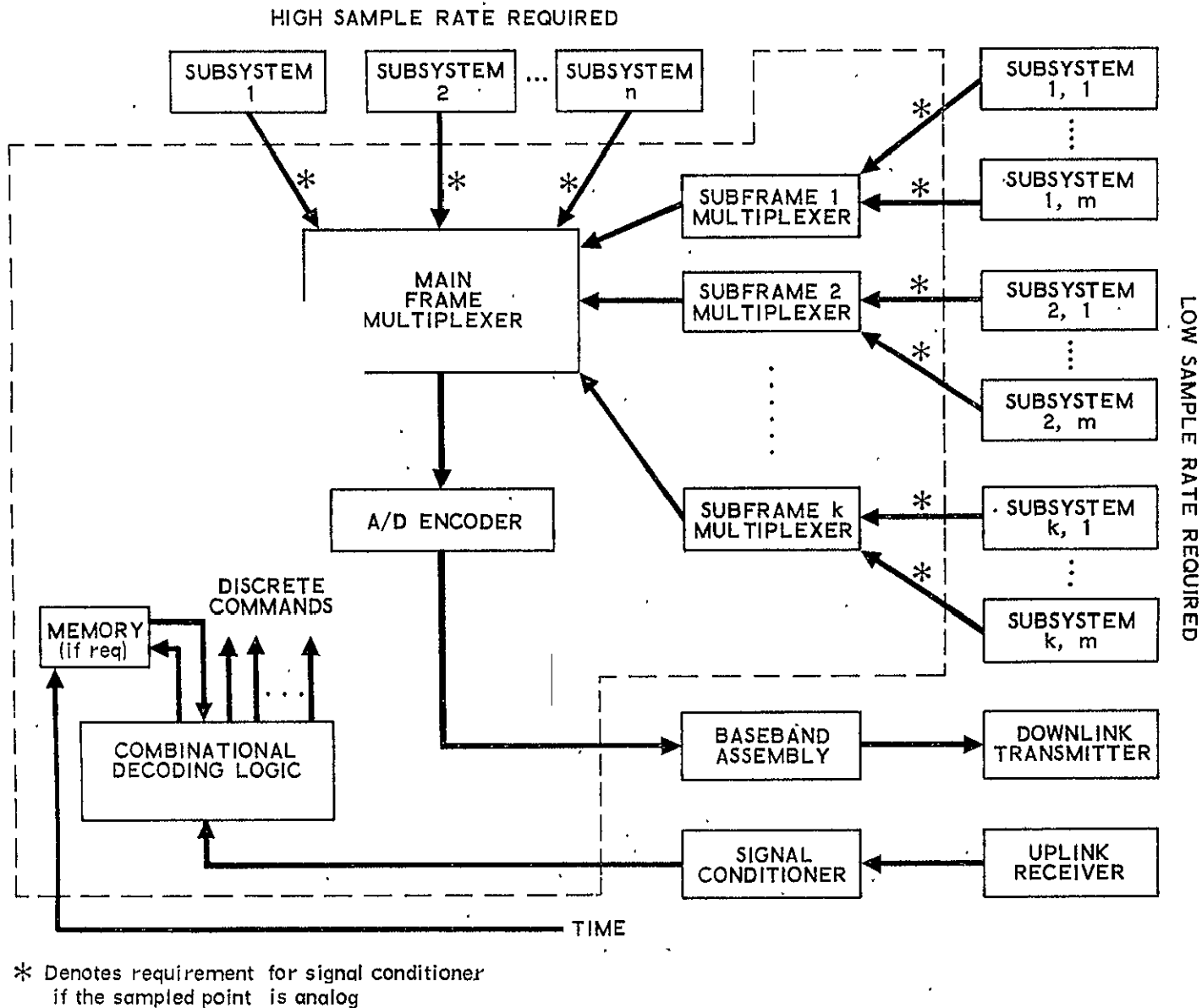


Figure 5-1. Special Purpose Processor Configuration Block Diagram

The purpose of the two categories is to maintain consistency with the Communication Subsystem which may have one or two downlinks which the Digital Telemetry Units (DTUs) are expected to service.

5.1.3 Configuration Compatibility

The Data Processing Subsystem configurations are generally consistent with the Communication Subsystem configurations. Table 5-1 summarizes the compatibility requirements between the two subsystems. It should be clear that the mission data, if provided, is combined with the "housekeeping" data when there is only one Communications downlink. When there are two downlinks, mission data is handled on the separate downlink. If mission data is not passed to the spacecraft, the separate downlink configurations in the Communication Subsystem are NOT designed.

5.1.4 Equipment Types

The list of equipment types from which the subsystem components will be selected is provided below:

- a. Digital Telemetry Unit
 1. Multiplexer
 2. Encoder
 3. Control
- b. Command Decoding and Distribution Unit
 1. Decoder
 2. Memory (if required)
 3. Clock
 4. Control
 5. Distribution logic
- c. General Purpose Data Processor

The elements of each component are stated in order to avoid misunderstanding as to whether any subcomponents have been left out of the model.

Table 5-1. Data Processing Configuration Compatibility

Communication Configuration	General Purpose Processor	Special Purpose Processors	
		1 DTU	2 DTUs
Uplink, plus downlink	Yes (1 Data Rate Computed)	Yes	No
Unified link, common antenna	Yes (1 Data Rate Computed)	Yes	No
Unified link, separate antennas	Yes (1 Data Rate Computed)	Yes	No
Unified link, common antenna plus downlink	Yes (2 Data Rates Computed)	No	Yes
Unified link, separate antennas plus downlink	Yes (2 Data Rates Computed)	No	Yes

Legend:

- Yes - Compatible
- No - Incompatible

5.2 INPUT DATA

The information required from the user to design the Data Processing Subsystem is identified in Table 5-2. The table indicates what data are used, whether the data are required or optional, and, if optional, the default value.

In addition to the user supplied data, the following information is required from the equipment data base for every component selected as part of the spacecraft design:

- a. Number of power switching commands
- b. Number of time tagged commands
- c. Number of other commands
- d. High rate telemetry
 1. Number of analog points
 2. Number of digital points
 3. Sample rate (bps)
 4. Word length (bits)
- e. Low rate telemetry
 1. Number of analog points
 2. Number of digital points
 3. Sample rate (bps)
 4. Word length (bits)

Table 5-3 summarizes the additional input data required by the Data Processing Subsystem model.

Table 5-2. Input Data Supplied by User

Symbol	Name	Required (R) or Optional (O)	Default Value
BTRMX	Maximum bit rate (bps)	O	1.024×10^6
SLSF	Special command synchronization flag (0 means no synchronization required, 1 means synchronization required)	O	0
TPRFL	Telemetry processing flag (0 means telemetry processed separately from general purpose processor, 1 means otherwise)	O	0
OPSMS	Mission data processing rate (ops/sec)	O	0
ARRAYN	Mission data for up to three equipments	O	0
NMSEQ	Number of mission equipment	O	0

Table 5-3. Input Data Requirements

User Input

(See Table 5-2)

Data Base

(Summarized in Text)

Stabilization and Control Subsystem

1. Number of Attitude Sensors
2. Data Operations Required Per Sensor
3. Data Processing Rate

Communication Subsystem

1. Configuration Type
2. Command Rate
3. Command Operations Required
4. Telemetry Operations Required

Reliability

1. Degree of Redundancy

5.3 SPECIAL PURPOSE PROCESSOR CONFIGURATION

5.3.1 Functional Description

The Special Purpose Processor Configuration components perform the following functions:

- a. Telemetry Data Processing - A typical Telemetry Data Processor (TDP) is shown in block diagram form in Figure 5-2. The TDP is divided into three major categories: multiplexers, encoder, and control. The multiplexers (mux) consist of one main frame mux having n channels being scanned at some rate R (usually specified in bits per second, which if divided by bits per channel, yield the channel sample rate) and some number of m channel subframe multiplexers being scanned at $1/n$ times the mainframe channel rate. All the subframes need not be m channels but, for the sake of simplicity, they generally are. The encoder consists of a switch to determine whether a given channel is presenting analog or digital information, an analog-to-digital converter for those channels whose input is analog, a one-word holding register, and a parallel-to-serial converter to clock out the digital words at the serial telemetry rate. The output goes to the Baseband Assembly Unit in the Communication Subsystem for transmission on the downlink.

The control logic generates all the timing signals, mux rates, sync words, etc., for handling the overhead and making everything operate synchronously.
- b. Command Decoding and Distribution - While it would appear to be as feasible to standardize the uplink as easily as the downlink, this has only been partially implemented. The uplink of SGLS has indeed been standardized as well as the downlink, but the standardization reaches only to the communications-to-data processing interface. This leaves the command decoding and distribution to be uniquely specified for each vehicle. The command decoding function often is included in the Electrical Integration Assembly of a spacecraft since this is a logical point from which command can be distributed.

Figure 5-3 is a block diagram of a typical command decode and distribution system. From this block diagram, it can easily be seen that the only parameters affecting the selection of the command unit is the number of commands and the synchronizing strategy. The uplink rate is not a determining factor since

5-9

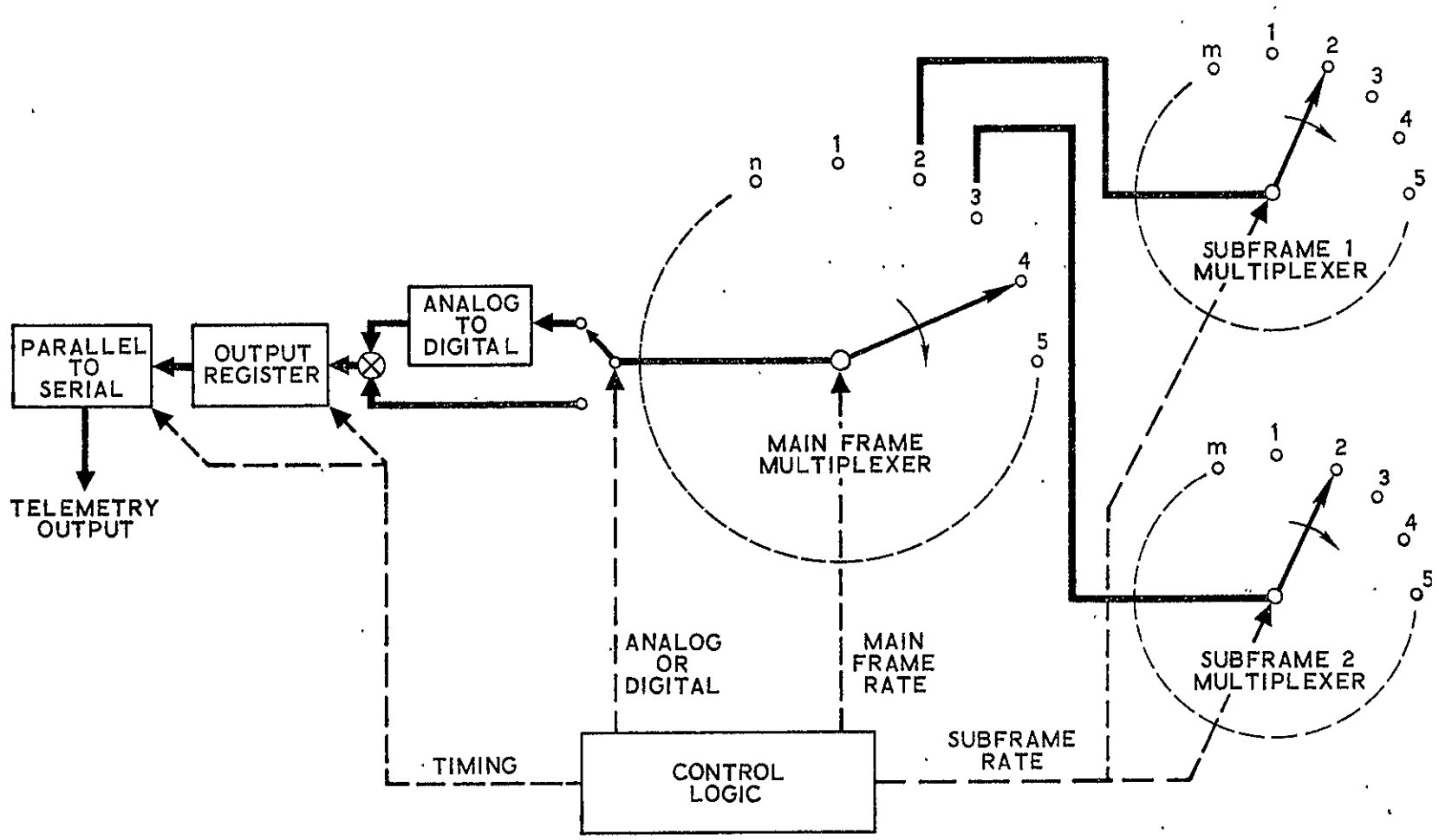


Figure 5-2. Telemetry Data Processor Block Diagram

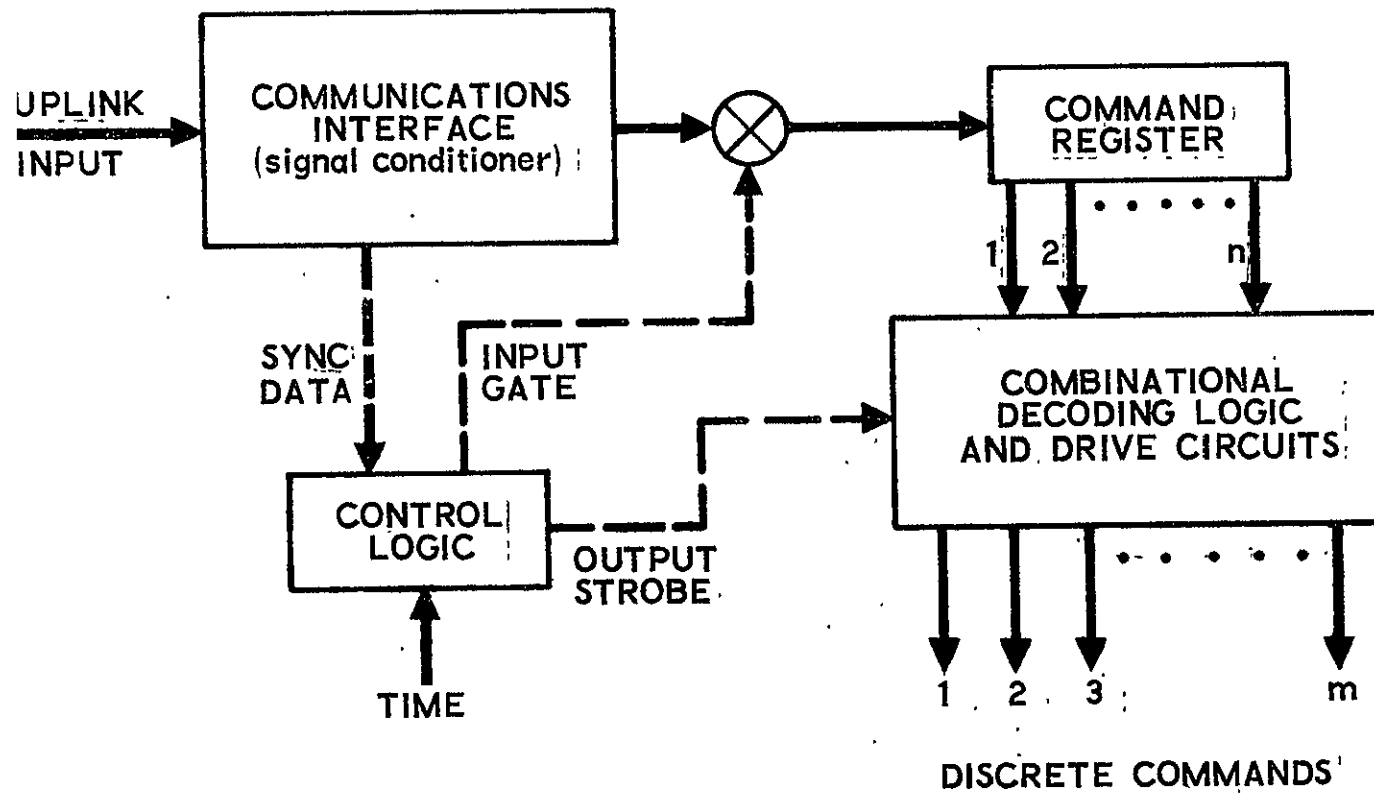


Figure 5-3. Command Decoding and Distribution Unit Block Diagram

uplink rates are orders of magnitude less than the speed capabilities of this type of logic. Synchronizing strategy is the method by which the control logic recognizes the presence of a command on the link, where it starts and where it stops. This allows the control logic to open and close the gate at the proper times into the command register and strobe the decoding logic. A further complication arises if commands can be time tagged (i. e. , sent with the time they are to be executed). This forces the decoding logic to include memory in which these commands are stored for later execution and requires time to be sent to the control logic. (Note: LST is an example of a time-tagged command system.) If commands are time tagged, the maximum number of commands which can be stored must be specified.

The only other requirement which impacts the command system is the total number of commands used. This requirement specifies the number of bits required in the command word and the number (or width) of decoding gates required. The number of bits (n) in the command word is related to the number of commands (m) by the following relationship:

$$2^n - 1 \geq m$$

The command word of all zeros should never be used due to its failure state.

- ∴ Mission Data Processing - Mission data processing (MDP) is the exception rather than the rule on most satellites and when it is included it is highly specialized and designed expressly to execute a specific algorithm. An example from among the listed example satellites is DSP which preprocesses and compresses the data from its primary mission sensor. The raw data rate is over 36 Mbps and this is reduced to less than 0.5 Mbps in the Data Processing Electronics Subsystem (DPES) by a combination of peak detection and thresholding. It must be pointed out, however, that the DSP DPES could not even remotely be considered for other satellites, even one whose mission was similar, because of the other highly specialized tasks performed by the DPES.

5.3.2 Design Equations

The parameters which determine the TDP requirements are seen to be quite simple, namely:

- a. Maximum scan rate
- b. Subscan rate
- c. Number of channels to be scanned at the maximum rate
- d. Number of channels to be scanned at the lower rate (hence the number of subframes)
- e. Word length

The first requirement may be driven by the Communication Subsystem. The bandwidth allocation may limit the maximum scan rate irrespective of the sampling requirements but these are tradeoffs which must be made in the design of the vehicle.

There are two approaches to the implementation of the hardware after the requirements are determined. The first approach is to standardize telemetry processors to one package versatile enough to meet all requirements of at least a large majority of space vehicles. This approach is taken by the Air Force Space Ground Link System (SGLS). The Digital Telemetry Unit (DTU) for SGLS can be delivered with word lengths ranging from 4 to 8 bits, bit rate from 7.8125 bps to 1024 kbps, mainframe size from 32 channels to 256 channels, and any number of subframes from 32 to 128 words long. The ranges of parameters are not, of course, continuously variable but are given in discrete steps of multiples of two times the base.

The second approach is to design special units, where necessary, for Telemetry Data Processing and on subsequent spacecraft try to choose a TDP from a previous design which will meet the new requirements. If an existing design cannot be adapted, however, a new unit must be designed. The disadvantages of the first approach is that, in its attempt to meet all (or most) requirements, it cannot be optimized for any and it must also be used

as a part of a standardized Communication Subsystem which may also place severe constraints on the vehicle design. The second approach has the disadvantage of often requiring new design when no previous design satisfies the current requirements.

There will always be cases which do not lend themselves easily to standardization, such as conformal coders and incremental (delta modulation) encoders. These cases would, however, require special design efforts under any circumstances; for purposes of the model it would appear most advantageous to elect the former option of having a standardized TDP from which the most optimum equipment for a given space vehicle can be selected from an inventory of standard items.

One assumption simplifies the approach to a large degree. The assumption is that the Stabilization and Control Subsystem will process sensor data in a special purpose device totally contained within the sub-System; hence, no S&C is considered.

Telemetry processing (either special or general purpose) is characterized by having various inputs from each of the other subsystems and one output to the Communications subsystem. The inputs represent points within each subsystem which are sampled, converted to digital form (if analog), and presented to the modulator of the Communication Subsystem for telemetering on the downlink. From this it is easily seen there are only three parts to the telemetry processor:

- a. Interface Signal Conditioning for Analog Data Points - This equipment is required to make voltage signals which may have widely differing ranges compatible with the input to the Analog-to-Digital converter (generally 0-5 volts). For the purpose of this model, it will be assumed that each of these interface conditioners costs the same as the others and that no conditioning is required for digital data points.
- b. Multiplexer (mux) which Samples Each Point in a Preprogrammed Manner - Actually this is made up of two separate multiplexers, the main mux and the sub-mux. The main mux samples at the highest rate necessary and the sub-mux (there may be one or more) samples at a fraction of the rate of the main mux and for those points not requiring high sample rates.

- c. Analog-to-Digital Converter - Analog-to-Digital Converter (ADC) converts the analog signals to digital form so that they can be transmitted on the down telemetry link. The ADC accepts analog signals from the mux and outputs the digital data.

The parameters which impact telemetry processing are:

- a. Word Length - The word length (mainly associated with the ADC) is predicated on the accuracy and range requirements of the sampled point. This parameter is supplied by the subsystem and is given either directly (in bits) or in granularity (e. g., 1 part in 256 requires 8 bits).
- b. Total Number of Points to be Sampled along with Each Point's Sample Rate Requirement - Each subsystem will supply a list containing an enumeration of points and the rate at which it must be sampled. These generally will be divided into two lists, one for high sample rates and the other for low sample rates. The former is associated with the main mux and the latter with the sub-mux. Along with each point must also be an indication of whether this point is analog (requiring interface signal conditioning) or digital. There is, then, a $4 \times n$ ($n = \text{total points}$) array containing:
 1. The name (number) of the point
 2. The sample rate (samples per second)
 3. The word length required (bits)
 4. Analog or digital

Command processing is considerably less demanding having basically only two requirements:

- a. Command word length is derived from the total number of commands in the system.
- b. If time tagged commands are required, a memory must be provided to hold the commands for later processing. The number of words in the memory is set by the maximum number of commands which must be stored at any one time for later processing.

5.3.3 Design Logic

The sequence followed in implementing the Design Algorithm is as follows:

- a. If the Communication Subsystem does not have a separate downlink for the mission equipment data, combine the mission equipment CDPI requirements with the data for the equipment selected from the data base.
- b. Order high rate telemetry points by sample rate required for each piece of equipment.
- c. If any telemetry points are analog, an analog-to-digital converter is required.
- d. Set the maximum bit rate.
- e. Determine the median data rate in the high rate telemetry table.
- f. If the highest rate is greater than twice the median, double the allocated telemetry points and halve the highest rate.
- g. Reorder high rate telemetry table, determine new median, and repeat test.
- h. If the highest rate is less than twice the median, set the sample rate equal to the highest rate.
- i. Determine the median word length required in the high rate table.
- j. If the maximum word length required is greater than twice the median, double the number of allocated telemetry points and halve the word length.
- k. Reorder high rate telemetry table by word length, determine new median, and repeat test.
- l. If the maximum word length required is less than twice the median, count the total number of words required and multiply by 1.20.
- m. If the number of words is greater than 256, abort the design attempt.
- n. Set the main frame length to the value of 2^n next larger than the number of words ($5 \leq n \leq 8$).
- o. Set word length to maximum required length.
- p. Take the product of number of words, word length, and sample rate to yield bit rate.

- q. Set bit rate to next highest multiple of 7.8125×2^n ($0 \leq n \leq 17$).
- r. If the bit rate is greater than the maximum bit rate, abort the design attempt.
- s. Order low rate telemetry points by sample rate required for each piece of equipment selected from the data base.
- t. Determine the median data rate in the low rate telemetry table.
- u. If the highest rate is greater than twice the median, double the allocated telemetry points and halve the highest rate.
- v. Reorder low rate telemetry table, determine new median, and repeat test.
- w. If the highest rate is less than twice the median, set the subframe rate equal to the highest rate.
- x. Set the subframe length equal to the mainframe rate divided by the subframe rate.
- y. Set the subframe length equal to the nearest lower value of 2^n ($5 \leq n \leq 7$).
- z. Multiply the total number of low rate telemetry points by 1.20.
- a. Set the number of subframes to the next larger integer of the number of low rate telemetry points divided by the subframe length.
- b. Select a DTU based on:
 1. Bit rate
 2. Word length
 3. Number of mainframe words
 4. Number of subframes
 5. Number of words per subframe
 6. Analog-to-digital converter requirement
- c. If the Communication Subsystem does have a separate downlink for the mission equipment data, repeat steps b through b using the mission equipment CDPI requirements.
- d. If the user specifies a special command synchronization requirement, print an exception report.
- e. Sum up total number of commands required.
- f. Multiply number of commands by 1.5.

- g. Raise the number of commands to next larger power of 2^n .
- h. Set the command word length equal to n .
- i. If time tagged-commands are specified in data base, state that time tagging is required and set the number of memory words required equal to the number of time tagged commands.

5.4 GENERAL PURPOSE PROCESSOR CONFIGURATION

5.4.1 Functional Description

The model of a general purpose (centralized) Data Processing Subsystem is characterized by having a general purpose computer aboard a satellite which has sufficient computing capability to handle all (or some well defined part) of the data processing tasks aboard that satellite. While it is clearly recognized that this represents a departure from the major portion of today's spacecraft, it is also recognized that this approach becomes more attractive as more capable spaceborne computers become available. The estimated rate in operations per second will be used to evaluate the utility of a given computer in a given satellite. Secondary considerations such as maximum memory size constraints and special radiation hardening requirements will not be included in this model.

5.4.2 Design Equations

The requirements of a general purpose Data Processing Subsystem are contained within the following driving functions:

- a. The number of inputs to the processor and the rate of data at that input.
- b. The number of outputs from the processor and the rate which each generates (no multiplier for operations is assumed since each output is assumed to be a result of an operation on an input).
- c. A 20-percent factor added to the total to account for software overhead attributed to the operating system (executive program).
- d. A 50-percent factor added for expansion and contingencies.

The first decision is whether to process the telemetry within the general purpose computer or to process it separately in the classical manner. The input for this decision is user supplied and could be to do both so a comparison can be made directly. One factor appearing is a term designated "Operations Required." For S&C this factor must be defined by the subsystem since different types of S&C require different computations. For telemetry the Operations required will be four and command will be six since these functions require no processing but only a storage and output plus decoding in the case of command. Telemetry rate has been previously computed and command rate is the rate at which commands can be sent which must be supplied by the Communication Subsystem (the rate must include any time required for a preamble). Mission Ops is user specified.

5.4.3 Design Logic

The sequence followed in implementing the Design Algorithm is as follows:

- a. Duplicate the logical operations specified for the Special Purpose Processor configuration. Do not select the DTUs from the data base at this point.
- b. Set the telemetry operations rate equal to the product of the bit rate and the number of telemetry operations required divided by the word length.
- c. If the user specifies that the telemetry is to be processed separately, set the telemetry operations rate equal to zero and select the telemetry DTU from the data base.
- d. Set the S&C operations rate equal to the product of the number of attitude sensors, the operations required per sensor, and the processing rate.
- e. Set the Command operations rate equal to the product of the command rate and the number of operations required per command.
- f. The total operations rate is equal to the sum of the operations rates for telemetry, S&C, command, and mission equipment.

- g. Multiply the total operations rate by 1.2 to account for overhead functions.
- h. Multiply the new total operations rate by 1.5 to allow for expansion.

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6. COMMUNICATION SUBSYSTEM

6.1 GENERAL

A design algorithm has been prepared which can be used to establish preliminary designs of the Communication Subsystem. The algorithm establishes the subsystem configuration, selected performance characteristics required of the components, and criteria for selecting acceptable components from the data base.

Some design considerations require knowledge of the standards employed; these standards were not available for the NASA Unified S-Band (USB) systems and, as a consequence, parts of the algorithm gravitate toward the Air Force Space Ground Link System (SGLS). However, the model has been made more general wherever possible. Reference 6-1 was used for information on the SGLS and Reference 6-2 was used for the NASA system.

The algorithm includes provisions for determining the characteristics of:

- a. Antennas
- b. Transmitters
- c. Receivers
- d. Command Signal Conditioners
- e. Baseband Assembly Units
- f. Diplexers

Command decoders, digital telemetry multiplexers, and recorders are not included because they were assigned to the Data Processing Subsystem. Encryption and decryption equipment are primarily of interest in DOD programs and are not included in the model.

6.1.1 Subsystem Function Description

The Communication Subsystem receives the uplink transmissions and supplies the demodulated signals to the Command Decoder for satellite

and mission command and control. The subsystem transmits the "house-keeping" and mission data. The data is received from the Data Processing Subsystem. In many designs, the subsystem receives a ranging signal from the ground stations and transmits a corresponding signal back to the ground.

6.1.2 Component Functional Descriptions

The Communication Subsystem components perform the following functions:

- a. Antenna
 - 1. Radiate and/or receive electromagnetic signals.
 - 2. Provide gain/directivity.
- b. Receiver
 - 1. Demodulate RF uplink signal and produce a frequency shift keyed (FSK) signal output for the SGLS or a command pulse train for the NASA USB system.
 - 2. May provide a ranging code output and a drive signal for coherent operation of a telemetry transmitter.
- c. Command Signal Conditioner - Convert FSK from SGLS receiver to binary signals.
- d. Baseband Assembly Unit - Modulate digital housekeeping data from the data processing equipment on subcarrier(s) and combine with pseudo random noise (PRN) ranging code from the receiver.
- e. Transmitter
 - 1. Modulate housekeeping or mission data on a carrier at the appropriate radio frequency and amplify the power to the required level.
 - 2. May accept a ranging code and a drive signal for coherent operation.
- f. Diplexer - Combine or distribute signals at different radio frequencies.

6.1.3 Subsystem Configurations

Five configurations have been established. The simplest configuration, which is depicted in Figure 6-1, is a non-unified subsystem with a

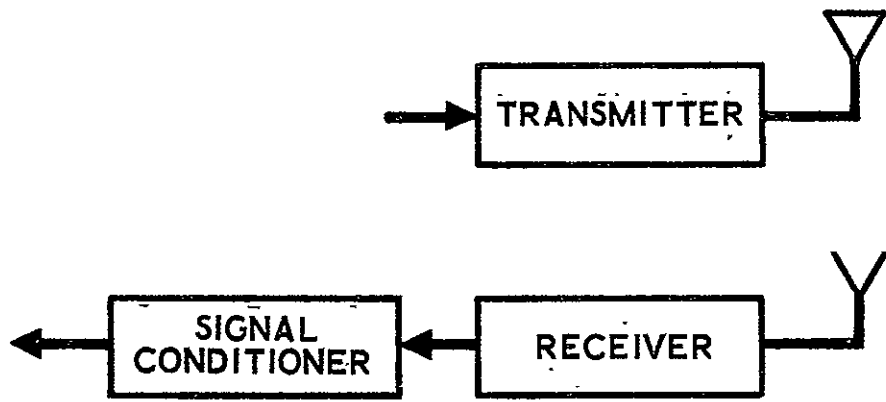


Figure 6-1. Separate Uplink and Downlink Configuration Block Diagram

separate uplink and downlink. Figures 6-2 and 6-3 present unified subsystem configurations having either a common antenna or separate antennas. Downlinks separate from the unified links are provided for the mission equipment data in the fourth and fifth subsystem configurations presented in Figures 6-4 and 6-5.

6.2 INPUT DATA

Input data for the subsystem model is required from the user and several subsystems. Inputs supplied by the user are listed in Table 6-2. As shown in Table 6-1, mandatory inputs (other than user-supplied) include mission and housekeeping data rate(s), vehicle stabilization type, and the amount of redundancy required.

Table 6-1. Input Data Requirements

User Input

(See Table 6-2)

Data Processing Subsystem

1. Mission data rate (bps)
2. Housekeeping data rate (bps)

Stabilization and Control Subsystem

1. Stabilization type (spin versus nonspin)

Reliability

1. Degree of redundancy

6.3 SUBSYSTEM DESIGN ALGORITHM

The Design Algorithm provides a means of establishing Communication Subsystem configuration, selected performance characteristics

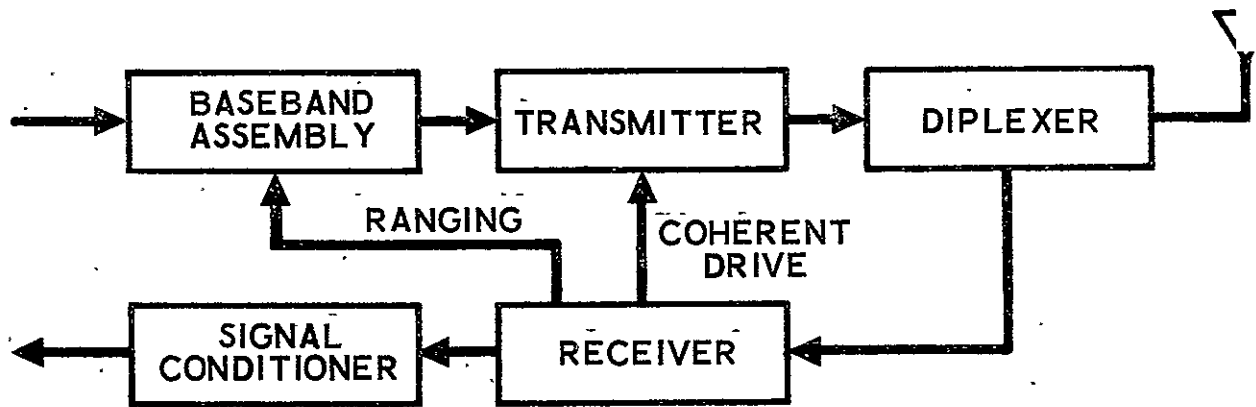


Figure 6-2. Unified Link, Common Antenna Configuration Block Diagram

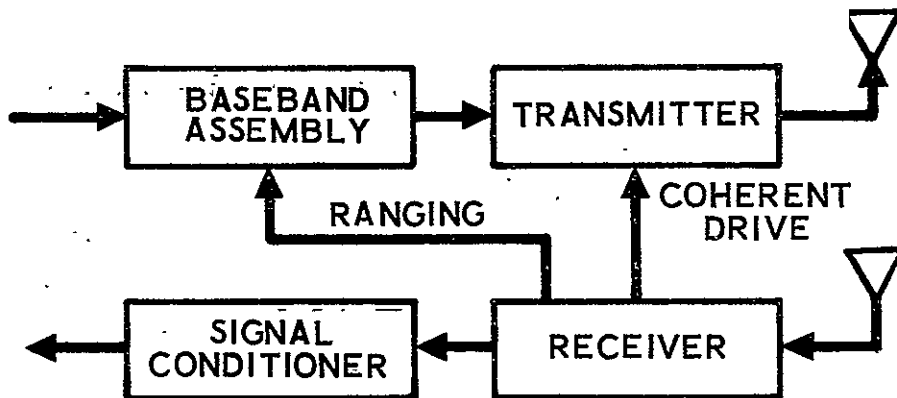


Figure 6-3. Unified Link, Separate Antennas Configuration Block Diagram

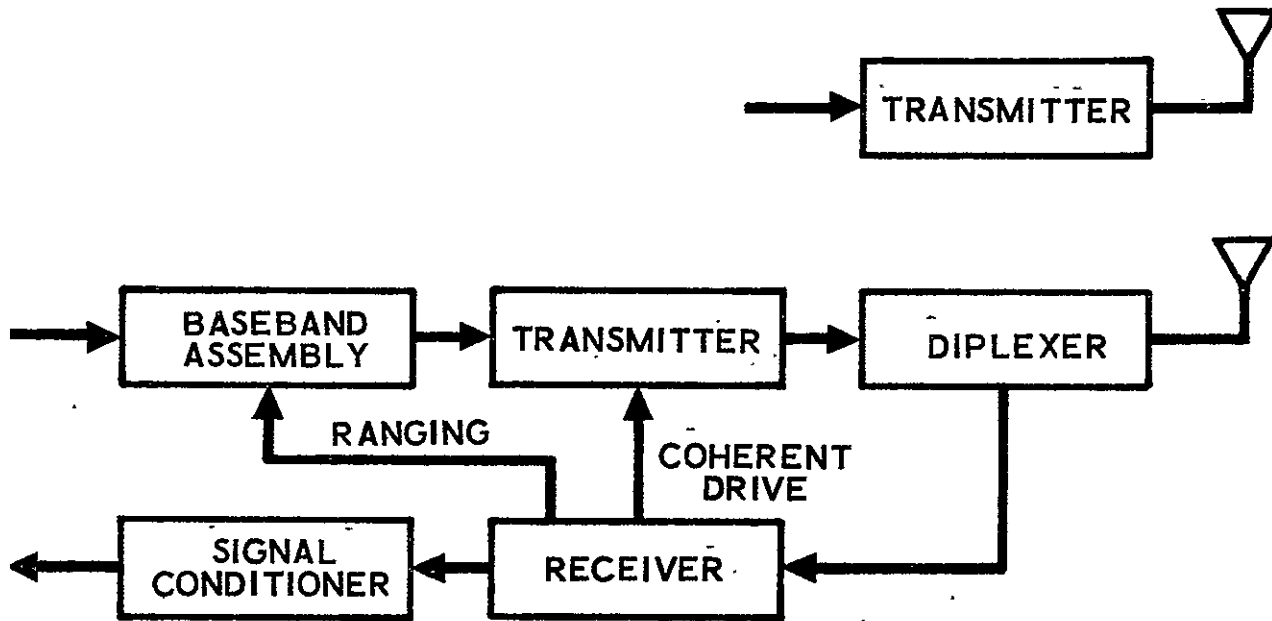


Figure 6-4: Unified Link, Common Antenna Plus Downlink Configuration Block Diagram

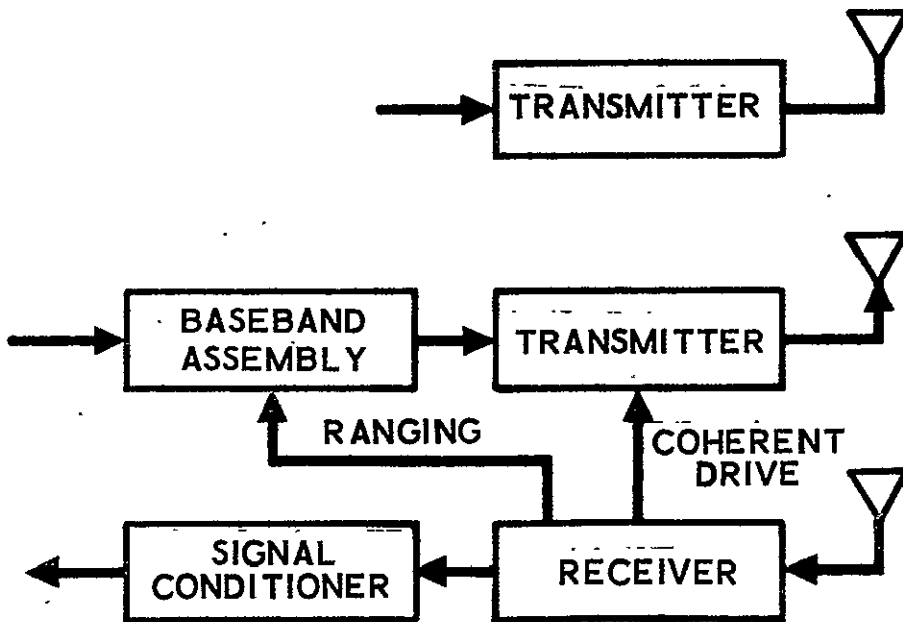


Figure 6-5. Unified Link, Separate Antennas Plus Downlink Configuration Block Diagram

Table 6-2. Input Data Supplied by User

Name	Required (R) or Optional (O)	Default Value
Compatibility requirement (USB/SGLS)	O	SGLS
Range and range rate requirement	O	No
Separate downlink requirement	O	No
Separate antenna requirement	O	No
NASA/AFSCF net	O	AFSCF
Nadir coverage requirement	O	No
Orbit apogee (nmi)	R	
Vehicle orientation (earth/non-earth)	R	
Frequency of each link (Mhz)	O	S-band
Signal-to-noise ratio (dB)	O	10
Ground station G/T	}	
or		
Receiving antenna gain (dB)		
Receiving system noise temperature ($^{\circ}$ K)	O	SCF USB 47.5 44
Downlink margin (dB)	O	220 170
Uplink carrier modulation	O	6
Uplink subcarrier modulation	O	PM
Downlink carrier modulation	O	FSK
Downlink subcarrier modulation	O	PM
Modulation index of ranging code (radians)	O	PSK
Modulation index of each subcarrier on downlink (radians)	O	0.1
Command bit or band rate (bps)	O	1 subcarrier 1.8 2 subcarriers 1.0
Transmitter circuit losses (dB)	O	1000
Type of command signal	O	1.0/1.5
Telemetry bandwidth (Hz)	O	Ternary
Polarization loss (dB)	O	$\pm 10^{10}$
Antenna off-axis loss (dB)	O	0
	O	0

required of the components, and criteria for selection of acceptable components from the data base.

6.3.1 Configuration Selection

The Design Algorithm will normally design all five subsystem configurations. If the input data so indicates, one or more of the configurations may be eliminated according to the user-supplied input requirements. This configuration test is outlined in Table 6-3.

6.3.2 Design Equations

6.3.2.1 Baseband Assembly Unit

The Baseband Assembly Unit (BBAU) is an SGLS component with characteristics and capabilities consistent with those described in Reference 6-1. The maximum data rates allowed on the subcarrier(s) are limited to those for which Reference 6-1 provides data on bit error rate degradation due to hardware.

A BBAU is used as part of the unified links. The BBAU selection is based on the number of input bit streams to the unified link and the bit rate of each stream.

For a single bit stream, a BBAU with a 1.024 MHz subcarrier is used for data rates up to and including 128 kbps, and a BBAU with a 1.7 MHz subcarrier is used for 256 kbps.

For two bit streams, a BBAU with a 1.024 MHz subcarrier and a 1.7 MHz subcarrier is used; the bit stream with the lower bit rate is assigned to the 1.024 MHz subcarrier and the other is assigned to the 1.7 MHz subcarrier. The maximum capability of the 1.024 MHz subcarrier is 128 kbps, and the 1.7 MHz subcarrier capability is 256 kbps. The frequency of the subcarrier associated with each bit stream must be noted from the data base for later use.

6.3.2.2 Downlink Effective Radiated Power

This section of the algorithm is a straightforward link analysis for effective radiated power (ERP). Atmospheric and rain attenuation are

Table 6-3. Communication Configuration Selection

Configurations	Ranging Requirement	Separate Mission Link Requirement	Separate Antenna Requirement
Uplink plus downlink	No	No	Yes
Unified link, common antenna	Yes	No	No
Unified link, separate antennas	Yes	No	Yes
Unified link, common antenna plus downlink	Yes	Yes	No
Unified link, separate antennas plus downlink	Yes	Yes	Yes

Legend:

- Yes - Acceptable
- No - Unacceptable

not included because it is not of great significance at the radio frequencies commonly in current use for telemetry transmissions. Ground antenna pointing loss is not included because it is considered to be negligible in terms of the accuracy that can be expected from this general procedure. The computation should be made for the worst combination of space loss (corresponding to maximum transmission range) and satellite antenna gain in the direction of the receiving station. However, if this combination is unknown, alternate procedures are included to perform the calculations based on assumptions. The computation is performed using the following equation, which is in units of dB.

$$\text{ERP} = \text{S/N} + \text{SL} + \text{K} + \text{B} - \text{G/T} + \text{M} + \text{L} \quad (6-1)$$

where

- ERP = Effective radiated power (dBW)
- S/N = Signal-to-noise ratio
- SL = Space loss
- K = Boltzman's constant (-228.6 dBW/Hz/°K)
- B = 10 log bandwidth (Hz)
- G/T = Gain-to-temperature ratio of receiving system
- M = Margin
- L = Other losses which include:
 - Transmitter circuit losses
 - Polarization loss
 - Satellite antenna off-axis loss
 - Modulation loss
 - Bit error rate degradation due to ground hardware

If S/N is not specified, assume a value of 10 dB.

Space loss (SL) is obtained from the ratio of the area of a sphere at the transmission range (S) to the area of isotropic antenna.

$$\text{SL} = \frac{4 \pi S^2}{\lambda^2 / 4 \pi} \quad (6-2)$$

where λ is the wavelength, and λ and S are in meters.

Also

$$\lambda = \frac{3 \times 10^2}{f} \quad (6-3)$$

where f is in MHz.

Simplifying and converting units results in

$$SL = 6017 S^2 f^2 \quad (6-4)$$

where S is in nautical miles

In units of dB, the equation becomes

$$SL = 37.8 + 20 \log S + 20 \log f \quad (6-5)$$

The default value of 2250 MHz for frequency will later result in the same frequency for all transmitters that are based on the default value. In reality, a number of different frequencies would be assigned to the transmitters; however, use of the default value of frequency will not introduce significant errors in the major subsystem characteristics.

For the user who does not know the transmission range, apogee (A) is converted into slant transmission range (S) for an observer at 0° elevation angle. The geometry is shown in Figure 6-6.

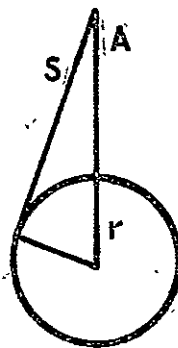


Figure 6-6. Orbit Geometry

Hence

$$S^2 = (A + r)^2 - r^2 \quad (6-6)$$

where $r = 3440$ nmi (6375 km)

$$S = \sqrt{A^2 + 6880A} \quad (6-7)$$

If bandwidth is unknown, * it is set equal to 10 log of the data rate in bits per second. (This assumes nonreturn-to-zero bit representation.)

If G/T is not specified, it may be computed from

$$G/T = G_R - 10 \log T \quad (6-8)$$

where

G_R = ground receiving (downlink) antenna gain (dB)

T = system noise temperature ($^{\circ}$ K)

If the noise figure (NF_{dB}) in dB is available, it should be converted to temperature in $^{\circ}$ K using

$$= \left(\text{antilog} \frac{NF_{dB}}{10} - 1 \right) 290 \quad (6-9)$$

Information for G/T of AFSCF stations may be obtained from Paragraph 4.1.3 of Reference 6-1. In the absence of other information, assume the use of the 46-ft antenna ($G_R = 47.5$ dB, $T = 220^{\circ}$ K).

Information for G/T of NASA stations may be obtained from Reference 6-2. In the absence of other information, assume the use of the 30-ft USB antenna with uncooled paramp ($G_R = 44$ dB, $T = 170^{\circ}$ K).

Margin is used to make allowances for miscellaneous losses not included in the analysis and may also be used to allow for nonoptimum system implementation. If not specified, use 6 dB.

If transmitter circuit losses are not specified, use 1.5 dB for unified link with a common antenna and 1.0 dB for all other links.

If polarization loss and satellite antenna off-axis loss are not specified, assume it to be 0 dB. (Assumptions for these losses are included in the reference values given later for antenna gain.)

*For the case of a unified link with two subcarriers, the dB values for bandwidth, modulation loss, and ground hardware degradation are summed for each subcarrier and the greater total is used.

The modulation loss for a separate downlink is 0 dB. For unified downlinks, the modulation loss for SGLS is calculated from the equations contained in Paragraph 4.1.2 of Reference 6-1. The equation for "TLM Mod. Loss" is used for a single subcarrier from the BBAU. For the case of two subcarriers from the BBAU, use the equation for "Subcarrier (1.024) Mod. Loss" for the 1.024 MHz subcarrier and "Subcarrier (1.7) Mod. Loss" for the 1.7 MHz subcarrier. If the modulation indices are not specified, the following may be assumed from Table 6-4.

Table 6-4. Modulation Indices

	PRN	Subcarrier
1 subcarrier	$\gamma = 0.1$ radian	$\beta = 1.8$ radian
2 subcarriers	$\gamma = 0.1$ radian	$\beta_1 = \beta_2 = 1.0$ radian

The absolute value of modulation loss is used.*

The bit error rate degradation due to ground hardware for SGLS is shown in Table 4-2 of Reference 6-1. Use the values for Carrier II for the separate downlink. For unified links, use the value(s) for the appropriate subcarrier(s).*

In the interest of simplicity, the link analysis for the unified downlink is limited to data transmission considerations. Analyses of the carrier and ranging signals are not included because they seldom have a significant effect on the sizing of the downlink. All default values of modulation indexes are within the SGLS limits cited in Reference 6-1. The

*For the case of a unified link with two subcarriers, the dB values for bandwidth, modulation loss, and ground hardware degradation are summed for each subcarrier and the greater total is used.

modulation index(es) were chosen so that the modulation loss is near minimum. For two subcarriers, the modulation indexes were made equal to provide equal modulation loss on the two subcarriers.

6.3.2.3 Antennas

Careful attention is usually given to the choice and design of satellite antennas because of the direct impact on the transmitter power. Depending on the total satellite electrical load, the transmitter power can have an impact throughout the satellite because of its influence on the electrical power subsystem. Each antenna shown in the configuration block diagrams should be considered separately. It is possible for each antenna on a satellite to be different from all other antennas on the same satellite.

User requirements for the antenna(s) should be used. If they are not available, the transmitting antenna selection may be made from Table 6-5. The code identifies appropriate antennas in the data base. Values of antenna gain are given for all antennas included in Table 6-5 except for the steerable parabola, which will be discussed later. Where a steerable parabola is chosen, a steering subsystem must be added to the block diagram as shown in Figure 6-7. The values of gain given include assumed influence of polarization loss and off-axis loss at acquisition. In the absence of better information, these values of gain can be used to size the transmitter. For the omni antenna, two values of gain are given. One value corresponds to the gain that will be exceeded over 87% of the radiation sphere and the second value corresponds to the gain that will be exceeded over 97% of the radiation sphere. In the absence of a requirement regarding coverage, use the average of the two values, i. e., -9 dB.

Where a separate receiving antenna is required and the type is not specified, a selection may be made from Table 6-6.

6.3.2.4 Transmitter Power for Fixed Antennas

The power in dBW required from the transmitter is obtained from:

$$P_T = ERP - G_T \quad (6-10)$$

Table 6-5. Transmitting Antenna Selection

Satellite Orientation Perigee Altitude	Earth		Non-Earth Oriented
	Nonspin Stabilized	Spin Stabilized	
Low (to 12,970 km) (7000 nmi)	Nadir coverage not required and large ground plane available. ** Monopole ($G_T = 2$ dB) (F1)	Omni* (B)	Omni* (B)
Medium (12,970 < 22,240 km) (> 7000 to < 12,000 nmi)	Nadir coverage required or large ground plane not available. ** Broadbeam conical spiral ($G_T = -1$ dB) (F2)		
High (22,240 km to synchronous) (12,000 nmi)	Helix ($G_T = 10$ dB) (F3)	Omni* (B)	Omni* (B)
	Parabola ($G_T = 15$ dB) (B)	biconical (A) ($G_T = 2$ dB)	Data Rate > 10 kbps Steerable Parabola Data Rate < 10 kbps Omni* (B)

* $G_T = -5$ dB (87%)
 $= -13$ dB (95%)

** Large ground plane must extend at least 5λ in all directions.

$$\lambda = \frac{984}{f} \text{ feet, } f \text{ in MHz}$$

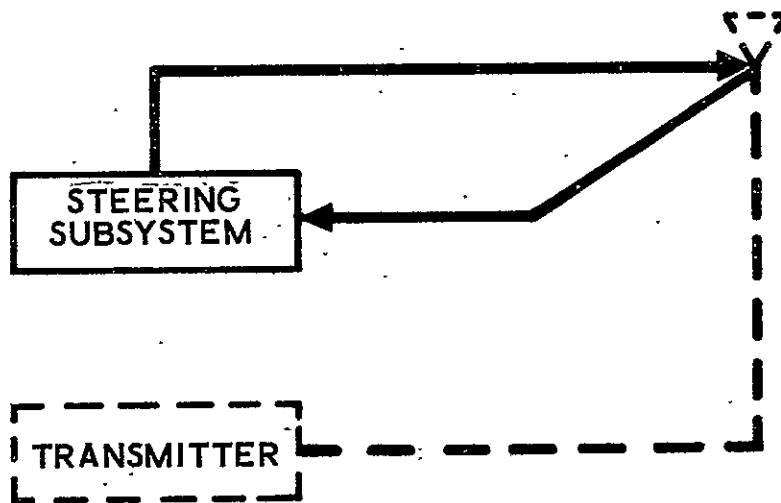


Figure 6-7. Antenna Steering Subsystem

Table 6-6. Separate Receiving Antenna Selection

Satellite Orientation		
Earth		Non-Earth Oriented
Nonspin Stabilized	Spin Stabilized	
Nadir coverage not required and large ground plane available. * Monopole (F1)	Omni (B)	Omni (B)
Nadir coverage required or large ground plane not available. * Broadbeam conical spiral. (F2)	Omni (B)	Omni (B)

* Large ground plane must extend at least 5λ in all directions.

$$\lambda = \frac{984}{f} \text{ feet, } f \text{ in MHz}$$

G_T = antenna gain in dB.

The transmitter power in watts (P_W) can be obtained from:

$$P_W = \text{antilog} \frac{P_T}{10} \quad (6-11)$$

6.3.2.5 Transmitter Power for Steerable Parabola

In the absence of performance requirements for the steerable parabola, a tradeoff between antenna gain and transmitter power is appropriate. The gain of the parabolic antenna in dB is given by:

$$G_T = 20 \log D + 20 \log f - 52.6 \quad (6-12)$$

where

D is the diameter of the parabola in feet

f is the frequency in MHz

If f is unknown, use 2250 MHz.

Using the relationship:

$$P_T = \text{ERP} - G_T \text{ and} \quad (6-13)$$

$$P_W = \text{antilog} \frac{P_T}{10}, \quad (6-14)$$

satisfactory combinations of transmitter power and antenna diameter may be obtained.

6.3.2.6 Transmitter Selection

The power output from the transmitter selected must equal or exceed the value of power required which was previously calculated.

For unified links, the transmitter must be capable of servicing a ranging code, usually PRN, and the input subcarrier frequency or frequencies from the selected baseband assembly unit.

For separate downlinks, the transmitter input data rate capability must equal or exceed the mission data rate from the data processing subsystem.

The transmitter frequency must comply with requirements.

In addition, the transmitter must be compatible with each of the following that are specified:

- a. Modulation type
- b. SGLS range and range rate compatibility
- c. USB range and range rate compatibility

6.3.2.7 Receiver Selection

The receiver must be compatible with:

- a. Operating frequency (if unspecified, assume S-band, 1750-1850 MHz)
- b. Command rate (if unspecified, assume 1000 baud for a ternary system or 1000 bps for a binary system)

Further, the receiver must be compatible with each of the following that are specified:

- a. SGLS range and range rate
- b. USB range and range rate
- c. Modulation type (of transmitted signal)

The general description of the command output should be noted from the data base for use in command signal conditioner selection.

6.3.2.8 Command Signal Conditioner Selection

The general description of the input (e.g., ternary FSK: 65, 76, 95 kHz) must be the same as the general description of the command output from the command receiver selected from the data base. Also, the command rate capability of the unit must equal or exceed the command rate requirement.

6.3.2.9 Diplexer Selection

The diplexer must be compatible with the receive (uplink) frequency and the transmit (downlink) frequency. Further, the transmit power rating of the diplexer must equal or exceed the power output from the transmitter which was selected from the data base.

6.3.2.10 Harness

A harness is necessary as a part of the Communication Subsystem. The harness weight is estimated in the Vehicle Sizing model.

6.3.3 Design Logic

The sequence followed in implementing the Design Algorithm is as follows. The components are selected on the basis of the parameters listed:

- a. If unified link, select baseband assembly unit.
 1. SGLS/USB compatibility
 2. Number of input bit streams
 3. Bit rate of each stream
- b. Compute the required effective radiated power (ERP).
- c. Select antenna(s).
 1. Vehicle stabilization (spin versus nonspin)
 2. Vehicle orientation (earth versus non-earth)
 3. Orbit altitude
 4. Nadir coverage requirement
- d. Compute required transmitter power.
- e. Select transmitter(s).
 1. SGLS/USB compatibility
 2. Range and range rate compatibility
 3. Modulation type
 4. Transmitter frequency
 5. Input data rate for a separate downlink
 6. Ranging code and input subcarrier frequencies for unified link.
 7. Transmitter power requirement
- f. Select receiver.
 1. SGLS/USB compatibility
 2. Range and range rate compatibility
 3. Modulation type (of transmitted signal)

4. Operating frequency
5. Command rate
- g. Select command signal conditioner.
 1. Command input
 2. Command rate
- h. For common antenna, select diplexer.
 1. Receive frequencies
 2. Transmit frequencies
 3. Transmitter power requirement

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7. ELECTRICAL POWER SUBSYSTEM

7.1 GENERAL

7.1.1 Subsystem Functional Description

The Electrical Power Subsystem (EPS) performs the following functions:

- a. Primary electrical power source
- b. Energy storage and secondary power capability
- c. Power conversion and conditioning
- d. Power distribution, control, and protection

For the purposes of the Systems Cost/Performance Model, the Electrical Power Subsystem will be divided into three elements:

- a. Electrical Power Sources - The Electrical Power Sources element of the subsystem will provide the electrical power generation and storage of electrical energy.
- b. Electrical Power Conditioning - The Electrical Power Conditioning element will provide the control over the electrical energy to ensure that it is regulated as to voltage, amperage, and frequency, and provides rectification where required. Included in this element are the assemblies for maintaining and controlling battery charging; however, the actual power source is not included.
- c. Electrical Power Distribution - The Electrical Power Distribution element of the subsystem provides the hardware interconnections between the electrical sources, conditioning equipment and the using subsystems and/or assemblies. This element includes the harness, the junction and breakout boxes, as well as power switching.

The first two elements, Sources and Conditioning, will be modeled in detail. The third element, Distribution, has been incorporated as a part of the Vehicle Sizing Model described in Section 9.

7.1.2. Subsystem Configurations

Two configuration models have been developed for Electrical Power Sources:

- a. Body Mounted Solar Arrays
- b. Oriented Paddle Mounted Solar Arrays

Electrical Power Conditioning consists of three basic configurations.

- a. Shunt Voltage Regulation
- b. Shunt and Discharge Voltage Regulation
- c. Series Load Regulation

Considering all combinations of configurations, the Electrical Power Subsystem model includes six configurations. Each of the basic five configurations is described briefly in the following sections.

7.1.2.1 Electrical Power Source Configurations

The Electrical Power Source Configurations require both solar arrays and batteries. The solar cell array is the primary power source for the spacecraft and is essentially a photovoltaic device which converts solar energy directly into electrical energy when exposed to sunlight. The solar array must generate sufficient power to satisfy most of the electrical power requirements and is supplemented by batteries which supply electrical power during eclipse periods when the array is not illuminated by the sun. Spacecraft batteries are charged using solar array power during sunlight periods, but they also deliver energy to spacecraft loads during peak load demands when solar array power output is insufficient to supply load demand.

7.1.2.2 Shunt Voltage Regulation Configuration

The Shunt Voltage Regulation Configuration is depicted in Figure 7-1. During daylight, the solar array generates power for the

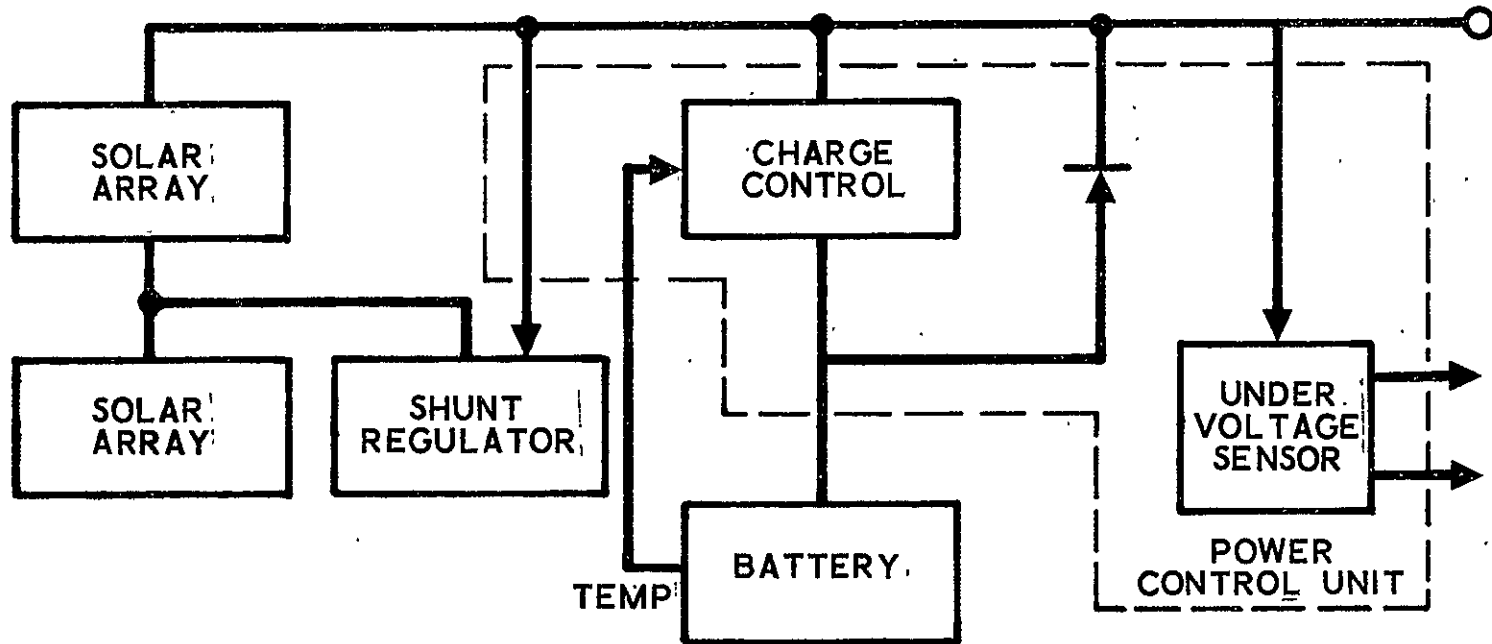


Figure 7-1. Shunt Voltage Regulation Configuration

vehicle loads as well as recharging batteries. The solar array is split so that a portion is shunted by transistor switches that dissipate excess energy and limit the bus voltage to 31.8 ± 0.2 V. The charge control unit provides several command and control functions for rapid charge, trickle charge, reconditioning and battery disconnect. After an eclipse, the charge control will connect the battery directly to the primary power bus until its temperature indicates full charge. The charge rate then drops to trickle until the battery temperature reduces sufficiently to cycle back to full rate, thus maintaining full charge.

During eclipse operation, the battery will support the primary power bus through the diode until the discharge current reaches a preset value, at which time the charge control bypasses the diode to eliminate the voltage drop. The bus voltage will be that of the battery and can drop to 25.3 volts. If the bus voltage drops below 24.75 volts, the undervoltage sensor will command off several non essential loads to reduce the drain on the battery. This is relatively simple EPS with a minimum of ground command or vehicle attitude control requirements.

7.1.2.3 Shunt and Discharge Voltage Regulation Configuration

The Shunt and Discharge Voltage Regulation Configuration is depicted in Figure 7-2. In this configuration, the bus voltage is limited to a small variation by means of a shunt regulator and a boost-discharge regulator which are controlled by the Central Control Unit (CCU). An essential feature of the CCU is to separate the operating ranges of the shunt, charge, and discharge regulators to avoid simultaneous operation and maximize the efficiency of solar array usage. For example, during the approach of an eclipse the CCU, sensing a falling bus voltage, turns on the boost-discharge regulator. At the end of the eclipse, as the solar array begins to share the bus load, the discharge regulator will begin to turn off. When the solar array can fully supply the load power, the

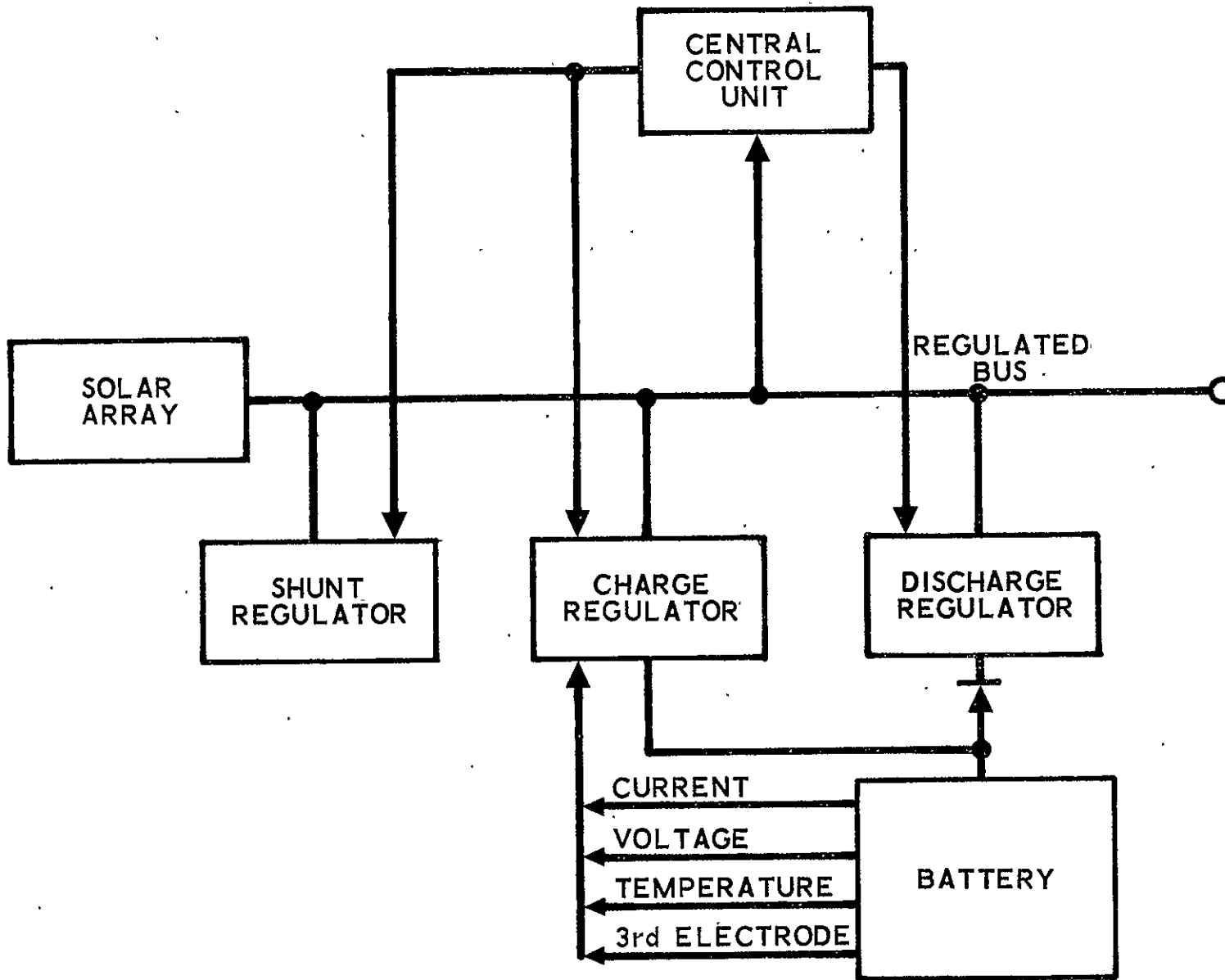


Figure 7-2. Shunt and Discharge Voltage Regulation Configuration

discharge regulator is turned completely off and the charge regulator is turned on. The CCU keeps the shunt regulator off until the battery has been fully charged or until the maximum charge rate has been reached.

The buck-charge regulator reduces the voltage applied to the battery to a value that will maintain the desired charge rate. The regulator senses various battery parameters and adjusts the charge rate to restore the battery to full capacity.

At the end of charge or when the array power is in excess of its normal needs, the bus voltage will begin to rise. The CCU will then turn on the shunt regulator which will apply power resistors to thermally dissipate the excess power and lower the bus voltage.

7.1.2.4 Series Load Regulation Configuration

The Series Load Regulation Configuration is depicted in Figure 7- During sunlight, the solar array generates power for the unregulated bus to charge the battery and supply the regulated bus loads through the load regulator. The unregulated solar array voltage will vary from 26 to 80 volts. The charge regulator reduces the unregulated bus to provide a temperature modified constant-current, constant-voltage characteristic. Charge rate reduction or termination is initiated by a third electrode signal or amp-hour meter.

The load regulator is a buck-boost switching type that closely controls the varying input into a 28 volt \pm 2 percent output as required by the vehicle and experiment loads.

During eclipse operation, the solar array voltage will drop until the battery diode is forward biased and the battery will supply the unregulated bus to support the vehicle loads. The load regulator will continue to control the load voltage.

7.1.3 Configuration Compatibility

Not all of the Electrical Power Subsystem configurations are suited to any particular spacecraft design.

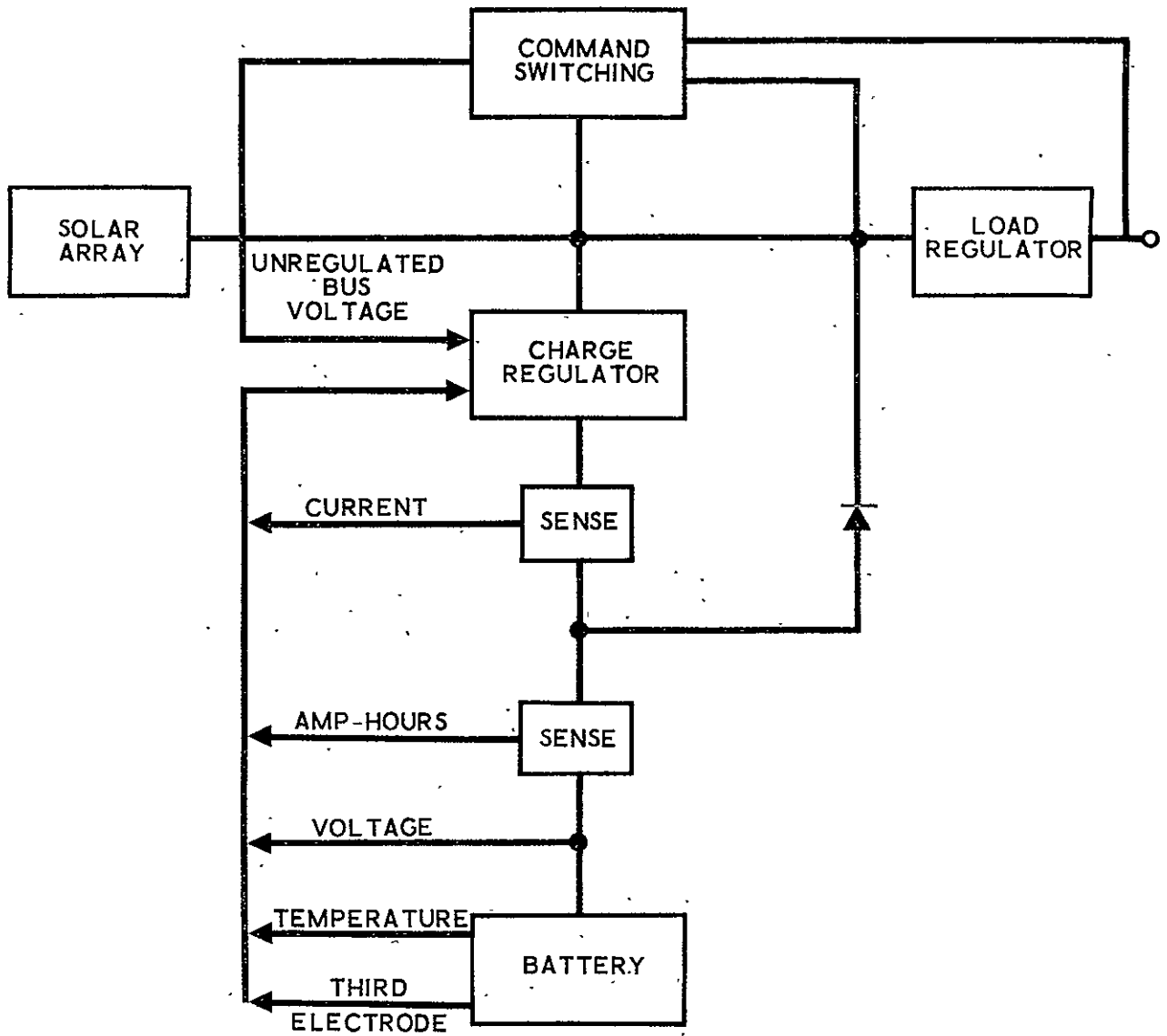


Figure 7-3. Series Load Regulation Configuration

Table 7-1 summarizes the limitations. In particular, oriented paddle mounted solar arrays are unsuited for a spinning vehicle. Likewise, an unregulated electrical power system is not well suited for spacecraft equipment requiring tight voltage regulation. In the latter case, the Shunt Voltage Regulation Configuration will be designed; however, any equipment selected as part of the spacecraft design and which requires tighter voltage regulation will be properly pointed out in the design description.

7.1.4 Equipment Types

The complete list of equipment types from which the subsystem components will be selected is provided below:

- a. Solar Arrays
- b. Batteries
- c. Battery Charge Regulators
- d. Battery Discharge Regulators
- e. Series Load Regulators
- f. Shunt Regulators
- g. Control Assemblies

The control assemblies provide selection or interconnection of EPS components and are designated by any of the following names:

- a. Central Control Unit
- b. Solar Power Distributor
- c. Power Distributor
- d. Power Control Unit
- e. Command Switching Unit

7.2 INPUT DATA

The information required from the user to design the Electrical Power Subsystem is identified in Table 7-2. This table indicates which data are used and whether the data are required or optional. If optional, the table indicates what default value is used in the event the data are not supplied.

Table 7-1. Electrical Power Configuration Compatibility

Configuration	Vehicle Orientation		Voltage Requirements	
	Spinning	Nonspinning	Unregulated	Regulated
Solar Arrays				
Body Mounted	Yes	Yes		
Oriented Paddles	No	Yes		
Voltage Regulation				
Shunt			Yes	No
Shunt and Discharge			Yes	Yes
Series			Yes	Yes

7-9

Table 7-2. Input Data Supplied by User :

Symbol	Name	Required (R) or Optional (O)	Default Value
h_p	Orbit perigee (feet)	R	
S	Average solar intensity (watt/meter ²)	O	1353
η_E	Battery watt-hr charge efficiency (dimensionless)	O	0.65
η_{CA}	Battery amp-hr charge efficiency (dimensionless)	O	0.75
η_I	Solar cell efficiency, at 28°C, AMO illumination (dimensionless)	O	0.105
λ_P	Solar array packing factor (active sur- face area/actual surface area) (dimensionless)	O	0.9
Δ_R	Radiation degradation factor (dimensionless)	O	0.2
Δ_F	Coverglass and coverglass adhesive transmissivity loss factor (dimensionless)	O	0.03
Δ_T	Temperature adjustment factor (dimensionless)	O	Table 7-3
Δ_I	Array fabrication loss factor (dimensionless)	O	0.02
Δ_M	Miscellaneous loss factor (dimensionless)	O	0.01
λ_G	Array orientation factor (dimensionless)	O	1 for sun oriented flat panels; $1/\pi$ for spinning oriented cyl- inder, $2/\pi$ for nonspinning oriented cylinders

Table 7-2. Input Data Supplied by User (Continued)

Symbol	Name	Required (R) or Optional (O)	Default Value
F_W	Array weight factor (dimensionless)	O	7.3 kg/m ² (1.5 lb)/ft for deployed panels; 3.4 kg/m ² (0.7 lb/ft ²) for body mounted arrays
λ_D	Average depth of discharge (at end of discharge)	O	0.30
K_1	Battery packing factor (dimensionless)	O	1.02
K_2	Battery structure weight factor (dimensionless)	O	1.40
λ_{DM}	Maximum depth of discharge (dimensionless)	O	0.70
V_C	Minimum allowable cell voltage (VDC)	O	1.10
R_{FD}	Temperature degradation factor (dimensionless)	O	1.20

In addition to the user supplied data, the following information is required from the equipment data base for every component selected as part of the spacecraft design:

- a. Average power requirement
- b. Minimum power requirement
- c. Voltage regulation requirement

The Reliability model must supply the following information:

- a. Number of batteries in parallel
- b. Number of series load regulators in parallel

The Stabilization and Control Subsystem model must indicate whether the spacecraft main body is spinning or nonspinning, and the Vehicle Sizing model must indicate the vehicle shape.

7.3 ELECTRICAL POWER SOURCE CONFIGURATIONS

There are two Electrical Power Source Configurations

- a. Body Mounted Solar Arrays
- b. Oriented Paddle Mounted Solar Arrays

These two configurations are discussed together in one section simply because of the similarity in the Design Equations and Logic. The only difference in the design approach is that the array orientation factor, λ_G , and weight factor, F_W , differ numerically for the two configurations. Other than this, the two design approaches are identical.

7.3.1 Functional Description

The Electrical Power Source Configuration components perform the following functions:

- a. Solar Array - The solar cell array is the primary power source for the spacecraft and is essentially a photovoltaic device which converts solar energy directly into electrical energy when exposed to sunlight. The solar array must generate sufficient

power to satisfy all of the electrical power demand plus sufficient energy to charge batteries which supply electrical power during periods when the array is not illuminated by the sun.

- b. Batteries - Spacecraft batteries are charged using solar array power during sunlight periods. They deliver energy to spacecraft loads during eclipses when solar array output is zero and during peak load demands during sunlight when solar array power output is insufficient to supply load demand.

7.3.2 Design Equations

7.3.2.1 Solar Arrays

A solar array will consist of a number of parallel-connected strings of series-connected solar cells. The cells are covered with glass sheets to minimize charged particle degradation. The cells may be bonded to metallic or plastic substrates. Array construction includes either designs in which cells are attached to spacecraft structure or cells are attached to flat panels which are deployed in orbit. The equations and design algorithm discussed in the remaining part of this section are applicable to all of the most common array designs. The algorithm does not account for the weights of solar array deployment mechanisms and stowage structure.

Solar array sizing requires the successive evaluation of the following equations:

$$P_S = \frac{P_L}{\eta_R} \left[1 + \frac{T_E}{T_S} \left(\frac{1}{\eta_D \eta_C \eta_E} \right) \right] \quad (7-1)$$

$$A = \frac{P_S}{(F_S) (S) (\eta_I)} \quad (7-2)$$

where

$$F_S = \lambda_G \lambda_P \left[(1 - \Delta_R) (1 - \Delta_F) (1 - \Delta_T) (1 - \Delta_I) (1 - \Delta_M) \right] \quad (7-3)$$

$$W_A = A F_W \quad (7-4)$$

and where

P_S - End-of-Mission Average Array Power Output

- P_L = Average Load Power Watts
 = Sum of the average power for all user and house-keeping subsystems (e. g., S&C, APS, CDPI, Thermal Control.) It is assumed that the array average power will be supplemented by batter power in order to meet peak power transients or low duty cycle demands.
- η_R = Power Distribution Loss Factor (array to loads)
 = Harness Efficiency x Load Regulator Efficiency (where applicable)
 = 98% x 85% (default values in place of actual design value)
- η_D = Discharge Regulator Efficiency
 = 85% (default value)
- η_C = Charge Regulator Efficiency
 = 98% for the unregulated bus configuration
 = 85% for the regulated bus configuration

and where

- S = Average solar intensity (1353 W/m^2)
- η_E = Battery watt-hr charge efficiency (dimensionless)
- η_{CA} = Battery amp-hr charge efficiency (dimensionless)
- T_E = Eclipse period (hr)
- T_S = Sunlight period (hr)
- η_I = Solar cell efficiency, at 28°C , AMO illumination (dimensionless)
- λ_P = Solar array packing factor (dimensionless)
- Δ_R = Radiation degradation factor (dimensionless)
- Δ_F = Coverglass and coverglass adhesive transmissivity loss factor (dimensionless)
- Δ_T = Temperature adjustment factor (dimensionless)

Δ_I	=	Array fabrication loss factor (dimensionless)
Δ_M	=	Misc. loss factor (dimensionless)
λ_G	=	Array orientation factor (dimensionless)
F_W	=	Array weight factor (kg/m ²)
F_S	=	Array sizing factor (dimensionless)
V_{DB}	=	Average battery discharge voltage
A	=	Array area (m ²)
W_A	=	Array weight (kg)

The energy balance equation [Eq. (7-1)] is based on the requirement that during sunlight portion of the orbit the solar array must supply sufficient energy to satisfy spacecraft load requirements and to recharge the battery which supplies the loads during the eclipse period.

For sizing purposes the worst case eclipse should be selected, i. e., maximum value of T_E/T_S ratio. The maximum ratio between T_E and T_S as a function of altitude for circular earth orbits is computed as follows:

$$S_e = \frac{102}{100} \sin^{-1} \left(\frac{h_e}{h_e + h_p} \right) \quad (7-5)$$

where S_e = Angular Size of Earth's Shadow (radians)

h_e = Earth Radius (km)

h_p = Orbit perigee (km)

and where

$\frac{102}{100}$ is due to the atmosphere

$$\frac{T_E}{T_S} = \frac{S}{\pi - S} \quad (7-6)$$

The sizing factor, F_S , accounts for array geometry and various array performance degradation factors. Temperature factors (Δ_T) which are required as input were estimated for: (a) spinning body-mounted arrays, (b) oriented flat panels of conventional construction, and (c) light-weight sun-oriented flat panels. Values for Δ_T are presented in Table 7-3 for low earth and synchronous orbits. As a first approximation, the array temperature factors for 12-hr elliptical orbits can be assumed to be approximately equal to the corresponding synchronous orbit cases.

Table 7-3. Typical Solar Array Temperature Correction Factors (Δ_T)*

Array Type	Low-Earth Orbits		Synchronous Orbits	
	10 Ω -cm	2 Ω -cm	10 Ω -cm	2 Ω -cm
Body mounted spinning	0.02	0.01	-0.06	-0.05
Body mounted non-spinning	0.06	0.04	0.03	0.02
Oriented panels (conventional construction)	0.13	0.11	0.10	0.08
Oriented Panels (Lightweight construction)	0.17	0.15	0.13	0.11

* The model assumes a standard solar cell resistivity of 10 Ω - cm.

The value of the radiation factor depends on the coverglass thickness and the magnitude of the charged particle radiation fluence that the array receives. For synchronous orbits, Δ_R will vary between approximately 0.2 and 0.3 for 7 and 15 year missions respectively. For low-earth orbits (< 780-km altitude), Δ_R is approximately 0.03 for one-year missions and approximately 0.05 for three-year missions (Ref. 7-1). The factor, Δ_R , will vary from 0.25 to 0.35 for 12-hr elliptical orbits for three and five-year missions respectively. A first order approximation of Δ_R can be obtained for each type of orbit by assuming a linear variation within the mission duration ranges discussed above.

7.3.2.2 Batteries

A battery assembly consists of multiple cells connected in series. One or more batteries may be required depending on the required battery capacity as well as thermal and reliability considerations. NiCd type batteries were selected as the reference battery type for this model and are commonly used for spacecraft power systems. One important factor affecting the life of this type of battery is temperature. High temperature accelerates performance degradation so that spacecraft thermal control systems are designed typically to maintain the batteries within an operating range of from 277 to 300°K.

The equations required to size and select the batteries are as follows:

a. Required Capacity

$$C_R = \frac{P_L T_E}{\lambda_D \eta_D} \quad (7-7)$$

$$C_A = C_R / V_{DB} \quad (7-8)$$

where V_{DB} = Average Battery Discharge Voltage
 = 27 V dc for configurations not using discharge regulators

and where $V_{DB} = 21$ V dc for configuration using discharge regulators.

b. Minimum Installed Capacity

$$C_I = C_A R_{FD} \quad (7-9)$$

c. Number of Cells in Series

$$N_C = V_{BM}/V_C \quad (7-10)$$

d. Unit Battery Weight and Volume

$$W_B = K_2 N_C W_{Cell} \quad (7-11)$$

$$V_B = K_1 N_C V_{Cell} \quad (7-12)$$

e. Total Battery Weight, Volume, and Capacity

$$W_{BT} = N_B W_B \quad (7-13)$$

$$V_{BT} = N_B V_B \quad (7-14)$$

$$C_T = N_B C_{Cell} \quad (7-15)$$

and where

$\lambda_D =$ Average depth of discharge (at end of discharge)

$K_1 =$ Battery packing factor (dimensionless)

$K_2 =$ Battery structure weight factor (dimensionless)

$\lambda_{DM} =$ Maximum depth of discharge (dimensionless)

$V_C =$ Minimum allowable cell voltage (V dc)

$R_{FD}^* =$ Temperature degradation factor (dimensionless)
battery temperature

* As a first approximation R_{FD} is assumed to be independent of depth of discharge and the number of charge/discharge cycles. It can also be assumed to vary linearly from 1 to 1.30 in the temperature range from 0 to 30°C.

V_{BM}	=	Minimum allowable battery voltage (V dc)
N_B	=	Number of batteries required in parallel (dimensionless)
C_A	=	Capacity required at end of mission (amp-hr)
C_R	=	Minimum required capacity (amp-hr)
C_I	=	Minimum installed battery capacity required (amp-hr)
C_{cell}	=	Capacity of selected cells (amp-hr)
N_C	=	Number of cells in series
V_B	=	Unit battery volume (m^3)
W_{BT}	=	Total battery weight (kg)
W_B	=	Unit battery weight (kg)
V_{BT}	=	Total battery volume (m^3)
V_{cell}	=	Volume of each cell (m^3)
W_{cell}	=	Weight of each cell (kg)
C_T	=	Total actual installed capacity (amp-hr)

This set of equations requires that the number of batteries, N_B , be computed by the reliability model and input to the power model. In general, N_B is computed by first assuming that at least two batteries are required as a minimum. Next a reliability value is computed based on no failures allowable in either battery during the mission. This result is then compared to required value. If the required value is exceeded, the algorithm determines that $N_B = 2$. If not, a third battery is added and the calculations are repeated. This procedure is repeated until a value of N_B is found such that the reliability requirement is met or exceeded. This approach was selected because it tends to minimize the number of iterations re-

quired to reach a solution and because it is compatible with available statistical models. It is conservative, since in reality, the most common type of cell failure (i. e., an electrical short circuit) does not result in a total loss of usable battery capacity, as assumed in the model.

Physical characteristics of commercially available cells are entered in the equipment data base. Cell volume and weight is obtained using the data in the equipment data base and a "look-up" procedure. Since only discrete values are given in the data base, the cell having the next highest capacity rating than that actually required is selected.

7.3.3 Design Logic

The flow logic or sequence that the Design Algorithm must follow is:

- a. Input data.
- b. Solve the energy balance equation for average array power output at end-of-mission.
- c. Compute the sizing factor which is a function of the array orientation and various degradation factors.
- d. Compute the required solar array area and weight.
- e. Compute the required minimum installed battery capacity.
- f. Determine number of cells in series.
- g. Select battery cells from the data base based on the number of parallel batteries being equal to two.
- h. Compute unit and total battery weight, volume, and capacity based on the total number of batteries as indicated by the Reliability model.

7.4 SHUNT VOLTAGE REGULATION CONFIGURATION

7.4.1 Functional Description

The Shunt Voltage Regulation Configuration components perform the following functions:

- a. Power Control Unit (PCU) - This assembly contains the bus voltage sensor for the shunt regulators and the switching controls for battery charging and reconditioning and non-essential load removal. The bus voltage sensor can turn on the shunt regulators to load the solar array and limit the array output voltage. The PCU also contains a dedicated charger for each battery, with capability to receive ground commands or operate autonomously. In addition, the charger can disconnect the battery from the bus and apply a reconditioning resistor to discharge it at a low rate.

During eclipse, the battery will discharge through the diode until the current exceeds a set limit that closes the relay contacts and bypasses the diode. If the primary bus voltage drops below 24.75 volts, the undervoltage sensor removes the nonessential loads to reduce the drain on the battery.

- b. Battery Charger - The charger is made up of a relay contact between the array bus and battery which is paralleled by a resistor and a diode. The relay contacts are closed for full charge, and through the resistor after the battery temperature indicates full charge and when the relay contacts are open.
- c. Shunt Regulator - The unit is a variable conductivity transistor circuit that shunts a section of the solar array to absorb excess energy and lower its output voltage. When the main bus voltage increases beyond 31.8 volts, the PCU senses the rise and turns on a shunt regulator to bypass part of the array current. Since the main bus voltage is the sum of the unshunted section plus the shunted section, the additional shunt current reduces the main bus voltage. The regulator operates primarily at the beginning of life of the solar array or at low temperature, with no battery charging and light payload use.

A block diagram of the Shunt Voltage Regulation Configuration was shown in Figure 7-1.

7.4.2 Design Equations

7.4.2.1 Battery Charge Regulator

For all configurations, the number of charge regulators is equal to the number of batteries. This is needed to prevent overcharge which reduces the battery life or could lead to destruction. The power handled

by the charger will vary throughout the charge cycle and is dependent upon the battery size, eclipse and sunlight periods, and battery recharge efficiency. The solar array power allocated for charging (P_C) is conditioned to provide a controlled current into the battery. A nominal maximum value of charge current in amperes for hermetically sealed NiCd. batteries is $C/2$, where C is the capacity in amp-hours. The charging power (P_C) is the sum of the charge power going into the battery (P_{CH}) and the power dissipated in each charger (P_{CD}) because of conversion losses.

For all EPS configurations, the distribution loss factor (η_R) from the array to the charger is the line efficiency and can be considered as 100 percent. The following is a summary of charge regulator designations and algorithms:

$$N_B = \text{Number of batteries} = \text{Number of chargers} \quad (7-16)$$

$$P_C = \text{Portion of solar array power allocated for battery charging}$$

$$= \frac{P_L}{\eta_R} \frac{T_E}{T_S} \frac{1}{\eta_D \eta_C \eta_E} \quad (7-17)$$

$$\eta_R = \text{distribution loss factor or line efficiency}$$

$$\approx 100 \text{ percent}$$

$$\eta_C = \text{Charge Regulator efficiency}$$

$$\approx 100 \text{ percent}$$

$$P_{CH} = \text{Charge power delivered to each battery}$$

$$= \frac{P_C \eta_C}{N_B} \quad (7-18)$$

$$= \frac{P_L}{N_B} \frac{T_E}{T_S} \left(\frac{1}{\eta_D \eta_E} \right)$$

$$\begin{aligned}
P_{CD} &= \text{Power to be dissipated by each charger into thermal control system} \\
&= \frac{P_C(1 - \eta_C)}{N_B} \quad (7-19) \\
&= \frac{P_L}{N_B} \frac{T_E}{T_S} \left(\frac{1}{\eta_D \eta_E} \right) \left(\frac{1}{\eta_C} - 1 \right)
\end{aligned}$$

To determine the charge current delivered by the regulator, the following equation can be used:

$$\begin{aligned}
I_{CH} &= \text{Charge current delivered by regulator} \\
&= \left(\text{default value of } \frac{C}{2} \right) \quad (7-20)
\end{aligned}$$

7.4.2.2 Shunt Regulator

The equation for the shunt regulator is intimately related to the design of the solar array. The equations provide the end-of-life array power (P_S). However, the array will be capable of delivering considerably more power at the beginning-of-life and immediately after the eclipse period when the array temperature is low. In order to limit the bus voltage under these initial conditions, shunt regulators are applied across the solar array.

Shunt regulators are variable conductivity dissipators that bypass a portion of the solar array current capability and reduce the array output voltage. In effect, the shunt regulators shift the operating point on the curve for array "E" vs "I" from high voltage-low current to lower voltage-higher current. Since the load current cannot be varied, the shunt regulators absorb the excess current through transistor circuitry. The sensor for the regulator detects the array voltage and keeps the shunt OFF for low voltages. When the array voltage increases beyond a limit value, the shunt turns ON gradually until it reaches a maximum value of shunt current. Regulators are operated in parallel with their limit value set at increasing array voltages, so that they turn on in sequence for increasing array voltage.

The number of regulators is dependent upon the anticipated excess solar array power at beginning-of-life and the maximum unit dissipating capability. If the maximum array power is given as P_{\max} and the minimum load power is $P_{L \min}$, then the number of shunt regulators needed is:

$$N_{SR} = \frac{P_{\max} - P_{L \min}}{\text{Max Unit Power Capacity}} \left\{ \begin{array}{l} \text{Select the next} \\ \text{larger integer} \\ \text{for any remainder} \\ \text{above 0.1} \end{array} \right. \quad (7-21)$$

7.4.3 Design Logic

The sequence followed in implementing the Design Algorithm is as follows:

- a. Compute the maximum charge current to be delivered by the battery charge regulator based on the selected battery charge capacity.
- b. Select a charge regulator from the data base possessing or exceeding the maximum charge current rating.
- c. Set the number of charge regulators equal to the number of batteries.
- d. Select the first shunt regulator from the data base.
- e. Select number of shunt regulators based on the anticipated excess solar array power at beginning-of-life and the regulator maximum dissipating capability.
- f. Call up configuration special hardware from the data base which is not differentiable, i.e., Power Control Unit.

7.5 SHUNT AND DISCHARGE VOLTAGE REGULATION CONSIDERATION

7.5.1 Functional Description

The Shunt and Discharge Voltage Regulation Configuration components perform the following functions:

- a. Discharge Regulator - The battery voltage is boosted to the bus level by means of the discharge regulator. This device uses a transistor switching circuit to generate a variable voltage across an inductor in series with the battery.
- b. Battery Charger - The charge current is limited according to the battery ambient temperature by reducing the bus voltage with a pulse-width modulated regulator. The bus voltage is switched across an averaging filter, with the output voltage controlled by varying the duty cycle of the transistor switch. When the battery sensors indicate full charge, the regulator is turned off.
- c. Shunt Regulator - The solar array characteristics vary with the temperature, solar intensity and life of the solar cells. If the load is insufficient to reduce the array voltage below a set level, the shunt regulator applies a dissipative circuit that loads down the bus to the set level. The regulator is made up of a sensor to detect the bus voltage and to control a variable conductivity transistor circuit that shunts the array.
- d. Central Control Unit (CCU) - The CCU is the decision making computer that senses the voltage of the regulated bus and sends operational commands to the shunt, charge and discharge regulators in order to optimize usage of the solar array and maintain the bus voltage regulation.

A block diagram of the Shunt and Discharge Voltage Regulation Configuration was presented in Figure 7-2.

7.5.2 Design Equations

7.5.2.1 Battery Charge Regulator

The charge regulators to be discussed here are nearly identical to the regulators discussed in Paragraph 7.4.2.1. The only substantial difference is a charge regulator efficiency, η_C , of 85 percent. For completeness, the equations will be repeated:

$$N_B = \text{Number of batteries} = \text{Number of chargers} \quad (7-22)$$

$$P_C = \text{Portion of solar array power allocated for battery charging}$$

$$= \frac{P_L}{\eta_R} \frac{T_E}{T_S} \frac{1}{\eta_D \eta_C \eta_E} \quad (7-23)$$

$$\begin{aligned}
\eta_R &= \text{Distribution loss factor or line efficiency} \\
&\approx 100 \text{ percent} \\
\eta_C &= \text{Charge Regulator efficiency} \\
&\approx 85 \text{ percent} \\
P_{CH} &= \text{Charge power delivered to each battery} \\
&= \frac{P_C \eta_C}{N_B} \tag{7-24} \\
&= \frac{P_L T_E}{N_B T_S} \left(\frac{1}{\eta_D \eta_E} \right)
\end{aligned}$$

$$\begin{aligned}
P_{CD} &= \text{Power to be dissipated by each charger into thermal control system} \\
&= \frac{P_C(1 - \eta_C)}{N_B} \tag{7-25} \\
&= \frac{P_L T_E}{N_B T_S} \left(\frac{1}{\eta_D \eta_E} \right) \left(\frac{1}{\eta_C} - 1 \right)
\end{aligned}$$

$$\begin{aligned}
I_{CH} &= \text{Charge current delivered by regulator} \\
&= (\text{default value of } C/2) \tag{7-26}
\end{aligned}$$

7.5.2.2 Shunt Regulator

The shunt regulator in the Shunt and Discharge Voltage Regulation Configuration is essentially identical to that in the Shunt Voltage Regulation Configuration.

The number of regulators is dependent upon the anticipated excess solar array power at beginning-of-life and the maximum unit dissipating capability. If the maximum array power is given as P_{\max} and the minimum load power is $P_{L \min}$, then the number of shunt regulators needed is:

$$N_{SR} = \frac{P_{max} - P_{L min}}{\text{Max Unit Power Capacity}} \left\{ \begin{array}{l} \text{Select the next} \\ \text{larger integer} \\ \text{for any remain-} \\ \text{der above 0.1} \end{array} \right. \quad (7-27)$$

The reader is directed to Paragraph 7.4.2.2 for a more complete discussion of the operation of Shunt Regulators.

7.5.2.3 Discharge Regulators

The regulator operates during eclipse periods and is not associated with the energy balance equation for the solar array. Each regulator adds a variable voltage in series with its battery so that the output voltage stays fairly constant for changes in load or battery voltage. The battery is the source of power (P_D) which is delivered to the regulator. The output is directly on the main bus and the line efficiency to the load can be considered as 100 percent. Therefore, the only power loss to be dissipated is that caused by the inefficiency of the regulator.

The following is a summary of discharge regulator designations and algorithms.

$$N_D = \text{Number of Discharge Regulators} = N_B \quad (7-28)$$

$$P_D = \text{Battery power delivered to each regulator}$$

$$= \frac{P_L}{N_D \eta_D} \quad (7-29)$$

$$\eta_D = \text{Discharge Regulator efficiency}$$

$$\approx 85 \text{ percent}$$

$$P_{DD} = \text{Power dissipated by each Discharge Regulator}$$

$$= \frac{P_L}{N_D} \left(\frac{1}{\eta_D} - 1 \right) \quad (7-30)$$

7.5.3 Design Logic

The sequence followed in implementing the Design Algorithm is as follows:

- a. Compute the maximum charge current to be delivered by the battery charge regulator based on the selected battery charge capacity.
- b. Select a charge regulator from the data base possessing or exceeding the maximum charge current rating.
- c. Set the number of charge regulators equal to the number of batteries.
- d. Set the number of discharge regulators equal to the number of batteries.
- e. Compute the power requirement to be dissipated by each discharge regulator.
- f. Select the discharge regulator from the data base possessing or exceeding the power dissipation rating.
- g. Select the first shunt regulator from the data base.
- h. Select number of shunt regulators based on the anticipated excess solar array power at beginning-of-life and the regulator maximum dissipating capability.
- i. Call up configuration special hardware from the data base which is not differentiable, i. e., Central Control Unit

7.6 SERIES LOAD REGULATION CONFIGURATION

7.6.1 Functional Description

The Series Load Regulation Configuration components perform the following functions:

- a. Battery Charger - The charge regulator uses a pulse width modulated switch to reduce the unregulated solar array bus to a voltage that will limit the maximum charge rate to an acceptable level. The charge continues until the battery temperature, voltage, third electrode signal or amp-hour capacity indicate recovery of energy discharged during the previous eclipse period. The maximum charge rate may be reduced if the array bus drops because of a high payload drain.
- b. Series Load Regulator - This unit is of the buck-boost type using semiconductor switching circuits to generate a voltage in opposition or supporting the unregulated bus voltage. The magnitude and polarity of the buck-boost voltage is controlled by the average regulated voltage. In addition, the regulator current can be used as a control function to provide load sharing amongst several parallel regulators.

- c. Command Switching - This unit is coordinated with the ground command capability of the entire vehicle and is used for power management of the EPS under varying conditions of load demand or component failure over the anticipated life.

A block diagram of this Series Load Regulation Configuration was presented in Figure 7-3.

7.6.2 Design Equations

7.6.2.1 Battery Charge Regulator

The charge regulators to be discussed here are nearly identical to the regulators described in Paragraphs 7.4.2.1 and 7.5.2.1. The only significant difference is the charge regulator efficiency, η_C , equal to 85 percent.

7.6.2.2 Series Load Regulator

The load regulator is a series element that must regulate the input power from the solar array or the battery. Using the energy balance equation for the array, the input to the regulator is the power delivered to the load (P_L) modified by the distribution loss factor (η_R). In this case, different from the previous components, the distribution loss or line efficiency is the efficiency of the load regulator (η_{LR}). Another difference is that the number of regulators is not related to any other EPS component. The number is dependent upon the average load power (P_L) and the unit handling capacity.

The following is a summary of load regulator designations and algorithms:

$$N_R = \frac{P_L}{\text{unit handling capacity}} \quad \left\{ \begin{array}{l} \text{select an integer greater} \\ \text{than two. If remainder is} \\ \text{greater than 0.2, select} \\ \text{next larger integer.} \end{array} \right. \quad (7-31)$$

For P_L <1000 watts use unit capacity of 225 watts
 >1000 watts use unit capacity of 350 watts

$$\begin{aligned}
 P_{LR} &= \text{Total power input to Load Regulators} \\
 &= \frac{P_L}{\eta_{LR}} \quad (7-32)
 \end{aligned}$$

$$\begin{aligned}
 \eta_{LR} &= \text{Load Regulator efficiency} \quad (7-33) \\
 &\approx 90 \text{ percent}
 \end{aligned}$$

$$\begin{aligned}
 P_{LRD} &= \text{Power dissipated by all Load Regulators} \\
 &= P_L \left(\frac{1}{\eta_{LR}} - 1 \right) \quad (7-34)
 \end{aligned}$$

Note that the peak power handling capacity is usually twice the average given in the equipment data base.

7.6.3 Design Logic

The sequence followed in implementing the Design Algorithm is as follows:

- a. Compute the maximum charge current to be delivered by the battery charge regulator based on the selected battery charge capacity.
- b. Select a charge regulator from the data base possessing or exceeding the maximum charge current rating.
- c. Set the number of charge regulators equal to the number of batteries.
- d. Select a series load regulator from the data base.
 1. For a total power load less than 1000 watts, use a unit capacity of 225 watts.
 2. For a total power load greater than 1000 watts, use a unit capacity of 350 watts.
- e. Select number of load regulators based on total power load and unit handling capacity. Select an integer number greater than two. If remainder is greater than 0.2, select next larger integer number.
- f. Call up configuration special hardware from the data base which is not differentiable:
 1. Solar Power Distributor
 2. Power Distributor

8. THERMAL CONTROL SUBSYSTEM

8.1 GENERAL

A Design Algorithm for sizing the Thermal Control Subsystem (TCS) is herein presented. The algorithm addresses only the basic "house-keeping" subsystems and does not address any mission equipment thermal control requirements. The approach taken in this model is to synthesize a Thermal Control Subsystem for potential spacecraft orbits and configurations without quantitatively optimizing the subsystem. The approach is qualitative in terms of the equipment used in the thermal control subsystem. Internal component related effects were not modeled. Any perturbation in the thermal control approach, however, will not significantly impact the overall spacecraft.

8.1.1 Subsystem Functional Description

The Thermal Control Subsystem is designed to provide an environment favorable to the operation of scientific instruments and other equipment vital to the mission by limiting temperature variations in that equipment. The onboard thermal environment is determined by the magnitude and distribution of radiation inputs from the sun and the planets, heat from internal sources (rockets, isotope heaters, and nuclear power sources), and heat from spacecraft electrical operations. The impact of these inputs is affected by the characteristics of the heat transfer paths within the spacecraft and the heat radiation characteristics of its external surfaces. The common purpose of the Thermal Control Subsystem components is to modify the heat transfer to and from each spacecraft element so that its temperature will remain within the allowable range during the entire life of the mission. Temperature stability and temperature gradients are also primary concerns in the design of the Thermal Control Subsystem.

8.1.2 Component Functional Descriptions

The Thermal Control Subsystem components perform the following functions:

- a. Phase Change Material (Thermal Mass). Phase change materials are those that can change from one physically distinct and mechanically separable state to another distinct form such as from a definite crystalline to a liquid state. Phase change materials used for temperature control are those whose melting point is close to the desired temperature of a component. Then the latent heat associated with the phase change provides a large thermal inertia when the temperature of the attached component is passing through the melting point. However, the phase change material cannot prevent a further temperature rise when all the material is melted. Phase change materials are used in electronic component thermal control systems to enable cyclically operating components to remain very nearly isothermal at all times, in thermal energy storage devices to store energy isothermally for later release, and in space flight experiments to maintain thermal stability.
- b. Insulation. Thermal insulation is designed to reduce the rate of heat flow per unit area between two boundary surfaces at specified temperatures. Insulation may be a single, homogeneous material such as a low-thermal-conductivity foam or an evacuated, multilayer, insulation system in which each layer acts in a low-emittance radiation shield and is separated by low-conductance spacers. Multilayer, evacuated insulations are widely used in the thermal control of spacecraft and components to (1) minimize heat flow to or from the component, (2) reduce the amplitude of temperature fluctuations in components because of time-varying external radiative heatfluxes, and (3) minimize the temperature gradients in components caused by varying directions of incoming external radiative heat.
- c. Heaters. Electrical heaters (resistance elements) are commonly used to maintain component temperatures close to desired values. The heater is typically part of a closed-loop system that includes a temperature sensing element and an electronic temperature controller. Electrical heaters are used in on-off control modes, ground-controllable modes (including command models), or simply in continuously-on modes.
- d. Radiators. The external surfaces of a spacecraft radiatively couple the spacecraft to space, the only heat sink available. Because these surfaces are also exposed to external sources

of energy, their radiative properties must be selected to achieve the balance at the desired temperature between internally-dissipated and external sources of power and the heat rejected to space. The two properties of primary importance are the emissivity of the surface ϵ and the solar absorptivity α . Two or more coatings can be combined in an appropriate pattern to obtain some desired average surface values of α and ϵ ; e.g., a checkerboard pattern of white paint and polished metal.

- e. Louvers. Louvers provide a simple reliable method of active temperature control by varying the effective emittance of a spacecraft radiator with temperature. The most commonly used configuration consists of a series of polished aluminum blades arranged in venetian blind fashion over a high emittance radiator. Each blade is attached to a shaft supported at the ends by bearings. A bimetallic spring attached to the shaft of each blade varies the blade angle with temperature and changes the exposure to space of the radiator surface. Other mechanical configurations and temperature actuators have been used.
- f. Heat Pipes. In its basic form, a heat pipe is a very simple, self-contained device. The walls of an enclosure are lined with a "wicking" material saturated with a "working fluid." Heat is then conducted from a source such as electronics through the heat pipe walls and into the working fluid. The additional heat causes the evaporation of working fluid which then travels by the induced pressure gradient to a colder portion of the pipe. The vapor carries with it the latent heat of vaporization which is released as the vapor condenses in a colder portion of the pipe. The heat is then conducted through the wall to a heat rejection system such as a radiator. Meanwhile, the condensed fluid is pumped back to the hot end by the capillary action of the wicking material to complete the cycle. In some applications, heat pipes exhibit an effective thermal conductivity that exceeds solid copper by orders of magnitude. The heat pipe in this basic form is useful in "isothermalizing" spacecraft structures such as equipment shelves and telescope optical tubes by conducting thermal energy efficiently from hotter to colder regions. This basic heat pipe has a fixed, high conductance and must, therefore, be designed for given heat source and sink conditions. Deviation from these conditions results in the overcooling or overheating of the heat source. A heat pipe, however, that is designed to vary its effective conductance in response to changing conditions is the "variable conductance heat pipe" which can be used to control the source at a near constant temperature. Feedback control provides greater sensitivity than that available with standard variable conductance techniques. Heat pipes are designed to provide heat transfer in one direction only (thermal diode).

8.2 INPUT DATA

The inputs for the thermal control algorithm are shown in Table 8-1.

Table 8-1. Input Data Requirements

Source	Name	Required (R) or Optional (O)	Default Value
User	Orbit perigee	R	
User	Orbit apogee	R	
User	Orbit inclination	O	0 rad
User	Orbit beta angle	O	0 rad
User	Orientation (solar or earth)	R	
User	Spin axis orientation (required for spinning vehicles)	R	
S&C	Configuration	R	
VS	Vehicle shape	R	

These inputs are used to calculate the Thermal Control Subsystem component description. The orbit perigee and apogee are used to determine whether the spacecraft is in low earth orbit, near synchronous, or in a highly elliptical orbit. Inclination and beta angle are used to determine eclipse conditions. Vehicle configuration, vehicle dimensions and stabilization and orientation information are necessary to determine a characteristic dimension for heat pipe sizing, and to determine available area for heat rejection. Also, stabilization and orientation information is necessary to determine environmental heat loads.

Minimum and maximum power for each component selected from the equipment data base is required to determine heat rejection

requirements. Component temperature limits provided in the data base data will determine the allowable temperatures for each particular subsystem. If component temperatures are not specified, the default temperatures -17°C and $+54^{\circ}\text{C}$ will be used except that battery default temperatures are $+6^{\circ}$ and $+32^{\circ}\text{C}$.

8.3 DESIGN EQUATIONS

Tables 8-2 and 8-3 show preliminary calculations and inputs of constant values for the program. A vehicle characteristic dimension is determined based on the spacecraft geometry and size. This characteristic dimension should be in meters. The other program inputs are in engineering units. Consequently, a conversion must be made at the end of the algorithm in order to obtain the current metric dimensions.

The spacecraft orbit, orientation, stabilization, etc., are used to select the Thermal Control components. The approach used in the algorithm is to group the various spacecraft into low earth, synchronous, and intermediate altitude orbits. The spacecraft are further classified as to their orientation (earth, solar, inertial, etc.) and as to their stabilization approach (three-axis, dual spin, etc.). Orbit inclination and eclipse conditions are also considered. Once the classification has been completed, the appropriate equations are used to size the components.

Thermal control of batteries is handled slightly differently from the rest of the spacecraft subsystems. The equations for battery thermal control are presented at the end of Section 8.3.

All the sizing equations are in engineering units, except for the heat pipe equations. The conversion factors necessary to characterize the Thermal Control Subsystem in metric units are given in Table 8-4.

Table 8-2. Vehicle Configuration Characteristic Length

Configuration	Characteristic Length
Rectangular	Middle length of vehicle's three dimensions
Cylindrical	0.75 of the axial length
Spherical	π times the sphere radius

Table 8-3. Internally Stored Data

Symbol	Description	Value (Source)	Units
Q_{\max}	Maximum internal power dissipation	(data base)	Btu/hr
Q_{\min}	Minimum internal power dissipation	(data base)	Btu/hr
σ	Stefan-Boltzmann constant	0.1714×10^{-8}	Btu/hr-ft ² - R ⁴
A	Radiator area		ft ²
Q_s	Solar constant	442	Btu/hr-ft ²
Emiss.	Earth emission	60	Btu/hr-ft ²
Alb.	Albedo of earth	155	Btu/hr-ft ²
K	Factor	1.0	N. D.
T_{\max}	Maximum temperature	(data base)	Rankine (R)
T_{\min}	Minimum temperature	(data base)	Rankine (R)
α / ϵ	Conventional radiator emissivity/absorptivity	0.30/0.75	
α / ϵ	OSR radiator emissivity/absorptivity	0.08/0.73	
L	Characteristic length	(data base)	Meters

Table 8-4. . Metric Conversion Factors

To convert	Into	Multiply by
Btu/hr	watts	0.2931
sq. ft	sq m	0.0929
°F	°C	$C = (F - 32) / 1.8$
R	F	$F = R - 460$
lb	kg	0.454

8.3.1 Orbits of Synchronous Altitude or Greater

8.3.1.1 Solar Inertially Oriented Vehicle

a. Spinning Vehicle

1. Spin Axis Pointed Toward Sun

$$\text{Conventional Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau^4_{\max}}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau^4_{\min} - Q_{\min} \right\}$$

2. Spin Axis Normal to Sun

$$\text{Conventional Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau^4_{\max}}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau^4_{\min} - Q_{\min} \right\}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

b. Three-Axis Stabilized Vehicle

$$\text{Conventional Radiator Area} = \frac{2Q_{\max}}{\sigma \epsilon \tau^4_{\max}}$$

$$\text{Heater Power} = 1.25 \left\{ \frac{\sigma \epsilon A \tau^4_{\min}}{2} - Q_{\min} \right\}$$

$$\text{Diode Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41} \quad (2 \text{ req'd})$$

8.3.1.2 Earth Oriented Vehicle (Near Equatorial)

a. Three-Axis Stabilized Vehicle

$$\text{Conventional Radiator Area} = \frac{2 Q_{\max}}{\sigma \epsilon \tau^4_{\max}}$$

$$\text{Heater Power} = 1.25 \left\{ \frac{\sigma \epsilon A \tau^4_{\min}}{2} - Q_{\min} \right\}$$

$$\text{Diode Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41} \quad (2 \text{ req'd})$$

b. Yaw Spin Stabilized Vehicle

$$\text{Optical Surface Reflector (OSR) Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - \frac{Q_s \alpha}{\pi}}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon \tau_{\min}^4 - Q_{\min} \right\}$$

c. Dual Spin or Normal Spin Vehicle

$$\text{Conventional Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon \tau_{\min}^4 - Q_{\min} \right\}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

3.2 Orbits Less than 926 Kilometers (500 nm)

3.2.1 Orbit Inclination is Less Than 30°

a. Solar Inertially Oriented Vehicle

1. Three-Axis Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - (E_{\text{miss}}) \epsilon - (Alb) \alpha}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon \tau_{\min}^4 - Q_{\min} \right\}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

2. Spin Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - 0.5[(E_{\text{miss}}) \epsilon + (Alb) \alpha]}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon \tau_{\min}^4 - Q_{\min} \right\}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

b. Earth Oriented Vehicle

1. Three-Axis Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - \alpha Q_s}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau_{\min}^4 - Q_{\min} \right\}$$

$$\text{Phase Change Material (PCM) Mass} = \frac{0.26 Q_s \alpha AK}{40}$$

$$\text{Isothermalizer Heat Pipe} = \frac{Q_{\max} L}{3.41}$$

2. Spin Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\alpha \epsilon \pi_{\max}^4 - \frac{(Emiss) \epsilon \cdot [Q_s + (Alb) \alpha]}{\pi}}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau_{\min}^4 - Q_{\min} - \frac{(Emiss) \epsilon}{\pi} \right\}$$

8.3.2.2 Orbit Inclination is Greater Than 30°

a. Solar Inertially Oriented Vehicle

1. Three-Axis Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - (Emiss) \epsilon - (Alb) \alpha}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau_{\min}^4 - Q_{\min} - (Emiss) \epsilon \right\}$$

$$\text{PCM Mass} = \frac{0.26 \alpha (Alb) AK}{40}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

2. Spin Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - (\text{Emiss}) \epsilon - (\text{Alb}) \alpha}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau_{\min}^4 - Q_{\min} - (\text{Emiss}) \epsilon \right\}$$

$$\text{PCM Mass} = \frac{0.26 \alpha (\text{Alb}) AK}{40}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

b. Earth Oriented Vehicle

1. Three-Axis Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - \alpha Q_s}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau_{\min}^4 - Q_{\min} \right\}$$

$$\text{PCM} = \frac{0.26 Q_s \alpha AK}{40}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

2. Spin Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - \frac{(\text{Emiss}) \epsilon}{\pi} - \frac{[Q_s + (\text{Alb}) \alpha]}{\pi}}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon \tau_{\min}^4 - Q_{\min} - \frac{(\text{Emiss}) \epsilon}{\pi} \right\}$$

8.3.3 All Other Orbits

8.3.3.1 Solar Inertially Oriented Vehicle

a. Three-Axis Stabilized Vehicle

$$\text{Conventional Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - (\text{Emiss}) \epsilon}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau_{\min}^4 - Q_{\min} - (\text{Emiss}) \epsilon \right\}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

b. Spin Stabilized Vehicle

$$\text{Conventional Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - (\text{Emiss}) \epsilon}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau_{\min}^4 - Q_{\min} - (\text{Emiss}) \epsilon \right\}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

8.3.3.2 Earth Oriented Vehicle

a. Three-Axis Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - Q_s \alpha}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau_{\min}^4 - Q_{\min} \right\}$$

$$\text{PCM} = \frac{0.26 \alpha Q_s AK}{40}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

Spin Stabilized Vehicle

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon \tau_{\max}^4 - Q_s \alpha}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A \tau_{\min}^4 - Q_{\min} \right\}$$

$$\text{PCM} = \frac{0.26 \alpha Q_s AK}{40}$$

$$\text{Heat Pipe Capacitance} = \frac{Q_{\max} L}{3.41}$$

8.3.4 Battery Thermal Control

$$\text{OSR Radiator Area} = \frac{Q_{\max}}{\sigma \epsilon (\tau_{\max} - 30)^4 - Q_s \alpha}$$

$$\text{Heater Power} = 1.25 \left\{ \sigma \epsilon A (\tau_{\min} + \Delta\tau)^4 - Q_{\min} \right\}$$

$$\Delta\tau = 1.7 (Q_{\min} + \text{Heater Power})$$

$$\text{Variable Conductance Heat Pipe} = \frac{Q_{\max} L}{3.41}$$

8.4 DESIGN LOGIC

The logic within the Design Algorithm is a set of logical tests which, for the specified conditions, point the way to the proper components and the equations necessary to describe the physical attributes of the components. Table 8-5 presents the logical tests which are performed at each branch or tier in the logic tree. Thus, Table 8-5 indicates the first test is based on orbit altitude; there are three ranges of altitudes, and for each altitude range there are four or five further tests to be performed.

Table 8-5. Thermal Control Logic

Logic Tier	Test	Options
1	Orbit Altitude	Earth synchronous altitude or greater
2	Orientation	Solar Earth
3	S&C Type	Dual Spin or normal spin Yaw spin Three-axis stabilized
4	Spin Orientation	Spin axis pointed toward sun Spin axis normal to sun

Logic Tier	Test	Options
1	Orbit Altitude	Less than earth synchronous and greater than 926 km (500 nm)
2	Orientation	Solar Earth
3	S&C Type	Spinning vehicle Three-axis stabilized

Logic Tier	Test	Options
1	Orbit Altitude	Less than 926 km (500 nm)
2	Inclination	Less than or equal to 30 degrees Greater than 30 degrees
3	Orientation	Solar Earth
4	S&C Type	Spinning vehicle Three-axis stabilized

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9. VEHICLE SIZING

9.1 GENERAL

9.1.1 Functional Description

The Vehicle Sizing Model determines the satellite structural weight, the total weight, the satellite volume, dimensions, center of gravity locations and the satellite inertial characteristics.

The model is able to accept inputs describing:

- a. Mission Equipment
- b. External Equipment
- c. Solar Arrays

A typical satellite configuration is depicted in Figure 9-1.

The mission equipment is limited to two items in number and can be mounted on either the forward (+ x) or aft (- y) end of the vehicle.

Up to nine external equipments can be specified. The external equipment can be located anywhere on the surface of the vehicle. Clearly, the external equipment can include mission equipment descriptions.

The solar arrays can be mounted either on the vehicle body or on oriented paddles. The body-mounted solar arrays can be mounted on the forward (+ x) end, center, or aft (- x) end of the vehicle surface. Paddle-mounted solar arrays are assumed to be mounted in the x - y plane parallel to the y axis and attached to either the front (+ x) end, center, or aft (- x) end of the vehicle.

9.1.2 Vehicle Configurations

The Vehicle Sizing Model has the ability to design any of three general vehicle configurations:

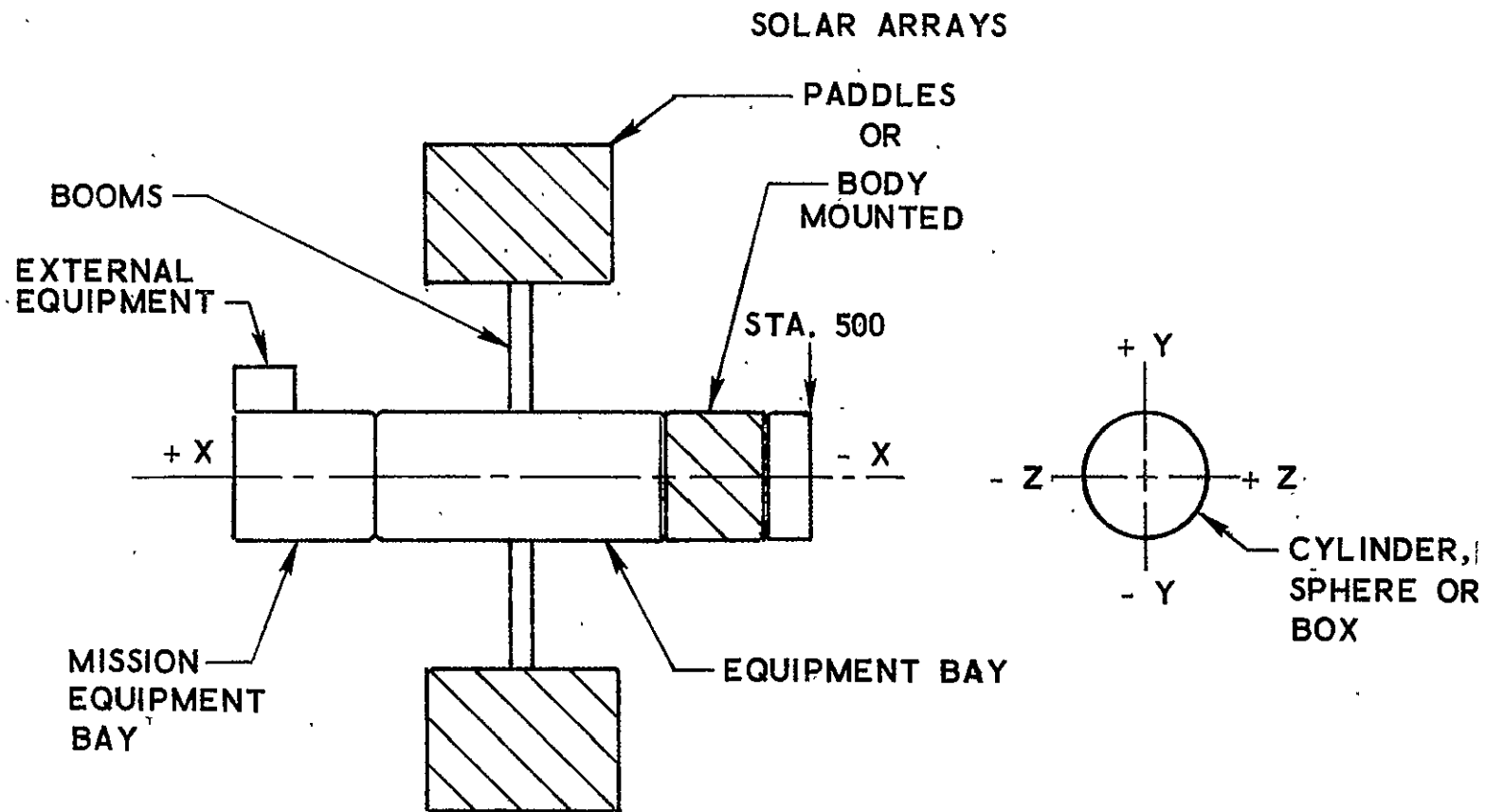


Figure 9-1. Typical Spacecraft Configuration

- a. Cylindrical
- b. Box (square end)
- c. Spherical

The vehicle configuration is always tested against the Stabilization and Control Subsystem configuration to prevent a spinning, box-shaped vehicle from being designed. This test is summarized in Table 9-1.

Table 9-1. Vehicle Shape Compatibility

S&C Configuration	Cylinder	Sphere	Box
Spinning	Yes	Yes	No
3-Axis	Yes	Yes	Yes

Legend:

- Yes - Compatible
- No - Incompatible

9.1.3 Equipment Types

Equipment weights which are determined by the Vehicle Sizing Model include the following:

- a. Equipment Bay Structure
- b. Booms, Extensions, and Mechanisms
- c. Wiring Harness
- d. Thermal Control Subsystem
- e. Adapter

9.2 INPUT DATA

The information required from the user to design the vehicle is identified in Table 9-2. The table indicates which data are used and whether the data are required or optional. If optional, the table indicates what default value is used in the event the data are not supplied.

Table 9-2. Input Data Supplied by User

Symbol	Name	Required (R) or Optional (O)	Default Value
EQPF	Equipment packing factor	O	2
MB12SH	Mission equipment bay shape (1 means cylinder, 2 means box)	O	1
EQM1WT	Mission equipment bay #1 weight (lb)	O	350
EQM1XL	Mission equipment bay #1 length (in.)	O	40
EQM1YL	Mission equipment bay #1 width (in.)	O	40
EQM1ZL	Mission equipment bay #1 height (in.)	O	40
EQM2WT	Mission equipment bay #2 weight (lb)	O	0
EQM2XL	Mission equipment bay #2 length (in.)	O	40
EQM2YL	Mission equipment bay #2 width (in.)	O	40
EQM2ZL	Mission equipment bay #2 height (in.)	O	40
ISBOFG	Solar array boom orientation (0 means not oriented, 1 means oriented)	O	0
NUMEEQ	Number of external equipments (maximum of nine)	O	0
EEQWT(i)	External equipment #i weight (lb)	O	0
EEQXL(i)	External equipment #i volume (ft ³)	O	0
EM1YCG	Miss equip #1 c.g. "y" location (in.)	O	0
EM1ZCG	Miss equip #1 c.g. "z" location (in.)	O	0
EM2YCG	Miss equip #2 c.g. "y" location (in.)	O	0
EM2ZEG	Miss equip #2 c.g. "z" location (in.)	O	0
CGEEX(i)	Location of ith external equipment (1 means front, 2 means center, 3 means aft)	O	2

Table 9-2. Input Data Supplied by User (Continued)

Symbol	Name	Required (R) or Optional (O)	Default Value
EELDC(i)	Location of ith external equipment (1 means right, 2 means left, 3 means top, 4 means bottom when viewed from aft end)	O	2
XCGSA1	Location of solar paddles (1 means front, 2 means center, 3 means aft)	O	1
XCGSA3	Location of body mounted solar array (1 means front, 2 means center, 3 means aft)	O	1
DIAMAX	Maximum satellite diameter (in.)	O	180
SLGMAX	Maximum satellite length (in.)	O	720
SWTMAX	Maximum satellite launch weight (lb)	O	65,000

The Vehicle Sizing Model obtains the weight and volume from the data base for each component selected. The Reliability model indicates the total number of each component type. The Stabilization and Control Subsystem indicates whether the vehicle is spinning or three-axis stabilized. Total solar array area is supplied by the Electrical Power Subsystem model.

9.3 DESIGN EQUATIONS

The empirical weight equations used in the model were developed by correlating actual satellite data with a theoretical model using a regression analysis computer routine. Numerous parameters were compared to determine their relative effect on weight, and those which had a low influence were deleted from the equations for simplification.

The data shown in Figure 9-2 was used to produce the following structural equations:

$$\text{Structural Weight} = K \left[(\text{EQWT})^{0.9} (\text{L/D})^{0.24} \right] 1.096 \quad (9-1)$$

where:

- K = Density Coefficient
 - = 0.218 for satellites with sidewalls (i. e., with body mounted solar arrays)
 - = 0.129 for satellites with paddle mounted solar arrays, which do not require the large body area needed for body mounted arrays
- EQWT = Equipment weight carried by the structure
- L/D = Length/diameter ratio of the structural shape (i. e., long, slender structures weigh more than short, wide structural shapes.)

The electrical harness weight, the structural thermal protection, and the equipment boom weight were obtained in a similar manner.

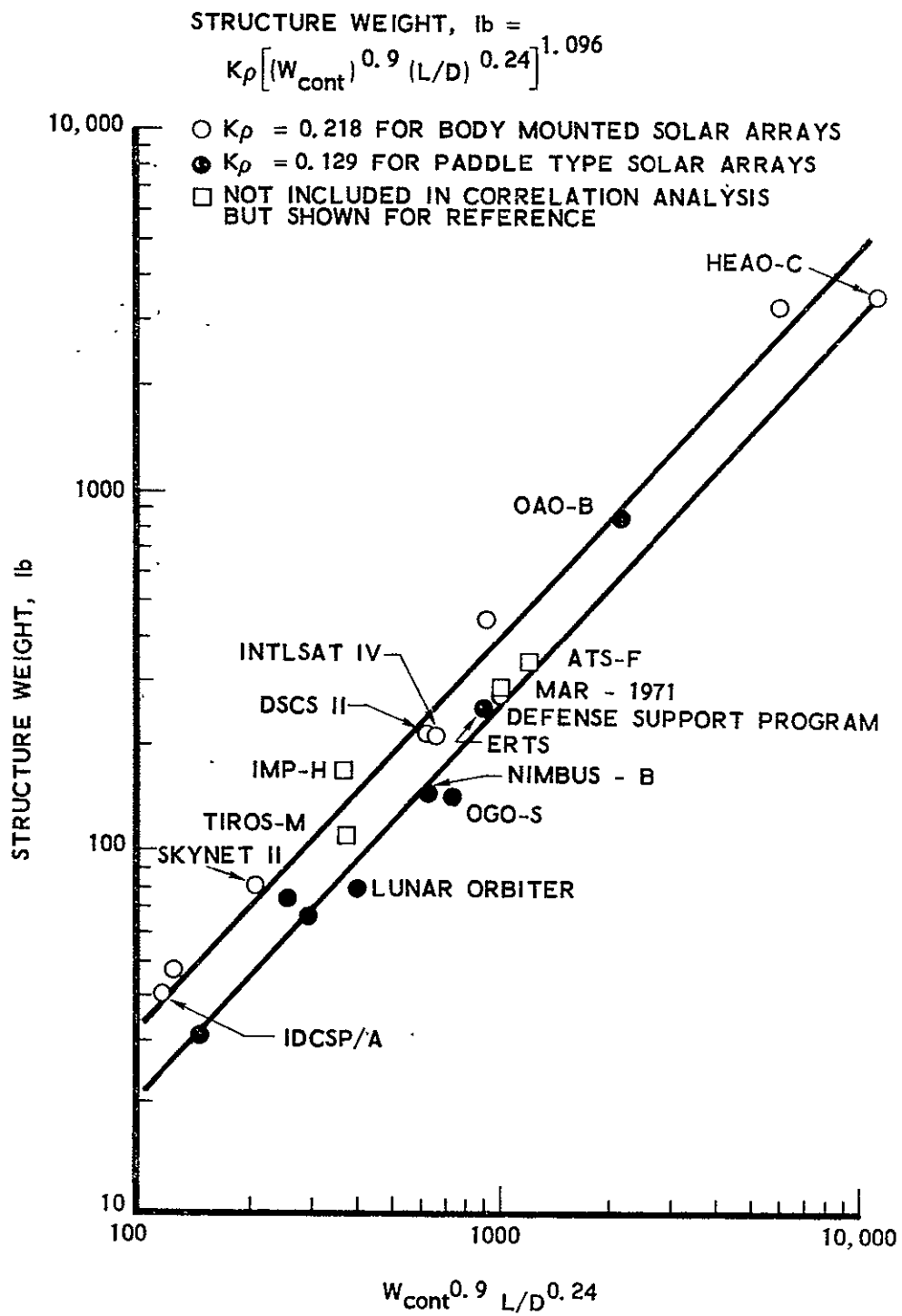


Figure 9-2. Structure Weight Correlation

9.3.1 Mission Equipment

The mission equipment weight is supplied by the user:

$$\begin{aligned} \text{EQM1WT} &= \text{Mission Equipment \#1 Weight (lb)} \\ \text{EQM2WT} &= \text{Mission Equipment \#2 Weight (lb)} \\ \text{EQMWT} &= \text{EQM1WT} + \text{EQM2WT} \end{aligned} \quad (9-2)$$

The dimensions of the mission equipment bays are also provided by the user:

$$\begin{aligned} \text{EQM1XL} &= \text{Mission Equipment Bay \#1 Length (in.)} \\ \text{EQM1YL} &= \text{Mission Equipment Bay \#1 Width (in.)} \\ \text{EQM1ZL} &= \text{Mission Equipment Bay \#1 Height (in.)} \\ \text{EQM2XL} &= \text{Mission Equipment Bay \#2 Length (in.)} \\ \text{EQM2YL} &= \text{Mission Equipment Bay \#2 Width (in.)} \\ \text{EQM2ZL} &= \text{Mission Equipment Bay \#2 Height (in.)} \end{aligned}$$

a. Mission Equipment Volume. It is necessary to know the total mission equipment bay volume to determine the electrical harness weight.

1. If the mission equipment bay is a cylinder, volumes of equipment bays #1 and #2 are:

$$\begin{aligned} \text{EQM1VL} &= 0.785 \times \text{EQM1YL}^{2.0} \times \text{EQM1XL} \\ \text{EQM2VL} &= 0.785 \times \text{EQM2YL}^{2.0} \times \text{EQM2XL} \end{aligned} \quad \left. \vphantom{\begin{aligned} \text{EQM1VL} \\ \text{EQM2VL} \end{aligned}} \right\} (9-3)$$

and the total volume is:

$$\text{EQMVOL} = \text{EQM1VL} + \text{EQM2VL} \quad (9-4)$$

2. If the mission equipment bay is a box, volumes of equipment bays #1 and #2 are:

$$\begin{aligned} \text{EQM1VL} &= \text{EQM1XL} \times \text{EQM1YL} \times \text{EQM1ZL} \\ \text{EQM2VL} &= \text{EQM2XL} \times \text{EQM2YL} \times \text{EQM2ZL} \end{aligned} \quad \left. \vphantom{\begin{aligned} \text{EQM1VL} \\ \text{EQM2VL} \end{aligned}} \right\} (9-5)$$

and the total volume is:

$$\text{EQMVOL} = \text{EQM1VL} + \text{EQM2VL} \quad (9-6)$$

- b. Mission Equipment Structure. If the mission equipment bay structure is given as input, this section should be skipped and the input data used. Otherwise, with the mission equipment weight (EQM1WT), the length of the mission equipment bay (EQM1XL), and the diameter (EQM1YL) known, the weight of the mission equipment bays can be determined as shown below.

1. If the mission equipment bay is a cylinder:

$$\left. \begin{aligned} \text{EQM1S} &= 0.218 \times [(\text{EQM1WT})^{0.9} \times (\text{EQM1XL}/\text{EQM1YL})^{0.24}]^{1.096} \\ \text{EQM2S} &= 0.218 \times [(\text{EQM2WT})^{0.9} \times (\text{EQM2XL}/\text{EQM2YL})^{0.24}]^{1.096} \end{aligned} \right\} (9-7)$$

2. If bays 1 and 2 are boxes, an additional step is necessary to obtain the length to diameter ratio. The longest side is selected as the length (EM1LNG) regardless of the direction (x, y, or z). Then the diagonal of the other two sides (EM1DIA) is used as the diameter.

Select: Longest of EQM1XL or EQM1YL or EQM1ZL = EM1LNG

Remaining Terms = EM1HT and EM1WD

Then: $\text{EM1DIA} = (\text{EM1HT}^2 + \text{EM1WD}^2)^{0.5}$ (9-8)
(Do same for second bay)

$$\left. \begin{aligned} \text{EQM1S} &= 0.218 \times [(\text{EQM1WT})^{0.9} \times (\text{EM1LNG}/\text{EM1DIA})^{0.24}]^{1.096} \\ \text{EQM2S} &= 0.218 \times [(\text{EQM2WT})^{0.9} \times (\text{EM2LNG}/\text{EM2DIA})^{0.24}]^{1.096} \end{aligned} \right\} (9-9)$$

3. A ten-percent factor is carried to account for mounting equipment. Therefore, the final mission equipment bay structure weight is:

$$\left. \begin{aligned} \text{EQM1ST} &= \text{EQM1S} + (0.10 \times \text{EQM1WT}) \\ \text{EQM2ST} &= \text{EQM2S} + (0.10 \times \text{EQM2WT}) \end{aligned} \right\} (9-10)$$

9.3.2 Solar Arrays

The solar array area (SAAREA) required to produce the necessary power (watts) must be read from the input data.

- a. Oriented Paddles. Assume two flexible, oriented paddles of equal areas:

$$\begin{aligned} \text{First paddle area (ft}^2\text{)} &= \text{SAREA1} \\ &= \text{SAAREA}/2 \\ \text{Second paddle area (ft}^2\text{)} &= \text{SAREA2} \\ &= \text{SAAREA}/2 \end{aligned} \quad \left. \vphantom{\begin{aligned} \text{First paddle area (ft}^2\text{)} \\ \text{Second paddle area (ft}^2\text{)} \end{aligned}} \right\} (9-11)$$

The flexible arrays are assumed to be 2.44 m (8 ft) wide. Therefore, the length of each paddle is:

$$\begin{aligned} \text{Paddle length} &= \text{SA1YL (in.)} \\ &= 0.75 \text{ SAAREA} \end{aligned} \quad \left. \vphantom{\begin{aligned} \text{Paddle length} \\ \text{Paddle length} \end{aligned}} \right\} (9-12)$$

This assumes the paddle is extended in the y direction. Then, assume the 2.44 m (96 in.) dimension is in the x direction and a 2.54 cm (1 in.) thickness:

$$\begin{aligned} \text{SA1XL} &= 96.0 \\ \text{SA1ZL} &= 1.0 \end{aligned} \quad \left. \vphantom{\begin{aligned} \text{SA1XL} \\ \text{SA1ZL} \end{aligned}} \right\} (9-13)$$

- b. Body Mounted Arrays (SA3WT). The solar arrays are mounted on the equipment bay sidewall and are unoriented. If the equipment bay is a cylinder, the available solar array area is the total equipment bay sidewall area if the vehicle is spinning and one-half the equipment bay sidewall area if the vehicle is not spinning.* If the equipment bay is a square-ended box, the minimum available area for power generation is one side of the box. In both cases, a check is made to see if the solar array length (SA3XL) exceeds the equipment bay length (EQBLG). If this happens, the model indicates that additional array area is required.

If the equipment bay is a sphere, only the projected area is available for power generation. If the vehicle diameter required for the solar arrays exceeds the diameter originally calculated to contain the equipment volume, then the model again indicates that additional array area is required.

- c. Solar Array Booms. The solar arrays are always placed in the x-y plane and the extension from the body (on the y axis) is assumed to be 61.0 cm (24 in.). Therefore:

* The solar array area calculated by the Electrical Power Subsystem model takes into account the vehicle shape and whether the vehicle is spinning.

$$\text{SABOLG} = 24.0 \quad (9-14)$$

and the boom weight is:

$$\begin{aligned} \text{SABOOM} &= 5.0 + (0.213 \times \text{SABOLG} \times 2.0) \\ &= 6.89 \text{ kg (15.2 lb)} \end{aligned} \quad (9-15)$$

If the boom is oriented, a drive mechanism (SADRIV) must be added. Its weight is a function of the solar array weights (SA1WT and SA2WT):

$$\text{SADRIV} = 0.166 \times (\text{SA1WT} + \text{SA2WT}) \quad (9-16)$$

Finally, the system weight is the sum of the boom and the drive:

$$\text{SABMWT} = \text{SABOOM} + \text{SADRIV} \quad (9-17)$$

9.3.3 Equipment Bay

Several preliminary steps are necessary in order to determine the equipment bay shape and size.

- a. Subsystem Weights. The weights and volumes of the various pieces of equipment which will be carried in the equipment bay must be read from the data base. The subsystem total weight is the sum of the various component weights.

Stabilization + Control Weight (STABWT)

$$\text{STABWT} = \text{STA1WT} + \text{STA2WT} + \dots \quad (9-18)$$

Auxiliary Propulsion Inerts Weight (ACINWT)

$$\text{ACINWT} = \text{ACI1WT} + \text{ACI2WT} \dots \quad (9-19)$$

Auxiliary Propulsion Propellant Weight (ACSWP)

$$\text{ACSWP} = \text{ACS1WP} + \text{ACS2WP} + \dots \quad (9-20)$$

Communication Weight (COMWT)

$$\text{COMWT} = \text{COM1WT} + \text{COM2WT} + \dots \quad (9-21)$$

Data Processing Weight (DATAWT)

$$\text{DATAWT} = \text{DAT1WT} + \text{DAT2WT} + \dots \quad (9-22)$$

Electrical Power Weight (ELPWT)

$$\text{ELPWT} = \text{ELP1WT} + \text{ELP2WT} + \dots \quad (9-23)$$

Thermal Control Weight (TCWT)

$$\text{TCWT} = 0.025 \times \text{EQWT} \quad (9-24)$$

Equipment Weight (EQWT)

$$\text{EQWT1} = \text{STABWT} + \text{COMWT} + \text{DATAWT} + \text{ELPWT} + \text{TCWT} \quad (9-25)$$

$$\text{EQWT} = \text{EQWT1} + \text{ACINWT} + \text{ACSWP} \quad (9-25a)$$

- b. Subsystem Volumes. The subsystem volumes are obtained by a summation of the component volumes which are read from the data base.

$$\text{STAVOL} = \text{STA1VL} + \text{STA2VL} + \dots \quad (9-26)$$

$$\text{ACSVOL} = \text{AC1VL} + \text{AC2VL} + \dots \quad (9-27)$$

$$\text{COMVOL} = \text{COM1VL} + \text{COM2VL} + \dots \quad (9-28)$$

$$\text{DATVOL} = \text{DAT1VL} + \text{DAT2VL} + \dots \quad (9-29)$$

$$\text{ELPVOL} = \text{ELP1VL} + \text{ELP2VL} + \dots * \quad (9-30)$$

$$\text{EQVOL} = \text{STAVOL} + \text{ACSVOL} + \text{COMVOL} + \text{DATVOL} + \text{ELPVOL} \quad (9-31)$$

- c. Equipment Bay Volume. The actual volume required in the equipment bay itself must be determined. This volume must be greater than the sum of the component volumes because the shapes of the equipment packages makes it difficult to stack them efficiently and because it is not possible to put a flat-sided box closely against a curved sidewall. In addition, room must be left for cable runs and access to the equipment. The factor used to obtain this additional volume is the packing factor, and a value of 2.0 is used as a default value in this program. The factor of 2.0 is based upon a very tightly packed satellite. If this is not acceptable or desirable because of thermal control problems, then the value should be raised as suggested in Figure 9-3.

Equipment Bay Volume (EQBVOL) (ft³)

$$\text{EQBVOL} = \text{EQVOL} \times \text{EQPF} \quad (9-32)$$

where

$$\text{EQPF} = \text{Equipment packing factor}$$

- d. Equipment Bay Dimensions. The input data is read to determine the desired shape of the equipment bay. In addition, the maximum allowable diameter (or diagonal if a square-ended box) must be read from the input data. With the total volume, the shape,

* Do not include Solar Array Volume in Electric Power Volume (ELPVOL) since it is not carried in the equipment bay.

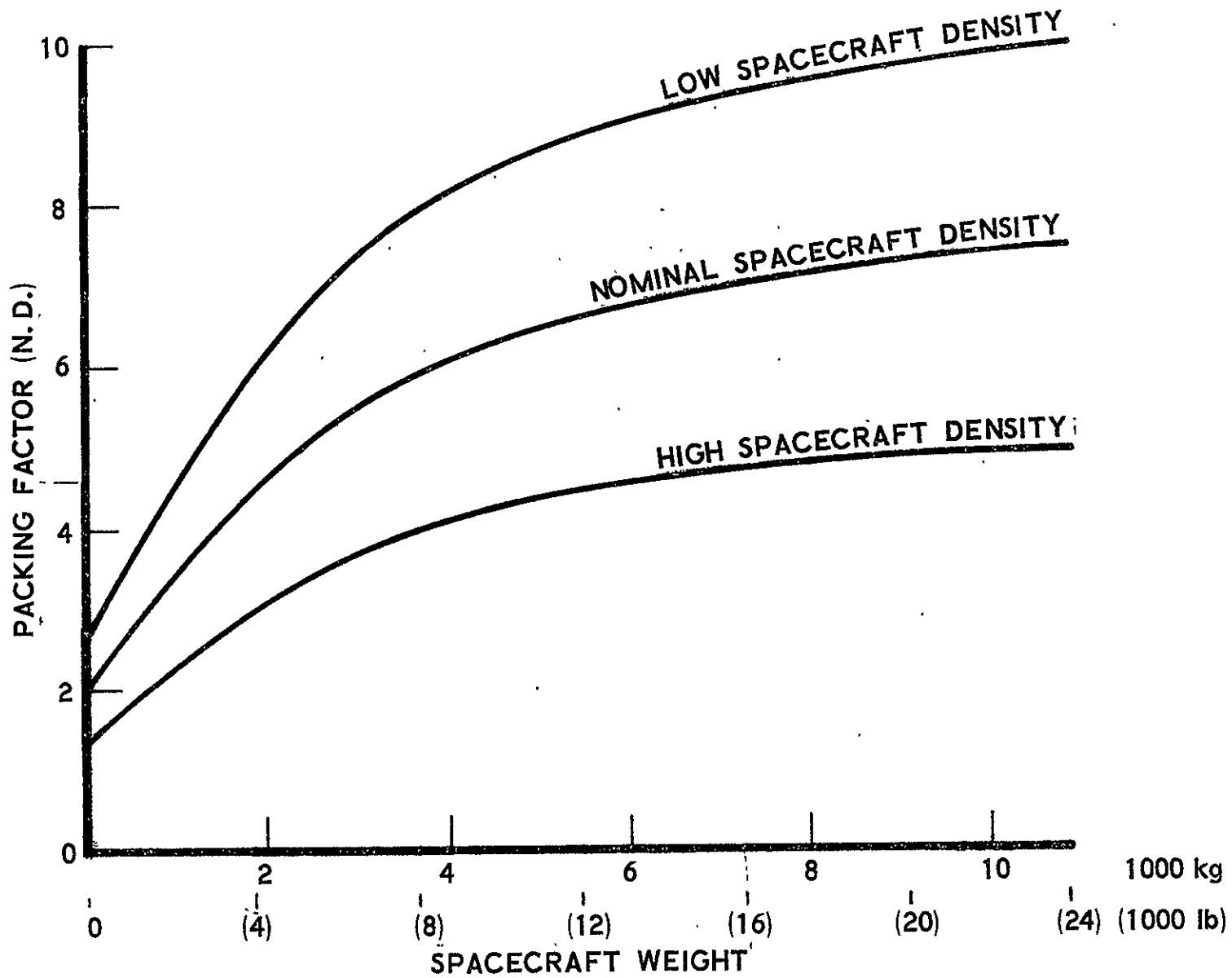


Figure 9-3. Representative Spacecraft Packing Factors

and the maximum diameter available, the volume determination can be made as presented in the following material.

It should be noted that a length-to-diameter ratio of one will be used until the maximum diameter is reached. Then the diameter will be fixed and the length extended until the required volume is obtained.

Note that if the satellite attitude control requires that the satellite be a spinning body, then the shape must be a cylinder and solar arrays are body-mounted.

The maximum allowable diameter (DIAMAX) is read from the input list. Note that the default value for DIAMAX is 4.57 m (180 in.).

Since the equipment bay volume (EQBVOL) has been determined, the required satellite diameter (SATDAM) and the equipment bay length (EQBLG) can be determined as shown below.

1. Cylindrically Shaped Vehicle (Fig. 9-4)

$$\text{SATDAM} = [(\text{EQBVOL} \times 1728) / 0.785]^{0.333} \quad (9-33)$$

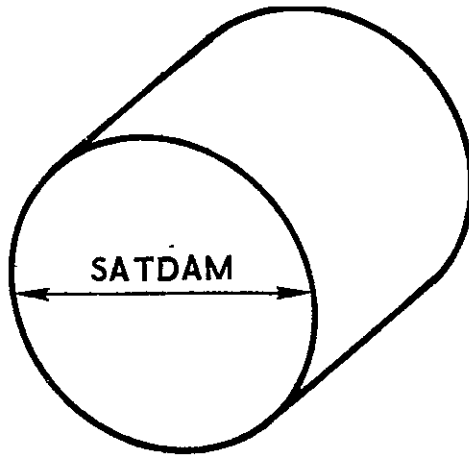


Figure 9-4. Cylindrically Shaped Vehicle

$$EQBLG = SATDAM \quad (9-34)$$

If SATDAM exceeds DIAMAX, change SATDAM to DIAMAX

$$EQBLG = (EQBVOL \times 1728) / (0.785 \times SATDAM^2) \quad (9-35)$$

2. Box Shaped Vehicle (Fig. 9-5)

For Length/Diagonal = 1.0 (End = Square)

$$EQBLG = (2 \times EQBVOL \times 1728)^{0.333} \quad (9-36)$$

$$EQBDIA = EQBLG \quad (9-37)$$

If EQBDIA exceeds DIAMAX, change EQBDIA to DIAMAX and

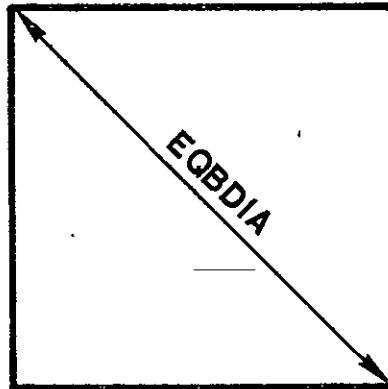


Figure 9-5. Box Shaped Vehi

$$EQBSID = EQBDIA / 1.414 \text{ and} \quad (9-38)$$

$$EQBLG = (EQBVOL \times 1728) / EQBSID^{2.0} \quad (9-39)$$

3. Spherically Shaped Vehicle (Fig. 9-6)

$$\text{SATDAM} = [(\text{EQBVOL} \times 1728)/0.5235]^{0.333} \quad (9-40)$$

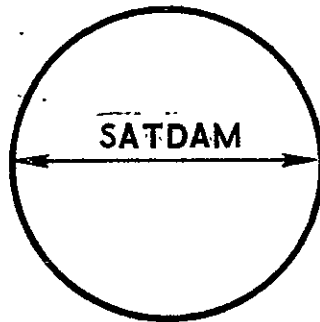


Figure 9-6. Spherically Shaped Vehicle

If SATDAM exceeds DIAMAX, the model aborts this configuration and goes on to the next case.

The satellite length (inches) is the summation of the equipment baylength (EQBLG) plus the mission bay lengths (EM1XLG and EM2XLG).

$$\text{SATLG} = \text{EQBLG} + \text{EQM1XL} + \text{EQM2XL} \quad (9-41)$$

The satellite length is finally compared with the maximum allowable length (SLGMAX) specified by the user and the design is aborted if SATLG is greater than SLGMAX.

9.3.4 Equipment Bay Structural Weight

With the equipment weight (EQWT) and the length (EQBLG) to diameter (SATDAM) known, the equipment bay structure can be calculated. Note that different multipliers are used when body-mounted solar arrays are used instead of paddle-mounted solar arrays.

a. Body-Mounted Solar Arrays

$$EQBST = 0.218 \times [(EQWT)^{0.9} \times (EQBLG/SATDAM)^{0.24}]^{1.096} \quad (9-42)$$

b. Paddle-Mounted Solar Arrays

$$EQBST = 0.129 \times [(EQWT)^{0.9} \times (EQBLG/SATDAM)^{0.24}]^{1.096} \quad (9-43)$$

A ten-percent factor is added to account for the equipment mounts and supports. Therefore, the final equipment structural weight is:

$$EQBSTR = EQBST + (0.10 \times EQWT) \quad (9-44)$$

9.3.5 Total Equipment Bay Weight

The total equipment bay weight (EQBWT) is the sum of the equipment weight (EQWT) and the structural weight (EQBSTR).

$$EQBWT = EQWT + EQBSTR \quad (9-45)$$

9.3.6 Wiring Harness Weight

The electrical harness weight (HARNWT) is a function of the power consuming equipment weight (not including batteries) and the equipment bay volumes (Ref. 9-1).

$$HARNWT = 0.013 \times (EQWT - ACSWP + EEQTWT + EQMWT)^{1.31} \times (EQBVOL + EQMVOL + EEQVOL)^{0.16} \quad (9-46)$$

9.3.7 Thermal Control Weight

The structural thermal protection system (STTPS) is a function of the equipment bay structural weight (EQBSTR).

$$\text{STTPS} = 0.025 \times \text{EQBSTR} \quad (9-47)$$

9.3.8 Spacecraft Gross Weight

The subsystem weights are listed in two parts. The external equipment weights are shown first as SUBWT1. The equipment bay subsystems are then added to form SUBWT2.

$$\begin{aligned} \text{SUBWT1} = & \text{EQMWT} + \text{EQM1ST} + \text{EQM2ST} + \text{SA1WT} + \text{SA2WT} + \text{SA3WT} + \\ & \text{EEQ1WT} \dots + \text{EEQ9WT} + \text{SABMWT} \end{aligned} \quad (9-48)$$

$$\text{SUBWT2} = \text{SUBWT1} + \text{EQWT} + \text{HARNWT} + \text{STTPS} + \text{EQBSTR} \quad (9-49)$$

$$\text{CONTIN} = 0.15 \times \text{SUBWT2} \quad (9-50)$$

The satellite gross weight (SATWT) is the sum of the system weight (SUBWT2) plus the contingency (CONTIN).

$$\text{SATWT} = \text{SUBWT2} + \text{CONTIN} \quad (9-51)$$

9.3.9 Adapter Weight

The adapter weight (SATADP) is a function of the satellite gross weight (SATWT).

$$\text{SATADP} = 0.012 \times \text{SATWT} \quad (9-52)$$

9.3.10 Spacecraft Launch Weight

Finally, the satellite launch weight (SATTWT) is the sum of the satellite gross weight (SATWT) and the adapter weight (SATADP).

$$\text{SATTWT} = \text{SATWT} + \text{SATADP} \quad (9-53)$$

The launch weight (SATTWT) is compared with the maximum allowable weight (SWTMAX) specified by the user. If SATTWT exceeds SWTMAX, the design is aborted.

9.3.11 Center-of-Gravity Location

The model determines the inertial characteristics of the satellite, such as center of gravity (c. g.), individual component inertias about their own c. g. 's (called incremental inertias) for the three axes (x, y, z), and the total inertia of the vehicle (about the three axes).

- a. Mission Equipment. Several rules must be observed in locating the mission equipment bays. Two locations are available:

- Number 1 = Forward of the Equipment Bay on the Equipment Bay centerline.
Number 2 = Behind the Equipment Bay on the Equipment Bay centerline.

These locations are shown in Figure 9-7.

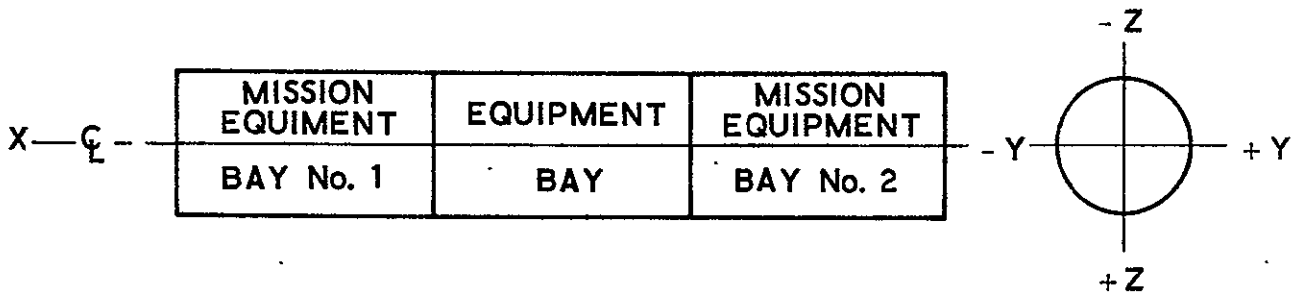


Figure 9-7. Mission Equipment Locations

The forward mission equipment bay (No. 1) "x" c. g. is determined by combining the station at the aft end of the equipment bay (500) plus the equipment bay length (EQBLG) and half the length of the mission equipment bay length (EQM1XL/2). Again the c. g. is assumed to be at the midpoint of the bay.

$$\begin{array}{l}
 \text{EM1XCG} = 500 + \text{EQBLG} + (\text{EQM1XL}/2) \\
 \text{EM1YCG} = \text{Input} = 0.0 \text{ Default value} \\
 \text{EM1ZCG} = \text{Input} = 0.0 \text{ Default value} \\
 \text{EM2XCG} = \text{Input} = 500 - (\text{EQM2XL}/2) \\
 \text{EM2YCG} = \text{Input} = 0.0 \text{ Default value} \\
 \text{EM2ZCG} = \text{Input} = 0.0 \text{ Default value}
 \end{array}
 \left. \begin{array}{l} \\ \\ \\ \\ \\ \end{array} \right\} \begin{array}{l} (9-54) \\ \\ \\ (9-55) \\ \end{array}$$

b. Equipment Bay. The equipment bay "x" c.g. is assumed to be at the midpoint of the equipment bay. Since the aft end of the equipment bay is always station 500, the equipment bay "x" c.g. is 500 plus half the equipment bay length (EQBLG).

$$\text{EBXCG} = 500 + (\text{EQBLG}/2) \quad (9-56)$$

The equipment bay "y" and "z" c.g.'s are assumed to be on the equipment bay centerline which is station zero.

$$\begin{array}{l}
 \text{EBYCG} = 0 \\
 \text{EBZCG} = 0
 \end{array}
 \left. \right\} (9-57)$$

c. Equipment Bay Structure. The main equipment bay structure and equipment c.g. is assumed to be at the midpoint of the bay and the aft end of the bay is x station 500. The y and z coordinates are zero at the centerline.

Therefore:

$$\begin{array}{l}
 \text{STRXCG} = 500 + (\text{EQBLG}/2) \\
 \text{STRYCG} = 0 \\
 \text{STRZCG} = 0
 \end{array}
 \left. \right\} (9-58)$$

d. External Equipment. The external equipment packages are assumed to be cubes and the volume of each (in. ³) is:

$$\text{EE1VL} = (\text{EEQ1WT}/\text{DENS}) \times 1728 \quad (9-59)$$

where $\text{DENS} = 0.24 \text{ g/cm}^3 \text{ (15 lb/ft}^3\text{)}$

Note: DENS may be a variable.

Since each package is a cube, the length of the side (in inches) is:

$$EE1SID = EE1VL^{0.333} \quad (9-60)$$

Then, the "x" location (and c.g.) is selected as follows:

$$\begin{aligned} \text{If forward, } EE1XCG &= 500 + EQBLG \\ \text{If center, } EE1XCG &= EQBLG/2 + 500 \\ \text{If aft, } EE1XCG &= 500 \end{aligned} \quad \left. \vphantom{\begin{aligned} \text{If forward, } EE1XCG &= 500 + EQBLG \\ \text{If center, } EE1XCG &= EQBLG/2 + 500 \\ \text{If aft, } EE1XCG &= 500 \end{aligned}} \right\} (9-61)$$

The external equipment packages are located either on top (-z) on the right side (+y) on bottom (+z) or on the left side (-y) of the equipment bay.

On the right side:

$$EE1YCG = SATDAM/2 + EE1SID/2 \quad (9-62)$$

On the left side:

$$\begin{aligned} EE1YCG &= SATDAM/2 - EE1SID/2 \\ EE1ZCG &= 0 \end{aligned} \quad \left. \vphantom{\begin{aligned} EE1YCG &= SATDAM/2 - EE1SID/2 \\ EE1ZCG &= 0 \end{aligned}} \right\} (9-63)$$

If the package is to be located on the top or bottom:

$$EE1YCG = 0 \quad (9-64)$$

On the top:

$$EE1ZCG = -SATDAM/2 - EE1SID/2 \quad (9-65)$$

On the bottom:

$$EE1ZCG = SATDAM/2 + EE1SID/2 \quad (9-66)$$

If the package is to be located on either side:

$$EE1ZCG = 0 \quad (9-67)$$

C.G.'s for the other external equipment packages are handled in a similar manner.

- e. Solar Array. * The solar arrays are located in the xy plane. Each solar array boom lies on the y axis and is 61 cm (24 in.) long. The number one solar array is on the right side (+y) of the vehicle and the number two solar array is on the left side (-y).

Therefore:

SA1XCG is computed as in the external equipment case.

$$\left. \begin{aligned} \text{SA1YCG} &= \text{SATDAM}/2 + 24 + \text{SA1YL}/2 \\ \text{SA1ZCG} &= 0 \end{aligned} \right\} (9-68)$$

and:

$$\left. \begin{aligned} \text{SA2XCG} &= \text{SA1XCG} \\ \text{SA2YCG} &= -\text{SATDAM}/2 - 24 - \text{SA2YL}/2 \\ \text{SA2ZCG} &= 0 \end{aligned} \right\} (9-69)$$

The body-mounted array c.g.'s are:

SA3XCG is computed as in the external equipment case.

$$\left. \begin{aligned} \text{SA3YCG} &= 0 \\ \text{SA3ZCG} &= 0 \end{aligned} \right\} (9-70)$$

- f. Solar Array Boom. The solar array boom c.g. in the x direction is the same as the solar array 'x' c.g.:

$$\text{SABXCG} = \text{SA1XCG} \quad (9-71)$$

The 'y' c.g. is zero, since they are symmetrical about the centerline:

$$\text{SABYCG} = 0 \quad (9-72)$$

The 'z' c.g. is zero, since the solar arrays lie in the xy plane:

$$\text{SABZCG} = 0 \quad (9-73)$$

* The user should not locate both a solar array and an external equipment package at the same location (i. e., 1, 2, or 3). Locate the solar arrays first and, if an external equipment package is called out at the same location, move the external equipment package to the next location.

- g. Satellite Center of Gravity Calculations. The "x", "y", and "z" center of gravity of the entire satellite can now be determined:

$$\begin{aligned}
 \text{SATXCG} = & \quad [(\text{EQBSTR} \times \text{STRXCG}) \\
 & \quad + (\text{EQWT} \times \text{EBXCG}) \\
 & \quad + (\text{EQM1ST} + \text{EQM1WT}) \times \text{EM1XCG} \\
 & \quad + (\text{EQM2ST} + \text{EQM2WT}) \times \text{EM2XCG} \\
 & \quad + (\text{SA1WT} \times \text{SA1XCG}) \\
 & \quad + (\text{SA2WT} \times \text{SA2XCG}) \quad \quad \quad (9-74) \\
 & \quad + (\text{SA3WT} \times \text{SA3XCG}) \\
 & \quad + (\text{SABMWT} \times \text{SABXCG}) \\
 & \quad + (\text{EEQ1WT} \times \text{EE1XCG}) + \dots \\
 & \quad + (\text{EEQ9WT} \times \text{EE9XCG}) \\
 & \quad + (\text{HARNWT} \times \text{EBXCG}) \\
 & \quad + (\text{STTPS} \times \text{EBXCG})]/(\text{SATWT} - \text{CONTIN})
 \end{aligned}$$

$$\begin{aligned}
 \text{SATYCG} = & \quad [(\text{EQBSTR} \times \text{STRYCG}) \\
 & \quad + (\text{EQWT} \times \text{EBYCG}) \\
 & \quad + (\text{EQM1ST} + \text{EQM1WT}) \times \text{EM1YCG} \\
 & \quad + (\text{EQM2ST} + \text{EQM2WT}) \times \text{EM2YCG} \\
 & \quad + (\text{SA1WT} \times \text{SA1YCG}) \\
 & \quad + (\text{SA2WT} \times \text{SA2YCG}) \quad \quad \quad (9-75) \\
 & \quad + (\text{SA3WT} \times \text{SA3YCG}) \\
 & \quad + (\text{SABMWT} \times \text{SABYCG}) \\
 & \quad + (\text{EEQ1WT} \times \text{EE1YCG}) + \dots \\
 & \quad + (\text{EEQ9WT} \times \text{EE9YCG}) \\
 & \quad + (\text{HARNWT} \times \text{EBYCG}) \\
 & \quad + (\text{STPPS} \times \text{EBYCG})]/(\text{SATWT} - \text{CONTIN})
 \end{aligned}$$

$$\begin{aligned}
\text{SATZCG} = & \text{[(EQBSTR x STRZCG)} \\
& + (\text{EQWT x EBZCG}) \\
& + (\text{EQM1ST} + \text{EQM1WT}) \times \text{EM1ZCG} \\
& + (\text{EQM2ST} + \text{EQM2WT}) \times \text{EM2ZCG} \\
& + (\text{SA1WT x SA1ZCG}) \\
& + (\text{SA2WT x SA2ZCG}) \\
& + (\text{SA3WT x SA3ZCG}) \\
& + (\text{SABMWT x SABZCG}) \\
& + (\text{EEQ1WT x EE1ZCG}) + . . . \\
& + (\text{EEQ9WT x EE9ZCG}) \\
& + (\text{HARNWT x EBZCG}) \\
& + (\text{STTPS x EBYCG})]/(\text{SATWT} - \text{CONT})
\end{aligned}
\tag{9-76}$$

9.3.12 Incremental Moments of Inertia

The next step is to calculate the incremental moments of inertia of the various components, i. e., equipment bay, mission equipment bays, solar arrays, and external equipment. In order to do this, the dimensions of the various shapes must be read from the input data or (in the case of the equipment bay) from the earlier portion of the model.

The incremental or individual moments of inertia may now be calculated for the various components. These inertias are calculated about the c. g. of the component and later are transferred to the c. g. of the satellite. As an example, STRINX is the inertia of the equipment bay structure about the x axis, STRINY about the y axis, and STRINZ about the z axis.

a. Equipment Bay Structure

1. Cylindrical shell

$$EQBST1 = EQBSTR + STTPS \quad (9-77)$$

$$\begin{aligned} SATRAD &= \text{Satellite radius} \\ &= SATDAM/2 \end{aligned} \quad (9-78)$$

$$\begin{aligned} STRINX &= EQBST1 \times SATRAD^2 \\ STRINY &= EQBST1/2 \times (SATRAD^2 + EQBLG^2/6) \\ STRINZ &= STRINY \end{aligned} \quad (9-79)$$

2. Box-shaped shell

$$\begin{aligned} STRINX &= EQBSTR/12 \times (2 \times EQBSID)^2 \\ STRINY &= EQBSTR/12 \times (EQBSID^2 + EQBLG^2) + \\ &\quad \frac{(EQBSTR \times EQBSID \times EQBSID^2)}{6 \times (EQBSID + EQBSID)} \end{aligned} \quad (9-80)$$

$$STRINZ = STRINY$$

3. Spherical shell

$$\begin{aligned} STRINX &= 0.667 \times EQBSTR \times (SATDA3/2)^2 \\ STRINY &= STRINX \\ STRINZ &= STRINX \end{aligned} \quad (9-81)$$

b. Equipment Bay

1. Cylindrical vehicle

$$\begin{aligned} EQINX &= (EQWT/2) \times SATRAD^2 \\ EQINY &= (EQWT/12) \times [(3 \times SATRAD^2) + EQBLG^2] \\ EQINZ &= EQINY \end{aligned} \quad (9-82)$$

2. Box-Shaped Vehicle

$$\left. \begin{aligned} \text{EQINX} &= \text{EQWT}/12 \times (2 \times \text{EQBSID}^2) \\ \text{EQINY} &= \text{EQWT}/12 \times (\text{EQBSID}^2 + \text{EQBLG}^2) \\ \text{EQINZ} &= \text{EQINY} \end{aligned} \right\} (9-83)$$

3. Spherical Vehicle

$$\left. \begin{aligned} \text{EQINX} &= 0.40 \times \text{EQWT} \times (\text{SATDAM}/2)^2 \\ \text{EQINY} &= \text{EQINX} \\ \text{EQINZ} &= \text{EQINX} \end{aligned} \right\} (9-84)$$

c. External Equipment

EEQ1WT = External Equipment Number 1 - Weight
 EEQ1XL = External Equipment Number 1 - Length in x direction
 EEQ1YL = External Equipment Number 1 - Length in y direction
 EEQ1ZL = External Equipment Number 1 - Length in z direction

$$\left. \begin{aligned} \text{EE1INX} &= \text{EEQ1WT}/12 \times (\text{EEQ1YL}^2 + \text{EEQ1ZL}^2) \\ \text{EE1INY} &= \text{EEQ1WT}/12 \times (\text{EEQ1XL}^2 + \text{EEQ1ZL}^2) \\ \text{EE1INZ} &= \text{EEQ1WT}/12 \times (\text{EEQ1XL}^2 + \text{EEQ1YL}^2) \end{aligned} \right\} (9-85)$$

. to EE9INY

NOTE: Total External Equipment (EEQTWT) is sum of EEQ1WT + EEQ2WT + EEQ3WT . . . etc.

d. Solar Arrays. Arrays must be oriented with side parallel to axes. If oriented (movable), inertia calculations are for parallel position.

$$\left. \begin{aligned} \text{SA1INX} &= (\text{SA1WT}/12) \times (\text{SA1YL}^2 + \text{SA1ZL}^2) \\ \text{SA1INY} &= (\text{SA1WT}/12) \times (\text{SA1XL}^2 + \text{SA1ZL}^2) \\ \text{SA1INZ} &= (\text{SA1WT}/12) \times (\text{SA1XL}^2 + \text{SA1YL}^2) \end{aligned} \right\} (9-86)$$

$$\begin{aligned}
SA2INX &= (SA2WT/12) \times (SA2YL^2 + SA2ZL^2) \\
SA2INY &= (SA2WT/12) \times (SA2XL^2 + SA2ZL^2) \\
SA2INZ &= (SA2WT/12) \times (SA2XL^2 + SA2YL^2) \\
SA3INX &= SA3WT \times SATRAD^2 \\
SA3INY &= (SA3WT/2) \times [SATRAD^2 + (SA3XL^2/6)] \\
SA3INZ &= SA3INY
\end{aligned}
\left. \vphantom{\begin{aligned} SA2INX \\ SA2INY \\ SA2INZ \\ SA3INX \\ SA3INY \\ SA3INZ \end{aligned}} \right\} \begin{array}{l} (9-87) \\ (9-88) \end{array}$$

e. Mission Equipment

1. Cylindrical Shape

$$\begin{aligned}
EQM1ST &= \text{Mission Equipment Bay Structural Weight} \\
EQM1WT &= \text{Mission Equipment Bay Equipment Weight} \\
EM1RAD &= EQM1YL/2
\end{aligned}
\left. \vphantom{\begin{aligned} EQM1ST \\ EQM1WT \\ EM1RAD \end{aligned}} \right\} (9-89)$$

$$\begin{aligned}
EM1INX &= [(EQM1ST + EQM1WT)/2] \times EM1RAD^2 \\
EM1INY &= [(EQM1ST + EQM1WT)/12] \times [(3 \times EM1RAD^2) + EQM1XL^2] \\
EM1INZ &= EM1INY
\end{aligned}
\left. \vphantom{\begin{aligned} EM1INX \\ EM1INY \\ EM1INZ \end{aligned}} \right\} (9-90)$$

$$\begin{aligned}
EM2INX &= [(EQM2ST + EQM2WT)/2] \times EM2RAD^2 \\
EM2INY &= [(EQM2ST + EQM2WT)/12] \times [(3 \times EM2RAD^2) + EQM2XL^2] \\
EM2INZ &= EM2INY
\end{aligned}
\left. \vphantom{\begin{aligned} EM2INX \\ EM2INY \\ EM2INZ \end{aligned}} \right\} (9-91)$$

2. Box-Shaped

$$\begin{aligned}
EQM1XL &= \text{Length of Mission Equipment Bay in } x \text{ direction} \\
EQM1YL &= \text{Length of Mission Equipment Bay in } y \text{ direction} \\
EQM1ZL &= \text{Length of Mission Equipment Bay in } z \text{ direction}
\end{aligned}$$

$$\begin{aligned}
EM1INX &= [(EQM1ST + EQM1WT)/12] \times (EQM1YL^2 + EQM1ZL^2) \\
EM1INY &= [(EQM1ST + EQM1WT)/12] \times (EQM1ZL^2 + EQM1XL^2) \\
EM1INZ &= [(EQM1ST + EQM1WT)/12] \times (EQM1YL^2 + EQM1XL^2)
\end{aligned}
\left. \vphantom{\begin{aligned} EM1INX \\ EM1INY \\ EM1INZ \end{aligned}} \right\} (9-92)$$

$$\begin{aligned}
EM2INX &= [(EQM2ST + EQM2WT)/12] \times (EQM2YL^2 + EQM2ZL^2) \\
EM2INY &= [(EQM2ST + EQM2WT)/12] \times (EQM2ZL^2 + EQM2XL^2) \\
EM2INZ &= [(EQM2ST + EQM2WT)/12] \times (EQM2YL^2 + EQM2XL^2)
\end{aligned}
\left. \vphantom{\begin{aligned} EM2INX \\ EM2INY \\ EM2INZ \end{aligned}} \right\} (9-93)$$

9.3.13 Moments of Inertia

After the incremental inertias (I_o) about the component c. g. 's have been determined, the total inertia of the satellite is calculated about the complete satellite c. g. The equations take the form:

$$\left. \begin{aligned} I_x &= I_{ox} + WY^2 + WZ^2 \\ I_y &= I_{oy} + WZ^2 + WX^2 \\ I_z &= I_{oz} + WY^2 + WX^2 \end{aligned} \right\} (9-94)$$

a. Satellite Total Inertia about X Axis

$$\begin{aligned} \text{SATINX} &= \text{STRINX} + \text{EQBSTR} \times [(\text{SATYCG} - \text{STRYCG})^2 + (\text{SATZCG} - \text{STRZCG})^2] \\ &+ \text{EM1INX} + \text{EQM1TO} \times [(\text{SATYCG} - \text{EM1YCG})^2 + (\text{SATZCG} - \text{EM1ZCG})^2] \\ &+ \text{EM2INX} + \text{EQM2TO} \times [(\text{SATYCG} - \text{EM2YCG})^2 + (\text{SATZCG} - \text{EM2ZCG})^2] \\ &+ \text{EQINX} + \text{EQWT} \times [(\text{SATYCG} - \text{EQYCG})^2 + (\text{SATZCG} - \text{EQZCG})^2] \\ &+ \text{SA1INX} + \text{SA1WT} \times [(\text{SATYCG} - \text{SA1YCG})^2 + (\text{SATZCG} - \text{SA1ZCG})^2] \\ &+ \text{SA2INX} + \text{SA2WT} \times [(\text{SATYCG} - \text{SA2YCG})^2 + (\text{SATZCG} - \text{SA2ZCG})^2] \\ &+ \text{SA3INX} + \text{SA2WT} \times [(\text{SATYCG} - \text{SA3YCG})^2 + (\text{SATZCG} - \text{SA2ZCG})^2] \\ &+ \text{EE1INX} + \text{EEQ1WT} \times [(\text{SATYCG} - \text{EE1YCG})^2 + (\text{SATZCG} - \text{EE1ZCG})^2] \\ &+ \text{EE2INX} + \dots + \text{EE9INX} \\ &+ \text{SABMWT} \times [(\text{SATYCG} - \text{SABYCG})^2 + (\text{SATZCG} - \text{SABZCG})^2] \quad (9-95) \end{aligned}$$

where:

$$\left. \begin{aligned} \text{EQM1TO} &= \text{EQM1WT} + \text{EQM1ST} \\ \text{EQM2TO} &= \text{EQM2WT} + \text{EQM2ST} \end{aligned} \right\} (9-96)$$

b. Satellite Total Inertia about Y Axis

$$\begin{aligned} \text{SATINY} &= \text{STRINY} + \text{EQBSTR} \times [(\text{SATZCY} - \text{STRZCG})^2 + (\text{SATXCG} - \text{STRXCG})^2] \\ &+ \text{EM1INY} + \text{EQM1TO} \times [(\text{SATZCG} - \text{EM1ZCG})^2 + (\text{SATXCG} - \text{EM1XCG})^2] \\ &+ \text{EM2INY} + \text{EQM2TO} \times [(\text{SATZCG} - \text{EM2ZCG})^2 + (\text{SATXCG} - \text{EM2XCG})^2] \\ &+ \text{EQINY} + \text{EQWT} \times [(\text{SATZCG} - \text{EQZCG})^2 + (\text{SATXCG} - \text{EQXCG})^2] \end{aligned}$$

$$\begin{aligned}
& + SA1INY + SA1WT \times [(SATZCG - SA1ZCG)^2 + (SATXCG - SA1XCG)^2] \\
& + SA2INY + SA2WT \times [(SATZCG - SA2ZCG)^2 + (SATXCG - SA2XCG)^2] \\
& + SA3INY + SA3WT \times [(SATZCG - SA3ZCG)^2 + (SATXCG - SA3XCG)^2] \\
& + EE1INY + EEQ1WT \times [(SATZCG - EE1ZCG)^2 + (SATXCG - EE1XCG)^2] \\
& + EE2INY + \dots + EE9INY \\
& + SABMWT \times [(SATZCG - SABZCG)^2 + (SATXCG - SABXCG)^2] \quad (9-97)
\end{aligned}$$

c. Satellite Total Inertia about Z Axis

$$\begin{aligned}
SATINZ = & STRINZ + EQBSTR \times [(SATYCG - STRYCG)^2 + (SATXCG - STRXCG)^2] \\
& + EM1INZ + EQM1TO \times [(SATYCG - EM1YCG)^2 + (SATXCG - EM1XCG)^2] \\
& + EM2INZ + EQM2TO \times [(SATYCG - EM2YCG)^2 + (SATXCG - EM2XCG)^2] \\
& + EQINZ + EQWT \times [(SATYCG - EQYCG)^2 + (SATXCG - EQXCG)^2] \\
& + SA1INZ + SA1WT \times [(SATYCG - SA1YCG)^2 + (SATXCG - SA1XCG)^2] \\
& + SA2INZ + SA2WT \times [(SATYCG - SA2YCG)^2 + (SATXCG - SA2XCG)^2] \\
& + SA3INZ + SA3WT \times [(SATYCG - SA3YCG)^2 + (SATXCG - SA3XCG)^2] \\
& + EE1INZ + EEQ1WT \times [(SATYCG - EE1YCG)^2 + (SATXCG - EE1XCG)^2] \\
& + EE2INZ + \dots + EE9INZ \\
& + SABMWT \times [(SATYCG - SABYCG)^2 + (SATXCG - SABXCG)^2] \quad (9-98)
\end{aligned}$$

9.4 DESIGN LOGIC

The sequence followed in implementing the design algorithm is as follows:

- a. Determine mission equipment bay structure, booms and mechanisms.
- b. Determine total weight and volume of equipment in the equipment bay by summation of individual subsystem weights and volumes.
- c. Use shape specified plus maximum diameter to determine equipment bay length.
- d. Use equipment bay weight of equipment and length/diameter ratio to determine equipment bay structural weight.
- e. Calculate harness and thermal control weight.
- f. Determine satellite gross weight.

- g. Calculate adapter weight.
- h. Determine satellite launch weight (gross weight plus adapter)
- i. Calculate center of gravity.
- j. Calculate moments of inertia.

10. STRUCTURE

10.1 GENERAL

10.1.1 Subsystem Description

Satellite equipment can be classified into two general categories, viz., mission equipment and support equipment. Mission equipment encompasses all equipment which performs a specific function related to the satellite purpose. Cameras, heat sensing devices, and telescopes are examples of mission equipment. Support systems equipment includes all equipment which perform supporting functions within the satellite. Examples are: Stabilization and Control, Auxiliary Propulsion, Communication and Instrumentation, Data Processing, Electrical Power, and Thermal Control.

For purposes of the model, the structure containing the mission equipment is considered separate from the structure containing the support systems equipment. The two structures will be referred to as the mission equipment bay and the systems equipment bay, respectively. Satellites considered in the model will be constructed as a single cylindrical or box type equipment bay with externally attached mission equipment bays as shown in Figure 10-1. A maximum number of eleven mission equipment bays are allowed including the possibility for one at the forward end and one at the aft end on the center line of the systems equipment bay. The remaining mission equipment (or external equipment) bays are laterally positioned at the forward and aft ends as well as in the middle of the systems equipment bay side wall.

The location of most support systems equipment will not be specified with sufficient accuracy to be considered in the structural design

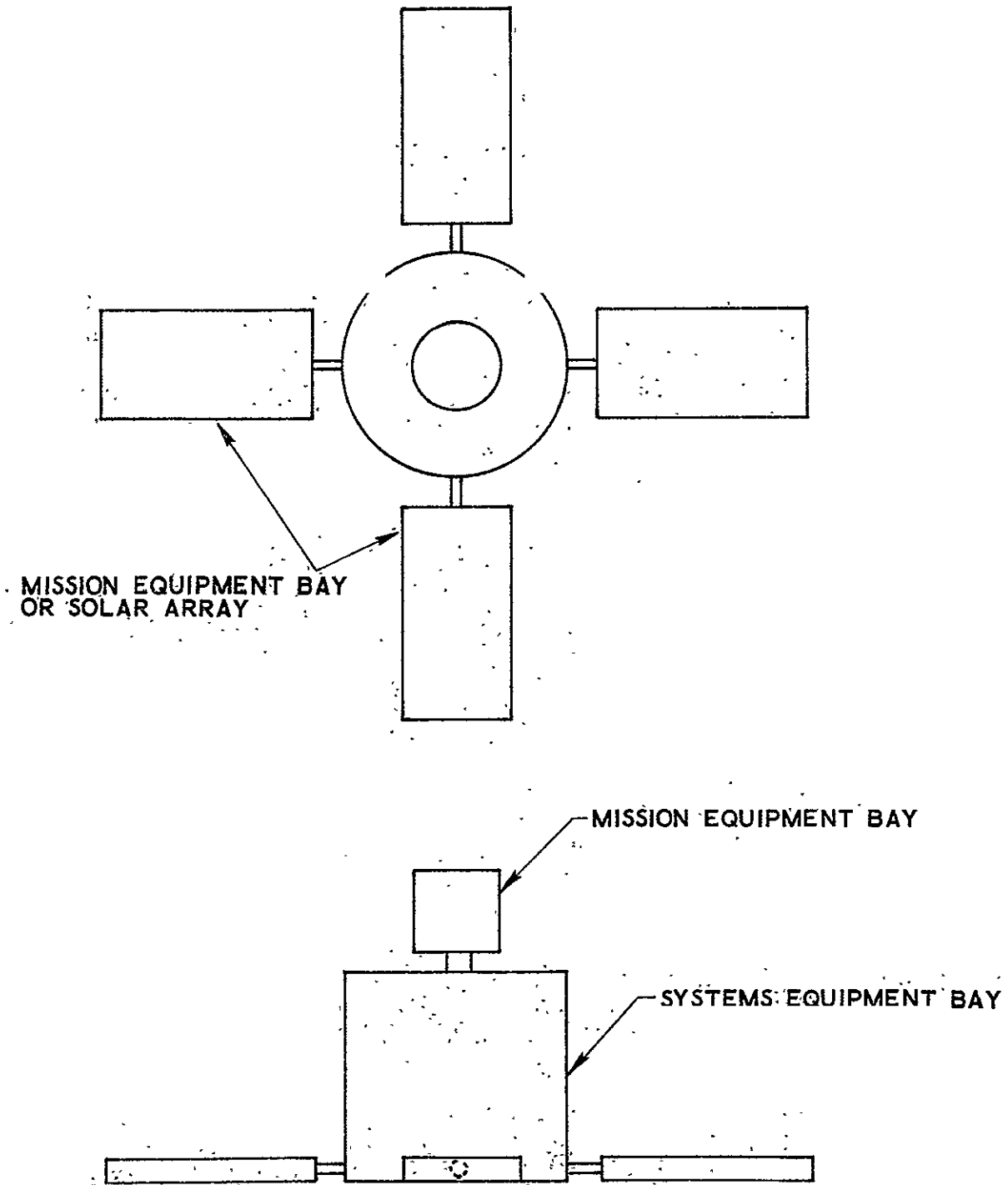


Figure 10-1. Typical Satellite Configuration

process. The exception is the location of paddle mounted solar arrays. These may be located laterally with respect to the systems equipment bay structure, at the forward, aft, or middle of the side wall.

Due to the individuality of the mission equipment bays, only the systems equipment bay structure is considered in the model. However, the mission equipment bay and solar array attachment booms will be included.

10.1.2 Subsystem Configurations

There are three types (configurations) of construction appropriate for the systems equipment bay structure. These include the monocoque shell, the stringer and frame stiffened semi-monocoque shell, and the truss structure. Due to the interest expressed by NASA in the semi-monocoque structure, the Systems Cost/Performance Model includes this configuration or type of construction. Both ends of the systems equipment bay structure are closed with the use of thin plate end covers.

The semi-monocoque structure is a thin shell stiffened with beam-like longitudinal stiffeners and ring-like frames as shown in Figure 10-2. Both the stringers and frames are assumed to have rectangular cross sections.

The lateral location of a mission equipment bay in the middle of the systems equipment bay thin shell side wall requires a structural member to transmit loads to other parts of the systems equipment bay structure. For this purpose, a stiff ring or rib-like plate (henceforth called a midsection bulkhead) will be sized and located in the middle of the system equipment bay.

The mission equipment bay and solar array attachment booms are designed as thin walled tubes. These tubes are assumed to have circular cross sections.

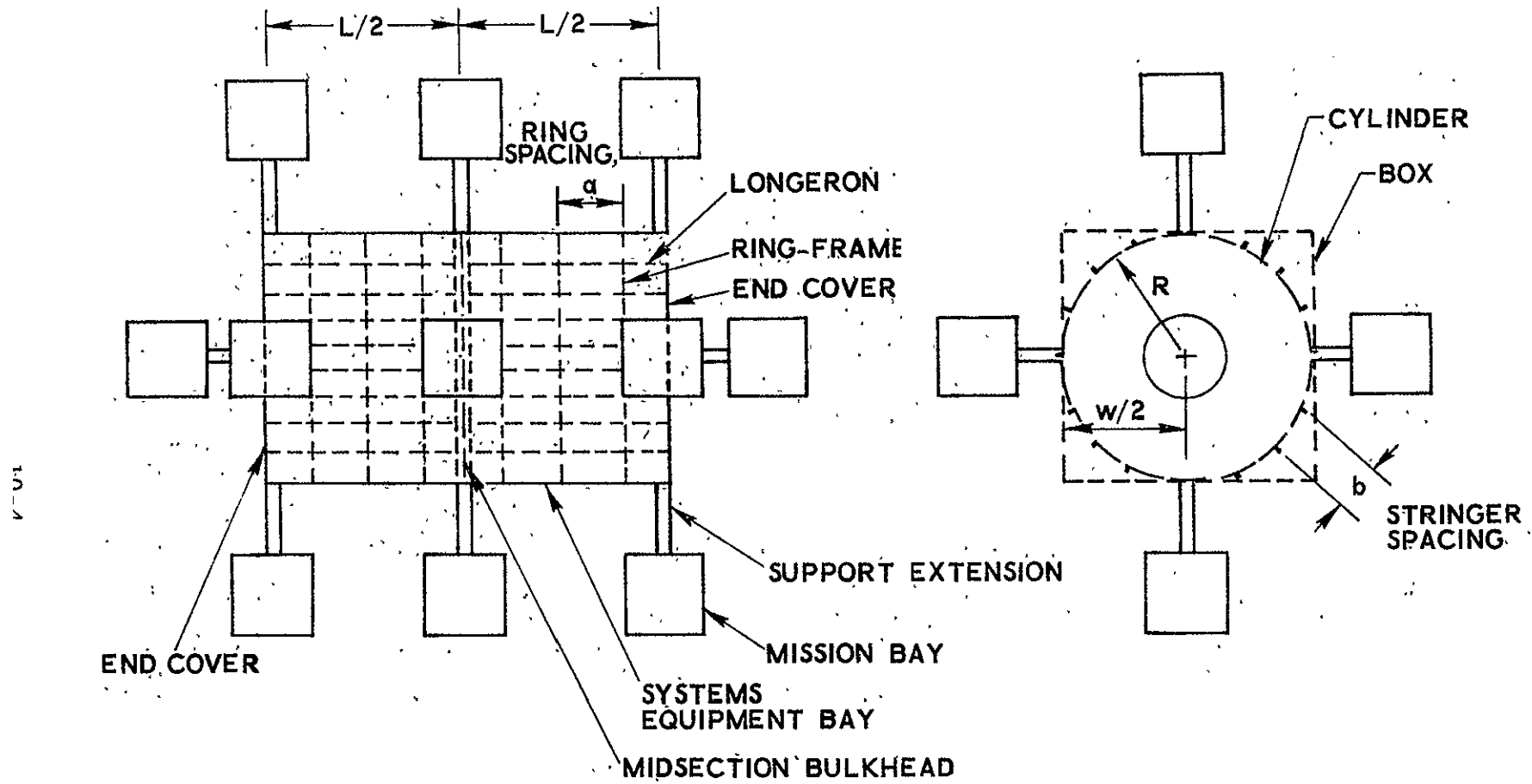


Figure 10-2. Satellite Structural Idealization

10.1.3 Configuration Compatibility

The Structural configuration must be consistent with the vehicle shape. Table 10-1 summarizes the compatibility requirements. The only special requirement is that a spherically shaped vehicle is generally constructed using a truss structure.

Table 10-1. Structural Configuration Compatibility

Structural Configuration	Vehicle Shape		
	Cylinder	Sphere	Box
Monocoque	Yes	No	Yes
Semi-Monocoque	Yes	No	Yes
Truss	Yes	Yes	Yes

Legend: Yes - Compatible
No - Incompatible

10.1.4 Equipment Types

A design algorithm for a semi-monocoque circular cylinder, or box-like satellite equipment bay structure has been developed. The design algorithm includes methods for sizing the elements of a stringer-frame stiffened shell with plate-like end covers and thin walled tubular mission equipment bay and solar array extension booms. In particular, the following quantities are determined:

- a. Systems equipment bay structure.
 1. Skin thickness, t
 2. Stringer width, t_s

3. Stringer height, b_s
 4. Number of stringers, n
 5. Frame width, t_f
 6. Frame height, b_f
 7. Number of frames, m
 8. Forward cover thickness, t_e
 9. Aft cover thickness, t_a
 10. Midsection bulkhead thickness, (when required)
- b. Mission equipment bay and solar array extension booms.
1. Nominal tubular radius, r
 2. Tubular wall thickness, t_w

With this information, the preliminary design of the satellite equipment bay structure is adequately defined for the purposes of the model.

10.1.5 Design Criteria

The design of semi-monocoque structures requires consideration of three different failure modes. These failure modes are:

- a. Material failure.
- b. Local buckling between frames.
- c. General instability of the total structure.

In general, the latter two modes are critical for compression loaded shell structures.

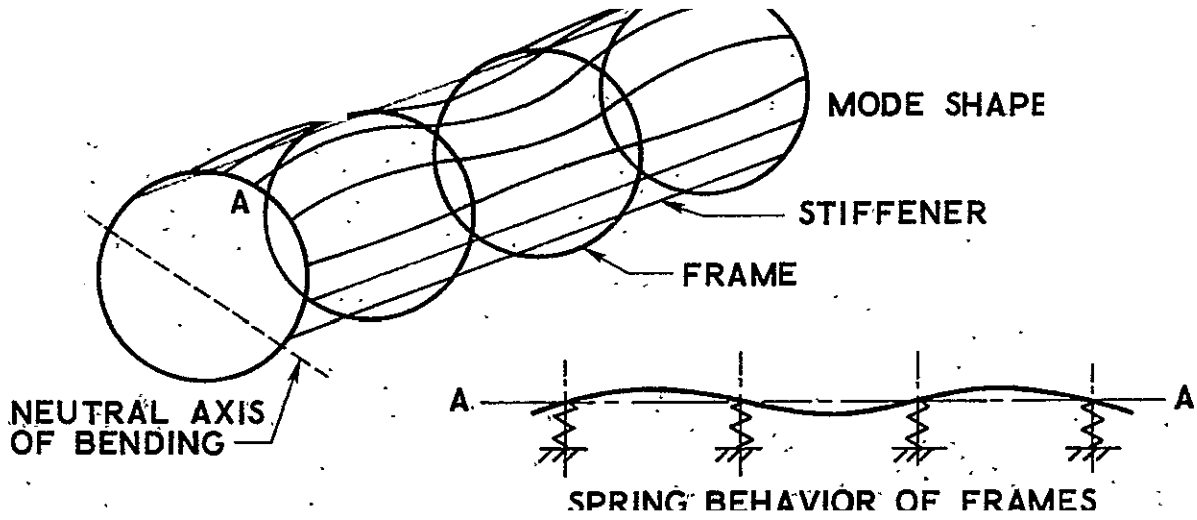
The first type of failure can be avoided by designing the structure such that under no circumstances will a limiting parameter describing the material failure be exceeded. Such a parameter could, for example, be the yield stress or ultimate stress of the material used in the design. In the following design algorithm, the limiting material parameter is the yield stress.

Designing the structure to support a given load without the occurrence of a stability failure as categorized in failure modes (b) and (c) above is not a simple task. Type (b) failure modes may be subdivided into subclasses of failure. For example, the sheet forming the skin between the stringers and frames may buckle as an edge stiffened plate, or the stringer itself may buckle as a consequence of column instability, or finally, the skin-stringer combination may buckle between frames as a result of longitudinally stiffened panel instability as shown in Figure 10-3. Of course, the general instability, mode (c) above, is characterized by buckling of the skin-stringer-frame combination (Fig. 10-3).

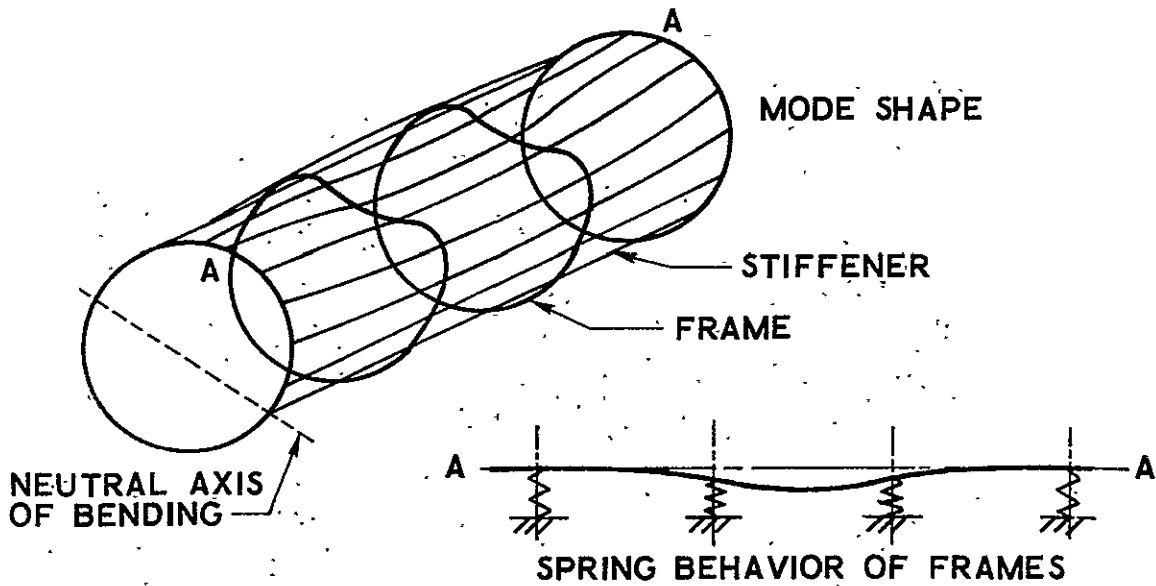
For design purposes, each type of instability (edge stiffened plate, column, longitudinally stiffened panel or general instability) may be characterized by a quantity termed critical stress. The critical stress is defined to be that stress level above which the structural element being analyzed will buckle. Since the structure is limited to function in an elastic manner, the critical stress describing each mode of stability failure must be less than or equal to the elastic limit of the structural material.

There is no real problem in designing a semi-monocoque structure capable of carrying a given load without material yield or undergoing a stability failure. Since a satellite structure is of interest here, the problem is to design a structure of minimum weight. Using the constraint that all modes of instability are avoided, a minimum weight structure can be achieved if the structure is designed in such a way that all modes of instability occur simultaneously and that the resulting critical stress be less than or equal to the material yield strength but greater than the design stress resulting from the structural loads.

The factor of safety selected for the model is 1.25, a value commonly used in design of satellite structures. This factor accounts for uncertainties in material properties, fabrication, loads, analysis,



PANEL INSTABILITY



GENERAL INSTABILITY

Figure 10-3. Semi-Monocoque Cylinder Failure Modes

The satellite structure is also assumed to be constructed of a single isotropic, homogeneous material such as aluminum. For design purposes, the material is completely determined by the specification of Young's modulus E , Poisson's ratio ν , weight density ρ , and yield stress σ_y .

10.2 INPUT DATA

The information required from the user to design the Structure is identified in Table 10-2. The table indicates what data is used, whether the data is required or optional, and, if optional, the default value.

In addition to the user supplied data, the data outlined in Table 10-3 must be supplied by the Vehicle Sizing model.

Table 10-2. Input Data Supplied by User

Symbol	Name	Required (R) or Optional (O)	Default Value
	Location of mission equipment	R	
	Location of external equipment	R	
	Location of solar arrays	R	
W_e	Mission equipment weight (lb)	R	
l_e	Mission equipment and/or solar array extension (in.)	O	24
C_a	Axial number of gravity accelerations	O	10
C_e	Lateral number of gravity accelerations	O	5
E	Young's modulus (psi)	O	10^7
ν	Poisson's ratio	O	0.33
ρ	Weight density (lb/in ³)	O	0.10
σ_y	Yield stress (psi)	O	3×10^4

Table 10-3. Input Data Supplied by Vehicle Sizing

Symbol	Name
	Vehicle shape
	Solar array (Electrical Power) configuration
W	Total satellite weight (lb)
W_b	Equipment bay weight (lb)
L	Equipment bay length (in.)
R	Cylindrical equipment bay radius (in.)
w	Box shaped equipment bay width (in.)
R	Spherical equipment bay radius (in.)
W_e	Solar array weight (lb)

10.3 SEMI-MONOCOQUE CYLINDRICAL STRUCTURE

10.3.1 Structural Loads

The loads applied to the satellite structure result from axial and lateral accelerations induced by the booster or kick stages during the ascent conditions. The maximum g-loading in the axial and lateral directions is denoted by c_a and c_e , respectively.

In computing the design loads for the semi-monocoque structure, the satellite is represented as a cantilevered shell with concentrated masses. Therefore, the design loads will be limited to axial loads due to the axial acceleration coupled with the bending and shear loads caused by the lateral acceleration.

When the midsection bulkhead is not required, the total satellite weight (W) is located on the forward end of the systems equipment bay. For this case, the design load (P) in the axial direction is:

$$P = 1.25 c_a W \quad (10-1)$$

The lateral shear load (T) and bending moment (M) are:

$$T = 1.25 c_e W \quad (10-2)$$

and

$$M = 1.25 c_e LW \quad (10-3)$$

respectively, where L denotes the total systems equipment bay length, and 1.25 is the safety factor.

For cases requiring the midsection bulkhead, the total satellite weight is divided into two parts: one-half at the midsection location and one-half at the forward end of the systems equipment bay. For this case, the maximum axial and lateral shear loads are unchanged from those given in Equations (10-1) and (10-2), but the bending moment is:

$$M = 1.25 \left(\frac{3}{4} \right) c_e LW \quad (10-4)$$

For the circular cylinder of radius R, the maximum compressive stress (σ) due to the combined axial load and bending moment is:

$$\sigma = \frac{P}{2\pi R\bar{t}} + \frac{M}{\pi R^2\bar{t}} \quad (10-5)$$

where \bar{t} , the equivalent thickness, is:

$$\bar{t} = t + \frac{A_s}{b} \quad (10-6)$$

with t, A_s , and b denoting the skin thickness, stringer cross-section area, and stringer spacing, respectively. Since the stress can also be defined in terms of the stress resultant (N) and equivalent thickness \bar{t} as:

$$\sigma = \frac{N}{\bar{t}} \quad (10-7)$$

Equation (10-5) may be rewritten as:

$$N = \frac{P}{2\pi R} + \frac{M}{\pi R^2} \quad (10-8)$$

10.3.2 Design Equations

10.3.2.1 Interframe Stiffened Panel

The first step in the design of the semi-monocoque cylinder, is to size the elements of the interframe stiffened panel. Assuming the interframe distance to be small in comparison to the cylinder radius, the effects of curvature can be ignored, and the stability criteria for flat wide compression columns is used. This type of structure experiences a flexural type of instability for which the critical stress (σ_p) is:

$$\sigma_p = \pi^2 E \left\{ \frac{\rho_p}{a} \right\}^2 \quad (10-9)$$

where a and ρ_p denote the distance between frames and the radius of gyration per unit width of the stiffened plate, respectively. For the present study, the longitudinal stiffeners are assumed to have rectangular cross section with thickness t_s and height b_s , and they are spaced close enough together to model the skin between stringers as a long thin plate. Denoting the stringer spacing by b , the plate buckling stress (σ_{cr}) is:

$$\sigma_{cr} = \frac{\pi^2}{3(1-\nu^2)} E \left(\frac{t}{b} \right)^2 \quad (10-10)$$

where t denotes the plate thickness and b the stringer spacing. Also, the stringers may collapse under load if the critical stress (σ_s) used by Almroth and Burns (Ref. 10-1):

$$\sigma_s = \frac{\pi^2 E}{24(1-\nu^2)} \left(\frac{t_s}{b_s} \right)^2 \quad (10-11)$$

is exceeded.

Using the same techniques of dimensional analysis used by Zahorski (Refs. 10-2 and 10-3) or Gerard (Ref. 10-4), the following combination of stresses given by Equation (10-7) along with Equations (10-9) and (10-10) can be derived:

$$\sigma_p \sigma_{cr} \sigma^2 = \frac{\pi^4}{3(1-\nu^2)} \left[\frac{\rho_p t}{bt} \right]^2 \left[\frac{NE}{a} \right]^2 \quad (10-12)$$

For optimum design, all critical stresses are equal to each other and to the applied stress. Denoting the optimum stress by $\bar{\sigma}$, Equation (10-12) implies the result:

$$\bar{\sigma} = \alpha \left[\frac{NE}{a} \right]^{1/2} \quad (10-13)$$

where α denotes the efficiency coefficient of the panel and is defined as:

$$\alpha = \left[\frac{\pi^4}{3(1-\nu^2)} \right]^{1/4} \left[\frac{\rho_p}{b} \frac{t}{t} \right]^{1/2} \quad (10-14)$$

For minimum weight, the optimum stress should be as large as possible without exceeding the material failure limit. However, for a given panel geometry which limits the value of α , the wide column structure can only support some fixed value of load N without an instability failure. The corresponding stress value is generally much less than that associated with material failure. Therefore, in order to optimize the design, the panel efficiency coefficient α is to be maximized within the limitations imposed by the geometrical constraints of the panel structure.

A plate stiffened with rectangular section stringers has an equivalent thickness and efficiency coefficient given by:

$$\bar{t} = t(1 + r_t r_b) \quad (10-15)$$

$$\alpha = \left[\frac{\pi^4}{12(1-\nu^2)} \right]^{1/4} \left[\frac{r_t r_b^3 (4 + r_t r_b)}{[1 + r_t r_b]} \right]^{1/4} \quad (10-16)$$

where:

$$r_t = \frac{t_s}{t} \quad (10-17)$$

$$r_b = \frac{b_s}{b} \quad (10-18)$$

Again noting that for optimum design all critical stresses for instability are equal, the use of Equations (10-10) and (10-11) gives the following result:

$$r_t = 2\sqrt{2} r_b \quad (10-19)$$

Substituting Equation (10-19) into Equation (10-14) and maximizing α with respect to r_b , the results are:

$$r_b = 0.671 \quad (10-20)$$

$$\alpha = 0.745 (1 - \nu^2)^{-1/4} \quad (10-21)$$

Substituting Equation (10-20) into (10-19),

$$r_t = 1.90 \quad (10-22)$$

Using Equations (10-11), (10-15), (10-17), (10-18), (10-20) and (10-22), the optimum dimensions for the rectangular section stringer stiffened plate-like wide column are given as follows:

$$t = 0.44 \bar{t} \quad (10-23)$$

$$t_s = 1.90 t \quad (10-24)$$

$$b_s = \left\{ \frac{\pi^2 E}{24 (1 - \nu^2)} \frac{\bar{t}}{N} \right\}^{1/2} t_s \quad (10-25)$$

$$b = 1.49 b_s \quad (10-26)$$

It is observed that given the load N and the equivalent thickness \bar{t} , the dimensions for the elements composing the stiffened panel are completely determined.

For subsequent use, Equation (10-7) combined with (10-13) gives the result:

$$\bar{t} = \left[\frac{Na}{\alpha^2 E} \right]^{1/2} \quad (10-27)$$

The equivalent thickness \bar{t} is related to the frame spacing a and the panel cross-sectional geometry represented by the panel efficiency coefficient α .

10.3.2.2 Frame Stiffener

Having developed the sizing methods for the interframe stiffened panel side wall, the second step in the cylinder design is to size the frame stiffeners. The controlling criterion for the frame design is the avoidance of general instability of the total cylinder structure. By evaluating the results of a large number of bending tests of stringer-frame stiffened cylinders, Shanley (Ref. 10-5) developed a stiffness criteria $(EI)_f$ for the frames given by the relation:

$$(EI)_f = \frac{C_f MD^2}{a_c} \quad (10-28)$$

where M , D and a_c are the applied bending moment, diameter and frame spacing of the stiffened cylinder, respectively. In order to avoid the occurrence of general instability, the constant C_f was empirically determined to require a value greater than 6.25×10^{-5} . By transforming the bending moment into an equivalent resultant membrane load per unit distance of cylinder circumference, the frame stiffness criteria can be rewritten as:

$$(EI)_f = \frac{4\pi C_f R^4 N}{a_c} \quad (10-29)$$

It is interesting to note Gerard (Ref. 10-6) represented the stiffened cylinder as a compressed elastic column beam supported at discrete points by elastic springs, and analytically found the minimum value of C_f to be 6.84×10^{-5} .

As a means of obtaining an optimum design for the stiffened cylinder, a quantity called solidity is defined. Solidity Σ_c is the ratio of the volume occupied by the structure to the volume enclosed by the structure, i.e.,

$$\Sigma_c = \frac{2\pi R \bar{t} a_c + 2\pi R A_f}{\pi R^2 a_c} \quad (10-30)$$

where A_f is the frame cross-sectional area. Note that solidity is directly proportional to structural weight.

Equation (10-29) can be rewritten as:

$$A_f \rho_{fc}^2 = \frac{4 \pi C_f R^4 N}{E a_c} \quad (10-31)$$

Using Equations (10-27), (10-30) and (10-31), the solidity can be rewritten as

$$\Sigma_c = \frac{2}{R} \left\{ \left[\frac{N a_c}{\alpha^2 E} \right]^{1/2} + \frac{4 \pi C_f R^4 N}{E a_c^2 \rho_{fc}^2} \right\} \quad (10-32)$$

Assuming the only open dimension in Equation (10-32) to be the frame spacing a_c , the solidity can be minimized with respect to this quantity. This minimization gives the following results:

a.
$$\left(\frac{a_c}{R} \right) = \left(16 \pi C_f \alpha \right)^{2/5} \left(\frac{R}{\rho_{fc}} \right)^{4/5} \left(\frac{N}{ER} \right)^{1/5} \quad (10-33)$$

b. Frame weight is one fourth the stringer stiffened panel weight, i. e.,

$$\frac{A_f}{a_c} = \frac{1}{4} \bar{t}_c \quad (10-34)$$

Equations (10-27) and (10-33) can be written as:

$$a_c = \frac{\alpha^2 E}{N} \bar{t}_c^2 \quad (10-35)$$

$$\rho_{fc} = \left[16 \pi C_f \right]^{1/2} \left(\frac{R}{a_c} \right) \left[\frac{\alpha^2 N}{E a_c} \right]^{1/4} \quad (10-36)$$

For rectangular frame cross section, the height (b_{fc}) is given as:

$$b_{fc} = 3.464 \rho_{fc} \quad (10-37)$$

while the width (t_{fc}) is given by:

$$t_{fc} = \frac{A_f}{b_{fc}} \quad (10-38)$$

Observe, for a given load N and with the use of Equations (10-23) through (10-26) along with Equations (10-34) through (10-38), the stringer-frame stiffened circular cylindrical shell is completely specified once the equivalent thickness \bar{t} is determined

10.3.2.3 Stiffened Cylinder Equivalent Thickness

The method for determining the equivalent thickness is based upon a comparison of the weight efficiencies of stiffened and unstiffened cylinders having the same length and diameter, and required to carry the same compressive loads. Based upon past design data, the stiffened cylinder is conservatively found to be at least three times as efficient as an unstiffened cylinder and comparing weights, the stiffened cylinder thickness can be determined.

The critical compressive stress (σ_{mc}) defining instability of the unstiffened cylinder is given by:

$$\sigma_{mc} = \frac{\pi^2 k_c E}{12(1 - \nu^2)} \left(\frac{t_{mc}}{L} \right)^2 \quad (10-39)$$

where k_c is a coefficient describing the relationship between the stress found by theory and those determined experimentally and t_{mc} denotes the monocoque shell thickness. By defining the curvature parameter (Z_L) as:

$$Z_L = \sqrt{(1 - \nu^2)} \frac{L^2}{R t_{mc}} \quad (10-40)$$

and using the procedure developed by Batdorf et al, which is described in Gerard (Ref. 10-6) or Gerard and Becker (Ref. 10-7), the coefficient k_c is conservatively determined as follows:

For $Z_L \leq 31$,

$$k_c = 4 \quad (10-41)$$

and for $Z_L > 31$,

$$k_c = 0.16 Z_L \quad (10-42)$$

From Equations (10-39) through (10-42), the thickness (t_{mc}) of the unstiffened cylinder is determined to be the following:

For $Z_L \leq 31$,

$$t_{mc} = 0.672 \left\{ \frac{(1 - \nu^2) NL^2}{E} \right\}^{1/3} \quad (10-43)$$

and for $Z_L > 31$,

$$t_{mc} = 2.76 \left\{ \sqrt{1 - \nu^2} \frac{NR}{E} \right\}^{1/2} \quad (10-44)$$

where:

$$N = \sigma_{mc} t_{mc} \quad (10-45)$$

Comparing the weights of the stiffened and unstiffened cylinders, the following relationship can be derived:

$$\bar{t}_c + \frac{A_f}{a_c} = \frac{1}{3} t_{mc} \quad (10-46)$$

Substituting Equations (10-34) into (10-46), the equivalent thickness (\bar{t}_c) of the stiffened cylinder is related to that of the unstiffened cylinder by:

$$\bar{t}_c = \frac{4}{15} t_{mc} \quad (10-47)$$

Using Equations (10-43) or (10-44) to determine the unstiffened cylinder thickness, followed by the determination of the equivalent stiffened cylinder thickness given by Equation (10-47), the design of the stringer-frame stiffened cylinder is completed with the use of Equation's (10-23) through (10-26) coupled with Equations (10-35) through (10-38).

10.3.3 Design Logic

The design sequence for the stringer-frame stiffened circular cylinder is developed in three steps. First, assuming the inter frame distance to be small in comparison with the cylinder radius, effects of curvature are ignored and the flat wide compression column stability criterion is used for the design of the longitudinally stiffened panel between frames. Using techniques applicable to the optimum design of flat wide compression columns, the dimensions of the stiffened panel elements are determined in terms of the equivalent panel thickness \bar{t} .

The second step in the cylinder design is to size the frame stiffeners. By application of a criterion for avoidance of general instability coupled with optimum design procedures, the frame dimensions are determined. Analogous to the stiffened panel dimensions, the frame dimensions are also found to be functions of the equivalent panel thickness.

Since the longitudinally stiffened panel dimensions and the frame dimensions are observed to be dependent only on the equivalent panel thickness, the stiffened cylinder design is complete once this quantity is determined. The equivalent panel thickness is found by first sizing the thickness of a monocoque shell having the same length and diameter as the semi-monocoque shell and, based on past design data, sizing the equivalent panel thickness by assuming the semi-monocoque structure to be at least three times as weight-efficient as the monocoque shell.

The detailed sequence followed in implementing the design algorithm is as follows:

- a. Input data.
- b. Compute axial load and bending moment.
- c. Compute equivalent axial load.
- d. Size equivalent monocoque cylinder.
 1. Assume short cylinder and compute shell thickness
 2. Compute curvature parameter and test short cylinder assumption
 3. If assumption fails test, assume long cylinder and recompute shell thickness

- e. Compute equivalent thickness of stiffened cylinder.
- f. Size skin-stringer assembly.
 - 1. Skin thickness
 - 2. Stringer thickness
 - 3. Stringer height
 - 4. Stringer spacing
 - 5. Panel efficiency
- g. Size cylinder frames.
 - 1. Frame spacing
 - 2. Frame radius of gyration
 - 3. Frame area
 - 4. Frame height
 - 5. Frame thickness

10.4 SEMI-MONOCOQUE BOX STRUCTURE

10.4.1 Structural Loads

For the box shaped structural configuration, the critical applied compressive stress is:

$$\sigma = \frac{P}{4wt} + \frac{3M}{4w^2t} \quad (10-48)$$

Analogous to the circular cylinder, the resultant load per unit length of the cross-section perimeter is:

$$N = \frac{P}{4w} + \frac{3M}{4w^2} \quad (10-49)$$

10.4.2 Design Equations

10.4.2.1 Interframe Stiffened Panel

Assuming the box structure to be constructed of four stringer-frame stiffened flat panels attached along their edges, plate theory can be used to develop the design process for the box. For conservatism, the common edges of each panel are assumed to be simply supported. As for the

circular cylinder, the skin-stringer structure between frames is sized using wide column criteria. Therefore, Equations (10-23) through (10-27) also apply to the design of the interframe stiffened panel of the box structure.

10.4.2.2 Frame Stiffener

Using beam theory, Langhaar (Ref. 10-8) analytically showed the minimum frame stiffness $(EI)_f$ required to ensure against the occurrence of general instability of flat stringer-frame stiffened panels with simply supported edges is given by:

$$(EI)_f = \frac{4}{\pi^4} \frac{w^4 N}{a_b^2} \quad (10-50)$$

In this equation, w and a_b represent the panel width and frame distance, respectively. Applying the definition for radius of gyration, Equation (10-50) can be rewritten as:

$$A_f = \frac{4}{\pi^4} \frac{w^4 N}{E a_b^2 \rho_{fb}} \quad (10-51)$$

Assuming the frames to have a rectangular cross section of height b_{fb} and thickness t_{fb} the solidity is given by:

$$\Sigma_b = \frac{w t_{fb} + A_f w}{w a_b b_{fb}} \quad (10-52)$$

Substituting into Equation (10-52) the panel effective thickness and the frame area given by Equations (10-27) and (10-51), respectively, and minimizing with respect to the frame spacing, the following results are obtained:

$$a_b = \left(\frac{16}{\pi^4} \right)^{2/5} \left(\frac{w^2}{\rho_{fb}} \right)^{4/5} \left(\frac{\alpha^2 N}{E} \right)^{1/5} \quad (10-53)$$

$$\frac{A_f}{a_b} = \frac{1}{4} \bar{t}_b \quad (10-54)$$

Therefore, frame weight is one-fourth the stringer stiffened panel weight between frames.

Equations (10-27) and (10-53) can be rewritten as:

$$a_b = \frac{\alpha^2 E}{N} \bar{t}_b^2 \quad (10-55)$$

$$\rho_{fb} = \frac{4}{\pi^2} \left(\frac{w^2}{a_b} \right) \left(\frac{\alpha^2 N}{E a_b} \right)^{1/4} \quad (10-56)$$

With the specification of a rectangular section frame, the height is given by:

$$b_{fb} = 3.464 \bar{\rho}_{fb} \quad (10-57)$$

and the width is given by:

$$t_{fb} = \frac{A_f}{b_{fb}} \quad (10-58)$$

As with the stiffened cylinder, for a given load N and with the use of Equations (10-23) through (10-26) along with Equations (10-54) through (10-58), the stringer-frame stiffened panel is completely specified once the equivalent thickness \bar{t}_b is determined. The design of the box structure is complete with the sizing of each of the panels as specified.

10.4.2.3 Stiffened Panel Equivalent Thickness

Analogous to the determination of the circular cylinder equivalent thickness, the flat panel equivalent thickness is determined by comparison of the structural weights of stiffened and unstiffened panels. In the elastic region of material capability, the analysis of panel efficiencies discussed in Gerard (Ref. 10-9) shows the stiffened panel to be conservatively three times as weight-efficient as an unstiffened panel.

The critical compressive stress defining instability of the unstiffened panel is given by:

$$\sigma_{mb} = \frac{\pi^2 k_c E}{12 (1 - \nu^2)} \left(\frac{t_{mb}}{w} \right)^2 \quad (10-59)$$

where k_c is a coefficient describing the mode of instability experienced by the panel.

For $L/w \leq 0.5$:

$$k_c = \left(\frac{w}{L}\right)^2 \quad (10-60)$$

and for $L/w > 0.5$:

$$k_c = 4 \quad (10-61)$$

Using Equations (10-57), (10-60) and (10-61), the thickness of an unstiffened panel is determined by the following:

For $L/w \leq 0.5$:

$$t_{mb} = \left\{ \frac{12(1-\nu^2)NL^2}{\pi^2 E} \right\}^{1/3} \quad (10-62)$$

and for $L/w > 0.5$:

$$t_{mb} = \left\{ \frac{3(1-\nu^2)Nw^2}{\pi^2 E} \right\}^{1/3} \quad (10-63)$$

whereas, for the cylinder:

$$N = \sigma_{mb} t_{mb} \quad (10-64)$$

Comparing the weights of the stiffened and unstiffened panels, the following relationship can be derived:

$$\bar{t}_b + \frac{A_f}{a_b} = \frac{1}{3} t_{mb} \quad (10-65)$$

Substituting Equation (10-54) into (10-65), the equivalent thickness of the stiffened panel is related to the unstiffened panel by:

$$\bar{t}_b = \frac{4}{15} t_{mb} \quad (10-66)$$

Using Equation (10-62) or (10-63) to determine the unstiffened panel thickness followed by the determination of the equivalent stiffened panel thickness given by Equation (10-66), the design of the stringer-frame stiffened box structure is completed with the use of Equations (10-23) through (10-26) coupled with Equations (10-54) through (10-58).

10.4.3 Design Logic

The procedure for sizing the box type semi-monocoque structure is exactly analogous to the stiffened circular cylinder. The side panels between frames are sized as longitudinally stiffened wide columns while the frames are sized to avoid general collapse of the stiffened structure. The dimensions for the structural elements are again determined in terms of the equivalent panel thickness. As for the stiffened cylinder, the equivalent panel thickness is found by comparing the relative weight-efficiencies of stiffened and unstiffened structures.

The detailed sequence followed in implementing the algorithm is as follows:

- a. Input data.
- b. Compute axial load and bending moment.
- c. Compute equivalent axial load.
- d. Compute equivalent monocoque box shell thickness based on the length/width ratio.
- e. Compute equivalent thickness of stiffened box.
- f. Size skin-stringer assembly.
 1. Skin thickness
 2. Stringer thickness
 3. Stringer height
 4. Stringer spacing
 5. Panel efficiency
- g. Size frames.
 1. Frame spacing
 2. Frame radius of gyration
 3. Frame area
 4. Frame height
 5. Frame thickness

10.5 END COVERS AND MIDSECTION BULKHEAD

10.5.1 Structural Loads

The actual loads applied to the end covers and midsection bulkhead cannot be specifically determined since location and attachment of support systems equipment are not specified. For this reason, a portion of total satellite weight (W) is assumed uniformly distributed over the surface of the forward and aft end covers as well as the midsection bulkhead. In general, the uniform applied load (w) for these structural elements is written in the form:

$$w = \frac{1.25 f_c W}{A} \quad (10-67)$$

where f and A denote the fractional portion of the total satellite weight and the area of each of the above elements, respectively. The safety factor is 1.25. The quantity f is assumed as follows:

- a. Forward end cover: $f = 0.25$
- b. Midsection bulkhead: $f = 0.50$
- c. Aft end cover: $f = 0.25$

10.5.2 Design Equations

Basically, the forward and aft equipment bay end covers are assumed to be flat plate-like elements which extend across the ends of the stiffened cylinder or box type structure. Thus, the lateral dimensions are fixed and the thickness is to be determined. The boundary conditions for the covers are assumed to be simple support.

First, consider the circular cylindrical systems equipment bay structure. For a uniformly loaded circular flat plate with simply supported edge, the maximum tensile stress (σ_{\max}) is given by Roark (Ref. 10-10) as:

$$\sigma_{\max} = \frac{3F}{8\pi t^2} (3 + \nu) \quad (10-68)$$

where t and F denote the plate thickness and total applied load, respectively. Since plate thickness is the quantity of interest, Equation (10-68) can be inverted to give:

$$t = \left\{ \frac{3F}{8\pi\sigma_{\max}} (3 + \nu) \right\}^{1/2} \quad (10-69)$$

Recalling Equation (10-67), the total applied load on the forward cover is:

$$F = 1.25 c_a \frac{W}{4} \quad (10-70)$$

and similarly for the aft cover:

$$F = 1.25 c_a W/4 \quad (10-71)$$

where W denotes the total satellite weight.

Since the maximum allowable stress is the material yield stress σ_y , the forward end cover thickness (t_e) is:

$$t_e = \left\{ \frac{15 c_a W}{128 \pi \sigma_y} (3 + \nu) \right\}^{1/2} \quad (10-72)$$

Since the applied load on the aft cover is identical to that of the forward cover, the aft end cover thickness (t_a) is:

$$t_a = t_e \quad (10-73)$$

For the box-type systems equipment bay structure, the maximum tensile stress in a uniformly loaded square plate with simple supported edges is given in Roark (Ref. 10-10) by:

$$\sigma_{\max} = \frac{0.2208 F}{t^2} (1 + \nu) \quad (10-74)$$

By inverting this formula, substituting for F with Equation (10-70) and limiting stress to the yield value, the plate thickness of the forward end cover is found to be:

$$t_e = \left\{ 0.069 \frac{c_a W}{\sigma_y} (1 + \nu) \right\}^{1/2} \quad (10-75)$$

Analogously, using Equations (10-71) and (10-74), the aft end cover thickness is found to be:

$$t_a = t_c \quad (10-76)$$

The midsection bulkhead is arbitrarily chosen to be of annular-like shape having a centrally located hole of three-fourths the systems equipment bay lateral dimension in size. For simplicity, both the circular cylinder and box like structure midsection bulkheads will be sized as uniformly loaded annular plates. For the box structure, the radius is assumed to be:

$$R = \frac{W}{2} \quad (10-77)$$

The maximum stress induced in a uniformly loaded annular ring with simply supported edges from Roark (Ref. 10-10) is:

$$\sigma_{\max} = \frac{1.72 w_e R^2}{t^2} \left[0.418 + 0.036 \nu \right] \quad (10-78)$$

where:

$$w_e = \frac{0.455 c_a W}{R^2} \quad (10-79)$$

Inverting Equation (10-78) and substituting the material yield stress as the maximum allowable stress, the midsection bulkhead thickness (t_i) is:

$$t_i = \left\{ \frac{1.72 w_e R^2}{\sigma_y} \left[0.418 + 0.036 \nu \right] \right\}^{1/2} \quad (10-80)$$

The design of the midsection bulkhead is completed.

10.5.3 Design Logic

After the side wall structure of the satellite systems equipment bay has been sized, the end covers and midsection bulkhead are designed. These elements are sized as thin flat plates with simply supported boundary edges. The material yield stress is used as the maximum allowable design stress.

The sequence followed in implementing the design algorithm is as follows:

- a. Input data.
- b. Compute forward cover thickness.
- c. Compute aft cover thickness.
- d. Determine if midsection bulkhead is required (external equipment or solar array paddles mounted on the midsection).
 1. Compute applied load
 2. Compute midsection bulkhead thickness.

10.6 MISSION EQUIPMENT AND SOLAR ARRAY EXTENSIONS

10.6.1 Structural Loads

The loading condition for the mission equipment bay and solar array extension booms is dependent upon their location. For mission equipment bays located on the ends of the systems equipment bay along the satellite centerline, the applied load has an axial force component given by:

$$P_a = 1.25 c_a W_e \quad (10-81)$$

and a lateral bending moment component given by:

$$M_a = 1.25 c_e l_e W_e \quad (10-82)$$

where W_e and l_e are the mission bay total weight and the extension boom length, respectively. For mission equipment bays or solar arrays positioned laterally with respect to the systems equipment bay side wall structure, the critical applied load has two bending moment components:

one due to the axial acceleration and one due to the lateral acceleration. These two bending moments combine to form a single bending moment of the following magnitude:

$$M_{\ell} = 1.25 \ell_e W_e \left(c_a^2 + c_e^2 \right)^{1/2} \quad (10-83)$$

Again, note the 1.25 safety factor.

10.6.2 Design Equations

The mission equipment bay and solar array extensions are designed as thin walled circular tubes. The length of each extension ℓ_e is assumed fixed by geometrical constraints imposed by location and size of the mission equipment bay, or solar array with respect to the system equipment bay. The quantities to be sized are the tube radius r and the wall thickness t_w . The design algorithm uses appropriate considerations for structural stability and material failure.

For mission equipment bays located at the forward or aft end along the satellite centerline, the applied loads are the axial force and bending moment given by Equation (10-81) and (10-82), respectively. For mission equipment bay or solar arrays positioned laterally with respect to the systems equipment bay, applied load is the single equivalent bending moment given by Equation (10-83).

For a thin wall tube, the tube column flexural instability critical stress (σ_E) is given by:

$$\sigma_E = \frac{\pi^2 E}{2} \left(\frac{r}{\ell_e} \right)^2 \quad (10-84)$$

while the local critical buckling stress (σ_W) of the thin wall is given by:

$$\sigma_W = \frac{E}{4} \left(\frac{t_w}{r} \right) \quad (10-85)$$

The applied stress is:

$$\sigma = \frac{F}{2 \pi r t_W} \quad (10-86)$$

where F is the equivalent axial load. For the mission equipment bays located at the forward or aft end along the satellite centerline:

$$F = P_a + \frac{2M_a}{r} \quad (10-87)$$

where P_a and M_a are given by Equations (10-81) and (10-82), respectively. For mission bays or solar arrays positioned laterally with respect to the systems equipment bay, the equivalent axial load is:

$$F = \frac{2M}{r} \ell \quad (10-88)$$

Using techniques of optimum design, Gerard (Ref. 10-6) deduced the following optimum dimensions for compression loaded thin walled circular tubes:

$$t_W = \left(\frac{2 F}{\pi E} \right)^{1/2} \quad (10-89)$$

$$r = \left(\frac{F \ell_e^4}{2 \pi^5 E} \right)^{1/6} \quad (10-90)$$

The optimum stress (σ_o) is found to be:

$$\sigma_o = \left(\frac{\pi E^2 F}{16 \ell_e^2} \right)^{1/3} \quad (10-91)$$

For the satellite centerline mission bay location, Equations (10-87), (10-89), and (10-90) can be combined to give:

$$r^7 - \frac{P_a \ell_e^4}{2 \pi^5 E} r - \frac{M_a \ell_e^4}{\pi^5 E} = 0 \quad (10-92)$$

$$t_W = \left\{ \frac{2}{\pi E} \left[P_a + \frac{2 M_a}{r} \right] \right\}^{1/2} \quad (10-93)$$

Of the seven roots of Equation (10-92), only one is real and positive and corresponds to the desired tube radius. For laterally located mission equipment bays or solar arrays, Equations (10-88), (10-89) and (10-90) can be combined to give:

$$r = \left\{ \frac{M_e \ell_e^4}{\pi^5 E} \right\}^{1/7} \quad (10-94)$$

$$t_W = \left\{ \frac{4 M_\ell}{\pi E r} \right\}^{1/2} \quad (10-95)$$

Substituting the solutions of Equations (10-92) and (10-93) or (10-94) and (10-95) into Equation (10-91), the optimum stress is computed. By comparing the optimum stress with the material yield stress, the validity of using Euler column instability can be verified. If σ_0 is less than σ_y , the use of Equations (10-89), (10-90), and (10-91) is valid. If σ_0 exceeds σ_y , the use of Euler column instability and therefore Equations (10-89), (10-90) and (10-91) is not valid; however, the local buckling of the tube wall is still an applicable criterion.

If Euler column instability is not valid, the maximum applied stress as well as the critical buckling stress of the tube wall is limited by the material yield stress. Using the yield stress σ_y and Equations (10-85) and (10-86), the tube dimensions for the mission equipment bay extension located on the satellite centerline are found using the following:

$$t_W^3 - \frac{2P_a}{\pi E} t_W - \frac{16\sigma_y M_a}{\pi E^2} = 0 \quad (10-96)$$

$$r = \frac{Et_W}{4\sigma_y} \quad (10-97)$$

Note that Equation (10-96) has only one real positive root corresponding to the tube wall thickness. Analogously, using the yield stress σ_y and Equations (10-85) and (10-86), the tube dimensions for the mission equipment bay or solar array extensions laterally located with respect to the systems equipment bay are found from the following:

$$t_W = \left(\frac{16\sigma_y M_\ell}{\pi E^2} \right)^{1/3} \quad (10-98)$$

$$r = \frac{Et_W}{4\sigma_y} \quad (10-99)$$

Since the length ℓ_e is assumed specified, the design of the mission bay or solar array extension is complete once the nominal tube radius r and wall thickness, t_W are determined.

10.6.3 Design Logic

The design of the thin walled cylindrical tubes used for the mission equipment bay and solar array extension booms is accomplished in two steps. The procedure will start with the assumption that the design is governed by Euler column flexural instability coupled with local side wall instability criteria. If the critical instability stress found with this assumption exceeds the material yield stress, Euler column stability no longer applies. The design procedure will then be based on flexure of the tube as a simple beam coupled with local side wall instability. The maximum allowable stress specified for this condition is the material yield stress.

The detailed sequence followed in implementing the design algorithm is as follows:

- a. Input data .
- b. Compute axial load (if located on vehicle centerline) and bending moment.
- c. Assume applicability of Euler column stability.
- d. Compute nominal tube radius.
- e. Compute tube wall thickness.
- f. Check for applicability of Euler column stability.
- g. If Euler column stability not applicable, recompute tube wall thickness and nominal tube radius.

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11. RELIABILITY

11.1 GENERAL

The Reliability model improves the reliability of the spacecraft design through redundancy by optimum allocation of weight or cost reserves between redundant hardware and expendables, subject to constraints on total satellite weight or cost. The principle of operation is to increase the level of redundancy of a single module, and then to update the system Mean Mission Duration (MMD) or system reliability calculations to determine the change per unit of weight or cost. This is repeated for each module, and the "most profitable" redundancy is implemented. This process is repeated until requirements are met, resources are exhausted, or returns are no longer sufficiently profitable.

A sketch of the program data flow is shown in Figure 11-1. The input data is read, and an initial calculation of spacecraft expense, reliability at truncation time, $R(\text{TRUNC})$, and mean mission duration (MMD) is made. The term "expense" means cost or weight, whichever is selected as appropriate. $R(\text{TRUNC})$ represents the resultant spacecraft reliability functions, with consideration given to redundancy. The MMD represents the expected duration of the mission before failure and is given by

$$\text{MMD} = \int_0^{\text{TRUNC}} R(t) dt$$

where $R(t)$ is the spacecraft reliability function at time t , and $0 \leq t \leq \text{TRUNC}$.

These initial calculated values are printed, and the computer then adds a single redundant element to one of the online modules and calculates a new spacecraft reliability function, $R(t)$, and a new spacecraft

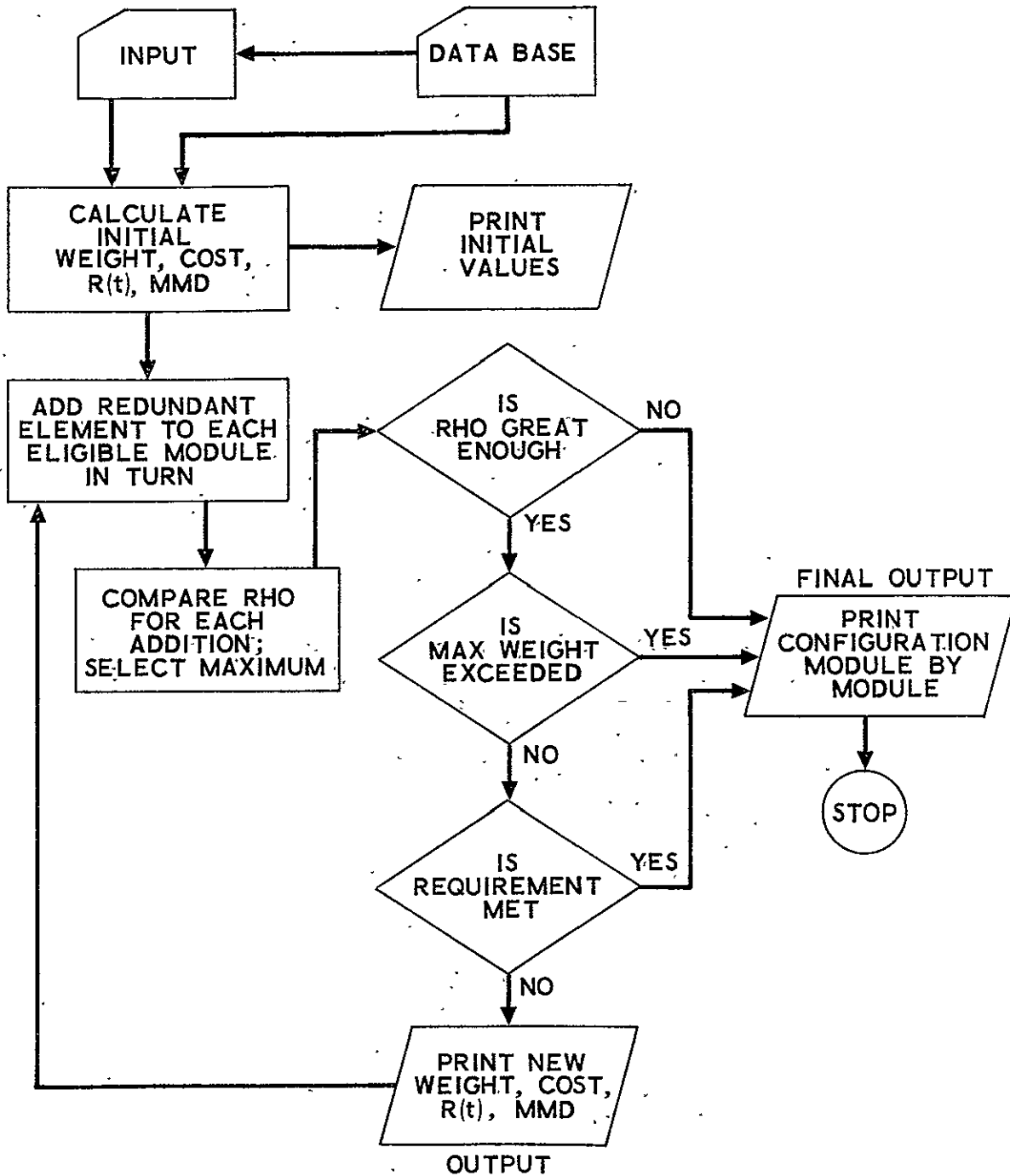


Figure 11-1. Reliability Model

expense. This is repeated for each module where redundancy is available, and the spacecraft MMDs and expenses are calculated in each case. The single redundant module offering the greatest payoff (RHO) as defined by:

$$RHO = \frac{\Delta R(TRUNC)}{\Delta \text{cost}}, \text{ or}$$

$$RHO = \frac{\Delta MMD}{\Delta \text{cost}}, \text{ or}$$

$$RHO = \frac{\Delta MMD}{\Delta \text{weight}}, \text{ or}$$

$$RHO = \frac{\Delta R(TRUNC)}{\Delta \text{weight}}$$

is selected. Then three tests are applied:

- a. Is RHO large enough? The threshold for RHO is preselected.
- b. Is the MMD or R(TRUNC) still short of the requirement?
- c. Is system weight less than the maximum allowable?

If these tests are passed, the new results [expense, R(TRUNC) and MMD] are printed and the computer begins the selection process again. This loop is retraced until one or more of the tests is failed. Then the final configuration is printed out, including expense, R(TRUNC), MMD, and a module-by-module description of the level of redundancy selected. This configuration is recognized as optimum subject to the MMD and weight or cost constraints imposed in the input.*

* It is recognized that the approach does not consider all possible configurations. In the model, cross-strapping is not a variable. Nor is it possible to increment redundancy in more than one mode within a given module. However, within the constraints and assumptions of the present general programming effort, the model is sufficiently accurate.

For modular redundancy to be effective and implementable, an equipment-status monitoring system must be included in system design. This monitoring system must have two characteristics:

- a. It must be comprehensive, i. e., the monitoring system must be sufficiently complete to provide a high probability of detecting any failure in the operating equipment.
- b. It must be fairly reliable to minimize the probability of false alarm and unnecessary corrective action.

The probability, P_D , of detection of a system or module failure, which is the probability of detection of an out-of-specification condition of any functional parameter, is determined as

$$P_D = P(X) P(M)$$

where

$P(X)$ = Probability that the out-of-specification parameter was monitored by the failure detection system subsequent to the failure.

$P(M)$ = Probability that the monitoring system is functioning properly at the time of the failure.

The concept is shown in Figure 11-2.

$P(X)$ is determined by system design. $P(X)$ may be represented as:

$$P(X) = \frac{\text{failure rate of failures detectable by monitoring system}}{\text{total failure rate}}$$

$$\triangleq \frac{\lambda_d}{\lambda_t}$$

$P(M)$ is the reliability of the monitoring system. In general, there will be more than one monitoring subsystem, each designed to monitor different parameters (e. g., voltage, pressure, temperature). $P(M_X)$ is the reliability of that portion of the monitoring system that monitors a given parameter X . Then, as in Figure 11-2:

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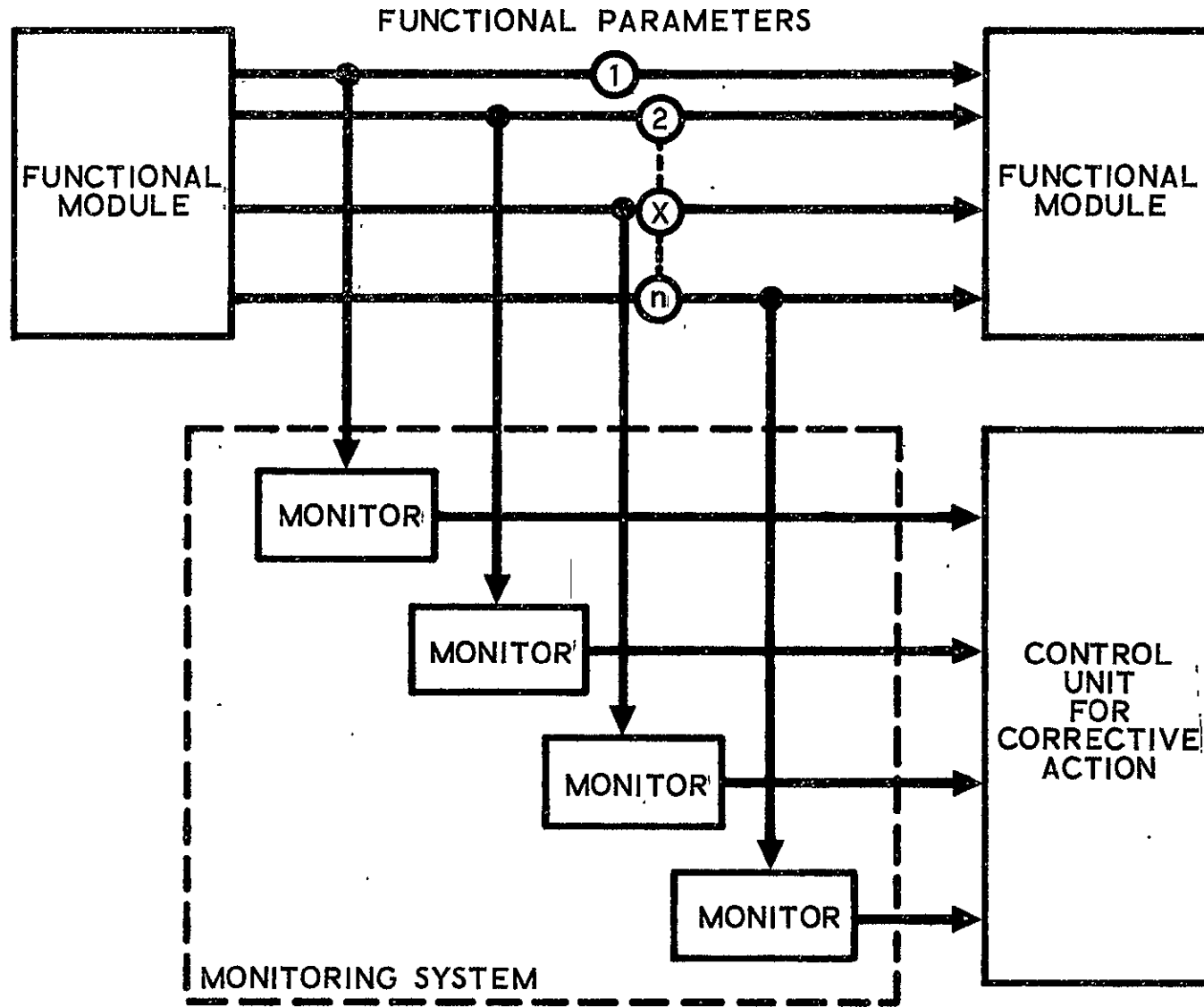


Figure 11-2. Failure Detection System

$$P(M_X) \equiv R_{M_X}(t) = \exp(-\lambda_{M_X} t)$$

and

$$R_M(t) = \prod_{X=1}^n R_{M_X}(t) = \exp\left(-\sum_{X=1}^n \lambda_{M_X} t\right)$$

where

λ_{M_X} = failure rate of that portion of the monitoring system assigned to parameter X, one of n parameters monitored

$R_M(t)$ = reliability (probability of successful operation to time t) of the entire monitoring system.

Then, the probability of detection $P_D(t)$ of any failure in the system to time t is given by

$$P_D(t) = \frac{\lambda_d}{\lambda_t} \exp\left(-\sum_{X=1}^n \lambda_{M_X} t\right)$$

A reliability diagram for this system is shown in Figure 11-3.

The false alarm rate refers to the frequency of failures:

- a. in the sensor/signal processor, which make up the monitoring subsystem, resulting in a command to the switch to change state
- b. in the system selection switch, which results in a state change without a command.

The result of such a failure, in either mode (a) or (b), is that an active, properly functioning unit is switched off-line. (In the single-string case, no switching would be done, but erroneous status reports would go to the user; sacrifice of mission objectives would result.) If redundancy is available, then this situation will not degrade performance immediately, but will likely result in a shortened mission duration. However, if redundancy has been invalidated through previous

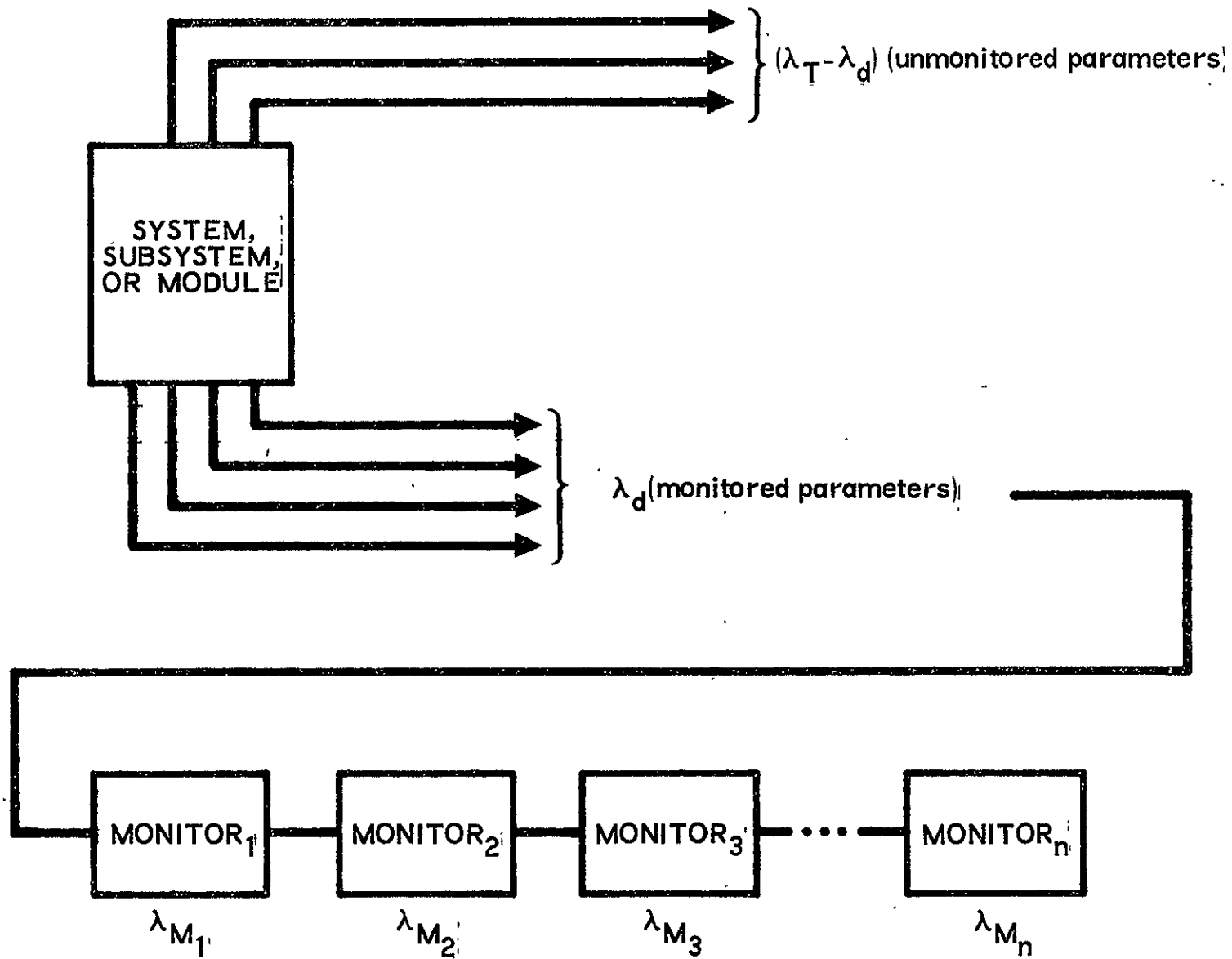


Figure 11-3. Failure Detection System Reliability Model

real failures, this inadvertant switching will degrade mission performance. In this latter case, the use of a "locked in" switch is to be recommended, i. e., a switch that cannot change state if no redundancy is present.

This approach does not consider false alarms resulting from a spurious output of the operating equipment into the monitoring system, such as an out-of-specification electrical spike caused by noise. Here, only hard failures of the monitor or switching subsystems are considered.

As a typical example of how the false alarm probability would be calculated, consider the model shown in Figure 11-4. For this case, the probability of a false alarm is given by

$$P_F(t) = 1 - \exp \left[(-\lambda_{M_F} + \lambda_{S_F})t \right]$$

where λ_{S_F} is the portion of the switch failure rate that is linked to a change of state without a command from the monitor subsystem and λ_{M_F} is that portion of the monitoring system failure rate which is linked to a false indication of failure being generated by the sensor.

The failure detection probabilities and false alarm probabilities are not calculated explicitly in the Reliability models, since monitoring and switching equipment have not been defined in detail. Rather, the total sense/switch failure rate is utilized in the standby mode only (see Table 11-4). Further refinements can be made where feasible (see footnote to Model 1).

11.2 INPUT DATA

The system must be described on a module-by-module basis, where a module is taken as the lowest subdivision of equipment which is a candidate for redundancy. The general model inputs provided by the user are defined in Table 11-1. Data for each piece of equipment selected from the data base is summarized in Table 11-2. Table 11-3 summarizes the data required from each subsystem. Table 11-4 lists the parameters which are fixed (hardwired) in the model.

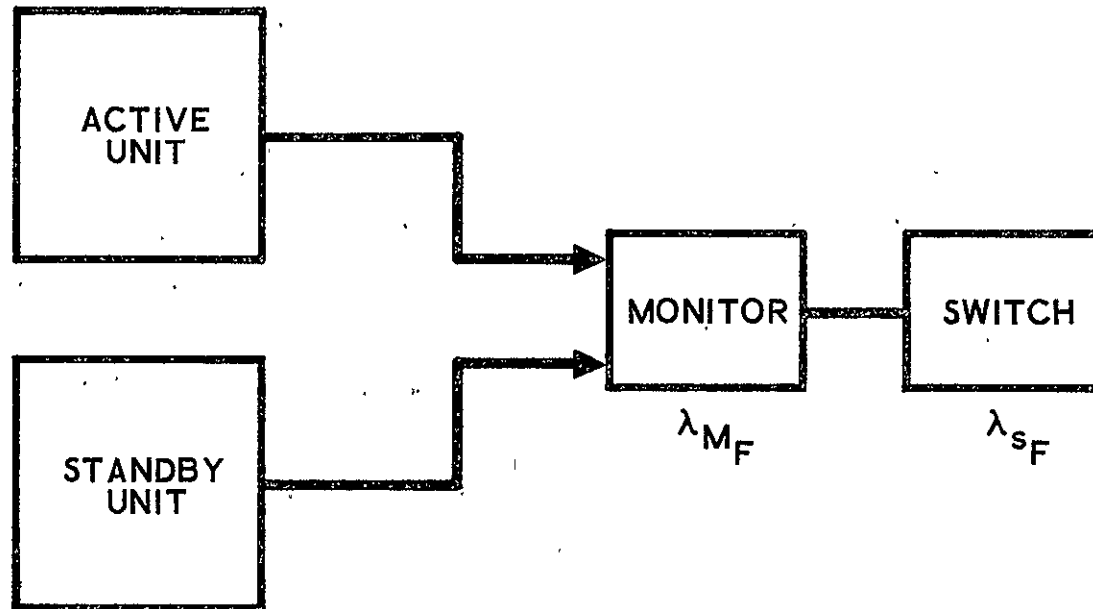


Figure 11-4. Reliability Model for Active Unit with Switched Standby Unit

Table 11-1. Input Data Supplied by User

Parameter Description	Option
Expense option	Weight or cost
Initial expense	Could be zero
Initial reliability	Could be 1.0
Maximum weight permitted	
Mission length	Truncation time
Requirements option	No single-point failures allowed or single-string start
System requirements	R(TRUNC), MMD
Subsystem requirements option	1 : yes; 2 : no
Subsystem requirements.	R(TRUNC)

Table 11-2. Data Supplied by Data Base

Symbol	Name
MODL	Failure model
λ	Module failure rate
μ	Module mean life
σ	Standard deviation of module life
q	Dormancy factor
N_M	Total number of redundant elements in module
EXPM	Module expense (weight or cost)

Table 11-3. Data Supplied by Subsystem Models

Subsystem	Symbol	Name
All	DC	Duty cycle
All	N	Number of active elements (initial)
All	M	Number of standby elements (initial)
All	R	Number of required elements
EP	T_B	Battery operating temperature
EP	D	Battery depth of discharge
EP	C	Battery cycle rate
EP	N_c	Number of cells per battery
EP	R_c	Number of cells required per battery.
S&C	N_o	Orbital mean motion
S&C	θ_{ax}	Roll control deadband
S&C	DT	Thruster on-time
VS	J_x	Roll <u>moment</u> of inertia
VS	D_x	Roll moment arm
APS	F	Low-level thruster force
APS	F_c	Thruster cycles per hour
APS	μ_e	Mean depletion time of expendables
APS	σ_e	Standard deviation of expendables depletion time
APS	D_e	Expendables expense increment

Table 11-4. Parameters Fixed in the Model

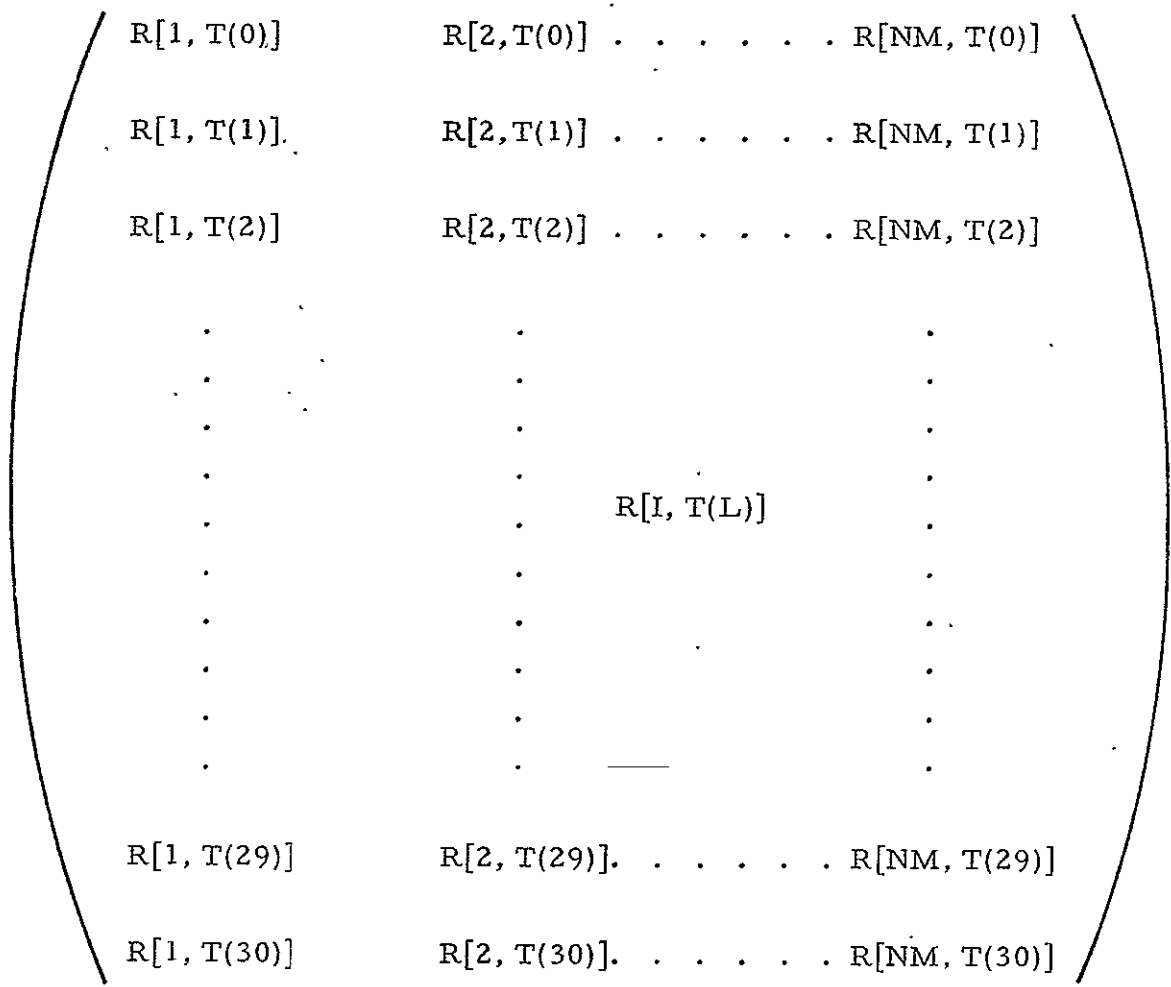
Symbol	Parameter Description	Value
λ_s	Sense/switch failure rate	120 failures/ 10^9 hr
RHOTH	Payoff threshold, MMD	0.2 hr/kg
	Payoff threshold, R(TRUNC)	2.2×10^{-5} /kg
q	Dormancy factor	0.5
Δ_μ	Expendables life increment	2190 hr
Δ_σ	Expendables life standard dev. increment	365 hr
R_C	Number of cells required per battery	$R_C = N_C$

11.3 FUNCTIONAL DESCRIPTION

When the baseline (starting point) system has been defined, the initial MMD determination is made as follows:

For Module I, the appropriate mathematical model is exercised. The value of the reliability function for this module is calculated for one value of time, $T(I)$. This process is repeated for each module in the system and for each of 31 time values ranging uniformly from $t = 0$ to TRUNC. These reliability-time values are stored in an array of dimension 31 by NM, where NM is the number of modules in the system. Call this array RARRAY (Figure 11-5). Then 31 system (spacecraft) reliability values are formed by taking the product of the NM values associated with each time value (the rows of RARRAY). This gives the system reliability-time curve which is then integrated (using a Simpson integration scheme) to give the MMD.

* These parameters are, in general, variable. For present considerations, however, their values are fixed. Changes to this program can consider values other than those listed.



Display of RARRAY

31 Rows, NM Columns

T(30) = Truncation Time (TRUNC)

NM = Number of Modules

Figure 11-5. Reliability Array

The value of system reliability at the time of truncation, $R(\text{TRUNC})$, is simply the product of NM module reliability values at the largest time point.

In the case where a system requirement exists on $R(\text{TRUNC})$ but not on MMD, the array RARRAY would be of dimension one by NM, with the single time point being the time of truncation.

The initial system expense (cost or weight) is merely the sum of the initial expenses of each module.

The incremented MMD and $R(\text{TRUNC})$ values, i. e., the new values resulting from the addition of redundancy, are determined as follows:

The level of redundancy of Module I is increased by adding one element in the appropriate mode (active, standby). The appropriate mathematical model is then exercised and new reliability values are calculated for Module I at 31 time points. The 31 values in RARRAY which correspond to Module I are replaced by the new set of 31 values and system MMD and/or $R(\text{TRUNC})$ values are calculated. The increments in MMD and/or $R(\text{TRUNC})$ resulting from the addition of redundancy to Module I are then calculated and stored in another array (RDELTA). Then the original 31 reliability values are replaced in RARRAY for Module I and the process is repeated for Module I + 1 until all NM modules have been modified in this way, and RDELTA has NM values. When one module, e. g. Module J, is selected as the most profitable module to be made redundant, the 31 reliability values in RARRAY for Module J are replaced by the 31 new values with the level of redundancy for Module J incremented by one element. The new RARRAY thus defined is the starting point for the next iteration.

The system expense increment for each redundant element added is simply the expense of the redundant element. Each value in RDELTA is divided by the expense of the appropriate redundant element. Then RDELTA is a collection of payoff values, the largest of which determines the most profitable module for redundancy.

In the case where R(TRUNC) requirements exist for subsystems, the same redundancy allocation procedure will be followed for each subsystem separately. For example, in subsystem K, there will be NM(K) modules. Then RARRAY (K) will contain NM(K) columns of the system RARRAY, each with 31 rows. Similarly RDELTA (K) will contain payoff values for subsystem K. When subsystem K and all other subsystems have been optimized with respect to redundancy, the system RARRAY will be formed and the system reliability-time curve generated and integrated as before to arrive at system-level R(TRUNC) and MMD.

If system MMD and/or R (TRUNC) requirements are specified by the user, and if redundancy implementation proceeds to the point that the requirements are met, the program stops.

A payoff threshold value, RHOTH, can be specified by the user. This means that the optimization process would stop if the payoff values were less than a predetermined amount (the threshold). For example, if system expense is measured in weight and the system reliability parameter of interest is MMD, then the payoff threshold value would be measured in hours per kilogram. This would correspond to the units of RDELTA. If redundancy implementation proceeds to the point where none of the values of RDELTA exceed RHOTH, the program stops.*

For Module I, a maximum allowable number of elements, initial plus redundant, is specified. When this limit is reached, no further redundancy is possible for Module I. The maximum number of elements allowable in the system is the sum of the modular maxima. When the system maximum has been reached, the program stops.

* It is conceivable that the initial slope could be less than the threshold. To avoid the "can't get started" situation, the slope threshold check could be suspended until some redundancy has been implemented, say 20 percent of the available expense (or some appropriate number), and then the slope can be checked at the last step. Or, alternatively, the slope can be tested at the 5th (or 10th or 20th) increment.

When one or more of the above tests results in program stoppage, the final configuration is printed out, including the number of elements of each module, modular and system expense, system MMD, and the system reliability-time curve. See Figures 11-6 and 11-7 for typical flow.

11.4 MATHEMATICAL MODELS

The mathematical models or algorithms mentioned in Paragraph 11.3 are described in the following paragraphs. The selection of a particular model for a given module depends on the nature of the hardware in the module, and on the operating mode for redundant elements within the module. The parameters used in the exercising of the models (failure rates, duty-cycles, etc.) are described in Tables 11-1 through 11-

The reliability-time function of a given module can normally be calculated assuming uniform operation with a given duty cycle throughout the system mission. There are some equipments, however, which are required to operate only during the first few hours or days of a mission and are not used thereafter. In this "pulse operation" case, the 31 reliability values will be determined as follows:

$$\begin{aligned} R[I, T(0)] &= 1.0 \\ R[I, T(L)] &= R(T_P), L = 1, 2, \dots, 31 \end{aligned}$$

where

T_P is the pulse width (duration of operation) and the R-values are determined from the appropriate mathematical models as defined below.

11.4.1 Model 1

Model 1 is a general algorithm for calculating the reliability of a module with N elements active, M standby, R required, with

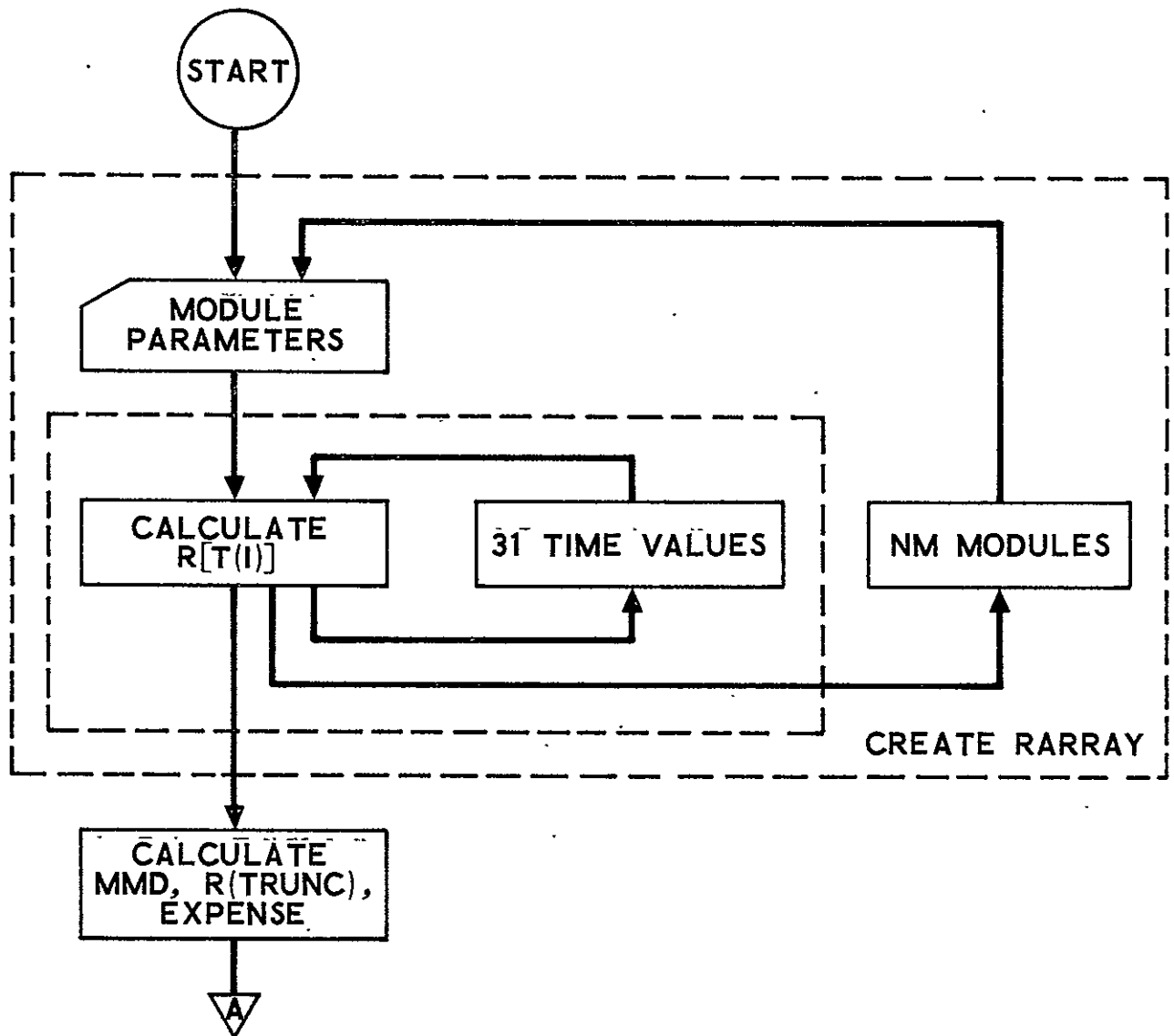


Figure 11-6. Typical Reliability Model Flow

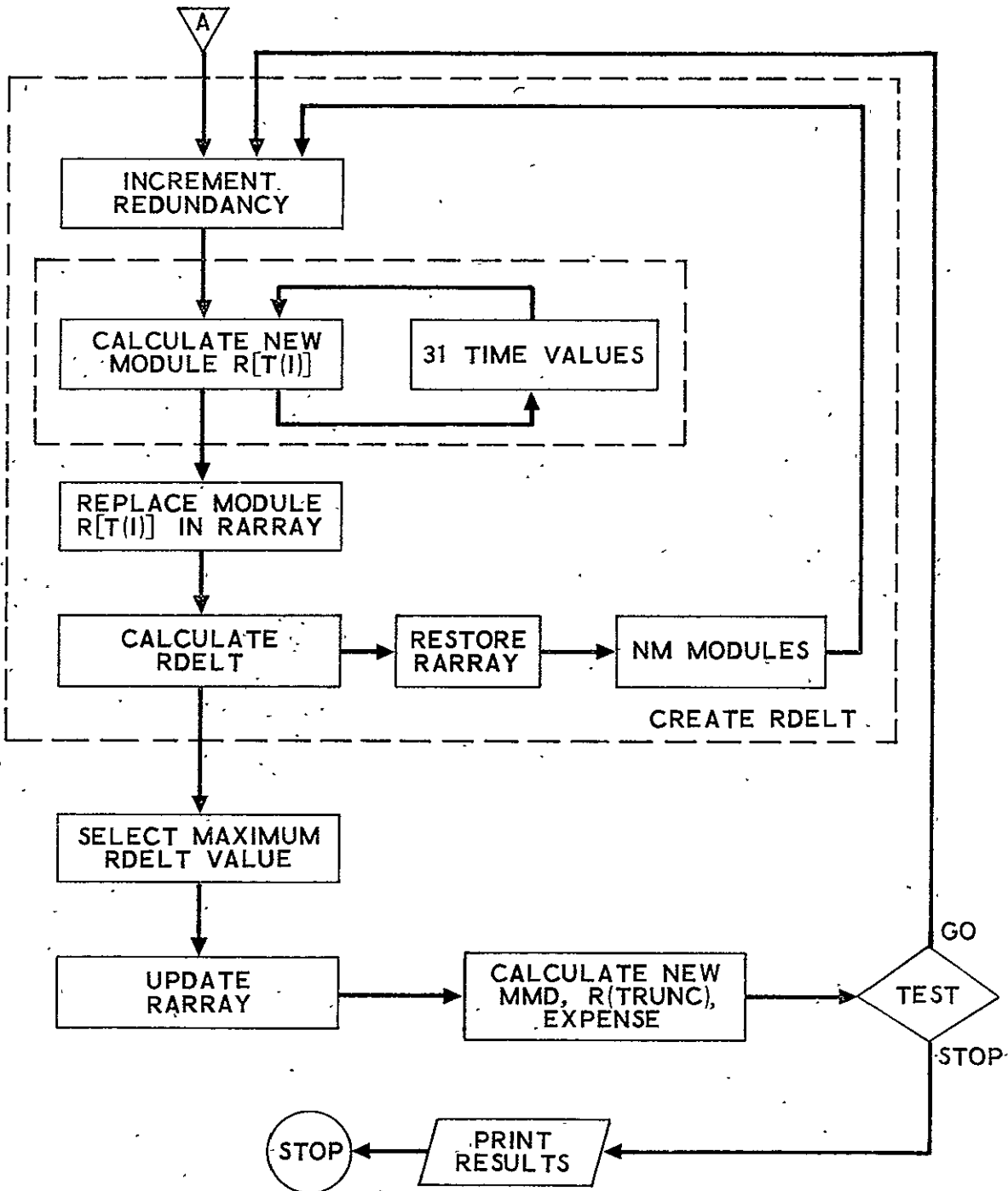


Figure 11-6. Typical Reliability Model Flow (Continued)

duty cycling.* It is applicable to elements with exponential reliability distributions. When Model 1 is used in implementing redundancy, a test is made on q (the dormancy factor). If $q = 1.0$, redundancy is implemented in the active mode. If $q \neq 1.0$, redundancy is implemented in the standby mode.

$$R[L, T(I)] = 1 - \sum_{k=0}^{R-1} \text{Pr}(k)$$

$$Q \equiv q + \frac{\lambda_s}{\lambda}$$

$$\bar{\lambda} = \lambda [\text{DC} + (1 - \text{DC}) q]$$

For $Q > 0$

$$\text{Pr}(k) = A_k \left\{ \sum_{i=k}^N \frac{e^{-i \bar{\lambda} t}}{B_i} + e^{-N \bar{\lambda} t} \sum_{j=1}^M \frac{e^{-j Q \lambda t}}{C_j} \right\}$$

* Model 1 considers, for constant failure rate devices, equipments with

1. N units active, of which R are required;
2. M units in standby, with $0 \leq q \leq 1$.
3. Duty cycling of active units as

$$\lambda_{\text{effective}} = \lambda_{\text{active}} [\text{DC} + (1 - \text{DC}) q]$$

4. Sensing/switching equipment, considered to be in the standby mode and in series with each standby unit (this is an approximation to reality).
5. The implementation philosophy is to switch-on (activate) a standby unit immediately upon failure of an active unit (i. e., maintain hot spares) rather than to allow N to shrink to R before activation of standby unit.

This model could be made more accurate and complete by a more comprehensive consideration of switching implementation; however, this is a second-order refinement from the standpoint of accuracy.

Where

$$\begin{aligned}
 A_k &= (-1)^k \frac{N!}{k!} [(N+Q)(N+2Q) \cdots (N+MQ)] \\
 B_i &= (-1)^i (i-k)! (N-i)! [(N-i+Q)(N-i+2Q) \cdots (N-i+MQ)] \\
 C_j &= (-1)^{N+j} j! (M-j)! Q^M [(jQ+1)(jQ+2) \cdots (jQ+N-k)]
 \end{aligned}$$

For $Q = 0$

$$\text{Pr}(k) = A_k \left\{ \sum_{i=1}^{N-k} B_i e^{-(N-i)\bar{\lambda}t} + \frac{e^{-N\bar{\lambda}t}}{k!} \sum_{j=1}^{M+1} \frac{C_j r^{j-1}}{(j-1)!} \right\}$$

Where

$$\begin{aligned}
 A_k &= (-1)^{N-k} N^M N! \\
 B_i &= \frac{(-1)^i}{i^M i! (N-k-i)!} \\
 C_j &= \lambda^{j-1} \sum_{\ell=1}^{N-k} \frac{(-1)^{\ell-1}}{\ell^{(M+2-j)} (N-k-\ell)! (\ell-1)!}
 \end{aligned}$$

Where

$R [L, T(I)]$ = Reliability of module L at time T (I)
 $\text{Pr}(k)$ = Probability that module L is in the k^{th} operability state
 Q = Effective dormancy factor
 $\bar{\lambda}$ = Effective active failure rate, when duty cycled

11.4.2 Model 2

This model is used to calculate the reliability of a module with N elements active and R required. It is applicable to elements with normally distributed lifetimes, with mean μ and variance σ^2 .

$$R[L, T(I)] = \sum_{X=R}^N \binom{N}{X} (RN)^X (1-RN)^{N-X}$$

where RN is The elemental reliability function, given by

$$RN = 1 - P \left[\frac{T(I) - \mu}{\sigma} \right]$$

P(z) is Cumulative normal distribution function given by

$$P(z) = \frac{1}{\sqrt{2\pi}} \int_{-\infty}^z e^{-y^2/2} dy$$

Note that

$$P(z) \cong 0.5 + \frac{Z}{2(Z^2 + 16V^8)^{1/2}}$$

where

$$V = \frac{1.18145}{1.57926 + 0.0594375 Z^2 + 0.00390625 Z^4}$$

11.4.3 Model 3

Model 3 is applicable to a binomial battery module, where there are N batteries active with R required. The battery cell reliability function is derived empirically in Reference 11.1.

$$R[L, T(I)] = \sum_{K=R}^N \binom{N}{K} (RB)^K (1 - RB)^{N - K}$$

where

RB = The reliability of a single battery, with R_c out of N_c cells required; RB is given by:

$$RB = \left[\sum_{X=R_c}^{N_c} \binom{N_c}{X} (RW)^X (1 - RW)^{N_c - X} \right] \left[1 - P\left(\frac{T(I) - 43800}{8760}\right) \right]$$

Here, RW is the cell reliability given by:

$$RW = \text{EXP} \left[- (C \cdot T(I) / AB)^{BB} \right]$$

where

$$AB = \text{EXP} \left[- 11.380958 + 0.23896921 T_B - 0.54986583D - 0.00050646174 T_B^2 + 0.019307737 D^2 - 0.0002374105D^3 \right]$$

$$BB = \text{EXP} [-138.10332 + 0.95927099 T_B - 0.19804227 D \\ - 0.0016717786 T_B^2 - 0.0019619976 D^2 \\ + 0.0011242688 T_B D]$$

$$= \text{TRUNC} \cdot \frac{N_0}{2\pi}$$

$$\text{and } P(Z) = \frac{1}{\sqrt{2\pi}} \int_0^Z e^{-y^2/2} dy \text{ as before.}$$

11.4.4 Model 4

This model is applicable to a module whose life is normally distributed. It is distinct from Model 2 because redundancy is added by extending the mean life and the variance. This condition is applicable when expendables are incremented:

$$R [L, T(i)] = 1 - P \left[\frac{T(i) - \mu_e}{\sigma_e} \right]$$

where

$$P(Z) = \frac{1}{\sqrt{2\pi}} \int_{-\infty}^Z e^{-y^2/2} dy$$

Redundancy is implemented by adjusting μ_e and σ_e as follows:

$$\mu_e \text{ is replaced by } \mu_e + k \cdot \Delta_M$$

$$\sigma_e \text{ is replaced by } [\sigma_e^2 + (k \cdot \Delta_\sigma)^2]^{1/2}$$

where k is the number of increments added.

11.4.5 Model 5

This model is an extension of Model 1 for those equipments whose failure rate is based on the number of operational cycles, rather than the number of operational hours. A conversion factor (number of cycles per hour), which is derived from the design considerations of the hardware subsystems, is used to convert from the cycle base to the time base. Then Model 1 is exercised to determine the reliability-time function.

This model translates the failure rate:

$$\lambda = \lambda_c \cdot F_c$$

where

λ_c is in units of failures per 10^9 cycles

F_c is in units of 10^9 cycles per 10^9 hour.

λ , then, is in units of failures per 10^9 hours.

The algorithm for this model is that of Model 1, using λ , the translated failure rate.

For thrusters, F_c is given by
$$F_c = (14.1) \frac{F \cdot D_x \cdot DT}{J_x \cdot \theta_{ax}}$$

11.4.6 Model 100

It may be desirable to optimize the redundancy of a system as described above, and then determine the reliability or MMD values when the entire system is made redundant in an active mode. Model 100 responds to a program executive command to calculate reliability and MM for such redundant systems where one is required and one or two are present.

$$R [T(I)] = 1 - [1 - R_s [T(I)]]^2$$

where

$$R_s [T(I)] = \text{Reliability of single system at time } T(I)$$

It may be that a requirement exists for the elimination of all single-point failures, i. e., the requirement may be that every functional element in the system have at least one backup element. This requirement could be satisfied by requiring redundancy in all modules at the start. A user option permits selection of initial redundancy or bare bones system.

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12. COST*

12.1 BACKGROUND

Previous cost models that use parametric estimating methods usually have been subsystem oriented. A deficiency to such estimating methods is that independent variables in a cost-estimating relationship (CER) always influence cost in the same way regardless of the makeup or numbers of detail components in a subsystem. Thus, if weight is the independent variable, which is frequently the case, each additional pound added or subtracted always changes cost by a fixed amount despite the fact that the individual components that are changed may vary considerably in their cost per pound. An estimating system that relies on a finer grain of detail, i. e., on major assemblies or components, should help to alleviate the problem. The cost model described in this section is part of an overall model designed to meet the requirements for such an estimating system.

12.2 COST DATA BASE

A substantial amount of cost and related technical data has been collected, analyzed, and used in numerous studies for NASA over the past several years. Such studies, commencing with the STS, have required the development of cost models for launch vehicles and satellite systems. Satellite data, covering both NASA and DOD satellite programs, were originally oriented to produce subsystem cost information; however, during the course of data collection, it was apparent that for many programs component information could also be obtained with little additional effort. Moreover, certain studies for DOD required an examination of component cost, quantity and related technical data. As a consequence, the satellite cost data base at Aerospace grew to include considerable amounts of component information. Such information forms the raw data base used for the Systems Cost/Performance Model.

*The term "component" as used in this section refers to subsystem components, i. e., assemblies.

The raw cost data referred to above are run through a computer model to guarantee comparability with respect to (1) adjustments for yearly price changes, (2) allocations to components of program management, system engineering and integration costs, (3) separation of recurring and non-recurring costs, and (4) cost-quantity adjustments based on total numbers of full prototype and flight vehicles.* After application of the computer program each identifiable component is shown in terms of weight, quantity per satellite, and estimates of engineering design and development cost and average unit cost (normalized to cumulative average for the first 5 units including prototypes). An example of the computer program output covering a selected sample of components is contained in Table 12-1. Finally, the computer cost figures are adjusted manually to eliminate average amounts of program management, system engineering, integration and quality control costs prior to use in CER derivation. Each of the steps mentioned above are elaborated upon in the next section on data adjustment.

12.3 COST DATA ADJUSTMENTS

To maintain comparability, certain adjustments are routinely performed on raw cost data. In many instances, these adjustments have been performed with the aid of computer programs; in other cases, manual computations are used.

12.3.1 Yearly Price Changes

Actual cost data come from numerous programs that have occurred at various times in the past. Price indexes constitute the principal means of adjusting data to a common base; for this study the base year is 1971. Thus, all basic cost that is used either directly within the model or in CERs that support the cost model is input in terms of constant 1971 dollars. Of course, the output of the model can be expressed in other base

* Three satellite programs, Tiros-M, DSP and DSCS-II, currently are fully converted to a component cost data output system. Portions of the data output on five satellite programs will provide partial component information; Nimbus, OGO, Vela, Vasp, Pioneer. Four other programs, OAO, Lunar Orbiter, ATS and Program 191 require substantial analysis and recoding to produce component data.

Table 12-1. Example of Computer Output for Selected Components.

	<u>Control Elec.</u>	<u>Damper</u>	<u>Reaction Wheel</u>	<u>APT Camera</u>	<u>AVCS Camera</u>
Hours					
Des. Eng.	34580	2324	9437	5956	9962
Dev. Test	28292	1099	13254	11502	31145
Unit Eng.	8179	549	2232	1409	2356
Unit Prod.	17644	3845	585	7587	4009
Cost (1971 Dollars)					
Des. Eng.	705480	44906	191899	100461	179276
Dev. Test	385797	12859	316289	134707	401813
Dev. Matl.	78158	317	18662	32098	114902
Total Devel. Cost	1169435	58082	526850	267266	695991
Unit Eng.	166878	10622	45393	23763	42407
Unit Prod.	239524	52934	9818	97355	57076
Unit Matl.	109178	17258	327	49945	16756
Total Unit Cost	515580	80814	55538	171063	116239
Unit Cost/Pound	42965	5772	1322	8772	7749
Component Wt.	12	14	42	19	15
Qt. Per Sat.	1	1	1	2	2

years at the discretion of the user, e. g., 1975 dollars. The price index procedures used in all such adjustments are based on the method described in Reference 12-1. The actual application of price index factors is done automatically with the aid of a computer program that also makes the adjustments described in the next three subsections.

For the cost model, projections of satellite prices in terms of future years are usually needed, and the same rate of increase in price adjustments for past years is planned for use in projecting the future. The text table below contains price index factors that will be used in the model for the years 1971 through 1975.

<u>Fiscal Year</u>	<u>Price Index Factor</u>
1971	1.000
1972	1.050
1973	1.100
1974	1.176
1975	1.227

12.3.2 Allocations to Component Cost

The raw data are organized so that three types of cost are an input to the computer program: (1) component identified, (2) subsystem identified, and (3) other cost. Component identified cost refers to all costs that can be directly related to a particular component or assembly through a work breakdown or job order system of cost accounting. Such accounting systems are used by most contractors and identify varying proportions of cost from program to program. Similarly, certain other effort can be identified only by subsystem. Examples would be electrical power subsystem qualification testing, reaction control system design or inspection of communications equipment. Still other cost falls into overall system categories such as receiving inspection, system checkout and final assembly, or system engineering.

The computer program makes two allocations, the first to prorate subsystem identified costs to components using component identified cost as a base and the second to allocate other costs, such as program management and system engineering, to the revised component base. Allocations are made on the basis of like cost elements, i. e., engineering (hours and dollars), development (hours and dollars), production (hours and dollars) and material cost.

Allocations are made so that all costs are attributable to components or subsystems in a systematic way. Differences in accounting practices of various contractors are thereby minimized and comparability of important cost influencing categories is improved.

12.3.3 Separation of Recurring and Nonrecurring Cost

With the exception of engineering, many functional cost elements lend themselves to categorization into nonrecurring and recurring cost as illustrated by the cost data format used for inputs to the computer cost data program (see Fig. 12-1). For example, DEV cost includes all manufacturing type cost associated with support of development and development testing (nonrecurring) and MFG cost includes all manufacturing-type cost for production of prototype and flight hardware (recurring). Typically, contractor job order or work breakdown systems identify such costs. However, it is difficult to segregate engineering costs by such categories because contractors define DDT&E (nonrecurring) and sustaining or production (recurring) engineering in different ways. Moreover, job order or work breakdown classifications relating to engineering functions are not applied or used uniformly on different satellite programs.

To maintain consistency in the treatment of engineering categorization two assumptions have been made: (1) engineering costs increase if a larger quantity of satellites is manufactured and flown on a given satellite program, i. e., costs are related to quantity produced, and (2) the cost-quantity relationship is quantifiable in terms of a log-linear cumulative average function with a b value of -0.515, i. e., a 70-percent curve. Thus,

INPUT FORM FOR COST DATA BASE
(KEYPUNCH ALL UNDERLINED ITEMS)

¹ PROJECT ⁹ _____ TASK ²⁵ _____

CROSS REFERENCE ⁴⁹ _____ TOTAL COST ⁵⁷ _____

SUBCONTRACT CIRCLE DEVELOPMENT ENG ⁶⁵ D COMBINED D&P MFG ⁶⁷ M
 CIRCLE YES ⁶⁵ Y ONE: PRODUCTION ENG ⁶⁵ P (CIRCLE IF APPLICABLE)
 ONE: NO ⁶⁵ N COMBINED D&P ENG. ⁶⁵ C

SUBCONTRACT AND MATERIAL BURDEN ⁷³ _____

	HOURS	DIR LAB \$	BURDEN	OV PR	DIR CHG	MATL	PURCHASED	SUBCONTRACT	G&A
¹ ENG	9	17	25	33	41	49	57	65	73
¹ DEV	9	17	25	33	41	49	57	65	73
¹ TLG	9	17	25	33	41	49	57	65	73
¹ MFG	9	17	25	33	41	49	57	65	73
¹ QC	9	17	25	33	41	49	57	65	73
¹ CLER	9	17	25	33	41	49	57	65	73
¹ OTHER	9	XXXXXXXX	XXXXXXXX	33	41	49	57	65	73
¹ TOTAL	9	17	25	33	41	49	57	65	73

12-6

X-5893

Figure 12-1. Input Form for Cost Data Base

using such a function, the engineering required for the first unit produced can be estimated, which is also a surrogate for Design Engineering. Similarly, the engineering required for the total quantity produced (including prototype) less Design Engineering will give Production Engineering. Little data exist to verify the correctness of the 70-percent curve; however, generally so few satellites are produced for a given project that any error in assumption about the curve slope will have negligible effects on cost estimates.

12.3.4 Cost-Quantity Effects

A particular satellite program may have any number of flight and prototype units. For comparability, costs must be adjusted to a common base. The base judged to be most appropriate is five units because that figure tends to represent a reasonable average of the satellite programs analyzed. (The theoretical first unit is often selected in other cost analysis work; however, the average cost for five units is so close to average actual experience it is deemed the most appropriate for satellites.)

Manufacturing effort typically follows a cost-reduction function when related to quantity produced. Because of the relatively few satellites normally produced as part of a given program, no large body of data exists that would aid in accurately measuring such cost-reduction functions. A 90 percent log-linear cumulative average curve (i. e., $b = -0.152$) has been assumed, based on preliminary data from the OGO program. Again, the exactness of the cost-quantity slope should not materially affect any cost estimates because the quantity of satellites on a program is generally small.

12.3.5 Adjustments for Program Management and System Engineering

The computer program allocates overall system cost categories (i. e., Program Management, System Engineering and Integration, and Quality Control) to components as previously described; the purpose of the allocations is to achieve comparability of total component cost from various programs. For the Cost Model, such categories as Program Management,

etc., are treated separately; thus, any component data base must exclude them when CERs are to be developed. Accordingly, the computer-adjusted costs described above are reduced by applicable percentages that are based on an examination of data from all available satellite programs. Engineering, Development (Test and Evaluation), and Production costs are reduced by 34.2, 30.5 and 27.5 percent, respectively.

12.4 COST MODEL DEVELOPMENT

After all of the adjustments are made, the data are ready for use in developing the CERs needed by the cost model. CERs are applied in two ways; one directly within the cost model and the other outside the model. Certain items within the model are basically a function of the types of components that the model identifies as requirements. These items are Structure, Thermal Control, Solar Arrays, Power Control Equipment, Power Converters, Wiring Harness, and Attitude Control Propulsion Tankage and Feed Systems. Because the requirements for these items tend to be variable, it is prudent to allow costs for them to be changeable depending on the independent variables that relate to cost. Costs for all other components are an input to the program cost data bank and the cost comes from component CERs used outside of the model. Further, the use of CER generated cost, rather than contractor or vendor cost, serves to dispel any possible concern over the proprietary aspect of the program cost data base. For convenience, components are classified as CER or catalog within the model; the former denotes internal CER cost items, the latter are cost data bank items.

12.4.1 CER Categories

Three categories of cost are required for CER items and input for catalog items: Design Engineering, Test and Evaluation and Unit Production. (A fourth category, Production Engineering, is a function of Design Engineering and is not a direct input or CER requirement.) Besides individual component cost, the model must generate the cost of Program Management and related categories, Ground Support Equipment, Satellite Launch

Cost, and Fee or Profit. Factors, percentages and all of the component CERs stored within the model are required for estimating such items. Finally, factors that relate the effect of redundancy on cost are required as part of the model.

- a. Design Engineering. Design Engineering cost is defined as the cost of all engineering effort connected with the planning, design, development and qualification of a particular component or assembly. As previously explained, contractor accounting systems cannot guarantee comparability of costs; thus, a standard procedure is applied to the cost of engineering on past programs to achieve a split between nonrecurring (Design Engineering) and recurring (Production Engineering). Even though comparability of costs should be enhanced by such a procedure, variability must inevitably be expected. Factors that account for cost variability are rework, redesign schedule alterations, and amounts of design inheritance from similar components previously developed. When CERs are applied, such variability tends to be normalized.
- b. Test and Evaluation. Testing encompasses all developmental and qualification effort required as part of DDT&E. Test and evaluation cost includes all nonengineering labor and materials connected with building tooling and test equipment, manufacturing parts for test and conducting development and qualification testing.
- c. Unit Production. The cost of all manufacturing labor and materials used to fabricate, assemble, checkout and acceptance test full prototype and flight units is defined as Unit Production cost.

12.4.2 CER Development - Generalized Procedure

The procedure for developing CERs consists of examining all pertinent data and determining if any relationship exists between a component's physical or performance characteristics (explanatory variables) and its cost. The engineering portion of the over-all model produces weight, quantity required per satellite and other physical and performance data that can be used as explanatory variables; thus, a first step is to check for relationships between the engineering model output and cost. Next, the relationship, if it appears useful, must be quantified. Typically a formula of the type $C_E = K_E \times E^b$ is used to quantify the relationship where C_E is a

cost category (Design Engineering for example), K_E is a constant for Design Engineering, X is the explanatory variable (weight for example), and b_E is the exponent of the variable. In addition, factors are introduced into the equation to give effect to the selection of different types of satellites or subsystems, for example, body mounted versus paddle mounted solar cells would require factors equal to 1 and 4, respectively.

Because of the relatively small size of the sample of data points currently available, it is not possible to apply rigorous statistical procedures in the development of CERs. Accordingly, manual plots on log-log graphs of costs versus explanatory variables have been used. When the cost data base is augmented, it may be possible to improve the CERs (e. g., increase the accuracy or resort to greater stratification by satellite or subsystem type.) and it may be possible to apply regression methods to the data.

One way of expanding the data base is to convert existing detailed cost data at Aerospace to a component format through the process of recoding contractor job number or work-breakdown identifiers. Another method is to obtain cost data for additional completed projects. A third method is to use contractor estimates of cost for future satellite components. (Such estimates, however, are likely to be unreliable if past experience is a guide.)

Within the overall model, the program cost data bank includes an estimated cost for each component (or assembly) that is cataloged. In addition to catalog items, certain assemblies or subsystems are estimated by CERs that are internal to the model, as previously explained. All costs, whether catalog or CER type estimates, exclude System Engineering and Integration, Quality Control and Program Management. Such costs are calculated by means of average percentages applied to the basic component total cost categories. The percentages were derived from an examination of all available satellite programs. Table 12-2 provides all percentages, and the cost bases to which they are applied, that are used for estimating system-oriented costs.

Table 12-2. Percentage Allocations of System Engineering, Quality Control and Program Management

<u>Base Cost Category</u>	<u>System Eng. & Integ.</u>	<u>Quality Control</u>	<u>Program Management</u>
Design engineering	32	1.5	19
Test and evaluation	27	14	2
Production engineering	32	1.5	19
Production (mfg.) unit	22	14	2

The net effect is that the original cost data (examples are shown in Table 12-1) are adjusted downward to subtract such system costs prior to CER development and the model adds back in the costs by applying the noted percentages to the CER derived cost base.

12.4.3 CER Development*

The Cost Model CER items, currently made up of Structures, Thermal Control, Solar Array, Power Control Equipment, Power Converters, Wiring Harness and Propulsion Tankage and Feed System, are handled internally by the model; all other items, i. e., catalog items, are input to the model cost data base. As previously explained, all of the costs are based on CERs; however, at this juncture either insufficient data exist or no data are available to allow development of CERs for certain components. Preliminary estimates have been made for data base catalog items where no CER is currently available, thereby allowing the model to be run.

The subsections that follow contain descriptions of each CER (or concludes that because of insufficient data no CER can be developed at this time). Component CERs are grouped by major subsystem and include the equation plus pertinent remarks concerning factors that should be applied when different types of satellites are being considered. Log-log plots of the data for each CER are contained in Figures 12-2 through 12-63 and are located in the back of Section 12 for easy reference. Three CERs are

*The Cost Model CERs in the Computer Program are preliminary versions in that insufficient time remained in the current contract period to code and debug the CERs provided in this document.

needed for each component - Design Engineering, Test and Evaluation, and Unit Production. All CERs are considered in terms of constant 1971 dollars. The following listing of CERs shows C_E , C_T and C_P , Design Engineering, Test and Evaluation, and Unit Production, respectively. The independent variable is often component weight (denoted by W); otherwise the variable will be mentioned. When differences in satellite characteristics appear to influence costs, an adjustment factor F is used. Where information to develop a CER is currently insufficient, the notation NA (not available) is given.

a. Stabilization and Control

1. Sun Sensor Assemblies (Figs. 12-2 to 12-4)

$$C_E = 254,845 W^{.677} \quad (12-1)$$

$$C_T = 160,000 W^{.447} \quad (12-2)$$

$$C_P = 7,153 W^{.521} F_P \quad (12-3)$$

The plotted data (Fig. 12-4) suggest that the type of control system applicable to a particular satellite can help to identify cost. The factor F_P is used to adjust cost; it is 1.0, .3.3, or 7.5 for spin, 3-axis or oriented solar paddle systems, respectively.

2. Control Electronic Assemblies (Figs. 12-5 to 12-7)

$$C_E = 183,910 W^{.633} \quad (12-4)$$

$$C_T = 122,780 W^{.633} \quad (12-5)$$

$$C_P = 29,420 W^{.669} \quad (12-6)$$

3. Earth Sensor Assemblies (Figs. 12-8 to 12-10)

$$C_E = 36,000 W^{.514} F_E \quad (12-7)$$

where $F_E = 13.5$ if satellite is 3-axis controlled or has paddle-mounted solar arrays; $F_E = 1.0$ for spin-stabilized satellites.

$$C_T = 43,440 W^{.647} F_T \quad (12-8)$$

where $F_T = 1.0$ for spin satellites, or 2.4 for 3-axis of oriented paddle arrays.

$$C_P = 11,550 W^{.767} F_P \quad (12-9)$$

where $F_P = 1.0$ for spin, 2.2 for 3-axis and 5.0 for oriented paddle arrays.

4. Rate Gyro Assemblies (Figs. 12-11 to 12-13)
- $$C_E = 134,780 W \cdot 630 \quad (12-10)$$
- $$C_T = 51,420 W \cdot 710 \quad (12-11)$$
- $$C_P = 22,420 W \cdot 707 \quad (12-12)$$
5. Reaction Wheel Assemblies (Figs. 12-14 to 12-16)
- $$C_E = 16,730 W \cdot 745 \quad (12-13)$$
- $$C_T = 9,350 W \cdot 855 \quad (12-14)$$
- $$C_P = 7,690 W \cdot 632 \quad (12-15)$$
6. Control Moment Gyros NA
7. Star Sensors NA
8. Nutation Dampers NA

b. Auxiliary Propulsion

1. Thrusters (Figs. 12-17 to 12-19)
- $$C_E = 150,000 W \cdot 748 \quad (12-16)$$
- $$C_T = 150,000 W \cdot 748 F_T \quad (12-17)$$
- $$F_T = 1.00 \text{ when thrust is greater than or equal to thruster weight}$$
- $$= 0.25 \text{ when thrust is less than thruster weight}$$
- $$C_P = 11,490 W \quad (12-18)$$
2. Tanks (includes fill and drain valves and relief valves) (Figs. 12-20 to 12-22)
- $$C_E = 129,200 TWW \cdot 272 F_E \text{ (dual-spin satellites)} \quad (12-19)$$
- $$F_E = 0.507$$
- $$TWW = \text{total Reaction Control System wet weight}$$
- $$C_E = 545,640 TWW \cdot 222 F_E \text{ (for all other satellites)} \quad (12-20)$$
- $$F_E = 0.268$$
- $$C_T = 24,160 TDW \cdot 675 F_T \quad (12-21)$$
- $$F_T = 0.325 \text{ for dual spin satellites}$$
- $$F_T = 0.619 \text{ for all other satellites}$$
- $$TDW = \text{total Reaction Control System dry weight}$$

$$C_P = 14,000 \text{ TDW} \cdot 0.668 F_P \quad (12-22)$$

$$F_P = 0.615 \text{ for dual spin satellites}$$

$$F_P = 0.840 \text{ for all other satellites}$$

Detail cost data concerning Reaction Control tankage (including fill, drain and relief valves) is currently available from only two satellite programs. In order to broaden the data base, cost and weight information for total reaction control system was used as a reference. Ratios of tank costs to total system costs for the two satellites were calculated and in turn were applied to total reaction control costs to derive the tank CERs. This procedure allowed the use of data from eight programs.

c. Data Processing

1. Digital Telemetry Units (Figs. 12-23 to 12-25)

$$C_E = 43,170 W \cdot 0.668 F_E \quad (12-23)$$

$$F_E = 1.0 \text{ for low data rates}$$

$$F_E = 2.7 \text{ for high data rates}$$

$$C_T = 28,780 W \cdot 0.516 F_T \quad (12-24)$$

$$F_T = 1.0 \text{ for comsats}$$

$$F_T = 1.9 \text{ for all other satellites}$$

$$C_P = 6,894 W \cdot 0.687 F_P \quad (12-25)$$

$$F_P = 1.0 \text{ for comsats}$$

$$F_P = 4.5 \text{ for planetary satellites}$$

$$F_P = 3.0 \text{ for other satellites}$$

2. Tape Recorders (Figs. 12-26 to 12-28)

$$C_E = 77,720 W \cdot 0.621 \quad (12-26)$$

$$C_T = 71,690 W \cdot 0.557 \quad (12-27)$$

$$C_P = 15,730 W \cdot 0.548 \quad (12-28)$$

3. General Purpose Processors NA

d. Electrical Distribution

1. Command Decoding and Distribution Units
(Figs. 12-29 to 12-31)

$$C_E = 140,390 W \cdot 549 \quad (12-29)$$

$$C_T = 71,540 W \cdot 628 \quad (12-30)$$

$$C_P = 43,914 W \cdot 622 \quad (12-31)$$

2. Wiring Harness (Figs. 12-32 to 12-34)

$$C_E = 3,920 W \cdot 715 F_E \quad (12-32)$$

$F_E = 1.0$ for pure spin satellites

$F_E = 5.0$ for all other satellites

$$C_T = 6,000 W \cdot 585 F_T \quad (12-33)$$

$F_T = 1.0$ for pure spin satellites

$F_T = 5.0$ for all other satellites

$$C_P = 2,050 W \cdot 745 F_P \quad (12-34)$$

$F_P = 1.0$ for pure spin satellites

$F_P = 3.0$ for all other satellites

e. Communications

1. Antenna (Figs. 12-35 to 12-37)

$$C_E = 67,970 W \cdot 593 \quad (12-35)$$

$$C_T = 49,850 W \cdot 699 \quad (12-36)$$

$$C_P = 12,000 W \cdot 569 \quad (12-37)$$

2. Receivers (Figs. 12-38 to 12-40)

$$C_E = 39,500 W \cdot 467 \quad (12-38)$$

$$C_T = 69,060 W \cdot 6569 \quad (12-39)$$

$$C_P = 22,190 W \cdot 329 \quad (12-40)$$

3. Diplexers (Figs. 12-41 to 12-43)

$$C_E = 10,940 W \cdot 454 \quad (12-41)$$

$$C_T = 6,680 W \cdot 645 \quad (12-42)$$

$$C_P = 7,820 W \cdot 619 \quad (12-43)$$

- 4. Baseband Assembly Units NA
- 5. Transmitters NA
- 6. Command Signal Conditioners NA

f. Electrical Power

1. Batteries (Figs. 12-44 to 12-46)

$$C_E = 31,000 \text{ AMPH} \cdot^{.276} \text{ for battery systems with capacities } \leq 15 \text{ amp-hr} \quad (12-44)$$

AMPH = amp-hr of total battery system in satellite

$$C_E = 42,250 \text{ AMPH} \cdot^{.582} \text{ for batteries with capacities } > 15 \text{ amp-hr} \quad (12-45)$$

$$C_E = 99,900 \text{ AMPH} \cdot^{.589} \text{ for 3-axis controlled and oriented paddle satellites} \quad (12-46)$$

$$C_T = 32,480 \text{ W} \cdot^{.469} F_T \quad (12-47)$$

$F_T = 6.0$ for 3-axis controlled or oriented paddle satellites

$F_T = 1.0$ for all other satellites

$$C_P = 11,470 \text{ W} \cdot^{.400} F_P \quad (12-48)$$

$F_P = 4.0$ for 3-axis and oriented paddle satellites

$F_P = 1.0$ for all other satellites

2. Power Control Equipment (Figs. 12-47 to 12-49)

$$C_E = 51,383 \text{ W} \cdot^{.587} F_E \quad (12-49)$$

$F_E = 3.2$ for 3-axis and oriented paddle satellites

$F_E = 1.0$ for all other satellites

$$C_T = 87,500 \text{ W} \cdot^{.301} F_T \quad (12-50)$$

$F_T = 3.1$ for 3-axis and oriented paddle satellites

$F_T = 1.0$ for all other satellites

$$C_P = 36,660 W \cdot 182 F_P \quad (12-51)$$

$F_P = 4.0$ for 3-axis and oriented paddle satellites

$F_P = 1.0$ for all other satellites

3. Solar Array (Figs. 12-50 to 12-52)

$$C_E = (41,500 SF \cdot 627 - 34,100 W \cdot 50) F_E \quad (12-52)$$

$F_E = 4.0$ for deep space or lunar satellites

$F_F = 1.0$ for all other satellites

SF = square feet of solar array

$$C_T = 34,100 W \cdot 50 F_T \quad (12-53)$$

$F_T = 4.0$ for deep space or lunar satellites

$F_T = 1.0$ for all other satellites

$$C_P = 42,678 SF \cdot 444 F_P \quad (12-54)$$

$F_P = 2.0$ for oriented paddle satellites

$F_P = 1.0$ for all other satellites

4. Power Converters (Figs. 12-53 to 12-55)

$$C_E = 82,800 W \cdot 620 \quad (12-55)$$

$$C_T = 48,640 W \cdot 620 \quad (12-56)$$

$$C_P = 14,870 W \cdot 738 \quad (12-57)$$

5. Shunt Regulators

NA

Total Structure

1. Structure (Figs. 12-56 to 12-58)

$$C_E = (139,000 W \cdot 393) (F_E) (STF) \quad (12-58)$$

$F_E = 2.5$ for 3-axis and oriented paddle satellites

$F_E = 1.0$ for all other satellites

STF = $1 + \left[\frac{\text{STRF} - (\text{structure weight}/\text{total satellite weight})}{\text{STRF}} \right]$, if STRF \geq structure weight/
total satellite weight; otherwise

STF = $1 - \left\{ \left[\frac{(\text{structure weight}/\text{total satellite weight}) - \text{STRF}}{(\text{structure weight}/\text{total satellite weight})} \right] \right\}$

STRF = $.5054 (\text{total satellite weight})^{-.168}$

$$C_T = (48,900 W \cdot 410) (F_T) (STF) \quad (12-59)$$

$F_T = 3.0$ for 3-axis and oriented paddle satellites

$F_T = 1.0$ for all other satellites

$$C_P = 53,545 W \cdot 263 F_P \quad (12-60)$$

$F_P = 4.0$ for 3-axis and oriented paddle satellites

$F_P = 1.0$ for all other satellites

Thermal Control (Figs. 12-59 to 12-61)

$$C_E = 91,287 W \cdot 50 F_E \quad (12-61)$$

$F_E = 2.0$ for 3-axis satellites

$F_E = 1.0$ for all other satellites

$$C_T = 69,338 W \cdot 50 \quad (12-62)$$

$$C_P = 9,400 W \cdot 566 F_P \quad (12-63)$$

$F_P = 2.0$ for 3-axis and oriented paddle satellites

$F_P = 0.4$ for pure spin satellites

$F_P = 1.0$ for all other satellites

12.4.4 Other CERs

Costs for ground support equipment and launch operations are satellite oriented. The CERs developed for these items are related to Spacecraft total Design Engineering costs (SAT_E) or average unit costs (SAT_5) for five spacecraft.

a. Ground Support Equipment (Fig. 12-62)

$$C_G = 49.72 SAT_E \cdot 689 F_G \quad (12-64)$$

$F_G = 2.121$ for 3-axis and oriented paddle satellites

$F_G = 0.409$ for communication satellites

$F_G = 1.000$ all other satellites

b. Launch Operations (Fig. 12-63)

$$C_L = 310 SAT_5 \cdot 588 \quad (12-65)$$

12.4.5 Cost Factors for Redundancy

The question of redundancy and its relation to cost is handled in two ways depending on whether or not the component affected is a CER or catalog item. If a CER item is involved, no further action is taken by the model because the independent variable in the CER will reflect the effects of redundancy. For catalog items, the following procedure is applied. First, Design Engineering is adjusted to reflect the change in development test provided by engineering; this is accomplished by comparing the quantity of components required (including redundant components) per satellite with the base quantity per satellite and using the results with Equation (12-66) to develop a cost adjustment factor. (The adjusted Design Engineering will also have an effect on Unit Production Engineering cost because the two are directly related.) Next, the Test and Evaluation cost category is adjusted in the same manner as Design Engineering, except that the numerical constants in Equation (12-67) are different. Finally, unit costs are affected; however, the effects are indirect. Unit Engineering will be altered because of its relationship to Design Engineering and different quantities per satellite will result in associated learning curve changes. Accordingly, no additional factors are applied to unit costs.

The formulas used for DDT&E adjustment factors are as follows:

Design Engineering Factor (F_{R1})

$$F_{R1} = 0.8875 + 0.1125 F_Q \quad (12-66)$$

where $F_Q = (\text{base quantity} + \text{redundant quantity}) / \text{base quantity}$

Test and Evaluation Factor (F_{T1})

$$F_{T1} = 0.3 + 0.7 F_Q \quad (12-67)$$

The numerical constants in Equations (12-66) and (12-67) were derived from the results of a DOD study (Ref. 12-2) that examined the changes in cost of DDT&E and production as redundancy was added to or subtracted from a baseline satellite configuration. The equations are applied in the same manner to all catalog items.

12.5 COMPUTER COST MODEL

The purpose of the Cost Model computer program is to provide a method of interacting with the engineering model output to produce cost estimates rapidly. The objective of this section is to outline the major steps in the program that make up the cost model.

For each component selected from a particular design, the computer program directly accepts as inputs certain outputs from the engineering model such as weight, performance and quantity information. To produce cost estimates, the program first considers all components identifiable as CER items and applies internally stored CERs. Next, costs for catalog items are obtained from a stored cost data base. The effects of redundancy are then estimated through the application of stored factors. All component costs are summed by major category (i. e., Design Engineering, Test and Evaluation, Unit Engineering and Unit Production) for subsystems and total spacecraft. The costs of System Engineering and Integration, Quality Control and Program Management are calculated by applying the factors discussed in Paragraph 12.4.2 to spacecraft total cost categories. Finally, quantity data concerning the number of full qualification units and total number of flight vehicles are used to calculate total DDT&E, Investment and Operations cost. Other costs such as Ground Support Equipment (GSE) and Launch Support are also covered by the program. The subsections that follow present a detailed discussion of exactly how the program accomplishes all of the above steps.

12.5.1 Program Inputs

Depending on the source, inputs can be classed in three ways; (1) user, (2) program stored, and (3) engineering model (output). User inputs deal with type of mission, cost of Mission Equipment (the computer model treats only the basic spacecraft), fee or profit percentages and quantities required for calculating total program cost. Default or nominal values are provided in case the user cannot ascertain a particular input value. The inputs, program acronyms and default values are presented in Table 12-3.

Table 12-3. Cost Model User Inputs

<u>Item</u>	<u>Acronym</u>	<u>Default Value</u>
Number of Qualification Satellites	QV	1
Number of Flight Satellites	FV	4
Mission Equipment DDT&E Cost*	ME _R	0
Mission Equipment Average Unit Cost*	ME _U	0
Mission Equipment Weight	ME _W	0
Contractor Fee or Profit Percentage	FEE	.07
Type of Mission		
- Communications	COM	NA
- Earth Observation	EO	NA
- Lunar	LUN	NA
- Planetary	PLN	NA

There are two types of program-stored inputs; the first includes catalog item cost data and the second covers factors that operate on component costs. Outside of the model, cost data are calculated for each catalog component that can be identified by the engineering model; the total information thus identified constitutes the program cost data

* Must include applicable fee or profit

base. The catalog item inputs and appropriate program acronyms for each are listed below:

<u>Catalog Input</u>	<u>Program Acronym</u>
Design Engineering Cost	COMPE _R
Test and Evaluation Cost	COMPT _R
Unit Production Cost (Cumulative Average for 5)	P ₅
Reference Quantity per Satellite	Q _{REF}

The first three items have been defined and discussed in Paragraph 12.4.1. The fourth item refers to the quantity per satellite nominally required by satellites for a particular component; an example would be batteries where typically Q_{REF} equals two. The need for such information arises from a requirement to give effect to redundancy within the Cost Model; its use is included as part of the redundancy factor discussion in Paragraph 12.4.5.

Factors are stored in the program and are used to: (1) adjust cost to a particular base year, and (2) to treat redundancy. Base year costs are currently stated in terms of 1971 dollars; the price index factor is PI and is applied to all catalog and CER component costs. The effect on component development cost of redundancy is calculated by applying factors F_{RI} to Design Engineering and F_{TI} to Test and Evaluation. Values for these factors are derived within the model by comparing the quantity per satellite for each component in a particular design with the reference quantity, Q_{REF} . The equations for these redundancy factors are stored within the model.

The last major set of inputs needed by the Cost Model is internally derived from the engineering model output. The model produces for each component (i) in a particular design its weight (W_i), quantity ($NCHOSE_i$) and, for selected CER items, physical or performance variables such as square footage of solar arrays, total APS subsystem dry and wet weight, and total satellite wet weight.

12.5.2 Program

The program first operates on all CER items and then treats catalog items. For CER items, the cost equation for Design Engineering takes the general form:

$$\text{COMPE} = \text{COMPE}_R F_{RE2} \text{PI} \quad (12-68)$$

where

$$\text{COMPE}_R = K_E X_E^{b_E} \quad (12-69)$$

and

- F_{RE2} = factor for type of subsystem
- PI = price index
- K_E = constant
- X_E = independent variable
- b_E = power term

For Test and Evaluation, the general form is:

$$\text{COMPT} = K_T X_T^{b_T} F_{RT2} \text{PI} \quad (12-70)$$

where

- K_T = constant
- X_T = independent variable
- b_T = power term
- F_{RT2} = factor based on subsystem type

From Unit Engineering, the general form of the equation is:

$$\text{COMP}_{UE} = \text{COMPE} (Q^{.485} - 1) / \text{QPI} \quad (12-71)$$

where $Q = QV + FV$.

The general form for Unit Production is:

$$\text{COMP}_{UP} = K_P X_P^{b_P} Q^{.152} F_{UP2} \text{PI} / .783 \quad (12-72)$$

where

- K_P = constant
- X_P = independent variable
- b_P = power term
- F_{UP2} = factor based on subsystem type

The CER constants, independent variables, power factors and satellite or subsystem type factors are presented in Tables 12-4 through 12-6.

For catalog items the equation for Design Engineering is:

$$COMPE = 1000 COMPE_R F_{RI}^{PI} \quad (12-73)$$

For Test and Evaluation, the equation is:

$$COMPT = 1000 COMPT_R F_{TI}^{PI} \quad (12-74)$$

For Unit Production, the equation is:

$$COMP_{UP} = 1000 P_5 Q_P^{.848} PI/.783 Q \quad (12-75)$$

and the equation for Unit Engineering is:

$$COMP_{UE} = 1000 COMPE (Q_P^{.485} - 1) / Q \quad (12-76)$$

where

- $F_{RI} = 0.8875 + 0.1125 F_Q$
- $F_{TI} = 0.3 + 0.7 F_Q$
- $Q_P = Q NCHOSE_i$
- $NCHOSE_i =$ Number per satellite for component (i)
- and $F_Q = NCHOSE_i / Q_{REF}$, ($F_Q \geq 1.0$).

- a. For each CER component (or assembly) the program, calculates COMPE*,

COMPT, $COMP_{UP}$, $COMP_{UE}$ using Equations (12-68) through (12-72), and

$$COMP_R = COMPE + COMPT \quad (12-77)$$

*For solar arrays only, $COMPE_R = K_E X_E^{b_E} - K_T X_T^{b_T}$

Table 12-4. Design Engineering CER Data

CER Item	Constant (K _E)	Variable. (X _E)	Power (b _E)	Factor (F _{RE2})	Satellite or Subsystem Type
Solar Array	41,500	SF	0.627	4.0	Lunar, Planetary or Paddles
	41,500	SF	0.627	1.0	Other
Wiring Harness	3,920	W	0.715	1.0	Pure Spin
	3,920	W	0.715	5.0	Other
Thermal	91,287	W	0.5	2.0	3-axis controlled
	91,287	W	0.5	1.0	Other
Converters	82,800	W	0.62	1.0	
Power Control Equipment	51,383	W	0.587	3.2	3-axis plus paddle arrays
	51,383	W	0.587	1.0	Other
Propellant Feed	545,640	TWW	0.222	0.268	Other
	129,200	TWW	0.272	0.507	Dual Spin
Structure	139,000	W	0.393	2.5 STF*	3-axis and paddle arrays
	139,000	W	0.393	1.0 STF*	Other

* $STF = 1 + (STRF - RATIO)/STRF$, if $STRF \geq RATIO$

$STF = 1 - (RATIO - STRF)/RATIO$, if $STRF < RATIO$

where $STRF = 0.5054$ (satellite wet weight)^{-0.168}

and $RATIO = \text{structure weight/satellite wet weight}$

Table 12-5. Test and Evaluation CER Data

CER Item	Constant (K_T)	Variable (X_T)	Power (b_T)	Factor (F_{RT2})	Satellite or Subsystem Type
Solar Array	34,100	W	0.5	4.0	Lunar, Planetary or Paddles
	34,100	W	0.5	1.0	Other
Wiring Harness	6,000	W	0.585	1.0	Pure Spin
	6,000	W	0.585	5.0	Other
Thermal	69,338	W	0.5	1.0	All
Converters	48,640	W	0.62	1.0	All
Power Control Equipment	87,500	W	0.301	3.1	3-axis plus oriented paddles
	87,500	W	0.301	1.0	Other
Propellant Feed	24,160	TDW	0.675	0.325	Dual Spin
	24,160	TDW	0.675	0.619	Other
Structure	48,900	W	0.41	3.0 STF*	3-axis plus oriented arrays
	48,900	W	0.41	1.0 STF*	Other

* See Table 12-3

Table 12-6. Unit Production CER Data

CER Item	Constant (K_P)	Variable (X_P)	Power (b_P)	Factor (F_{UP2})	Satellite or Subsystem Type
Solar Array	42,678	SF	0.444	2.0	Oriented Arrays
	42,678	SF	0.444	1.0	Other
Wiring Harness	2,050	W	0.745	1.0	Pure Spin
	2,050	W	0.745	3.0	Other
Thermal	9,400	W	0.566	2.0	3-axis plus oriented arrays
	9,400	W	0.566	0.4	Pure Spin
	9,400	W	0.566	1.0	Other
Converters	14,872	W	0.74	1.0	All
Power Control Equipment	36,600	W	0.182	4.0	3-axis plus oriented arrays
	36,600	W	0.182	1.0	Other
Propellant Feed	14,000	TDW	0.668	0.84	3-axis or oriented arrays
	14,000	TDW	0.668	0.615	Other
Structure	53,545	W	0.263	4.0	3-axis plus oriented arrays
	53,545	W	0.263	1.0	Other

$$\text{COMP}_U = \text{COMP}_{UP} + \text{COMP}_{UE} \quad (12-78)$$

$$\text{COMP}_{5P} = 0.783 \text{ COMP}_{UP} Q^{.152} \quad (12-79)$$

$$\text{COMP}_{5E} = 0.2365 \text{ COMP}_{UE} (Q/Q^{.485} - 1) \quad (12-80)$$

where

COMP_R = sum of Design Engineering and Test and Evaluation

COMP_U = sum of Unit Engineering and Unit Production

COMP_{5P} = cumulative average Unit Production cost for first five satellites

COMP_{5E} = cumulative average Unit Engineering cost for five

- b. For each catalog item (component) the program, calculates COMPE , COMPT , COMP_{UP} , COMP_{UE} , COMP_R , COMP_U , Equations (12-78) through (12-78), and

$$\text{COMP}_{5P} = 200 P_5 Q_5^{.848} \text{PI} / .783 \quad (12-81)$$

$$\text{COMP}_{5E} = 200 \text{COMPE} (Q_5^{.485} - 1) \quad (12-82)$$

where

Q_5 is five times the quantity per satellite of component (i).

- c. For each subsystem, summations are performed as exemplified by the following equations:

$$\left. \begin{aligned} \text{SUB}_E &= \Sigma \text{COMPE} \\ \text{SUB}_T &= \Sigma \text{COMPT} \\ \text{SUB}_R &= \Sigma \text{COMP}_R \\ \text{SUB}_{UE} &= \Sigma \text{COMP}_{UE} \\ \text{SUB}_{UP} &= \Sigma \text{COMP}_{UP} \\ \text{SUB}_U &= \Sigma \text{COMP}_U \\ \text{SUB}_{5E} &= \Sigma \text{COMP}_{5E} \\ \text{SUB}_{5P} &= \Sigma \text{COMP}_{5P} \end{aligned} \right\} (12-83)$$

- d. For the basic spacecraft, the following categories are summed from the subsystem totals:

$$\begin{array}{rcl}
 \text{SYS}_E & = & \Sigma \text{SUB}_E \\
 \text{SYS}_T & = & \Sigma \text{SUB}_T \\
 \text{SYS}_R & = & \Sigma \text{SUB}_R \\
 \text{SYS}_{UE} & = & \Sigma \text{SUB}_{UE} \\
 \text{SYS}_{UP} & = & \Sigma \text{SUB}_{UP} \\
 \text{SYS}_U & = & \Sigma \text{SUB}_U \\
 \text{SYS}_{5E} & = & \Sigma \text{SUB}_{5E} \\
 \text{SYS}_{5P} & = & \Sigma \text{SUB}_{5P}
 \end{array} \quad \left. \vphantom{\begin{array}{rcl} \text{SYS}_E \\ \text{SYS}_T \\ \text{SYS}_R \\ \text{SYS}_{UE} \\ \text{SYS}_{UP} \\ \text{SYS}_U \\ \text{SYS}_{5E} \\ \text{SYS}_{5P} \end{array}} \right\} (12-84)$$

- e. System oriented costs are estimated using the previous summations as bases for calculations. All of the system costs are added to the base after all calculations have been made, i. e., no system cost is in a base used for estimating any other system cost. The equations for Quality Control are:

$$\begin{array}{rcl}
 \text{QC}_R & = & 0.015 \text{SYS}_E + 0.14 \text{SYS}_T \\
 \text{QC}_U & = & 0.015 \text{SYS}_{UE} + 0.14 \text{SYS}_{UP} \\
 \text{QC}_5 & = & 0.015 \text{SYS}_{5E} + 0.14 \text{SYS}_{5P}
 \end{array} \quad \left. \vphantom{\begin{array}{rcl} \text{QC}_R \\ \text{QC}_U \\ \text{QC}_5 \end{array}} \right\} (12-85)$$

where QC is Quality Control and the subscripts R, U and 5 stand for RDT&E, average Unit Cost and average cost for the first five spacecraft, respectively.

For System Engineering and Integration (SEI)

$$\begin{array}{rcl}
 \text{SEI}_R & = & 0.32 \text{SYS}_E + 0.27 \text{SYS}_T \\
 \text{SEI}_U & = & 0.32 \text{SYS}_{UE} + 0.22 \text{SYS}_{UP} \\
 \text{SEI}_5 & = & 0.32 \text{SYS}_{5E} + 0.22 \text{SYS}_{5P}
 \end{array} \quad \left. \vphantom{\begin{array}{rcl} \text{SEI}_R \\ \text{SEI}_U \\ \text{SEI}_5 \end{array}} \right\} (12-86)$$

For Program Management (PM)

$$\begin{array}{rcl}
 \text{PM}_R & = & 0.19 \text{SYS}_E + 0.02 \text{SYS}_T \\
 \text{PM}_U & = & 0.19 \text{SYS}_{UE} + 0.02 \text{SYS}_{UP} \\
 \text{PM}_5 & = & 0.19 \text{SYS}_{5E} + 0.02 \text{SYS}_{5P}
 \end{array} \quad \left. \vphantom{\begin{array}{rcl} \text{PM}_R \\ \text{PM}_U \\ \text{PM}_5 \end{array}} \right\} (12-87)$$

and for Tooling and Test Equipment (TOOL)

$$\text{TOOL}_R = \text{TOOL}_U = \text{TOOL}_5 = 0 \quad (12-88)$$

because such costs are subsummed under Test and Evaluation.

- f. Tool Spacecraft (SAT) cost is calculated by summations as follows:

$$\left. \begin{aligned} \text{SAT}_R &= \text{SYS}_R + \text{TOOL}_R + \text{QC}_R + \text{SEI}_R + \text{PM}_R \\ \text{SAT}_U &= \text{SYS}_U + \text{TOOL}_U + \text{QC}_U + \text{SEI}_U + \text{PM}_U \\ \text{SAT}_5 &= \text{SYS}_5 + \text{TOOL}_5 + \text{QC}_5 + \text{SEI}_5 + \text{PM}_5 \end{aligned} \right\} (12-89)$$

- g. Ground Support Equipment (GSE) cost is based on the equation:

$$\text{GSE} = 49.72 \text{SYS}_E^{.689} F_G \quad (12-90)$$

Unit Launch Support (L_N) cost is based on the equation

$$L_N = 310 \text{SAT}_5^{.588} \quad (12-91)$$

total Launch Support (L_{TOT}) is:

$$L_{TOT} = (FV) L_N \quad (12-92)$$

- h. Total Program Costs are calculated by adding Mission Equipment cost, which is an input by the user (or is set to 0 if default condition holds).

$$\text{PAY}_R = \text{SAT}_R + \text{ME}_R \quad (12-93)$$

$$\text{PAY}_{QUAL} = QV (\text{SAT}_U + \text{ME}_U) \quad (12-94)$$

$$\text{ME}_{INV} = (FV) \text{ME}_U \quad (12-95)$$

$$\text{SAT}_{INV} = (FV) \text{SAT}_U \quad (12-96)$$

$$\text{PAY}_{INV} = \text{SAT}_{INV} + \text{ME}_{INV} \quad (12-97)$$

In addition, fee or profit must be entered into the total as follows for Total DDTE:

$$\text{FEE}_R = \text{FEE} \left[\text{SAT}_R + (QV) \text{SAT}_U + \text{GSE} \right] \quad (12-98)$$

$$\text{DDTE} = \text{PAY}_R + \text{PAY}_{QUAL} + \text{GSE} + \text{FEE}_R \quad (12-99)$$

for Total Investment (NVEST):

$$\text{FEE}_{INV} = \text{FEE} (\text{SAT}_{INV}) \quad (12-100)$$

$$\text{NVEST} = \text{PAY}_{INV} + \text{FEE}_{INV} \quad (12-101)$$

for Launch Support (OPS):

$$\text{OPS} = L_{TOT} (1.0 + \text{FEE}). \quad (12-102)$$

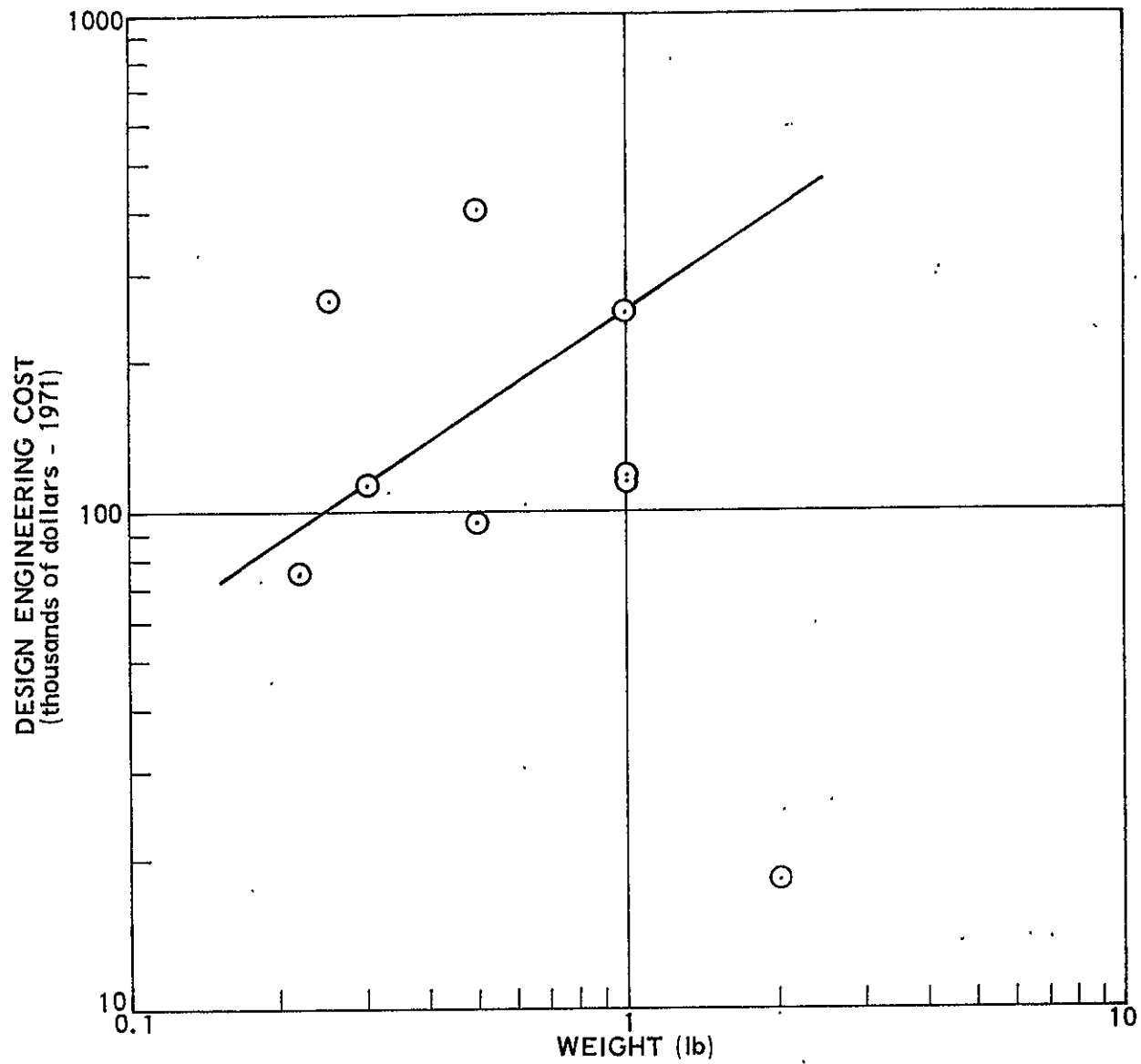


Figure 12-2. Sun Sensor Assemblies: Design Engineering Cost versus Weight

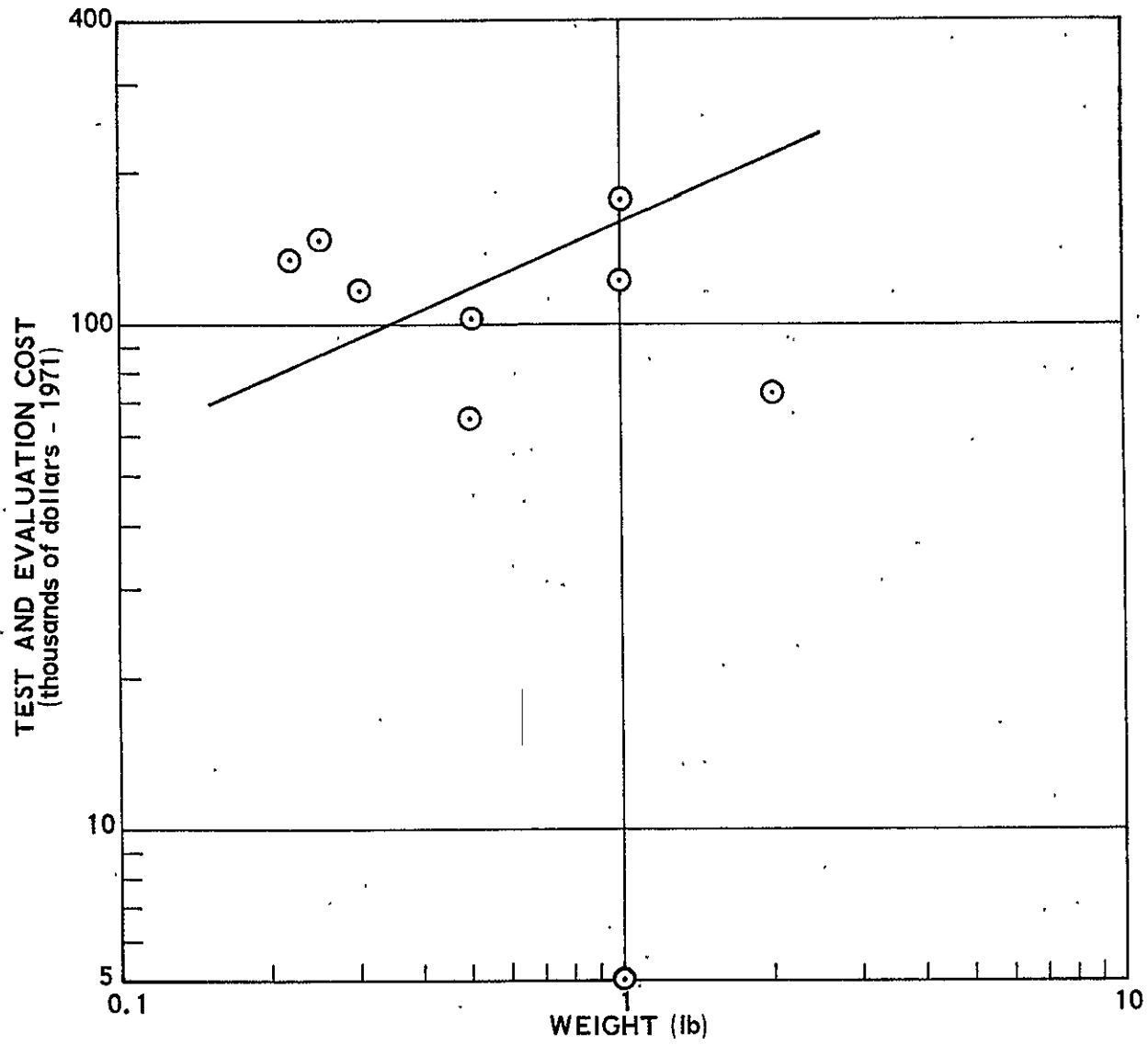


Figure 12-3. Sun Sensor Assemblies: Test and Evaluation Cost versus Weight

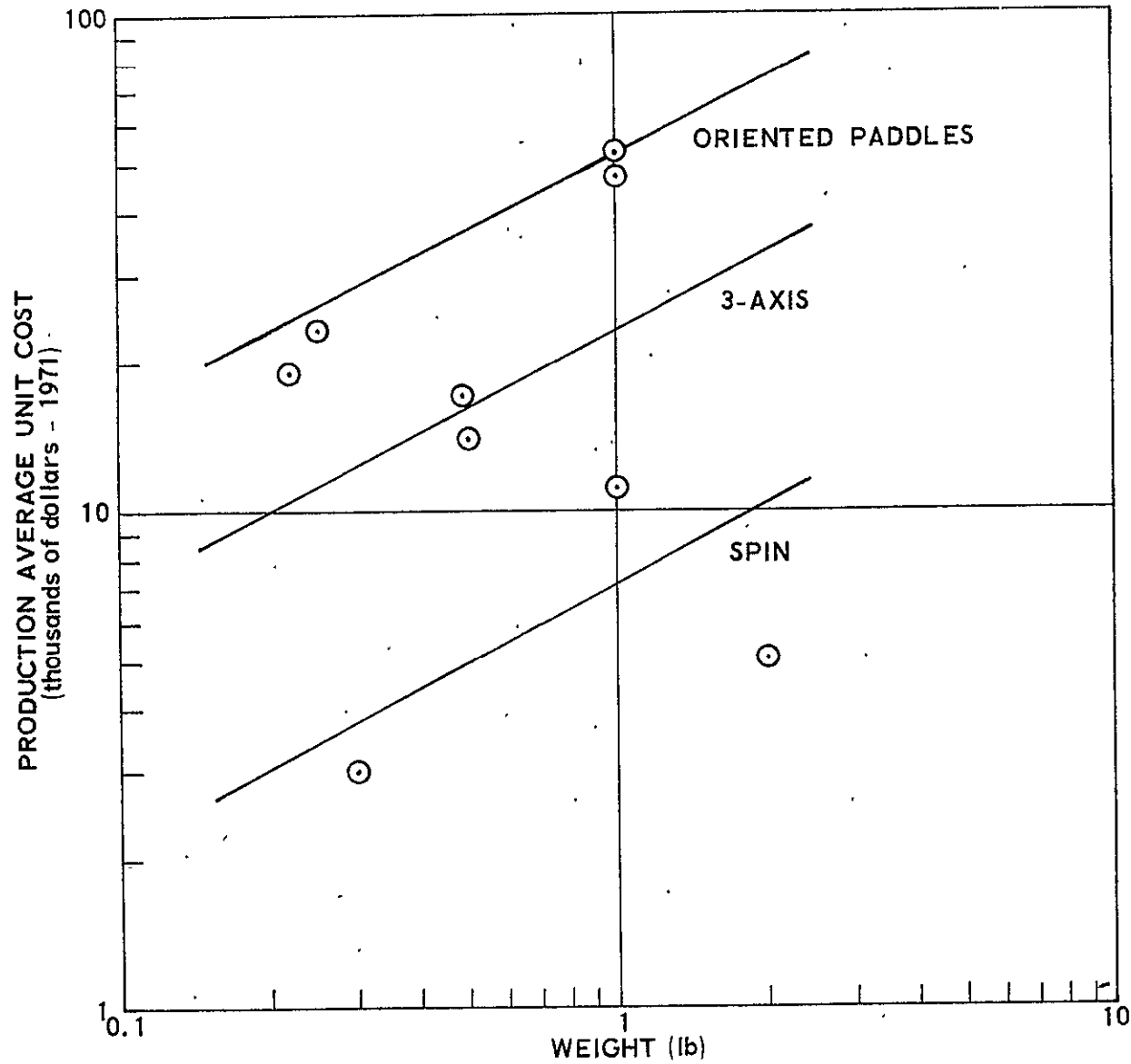


Figure 12-4. Sun Sensor Assemblies: Production Average Unit Cost versus Weight

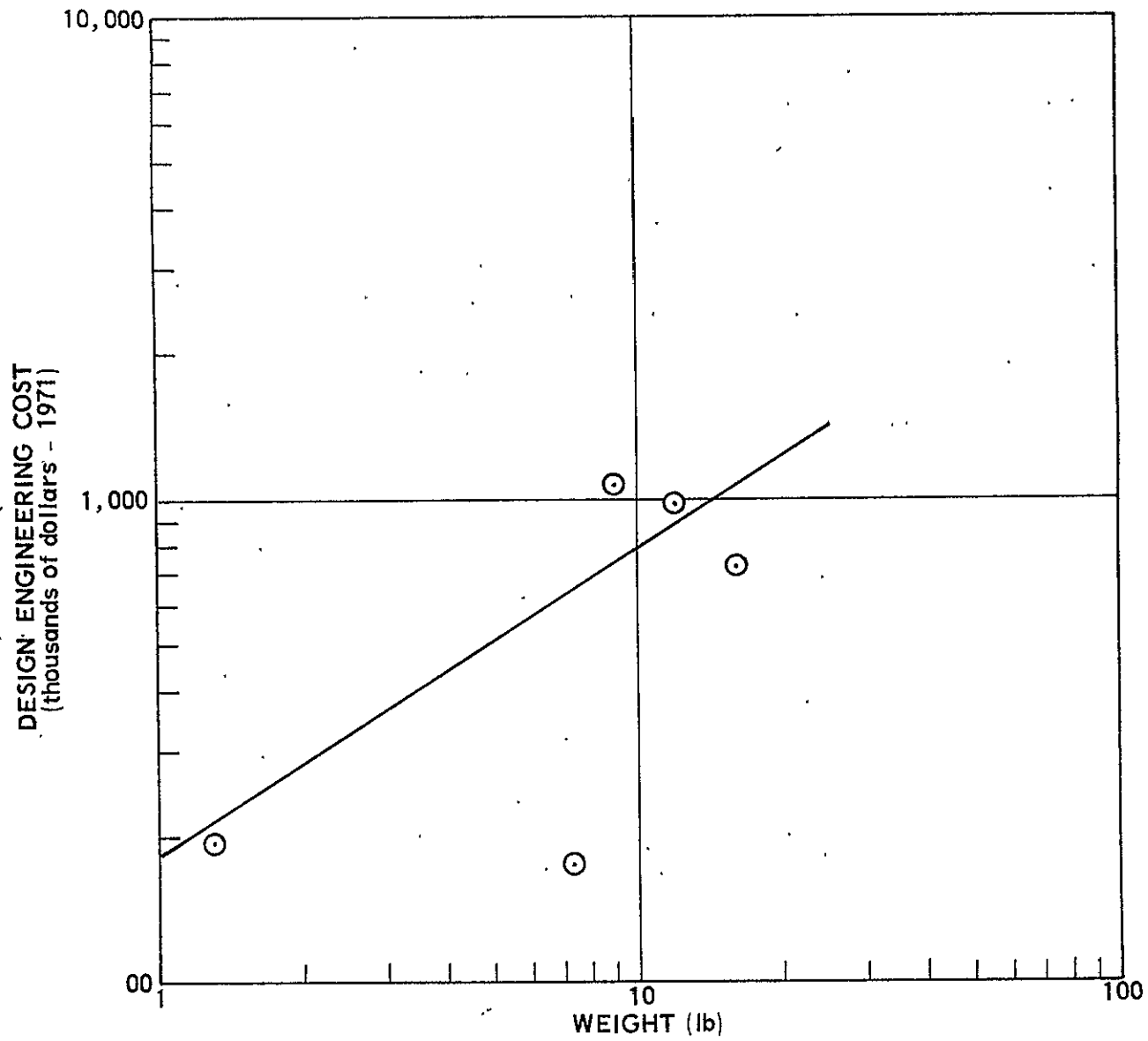


Figure 12-5. Control Electronics Assemblies: Design Engineering Cost versus Weight

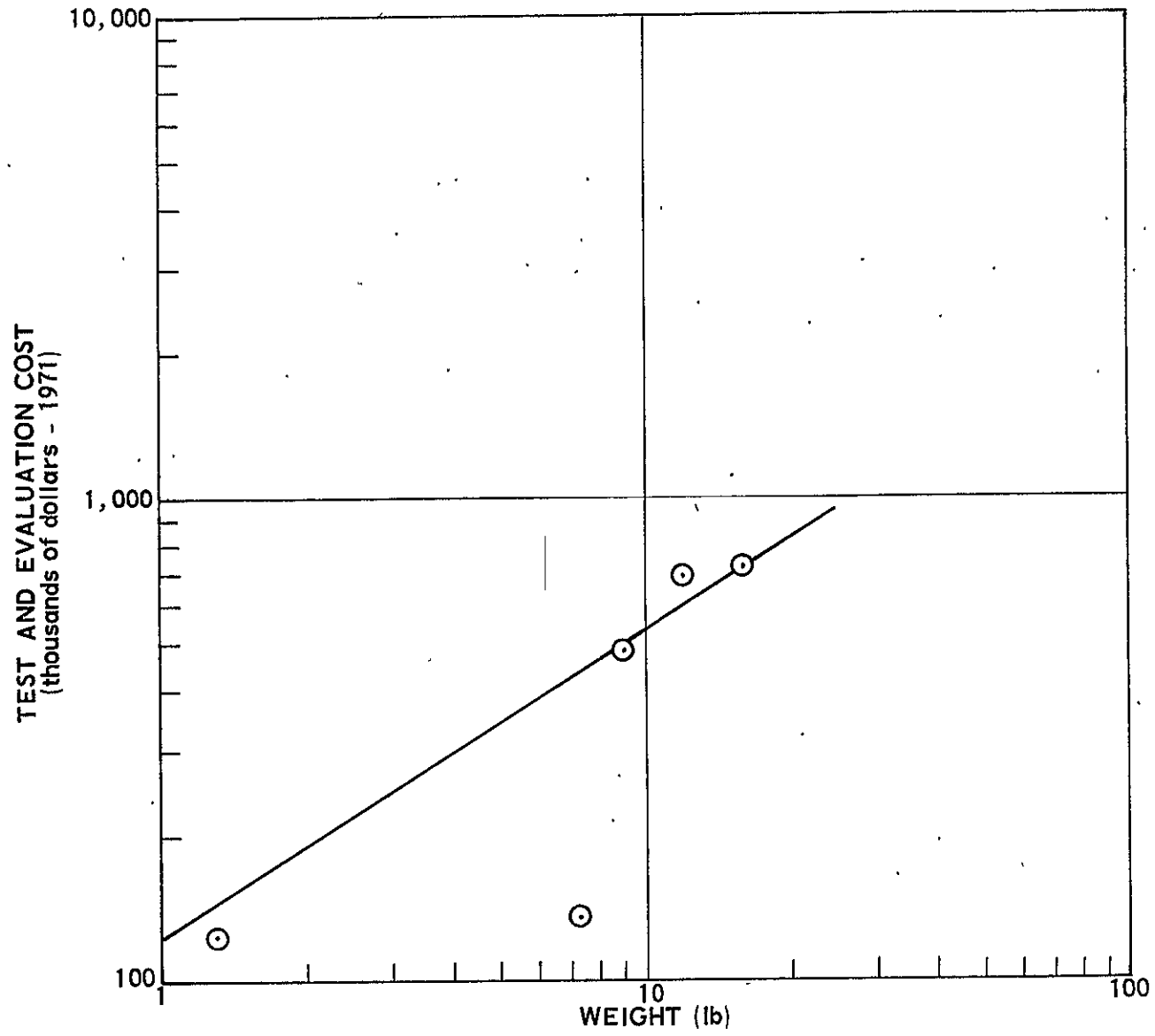


Figure 12-6. Control Electronics Assemblies: Test and Evaluation Cost versus Weight

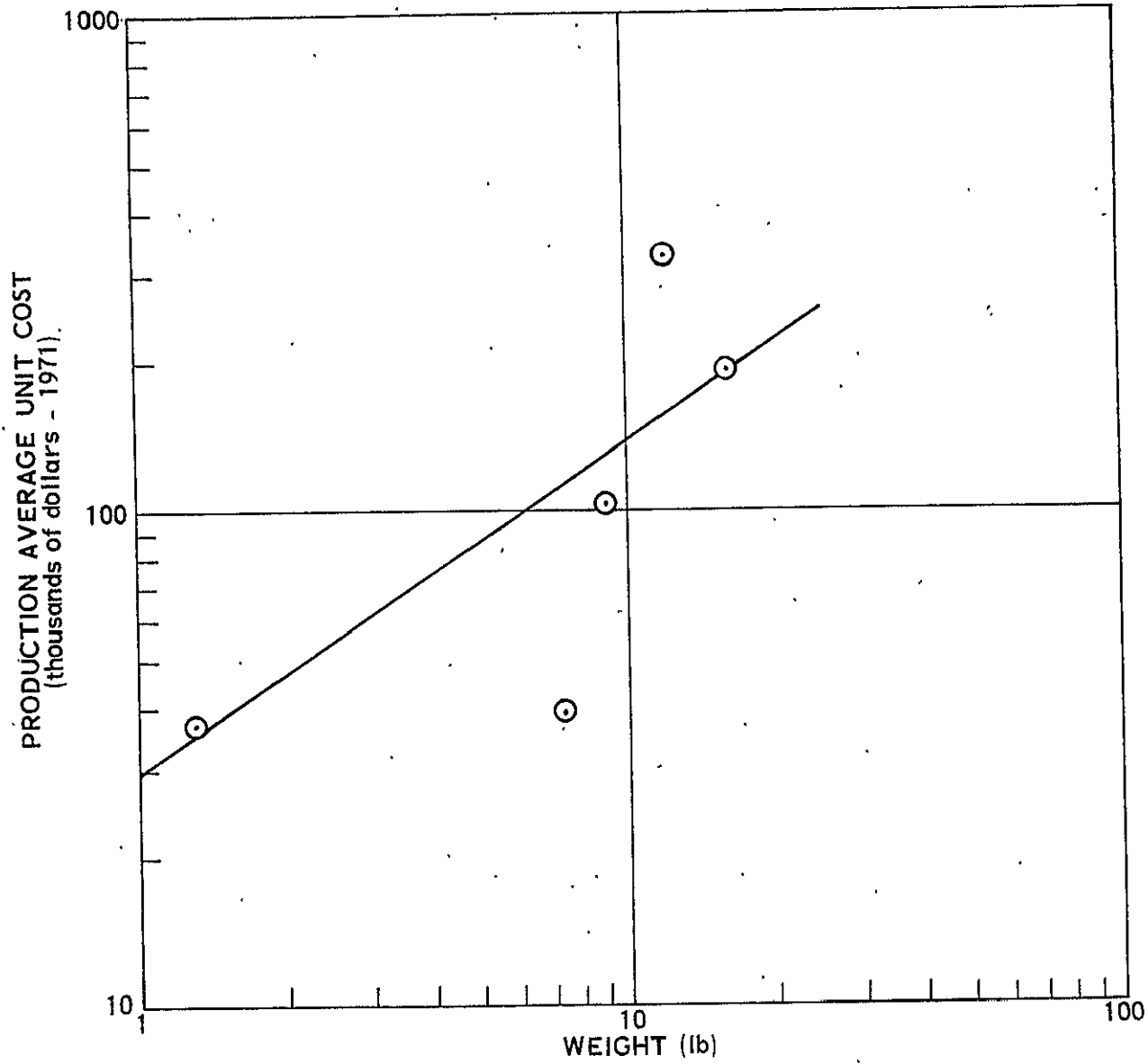


Figure 12-7. Control Electronics Assemblies; Production Average Unit Cost versus Weight

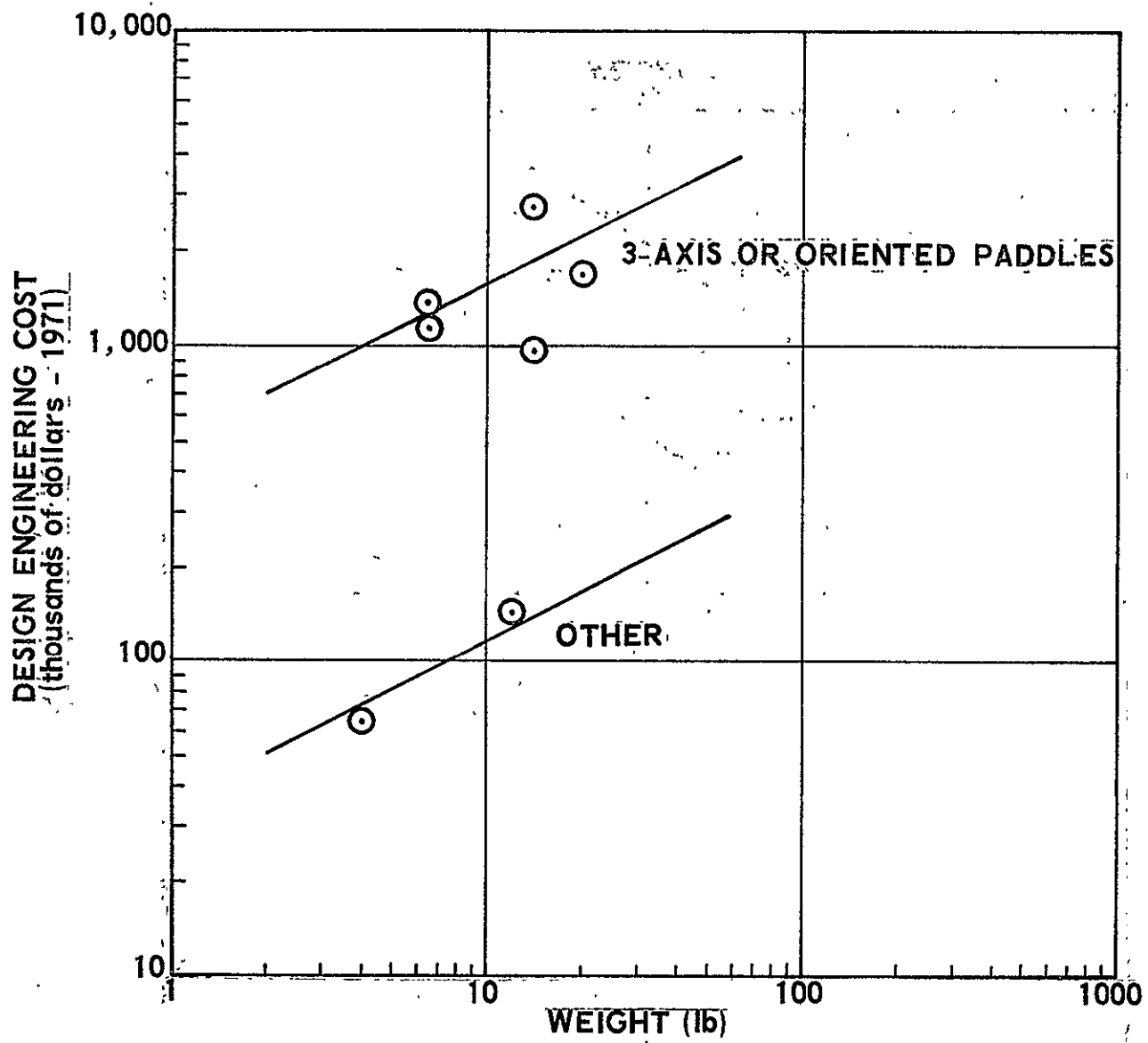


Figure 12-8. Earth Sensors: Design Engineering Cost versus Weight

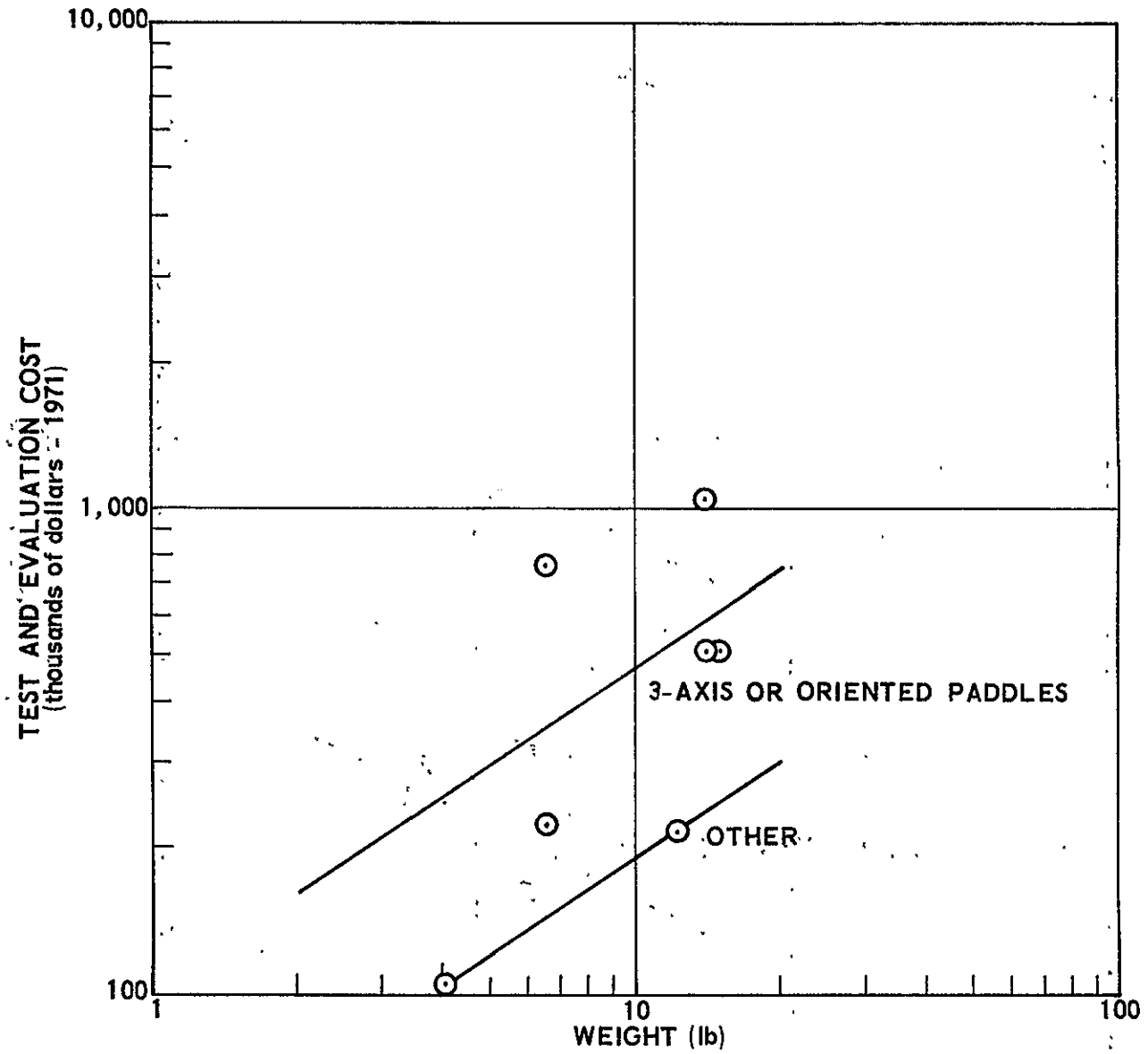


Figure 12-9. Earth Sensors: Test and Evaluation Cost versus Weight

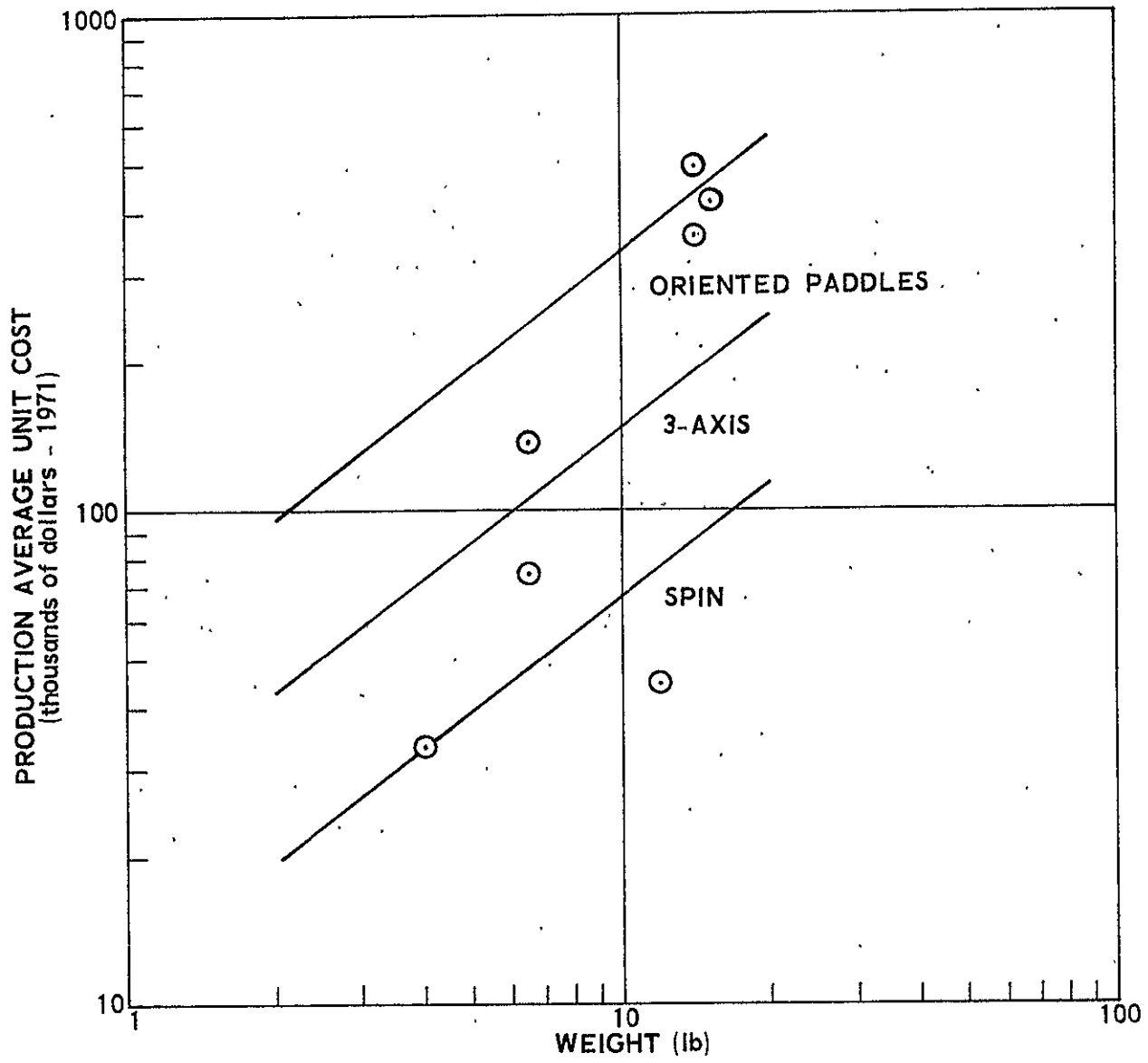


Figure 12-10. Earth Sensors: Production Average Unit Cost versus Weight

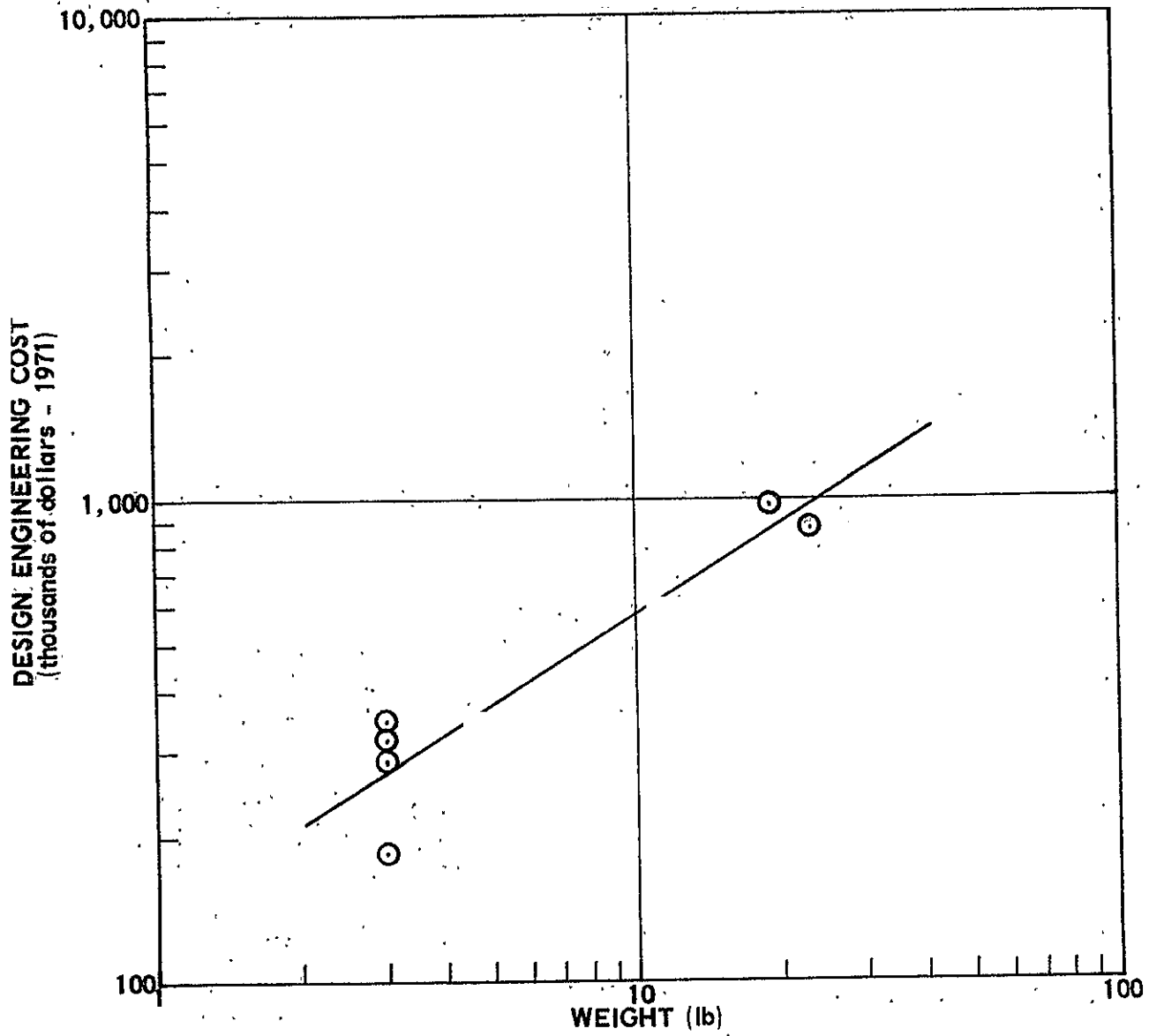


Figure 12-11. Rate Gyro Assemblies: Design Engineering Cost versus Weight

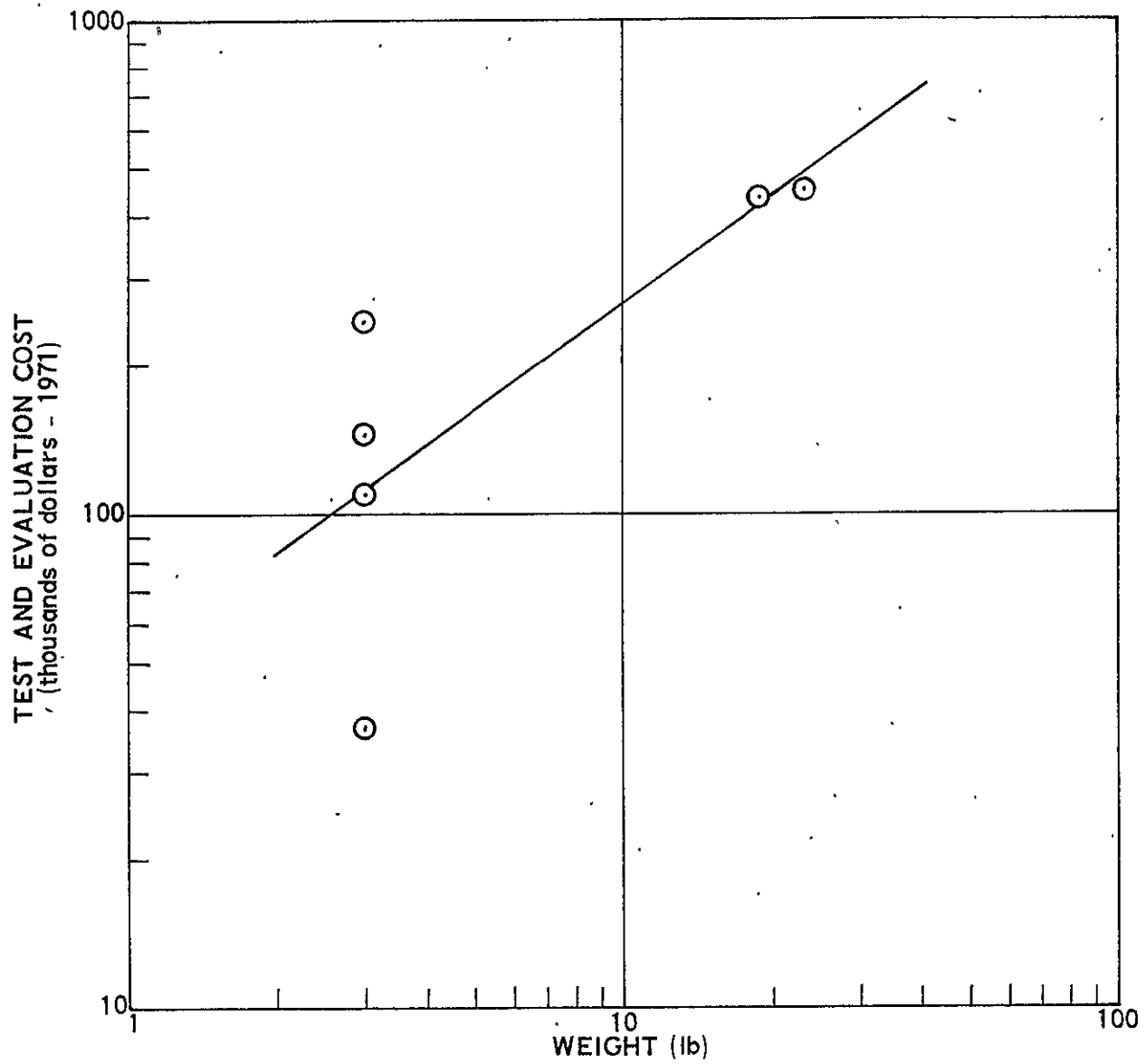


Figure 12-12. Rate Gyro Assemblies: Test and Evaluation Cost versus Weight

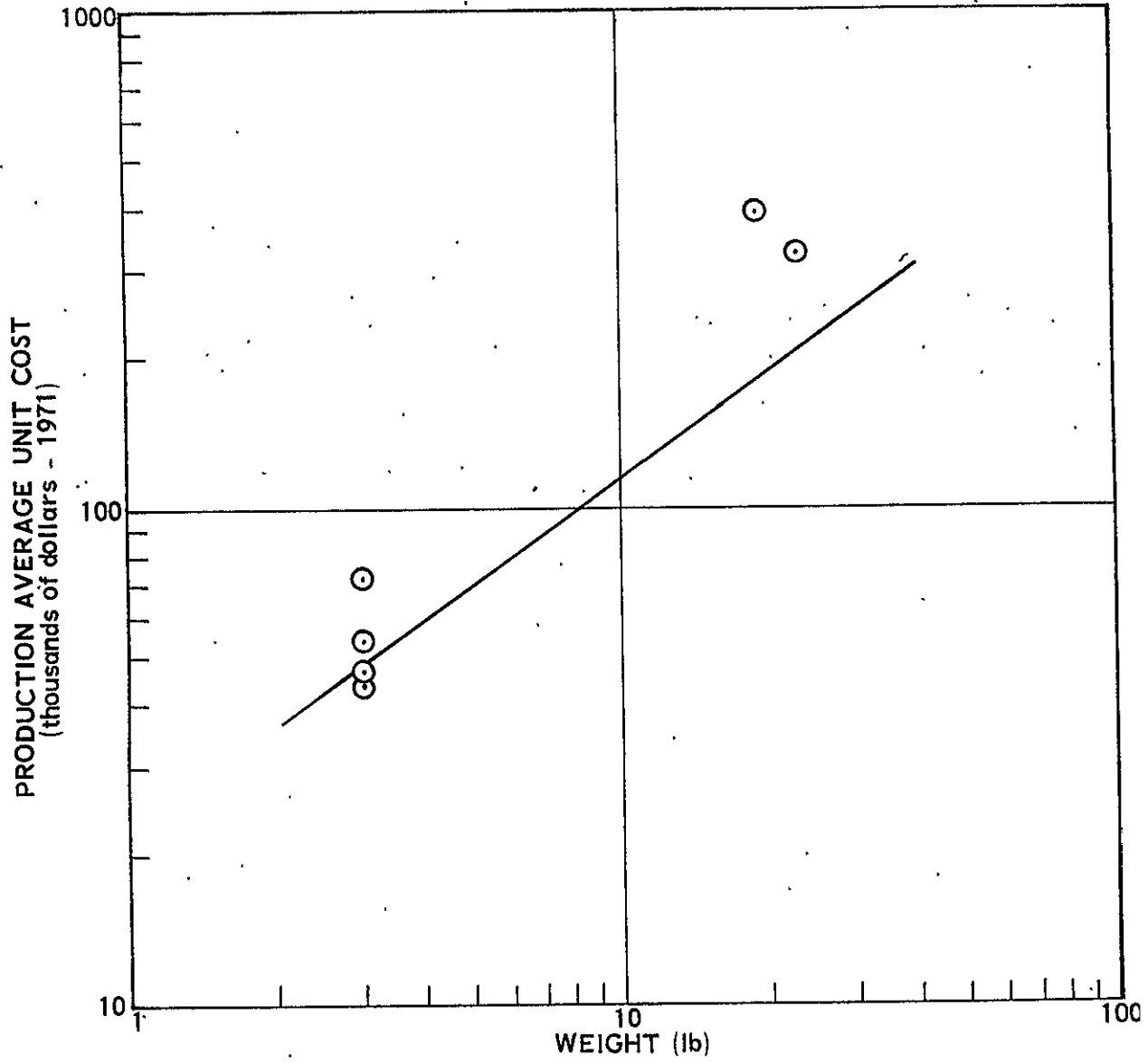


Figure 12-13. Rate Gyro Assemblies: Production Average Unit Cost versus Weight

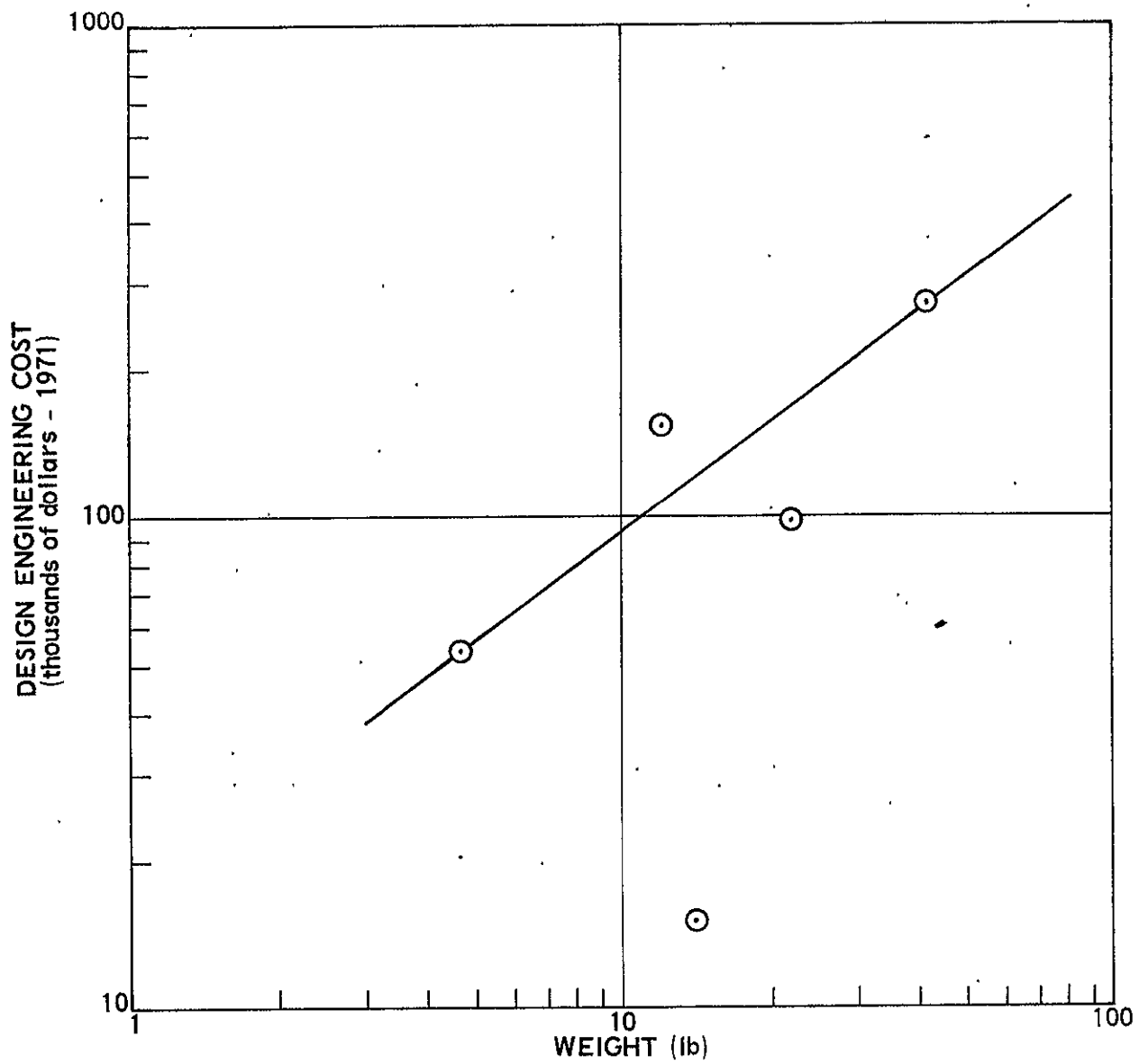


Figure 12-14. Reaction Wheel Assembly: Design Engineering Cost versus Weight

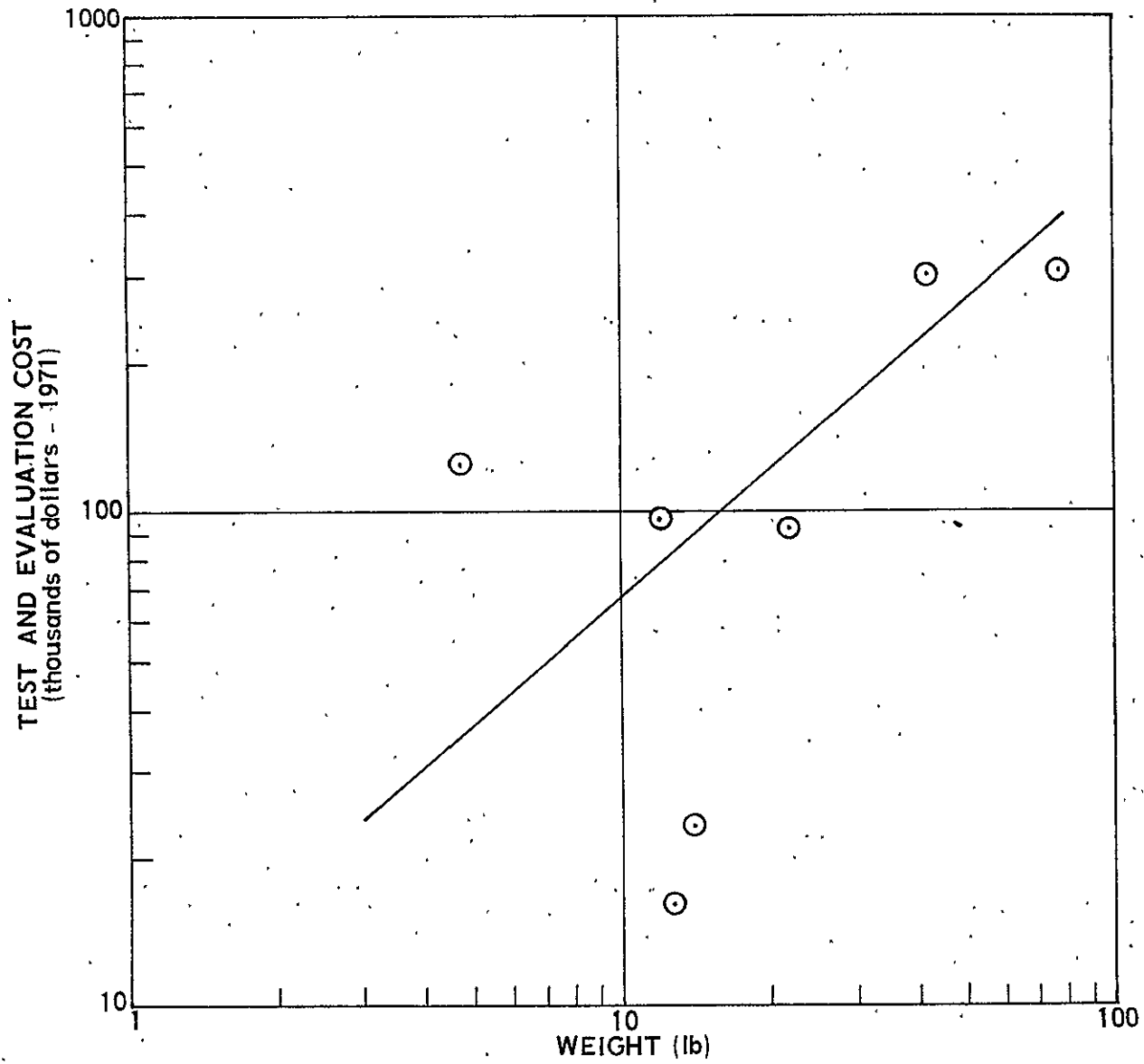


Figure 12-15. Reaction Wheel Assembly: Test and Evaluation Cost versus Weight

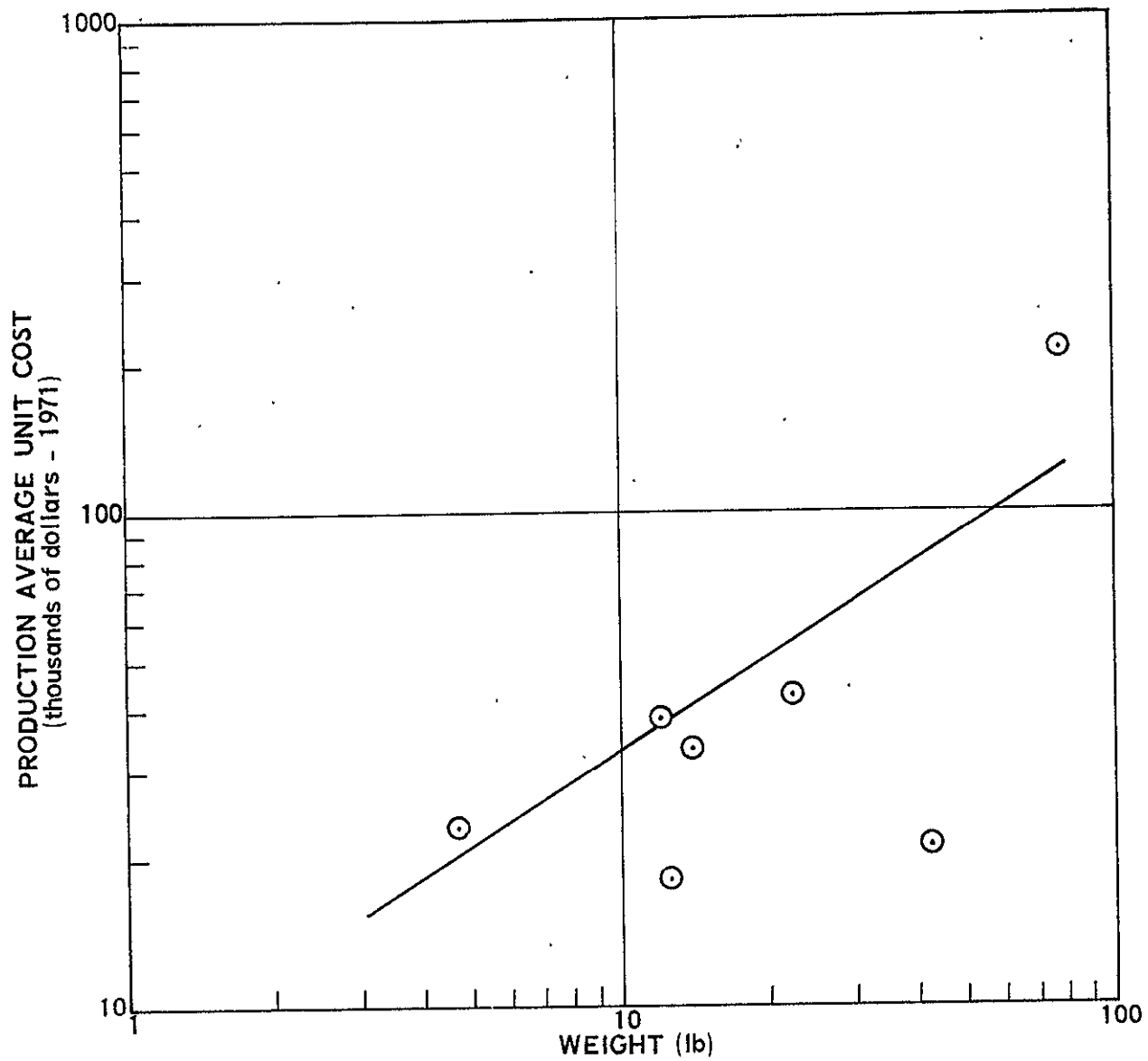


Figure 12-16. Reaction Wheel Assembly: Production Average Unit Cost versus Weight

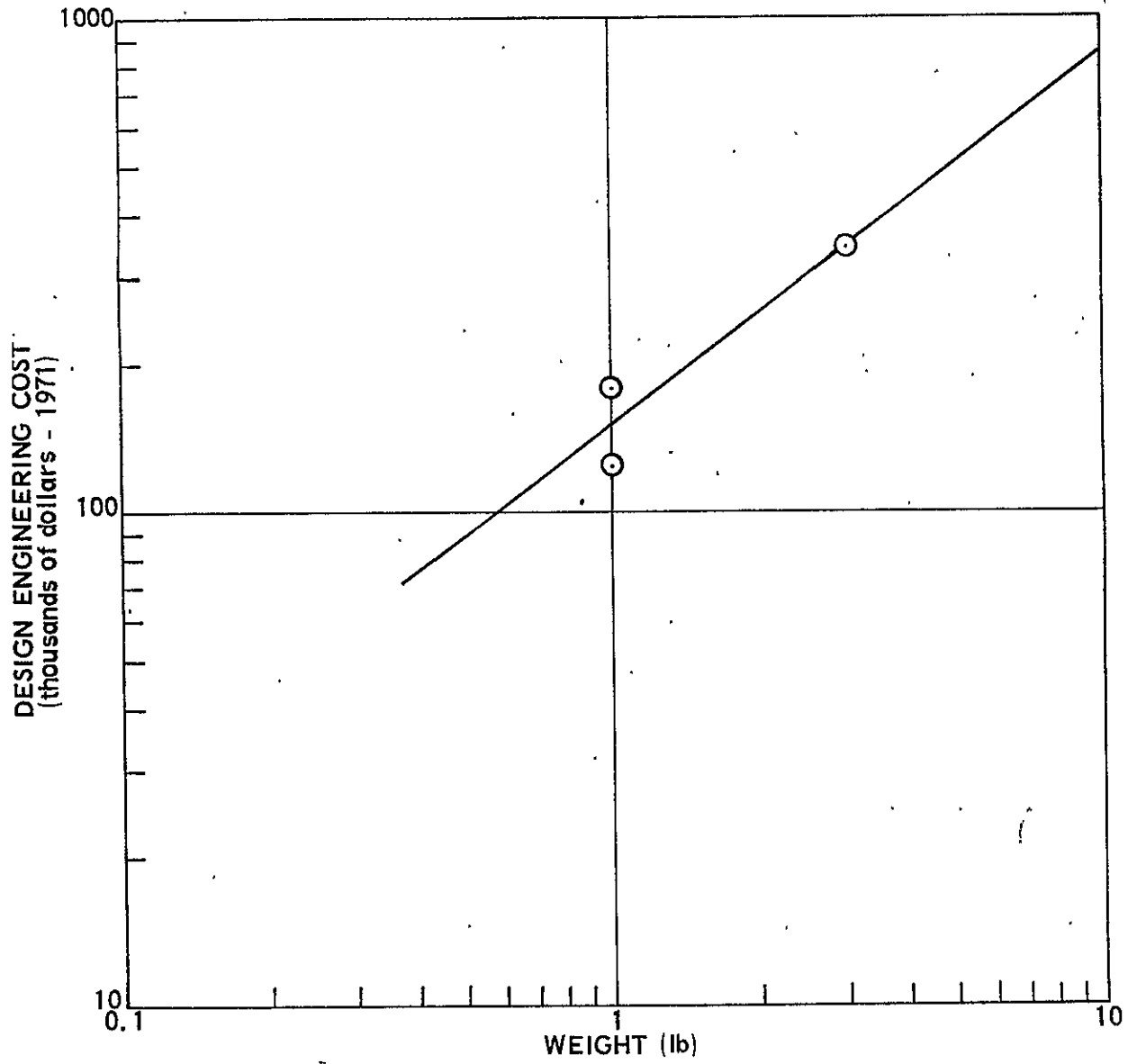


Figure 12-17. Thrusters: Design Engineering Cost versus Weight

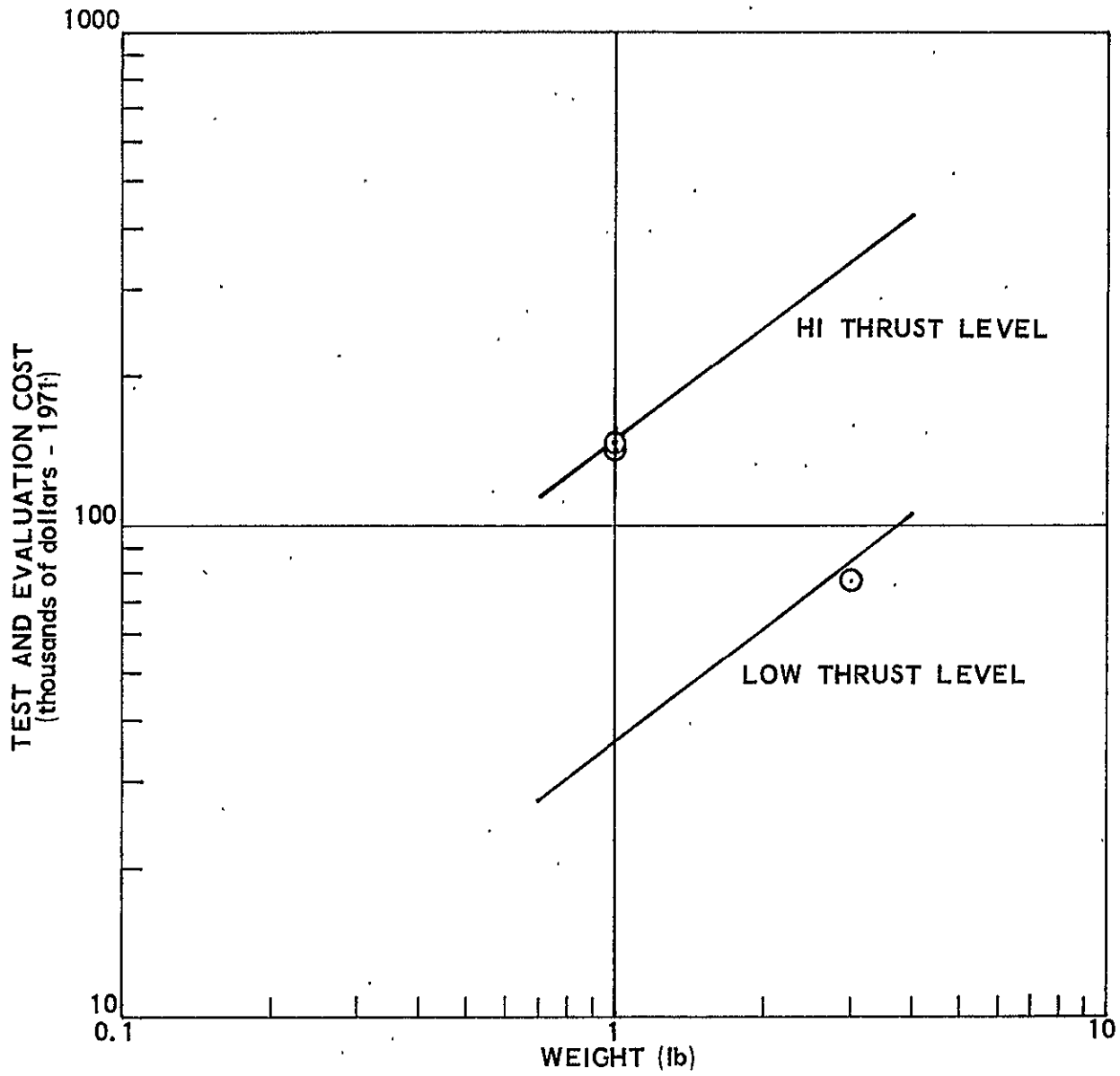


Figure 12-18. Thrusters: Test and Evaluation Cost versus Weight

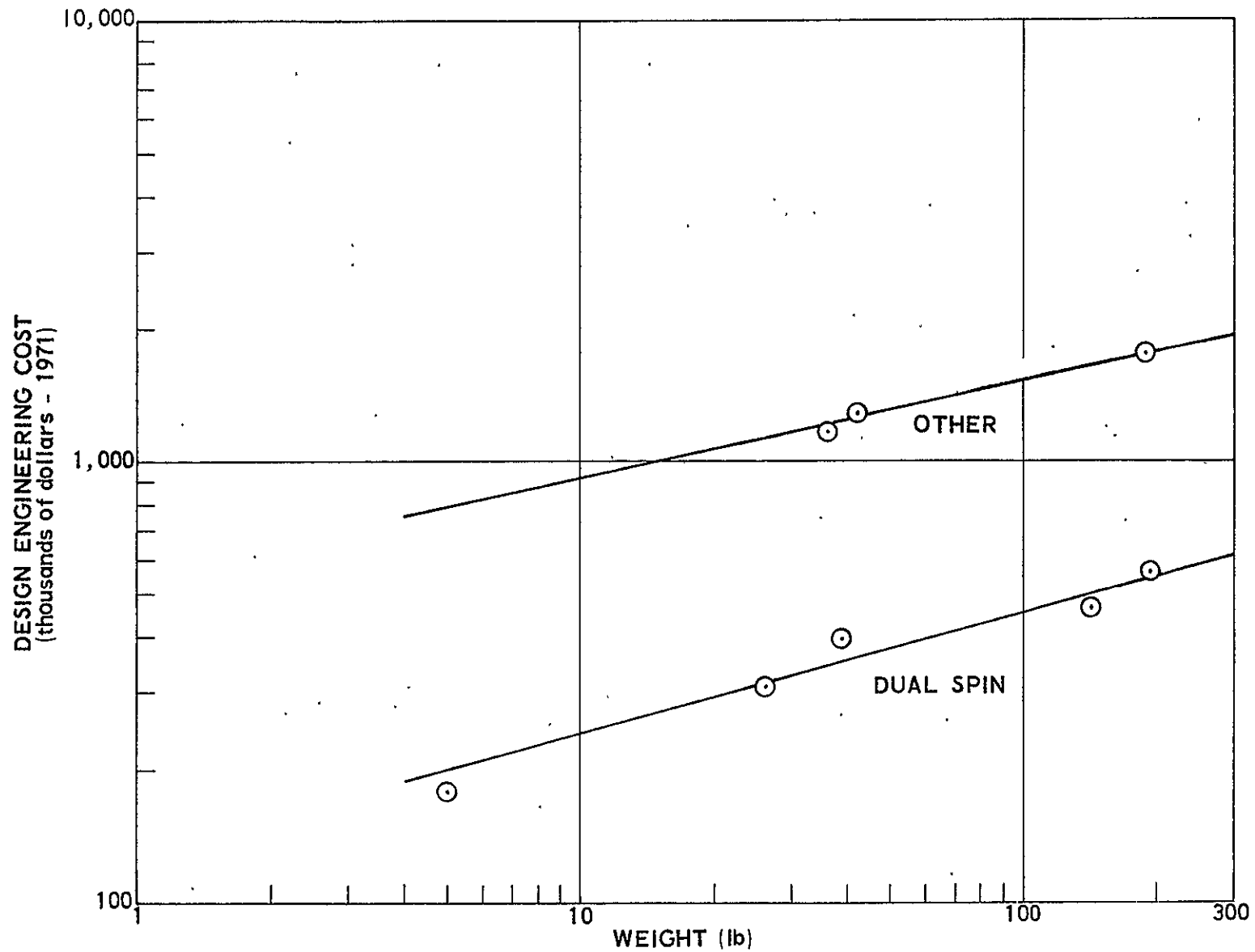


Figure 12-20. Total Reaction Control System: Design Engineering Cost versus Weight (Including Propellant)

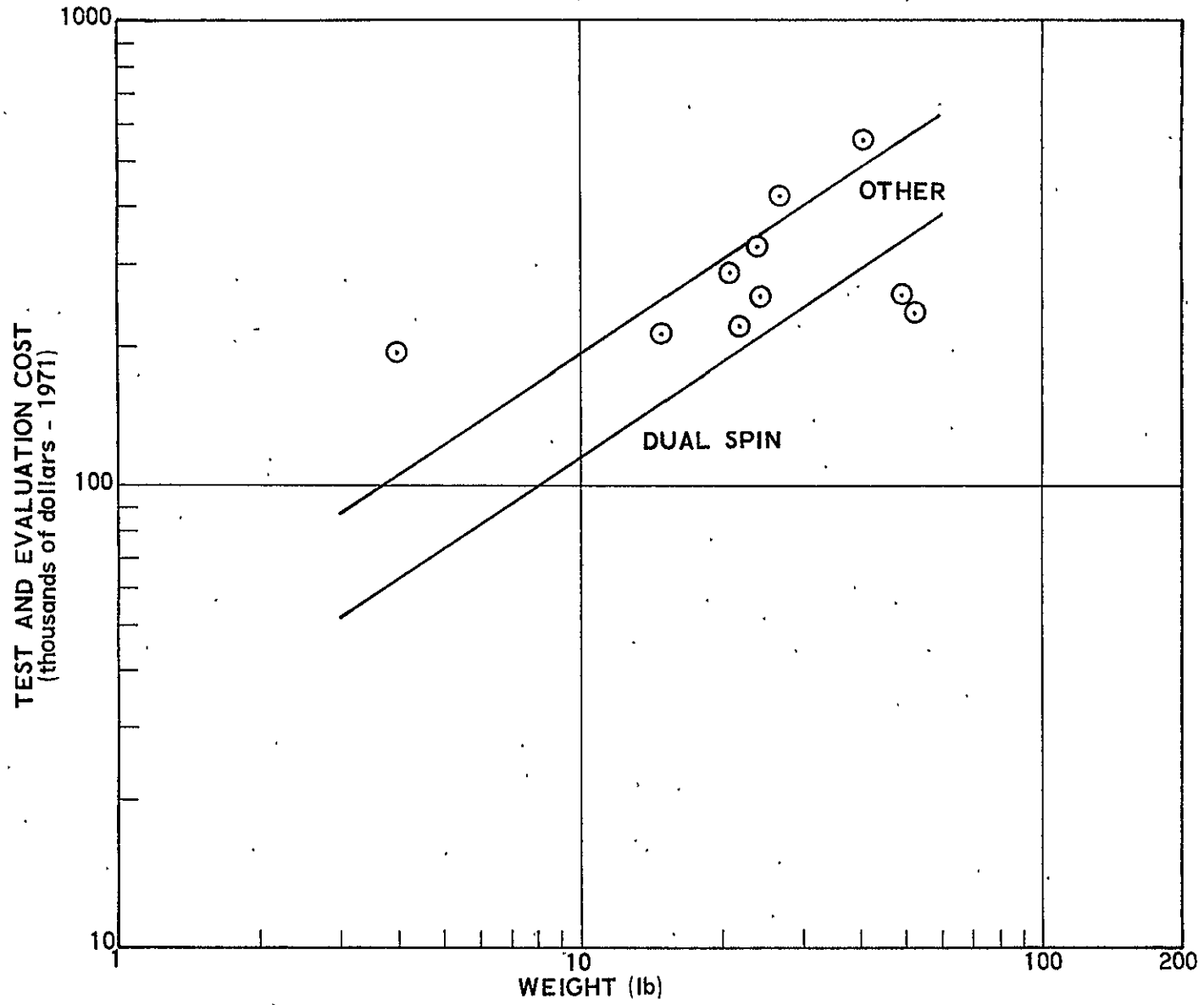


Figure 12-21. Total Reaction Control System: Test and Evaluation Cost versus Weight

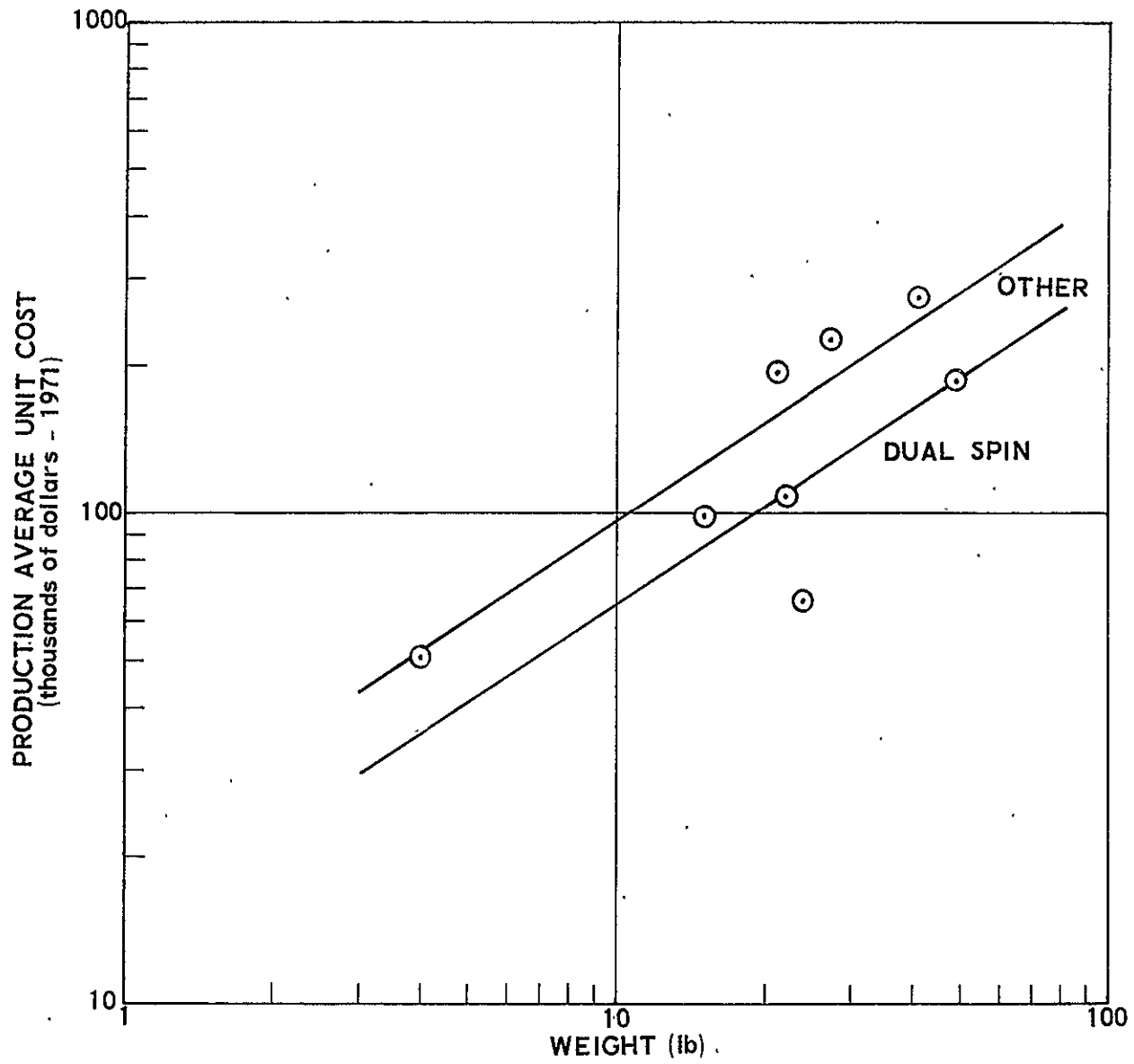


Figure 12-22. Total Reaction Control System: Production Average Unit Cost versus Weight

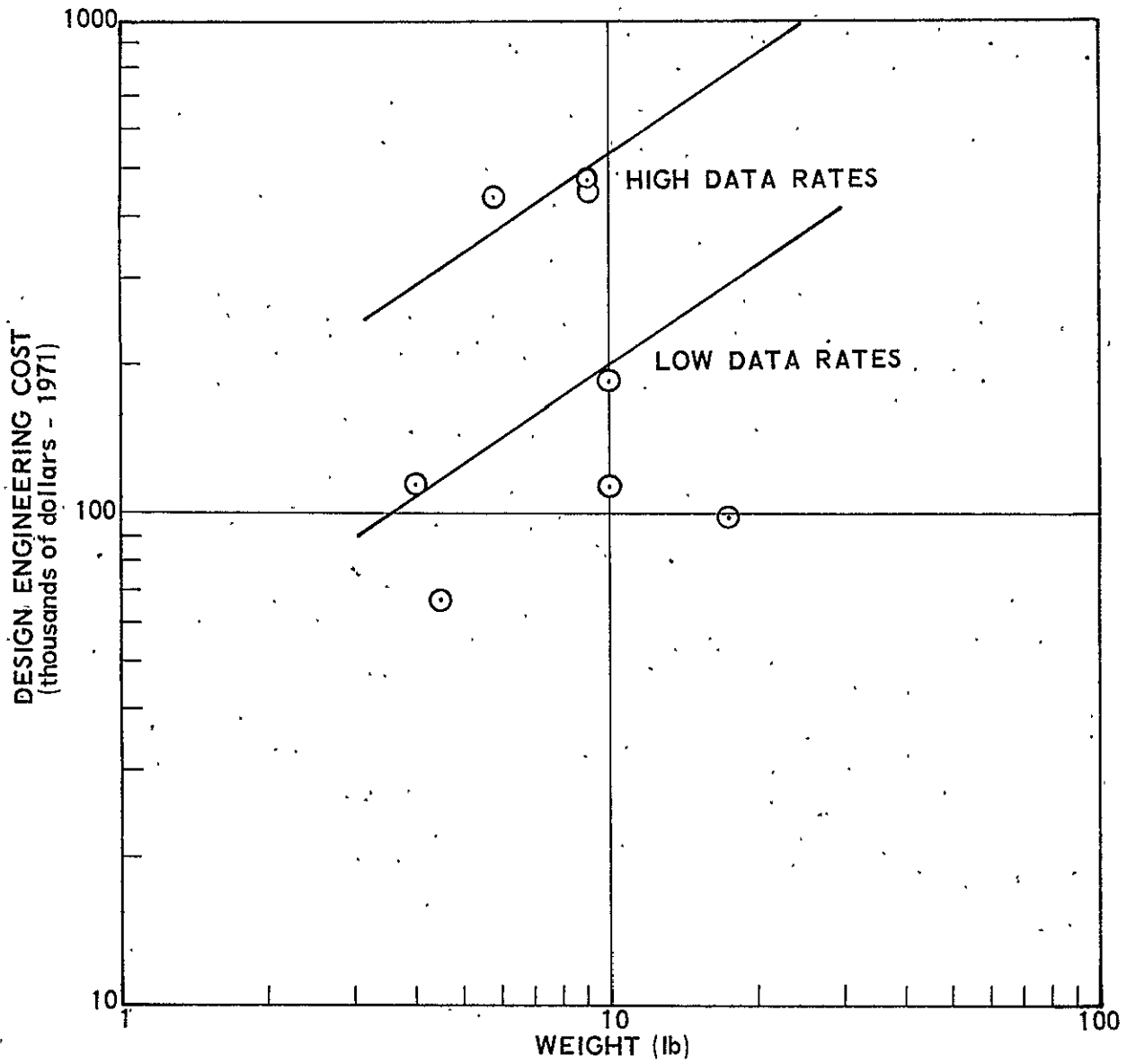


Figure 12-23. Digital Telemetry Units: Design Engineering Cost versus Weight

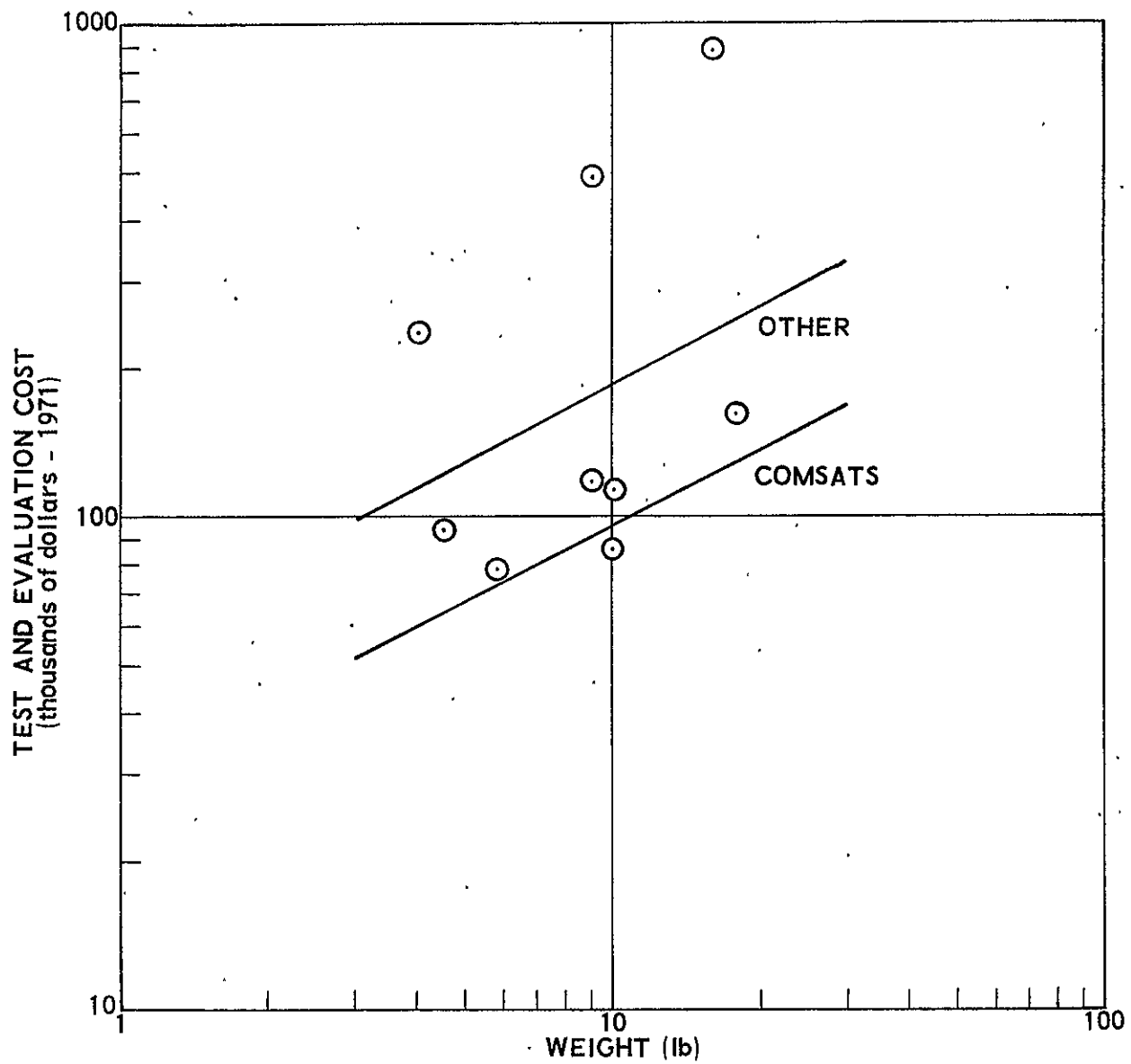


Figure 12-24. Digital Telemetry Units: Test and Evaluation Cost versus Weight

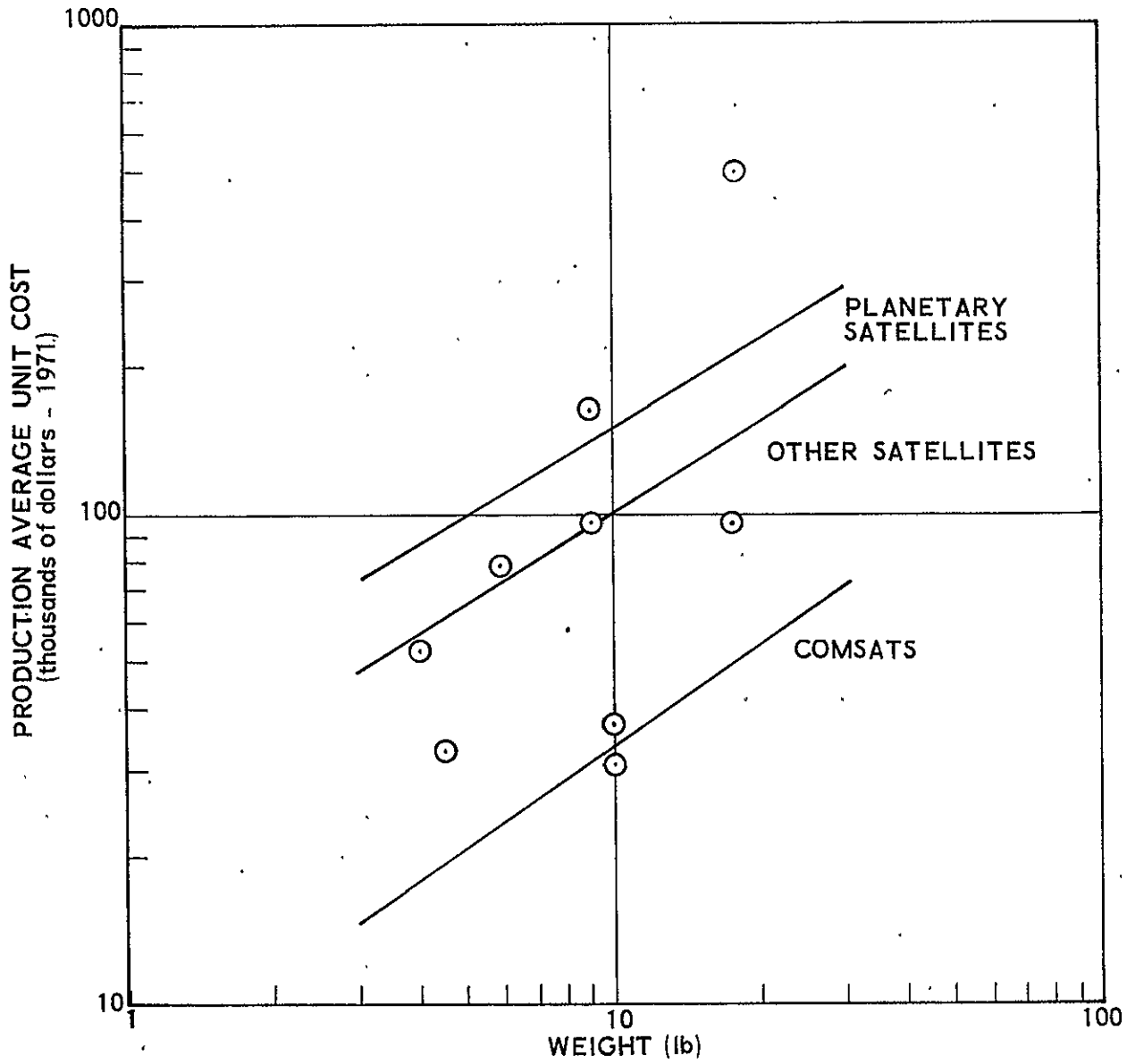


Figure 12-25. Digital Telemetry Units: Production Average Unit Cost versus Weight

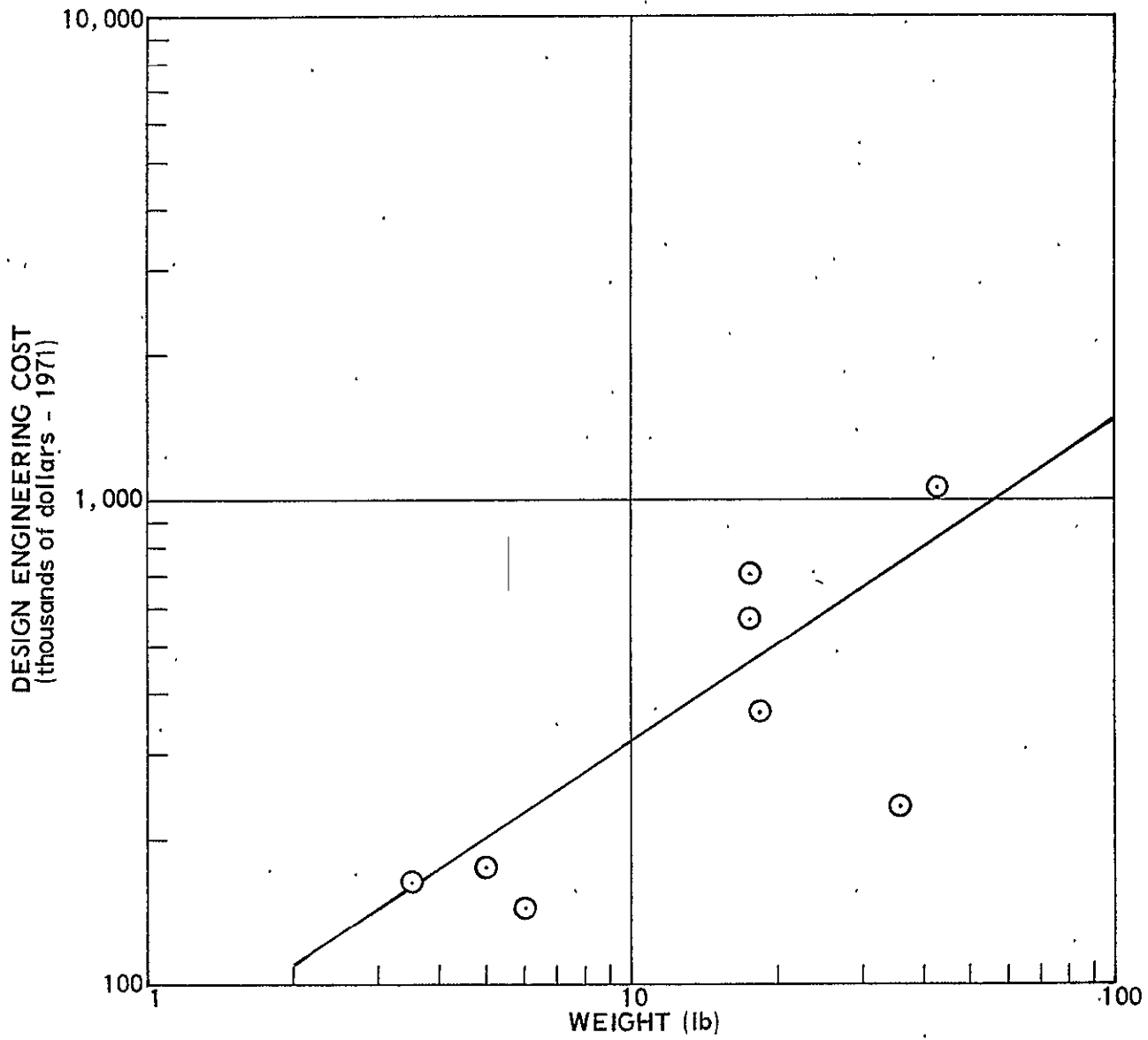


Figure 12-26. Tape Recorders: Design Engineering Cost versus Weight

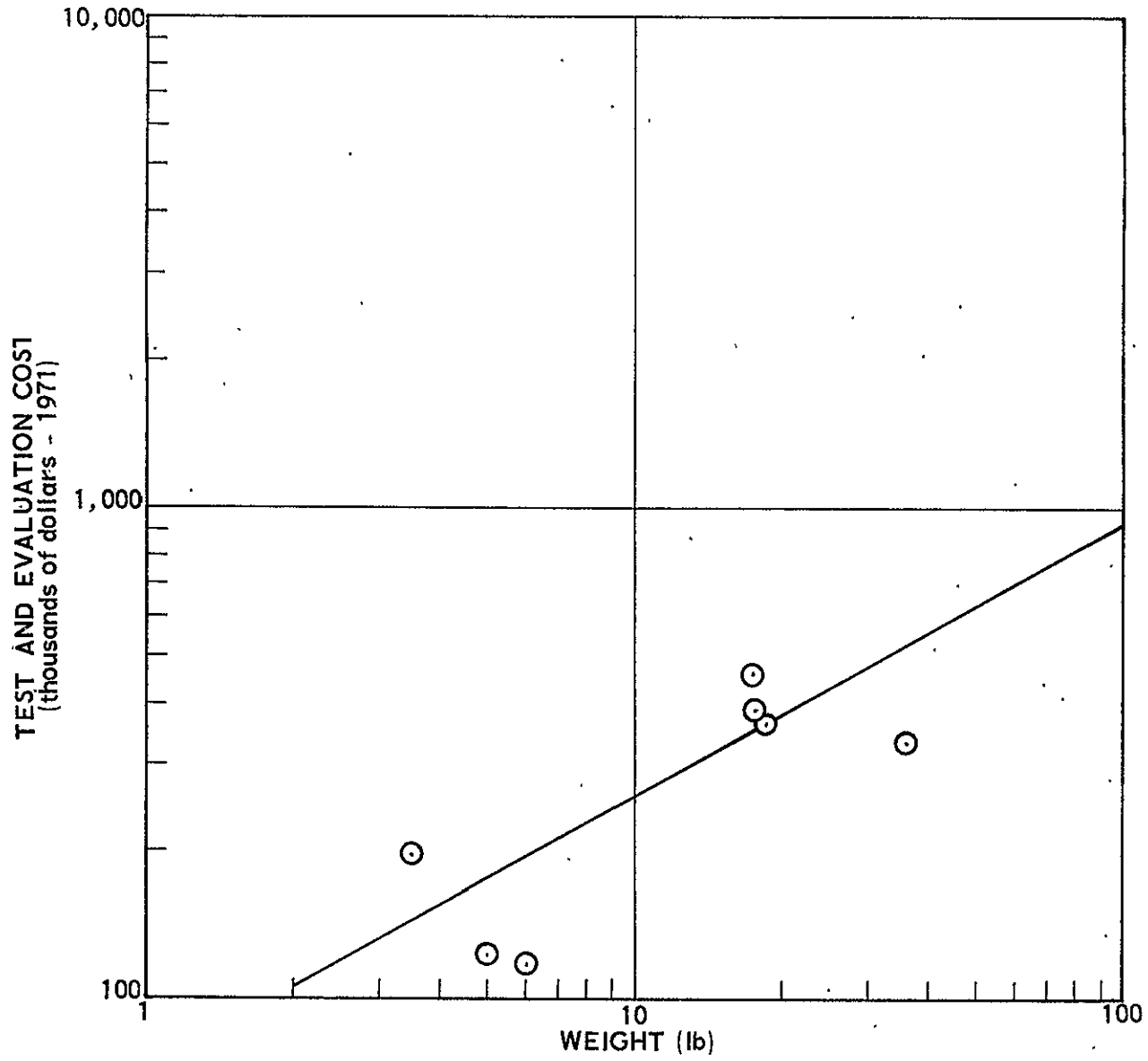


Figure 12-27. Tape Recorders: Test and Evaluation Cost versus Weight

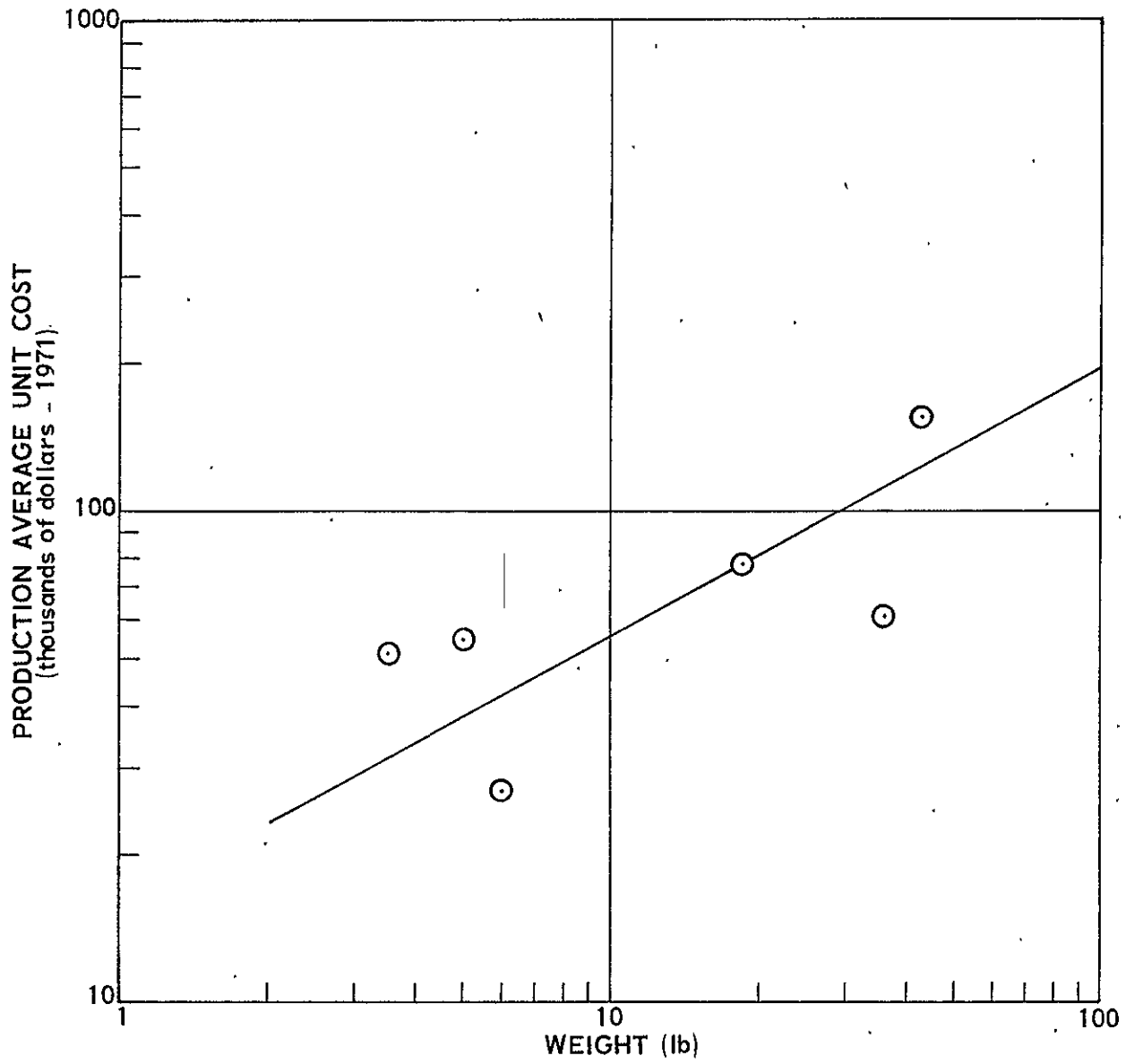


Figure 12-28. Tape Recorders: Production Average Unit Cost versus Weight

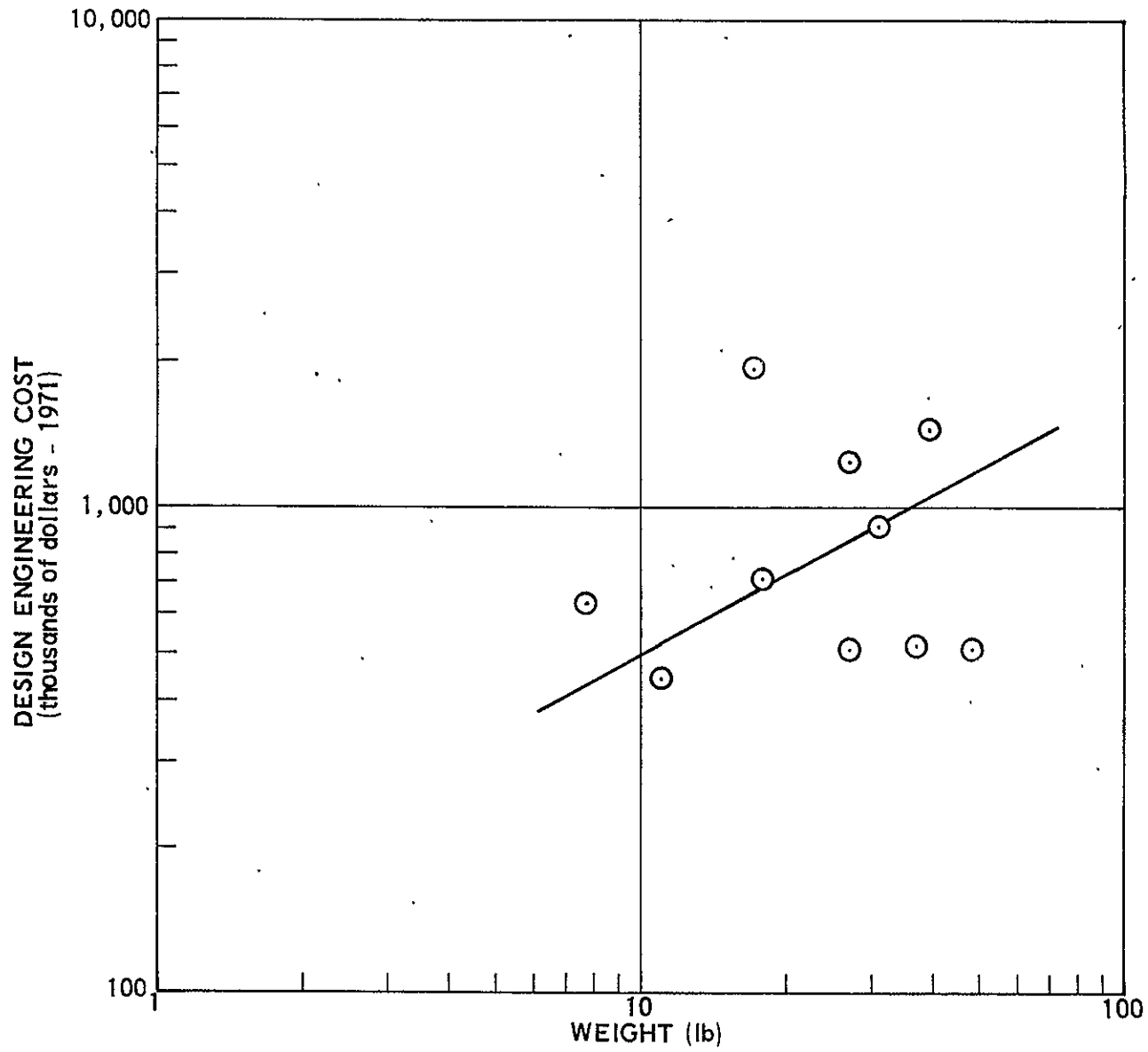


Figure 12-29. Command Decoding and Distribution Equipment: Design Engineering Cost versus Weight

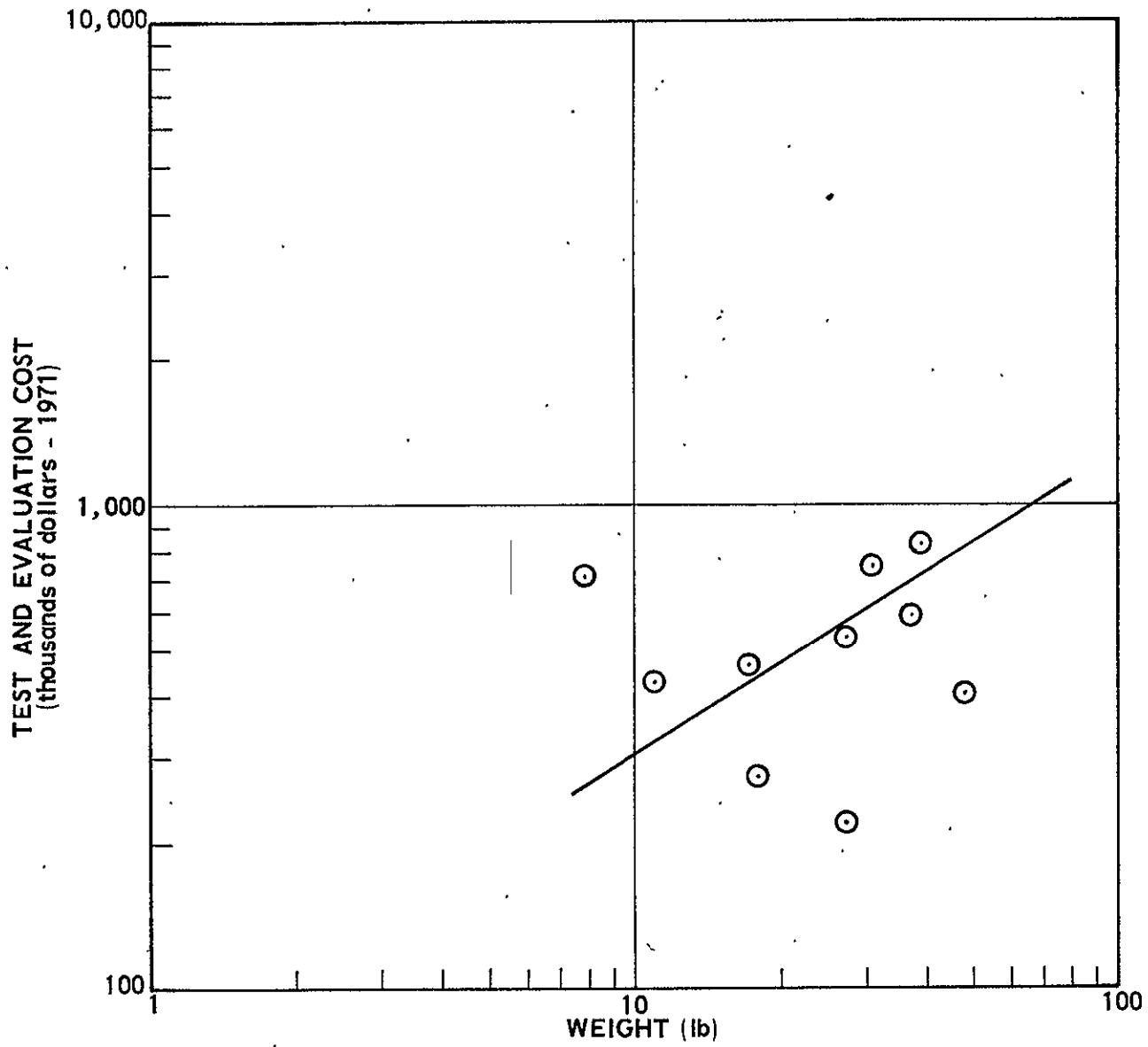


Figure 12-30. Command Decoding and Distribution Equipment: Test and Evaluation Cost versus Weight

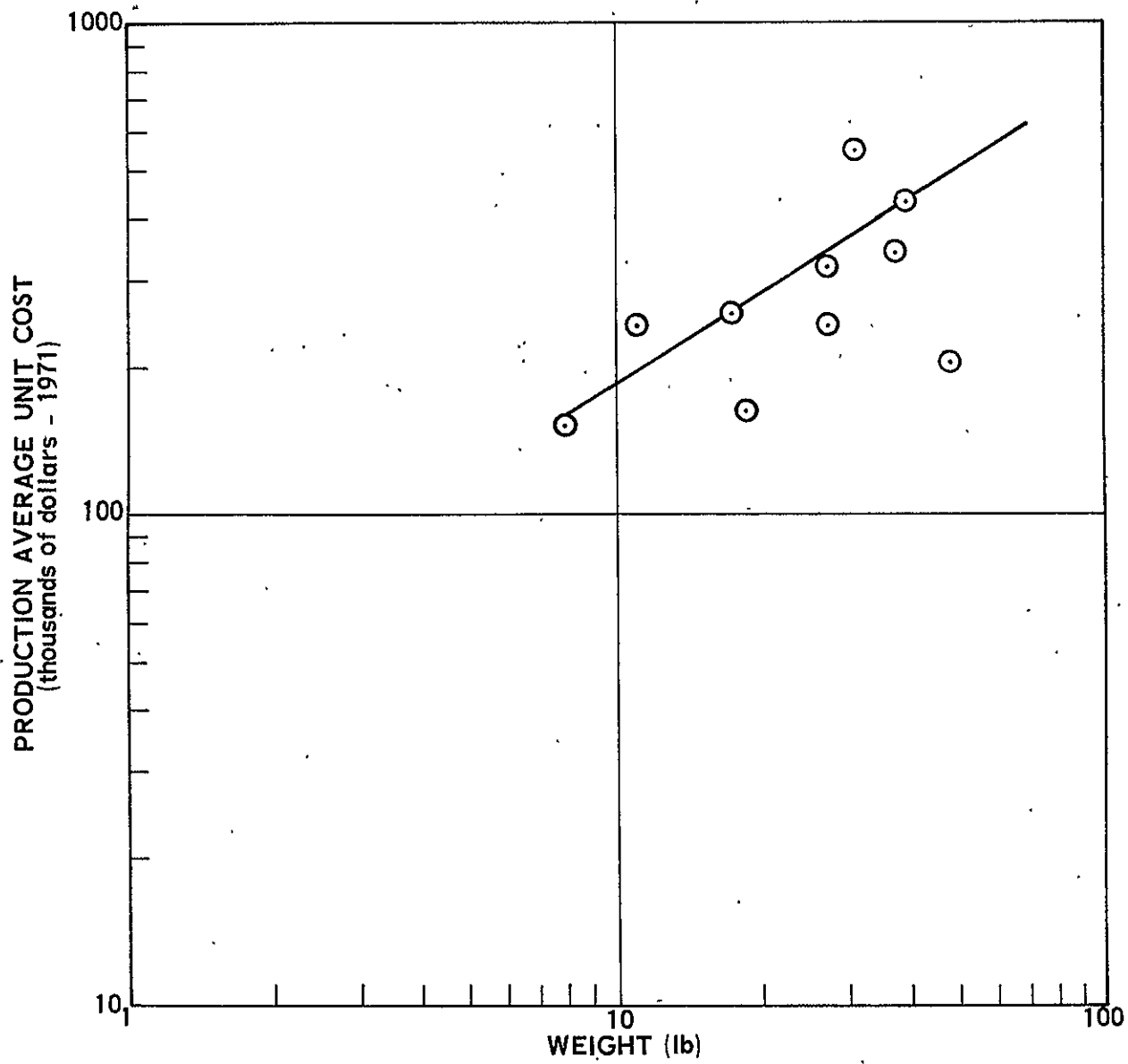


Figure 12-31. Command Decoding and Distribution Equipment: Production Average Unit Cost versus Weight

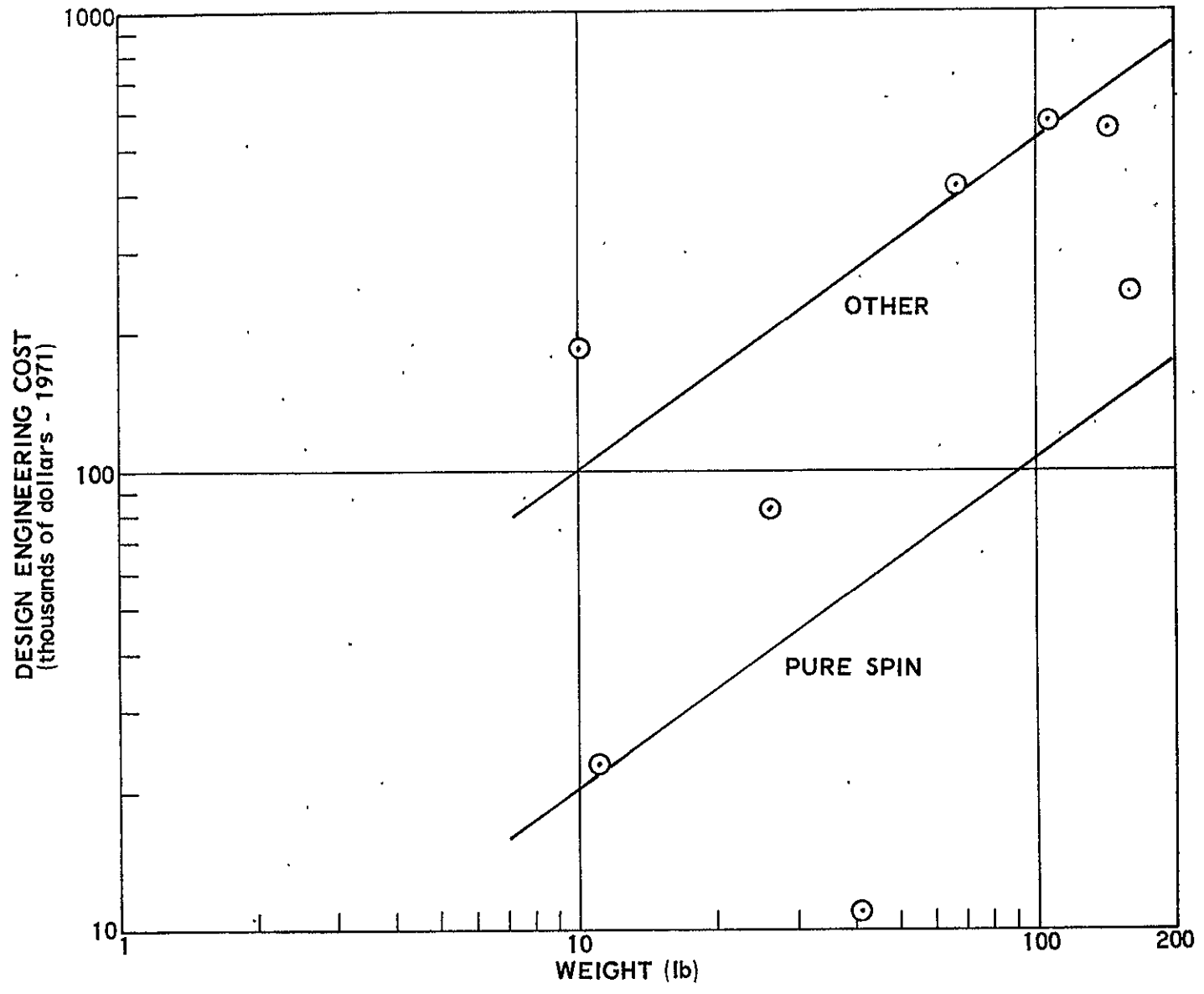


Figure 12-32. Wiring Harness: Design Engineering Cost versus Weight

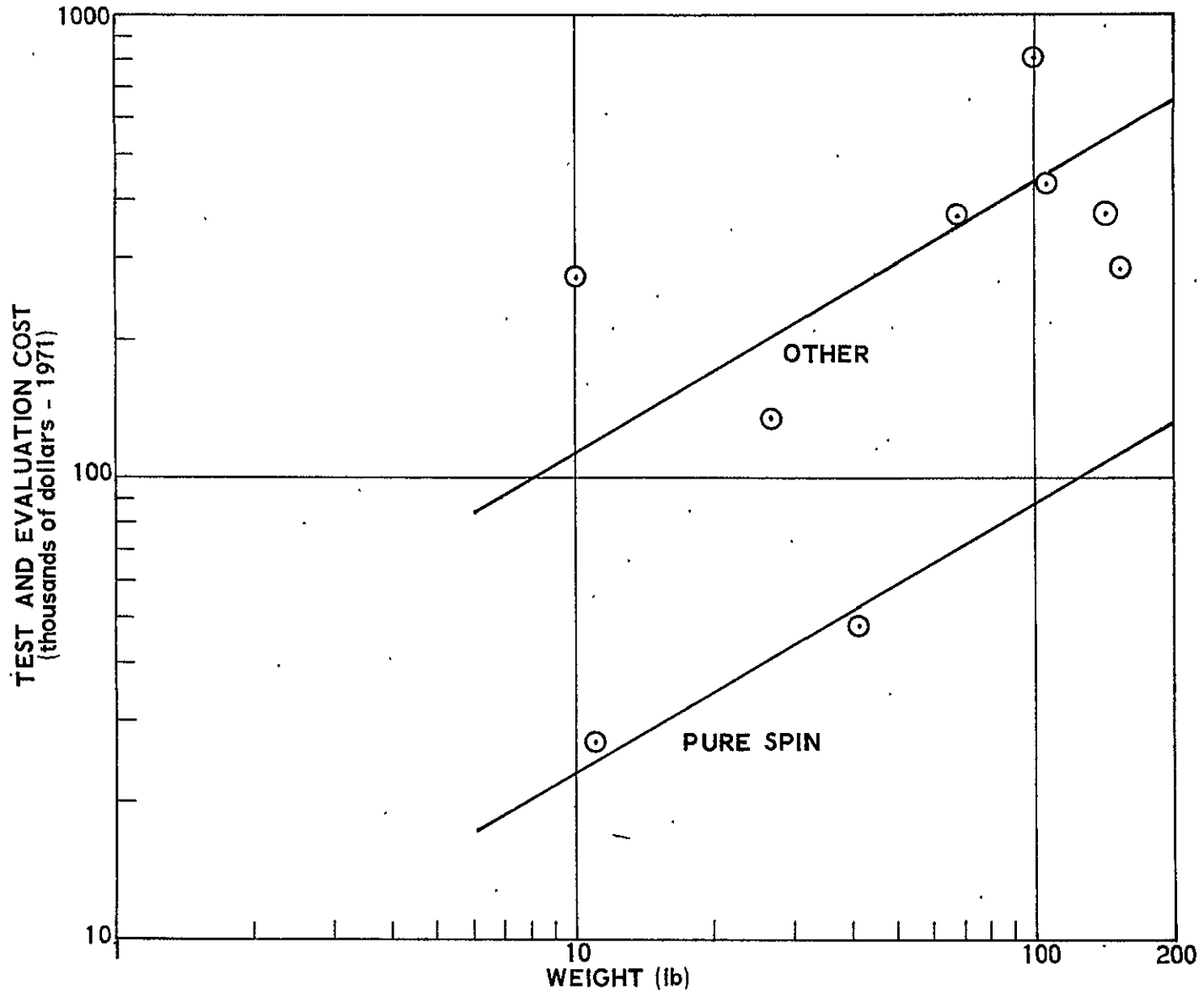


Figure 12-33. Wiring Harness: Test and Evaluation Cost versus Weight

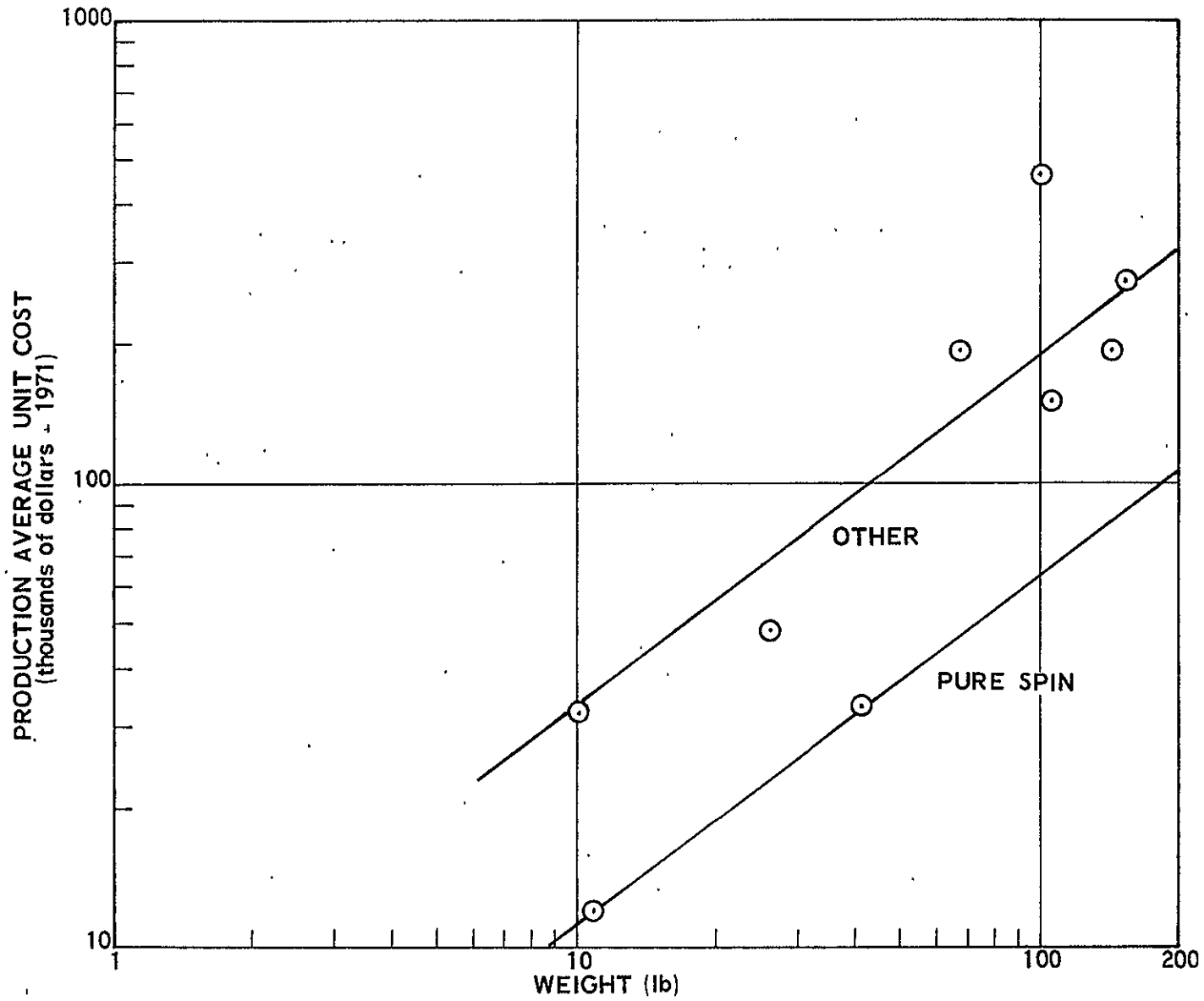


Figure 12-34. Wiring Harness: Production Average Unit Cost versus Weight

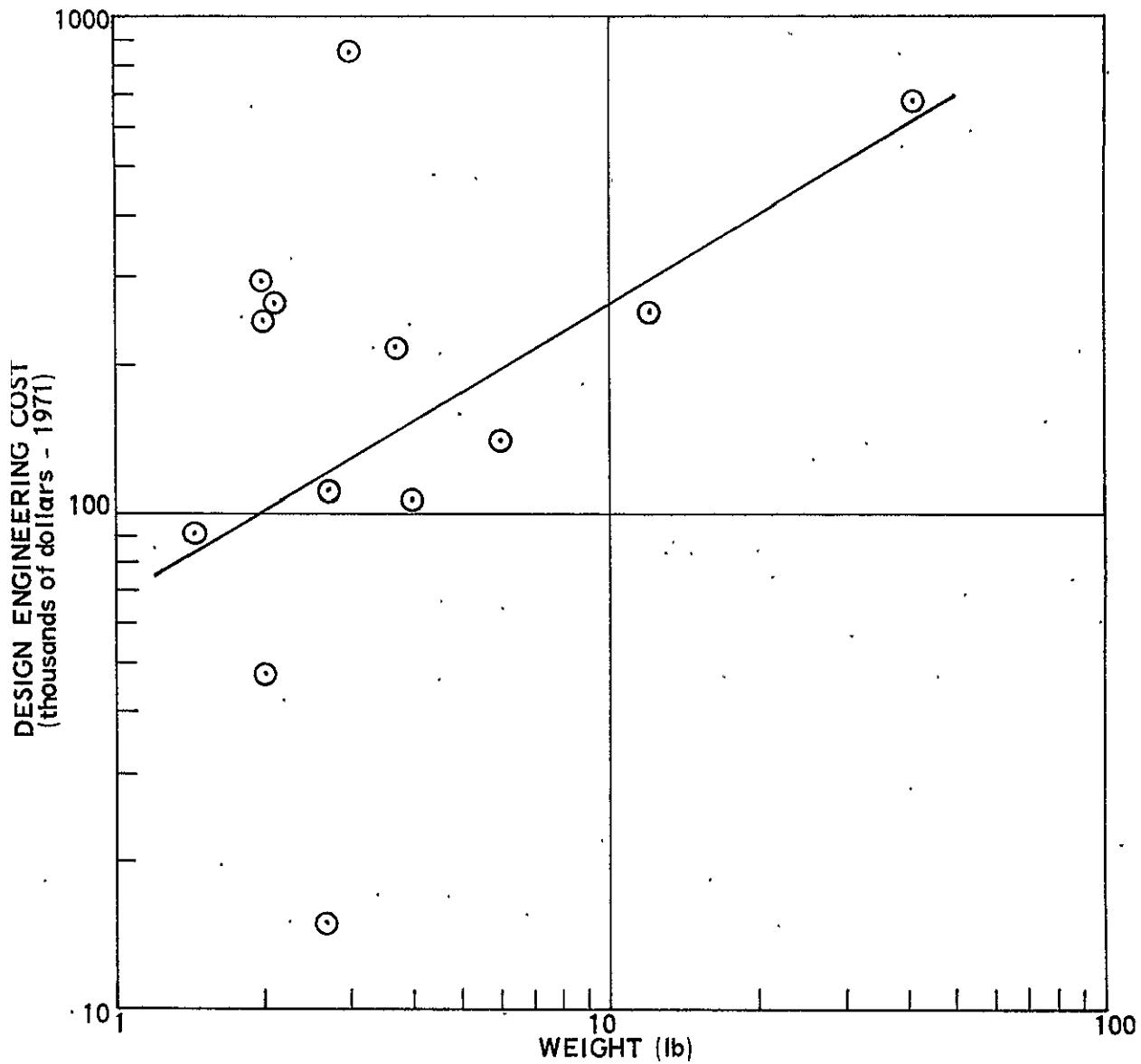


Figure 12-35. Antenna: Design Engineering Cost versus Weight

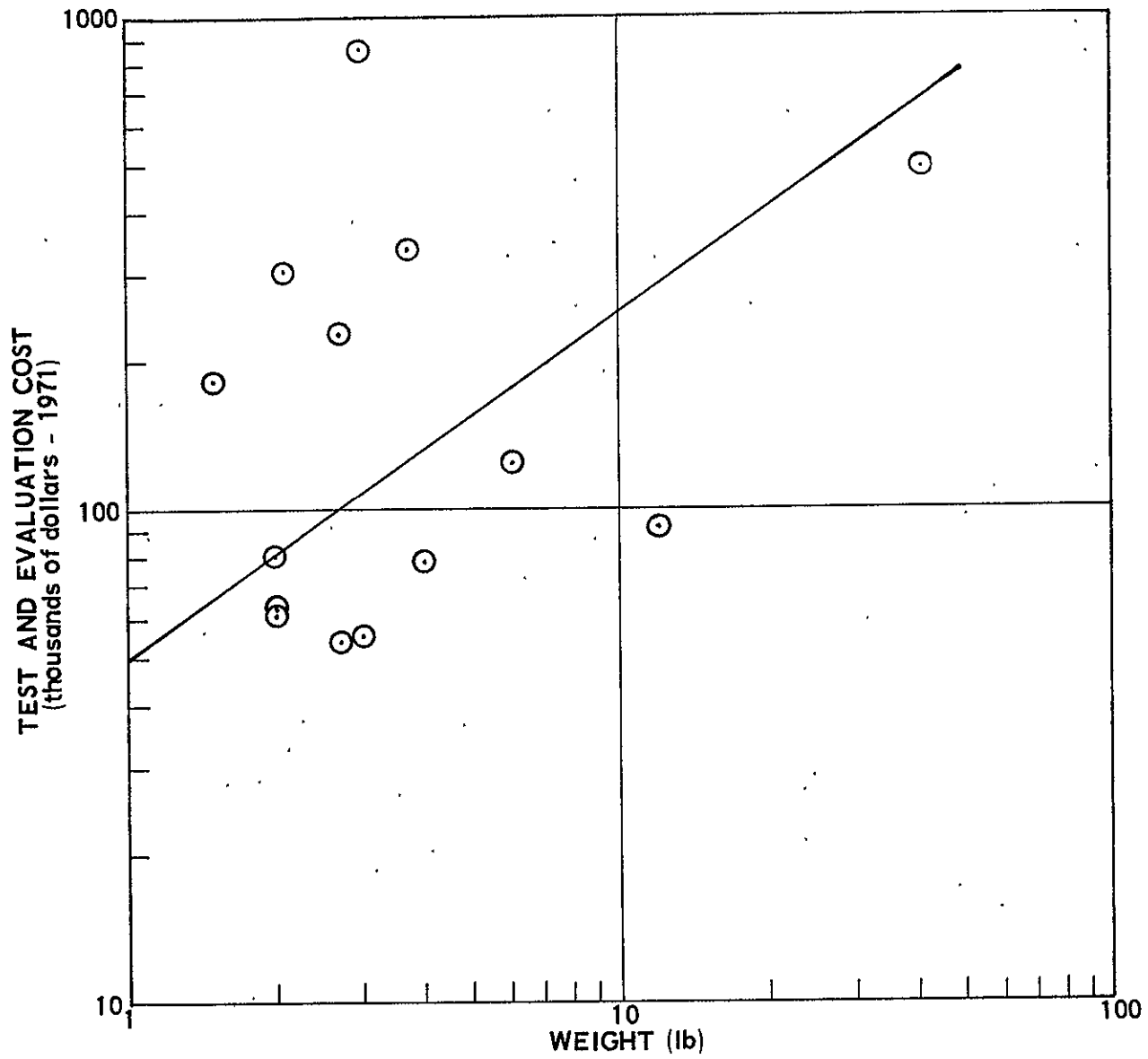


Figure 12-36. Antenna: Test and Evaluation Cost versus Weight

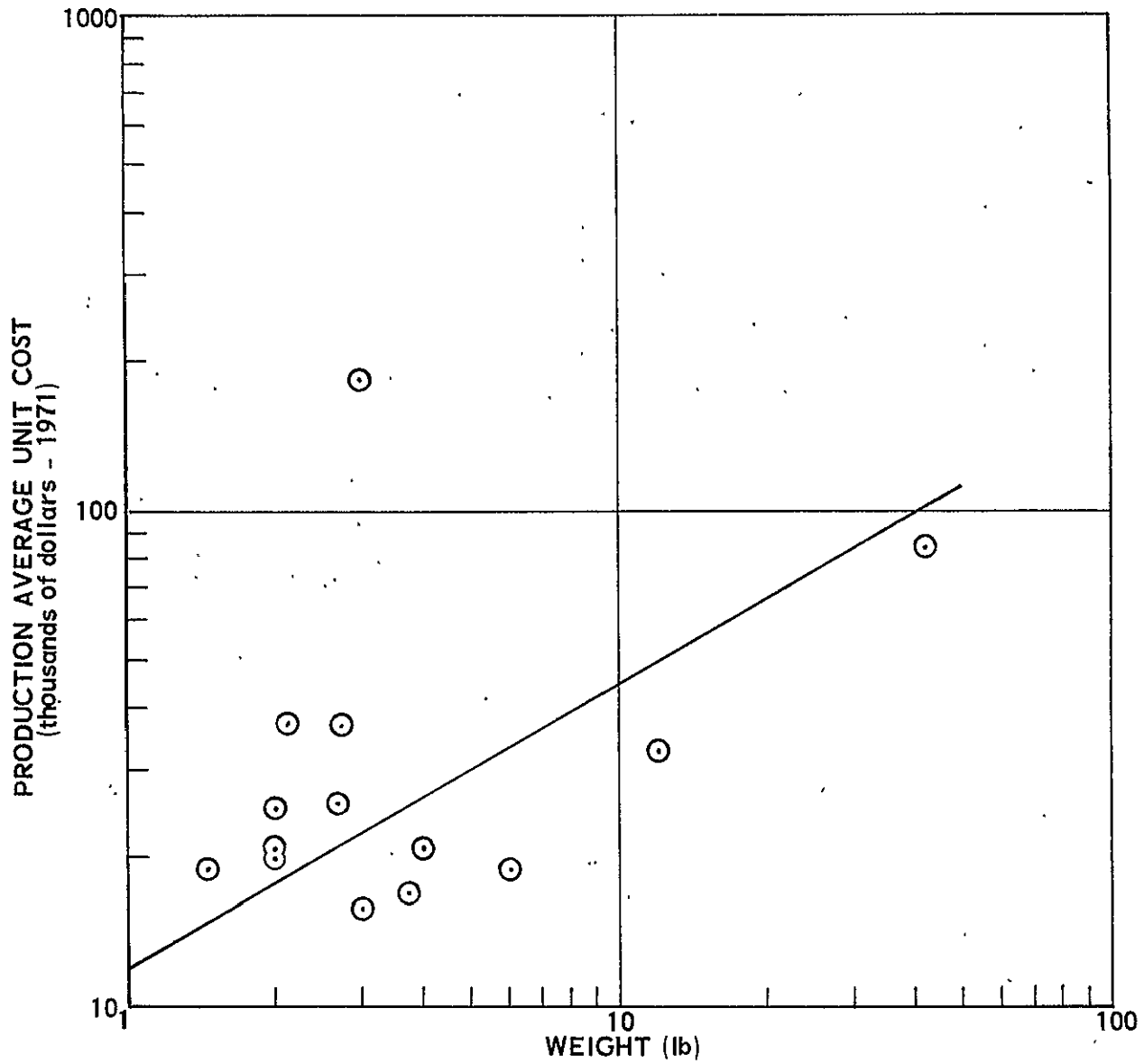


Figure 12-37. Antenna: Production Average Unit Cost versus Weight

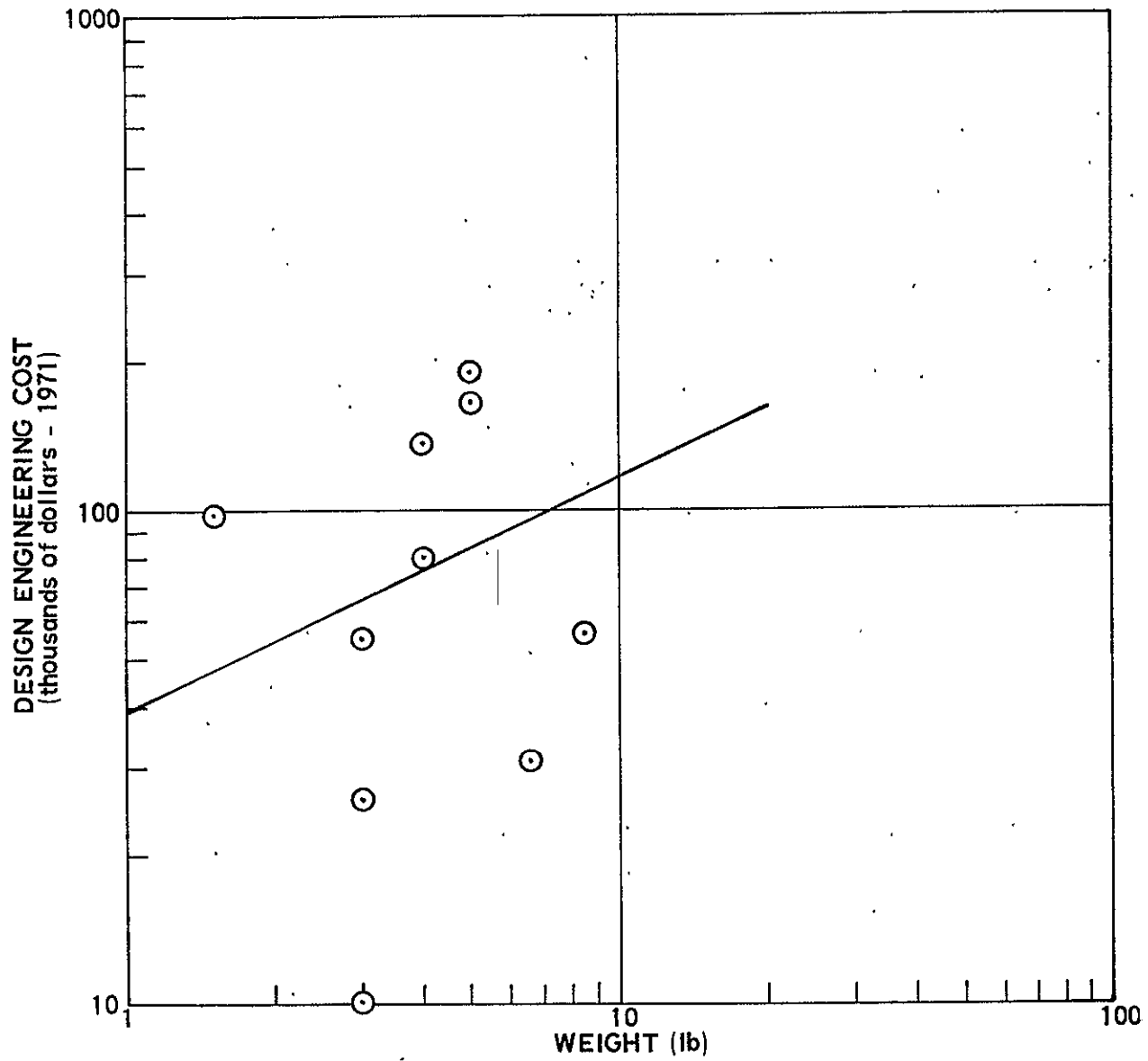


Figure 12-38. Receivers: Design Engineering Cost versus Weight

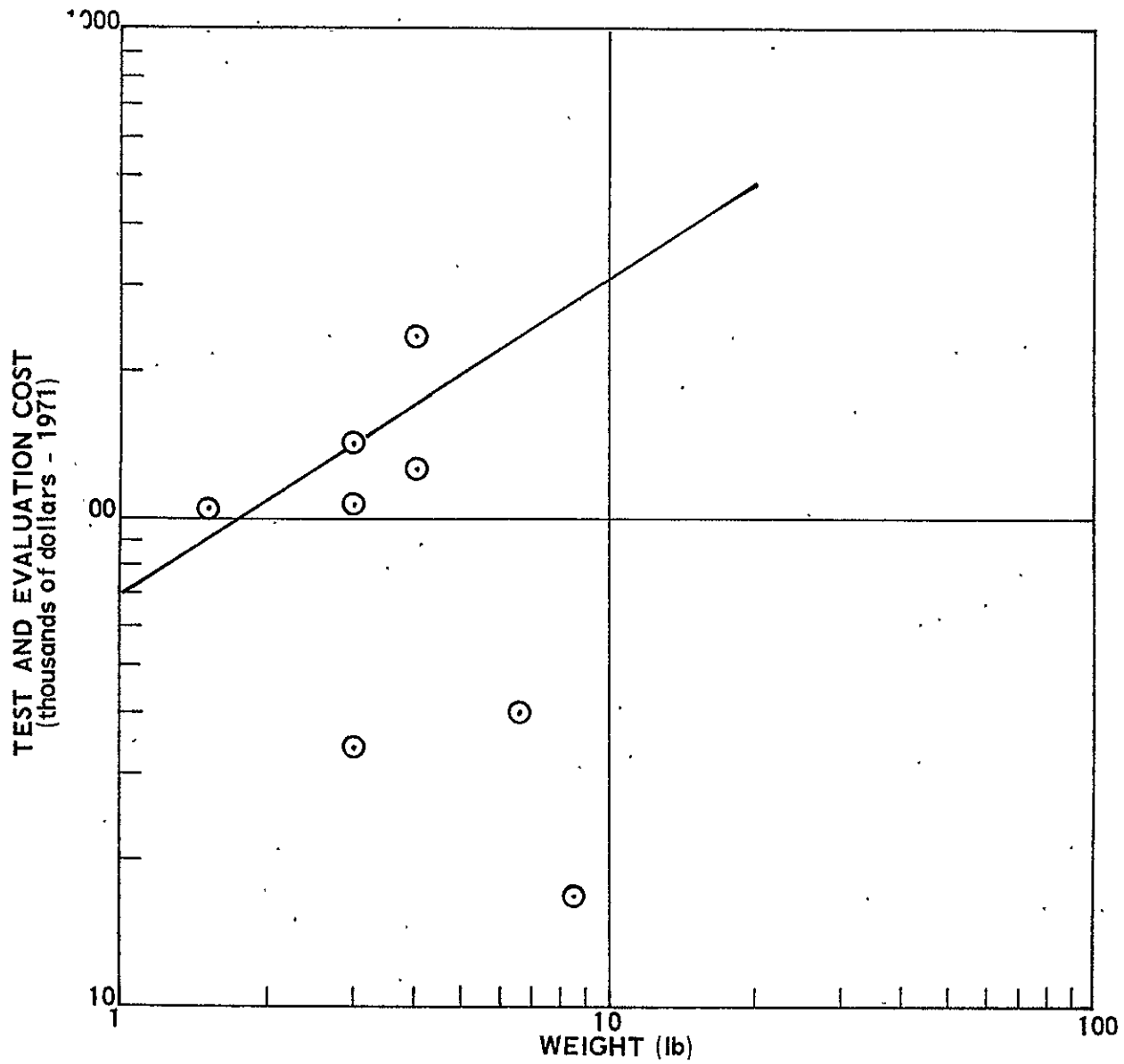


Figure 12-39. Receivers: Test and Evaluation Cost versus Weight

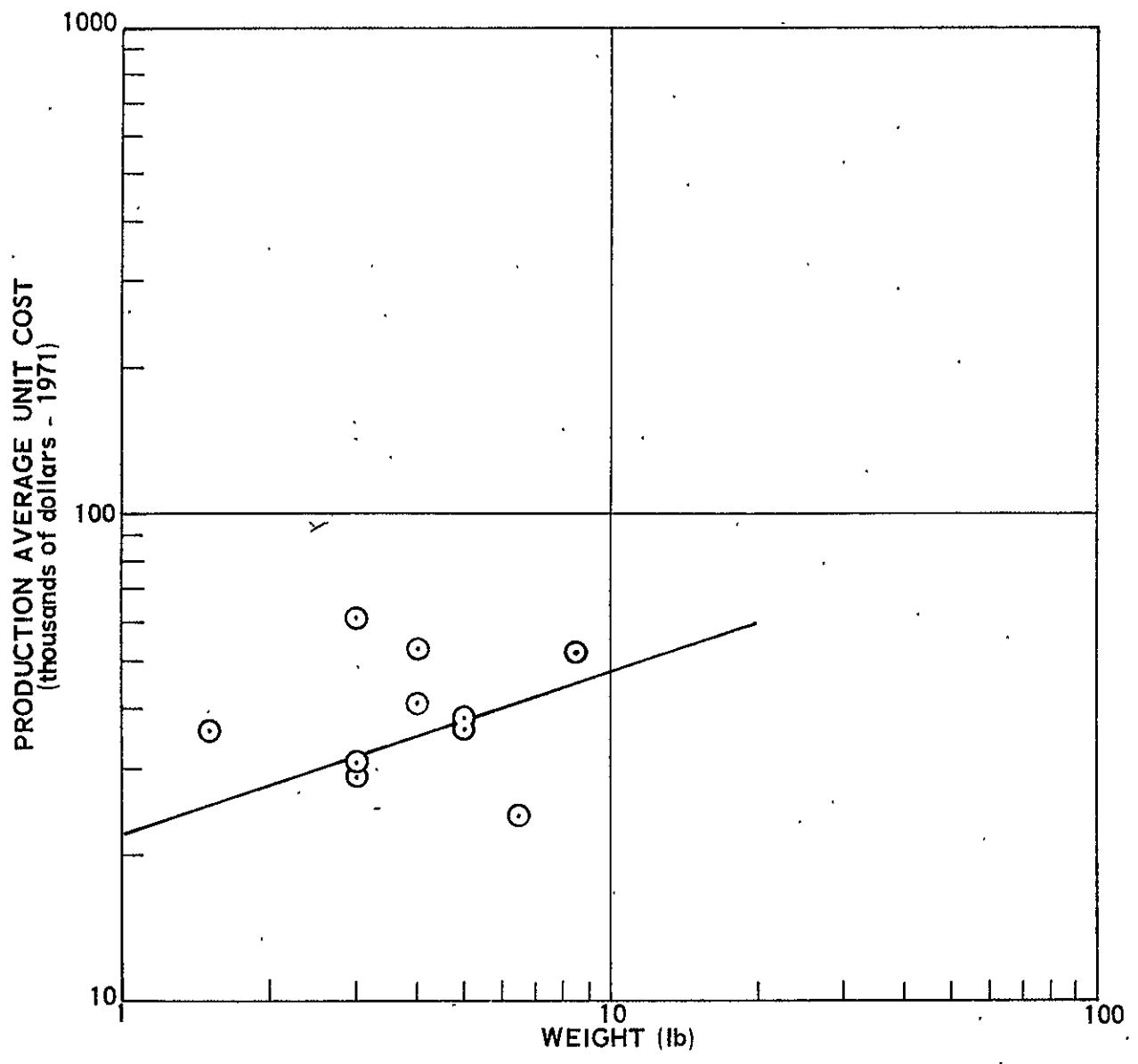


Figure 12-40. Receivers: Production Average Unit Cost versus Weight

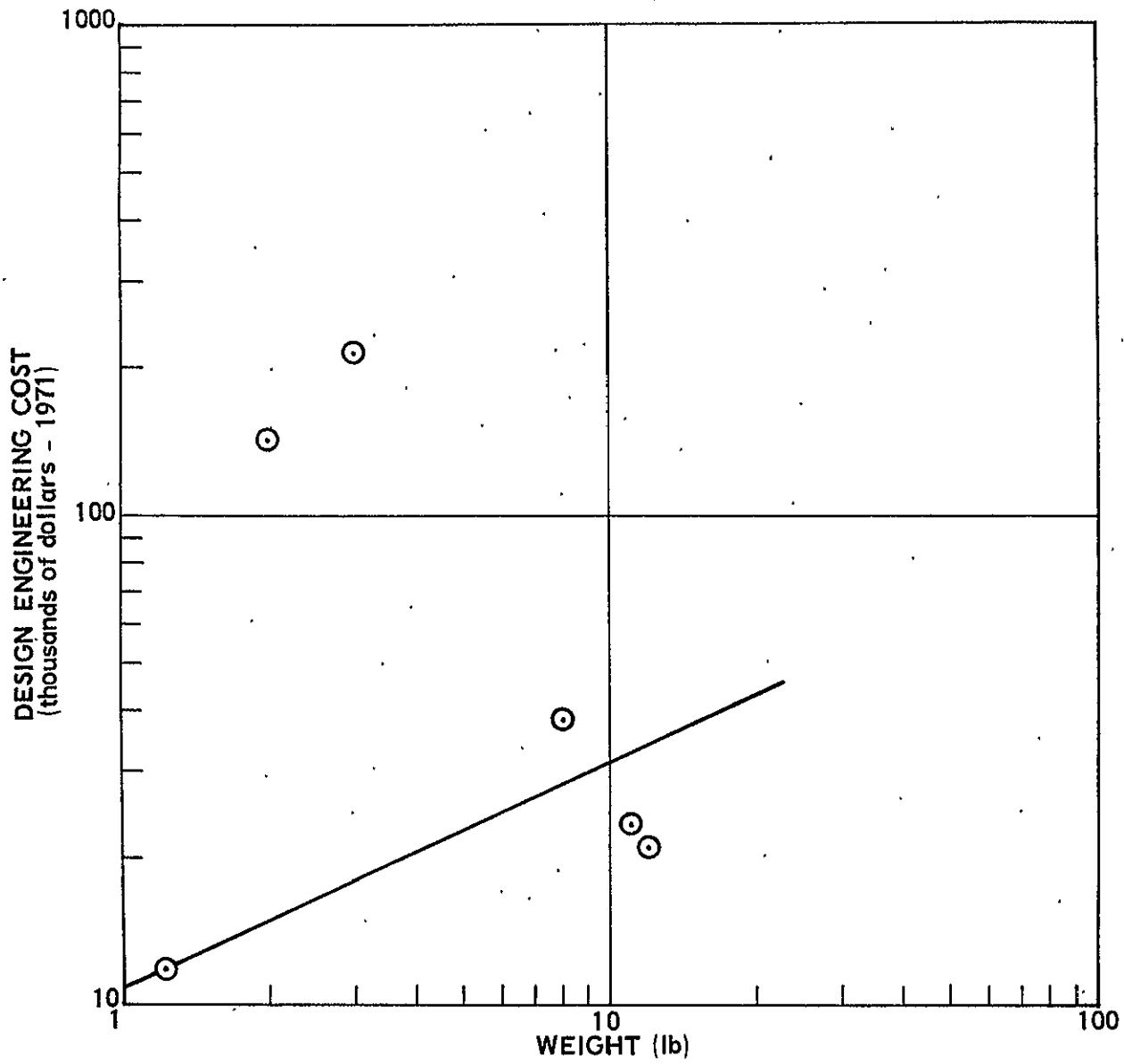


Figure 12-41. Diplexers: Design Engineering Cost versus Weight

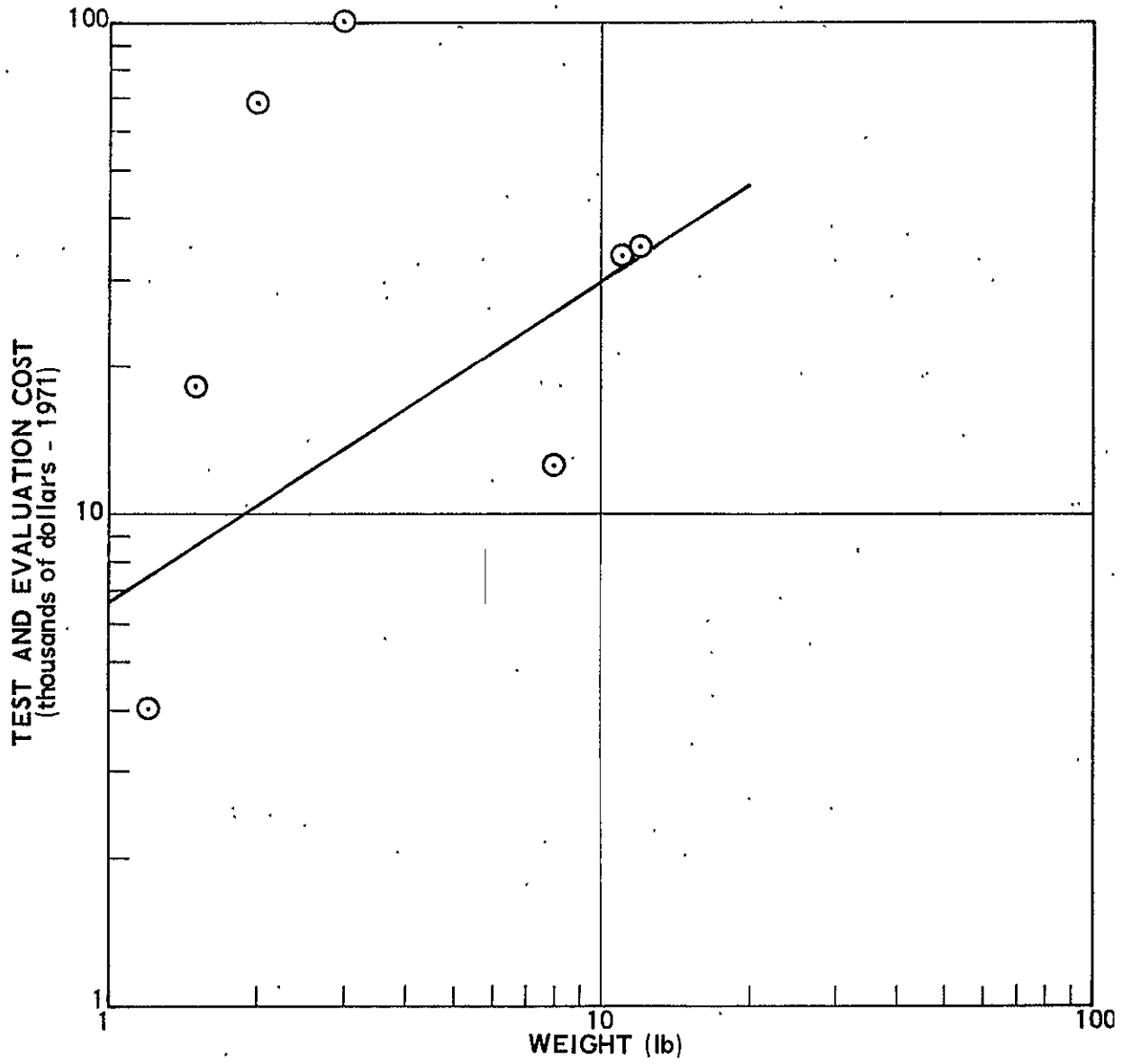


Figure 12-42. Diplexers: Test and Evaluation Cost versus Weight

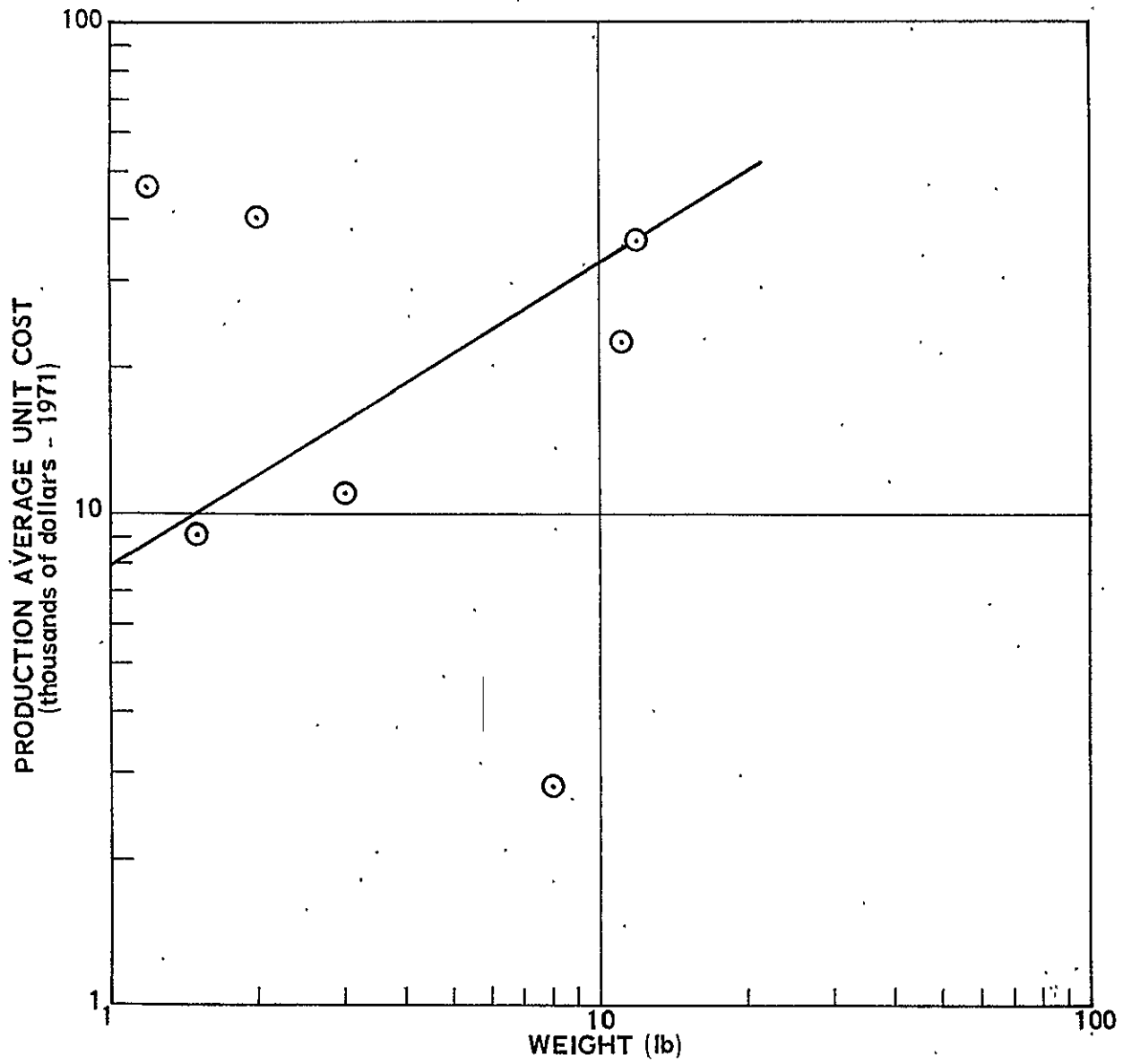


Figure 12-43. Diplexers: Production Average Unit Cost versus Weight

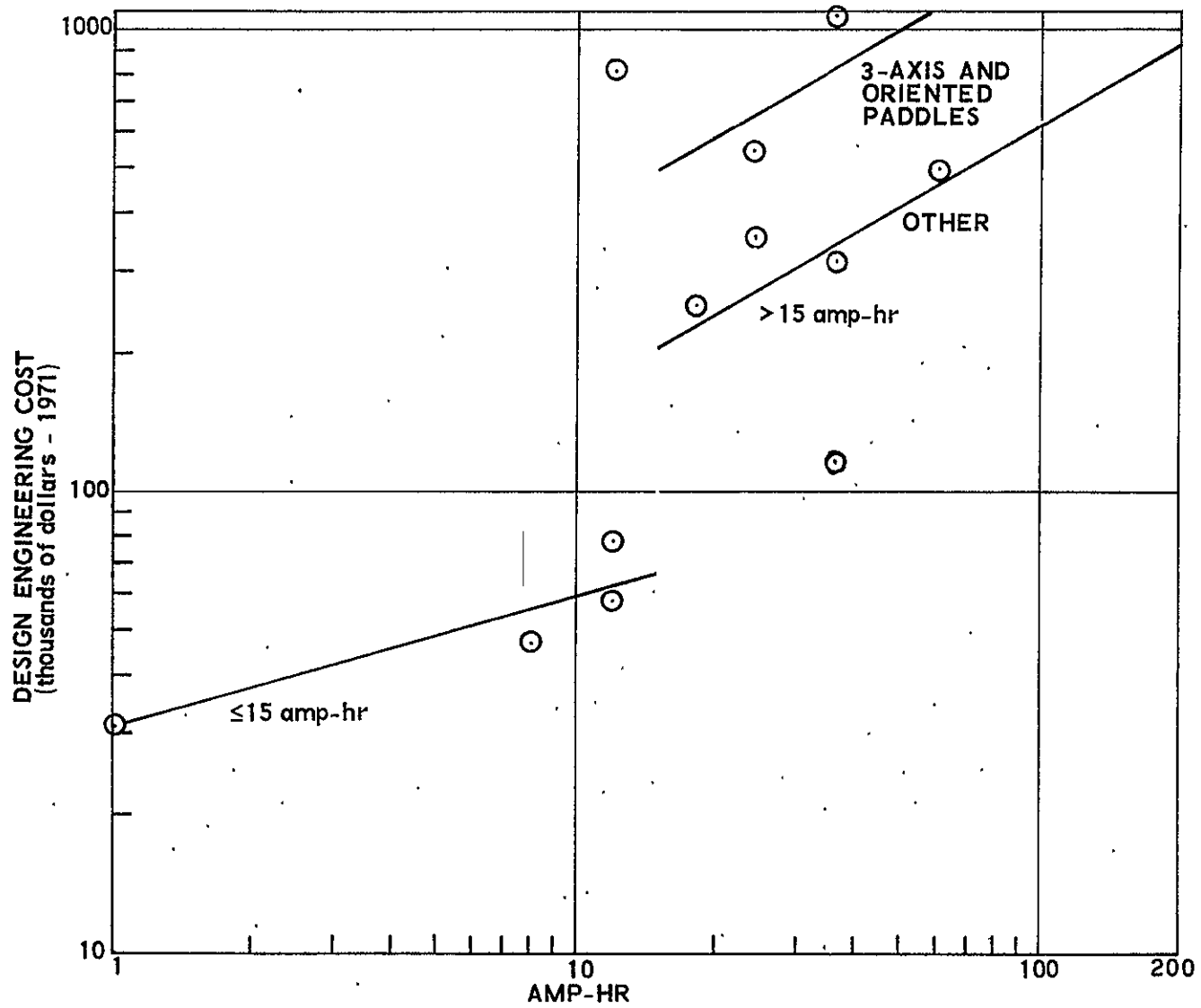


Figure 12-44. Batteries: Design Engineering Cost versus Ampere-Hours

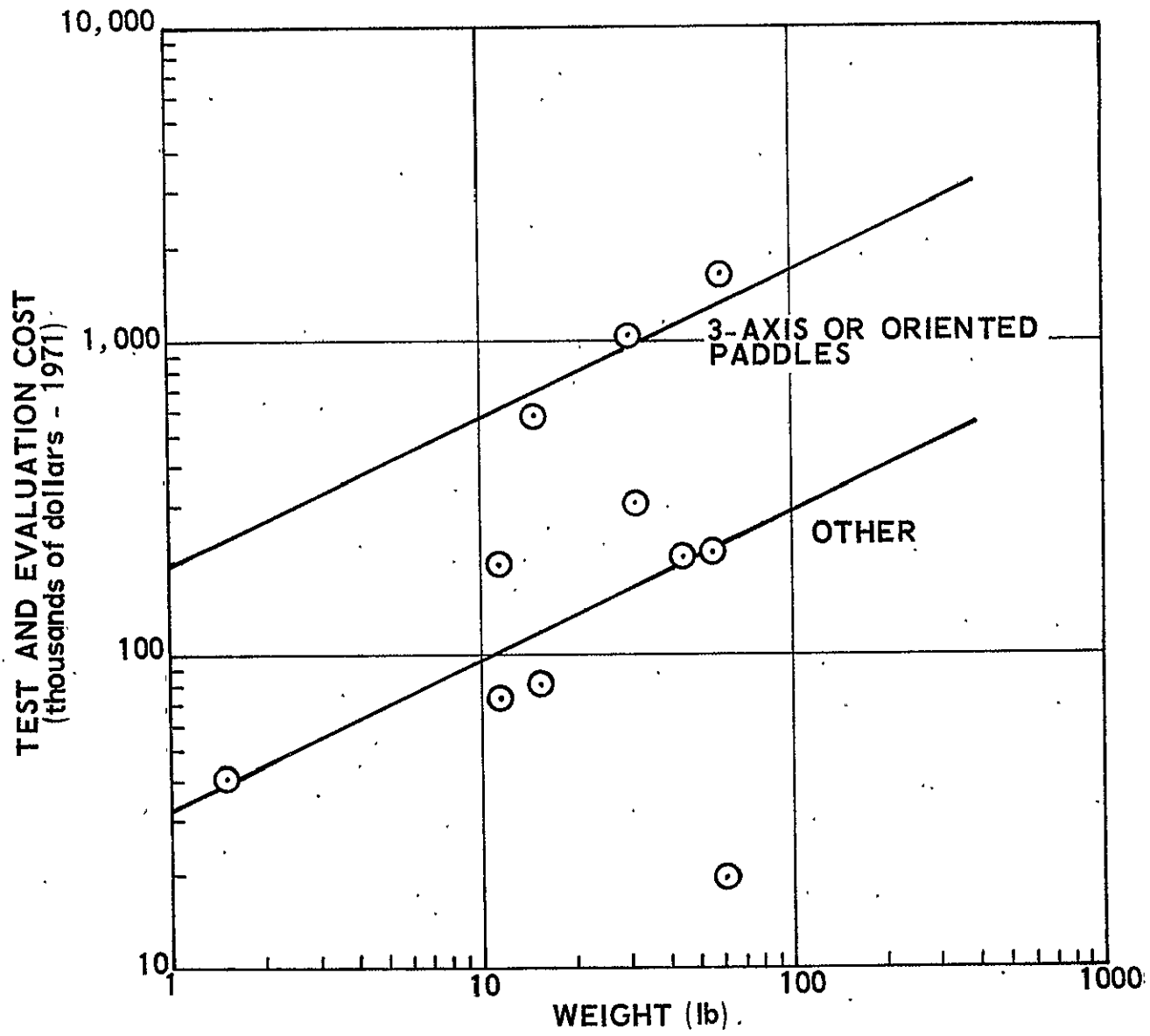


Figure 12-45. Batteries: Test and Evaluation Cost versus Weight

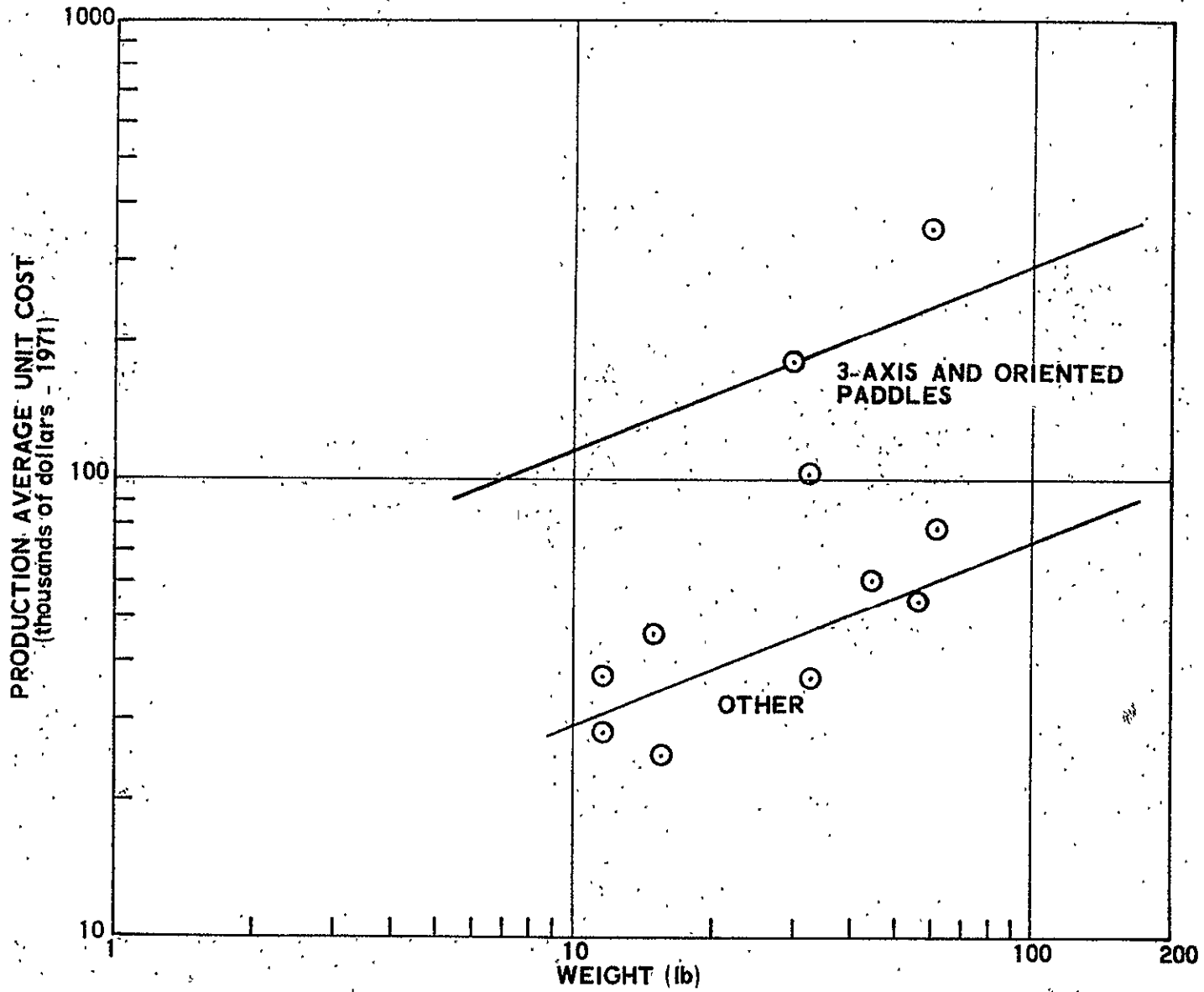


Figure 12-46. Batteries: Production Average Unit Cost versus Weight

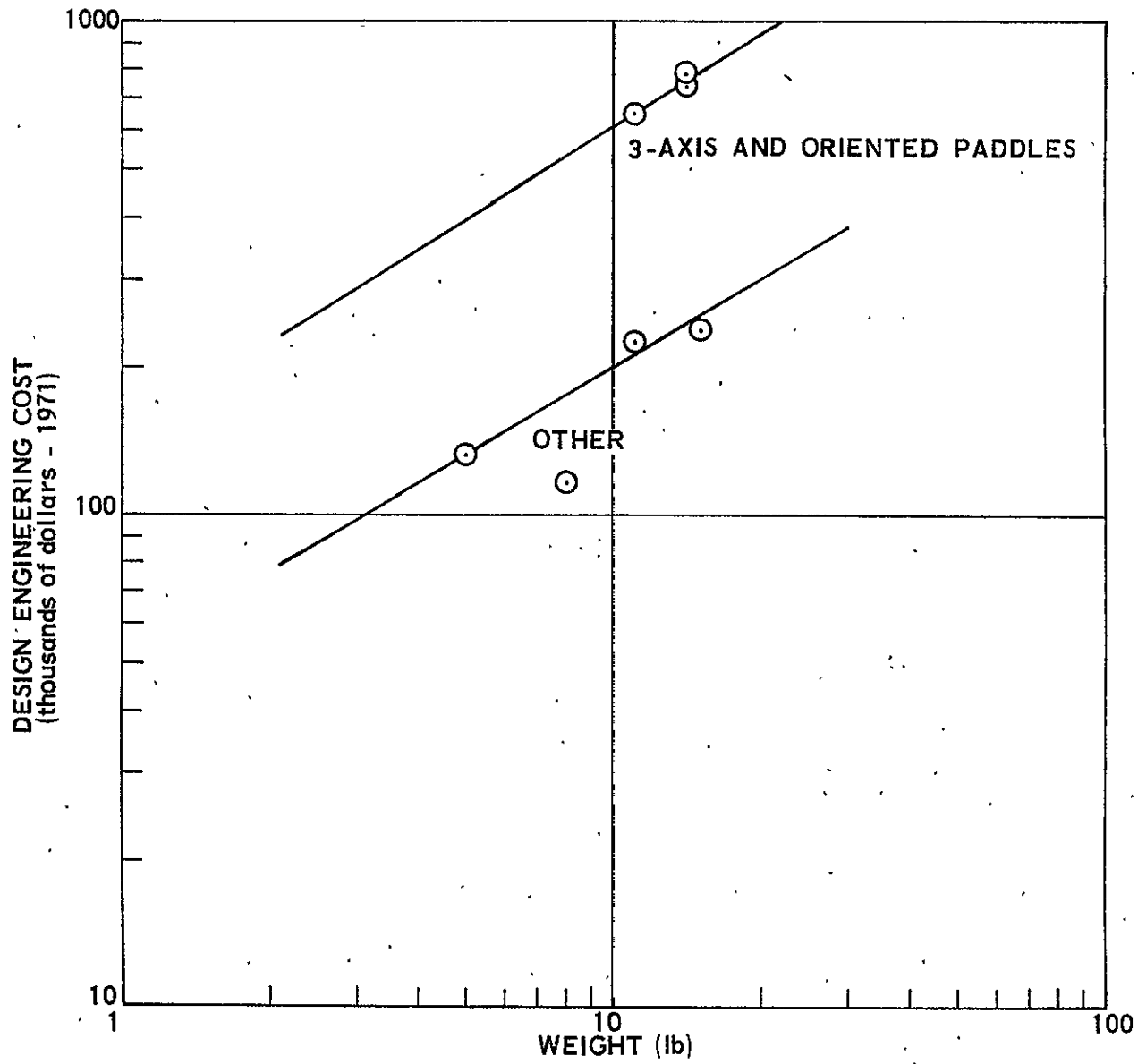


Figure 12-47. Power Control Equipment: Design Engineering Cost versus Weight

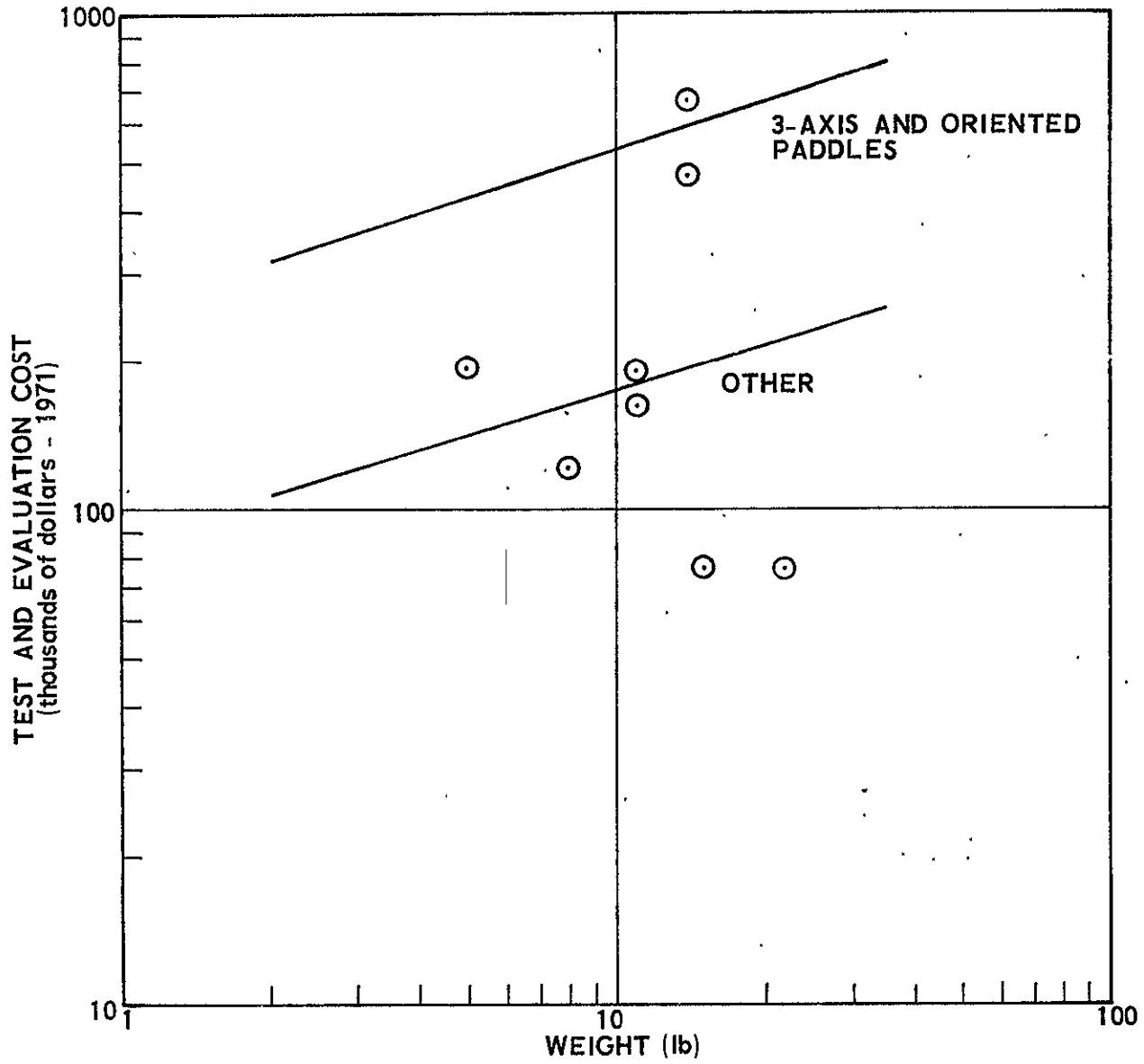


Figure 12-48. Power Control Equipment: Test and Evaluation Cost versus Weight

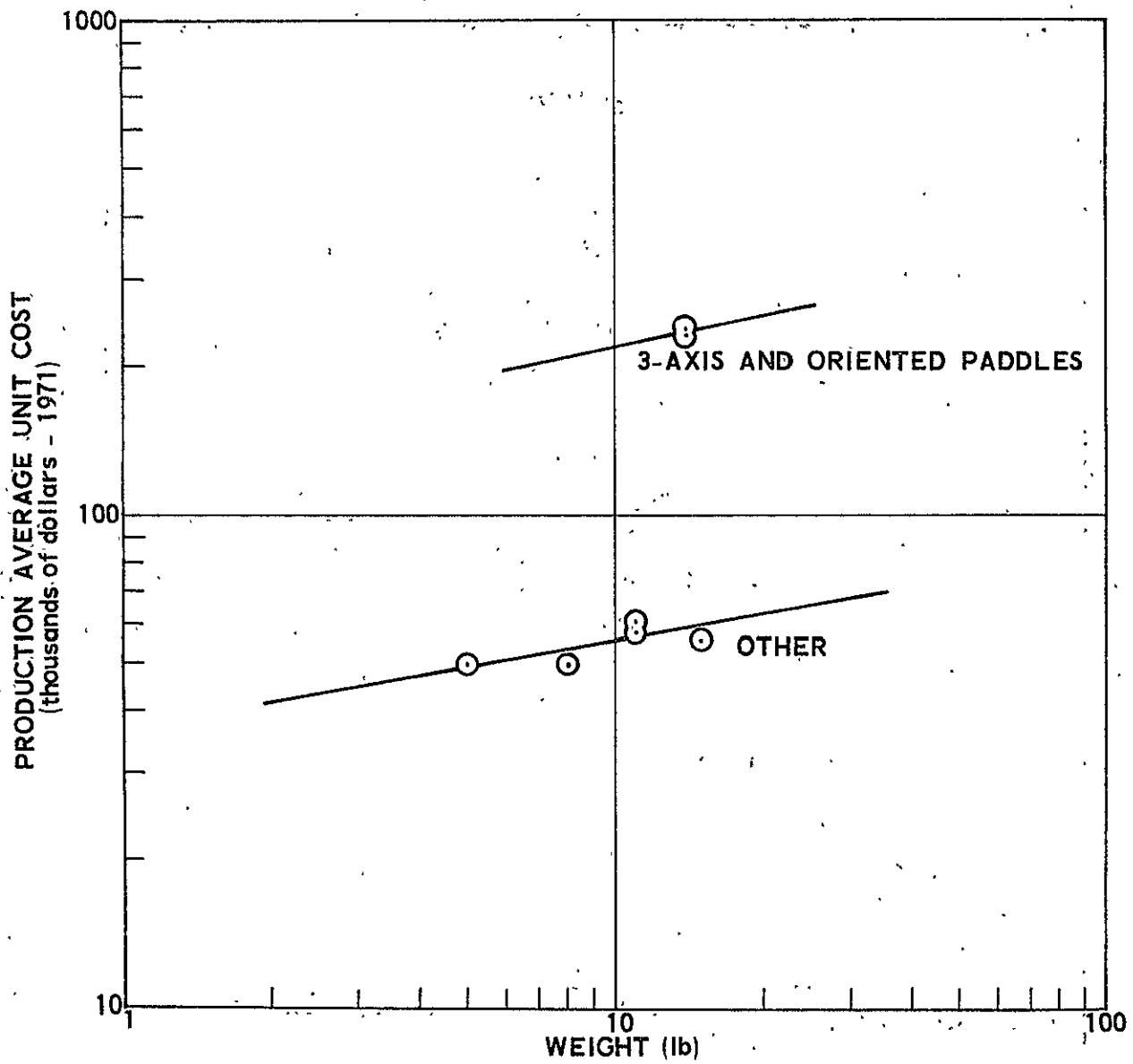


Figure 12-49. Power Control Equipment: Production Average Unit Cost versus Weight

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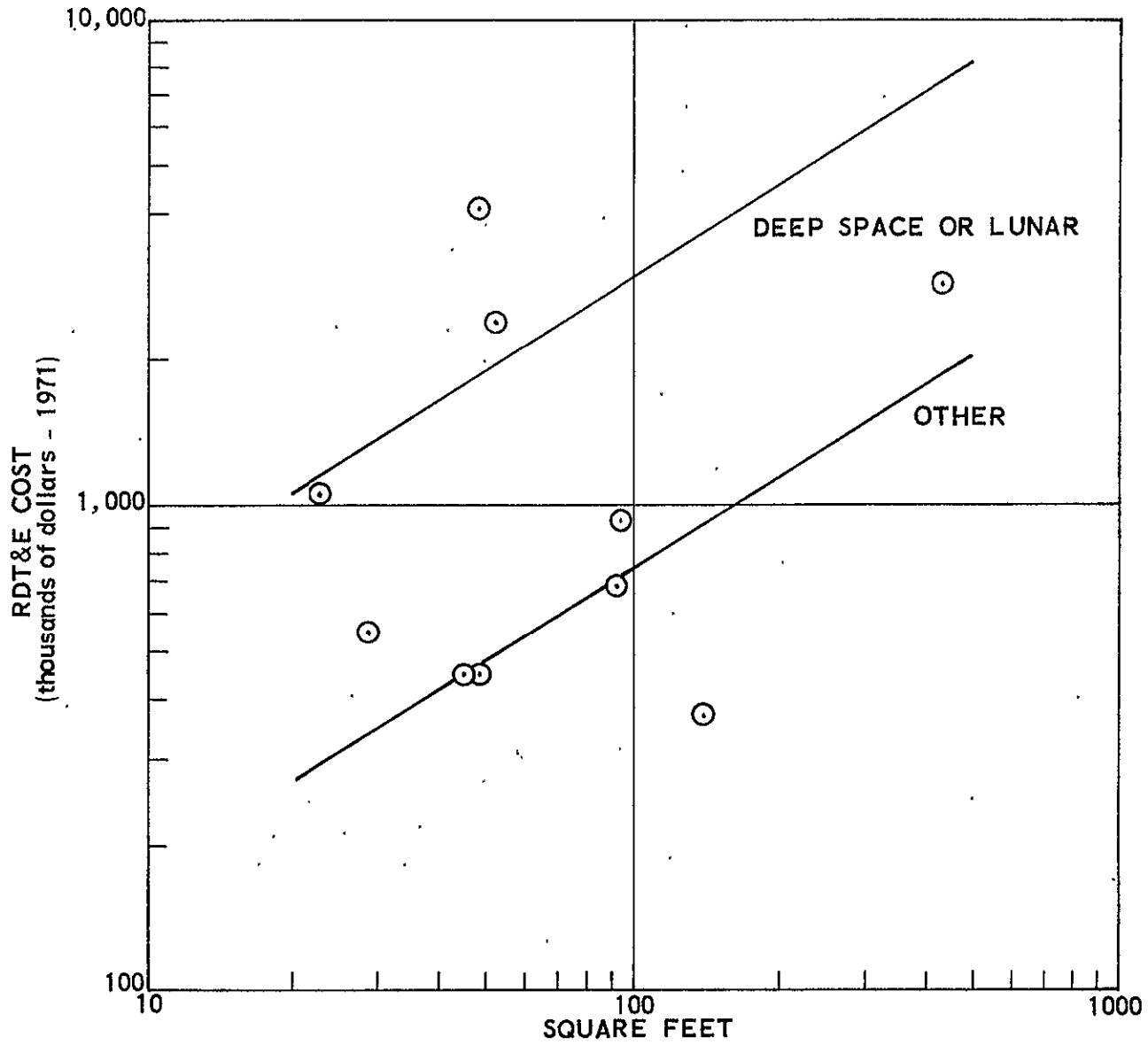


Figure 12-50. Solar Array: RDT&E Cost versus Square Feet

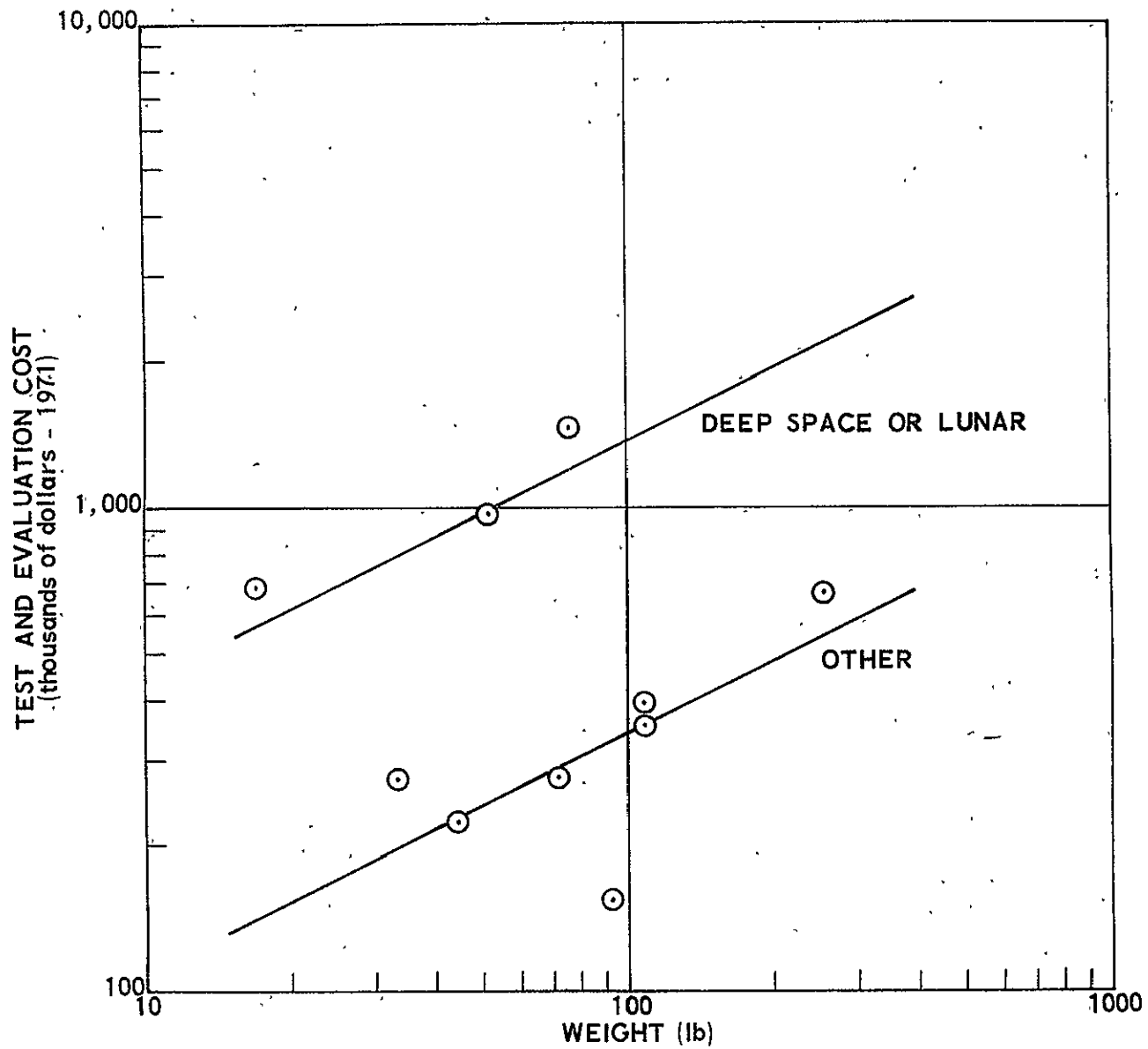


Figure 12-51. Solar Array: Test and Evaluation Cost versus Weight

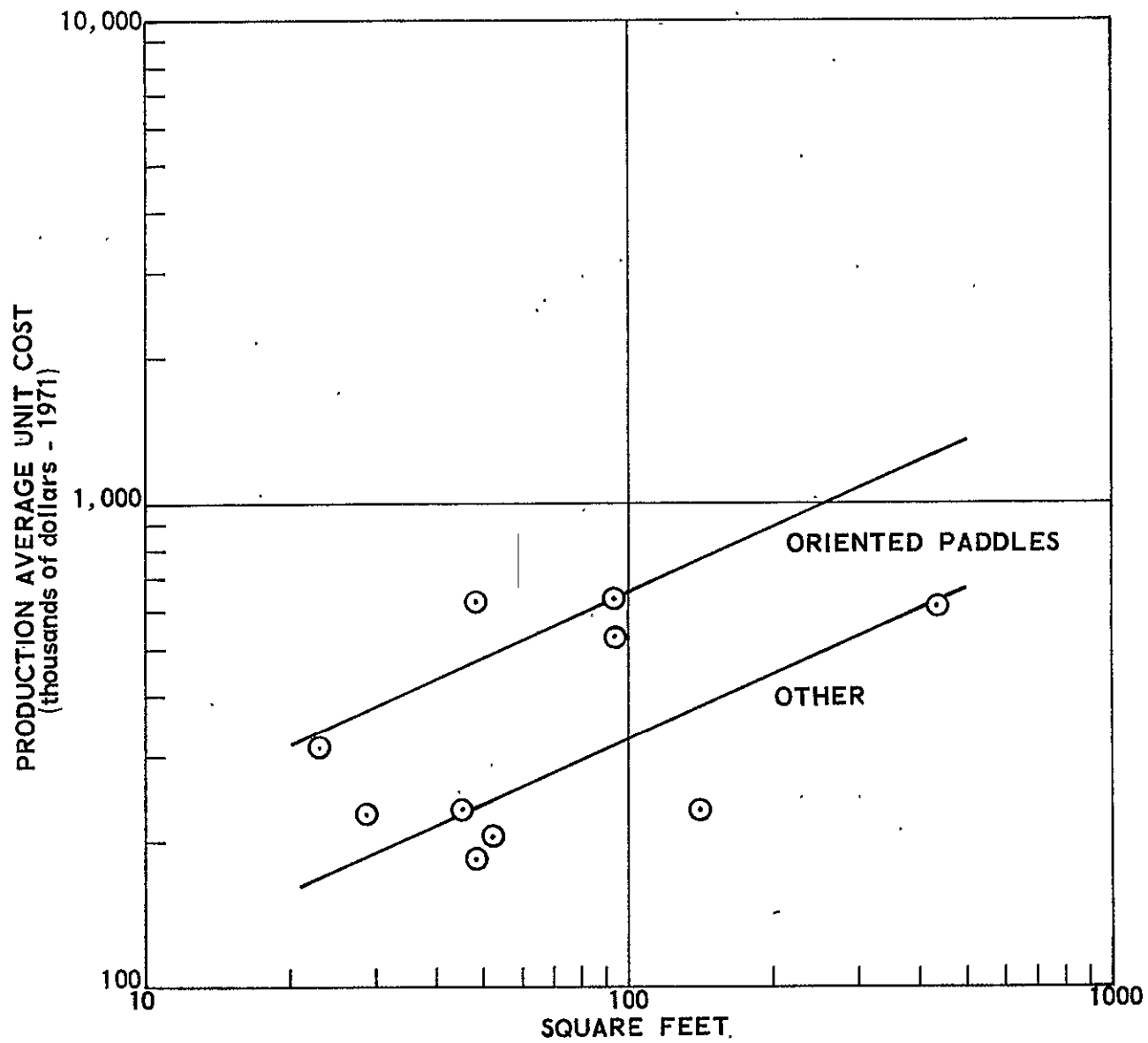


Figure 12-52. Solar Array: Production Average Unit Cost versus Square Feet

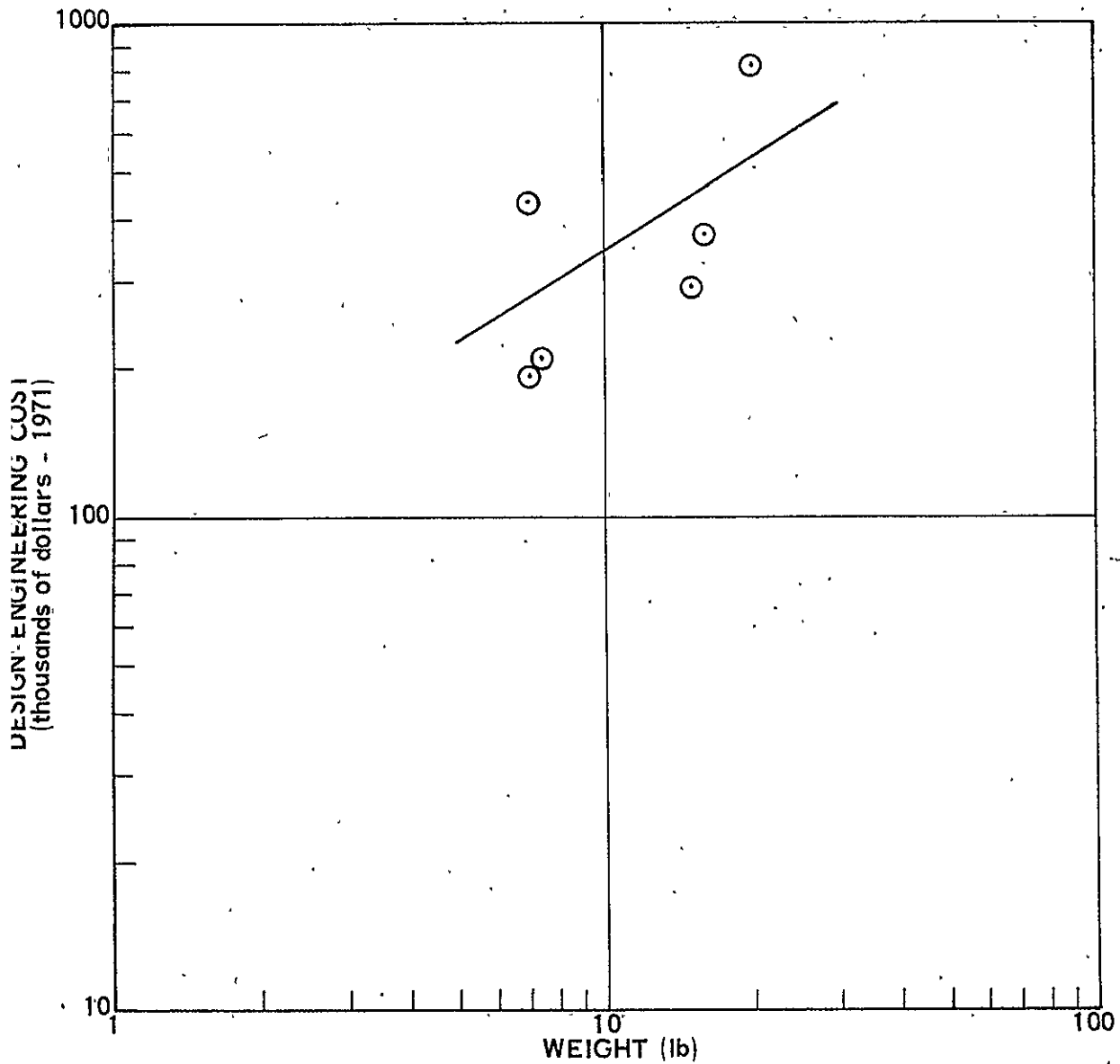


Figure 12-53. Power Converters: Design Engineering Cost versus Weight

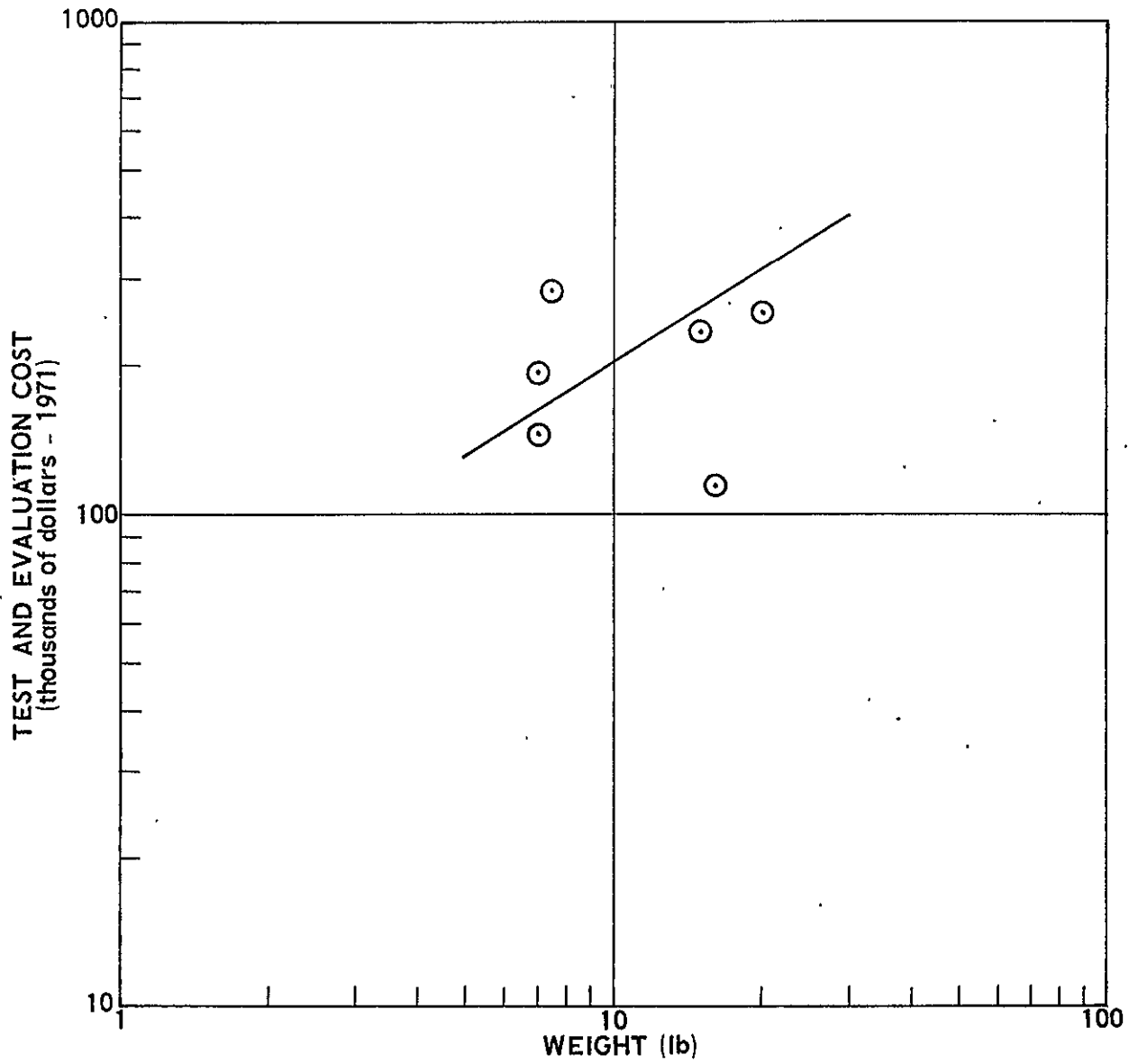


Figure 12-54. Power Converters: Test and Evaluation Cost versus Weight

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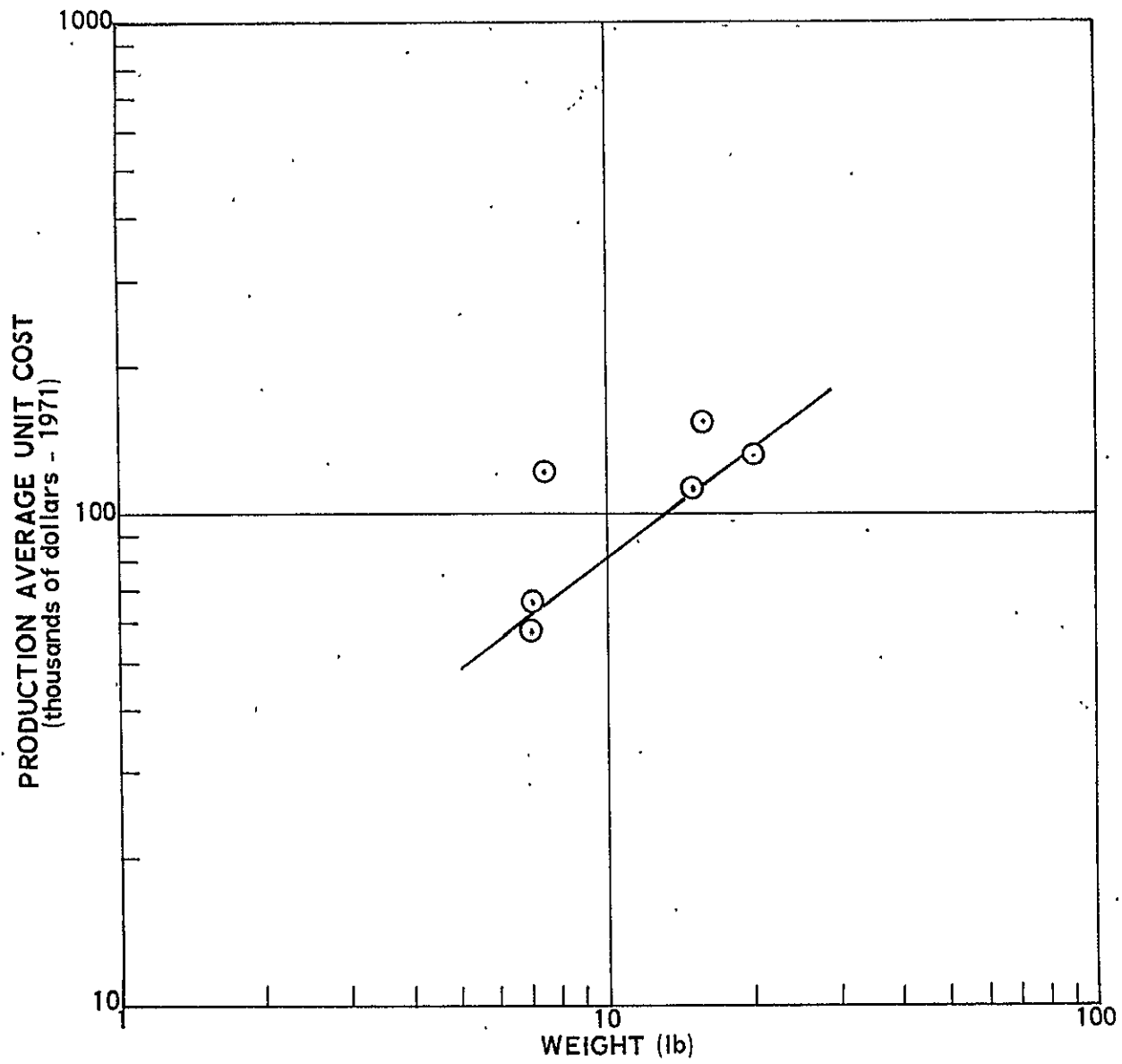


Figure 12-55. Power Converters: Production Average Unit Cost versus Weight

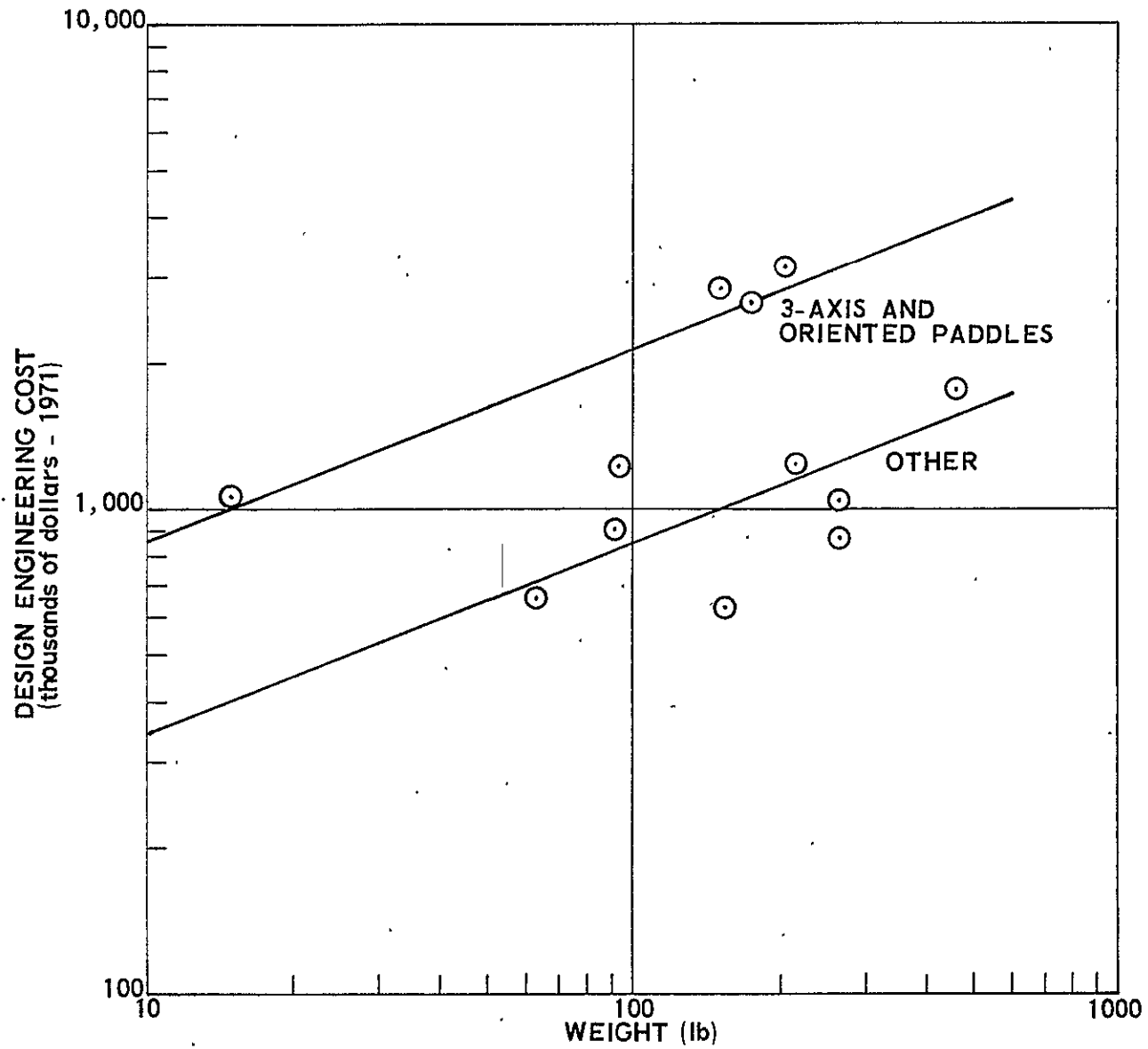


Figure 12-56. Structure: Design Engineering Cost versus Weight

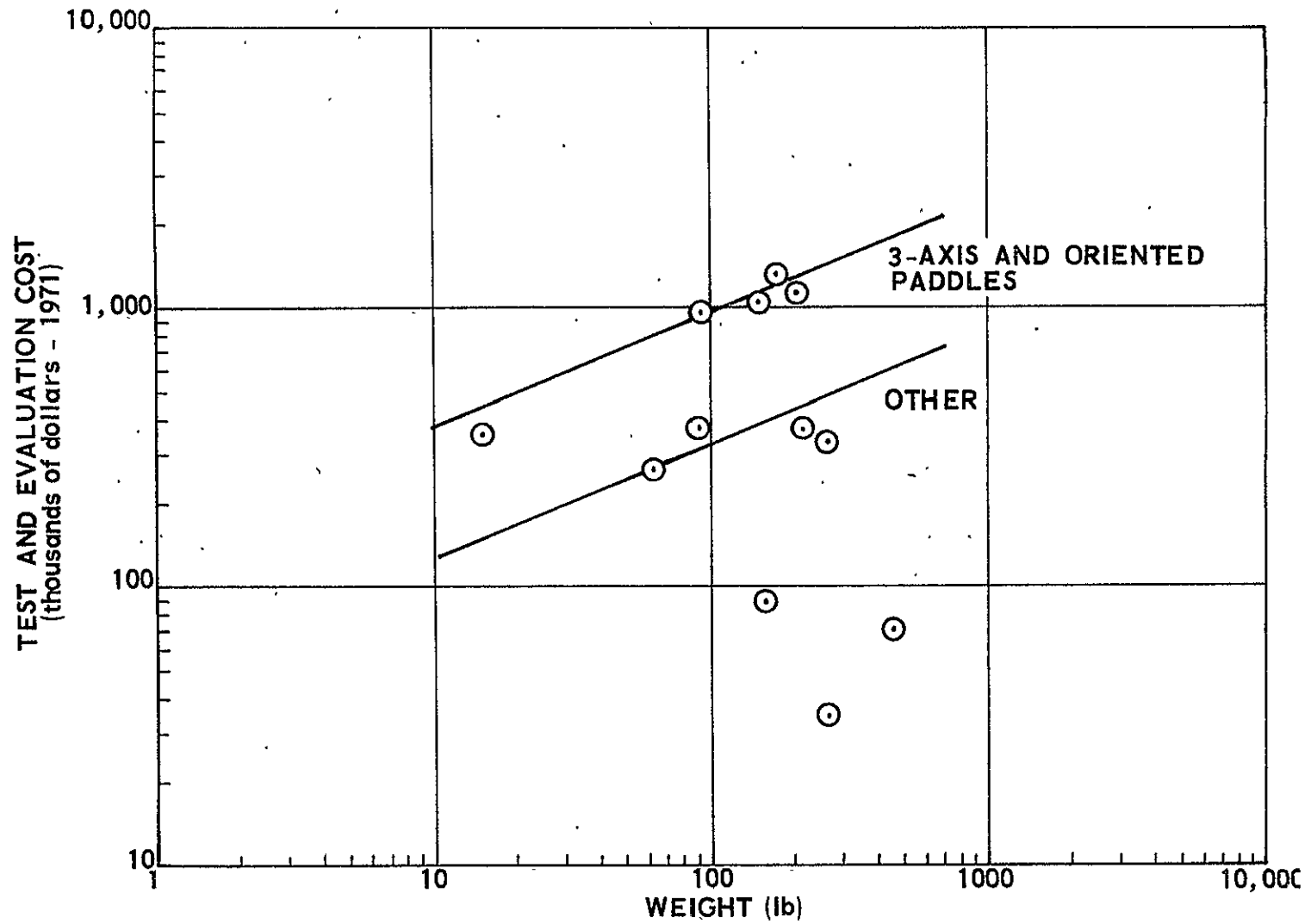


Figure 12-57. Structure: Test and Evaluation Cost versus Weight

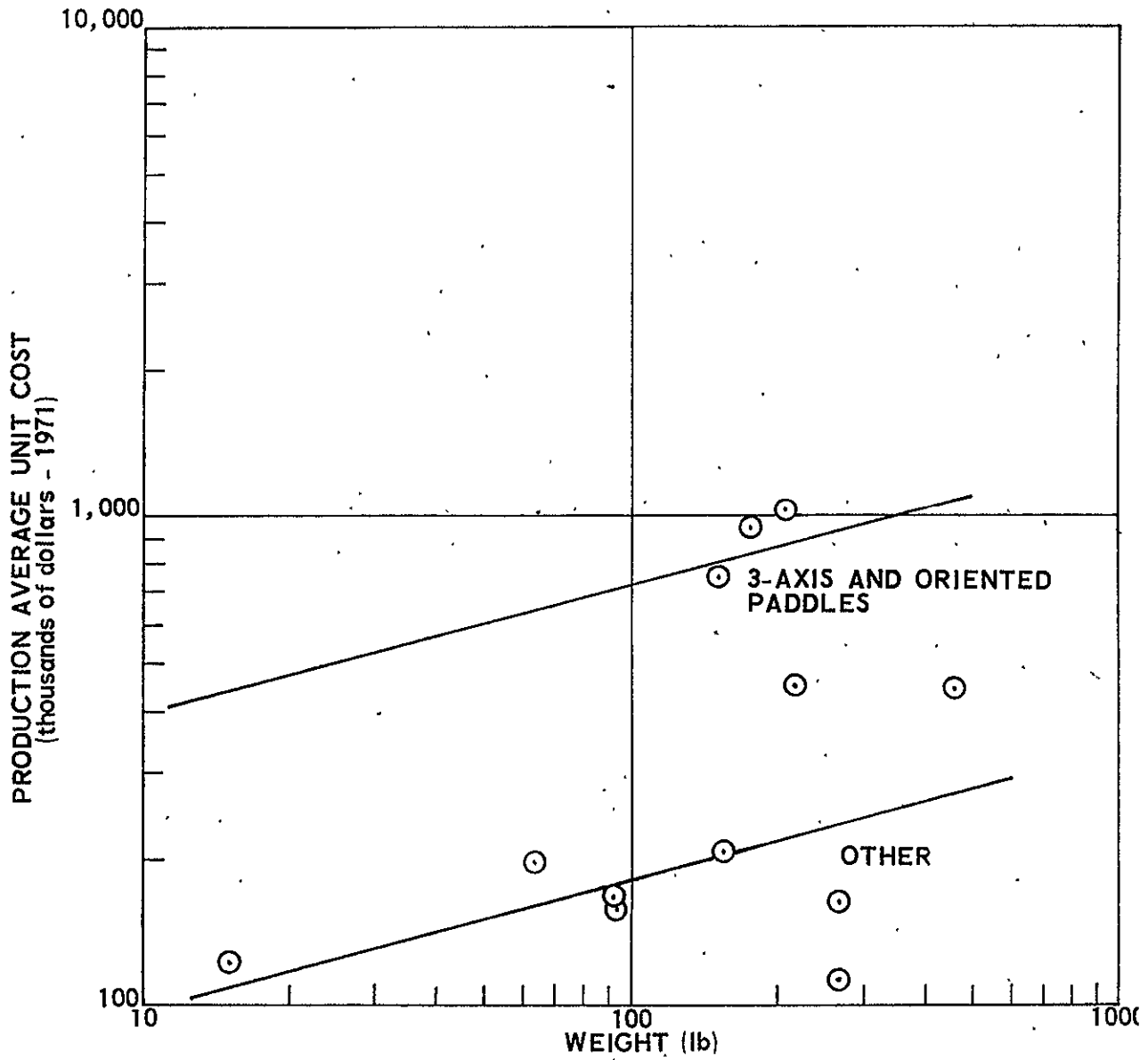


Figure 12-58. Structure: Production Average Unit Cost versus Weight

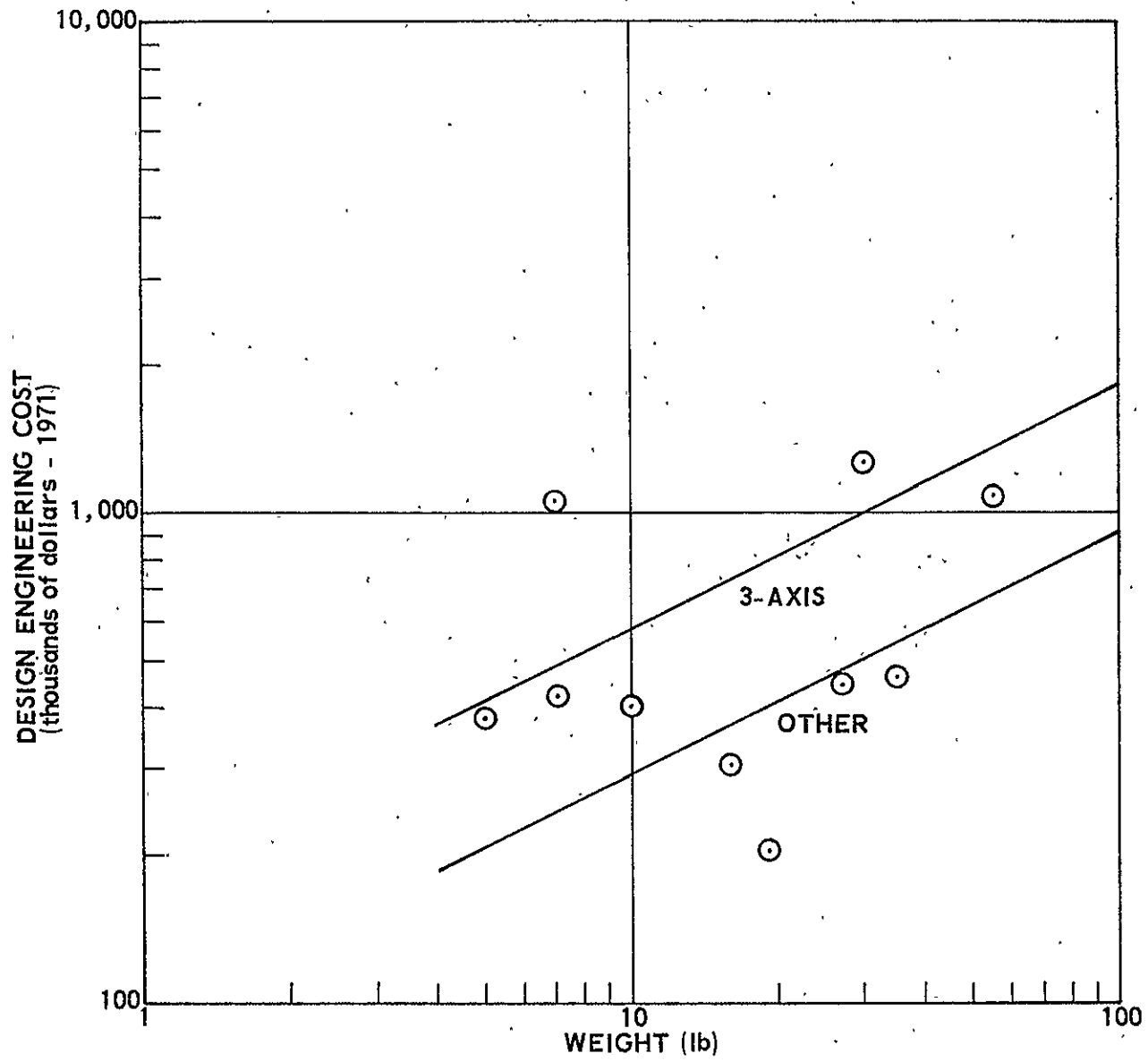


Figure 12-59. Thermal Control: Design Engineering Cost versus Weight

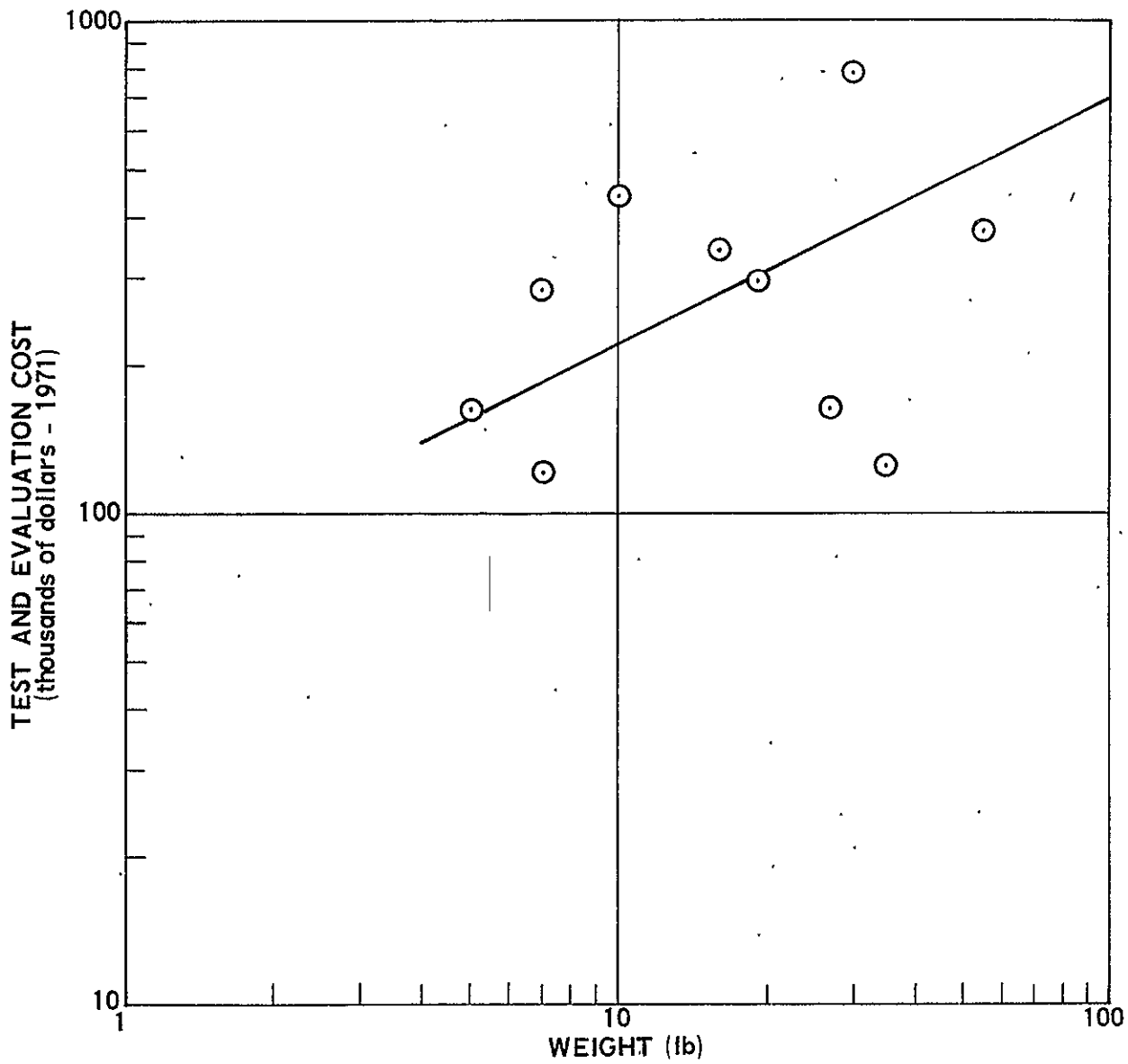


Figure 12-60. Thermal Control: Test and Evaluation Cost versus Weight

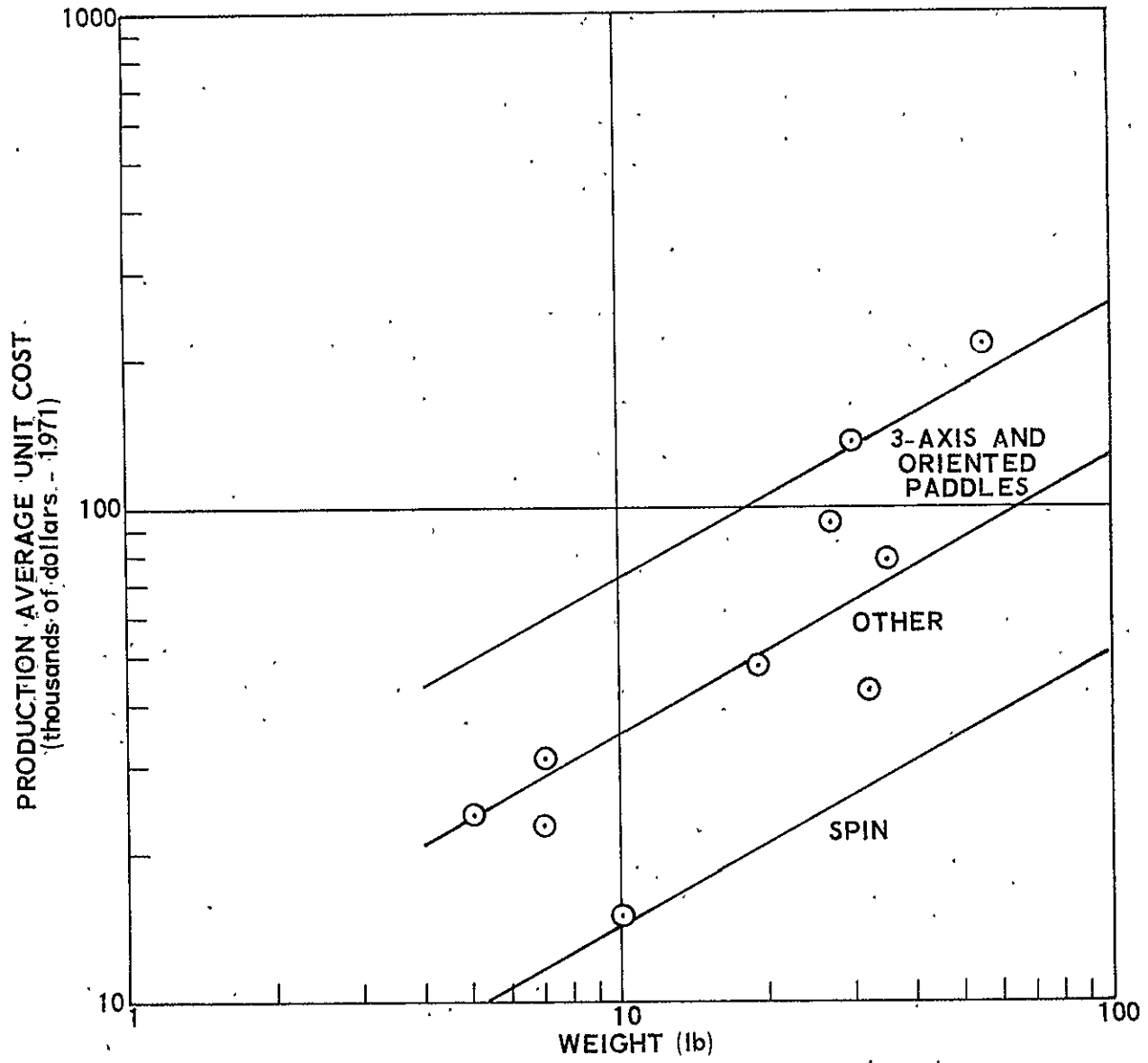


Figure 12-61. Thermal Control: Production Average Unit Cost versus Weight

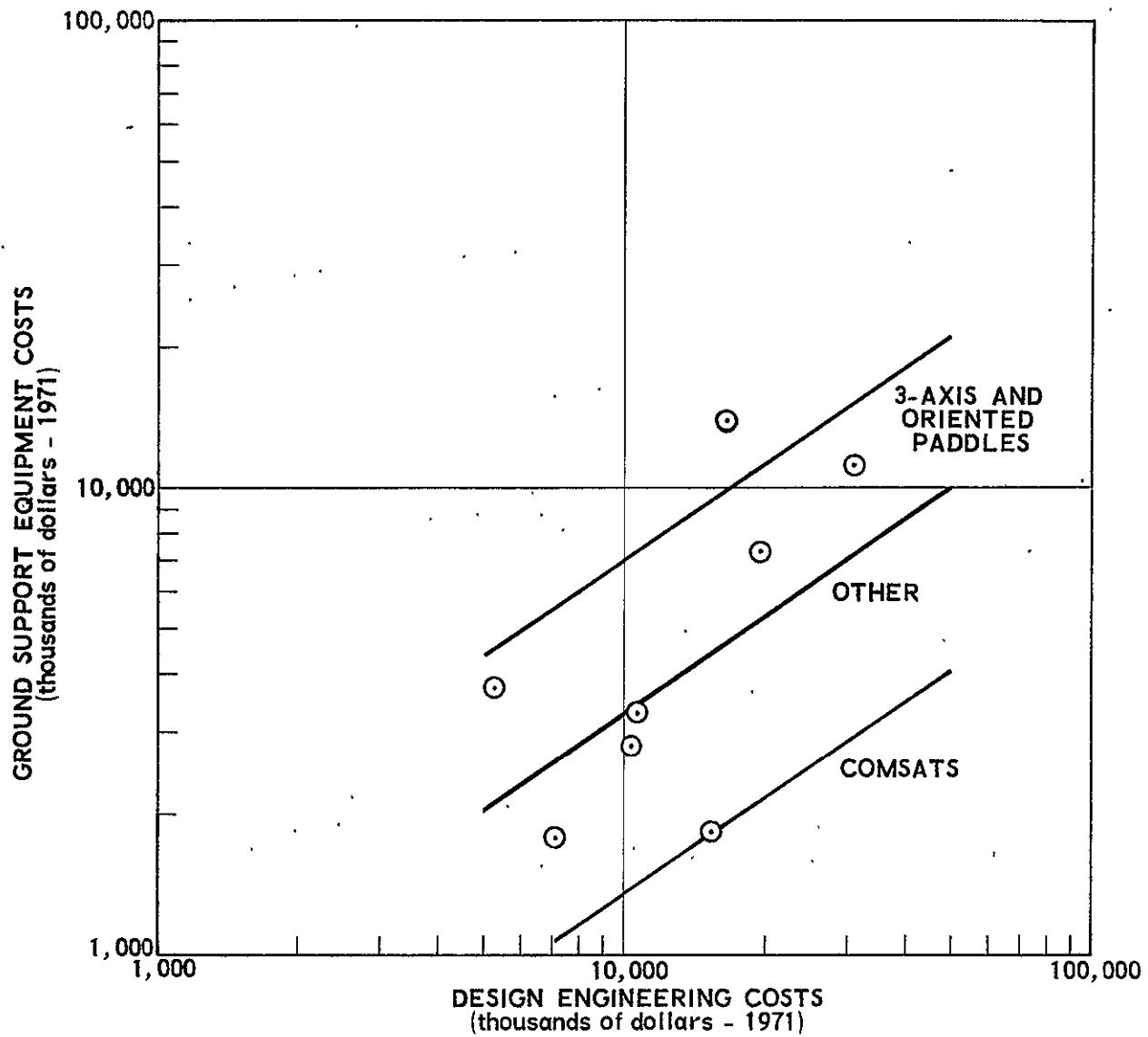


Figure 12-62. Ground Support Equipment Costs versus Satellite Design Engineering Costs

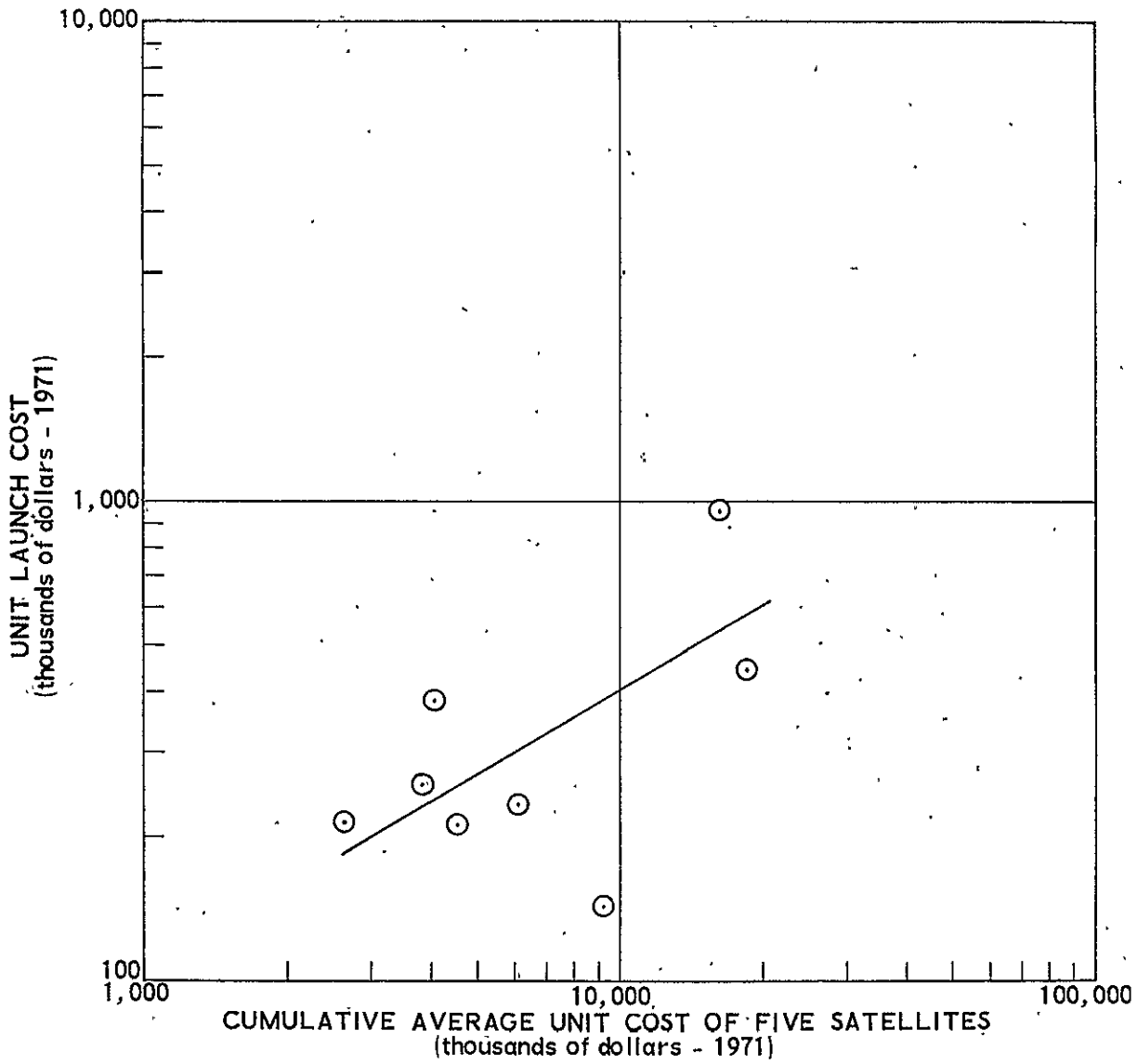


Figure 12-63. Unit Launch Cost versus Cumulative Average Unit Cost of Satellite

13. SCHEDULE

13.1 GENERAL

The method for determining nominal schedules is based upon the fact that component and subsystem direct DDT&E costs are measures of hardware complexities. Thus, it is feasible to devise a method of determining nominal schedules if direct DDT&E costs are known for the components of the various subsystems used in the system. Once these schedules are determined, total DDT&E costs may be determined by adding those terms dependent directly and primarily upon schedule duration (e.g., Project Management). Figure 13-1 diagrams the basic approach.

In order to provide credible results, the output data is limited to the following lead times:

- a. Component design and development (for major components)
- b. Subsystem development (for each subsystem)
- c. Component qualification
- d. Subsystem qualification
- e. System development, test, and flight readiness

Because the method is primarily empirical and has not been tested with adequate data, it must be considered preliminary.

13.2 SCHEDULE MODEL

13.2.1 Definitions

Activities not explicitly stated in either Figure 13-1 or the relationships given below are implicitly a part of the functions which are stated. The unstated activities include: (a) preparation and approval of test plans, specifications, schematics, layouts, and installation drawings; (b) fabrication of hardware, test equipment, checkout equipment, and mockups; (c) simulations; (d) special approvals (as for phases); and

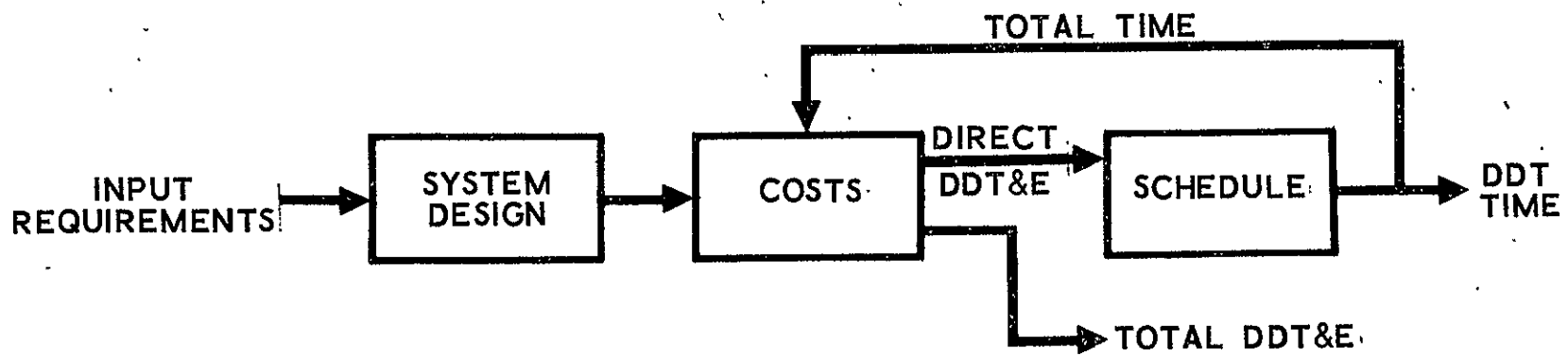


Figure 13-1. Nominal Schedule/Cost Computation Diagram

(e) analysis. Note that design is not separated from development; this is an example of where other activities are implicit, such as detail specification preparation and fabrication.

The term Direct DDT&E Cost as used in this model contains all DDT&E costs except those elements which are directly proportional to time such as Program Management. Total DDT&E cost is referred to as DDT&E Cost.

The term "base" value is used to refer to schedule values obtained from an actual component, subsystem, or satellite.

The following activities are explicitly used in the scheduling mode:

- a. Component Design and Development (τ_{cd}) - The lead time here includes all activities from the initiation of subsystem or component design (whichever comes first) through complete development of the component. This includes the preparation of the subsystem specification and layout, the detailed component specification, any approvals in the phase, and detailed component design.
- b. Component Qualification (τ_{cq}) - The definition includes items (if required) as test plan approval (which may be incorporated in the specification approval) and test report preparation and approval, as well as the actual tests and changes to qualify the component. This lead time includes redesign and shop time for normal changes to pass test.
- c. Subsystem Development (τ_{sd}) - This time incorporates all activities associated with the development of every subsystem from start of development through environmental tests which are in the nature of prequalification tests. The start depends upon availability of all components in a condition where they perform satisfactorily at least under laboratory conditions. Thus, time for usual changes is also a part of the nominal time.
- d. Subsystem Qualification (τ_{cq}) - This time is the equivalent of component qualification for every subsystem. It also ends at approval of the qualification test report.
- e. System Development, Test, and Flight Readiness (τ_f) - System development starts when spacecraft subsystems are ready for assembly into a system and ends when all physical and functional interfaces have been checked and approvals given to proceed into flight readiness test. It includes prequalification testing.

Flight readiness and checkout tests follow the development and are run with the complete satellite. This may entail a complete qualification test of the satellite, including vibration tests, acoustic tests, shock, space simulation, and check of all subsystems and mission equipment. Interface and performance with checkout equipment to be used at the launch site, EMI tests under launch site conditions, verification of compatibility with all AGE and AVE, and finally countdown and countup checks are typical of the functions of this activity, culminating in first flight.

13.2.2 Assumptions

- a. Component Direct DDT&E Cost is a direct measure of component complexity and is proportional to the direct charge engineering and technician manhours necessary to develop and qualify it.
- b. Lead time for development of each component and for each subsystem varies from a minimum value as a direct exponential of the ratio of men assigned to the men assigned in the base case. A starting value to be adjusted as data becomes available is that the ratio of men assigned to the men assigned in the base case varies directly as the square root of the lead time ratio. A starting value for the minimum value is that the minimum lead time is 70 percent of the base value.
- c. Lead time for qualification of each component and each subsystem varies linearly as a weak function of complexity from a minimum value. It therefore varies directly with DDT&E cost, as a consequence of assumption "a". A starting value for the minimum lead time is 90 percent of the base value.
- d. The state-of-art directly affects the component and subsystem development lead times. Each component has a state-of-art lead time factor, A, determined according to Table 13-1.

Table 13-1. State-of-Art Factors

Description	A
Off-the-shelf unit	.0.7
Similar to an existing unit	1.0
A new concept combining technology in use on other components	1.5
A new concept requiring reasonably predictable technology advance	2.0

A subsystem state-of-art factor, A_S , is determined independently of the values of A , using the same value judgment approach.

- e. Subsystem development lead time is a weak exponential function of the number of redundant components. A starting value for the exponent of the number of parallel components, whether active or standby, is $1/8$. A starting value for the minimum value is that the minimum lead time is 70 percent of the base value.
- f. Subsystem qualification lead time varies linearly as a weak function of complexity from a minimum value, having the same form as the component qualification function. A starting minimum value is 90 percent of the base value.
- g. System lead time is directly proportional to the pacing subsystem development lead time and its duration is a function of the degree of testing conducted with the entire spacecraft and also launch vehicle. Its value will range from about four for a system on which only essential development and flight readiness tests are conducted to a value of about five for tests which include complete qualification. System development effectively starts when all subsystem qualification tests and the system development tests are completed.

13.2.3 Schedule Equations

a. Component Development (τ_{cd})

$$\tau_{cd} = Ak_c C^a + c_o \quad (13-1)$$

in which:

- A = state-of-art factor for component
- c_o = constant starting value $\approx 0.7 (\tau_{cd})$ base
- k_c = constant for component type
- C = Direct DDT&E charges allocated by component. Effectively contains all direct charges (engineers, technicians, shop men) with burden, materials, overhead.
- a = constant, starting value = $2/3$

Equation 13-1 is derived using the assumption that men are assigned according to the relationship

$$\sqrt{\frac{\tau}{\tau_{base}}} = \frac{\text{Men}}{(\text{Men})_{base}} \quad (13-2)$$

and the fundamental

$$\tau = k'_c \frac{\text{Manhours}}{(\text{Men})} \quad (13-3)$$

$$(\text{manhours}) \sim C \quad (13-4)$$

from which the starting value for "a" emerges.

b. Component Qualification Test (τ_{cq})

$$\tau_{cq} = c_1 + c_2 C \quad (13-5)$$

in which c_1 and c_2 are constants, with a starting value of $c_1 \cong 0.9$ (τ_{cq}) base and c_2 dependent on each equipment type.

c. Subsystem Development (τ_{sd})

$$\tau_{sd} = k_s n^d A_s (C_s)^\alpha + c_3 \quad (13-6)$$

in which:

$$C_s = \sum_{i=1}^N C_i = \text{subsystem Direct DDT\&E Cost}$$

i = i th component of N components

d = constant, starting value = $1/8$

A_s = state-of-art factor for subsystem

n = Average redundancy for subsystem, active plus standby strings

c_3 = constant, starting value ≈ 0.7 (τ_{sd}) base

α = constant, starting value = $2/3$

The form of the equation follows from similarity with the component development relationship Equation 13-1.

d. Subsystem Qualification Test (τ_{sq})

$$\tau_{sq} = c_4 + c_5 n^d C_s \quad (13-7)$$

in which c_4 and c_5 are constants, with a starting value of $c_4 \cong 0.9$ (τ_{sq}) base and c_5 dependent on each equipment type.

e. System Tests (τ_f)

$$\tau_f = A_s k_f \tau_{sd} \quad (13-8)$$

in which:

k_f = constant which depends upon the extent of system testing required. The range is from 4 to 5.

13.2.4 Schedule Logic

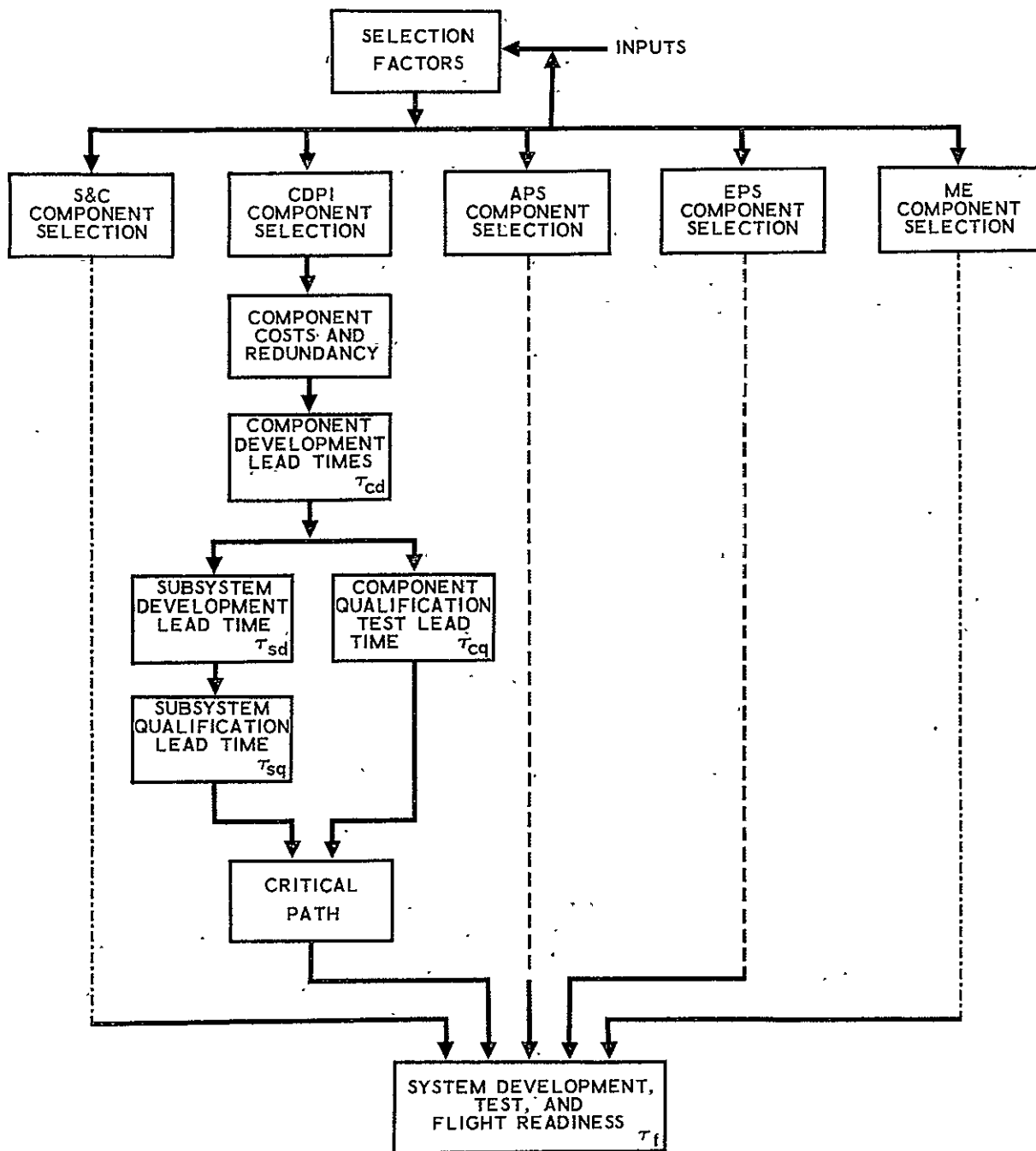
Figure 13-2 is the schedule activity flow diagram for the program. Detailed flow is shown for the CDPI subsystem; the flow for each subsystem follows the same logic. There are two main paths, the development path and the qualification path. The two converge at the flight readiness and checkout block. Where more than one input enters an activity block, it is intended that all preceding connected activities must be completed (i. e., for subsystem development and for system flight readiness and checkout).

Inputs to the scheduling activities enter for each subsystem at the component development lead times block. Principal inputs are selected component costs by component type and redundancy; thus information generated for developing costs is used as input for deriving schedules.

Other input data are semi-empirical or empirical coefficients, component state-of-art factors, and subsystem state-of-art factors.

Observe that payload structure and thermal control does not appear. The assumption here is that structure and thermal control do not constitute a critical path, so that no provision is made for their scheduling.

Subsystem qualification test starts upon completion of the subsystem development test. One or more components may not be 100 percent qualified at that point; this is not an unusual occurrence. On the other hand, a necessary condition for the start of flight readiness



NOTE: CDPI detailed flow is typical for all subsystems

Figure 13-2. Schedule Activity Flow Diagram

test is that all subsystems are qualified as well as developed. In actual fact, it is common to have to waive successful completion of some part of qualification test before first flight. This was not believed to be a desirable planning assumption.

The last block, System Development, Test, and Flight Readiness, brings the system to launch of first flight.

13.2.5 Representative Results

In order to demonstrate characteristic results, coefficients are derived from assumed starting values, and curves similar to those representing behavior of the computer outputs are presented. The example is for spacecraft A (Ref. 13-1), the subsystem (Mission Equipment), a Transponder; the component, a High Level Traveling Wave Tube (HLTWT). This component is typical of the components requiring a long qualification test lead time. The component was not actually the critical path item, but is treated as such in the example. In addition, average values for all components and subsystem lead times are given in parentheses. The assumed quantities are as follows:

a. Example component

τ_{cd}	=	8.7(9.5) months
τ_{cq}	=	4.3(5.0) months
A	=	1.0
c_o	=	6.0(6.5) months
c_l	=	3.0(3.5) months

b. Example subsystem

$$\begin{aligned}\tau_{sd} &= 2.7 (2.8) \text{ months} \\ c_3 &= 1.9 (2.0) \text{ months} \\ \tau_{sq} &= 3.0 (3.0) \text{ months} \\ c_4 &= 2.4 (2.5) \text{ months} \\ n &= 2 \\ A_s &= 1.0\end{aligned}$$

c. Example system

$$\tau_f = 14.5 \text{ months}$$

Other starting values: $a = 2/3$, $\alpha = 2/3$, $d = 1/8$

Using the above numbers, the data presented in Table 13-2 and Figures 13-3 and 13-4 were calculated. Figures 13-3 and 13-4 show the variation in lead times with direct DDT&E cost ratios from the base value. Table 13-2 compares the component example and the average component cases.

Figure 13-5 shows a schedule for the nominal case and a variation where (1) the HLTWT component (considered here as the critical path component for illustration) costs 40 percent more than the base case component but has the same state-of-art factor, and (2) the subsystem state-of-art factor, $A_s = 1.2$, because a new component requires more subsystem development.

13.2.6 Discussion

Schedules are usually dependent upon critical paths, as has long been recognized and used in such management tools as PERT. The method defined above recognizes this principle and adapts it in the summation of critical paths.

Consider the bar chart examples of Figure 13-5. The critical path subsystem development is completed before the initiation of subsystem

Table 13-2. Comparison of Nominal Schedules from Typical Components and from Subsystem Summary (Transponder)

Lead Time	Months	
	HLTWT	Subsystem
Component Development	8.7	9.5
Component Qualification	(4.3)	(5.0)
Subsystem Development	2.7	2.8
Subsystem Qualification	3.0	3.0
Subsystem Tests	14.5	14.5
DDT&E Program	28.9	29.8

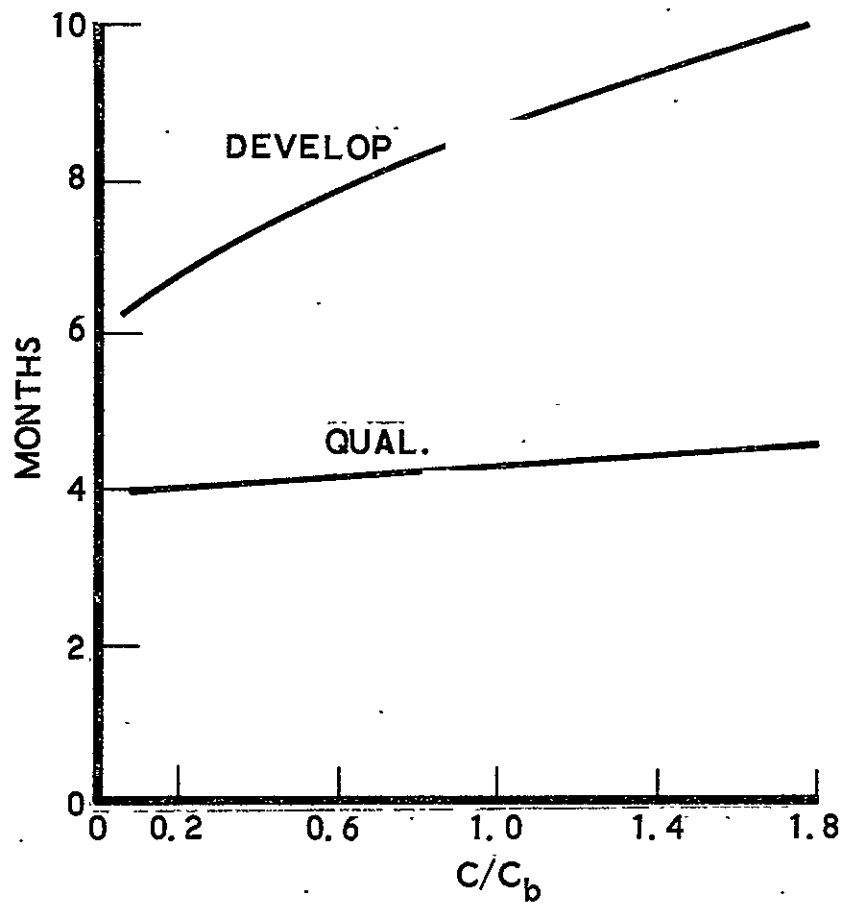


Figure 13-3. High-Level Traveling Wave Tube Development and Qualification Lead Time

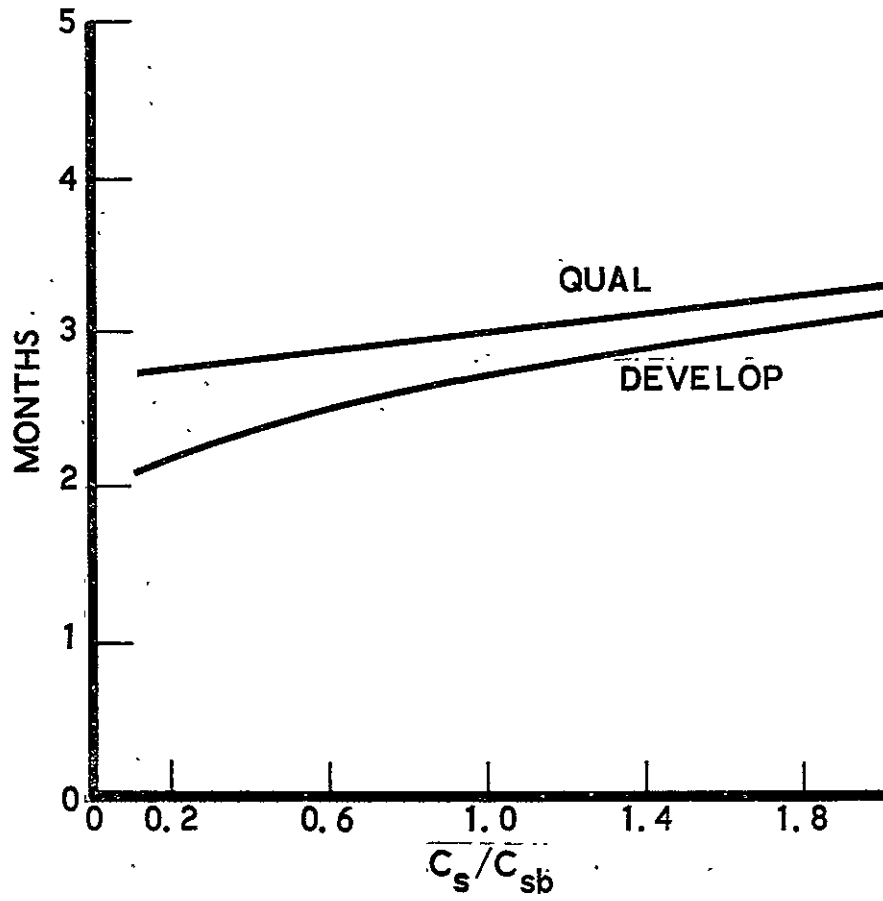


Figure 13-4. Transponder Development and Qualification Lead Time

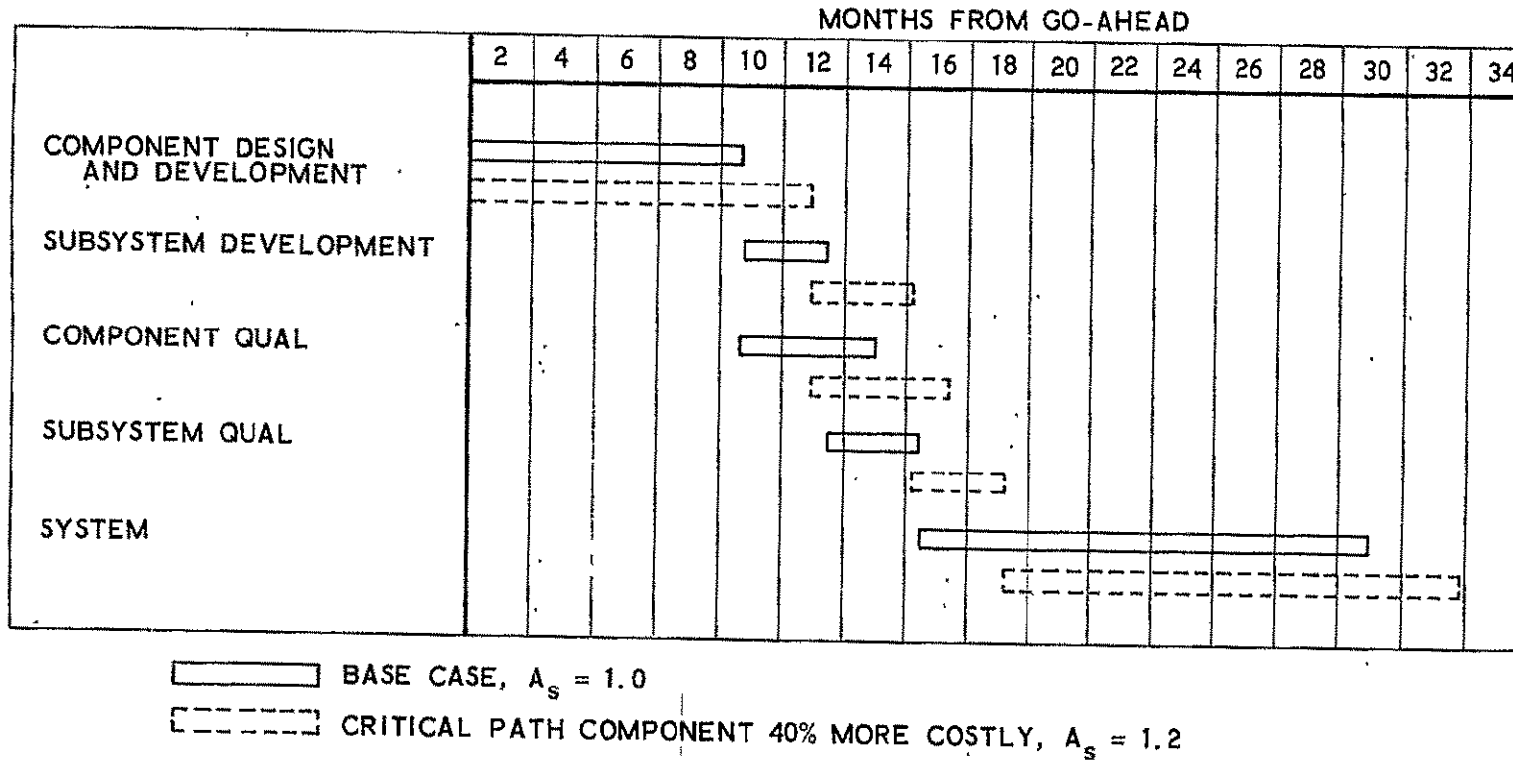


Figure 13-5. Example Schedule Output

qualification. The system development is completed later as one would expect. The system test time is large compared with other lead times; attempts to separate development time from flight readiness or system qualification time have been unsuccessful because the differing approaches used for different systems make correlation of results meaningless. For example, spacecraft qualification may be done separately or as part of the satellite qualification. Therefore, the end of development is a grey area which depends upon the program approach and it is believed a finer breakdown should not be sought.

Next, on Figure 13-5, observe the usefulness of the model in response to a change in input data. The solid lines are for the base value case, and the dashed lines for the variation. The critical path component of the variation has the same function as the base case, but is more complex, so that DDT&E cost is increased by 40 percent. Some technology advance is introduced ($A_s = 1.2$). The result is that the system should be ready for flight about three months later ($\approx 10\%$ increase in total lead time) than the less expensive base value system.

Now refer to Figures 13-3 and 13-4, which present example plots of component and subsystem lead times as functions of DDT&E costs. The rationale behind the shapes of the development curves bears repeating, although the actual shape may eventually be developed empirically. In Paragraph 13.2.2 it was stated that men would be added to the base number as the square root of the lead time as the effort required to develop the item increases. This simply recognizes that development lead time of a component or subsystem type cannot be maintained constant as complexity is increased by merely adding manpower. For example, assume the base value of average men charging directly to component design and development is four (4), and the complexity of the component is increased to 1.4 times the DDT&E cost. This will result in a corresponding increase in development lead time unless manpower is increased. The average number of men assigned to expedite the development would be 4.5 (rather than 4.0), which

would result in the development lead times being 25 percent longer, rather than 40 percent longer. This results from calculations following Equations 13-2 through 13-4:

$$\begin{aligned}(\text{cost ratio})^{2/3} &= (1.4)^{2/3} && 1.25 \\ \text{men ratio} &= \sqrt{1.25} && = 1.12 \\ 1.12 \times 1.25 &= && 1.4\end{aligned}$$

This rationale results in the trend shown in Figure 13-3. During the test phase for component development, it is believed to be representative. For example, if an electro-mechanical component is the responsibility of a development engineer, test technicians, part-time supervision, model shop men, etc., amounting to an average of four men, the development may be expedited by adding some engineering assistance and working some overtime, the manpower increase being about one man. Further increases would probably be quite inefficient for, even with several test models, only a limited number of changes resulting from test can be handled simultaneously. Thus, the increment seems compatible with a nominal schedule.

As data is accumulated from various programs, actual values of the coefficients for development can be developed. Meanwhile, a method has been demonstrated which can be used with a very limited amount of data. The same objective has been applied to the other terms.

Consider the qualification test line for the example component of Figure 13-3. Note that it is straight and has a minimum value (only 10 percent less than the nominal, or base value). The minimum value is a function of the number of tests, or environments, to which the component will be subjected. In the nominal case, it is common practice to fabricate a fixed number of qualification test units for each component (e. g., six), so that simultaneous testing of different environmental sequences may be conducted, and so that changes may be made when a failure occurs without

holding up the entire test program. The number of environments is generally fixed, as well as the duration of a successful test sequence. Typically, a component fails to pass one or more tests of the qualification test sequences, requiring some changes to pass them. This is a nominal occurrence, but one which normally requires nominal redesign of some readily changed parts to achieve success. This is particularly true if the development test program has adequately tested behavior under environmental extremes as has been assumed here. The minimum time encompasses these effects.

The second part of the component qualification test equation is dependent upon component complexity. This is to account for items whose development times depend upon complexity, and therefore, the DDT&E cost of the component. One effect is that more complex components having the same function contain either more parts, more precise parts, require tighter performance tolerances, a larger capacity, or a higher power, for example. These complexities tend to make it more difficult to stay within performance limits over the range and duration of environments. Another related effect is inherent in the meaning of component as applied to the model. According to the meaning used, a "component" may be comprised of more than one box or subassembly, each of which requires development. For example, a propulsion subsystem "component" would be a "thruster." In one satellite, there may be two sizes of hydrazine thruster; one low-thrust unit used for attitude control and another somewhat higher thrust unit used only for station-keeping. The attitude control unit may be more costly to develop because of the much larger number of cycles which must be accumulated to meet the design life. The lower thrust unit would take longer to develop and would therefore have a higher DDT&E cost. It would also take somewhat longer to qualify both because of the longer test time needed and because it is less likely to meet requirements on the first attempt. Similarly, a computer memory required to store 10^5 bits will take less time to qualify than one required to store 10^6 bits, and therefore, the lead time for the more complex unit will be longer.

A linear variation with cost is assumed for qualification unless data show that an exponential is needed.

Next, note the development curve for the subsystem in Figure 13-4. The wide variation in subsystem complexity for any function is well known, so that it is to be expected that the lead time for development will be a function of subsystem DDT&E cost. By the same reasoning discussed for component development, however, time cannot be bought back entirely by adding manpower. Using the same reasoning as before, the variation is expected to be in the neighborhood of the $2/3$ power of cost.

The computer program written for this schedule model will utilize computations for costs as input data. This is nearly the inverse of normal procedure for bottom-up costing, where schedules are normally made up first (or simultaneously), and manpower second to avoid the problem of failure to account for standby time (e.g., time spent by development engineers awaiting changes to proceed through the shop). However, the procedure has merit in that:

- a. It uses the cost data base of a cost program.
- b. It requires schedule data which should be available from previous program files.
- c. Component scheduling data need be accurate only where lead times form critical paths.
- d. Only the more significant scheduling mileposts are computed to avoid unwarranted detail.

Note, on Figure 13-2, that the mission equipment path is shown even though it is not a part of the current effort. The reasons for this are:

- a. Mission equipment is often pacing
- b. The program techniques should be similar for similar types of mission equipment (i.e., it will be necessary to gather data on types of mission equipment, because there will be types. just as there are four major subsystems).

13.3 SCHEDULE DATA

For the purpose of this study, all spacecraft coefficients have been calculated with the coefficients of cost dependent terms normalized to the base value component and subsystem costs, C_b and C_{sb} , respectively. Table 13-3 contains component data (Ref. 13-1), and Table 13-4 contains subsystem data (Ref. 13-2). (The unit of cost, C_b and C_{sb} , is millions of dollars.) The values for the state-of-art factors would make use of the best opinions of the Spacecraft A program engineers to establish satisfactory judgmental values.

Insertion of the Table 13-3 and 13-4 inputs into a computer program with the A and A_s both set equal to unity will mean that all new programs considering the use of a component of the same type or subsystem of the same type will give correct relative schedules, even though the "nominal" base value actually had a slightly longer lead time than a true nominal. This is a satisfactory assumption to test a computer program.

The major assumptions used in compiling data from PERT time charts for Spacecraft A follow. (This detail was found to be necessary because of inconsistency of terminology on the various PERT charts.)

- a. Component development time is measured from component go-ahead to completion of engineering model (or qualification model where so defined) pre-environmental functional tests, or the last engineering model test, whichever is later. Breadboard tests, in general, are not considered because development is defined here as incomplete until the test model is essentially the same design as the qualification test model. As with all rules, some judgment was necessary in exceptional cases; for example, where the pre-environmental test data was not recorded and the breadboard test date was available (the Command Receivers), an estimate was made of the probable development/qualification ratio, which was applied against the known sum of development and qualification time.
- b. Component qualification is measured from completion of component development to completion of component engineering model (or qualification model if so defined) post-environmental test (or qualification if so defined.)

Table 13-3. Component Schedule Coefficient and Lead Time Summary

SS/COMPONENT	c_o	c_1	$Ak_c C_b^{2/3}$	$c_2 C_b$	τ_{cd}	τ_{cq}	τ_{sd}	τ_{sq}	τ_f	REMARKS
CONTROLS										
Despin Mech. Assy	7.8 ⁽³⁾	1.2	3.3	.13	11.1	1.3	7.0		11.9	τ_f here = $\tau_{sq} + \tau_f$
Biax Drive & Gimbal Dr. Elect	7.5	3.9	3.1	.43	10.7	4.3	8.5		11.5	
Despin Electronics Assy	7.3	3.8	3.1	.42	10.4	4.2	7.7		11.9	
Control Timing Assy	8.3	2.6	3.6	.29	11.9	2.9	7.2		10.8	
Valve Driver Assy	4.7	6.3	2.0	.70	6.7	7.0	11.3		11.9	
Earth Sensor Assy (2)	5.4	4.3	2.9	.48	7.8	4.8	8.1		↓	
Sun Sensor Assy	4.6	3.4	2.9	.38	6.6	3.8	8.4		↓	
Propulsion S.S. (2)	4.0	2.2	2.9	.25	5.7	2.5	8.2		↓	
ELECTRICAL										
Solar Array (2)	8.7	1.4	4.4	.16	12.4	1.6		7.0	9.0	τ_{sq} here = $\tau_{sd} + \tau_{sq}$
Shunt Element Assy	9.5	2.5	4.1	.25	13.6	2.5		6.1	8.5	
Power Control Unit(2)	8.1	2.3	4.0	.25	11.6	2.5		6.3	9.1	
Battery (2)	6.9	2.1	3.5	.23	9.8	2.3		8.0	↓	
Converter & Electrical Integ(2)	5.3	2.5	2.8	.28	7.5	2.8		10.1	↓	
Reset Gen. & Switching Logic(2)	3.5	1.9	3.7	.21	5.0	2.1		7.2	↓	
THERMAL										
	NA				0	0	14.4 ⁽¹⁾	2.9	12.9	See SS Summary
CDPI (TT&C)										
Dual Baseband-Signal Cond.	7.3	1.8	3.1	.20	10.4	2.0	2.0		18.3	τ_f here = $\tau_f + \tau_{sq}$
Encoder-Multiplier	6.2	1.8	2.8	.20	8.9	2.0	3.0		18.3	
Command Receivers	4.2	7.1	1.8	.79	6.0	7.9	5.7		18.3	
Transponder-Converter	7.3	1.7	3.2	.19	10.5	1.9	1.2		18.1	
Telemetry Transmitter	7.3	1.7	3.2	.19	10.5	1.9	1.2		18.1	
Diplexer	6.0	1.5	2.6	.17	8.6	1.7	3.0		18.3	
TRANSPONDER										
E. C. Transmit Ant.	6.6	.45	2.8	.05	9.5	0.5	2.8	3.0	14.5	Mission Equip't
E. C. Receive Antenna	6.6	.63	2.8	.07	9.4	0.7	2.9	3.2	↓	
N. C. Antenna	6.6	3.6	2.8	.40	9.4	4.0	5.3	0.6	↓	
Transmit Filter, Coupler, Sw(2)	3.9	2.9	2.9	.32	5.6	3.2	2.3	3.1	↓	
Receive Filter, Amplifier, Sw(2)	5.6	4.5	2.6	.50	8.0	5.0	3.5	3.0	↓	
Diplexer / Equalizer	6.4	0	2.8	0	9.2	0	3.2		↓	
Low Level TWT(2)	4.3	2.6	3.0	.29	6.1	2.9	2.2		↓	
High Level TWT(2)	5.3	3.9	2.6	.43	7.6	4.3	2.7		↓	
Mixer Channel Combiner(2)	5.9	3.1	3.3	.35	8.4	3.5	2.4		↓	
Frequency Generator(2)	6.4	4.5	2.9	.50	9.2	5.0	0.7		↓	

(1) Includes τ_{cd} , τ_{cq}

(2) Started later than contract go-ahead.

(3) All data presented in units of months.

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Table 13-4. Subsystem Schedule Coefficient Summary

S/C SUBSYSTEM	c_3	c_4	$A_s k_s C_{sb}^{2/3}$	$c_5 C_{sb}$	$A_s k_f$	c_o	c_l	$A_k C_b^{2/3}$	$c_2 C_b$	τ_{sd}	τ_{sq}	τ_f
S/C A												
Controls	5.8 ⁽¹⁾	NA	2.2	NA	1.40	SEE COMPONENT DATA				8.3	-	11.6
Electrical	NA	6.8	NA	0.69	1.20	"				-	7.5	9.0
Thermal	10.1 ⁽²⁾	2.6	3.9 ⁽²⁾	0.27	0.90	"				14.4	2.9	12.9
CDPI	1.8	NA	0.70	NA	7.0	"				2.6	-	18.2
Transponder	2.0	2.7	0.76	0.27	5.2	"				2.8	3.0	14.5
S/C BI	1.0	5.2	0.41	0.69	5.2	11.2	?	4.8	?	1.5	7.5	5.9
BII	1.0	5.4	0.41	0.71	6.5	11.4	?	4.9	?	1.5	7.7	9.8

(1) All data presented in units of months.

(2) These terms include component development

- c. Subsystem development is measured from end of component development to final engineering model subsystem test. In some cases the complete subsystem was first assembled on the spacecraft, in which case, the subsystem test event preceding start of system qualification test was used as completion of subsystem development (see S&C and CDPI).
- d. Subsystem qualification test is in some instances on the spacecraft, in some on a separate assembly of components. In one instance, Electrical, the development completion could not be determined from recorded events, so that only the subsystem qualification completion was recorded; here the subsystem qualification lead time is defined as the time from component development completion to the end of subsystem qualification.
- e. The values of system test lead time were determined from a memorandum-recorded end date minus the earlier PERT chart dates for completion of subsystem qualification tests.

As the foregoing is intended to suggest, judgment was necessary in selecting events consistent with the definitions. As discussed, the separate pacing lead times which add up to the total DDT&E time (i. e., for pacing items $\tau_{cd} + \tau_{sd} + \tau_{sq} + \tau_f = \tau_t$) may be unprecise because of the need for judgment in selecting events from a variety of terminology; however, the program time, τ_t , should be quite representative.

As noted in Table 13-3, second footnote, some of the components were started after contract go-ahead, while many were started from the program outset. This choice was made by the developer at the beginning of the program, and was therefore a prediction tempered by convenience. Thus, the component development lead times are inexact. The items started at program go-ahead are not necessarily pacing; an example is the thermal subsystem. Analysis of the thermal problem and design of the thermal control starts with a best estimate at the beginning, and thereafter the estimates are revised as electrical loads, structure, and mission operations are refined; it is therefore generally not pacing, even though it is started immediately. In the case of subcontracted components contractor may elect to start work on specifications and requests for proposal at contract go-ahead, even though the items may not be pacing.

No attempt was made to identify these items, for it would take a detailed knowledge of the engineering judgments made to adjust the lead times. Some of the items started at contract go-ahead may have development lead times somewhat above nominal. The error is therefore generally on the conservative side.

Items started some time after the program go-ahead will not satisfy $\tau_t = \tau_{cd} + \tau_{sd} + \tau_{sq} + \tau_f$, because of the delta between program start and item start. Note that this delta was not applied to data presented in Table 13-2, where τ_{cd} was measured from program go-ahead for every component subsystem.

These facts simply point out the desirability of gathering data from a number of programs.

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14. DATA BASE DESCRIPTION

Data on selected payload equipments (components) have been collected for the express purpose of exercising the Systems Cost/Performance Model. The reader should take careful note that, although most of the data is accurate, approximations based on engineering judgment and experience are used wherever actual data were unavailable. The approximations are justified by the original purpose of the study which was to develop a cost/performance model. Assuming that the model is successfully completed and is accepted for use by a body of users, the data base must be expanded and approximations replaced by actual data. The following paragraphs are devoted to an explanation of how the data base is organized and how to interpret the information.

14.1 GENERAL

The model selects equipment for a specific design in one of three ways:

- a. Most equipment is selected from the data base on the basis of technical performance.
- b. Some equipment which cannot be differentiated on the basis of technical performance is called up from the data base on a first-called basis in order to provide a complete design description.*
- c. Certain equipment is not amenable to cataloging in the data base. This equipment is identified and specific parameters are determined. Examples include the wiring harness and the Thermal Control Subsystem components.

The equipments are organized according to the following subsystems which use the specific components:

* This category should be eliminated in improved versions of the Systems Cost/Performance Model.

- a. Stabilization and Control
- b. Auxiliary Propulsion
- c. Data Processing
- d. Communication
- e. Electrical Power

Thermal Control and Structures do not select equipment from the data base, but do, instead, estimate the parameters describing their respective components.

The data base contains information on the following types of equipment for each subsystem:

- a. Stabilization and Control Subsystem
 - 1. Despin Mechanical Assemblies
 - 2. Despin Electronic Assemblies
 - 3. Valve Driver Assemblies
 - 4. Sun Sensor Assemblies
 - 5. Nutation Dampers
 - 6. Gimbal Electronic Assemblies
 - 7. Bi-axial (Gimbal) Drive Assemblies
 - 8. Control Electronics Assemblies
 - 9. Earth Sensor Assemblies
 - 10. Rate Gyro Assemblies
 - 11. Reaction Wheel Assemblies
 - 12. Rate Integrating Gyro Assemblies
 - 13. Control Moment Gyros
 - 14. Star Sensor Assemblies
- b. Auxiliary Propulsion Subsystem
 - 1. Thrusters
 - 2. Isolation Valves
 - 3. Filters
 - 4. Pressure Regulators

5. Tanks
 6. Fill and Drain Valves
 7. Relief Valves
- c. Data Processing Subsystem
1. General Purpose Processors
 2. Digital Telemetry Units
 3. Tape Recorders
 4. Command Decoding and Distribution Units
- d. Communication Subsystem
1. Baseband Assembly Units
 2. Antennas
 3. Transmitters
 4. Receivers
 5. Command Signal Conditioners
 6. Diplexers
- e. Electrical Power Subsystem
1. Shunt Regulators
 2. Batteries
 3. Battery Chargers
 4. Discharge Regulators
 5. Series Load Regulators
 6. Power Control Equipment
 7. Solar Arrays
 8. Power Converters

An example of an equipment description in the data base is provided in Table 14-1. The data sheet for each component states which subsystem utilizes the component, which configurations require the component, which equipment type the component is categorized as, and the data base identifier or code number assigned to the component.

Table 14-1. Data Base Example

Subsystem: Auxiliary Propulsion (0808)
 Configuration: Monopropellant
 Equipment Type: Thruster (TRW 404620)
 Performance

Technical Characteristics

1.	Thrust Level (N)	18
2.	Pulse Life (cycles)	93,000
3.	Inlet Pressure (N/m ²)	4.14×10^6
4.	Total Impulse (N-sec)	6.49×10^4
5.	ISP (sec)	230
6.		
7.		
8.		
9.		
10.		

Power

Average Power (watts):	(near zero)
Maximum Power (watts):	5.5
Minimum Power (watts):	0.0
Nominal Voltage (volts):	28.0
Maximum Voltage (volts):	32.6
Minimum Voltage (volts):	26.0
Converter/Inverter Requirement (flag):	N.A.

Weight (Kg): 0.3

Volume (cc): 1700

Vibration

Random (g, rms):	19.5
Non-random (g):	10.5

Temperature

Maximum (deg K):	322
Minimum (deg K):	278

Pressure (N/m²): (Unknown)

Table 14-1. Data Base Example (Continued)

Performance (continued)

CDPI

Power Switching Commands (No.):	0
Time Tagged Commands (No.):	0
Other Commands (No.):	0
High Rate Telemetry	
Number of Analog Points (No.):	0
Number of Digital Points (No.):	0
Sample Rate (sec ⁻¹):	0
Word Length (bits):	0
Low Rate Telemetry	
Number of Analog Points (No.):	2
Number of Digital Points (No.):	0
Sample Rate (sec ⁻¹):	1
Word Length (bits):	8

Safety

Failure Model (flag):	5
Failure Parameters	
Failure Rate or Mean (x 10 ^{±9} hr):	1700
Standard Deviation (x 10 ⁺⁹ hr):	N. A.
Dormancy Factor (N. D.):	0.1
Total Number of Redundant Elements (No.):	12

Cost

Design Engineering (\$1000):	127
Test and Evaluation (\$1000):	150
Unit Production (\$1000):	9
Reference Quantity (No.):	4
Factor (N. D.):	1

Schedule

Development Lead Time Constant (months):	3.0
Development Lead Time Variable (months):	1.0
Qualification Lead Time Constant (months):	1.5
Qualification Lead Time Variable (months):	0.1
State-of-Art Factor (N. D.):	1.0

The data describing the component consist of the following four types:

- a. Performance
- b. Safety
- c. Cost
- d. Schedule

14.2 PERFORMANCE DATA

The performance data are separated into eight categories:

- a. Technical Characteristics
- b. Power
- c. Weight
- d. Volume
- e. Vibration
- f. Temperature
- g. Pressure
- h. Communication, Data Processing, and Instrumentation (CDPI)

The technical characteristics are peculiar to each equipment type. Generally speaking, the technical characteristics provide the data required to select or differentiate among the components and additional data for the component which, if selected, provide information for design of the remainder of the subsystem. The technical characteristics required by the model for each type of equipment are as follows:

- a. Despin Mechanical Assembly
 - 1. Bearing and motor friction (mrad, 3σ)
 - 2. Bearing runout (mrad, 3σ)
- b. Despin Electronic Assembly. (None, this component is selected as a supplement to the Despin Mechanical Assembly.)
- c. Valve Drive Assembly
 - 1. Number of valves

- d. Sun Sensor Assembly. (None)
- e. Nutation Damper. (None)
- f. Gimbal Electronic Assembly
 - 1. Resolver accuracy (mrad, 3σ)
- g. Bi-axial (Gimbal) Drive Assembly
 - 1. Drive quantization (mrad, 3σ)
 - 2. Gimbal drive error (mrad, 3σ)
 - 3. Biax droop error (mrad, 3σ)
- h. Control Electronic Assemblies (Spinning Vehicle)
 - 1. Programmer sine wave (mrad, 3σ)
 - 2. Drive quantization and delay (mrad, 3σ)
 - 3. Measurement compensation (mrad, 3σ)
 - 4. Pipper drift (mrad, 3σ)
 - 5. Quantization noise (mrad, 3σ)
 - 6. Controller error (mrad, 3σ)
- i. Control Electronic Assemblies (Three-Axis Controlled Vehicles)
 - 1. Pitch horizon scanner gain (sec^{-1})
 - 2. Roll horizon scanner gain to roll axis (sec^{-1})
 - 3. Roll horizon scanner gain to yaw axis (sec^{-1})
 - 4. Pitch feedback gain (sec^{-1})
 - 5. Roll feedback gain (sec^{-1})
 - 6. Roll to yaw coupling gain (sec^{-1})
- j. Earth Sensor Assembly
 - 1. Sensor noise (mrad, 3σ)
 - 2. Radiance irregularity (mrad, 3σ)
 - 3. Quantization error (mrad, 3σ)
 - 4. Sun interference (mrad, 3σ)
 - 5. Moon interference (mrad, 3σ)
 - 6. Threshold aging (mrad, 3σ)
 - 7. Null or bias error (mrad, 3σ)
 - 8. Maximum output frequency (rad/sec)
- k. Rate Gyro Assembly. (None)

1. Reaction Wheel Assembly
 1. Nominal momentum (M-kg-sec)
 2. Maximum momentum (M-kg-sec)
 3. Minimum momentum (M-kg-sec)
 4. Nominal speed (rpm)
 5. Maximum speed (rpm)
 6. Minimum speed (rpm)
- m. Rate Integrating Gyro Assembly
 1. G-insensitive gyro drive (mrad, 3σ)
 2. Total misalignment relative to vehicle (mrad, 3σ)
 3. Gyro scale factor error (N. D.)
- n. Single Gimballed Control Moment Gyro
 1. CMG momentum (M-kg-sec)
 2. Peak gimbal rate (rad/sec)
 3. Peak torquer torque (N-m)
- o. Star Sensor Assembly
 1. Type
 2. Sensor accuracy (mrad, 3σ)
 3. Mapper field of view (mrad²)
 4. Mapper sensitivity (visual magnitude)
- p. Thruster
 1. Thrust level (N)
 2. Pulse life (cycles)
 3. Inlet pressure (N/m²)
 4. Total impulse (N-sec)*
 5. ISP (sec)*
 6. Mixture ratio (N. D.)**
- q. Isolation Valve
 1. Maximum pressure (N/m²)
 2. Flow area (cm²)

* Applicable to monopropellant and bipropellant thrusters.

** Applicable to bipropellant thrusters only.

- r. Filter
 - 1. Maximum pressure (N/m^2)
 - 2. Flow resistance $[\text{N}/(\text{kg}\cdot\text{m})^2]$
- s. Pressure Regulator
 - 1. Maximum pressure (N/m^2)
 - 2. Flow area (cm^2)
 - 3. Minimum set point (N/m^2)
 - 4. Maximum set point (N/m^2)
- t. Tank
 - 1. Volume (cm^3)
 - 2. Maximum pressure (N/m^2)
- u. Fill and Drain Valve
 - 1. Maximum pressure (N/m^2)
- v. Relief Valve
 - 1. Minimum set point (N/m^2)
 - 2. Maximum set point (N/m^2)
 - 3. Maximum operating pressure (N/m^2)
- w. General Purpose Processor
 - 1. Instruction rate (kips)
 - 2. Word length (bits)
- x. Digital Telemetry Unit
 - 1. Bit rate (kbps)
 - 2. Word length (bits)
 - 3. Number of mainframe words
 - 4. Number of subframes
 - 5. Number of words per subframe
 - 6. Digital multiplexer (yes/no)
- y. Command Decoding and Distribution Units. (None)

- z. Baseband Assembly Unit
 - 1. Compatibility
 - 2. First data rate stream (kbps)*
 - 3. Second data rate stream (kbps)
 - 4. First subcarrier frequency (MHz)*
 - 5. Second subcarrier frequency (MHz)
 - 6. Transmitter requirement (T __)
- a. Antenna
 - 1. Frequency, high band max (MHz)
 - 2. Frequency, high band min (MHz)
 - 3. Frequency, low band max (MHz)
 - 4. Frequency, low band min (MHz)
 - 5. Type and equipment number
 - 6. On-axis gain (dB)
- b. Transmitter
 - 1. Special requirement code (T __)
 - 2. Compatibility
 - 3. Maximum frequency (MHz)
 - 4. Minimum frequency (MHz)
 - 5. Power output (watts)
 - 6. Unified or nonunified**
 - 7. First subcarrier frequency (MHz)
 - 8. Second subcarrier frequency (MHz)
 - 9. Input data rate (Mbps)
 - 10. Modulation type

* The first rate or frequency is the higher of the two if two are provided.

** Nonunified requires 7. and 8. to be blank.
 Unified requires 9. to be blank.

- c. Receiver
 - 1. Compatibility, range and range rate
 - 2. Maximum frequency (MHz)
 - 3. Minimum frequency (MHz)
 - 4. Modulation type
 - 5. Maximum command rate (baud or bps)
 - 6. Command output type
 - 7. F_1 (kHz)
 - 8. F_2 (kHz)
 - 9. F_3 (kHz)
 - 10. Signal conditioner requirement (SC __)
- d. Command Signal Conditioner
 - 1. Compatibility
 - 2. Special requirement code (SC __)
 - 3. Command input
 - 4. F_1 (kHz)
 - 5. F_2 (kHz)
 - 6. F_3 (kHz)
 - 7. Maximum command rate (baud)
- e. Diplexer
 - 1. Compatibility
 - 2. Maximum receive frequency (MHz)
 - 3. Minimum receive frequency (MHz)
 - 4. Maximum transmit frequency (MHz)
 - 5. Minimum transmit frequency (MHz)
 - 6. Maximum allowable transmit power (watts)
- f. Shunt Regulator
 - 1. Maximum power capacity (watts)
- g. Battery Cell
 - 1. Capacity (amp-hr)
 - 2. Watt/hour charge efficiency (N. D.)

- h. Battery Charger
 - 1. Current rating (amps)
 - 2. Efficiency (N. D.)
- i. Discharge Regulator
 - 1. Power capability (watts)
 - 2. Efficiency (N. D.)
- j. Series Load Regulator
 - 1. Output power (watts)
 - 2. Efficiency (N. D.)
- k. Power Converter
 - 1. Special requirement code (C__)

It is important for the reader to note that the fact that a component has no listed technical characteristic does not imply that the component has no important characteristics. The missing technical characteristics will be identified as the Systems Cost/Performance Model is improved.

The power data, which is required in order to design the Electrical Power Subsystem, includes three basic descriptions: the power requirements, the voltage requirements, and the conversion requirements. The average power is the average power required by the component during its active state. The maximum power is the power required either during peak load conditions or during any high power transient periods. The minimum voltage requirement exists during quiescent periods, powered-down periods, or the turned-off condition, if allowable. The voltage requirements are the specifications for which the equipment is rated, i. e., the nominal voltage and the maximum and minimum voltages for which the component will continue to perform within specifications. If the specific component is selected, the converter/inverter requirement flag identifies any need for special power conversion equipment. Since the requirement is identified by a flag, the number used should correspond to the identifier for the actual converter or inverter required.

The component weight which is required by the Vehicle Sizing model includes all weight which is essential to performing the functions associated with the component. Examples of additional functional weight include:

- a. Telemetry instrumentation
- b. Failure sensing and switches
- c. Interface equipment which is not ordinarily a separate component as selected by the Cost/Performance Model.

Weight which comes under different functional descriptions is not included. Examples are:

- a. Wiring harness
- b. Structural mountings

Volume is the direct counterpart of weight and is determined according to the same rules. The summation of the component volumes is used by the Vehicle Sizing Model to estimate the total volume of the payload.

The vibration specification includes both random and non-random categories. Although vibration is not used in the current Cost/Performance Model, the intent is to use the specification in future models.

The maximum and minimum temperature information are the temperature specifications for which the equipment is qualified. This information is used to design the Thermal Control Subsystem.

The pressure information is the ambient pressure for which the component is qualified. This information is also not used by the current model, but can be used in future models.

The CDPI information for each component is used for the express purpose of designing the Data Processing and Communication Subsystems. Command requirements are divided into three categories: power, time tagged, and other. The telemetry requirements are separated

into two categories, i. e., low rate and high rate telemetry requirements. The telemetry information includes:

- a. Number of analog telemetry points
- b. Number of digital telemetry points
- c. Sample rate
- d. Word length

14.3 SAFETY DATA

The intent in supplying the safety information is to indicate the failure mode, the numerics describing the failure mode, the redundancy type, and the maximum amount of redundancy. To this end, the failure model as stated in the data base indicates both the failure mode and the redundancy type. If the failure mode is modeled by an exponential, then the failure rate must be provided. Both the mean and standard deviation are supplied in the event of a normal (gaussian) failure mode. The dormancy factor must be provided for either failure mode. Because the Systems Cost/Performance Model can add an undesirable amount of redundancy (from an engineering point of view), the total allowable number of redundant elements is specified. This redundancy number includes both the original number of components as well as the components added for the purpose of increasing system reliability.

14.4 COST DATA

Component cost information must be supplied for each of the following three categories:

- a. Design engineering
- b. Test and evaluation
- c. Unit production

This information is entered into the data base from component (i.e., assembly) level CERs which have been developed external to the Systems Cost/Performance Model. An additional piece of information which must be provided is the reference quantity required to meet the performance requirements. Redundancy

is not included in the reference quantity. The nondimensional factor has been provided for use in future models where the effect of standardization or use of off-the-shelf hardware is to be incorporated.

14.5 SCHEDULE DATA

Component schedule data include both the development lead time and the qualification lead time. Each lead time is separated into constant and variable terms. Normally, the constant lead times will be exactly the same for all components of the same type. In addition, a state-of-art factor is provided based on the component being in a state of development somewhere between off-the-shelf and a new concept requiring an advance in technology.

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15. MODEL CHECKOUT AND RESULTS

The Systems Cost/Performance Model was checked out by programming the Model on the contractor's CDC 7600 and IBM 370 computers and performing both a thorough debugging and operation of the computer program. Programming of the Model served the purpose of checking out the interfaces between the Subsystems and the Reliability, Cost, and Schedule models. Operation of the computer program served the purpose of identifying errors in the Model. Finally, three test cases were used to check the Cost/Performance Model and the operation of the computer program. The three test cases were:

- a. Defense Satellite Communication System (DSCS-II)
- b. Earth Resources Technology Satellite (ERTS-A)
- c. Orbiting Solar Observatory (OSO-I)

The results of the three test cases are reported in the following three sections. Section 15.4 reports the results of a sensitivity analysis of the effects of reliability and power requirements on cost and weight.

15.1 DSCS-II CHECKOUT

The DSCS-II payload requirements were input to the Cost/Performance computer program to generate a design, along with the respective costs and schedules, which would satisfy the input requirements. Such a checkout is more thorough than may be apparent since the Cost/Performance Model will not necessarily reconstruct an identical DSCS-II design simply because the equipment is in the data base; rather it will configure alternate designs which meet the DSCS-II requirements based on well-defined procedures. The checkout effort consisted of subsystem specialists making a step-by-step investigation of the logic, calculations, and equipment selection in order to identify and correct errors.

The DSCS-II input parameters are listed in Table 15-1. A reproduction of the computer program output listing is provided in Figure 15-1. The output design consists of the following subsystem configurations:

- a. Dual spin
- b. Monopropellant propulsion
- c. Special purpose processing
- d. Separate uplink and downlink
- e. Body mounted solar arrays
- f. Shunt voltage regulation
- g. Cylindrical shape
- h. Single system redundancy

The equipment and the required quantities as selected by the computer program are summarized in Table 15-2. The subsystem weights and vehicle dimensions are compared with the actual values in Tables 15-3 and 15-4, respectively. Table 15-5 compares the Model's cost estimates with the equivalent cost estimates generated by subsystem (PALCM) CERs. The error in the total cost estimate (using preliminary CERs) is less than 23% relative to the actual DSCS-II costs.

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>General (MODE)</u>			
MICRO	0	0	Set to 0 for macro, set to 1, 2, 3, 4, or 5 for micro
ISTR1	1	1	Range of configurations for stabilization and control (must be equal for micro on another subsystem)
IEND1	1	5	
ISTR2	2	1	As above for Auxiliary Propulsion
IEND2	2	3	
ISTR3	2	1	As above for Data Processing and Instrumentation
IEND3	2	2	
ISTR4	2	1	As above for Communications
IEND4	2	5	
ISTR5	2	1	As above for Electrical Power
IEND5	2	6	
ISTR6	1	1	As above for Vehicle Sizing
IEND6	1	3	
ISTRTR	0	0	Reliability
IENDR	0	1	
<u>Stabilization and Control (USRSC)</u>			
DPHI		.25	Main eng. alignment to thrust axis (deg)
FE	3.5	4.1	Thrust (translational) (lb)
TSMALI	71.7	100.	Main eng. burn time (sec)
XNU		3.	Control system efficiency
PDOTO		1	Max, initial rate (deg/sec)

Table 15-1. DSCS-II Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Stabilization and Control (USRSC) (continued)</u>			
TAUX	1.578×10^8	62208000.	Times that disturbance torques are in effect (sec)
TAUY	1.578×10^8	62208000.	
TAUZ	1.578×10^8	62208000.	
T	60.	24.	Mission lifetime (months)
PHIRX	0.393	.75	Required accuracy on roll, pitch, and yaw axes (deg)
PHIRY	0.393	.75	
PHIRZ	0.393	.75	
PDOTX		1.	Max. Maneuver rates (deg/sec)
PDOTY		1.	
PDOTZ		1.	
XN		1.	Number of maneuver about roll, pitch, and yaw axes
YN		1.	
ZN		1.	
PDOTRX		.012	Required system rate accuracy (deg/sec)
PDOTRY		.012	
PDOTRZ		.012	
OMEGS	0	1.5708	Spin rate about yaw axis (rad/sec)
OMEGR	58.	60.	Spin rate of rotor (rpm)
PJ	71.	75.	Platform spin axis inertia (slug-ft ²)
XNN	10.	21.	Time between spin axis corrections (days)
K	1	1	0 if errors for spin axis relative to nadir 1 if errors for payload relative to nadir

Table 15-1. DSCS-II Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Stabilization and Control (USRSC) (continued)</u>			
MANV		1	1 = power flight control, 2 = stationkeeping, 3 = orbit correction control, 4 = vehicle slewing
IPAWAW		0	0 to 1 for payload yaw req. (no or yes)
EPI		.0001	Max. programmed pitchover rate (deg/sec)
AX		.05	Misalignment errors in mount- ing inertia measurement units (deg)
AY		.05	
AZ		.05	
EA		0.10	Antenna misalignment (deg)
EANT		0.1	Antenna elevation (rad)
ALPHA		12.0	Thruster offset in roll-yaw plane (deg)
TL		1.0	Time between unloading wheel momentum (day)
TACCEI		20.0	Acceleration time for maneuver- ing (sec)
XNNN		4.0	Number of single gimbale ^d gyros
THOLD		100,000.	Time vehicle in inertial hold (min)
PDOTAV		0.01	Aug. body rate for low orbit when high accuracy is required (deg/sec)
PDTST		0.0667	Max. rate at which star informa- tion is obtained (deg/sec)

Table 15-1: DSCS-II Input Requirements (Continued)

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>Stabilization and Control (USRSC) (continued)</u>			
PHIFOV		40.0	Max. range of attitude freedom required to track specific stars (deg)
ISAT			Earth pointing flag Equivalent to ISATOR in Thermal (do not input) (deg)
<u>Auxiliary Propulsion (USRAP)</u>			
CLIFE		50,000.	Cycle (or pulse) life (cycles)
<u>Data Processing and Instrumentation (USRDP)</u>			
BTRMX		1.024×10^6	Maximum bit rate (bit/sec)
SCSFL		0.	Special command synchronization flag (0 means no synchronization required, 1 means synchronization required)
TPRFL		0.	Telemetry processing flag (0 means telemetry processed separately, 1 means otherwise)
OPSMS		0.	Number of mission operations (ops/sec)
ARRAYN(11, 3)		(11, 0)	Mission data for up to three equipments:
	16		Power Switching Commands
			Time Tagged Commands
			Other Commands
			High Rate Telemetry
			Number of Analog Points
			Number of Digital Points
			Sample Rate (sec^{-1})
			Word Length (bits)
			Low Rate Telemetry
	68		Number of Analog Points
	16		Number of Digital Points
	0.0075		Sample Rate (sec^{-1})
	8		Word Length (bits)

Table 15-1. DSCS-II Input Requirements (Continued)

<u>FORTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Data Processing and Instrumentation (USRDP) (Continued)</u>			
MISPD		0	Mission data processing flag (1 means processing required, 0 means no such processing required)
NMSEQ	1	0	Number of mission equipment
<u>Communications (USRCM)</u>			
IOPTCM(3)	1, 0, 0	0, 0, 0	IOPTCM(1) is ranging IOPTCM(2) is separate link IOPTCM(3) is separate antenna (0 or 1 for no or yes)
LSGLS		1	Link SGLS flag (0 = no, 1 = yes)
LUSB		0	Link USB flag (0 = no, 1 = yes)
FREQ(2)		2250., 2250.	Frequency of downlink transmit- ters (Mhz)
APOGEE	19,323.	500.	Apogee (must be less than or equal to ALT) (nmi)
NET	.0	1	1 = NASA net, 0 = AFSCF net
NADIR	1	0	Nadir coverage flag (0 = no, 1 = yes)
FREQR		1800	Receiver frequency (Mhz)
COMRAT		1000.	Receiver command rate (baud)
BWIDTH (2)		-1.E10, -1.E10	Bandwidth for transmitter (Hz)

Table 15-1. DSCS-II Input Requirements (Continued)

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>Electrical Power (USREP)</u>			
OPTEMP		15.	Battery temp. (deg. c.)
IVOLT		0	Flag 0 = voltage need not be regulated 1 = voltage regulated
<u>Vehicle Sizing (USRVS)</u>			
EQPF	10.	2.	Equipment packing factor
MB12SH	1	1	Mission equipment bay shape (1 means cylinder, 2 means box)
EQM1XL	48.4	40.	#1 mission equipment bay length (in)
EQM1YL	108.2	40.	#1 mission equipment bay width (in)
EQM1XL	108.2	40.	#1 mission equipment bay height (in)
EQM2XL	0.	40.	#2 mission equipment bay length (in)
EQM2YL	0.	40.	#2 mission equipment bay width (in)
EQM2ZL	0.	40.	#2 mission equipment bay height (in)
ISBOFG	0	0	Solar array boom orientation (0 means not oriented, 1 means oriented)
NUMEEQ	0	0	Number of external equipments (Max = 9)
EEQWT(9)		0	External equipment weight (lb)
EEQVL(9)		0	External equipment volume (ft ³)
EM1YCG		0	
EM1ZCG		0	Mission equipment (in) C.G. 's
EM2YCG		0	
EM2ZCG		0	

Table 15-1. DSCS-II Input Requirements (Continued)

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>Vehicle Sizing (USRVS) (continued)</u>			
CGEEX(9)		2.	Location of external equipment (1 means front, 2 means center, 3 means aft end)
EELOC(9)		1.	Location of external equipment (1 means right, 2 means left, 3 means top, 4 means bottom)
XCGSA1		1.	Location of solar paddles (1 means front, 2 means center, 3 means aft end)
XCGSA3	2.	1.	Location of body mounted solar array (1 means front, 2 means center, 3 means aft end)
<u>Miscellaneous (USRI)</u>			
EQM1WT	181.	435.	Mission equipment weight (lb)
EQM2WT	0.	435.	Mission equipment weight (lb)
DIAMAX	108.	120.	Maximum satellite diameter (in)
ALT	19,323.	500	Altitude (nmi)
<u>Thermal (USRTH)</u>			
ISATOR		1	1 earth oriented 2 sun oriented 3 inertially oriented
ORBINC	2.5	28.5	Orbital inclination
<u>Reliability (USRRE)</u>			
KEOPT		1	Expense Option Indicator 1 expense is weight Otherwise expense is cost
RFIXED		1.0	Initial system reliability

Table 15-1. DSCS-II Input Requirements (Continued)

<u>FORTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Reliability (USRRE) (continued)</u>			
SYSLB	400.	0.0	Initial system weight (lb)
SLBMX	1650.	50000.0	Maximum system weight (lb)
ISPT		0	Single point failure requirements Option 0 not in effect Otherwise in effect
ISUB		0	Subsystem requirements option 1 at least one subsystem has a reliability spec. Otherwise no reliability specs on subsystems.
SPECI*	38.0	18.	Mean mission duration system requirement
SPEC(1)**		.9	R(TRUNC) requirement for S&C subsystem
SPEC(2)**		.9	R(TRUNC) requirement for AP subsystem
SPEC(3)**		.9	R(TRUNC) requirement for SPI subsystem
SPEC(4)**		.9	R(TRUNC) requirement for comm subsystem
SPEC(5)**		.9	R(TRUNC) requirement for EP subsystem
SPEC(6)**	0.236	.6	R(TRUNC) requirement for system
RFNL	0.7		Internal variable that sets ME reliability

* If SPEC 1 \leq 0.1, MMD MODE is skipped in RELY

** If SPEC(K) \leq 0.00001, R(TRUNC) MODE is skipped for subsystem K in RELY

Table 15-1. DSCS-II Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Schedules (USRSK)</u>			
SKDME(7, 3)		0	Schedule data for up to three mission equipments (mo)
<u>Structures (USRST)</u>			
CA		10.	Axial acceleration (g)
CE		5.	Lateral acceleration (g)
<u>Costs (USRCS)</u>			
NFV	6	4	Number of flight vehicles
NQV	1	1	Number of qualification vehicles
XMER	32,300,000.*	0.	Mission equipment DDT&E cost (\$)
SMEU	3,340,000.*	0.	Mission equipment average unit cost (\$)
FEEPCT		0.07	Contractor's fee percentage
IMETYP	1	2	Mission equipment type (1 means Communications, 2 means Earth Observatory)

* These numbers do not reflect DSCS-II costs, but are input for example purposes only.

SYSTEM DESCRIPTION -- DESIGN NUMBER 1

STABILIZATION AND CONTROL
 CONFIGURATION IDENTIFIER 1
 EQUIPMENT CODE IDENTIFIER 101 202 302 401 501 601 701 801 1401
 EQUIPMENT QUANTITIES 1 2 3 2 3 3 2 3 3
 CALCULATED ACCURACY 0.7500E 00 (DFG)
 IERR 0

AUXILIARY PROPULSION
 CONFIGURATION IDENTIFIER 2
 EQUIPMENT CODE IDENTIFIER 801 901 1001 499 201 1102 503 701 1201 001
 EQUIPMENT QUANTITIES 12 4 7 9 2 1 7 2 1 1 1
 TOTAL IMPULSE 0.1867E 05 (LB-SEC)
 IERR 0

DATA PROCESSING AND INSTRUMENTATION
 CONFIGURATION IDENTIFIER 2
 EQUIPMENT CODE IDENTIFIER 201
 EQUIPMENT QUANTITIES 2
 COMPUTER OPERATIONS RATE 0.0 (IPS)
 IERR 1

COMMUNICATIONS
 CONFIGURATION IDENTIFIER 2
 EQUIPMENT CODE IDENTIFIER 101 201 301 401 502 601 701 702
 EQUIPMENT QUANTITIES 3 2 3 3 3 2 2 2
 ENGINEERING DATA RATE 0.2000E 01 (KBPS)
 MISSION EQUIPMENT DATA RATE 0.0 (KBPS)

ELECTRICAL POWER
 CONFIGURATION IDENTIFIER 2
 EQUIPMENT CODE IDENTIFIER 101 205 301 1201
 EQUIPMENT QUANTITIES 2 6 6 2
 TOTAL AVERAGE POWER REQUIREMENT 0.1815E 03 (WATTS)
 SOLAR ARRAY AREA 0.7567E 02 (FT**2)
 MINIMUM INSTALLED BATTERY CAP. 0.5857E 01 (AMP-HR)

THERMAL CONTROL
 RADIATOR AREA 0.4590E 01 (FT**2) ; BATTERY RADIATOR AREA 0.1132E 01 (FT**2)
 TOTAL RADIATOR AREA 0.5722E 01 (FT**2)
 HEATER POWER 0.3066E 03 (BTU/HR) ; BATTERY HEATER POWER 0.1100E 03 (BTU/HR)
 TOTAL HEATER POWER 0.4166E 03 (BTU/HR)
 HEAT PIPE 0.1442E 05 (WATT-IN) ; VARIABLE CONDUCTANCE H.P. 0.1408E 04 (WATT-IN)
 IERR 1100C10111

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15-12

Figure 15-1. Computer Program Output Listing for DSCS-II

15-13

STRUCTURES

SKIN THICKNESS 0.6988E-02 (IN)
 STRINGER NO, THICKNESS, HT 601., 0.1328E-01 (IN), 0.3722E 00 (IN)
 FRAME NO, THICKNESS, HT. 5., 0.7535E-01 (IN), 0.8303E 00 (IN)
 END COVER THICKNESS - FORWARD 1.2391E 00 (IN), CENTER 0.0 (IN), AFT 0.2391E 00 (IN)

VEHICLE SIZING

CONFIGURATION IDENTIFIER 1
 LAUNCH WEIGHT, 0.1401E 04 (LBS), LENGTH 0.1157E 03 (IN),
 WIDTH 0.1059E 03 (IN), HEIGHT 0.1059E 03 (IN),
 IXX 0.1670E 07 (LB-IN**2), IYY 0.1771E 07 (LB-IN**2), IZZ 0.1771E 07 (LB-IN**2)

IERK

SAFETY,

REDUNDANCY CONFIGURATION 0

MEAN MISSION DURATION 0.3894E 02 (MO), RELIABILITY 0.2424E 00, RELIABILITY TRUNCATION TIME 0.6083E 02 (MO)

COST (ALL AMOUNTS ARE IN DOLLARS)

DDT+E

INVESTMENT (RECURRING)

DESIGN ENGINEERING	7804852.0	UNIT ENGINEERING	2180779.0
TEST AND EVALUATION	4192485.0	UNIT PRODUCTION	2605938.0
TOOLING AND TEST EQUIPMENT	0.0	TOOLING AND TEST EQUIPMENT	0.0
QUALITY CONTROL	802020.6	QUALITY CONTROL	397542.9
SYSTEMS ENGINEERING AND INTEGRATION	3818522.0	SYSTEMS ENGINEERING AND INTEGRATION	1271155.0
PROGRAM MANAGEMENT	1580770.0	PROGRAM MANAGEMENT	466466.8

COST CATEGORY

DDT+E

INVESTMENT

OPERATIONS

SPACECRAFT	11898040.	41531280.	
MISSION EQUIPMENT	52300000.	20040000.	
TOTAL PAYLOAD	51198640.	61571280.	
QUALIFICATION UNITS	6921882.		
G.S.E.	2235612.		
LAUNCH SUPPORT			1636781.
FLIGHT OPERATIONS			729502.
CONTRACTOR FEE	1964136.	2907189.	
PROGRAM TOTAL	62323248.	64478464.	2531921.

SCHEDULE

DESIGN AND COMPONENT DEVELOPMENT TIME 14.4 (MONTHS)
 SUBSYSTEM DEVELOPMENT TIME 9.4 (MONTHS)
 COMPONENT QUALIFICATION TIME 14.1 (MONTHS)
 SUBSYSTEM QUALIFICATION TIME 3.2 (MONTHS)
 SYSTEM DEVELOPMENT AND FLIGHT READINESS TIME 9.4 (MONTHS)
 SCHEDULE DURATION (TO LAUNCH) 41.4 (MONTHS)

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Figure 15-1. Computer Program Output Listing for DSCS-II (Continued)

Table 15-2. DSCS-II Equipment

Equipment Type	Data Base Identifier	Number		Actual Total
		Re-quired	Redun-dancy	
Despin Mechanical Assembly	0101	1	0	1
Despin Electronics Assembly				
Valve Driver Assembly	0202	1	1	2
Sun Sensor Assembly	0302	1	2	2
Nutation Damper	0401	1	1	1
Gimbal Electronics Assembly	0501	1	2	1
Control Timing Assembly	0601	1	2	2
Bi-Axial Drive Assembly	0701	2	0	2
Earth Sensor Assembly	0801	1	2	2
S&C Power Converter	1401	1	2	0
Monopropellant (Attitude Control) Thrusters	0807	6	6	6
Monopropellant (Translational) Thrusters	0807	2	2	2
Isolation Valve	0901	4	3	7
Filter	1001	9	0	7
Relief Valve	0701	1	0	1
Propellant Tank	1102	7	0	4
Propellant Filter Drain Valve	1201	1	0	1
Pressure Regulator	0499	1	1	0
Regulator Isolation Valve	0201	1	0	0
Pressurant Tank	0503	1	1	0
Pressurant Fill and Drain Valve	0601	1	0	0
Digital Telemetry Unit	0201	1	1	2
Baseband Assembly Unit	0101	1	2	2
Biconical (Omni) Antenna	0201	1	1	1
Transmitter (0.8 watt)	0301	1	2	2
Receiver	0401	1	2	2

Table 15-2. DSCS-II Equipment (Continued)

Equipment Type	Data Base Identifier	Number		Actual Total
		Re-quired	Redun-dancy	
Command Signal Conditioner	0502	1	2	2
Diplexer	0601	1	1	1
Transmitter Power Converter	0701	1	1	2
Receiver Power Converter	0702	1	1	2
Shunt Regulator	0101	1	1	2
Battery (10 amp-hr)	0205	2	4	3(12A-H)
Battery Charger	0301	2	4	3
Power Control Unit	1201	1	1	1

Table 15-3. DSCS-II Weights

Subsystem	Model Estimates		Actuals	
	kg	(lb)	kg	(lb)
Structures	148.7	327.9	129.7	285.9
Thermal Control	7.0	15.5	17.6	38.9
Electrical Distribution	85.4	188.3	54.0	119.0
Communication, Data Processing and Instrumentation	29.3	64.6	58.7	129.5
Electrical Power	101.4	223.5	93.8	206.7
Stabilization and Control	73.0	161.0	55.2	121.8
Auxiliary Propulsion	50.0	110.3	13.7	30.2
Expendable	50.8	111.9	55.2	121.8
Mission Equipment	82.1	181.0	82.1	181.0
Total Payload	627.8	1384.0	560.1	1234.8
Adapter	7.7	17.0	6.7	14.8
Launch Weight	635.5	1401.0	566.8	1249.6

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Table 15-4. DSCS-II Dimensions

<u>Dimensions</u>	<u>Model Estimates</u>		<u>Actuals</u>	
	<u>m</u>	<u>in.</u>	<u>m</u>	<u>in.</u>
Diameter	2.69	105.9	2.75	108.21
Equipment Bay Length	1.66	65.3	1.70	66.75
Total Length	2.89	113.7	3.03	119.15

Table 15-5. DSCS-II Cost Estimate Comparisons

	<u>Model Estimates</u> <u>(\$1000)</u>	<u>Subsystem CERs*</u> <u>(\$1000)</u>
DDT&E	(60,359)	(61,610)
Spacecraft	28,059	29,310
Mission Equipment	32,300	32,300
Investment	(61,571)	(49,610)
Spacecraft	41,531	29,570
Mission Equipment	20,040	20,040
Operations	(2,366)	(4,540)
Contractor Fee	(5,037)	(4,439)
TOTAL	(129,334)	(120,199)

*The subsystem level cost estimates were generated by the payload cost estimating model, PALCM.

The ERTS-A payload requirements were input to the Cost/Performance computer program to generate a design along with the cost and schedule estimates in order to make a cursory comparison of the computed results with the actual data for an independent program not in the data base. The ERTS input parameters are listed in Table 15-6. Figure 15-2 presents a reproduction of the computer program output listing. The output design consists of the following subsystem configurations:

- a. Three-axis mass expulsion with pitch momentum wheel
- b. Monopropellant propulsion
- c. Special purpose processing
- d. Unified uplink and downlink with common antenna
- e. Separate downlink
- f. Paddle mounted solar arrays
- g. Shunt and discharge voltage regulation
- h. Cylindrical vehicle
- i. Single system redundancy

The equipment selected by the computer program and the required quantities are summarized in Table 15-7. The subsystem weights and vehicle dimensions are compared with the actual values in Table 15-8 and 15-9, respectively.

Table 15-6. ERTS-A Input Requirements

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>General (MODE)</u>			
MICRO	0	0	Set to 0 for macro, set to 1, 2, 3, 4, or 5 for micro
ISTR1	5	1	Range of configurations for stabilization and control (must be equal for micro on another subsystem)
IEND1	5	5	
ISTR2	2	1	As above for Auxiliary Propulsion
IEND2	2	3	
ISTR3	2	1	As above for Data Processing and Instrumentation
IEND3	2	2	
ISTR4	4	1	As above for Communications
IEND4	4	5	
ISTR5	3	1	As above for Electrical Power
IEND5	3	6	
ISTR6	1	1	As above for Vehicle Sizing
IEND6	1	3	
ISTRTR	0	0	Reliability
IENDR	0	1	
<u>Stabilization and Control (USRSC)</u>			
DPHI		.25	Main eng. alignment to thrust axis (deg)
FE	1.0	4.1	Thrust (translational) (lb)
TSMALL	2400.	100.	Main eng. burn time (sec)
XNU		3.	Control system efficiency
PDOTO	.5.	1	Max, initial rate (deg/sec)

Table 15-6. ERTS-A Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Stabilization and Control (USRSC) (continued)</u>			
TAUX	31104000.	62208000.	Times that disturbance torques are in effect (sec)
TAUY	31104000.	62208000.	
TAUZ	31104000.	62208000.	
T	13.	24.	Mission lifetime (months)
PHIRX	0.7	.75	Required accuracy on roll, pitch, and yaw axes (deg)
PHIRY	0.7	.75	
PHIRZ	0.7	.75	
PDOTX		1.	Max. Maneuver rates (deg/sec)
PDOTY		1.	
PDOTZ		1.	
XN		1.	Number of maneuver about roll, pitch, and yaw axes
YN		1.	
ZN		1.	
PDOTRX	0.04	.012	Required system rate accuracy (deg/sec)
PDOTRY	0.04	.012	
PDOTRZ	0.04	.012	
OMEGS		1.5708	Spin rate about yaw axis (rad/sec)
OMEGR		60.	Spin rate of rotor (rpm)
PJ		75.	Platform spin axis inertia (slug-ft ²)
XNN		21.	Time between spin axis corrections (days)
K		1	0 if errors for spin axis relative to nadir 1 if errors for payload relative to nadir

Table 15-6. ERTS-A Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Stabilization and Control (USRSC) (continued)</u>			
MANV		1	1 = power flight control, 2 = stationkeeping, 3 = orbit correction control, 4 = vehicle slewing
IPAWAW		0	0 to 1 for payload yaw req. (no or yes).
EPI		.0001	Max. programmed pitchover rate (deg/sec)
AX		.05	Misalignment errors in mount- ing inertia measurement units (deg)
AY		.05	
AZ		.05	
EA		0.10	Antenna misalignment (deg)
EANT		0.1	Antenna elevation (rad)
ALPHA		12.0	Thruster offset in roll-yaw plane (deg)
TL		1.0	Time between unloading wheel momentum (day)
TACCEL		20.0	Acceleration time for maneuver- ing (sec)
XNNN		4.0	Number of single gimbaleed gyros
THOLD		100,000.	Time vehicle in inertial hold (min)
PDOTAV		0.01	Aug. body rate for low orbit when high accuracy is required (deg/sec)
PDTST		0.0667	Max. rate at which star informa- tion is obtained (deg/sec)

Table 15-6. ERTS-A Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Stabilization and Control (USRSC) (continued)</u>			
PHIFOV		40.0	Max. range of attitude freedom required to track specific stars (deg)
ISAT			Earth pointing flag Equivalent to ISATOR in Thermal (do not input) (deg)
<u>Auxiliary Propulsion (USRAP)</u>			
CLIFE		50,000.	Cycle (or pulse) life (cycles)
<u>Data Processing and Instrumentation (USRDP)</u>			
BTRMX		1.024×10^6	Maximum bit rate (bit/sec)
SCSFL	1	0.	Special command synchronization flag (0 means no synchronization required, 1 means synchronization required)
TPRFL		0.	Telemetry processing flag (0 means telemetry processed separately, 1 means otherwise)
OPSMS		0.	Number of mission operations (ops/sec)
ARRAYN(11, 3)		(11, 0)	Mission data for up to three equipments:
	23.		Power Switching Commands
	0.		Time Tagged Commands
	100.		Other Commands
			High Rate Telemetry
	106.		Number of Analog Points
	106.		Number of Digital Points
	500.		Sample Rate (sec^{-1})
	8.		Word Length (bits)
			Low Rate Telemetry
	280.		Number of Analog Points
	280.		Number of Digital Points
	1.		Sample Rate (sec^{-1})
	8		Word Length (bits)

Table 15-6. ERTS-A Input Requirements (Continued)

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>Data Processing and Instrumentation (USRDP) (Continued)</u>			
MISPD		0	Mission data processing flag (1 means processing required, 0 means no such processing required)
NMSEQ	1	0	Number of mission equipment
<u>Communications (USRCM)</u>			
IOPTCM(3)	1, 1, 0	0, 0, 0	IOPTCM(1) is ranging IOPTCM(2) is separate link IOPTCM(3) is separate antenna (0 or 1 for no or yes)
LSGLS		1	Link SGLS flag (0 = no, 1 = yes)
LUSB		0	Link USB flag (0 = no, 1 = yes)
FREQ(2)	2229.5, 2287.5	2250., 2250.	Frequency of downlink transmitters (Mhz)
APOGEE	500.	500.	Apogee (must be less than or equal to ALT) (nmi)
NET		1	1 = NASA net, 0 = AFSCF net
NADIR		0	Nadir coverage flag (0 = no, 1 = yes)
FREQR	2106.4	1800	Receiver frequency (Mhz)
COMRAT		1000.	Receiver command rate (baud)
BWIDTH (2)		-1.E10, -1.E10	Bandwidth for transmitter (Hz)

Table 15-6. ERTS-A Input Requirements (Continued)

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>Electrical Power (USREP)</u>			
OPTEMP		15.	Battery temp. (deg. c.)
IVOLT	1	0	Flag 0 = voltage need not be regulated 1 = voltage regulated
<u>Vehicle Sizing (USRVS)</u>			
EQPF	4.	2.	Equipment packing factor
MB12SH	1	1	Mission equipment bay shape (1 means cylinder, 2 means box)
EQM1XL	30.	40.	#1 mission equipment bay length (in)
EQM1YL	60.	40.	#1 mission equipment bay width (in)
EQM1XL	60.	40.	#1 mission equipment bay height (in)
EQM2XL	0.	40.	#2 mission equipment bay length (in)
EQM2YL	0.	40.	#2 mission equipment bay width (in)
EQM2ZL	0.	40.	#2 mission equipment bay height (in)
ISBOFG	1	0	Solar array boom orientation (0 means not oriented, 1 means oriented)
NUMEEQ	0	0	Number of external equipments (Max = 9)
EEQWT(9)	0.	0	External equipment weight (lb)
EEQVL(9)	0.	0	External equipment volume (ft ³)
EM1YCG		0	
EM1ZCG		0	Mission equipment (in) C.G. 's
EM2YCG		0	
EM2ZCG		0	

Table 15-6. ERTS-A Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Vehicle Sizing (USRVS) (continued)</u>			
CGEEX(9)	2.	2.	Location of external equipment (1 means front, 2 means center, 3 means aft end)
EELOC(9)	4.	1.	Location of external equipment (1 means right, 2 means left, 3 means top, 4 means bottom)
XCGSA1	2.	1.	Location of solar paddles (1 means front, 2 means center, 3 means aft end)
XCGSA3		1.	Location of body mounted solar array (1 means front, 2 means center, 3 means aft end)
<u>Miscellaneous (USRI)</u>			
EQM1WT	479.5	435.	Mission equipment weight (lb)
EQM2WT	0.	435.	Mission equipment weight (lb)
DIAMAX	60.	120.	Maximum satellite diameter (in)
ALT	500.	500	Altitude (nmi)
<u>Thermal (USRTH)</u>			
ISATOR	1	1	1 earth oriented 2 sun oriented 3 inertially oriented
ORBINC	99.1	28.5	Orbital inclination
<u>Reliability (USRRE)</u>			
KEOPT	1	1	Expense Option Indicator 1 expense is weight Otherwise expense is cost
RFIXED	1.0	1.0	Initial system reliability

Table 15-6. ERTS-A Input Requirements (Continued)

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
	<u>Reliability (USRRE) (continued)</u>		
SYSLB	530.	0.0	Initial system weight (lb)
SLBMX	3000.	50000.0	Maximum system weight (lb)
ISPT	0.	0	Single point failure requirements Option 0 not in effect Otherwise in effect
ISUB	0.	0	Subsystem requirements option 1, at least one subsystem has a reliability spec. Otherwise no reliability specs on subsystems.
SPECI*	12.	18.	Mean mission duration system requirement
SPEC(1)**		.9	R(TRUNC) requirement for S&C subsystem
SPEC(2)**		.9	R(TRUNC) requirement for AP subsystem
SPEC(3)**		.9	R(TRUNC) requirement for SPI subsystem
SPEC(4)**		.9	R(TRUNC) requirement for comm subsystem
SPEC(5)**		.9	R(TRUNC) requirement for EP subsystem
SPEC(6)**	0.5	.6	R(TRUNC) requirement for system
RFNL	1.0		Internal variable that sets ME reliability

* If SPEC 1 \leq 0.1, MMD MODE is skipped in RELY

** If SPEC(K) \leq 0.00001, R(TRUNC) MODE is skipped for subsystem K in RELY

Table 15-6. ERTS-A Input Requirements (Continued)

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>Schedules (USRSK)</u>			
SKDME(7, 3)		0	Schedule data for up to three mission equipments (mo)
<u>Structures (USRST)</u>			
CA		10.	Axial acceleration (g)
CE		5.	Lateral acceleration (g)
<u>Costs (USRCS)</u>			
NFV	2	4	Number of flight vehicles
NQV		1	Number of qualification vehicles
XMER		0.	Mission equipment DDT&E cost (\$)
SMEU		0.	Mission equipment average unit cost (\$)
FEEPCT		0.07	Contractor's fee percentage
IMETYP	2	2	Mission equipment type (1 means Communications, 2 means Earth Observatory)

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SYSTEM DESCRIPTION -- DESIGN NUMBER 1
STABILIZATION AND CONTROL
CONFIGURATION IDENTIFIER 5
EQUIPMENT CODE IDENTIFIERS 1601 2201 1801 1501
EQUIPMENT QUANTITIES 2 2 1 1
CALCULATED ACCURACY 0.4000E-01(DFG)
IERR 0
AUXILIARY PROPULSION
CONFIGURATION IDENTIFIER 2
EQUIPMENT CODE IDENTIFIERS 804 808 901 1001 442 201 1103 501 701 1201 601
EQUIPMENT QUANTITIES 6 2 5 9 2 1 1 1 1 1 1
TOTAL IMPULSE 0.1186L 05(LB-SEC)
IERR 0
DATA PROCESSING AND INSTRUMENTATION
CONFIGURATION IDENTIFIER 2
EQUIPMENT CODE IDENTIFIER 201 201
EQUIPMENT QUANTITIES 2 2
COMPUTER OPERATIONS RATE 0.0 (IPS)
IERR 1001
COMMUNICATIONS
CONFIGURATION IDENTIFIER 4
EQUIPMENT CODE IDENTIFIER 101 205 205 301 302 401 502 601 701 702
EQUIPMENT QUANTITIES 2 1 1 2 2 2 2 1 1 1
ENGINEERING DATA RATE 0.1280E 03(KBPS)
MISSION EQUIPMENT DATA RATE 0.5120E 01(KBPS)
ELECTRICAL POWER
CONFIGURATION IDENTIFIER 3
EQUIPMENT CODE IDENTIFIER 499 501 207 601 701
EQUIPMENT QUANTITIES 2 7 2 2 2
TOTAL AVERAGE POWER REQUIREMENT 0.1478E 03(WATTS)
SOLAR ARRAY AREA 0.4606E 02(FT**2)
MINIMUM INSTALLED BATTERY CAP. 0.1165E 02(AMP-HR)
THERMAL CONTROL
RADIATOR AREA 0.7309E 01 (FT**2) , BATTERY RADIATOR AREA 0.8750E 00 (FT**2)
HEATER POWER 0.5427E 03 (BTU/HR) , TOTAL RADIATOR AREA 0.8184E 01 (FT**2)
HEAT PIPE 0.1639E 05 (WATT-IN) , BATTERY HEATER POWER 0.8499E 02 (BTU/HR)
IERR 1110011011 , TOTAL HEATER POWER 0.6277E 03 (BTU/HR)
, VARIABLE CONDUCTANCE H.P. 0.1112E 04 (WATT-IN)

```

Figure 15-2. Computer Program Output Listing for ERTS-A

STRUCTURES

SKIN THICKNESS 0.8849E-02 (IN)
 STRINGER NO, THICKNESS, HT 402., 0.1681E-01 (IN), 0.3152E 00 (IN)
 FRAME NO, THICKNESS, HT. 5., 0.6382E-01 (IN), 0.7032E 00 (IN)
 END COVER THICKNESS - FORWARD 0.2547E 00 (IN), CENTER 0.4192E 00 (IN), AFT 0.2547E 00 (IN)

VEHICLE SIZING

CONFIGURATION IDENTIFIER 1
 LAUNCH WEIGHT, 0.1589E 04 (LBS), LENGTH 0.1163E 03 (IN),
 WIDTH 0.6000E 02 (IN), HEIGHT 0.6000E 02 (IN),
 IXX 0.8671E 06 (LB-IN**2), IYY 0.1661E 07 (LB-IN**2), IZZ 0.2017E 07 (LB-IN**2)
 IYR 0

SAFETY,

REDUNDANCY CONFIGURATION 0
 MEAN MISSION DURATION 0.1223E 02 (MO), RELIABILITY 0.7951E 00, RELIABILITY TRUNCATION TIME 0.1318E 02 (MO)

COST (ALL AMOUNTS ARE IN DOLLARS)

DDT+E		INVESTMENT (RECURRING)	
DESIGN ENGINEERING	11540206.0	UNIT ENGINEERING	3580022.0
TEST AND EVALUATION	5709257.0	UNIT PRODUCTION	4118257.0
TOOLING AND TEST EQUIPMENT	0.0	TOOLING AND TEST EQUIPMENT	0.0
QUALITY CONTROL	980748.9	QUALITY CONTROL	630256.2
SYSTEMS ENGINEERING AND INTEGRATION	5250504.0	SYSTEMS ENGINEERING AND INTEGRATION	2051623.0
PROGRAM MANAGEMENT	2308024.0	PROGRAM MANAGEMENT	762569.3
COST CATEGORY	DDT+C	INVESTMENT	OPERATIONS
SPACECRAFT	25848316.	22285456.	
MISSION EQUIPMENT	0.	0.	
TOTAL PAYLOAD	25848316.	22285456.	
QUALIFICATION UNITS	11142730.		
G.S.F.	16003617.		
LAUNCH SUPPORT			1470556.
FLIGHT OPERATIONS			1409414.
CONTRACTOR FEE	5709000.	1559981.	
PROGRAM TOTAL	56704000.	23845424.	3081509.

SCHEDULE

DESIGN AND COMPONENT DEVELOPMENT TIME 26.3 (MONTHS)
 SUBSYSTEM DEVELOPMENT TIME 14.8 (MONTHS)
 COMPONENT QUALIFICATION TIME 17.0 (MONTHS)
 SUBSYSTEM QUALIFICATION TIME 6.5 (MONTHS)
 SYSTEM DEVELOPMENT AND FLIGHT READINESS TIME 22.2 (MONTHS)
 SCHEDULE DURATION (TO LAUNCH) 71.8 (MONTHS)

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Figure 15-2. Computer Program Output Listing for ERTS-A (Continued)

Table 15-7. ERTS-A: Equipment

Equipment Type	Data Base Identifier	Number	
		Required	Redundancy
Horizon Sensor	1801	1	0
Electronic Error Processor	2201	1	1
Reaction Wheel Assembly	1301	1	0
Valve Driver Assembly	1601	1	1
Monopropellant (Attitude Control) Thrusters (0.5 lb)	0804	6	0
Monopropellant (Translational) Thrusters (1.0 lb)	0806	2	0
Isolation Valve	0901	4	1
Filter	1001	9	0
Relief Valve	0701	1	0
Propellant Tank	1103	1	0
Propellant Fill & Drain Valve	1201	1	0
Pressure Regulator	0499	1	1
Regulator Isolation Valve	0201	1	0
Pressurant Tank	0501	1	0
Pressurant Fill & Drain Valve	0601	1	0
Digital Telemetry Unit	0201	2	2
Baseband Assembly Unit	0101	1	1
Conical Spiral Antenna	0205	1	0
Transmitter (0.8 watts)	0301	1	1
Receiver	0401	1	1
Command Signal Conditioner	0502	1	1
Diplexer	0601	1	0
Transmitter (1.6 watts)	0302	1	1
Conical Spiral Antenna	0205	1	0
Transmitter Power Converter	0701	1	0

Table 15-7. ERTS-A Equipment (Continued)

Equipment Type	Data Base Identifier	Number	
		Required	Redundancy
Receiver Power Converter	0702	1	0
Battery (12.0 amp-hr)	0207	2	0
Shunt Regulator	0501	5	2
Discharge Regulator	0499	2	0
Battery Charger	0601	2	0
Central Control Unit	0701	1	1

Table 15-8. ERTS-A Weights

Subsystem	Model Estimates		Actuals	
	kg	(lb)	kg	(lb)
Structure	184.6	406.9	181.1	399.3
Thermal Control	4.6	10.2	36.4	80.2
Stabilization and Control	55.1	121.5	31.9	70.4
Auxiliary Propulsion	18.4	40.6	33.8	74.6
Expendables	30.4	67.0	36.6	80.7
Electrical Power	81.4	179.5	148.6	327.5
Electrical Distribution	89.7	197.7	89.3	196.8
Communications, Data Processing & Instrumentation	30.9	68.1	104.1	229.6
Mission Equipment	217.5	479.5	217.5	479.5
Total Payload	712.6	1571.0	879.3	1938.6
Adapter	8.2	18.0	61.9	136.5
Launch Weight	720.8	1589.0	941.2	2075.1

Table 15-9. ERTS-A Dimensions

Dimensions	Model Estimates		Actuals	
	m	in.	m	in.
Diameter	1.52	60.	1.52	60.
Equipment Bay Height	2.19	86.3		
Total Height	2.95	116.3	3.04	120.

OSO-I CHECKOUT

The OSO-I payload requirements were input to the Cost/Performance computer program as a test case to be evaluated by NASA. The OSO-I input parameters are listed in Table 15-10. The computer program output listing is reproduced in Figure 15-3. The output design consisted of the following subsystem configurations:

- a. Dual spin
- b. Cold gas propulsion
- c. Special purpose processing
- d. Unified uplink and downlink having common antenna
- e. Separate downlink
- f. Body mounted solar array
- g. Series load voltage regulation
- h. Cylindrically shaped vehicle
- i. Single system redundancy

The equipments selected by the computer program are listed in Table 15-11. The subsystem weights and vehicle dimensions are compared with the contractor's projected values in Tables 15-12 and 15-13, respectively.

Table 15-10. OSO-I Input Requirements

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>General (MODE)</u>			
MICRO	0	0	Set to 0 for macro, set to 1, 2, 3, 4, or 5 for micro
ISTR1	1	1	Range of configurations for stabilization and control (must be equal for micro on another subsystem)
IEND1	1	5	
ISTR2	1	1	As above for Auxiliary Propulsion
IEND2	1	3	
ISTR3	2	1	As above for Data Processing and Instrumentation
IEND3	2	2	
ISTR4	4	1	As above for Communications
IEND4	4	5	
ISTR5	6	1	As above for Electrical Power
IEND5	6	6	
ISTR6	1	1	As above for Vehicle Sizing
IEND6	1	3	
ISTRTR	0	0	Reliability
IENDR	0	1	
<u>Stabilization and Control (USRSC)</u>			
DPHI	0.	.25	Main eng. alignment to thrust axis (deg)
FE	0.	4.1	Thrust (translational) (lb)
TSMALL	0.	100.	Main eng. burn time (sec)
XNU	12.	3.	Control system efficiency
PDOTO		1	Max, initial rate (deg/sec)

Table 15-10. OSO-I Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Stabilization and Control (USRSC) (continued)</u>			
TAUX	31104000.	62208000.	Times that disturbance torques are in effect (sec)
TAUY	31104000.	62208000.	
TAUZ	31104000.	62208000.	
T	13.	24.	Mission lifetime (months)
PHIRX	2.	.75	Required accuracy on roll, pitch, and yaw axes (deg)
PHIRY	2.	.75	
PHIRZ	2.	.75	
PDOTX	0.	1.	Max. Maneuver rates (deg/sec)
PDOTY	0.	1.	
PDOTZ	0.	1.	
XN	0.	1.	Number of maneuver about roll, pitch, and yaw axes
YN	0.	1.	
ZN	0.	1.	
PDOTRX		.012	Required system rate accuracy (deg/sec)
PDOTRY		.012	
PDOTRZ		.012	
OMEGS		1.5708	Spin rate about yaw axis (rad/sec)
OMEGR	5.	60.	Spin rate of rotor (rpm)
PJ	150.	75.	Platform spin axis inertia (slug-ft ²)
XNN	10.	21.	Time between spin axis corrections (days)
K	1	1	0 if errors for spin axis relative to nadir 1 if errors for payload relative to nadir

Table 15-10. OSO-I Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Stabilization and Control (USRSC) (continued)</u>			
MANV		1	1 = power flight control, 2 = stationkeeping, 3 = orbit correction control, 4 = vehicle slewing
IPAWAW		0	0 to 1 for payload yaw req. (no or yes)
EPI		.0001	Max. programmed pitchover rate (deg/sec)
AX		.05	Misalignment errors in mount- ing inertia measurement units (deg)
AY		.05	
AZ		.05	
EA		0.10	Antenna misalignment (deg)
EANT		0.1	Antenna elevation (rad)
ALPHA		12.0	Thruster offset in roll-yaw plane (deg)
TL		1.0	Time between unloading wheel momentum (day)
TACCEL		20.0	Acceleration time for maneuver- ing (sec)
XNNN		4.0	Number of single gimbaled gyros
THOLD		100,000.	Time vehicle in inertial hold (min)
PDOTAV		0.01	Aug. body rate for low orbit when high accuracy is required (deg/sec)
PDTST		0.0667	Max. rate at which star informa- tion is obtained (deg/sec)

Table 15-10. OSO-I Input Requirements (Continued)

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>Stabilization and Control (USRSC) (continued)</u>			
PHIFOV		40.0	Max. range of attitude freedom required to track specific stars (deg)
ISAT			Earth pointing flag Equivalent to ISATOR in Thermal (do not input) (deg)
<u>Auxiliary Propulsion (USRAP)</u>			
CLIFE		50,000.	Cycle (or pulse) life (cycles)
<u>Data Processing and Instrumentation (USRDP)</u>			
BTRMX	1.28×10^5	1.024×10^6	Maximum bit rate (bit/sec)
SCSFL	1	0.	Special command synchronization flag (0 means no synchronization required, 1 means synchronization required)
TPRFL		0.	Telemetry processing flag (0 means telemetry processed separately, 1 means otherwise)
OPSMS		0.	Number of mission operations (ops/sec)
ARRAYN(11, 3)		(11, 0)	Mission data for up to three equipments:
	28.		Power Switching Commands
	8.		Time Tagged Commands
	100.		Other Commands
			High Rate Telemetry
	106.		Number of Analog Points
	106.		Number of Digital Points
	1.		Sample Rate (sec ⁻¹)
	8.		Word Length (bits)
			Low Rate Telemetry
	280.		Number of Analog Points
	280.		Number of Digital Points
	0.0075		Sample Rate (sec ⁻¹)
	8.		Word Length (bits)

Table 15-10. OSO-I Input Requirements (Continued)

<u>FORTTRAN Name</u>	<u>Input</u>	<u>Default Value</u>	<u>Description</u>
<u>Data Processing and Instrumentation (USRDP) (Continued)</u>			
MISPD		0	Mission data processing flag (1 means processing required, 0 means no such processing required)
NMSEQ	1	0	Number of mission equipment
<u>Communications (USRCM)</u>			
IOPTCM(3)	1, 1, 0	0, 0, 0	IOPTCM(1) is ranging IOPTCM(2) is separate link IOPTCM(3) is separate antenna (0 or 1 for no or yes)
LSGLS		1	Link SGLS flag (0 = no, 1 = yes)
LUSB		0	Link USB flag (0 = no, 1 = yes)
FREQ(2)	2212.5, 2212.5	2250., 2250.	Frequency of downlink transmitters (Mhz)
APOGEE	300.	500.	Apogee (must be less than or equal to ALT) (nmi)
NET		1	1 = NASA net, 0 = AFSCF net
NADIR		0	Nadir coverage flag (0 = no, 1 = yes)
FREQR		1800	Receiver frequency (Mhz)
COMRAT	800	1000.	Receiver command rate (baud)
BWIDTH (2)		-1.E10, -1.E10	Bandwidth for transmitter (Hz)

Table 15-10. OSO-I Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Electrical Power (USREP)</u>			
OPTEMI		15.	Battery temp. (deg. c.)
IVOLT		0	Flag 0 = voltage need not be regulated 1 = voltage regulated
<u>Vehicle Sizing (USRVS)</u>			
EQPF	2.	2.	Equipment packing factor
MB12SH	2	1	Mission equipment bay shape (1 means cylinder, 2 means box)
EQM1XL	77.7	40.	#1 mission equipment bay length (in)
EQM1YL	83.	40.	#1 mission equipment bay width (in)
EQM1XL	7.	40.	#1 mission equipment bay height (in)
EQM2XL	0.	40.	#2 mission equipment bay length (in)
EQM2YL	0.	40.	#2 mission equipment bay width (in)
EQM2ZL	0.	40.	#2 mission equipment bay height (in)
ISBOFG		0	Solar array boom orientation (0 means not oriented, 1 means oriented)
NUMEEQ		0	Number of external equipments (Max = 9)
EEQWT(9)		0	External equipment weight (lb)
EEQVL(9)		0	External equipment volume (ft ³)
EM1YCG		0	
EM1ZCG		0	Mission equipment (in) C.G. 's
EM2YCG		0	
EM2ZCG		0	

Table 15-10. OSO-I Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Vehicle Sizing (USRVS) (continued)</u>			
CGEEX(9)		2.	Location of external equipment (1 means front, 2 means center, 3 means aft end)
EELOC(9)		1.	Location of external equipment (1 means right, 2 means left, 3 means top, 4 means bottom)
XCGSA1		1.	Location of solar paddles (1 means front, 2 means center, 3 means aft end)
XCGSA3	1.	1.	Location of body mounted solar array (1 means front, 2 means center, 3 means aft end)
<u>Miscellaneous (USRI)</u>			
EQM1WT	848.3	435.	Mission equipment weight (lb)
EQM2WT	0.	435.	Mission equipment weight (lb)
DIAMAX	86.	120.	Maximum satellite diameter (in)
ALT	300.	500	Altitude (nmi)
<u>Thermal (USRTH)</u>			
ISATOR	2	1	1 earth oriented 2 sun oriented 3 inertially oriented
ORBINC	33.	28.5	Orbital inclination
<u>Reliability (USRRE)</u>			
KEOPT	1	1	Expense Option Indicator 1 expense is weight Otherwise expense is cost
RFIXED	1.0	1.0	Initial system reliability

Table 15-10. OSO-I Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Reliability (USRRE) (continued)</u>			
SYSLB	848.3	0.0	Initial system weight (lb)
SLBMX	3100.	50000.0	Maximum system weight (lb)
ISPT	0	0	Single point failure requirements Option 0 not in effect Otherwise in effect
ISUB	0	0	Subsystem requirements option 1 at least one subsystem has a reliability spec. Otherwise no reliability specs on subsystems.
SPECI*	12.	18.	Mean mission duration system requirement
SPEC(1)**		.9	R(TRUNC) requirement for S&C subsystem
SPEC(2)**		.9	R(TRUNC) requirement for AP subsystem
SPEC(3)**		.9	R(TRUNC) requirement for SPI subsystem
SPEC(4)**		.9	R(TRUNC) requirement for comm subsystem
SPEC(5)**		.9	R(TRUNC) requirement for EP subsystem
SPEC(6)**	0.5	.6	R(TRUNC) requirement for system
RFNL	1.0		Internal variable that sets ME reliability

* If SPEC 1 \leq 0.1, MMD MODE is skipped in RELY

** If SPEC(K) \leq 0.00001, R(TRUNC) MODE is skipped for subsystem K in RELY

Table 15-10. OSO-I Input Requirements (Continued)

<u>FORTTRAN</u> <u>Name</u>	<u>Input</u>	<u>Default</u> <u>Value</u>	<u>Description</u>
<u>Schedules (USRSK)</u>			
SKDME(7, 3)		0	Schedule data for up to three mission equipments (mo)
<u>Structures (USRST)</u>			
CA		10.	Axial acceleration (g)
GE		5.	Lateral acceleration (g)
<u>Costs (USRCS)</u>			
NFV	1	4	Number of flight vehicles
NQV	0	1	Number of qualification vehicles
XMER	1,322,800.	0.	Mission equipment DDT&E cost (\$)
SMEU	155,067.	0.	Mission equipment average unit cost (\$)
FEEPCT		0.07	Contractor's fee percentage
IMETYP	0	2	Mission equipment type (1 means Communications, 2 means Earth Observatory)

SYSTEM DESCRIPTION -- DESIGN NUMBER 1

STABILIZATION AND CONTROL										
CONFIGURATION IDENTIFIER	1									
EQUIPMENT CODE IDENTIFIER	101	202	302	401	501	601	701	801	1401	
EQUIPMENT QUANTITIES	1	2	2	1	2	2	2	2	2	
CALCULATED ACCURACY	0.2000E 01(DEG)									
IERR	0									
AUXILIARY PROPULSION										
CONFIGURATION IDENTIFIER	1									
EQUIPMENT CODE IDENTIFIER	105	103	204	301	499	599	601	701		
EQUIPMENT QUANTITIES	6	2	4	4	2	1	1	1		
TOTAL IMPULSE	0.4272E 04(LB-SEC)									
IERR	1									
DATA PROCESSING AND INSTRUMENTATION										
CONFIGURATION IDENTIFIER	2									
EQUIPMENT CODE IDENTIFIER	201	201								
EQUIPMENT QUANTITIES	2	2								
COMPUTER OPERATIONS RATE	0.0 (IPS)									
IERR	1002									
COMMUNICATIONS										
CONFIGURATION IDENTIFIER	4									
EQUIPMENT CODE IDENTIFIER	101	203	203	301	302	401	502	601	701 702	
EQUIPMENT QUANTITIES	2	1	1	2	2	2	2	1	2 2	
ENGINEERING DATA RATE	0.1000E 01(KBPS)									
MISSION EQUIPMENT DATA RATE	0.4000E 01(KBPS)									
ELECTRICAL POWER										
CONFIGURATION IDENTIFIER	0									
EQUIPMENT CODE IDENTIFIER	201	205	301	1001	1101					
EQUIPMENT QUANTITIES	3	2	2	2	2					
TOTAL AVERAGE POWER REQUIREMENT	0.1768E 03(WATTS)									
SOLAR ARRAY AREA	0.1259E 03(FT**2)									
MINIMUM INSTALLED BATTERY CAP.	0.9577E 01(AHP-HR)									
THERMAL CONTROL										
RADIATOR AREA	0.9985E 01 (FT**2)				BATTERY RADIATOR AREA				0.1029E 01 (FT**2)	
					TOTAL RADIATOR AREA				0.1101E 02 (FT**2)	
HEATER POWER	0.6945E 03 (BTU/HR)				BATTERY HEATER POWER				0.9999E 02 (BTU/HR)	
					TOTAL HEATER POWER				0.7945E 03 (BTU/HR)	
HEAT PIPE	0.1590E 05 (WATT-IN)				VARIABLE CONDUCTANCE H.P.				0.1449E 04 (WATT-IN)	
IERR	1110011011									

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Figure 15-3. Computer Program Output Listing for OSO-I

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STRUCTURES
SKIN THICKNESS 0.9365E-02(IN)
STRINGER NO,THICKNESS,HT 459., 0.1779E-01 (IN), 0.3808E 00 (IN)
FRAME NO, THICKNESS, HT. 5., 0.7711E-01 (IN), 0.8496E 00 (IN)
END COVER THICKNESS - FORWARD 0.3204E 00(IN),CENTER 0.0 (IN),AFT 0.3204E 00(IN)
VEHICLE SIZING
CONFIGURATION IDENTIFIER 1
LAUNCH WEIGHT, 0.2515E 04(LBS), LENGTH 0.1288E 03(IN) ,
WIDTH 0.8276E 02(IN), HEIGHT 0.8276E 02 (IN),
IXX 0.1398E 07 (LB-IN**2), IYY 0.3173E 07 (LB-IN**2), IZZ 0.3802E 07 (LB-IN**2).
IERR 1
SAFETY,
REDUNDANCY CONFIGURATION 0
MEAN MISSION DURATION 0.1219E 02(MO),RELIABILITY 0.7908E 00,RELIABILITY TRUNCATION TIME 0.1318E 02(MO)
COST (ALL AMOUNTS ARE IN DOLLARS)
DDT+E INVESTMENT(RECURRING)
DESIGN ENGINEERING 8223823.0 UNIT ENGINEERING 3690373.0
TEST AND EVALUATION 4546857.0 UNIT PRODUCTION 1441224.0
TOOLING AND TEST EQUIPMENT 0.0 TOOLING AND TEST EQUIPMENT 0.0
QUALITY CONTROL 759921.4 QUALITY CONTROL 257126.9
SYSTEMS ENGINEERING AND INTEGRATION 3859282.0 SYSTEMS ENGINEERING AND INTEGRATION 1497988.0
PROGRAM MANAGEMENT 1653463.0 PROGRAM MANAGEMENT 729995.3
COST CATEGORY DDT+E INVESTMENT OPERATIONS
SPACECRAFT 19043360. 7618710.
MISSION EQUIPMENT 1322800. 155067.
TOTAL PAYLOAD 20366160. 7771777.
QUALIFICATION UNITS 0.
G.S.E. 5924349.
LAUNCH SUPPORT 255824.
FLIGHT OPERATIONS 117647.
CONTRACTOR FEE 1747738. 553170.
PROGRAM TOTAL 28038224. 8304946. 399613.
SCHEDULE
DESIGN AND COMPONENT DEVELOPMENT TIME 22.4(MONTHS)
SUBSYSTEM DEVELOPMENT TIME 2.9(MONTHS)
COMPONENT QUALIFICATION TIME 10.0(MONTHS)
SUBSYSTEM QUALIFICATION TIME 7.3(MONTHS)
SYSTEM DEVELOPMENT AND FLIGHT READINESS TIME 9.3(MONTHS)
SCHEDULE DURATION (TO LAUNCH) 41.9(MONTHS)

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Figure 15-3. Computer Program Output Listing for OSO-I (Continued)

Table 15-11. OSO-I Equipment

Equipment Type	Data Base Identifier	Number	
		Required	Redundancy
Despin Mechanical Assembly } Despin Electronics Assembly }	0101	1	0
Valve Driver Assembly	0202	1	1
Sun Sensor Assembly	0302	1	1
Nutation Damper	0401	1	0
Gimbal Electronics Assembly	0501	1	1
Control Timing Assembly	0601	1	1
Bi-Axial Drive Assembly	0701	2	0
Earth Sensor Assembly	0801	1	1
S&C Power Converter	1401	1	1
Cold Gas (Attitude Control) Thruster	0105	6	0
Cold Gas (Translational) Thruster	0103	2	0
Isolation Valve	0204	4	0
Filter	0302	9	0
Pressure Regulator	0499	1	1
Tank	0599	1	0
Fill and Drain Valve	0601	1	0
Relief Valve	0701	1	0
Digital Telemetry Unit	0201	2	2
Baseband Assembly Unit	0101	1	1
Omni Antenna	0203	1	0
Transmitter (0.8 watts)	0301	1	1
Receiver	0401	1	1
Command Signal Conditioner	0502	1	1
Diplexer	0601	1	0

Table 15-11. OSO-I Equipment (Continued)

Equipment Type	Data Base Identifier	Number	
		Required	Redundancy
Transmitter Power Converter	0701	1	1
Receiver Power Converter	0702	1	1
Transmitter (1.8 watts)	0302	1	1
Omni Antenna	0202	1	0
Battery (10.0 amp - hour)	0205	2	1
Series Load Regulator	0801	2	0
Battery Charger	0901	2	0
Solar Power Distributor	1001	1	1
Power Distributor	1101	1	1

C 1.6

Table 15-12. OSO-I Weights

Subsystem	Model Estimates		Contractor Estimates	
	kg	(lb)	kg	(lb)
Structure (includes adapter)	315.3	695.1	140.3	309.4
Thermal Control	5.9	12.9	7.0	15.4
Electrical Distribution	161.8	356.8	31.8	70.1
Electrical Power	67.9	149.7	56.3	124.2
Stabilization & Control (includes APS)	139.8	308.3	334.3	737.0
Expendables	32.8	72.3	11.8	26.0
Communications, Data Processing & Instrumentation	32.5	71.6	65.0	143.2
Mission Equipment	384.8	848.3	384.8	848.3
Launch Weight	1140.8	2515.0	1031.3	2273.6

Table 15-13. OSO-I Dimensions

Dimensions	Model Estimates		Contractor Estimates	
	m	in.	m	in.
Diameter	2.10	82.8	2.10	82.8
Equipment Bay Length	1.30	51.1	1.26	49.8
Total Length	3.27	128.8	3.24	127.6

The Systems Cost/Performance Model is an excellent tool for performing trade studies and sensitivity analyses. A series of computer runs were performed in order to determine the sensitivity of payload weight and cost to changes in reliability requirements and to changes in power levels.

Figure 15-4 presents the weight estimates generated by the Model as a function of the payload reliability. The input requirements correspond to the DSCS-II payload, and the nominal design weight is identified by a small circle. The minimum weight, single string system has a weight of 394.2 kg (869.1 lb) and a mean mission duration (MMD) of 21.1 months. In order to increase the MMD even slightly requires the addition of substantial redundancy for the "weak links." For MMDs between two and three years, the required increase in redundancy and, therefore, weight is less dramatic. However, as the MMD requirement approaches 39 months, certain equipment (e.g., the Despin Mechanical Assembly which was not allowed to be made redundant) prevents the payload MMD from being increased further. The net result of the analysis is an interesting and logical understanding of the impact of the mean mission duration requirement on the DSCS-II launch weight.

Figure 15-5 presents cost estimates generated by the Model for DSCS-II as a function of reliability. The cost estimates are relatively insensitive to changes in payload mean mission duration at low levels due to the inherent reliability of the single string system. However, attempts to increase reliability substantially causes costs to turn upward reflecting the diminishing returns and increasing costs of adding redundancy. The cost results generated by the Model provide more insight than the current CER approaches which are restricted to straight line approximations about the nominal cost as shown in the figure.

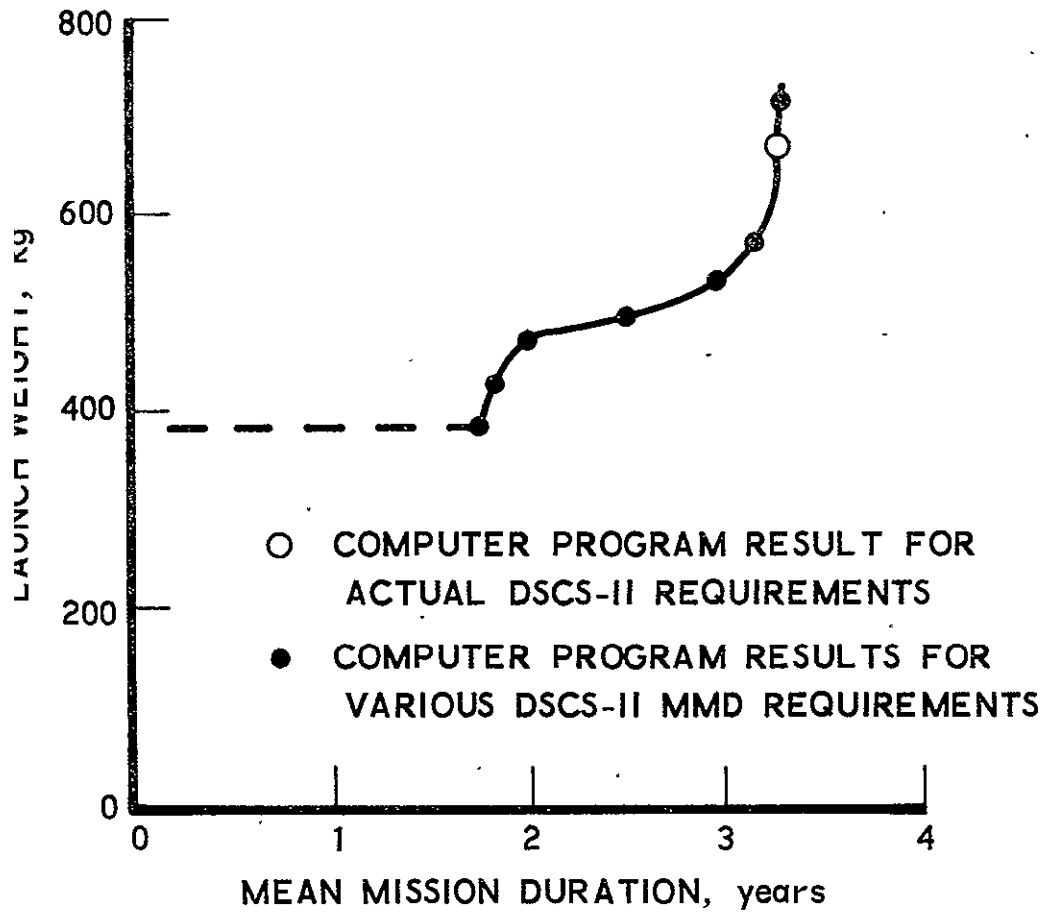


Figure 15-4. DSCS-II Weight versus Extended Life

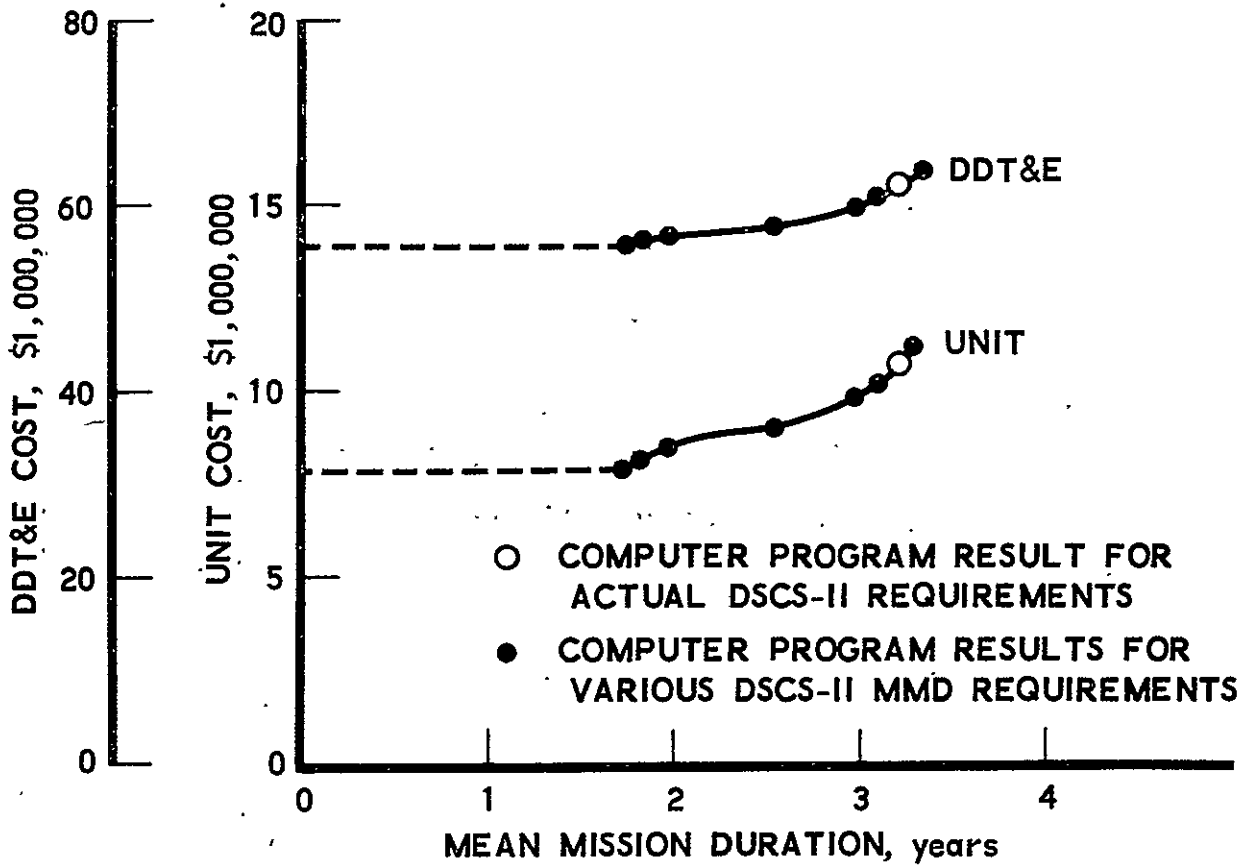


Figure 15-5. DSCS-II Cost versus Extended Life

Figure 15-6 represents the effect of changes in the mission equipment power requirement on the DSCS-II launch weight. The weight change reflects not only the change in the Electrical Power subsystem, but the changes in all other subsystems including S&C, APS, CDPI, Thermal, and Structure. The distinct advantage of the Systems Cost/Performance Model shows up in its ability to predict the effect on all subsystems of a change in another part of the vehicle. Figure 15-7 presents the cost estimates generated by the Model as a function of the payload electrical power requirement. Both the subsystems and the mission equipment requirements are included in the total power requirement.

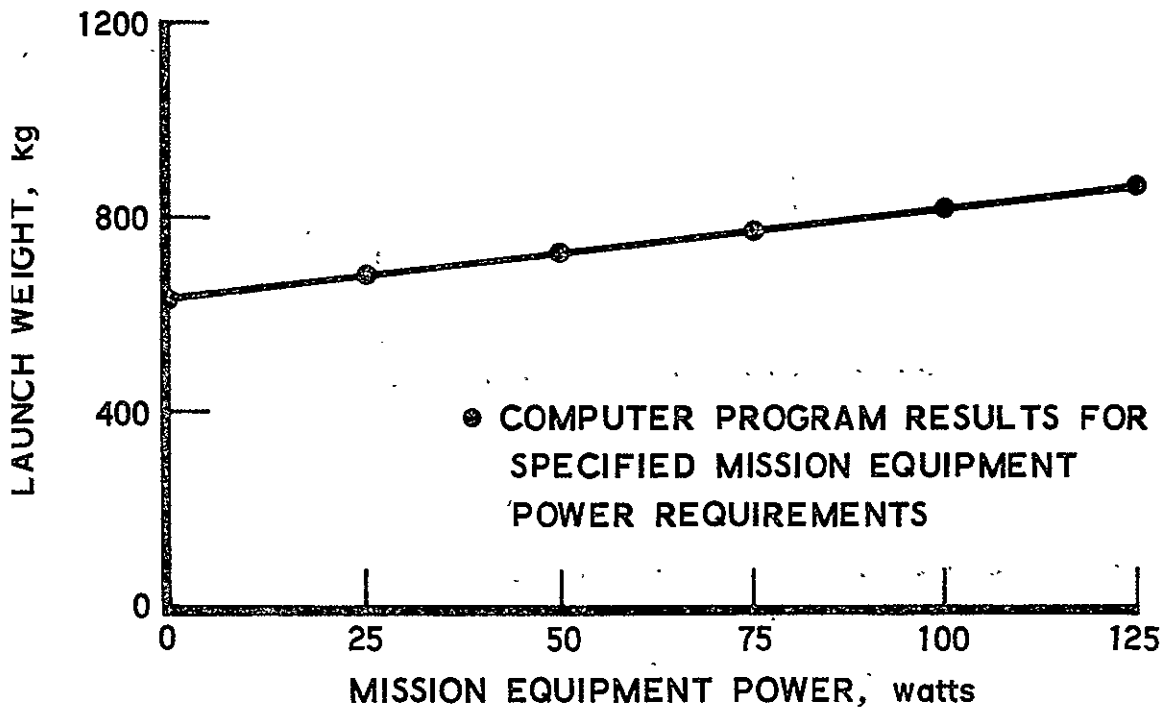


Figure 15-6. DSCS-II Weight versus Mission Equipment Power Requirement

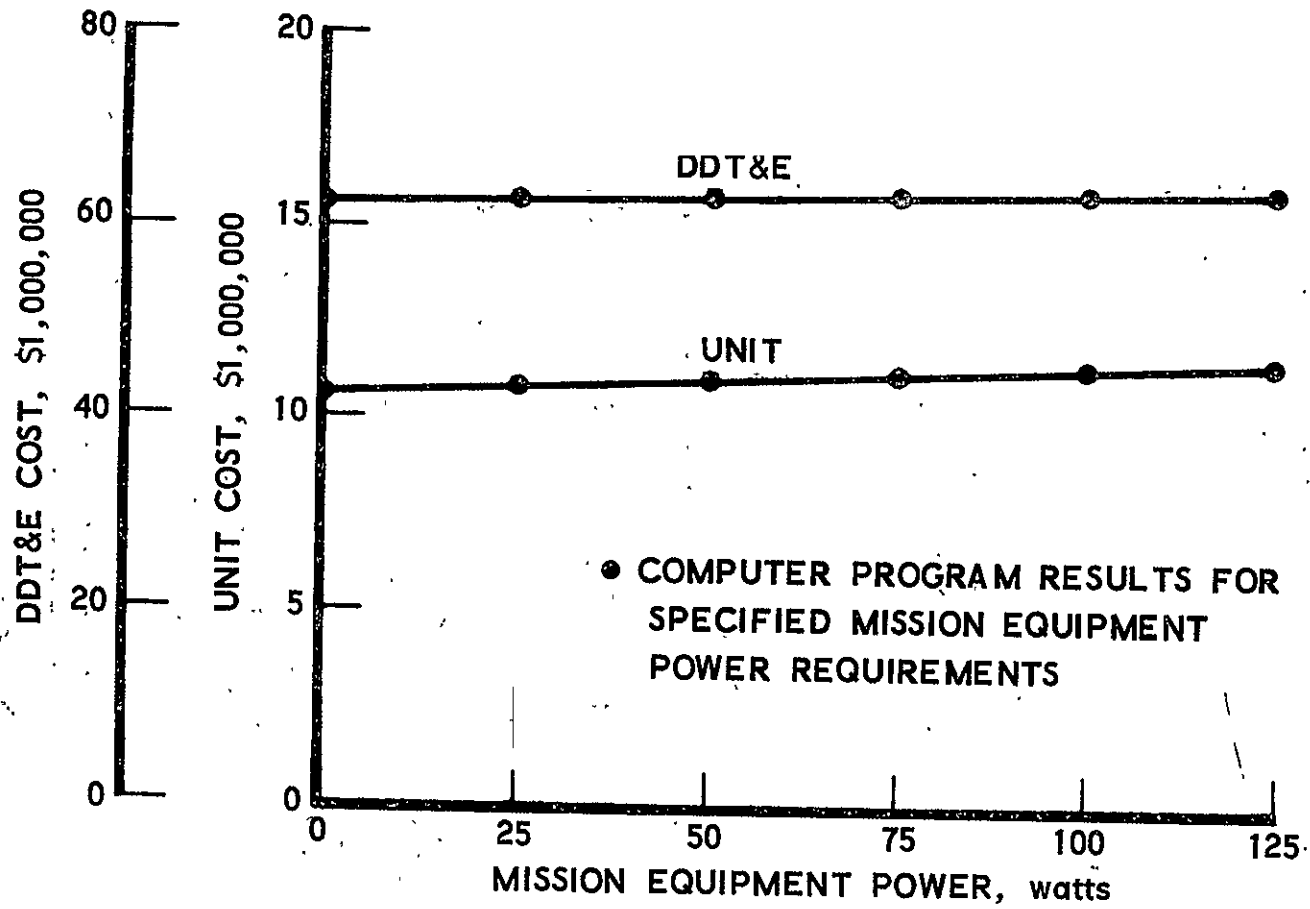


Figure 15-7. DSCS-II Cost versus Mission Equipment Power Requirement

16. CONCLUSIONS

The principal goal of this study was to identify and quantify the interrelationships between and within the performance, safety, cost, and schedule parameters in support of an overall effort by NASA to generate program models and methodology that provide insight into the effect of changes in specific system functional requirements on the total vehicle program.

So that this goal could be achieved, three objectives were established for this study. The first objective was to refine and improve the cost/performance methodology which was developed during the preceding fiscal year's study. The second objective was the application of the methodology to unmanned, automated payloads. The third objective was to implement the resulting model as a digital computer program.

In fulfilling the objectives, the Systems Cost/Performance Model was established. The Cost/Performance Model identifies acceptable payload designs for the following subsystems:

- a. Stabilization and Control
- b. Auxiliary Propulsion
- c. Communication, Data Processing and Instrumentation
- d. Electrical Power
- e. Thermal Control
- f. Structure

Redundancy is added to the payload design as necessary, and the costs and schedules required to design, develop, qualify, build, check-out, and prepare flight vehicles are estimated. The Model incorporates a data base comprised of assemblies with the requisite performance, safety, cost, and schedule information specified.

The Systems Cost/Performance Model was implemented as a digital computer program in Fortran IV. The program is designed to operate on either a CDC 7600 computer or a Univac 1108 computer.* Use of the computer program allows the user to establish specific designs and the related costs and schedules almost immediately. In addition, the user is able to determine the sensitivity of design, costs, and schedules to changes in requirements.

Three test cases were used to check the Cost/Performance Model and the operation of the computer program. The three test cases were:

- a. Defense Satellite Communication System (DSCS-II)
- b. Earth Resources Technology Satellite (ERTS-A)
- c. Orbiting Solar Observatory (OSO-I)

The results of these three test cases do indicate, in fact, that the current Model is capable of reasonable accuracy. The Model's accuracy is limited by the accuracy and completeness of the input data submitted by the user. For the given test cases, the physical descriptions and equipment selected corresponded quite well with the contractor's data. The error in the total cost estimate generated by the Model for the DSCS-II test case was less than 23% relative to the actual cost. At the same time, the model provided estimates and, therefore, insight into the effect of other variables (i. e., reliability and power requirements) on payload cost.

Generally speaking, the Cost/Performance Model should exceed the performance of "top-down" models. The Model uses a "bottom-up" approach and, therefore, designs the payload at the assembly level. Greater accuracy is achieved by the very nature of the more detailed design. This accuracy will be reflected in the cost and schedule model estimates. A second attribute of the Cost/Performance Model is the completeness of the design specified. Pieces of equipment are not forgotten,

*The Systems Cost/Performance Computer Program is currently operational on The Aerospace Corporation's CDC 7600 computer. The program is expected to be operational on MSFC's Univac 1108 computer in the near future.

and redundancy is automatically included in the specified design. In addition, the impact of all subsystem interfaces and interactions is properly modeled. The net result is a payload design which is as accurate and complete as one from a Pre-Phase A study and which is available to the Cost/Performance Computer Program user immediately.

The current Cost/Performance Model is limited to modeling spacecraft in earth orbit. More importantly, the current Model is limited in the range of payload designs it can generate by the limited amount of equipment in the data base. Accuracy of the cost estimates is limited by the relatively limited amount of cost data which could be reduced and processed to support the data base cost entries.

Because of the detailed nature of the Model, the uses of the System Cost/Performance Model far exceed that for "top-down" models.

The following uses of the model are suggested:

- a. Establish specific payload designs and the related costs and schedule to meet the program requirements.
- b. Determine the sensitivity of design, costs, and schedules to changes in requirements.
- c. Perform trade studies to identify optimal designs.
- d. Develop standardized designs using a data base consisting of standardized equipment.
- e. Identify low cost designs using a data base consisting of off-the-shelf equipment.
- f. Use current Model to establish mathematical relationships within and between performance, safety, cost, and schedule without the use of a discrete data base.
- g. Perform modularity studies by modifying the Model to assign equipment to modules.

The Model can readily be expanded in its scope to perform many other studies as well.

The computer program aids the designer in performing trade studies and simplifies the achievement of a balanced system design. The

Model will become a more versatile tool in terms of preliminary program planning and in actual program management as it becomes more fully developed.

17. RECOMMENDATIONS

The principal goal of this study, which was to identify and quantify the interrelationships between and within the performance, safety, cost, and schedule parameters for unmanned, automated payloads has been met. The cost/performance methodology developed during the preceding fiscal year's study has been refined and substantially improved. The application of the methodology to unmanned, automated payloads resulted in the current System Cost/Performance Model. The Cost/Performance Model has been implemented as an operational digital computer program. Having achieved these objectives, the following recommendations are made with respect both to improving the Model and verifying and validating it.

It is recommended that the Model be thoroughly verified and validated. The most useful validation procedure would be to use the Model on test cases selected from historical programs, operational programs, and new starts. Historical and current programs provide the most accurate data by which to validate the model. New start programs will test the applicability of the model as a preliminary planning tool.

Although the Model is operational, there are a number of improvements to the Model which should be implemented. The suggested improvements are listed below for applicable subsystem, reliability, cost, and schedule models:

- a. Subsystem Models
 1. Stabilization and Control
 - (a) Refine the disturbance torque portion of the model.
 - (b) Incorporate a magnetic torquer in the model.
 - (c) Improve the detail and accuracy of Dual Spin, Yaw Spin, and Three-Axis Mass Expulsion configurations.

2. Data Processing
 - (a) Refine the algorithm for selecting General Purpose Processors.
 - (b) Incorporate data compression in General Purpose Processors.
 - (c) Incorporate a tape recorder in the model.
 - (d) Incorporate an algorithm for selecting Command Distribution Units.

3. Communication
 - (a) Expand the model from the Air Force's Space Ground Link System (SGLS) to include NASA's Unified S-Band (USB), S-Band and VHF equipment.
 - (b) Expand the model to apply to interplanetary missions.

4. Electrical Power
 - (a) Refine the combined electrical/thermal relationships.

5. Structures
 - (a) Incorporate the truss structural configuration.
 - (b) Incorporate effects of strap-on solid kick stages.

6. Vehicle Sizing
 - (a) Incorporate provision for rotation of the vehicle relative to the normally defined axis.

7. Thermal Control
 - (a) Incorporate the effect of duty cycle in the model.

- b. Reliability
 1. Incorporate mission equipment in the model with provision for increasing reliability of the mission equipment.
 2. Incorporate a model of pulse-operation (short duration) modules.

3. Delete selection of redundancy based on Mean Mission Duration of system. (Selection based on system reliability is substantially faster and provides similar results in most cases.)
- c. Cost
1. Improve the accuracy and applicability of the data base and CERs based on additional data.
 2. Develop CERs for equipment not previously flown.
 3. Model the relationship between cost and schedule.
- d. Schedule
1. Improve the approach and accuracy of the model by collecting and processing additional schedule data.

In order to make the above improvements, it should be clear that additional cost, schedule, and technical data must be collected and processed. The focus of the current study was on developing a model rather than augmenting a data base. Only after the model was successfully developed and proven as a useful tool could data collection be justified at such a detailed level. On the other hand, lack of adequate data hindered the development of the current model. The Cost Model must be considered preliminary, and the Schedule Model cannot be considered operational until sufficient data have been collected to improve and validate the model. Hence, widespread use of the Systems Cost/Performance Model depends entirely on the collection of performance, safety, cost, and schedule data at the subsystem component (assembly) level.

It is recommended that the fiscal year 1975 effort include extension of the model to other space vehicle systems; improvement of the data base to be acceptable for performance, safety, cost, and schedule analyses; testing of the capability of the model to predict space vehicle interrelationships; and a user review to evaluate the potential of the model to assist in programmatic change control such as configuration management.

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NOMENCLATURE

Section 3

a_x, a_y, a_z	misalignment errors in mounting inertial measurement units relative to the vehicle x, y, and z axes, respectively (deg, 3σ)
b_1	drive quantization error in the Gimbal Drive Assembly (deg, 3σ)
b_2	gimbal drive error in the Gimbal Drive Assembly (deg, 3σ)
b_3	biax droop error in the Gimbal Drive Assembly (deg, 3σ)
C	value of the numerical integration term in the filtered roll horizon scanner noise power
c_1	programmer sine wave error in the Control Timing Assembly (deg, 3σ)
c_2	drive quantization and delay error in the Control Timing Assembly (deg, 3σ)
c_3	measurement compensation error in the Control Timing Assembly (deg, 3σ)
c_4	pipper drift error in the Control Timing Assembly (deg, 3σ)
c_5	quantization noise error in the Control Timing Assembly (deg, 3σ)
c_6	controller error in the Center Electrical Assembly (deg, 3σ)
D	vehicle diameter (ft)
d_T	distance from C. G. to main engine (ft)
d_x, d_y, d_z	gas jet lever arms on the roll, pitch and yaw axis, respectively (ft)

NOMENCLATURE (Continued)

db	deadband (deg)
db_x, db_y, db_z	deadbands on the roll, pitch and yaw axes, respectively (deg)
d_1	bearing and motor friction error in the Despin Mechanical Assembly (deg, 3σ)
d_2	bearing runout error in the Despin Mechanical Assembly (deg, 3σ)
E	antenna elevation angle (rad)
e	horizon scanner anomalies (deg, 3σ)
e_{AR}	attitude reference error (deg, 3σ)
e_{HS}	horizon scanner error (deg, 3σ)
e_a	antenna misalignment (deg, 3σ)
e_{beam}	antenna beam pointing error (deg, 3σ)
e_p	programmed pitchover rate error (deg/sec, 3σ)
e_1	horizon sensor noise (deg, 3σ)
e_2	horizon sensor radiance irregularity (deg, 3σ)
e_3	horizon sensor quantization error (deg, 3σ)
e_4	horizon sensor sun interference error (deg, 3σ)
e_5	horizon sensor moon interference error (deg, 3σ)
e_6	horizon sensor threshold aging error (deg, 3σ)
e_7	horizon sensor null or bias error (deg, 3σ)
F	gas jet force (assumed the same on all axes) (lb)
F_e	main engine thrust (lb)

NOMENCLATURE (Continued)

F_{\max}	maximum gas jet output thrust per axis (lb)
F_{\min}	minimum gas jet output thrust per axis (lb)
g_1	G-insensitive/gyro drift error, 24-hour stability (deg, 3σ)
g_2	total misalignment of the Rate Integrating Gyro Assembly relative to the vehicle (deg, 3σ)
H	momentum of the vehicle about the yaw axis for a yaw spin vehicle (ft-lb-sec)
$H_{F\phi}$	roll feedback gain (sec^{-1})
$H_{F\psi}$	roll to yaw coupling gain (sec^{-1})
$H_{F\theta}$	pitch feedback gain (sec^{-1})
$H_W(\text{nom})$	nominal value of the angular momentum of the reaction wheel (ft-lb-sec)
$H_W(\text{max})$	maximum value of the angular momentum of the reaction wheel (ft-lb-sec)
$H_W(\text{min})$	minimum value of the angular momentum of the reaction wheel (ft-lb-sec)
H_ϕ	roll horizon scanner gain to roll axis (sec^{-1})
H_ψ	roll horizon scanner gain to yaw axis (sec^{-1})
H_θ	pitch horizon scanner gain (sec^{-1})
\underline{h}	actual momentum of the individual CMGs selected from the hardware data base (ft-lb-sec)
h_{cap}	maximum momentum capability of the set of CMGs (ft-lb-sec)
h_{dist}	disturbance momentum requirement (ft-lb-sec)

NOMENCLATURE (Continued)

h_{man}	maneuver momentum requirement (ft-lb-sec)
h_{min}	minimum of the required momentum about the three axes (ft-lb-sec)
h_{req}	momentum storage requirement (ft-lb-sec)
$h_{req_x}, h_{req_y}, h_{req_z}$	momentum required on the roll, pitch, and yaw axes, respectively (ft-lb-sec)
h_1	resolver accuracy in the Gimbal Electronics Assembly (deg, 3σ)
J	inertia about the maneuver axis (slug-ft ²)
J_P	platform spin axis inertia (slug-ft ²)
J_R	rotor spin axis inertia (slug-ft ²)
J_x, J_y, J_z	vehicle roll, pitch and yaw inertias (slug-ft ²)
K	= 0 if errors are desired for spin axis relative to nadir, =1 if errors are desired for payload relative to nadir (dimensionless)
K_{sf}	gyro scale factor error, 24-hour stability (dimensionless)
K_1	horizon scanner output noise level of the noise power spectra (dimensionless)
K_2	product of the peak gimbal rate and the maximum maneuver rate (deg/sec ²)
$M_D(max)$	maximum disturbance torque (ft-lb)
$M_{D_x}(steady)$	} steady environment disturbance torque level about the vehicle roll, pitch and yaw axes, respectively (ft-lb)
$M_{D_y}(steady)$	
$M_{D_z}(steady)$	

NOMENCLATURE (Continued)

N	number of days between corrections in the spin axis pointing of the vehicle
N_m	average star density (steradian ⁻¹)
N_x, N_y, N_z	number of maneuvers about the roll, pitch, and yaw axes, respectively
n	number of skewed single-gimbaled control moment gyros
P	standby power of the CMG configuration (watts)
$P_{\phi N}, P_{\theta N}$	filtered roll and pitch horizon scanner noise power
$S_{\phi}(\omega), S_{\theta}(\omega)$	roll and pitch horizon scanner output noise power spectra
s_1	type of star sensor = 1 for star mapper 2 for body fixed (electronic) star tracker 3 for gimbaled star tracker
s_2	star sensor accuracy (deg, 3σ)
s_3	star mapper field of view (deg ²)
s_4	star mapper sensitivity (visual magnitude)
T	mission lifetime (months)
T_{accel}	acceleration torque requirement (ft-lb)
T_{dist}	disturbance torque requirement (ft-lb)
T_{max}	peak torquer torque which must be delivered at peak gimbal rate for a CMG configuration (ft-lb)
\bar{T}_{max}	actual peak torquer torque delivered by a CMG (ft-lb)
T_{req}	vehicle torque requirement which consists of the sum of the acceleration torque and the disturbance torque requirements (ft-lb)

NOMENCLATURE (Continued)

t	main engine burn time (sec)
t_{accel}	acceleration time for maneuvering (sec)
t_{hold}	time vehicle must be in inertial hold (min)
t_{ℓ}	time between unloading wheel momentum (days)
t_{max}	maximum time the gyros can be allowed to drift during inertial hold without exceeding the maximum pointing error requirement (sec)
t_{sc}	average time between star crossings (sec)
V	volume of a CMG configuration (ft^3)
W	weight of a CMG configuration (lb)
α	thruster offset angle in the roll-yaw plane for a Pitch Momentum Wheel configuration (deg)
Δd	lateral vehicle c.g. distance plus lateral thrust chamber c.g. distance from the reference axis (ft)
$\Delta db_x, \Delta db_y, \Delta db_z$	roll, pitch and yaw deadband tolerances, respectively (deg, 3σ)
Δt	minimum gas jet on time (sec)
$\Delta \theta_{\text{avg}}$	average body rate for low orbit during period when high accuracy is required (deg/sec)
η	control system efficiency
γ	CMG skew angle (deg)
ω_H	maximum horizon sensor output frequency (rad/sec)
ω_R	spin rate of rotor in a Dual Spin configuration (rpm)

NOMENCLATURE (Continued)

$\omega_W^{(max)}, \omega_W^{(min)}, \omega_W^{(nom)}$	maximum, minimum, and nominal reaction wheel spin speed (rpm)
ω_o	true orbital pitchover rate (rad/sec)
ω_{op}	programmed pitchover rate (rad/sec)
ω_s	spin rate about the z-axis for a Yaw Spin vehicle (rad/sec)
$\omega_x, \omega_y, \omega_z$	roll, pitch and yaw inertial rates, respectively (rad/sec)
φ	vehicle roll Euler angle (deg)
φ_{HS}	output from the roll channel of the horizon scanner (deg)
φ_s	roll horizon scanner error (deg, 3σ)
ψ	yaw Euler angle (deg)
σ	one-sigma value of a statistical distribution
σ_{HS}	one-sigma value of the horizon scanner noise
$\dot{\sigma}_{max}$	maximum gimbal rate requirement (rad/sec)
$\dot{\sigma}_{max}$	actual peak gimbal rate of a CMG (rad/sec)
τ_x, τ_y, τ_z	time that the roll, pitch, and yaw disturbance torques are in effect for a mission (sec)
θ	pitch Euler angle (deg)
θ_{CMG}	CMG angular error (deg, 3σ)
θ_{FOV}	maximum range of attitude freedom (full FOV) required to track specific guide stars over a wide range of vehicle motion (deg)

NOMENCLATURE (Continued)

θ_{HS}	pitch horizon scanner output (deg)
θ_R	minimum required system pointing accuracy (deg, 3σ)
$\dot{\theta}_{R_x}, \dot{\theta}_{R_y}, \dot{\theta}_{R_z}$	required system pointing accuracy about the roll, pitch, and yaw axes (deg, 3σ)
θ_s	pitch horizon scanner error (deg, 3σ)
θ_x	total pointing error for a Yaw Spin vehicle (deg, 3σ)
θ_e	total system attitude error for a CMG system (deg, 3σ)
θ_{e_b}	gyro bias drift error (deg, 3σ)
$\theta_{e_{sf}}$	gyro scale factor error (deg, 3σ)
$\dot{\theta}_{CMG}$	CMG angular rate error (deg/sec, 3σ)
$\dot{\theta}_D$	gyro drift rate (same as g_1) (deg/sec, 3σ)
$\dot{\theta}_{R_x}, \dot{\theta}_{R_y}, \dot{\theta}_{R_z}$	required system rate accuracy about the roll, pitch and yaw axes (deg/sec, 3σ)
$\dot{\theta}_{ST(max)}$	maximum vehicle rate at which star information must be obtained (deg/sec)
$\dot{\theta}_{avg}$	average body rate for low orbit during the period when high accuracy is required (deg/sec)
$\dot{\theta}_b$	gyro bias drift rate (deg/sec, 3σ)
$\dot{\theta}_{max}$	maximum maneuver rate (deg/sec)
$\dot{\theta}_o$	maximum initial rate, assumed the same on all axes (deg/sec)
$\dot{\theta}_x$	rate error about the roll axis in the Yaw Spin configuration, (deg/sec, 3σ)

NOMENCLATURE (Continued)

$\dot{\theta}_x(\text{max}), \dot{\theta}_y(\text{max}),$ $\dot{\theta}_z(\text{max})$	$\left. \vphantom{\dot{\theta}_x(\text{max}), \dot{\theta}_y(\text{max}), \dot{\theta}_z(\text{max})} \right\}$	maximum maneuver rates about the roll, pitch and yaw axes, (deg/sec, 3σ)
$\dot{\theta}_e$		total system rate, error for a CMG system (deg/sec, 3σ)

Section 4

$C_d A$		effective flow area (in ²)
F_{min}		minimum blowdown thrust (lb)
I_t		total impulse (lb-sec)
MR		mixture ratio (lb oxidizer/lb fuel)
P_{reg}		regulator set pressure (psia)
P_{ti}		thruster inlet pressure (psia)
R		flow resistance (sec ² /in-lb)
V		propellant volumes (in ³)
V_{pr}		regulated pressurant volume (in ³)
V_{prt}		pressurant tank volume (in ³)
W_f		fuel weight (lb)
W_o		oxidizer weight (lb)
W_p		propellant weight (lb)
W_{pr}		pressurant weight (lb)
ΔP_{filt}		filter pressure drop (psi)
ΔP_{iso}		isolation valve pressure drop (psi)
ΔP_1		line pressure drop (psia)

NOMENCLATURE (Continued)

\dot{w}	flow rate (lb/sec)
\dot{w}_{pr}	maximum flow rate (lb/sec)
ρ	propellant density (lb/in ³)
<u>Section 6</u>	
B	10 log bandwidth (Hz) (dBW)
D	diameter of the antenna parabola (ft)
ERP	effective radiated power (dBW)
F	frequency (MHz)
G_R	ground receiving (downlink) antenna gain (dB)
G/T	gain-to-temperature ratio of receiving system (dBW)
G_T	antenna gain (dB)
K	Boltzman's constant (-228.6 dBW/Hz/°K)
L	other losses (dBW)
M	margin (dBW)
NF _{dB}	noise figure (dB)
P _w	transmitter power (watts)
S	transmission range (m)
SL	space loss (dBW)
S/N	signal-to-noise ratio (dBW)
T	system noise temperature (°K)
λ	wavelength (m)

NOMENCLATURE (Continued)

Section 7

A	array area (m^2)
C_A	capacity required at end of mission (amp-hr)
C_{cell}	capacity of selected cells (amp-hr)
C_I	minimum installed battery capacity required (amp-hr)
C_R	minimum required capacity (amp-hr)
C_T	total actual installed capacity (amp-hr)
F_S	array sizing factor (dimensionless)
F_W	array weight factor (kg/m^2)
h_e	earth radius (km)
h_p	orbit perigee (km)
I_{CH}	charge current delivered by regulator (amps)
K_1	battery packing factor (dimensionless)
K_2	battery structure weight factor (dimensionless)
N_B	number of batteries required in parallel
N_C	number of cells in series
N_D	number of discharge regulators
P_C	array power allocated for charging (watts)
P_{CD}	power dissipated by charger (watts)
P_{CH}	battery recharge power (watts)

NOMENCLATURE (Continued)

P_D	battery discharge power during eclipse (watts)
P_{DD}	power dissipated by discharge regulator (watts)
P_L	average load power (watts)
P_{Lmin}	minimum load power (watts)
P_{LR}	array power allocated to the load (watts)
P_{LRD}	power dissipated by load regulator (watts)
P_{max}	maximum array power (watts)
P_S	EOL array output (watts)
P_{SRD}	power dissipated by shunt regulator (watts)
R_{FD}	battery temperature degradation factor (dimensionless)
S	average solar intensity (1353 W/m^2)
S_e	angular size of earth's shadow (radians)
T_E	eclipse period (hr)
T_S	sunlight period (hr)
V_B	unit battery volume (m^3)
V_{BM}	minimum allowable battery voltage (V dc)
V_{BT}	total battery volume (m^3)
V_C	minimum allowable cell voltage (V dc)
V_{cell}	volume of each cell (m^3)
V_{DB}	average battery discharge voltage (volts)

NOMENCLATURE (Continued)

W_A	array weight (kg)
W_B	unit battery weight (kg)
W_{BT}	total battery weight (kg)
W_{cell}	weight of each cell (kg)
Δ_F	coverglass and coverglass adhesive transmissivity loss factor (dimensionless)
Δ_I	array fabrication loss factor (dimensionless)
Δ_M	miscellaneous loss factor (dimensionless)
Δ_R	radiation degradation factor (dimensionless)
Δ_T	temperature adjustment factor (dimensionless)
η_C	charge regulator efficiency (dimensionless)
η_{CA}	battery amp-hr charge efficiency (dimensionless)
η_D	discharge regulator efficiency (dimensionless)
η_E	battery watt-hr charge efficiency (dimensionless)
η_I	solar cell efficiency, at 28°C, AMO illumination (dimensionless)
η_{LR}	load regulator efficiency (dimensionless)
η_R	power distribution loss factor (array to loads)
λ_D	average depth of discharge (at end of discharge)
λ_{DM}	maximum depth of discharge (dimensionless)
λ_G	array orientation factor (dimensionless)
λ_P	solar array packing factor (dimensionless)

NOMENCLATURE (Continued)

Section 8

α	solar absorptivity of the surface (dimensionless)
ϵ	emissivity of the surface (dimensionless)

Section 9

EQWT	equipment weight carried by the structure (lb)
K	density coefficient (dimensionless)
L/D	length/diameter ratio of the structural shape (dimensionless)

Section 10

A_f	frame stiffener area (in ²)
A_s	longitudinal stiffener area (in ²)
a	frame spacing (in.)
b	longitudinal stringer spacing (in.)
b_f	frame cross-section height (in.)
b_s	stringer cross-section height (in.)
c_a	number of "g" accelerations in axial direction
c_e	number of "g" accelerations in lateral direction
D	cylinder diameter (in.)
F	total force on end cover or midsection bulkhead (lb)
f	proportion of satellite weight on end covers or midsection bulkhead (dimensionless)

NOMENCLATURE (Continued)

I_f	area moment of frame stiffener (in ⁴)
k_c	correction factor for critical stress (dimensionless)
L	systems equipment bay length (in.)
l_e	mission equipment bay or solar array extension boom length (in.)
M	applied moment to systems equipment bay (in.-lb)
M_a	applied moment to center line extension boom (in.-lb)
M_ℓ	applied moment to lateral extension boom (in.-lb)
N	stress resultant (lb/in.)
P	axial load applied to systems equipment bay (lb)
P_a	axial load applied to center line extension boom (lb)
R	cylinder radius (in.)
r	extension boom radius (in.)
r_b	ratio of stringer height and stringer spacing
r_t	ratio of stringer thickness and skin thickness
T	lateral shear load applied to systems equipment bay (lb)
t	skin thickness of stiffened shell (in.)
\bar{t}	equivalent thickness of stiffened shell (in.)
t_a	aft cover thickness (in.)
t_e	forward cover thickness (in.)
t_f	frame stiffener thickness (in.)
t_i	midsection bulkhead thickness (in.)

NOMENCLATURE (Continued)

t_m	monocoque shell thickness (in.)
t_s	longitudinal stiffener thickness (in.)
t_w	tubular wall thickness of extension boom (in.)
W	total satellite weight (lb)
W_e	mission equipment weight (lb)
w	width of systems equipment bay box structure (in.)
Z_L	curvature parameter of monocoque cylinder (dimensionless)
α	panel efficiency coefficient (dimensionless)
β	critical stress correction factor for combined bending and axial compression of plates
ν	Poisson's ratio (dimensionless)
w_e	uniform load on midsection bulkhead (psi)
ρ	material weight density (lb/in ³)
ρ_f	frame stiffener radius of gyration (in.)
ρ_p	radius of gyration per unit width of stiffened panel (dimensionless)
Σ	solidity (dimensionless)
σ	applied stress (psi)
$\bar{\sigma}$	optimum stress (psi)
σ_{cr}	critical stress for monocoque plate (psi)
σ_m	critical stress for monocoque shell (psi)
σ_p	critical stress for stiffened panel (psi)

NOMENCLATURE (Continued)

σ_s	critical stress for longitudinal stringer (psi)
σ_w	critical stress for tubular wall (psi)
σ_y	yield stress (psi)
 <u>Section 11</u>	
$P_D(t)$	probability of detection (dimensionless)
$P(M)$	probability that the monitoring system is functioning properly at the time of the failure (dimensionless)
$Pr(k)$	probability that module L is in the kth operability state (dimensionless)
$P(X)$	probability that the out-of-specification parameter was monitored by the failure detection system subsequent to the failure (dimensionless)
Q	effective dormancy factor (dimensionless)
q	dormancy factor (dimensionless)
$R[L, T(I)]$	reliability of module L at time T (I) (dimensionless)
$R_M(t)$	reliability (probability of successful operation to time t) of the entire monitoring system (dimensionless)
$R_s [T(I)]$	reliability of a single system at time T (I) (dimensionless)
$R(t)$	spacecraft reliability function (dimensionless)
t	time (sec)
λ	translated failure rate (failures/sec)
$\bar{\lambda}$	effective active failure rate, when duty cycled (failures/sec)

NOMENCLATURE (Continued)

λ_{M_F}	portion of the monitoring system failure rate which is linked to a false indication of failure being generated by the sensor (failures/sec)
λ_{M_X}	failure rate of that portion of the monitoring system assigned to parameter X, one of n parameters monitored (failures/sec)
λ_{S_F}	portion of the switch failure rate that is linked to a change of state without a command from the monitor subsystem (failures/sec)
μ	mean (sec)
σ^2	variance (sec ²)

Section 13

A	component state-of-art factor (dimensionless)
A_s	subsystem state-of-art factor (dimensionless)
a	constant exponential power for component development lead time (dimensionless)
C	direct DDT&E charges allocated by component (dollars)
C_b	base value component costs (dollars)
C_s	subsystem direct DDT&E cost (dollars)
C_{sb}	base value subsystem costs (dollars)
c_0	component development lead time constant (months)
c_1	component qualification lead time constant (months)
c_2	constant multiplier for component qualification (months/dollar)
c_3	subsystem development lead time constant (months)
c_4	subsystem qualification lead time constant (months)

NOMENCLATURE (Continued)

c_5	constant multiplier for subsystem qualification (months/dollar)
d	constant exponential power for subsystem development (dimensionless)
i	i th component of N components (dimensionless)
k_c	constant for component type (months/dollar)
k_f	constant which depends upon the extent of system testing required (dimensionless)
n	average redundancy for subsystem, active plus standby strings (dimensionless)
α	constant exponential power for subsystem development lead time (dimensionless)

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GLOSSARY

APS	Auxiliary Propulsion Subsystem
APT	Automatic Picture Transmission
ATS	Applications Technology Satellite
AVCS	Automatic Vidicon Camera Subsystem
BBAU	Baseband Assembly Unit
CCU	Central Control Unit
CDPI	Communications, Data Processing and Instrumentation
CER	Cost Estimating Relationship
CLER	Clerical
CMG	Control Moment Gyros
COM	Communications
DDT&E	Design, Development, Test and Evaluation
DEV	Development
DOD	<u>Department of Defense</u>
DPES	Data Processing Electronics Subsystem
DSCS-II	Defense Satellite Communications System-II
DSP	Defense Satellite Program
DTU	Digital Telemetry Unit
ENG	Engineering
EO	Earth Observation Mission
EOS	Earth Observatory Satellite
ERP	Effective Radiated Power
ERTS-A	Earth Resources Technology Satellite-A
FOV	Field of View
FSK	Frequency Shift Key
FV	Number of Flights Vehicles

G&A	General and Administrative Expense
GRA	Gyro Reference Assembly
GSE	Ground Support Equipment
HEAO	High Energy Astronomy Observatory
HLTWT	High Level Traveling Wave Tube
LST	Large Space Telescope
LUN	Lunar Mission
LV/OP	Local Vertical/Orbit Plane
MDP	Mission Data Processing
ME	Mass Expulsion
ME _R	Mission Equipment DDT&E Cost
ME _U	Mission Equipment Average Unit Cost
ME _W	Mission Equipment Weight
MFG	Manufacturing
MMD	Mean Mission Duration
MUX	Multiplexer
NA	Not Available
NCHOSE	Vectors Specifying the Equipment Quantities
OAO	Orbiting Astronomical Observatory
OGO	Orbiting Geophysical Observatory
OSO-I	Orbiting Solar Observatory-I
OSR	Optical Surface Reflectors
PCM	Phase Change Material
PCU	Power Control Unit
PLN	Planetary Mission
PM	Phase Modulation
PRN	Pseudo Random Noise
PSK	Phase Shift Key
QC	Quality Control
QV	Number of Qualification Vehicles

RF	Radio Frequency
RSS	Root-Sum-Square
S&C	Stabilization and Control
SGLS	Air Force Space Ground Link System
TCS	Thermal Control Subsystem
TDP	Telemetry Data Processing
TLG	Tooling

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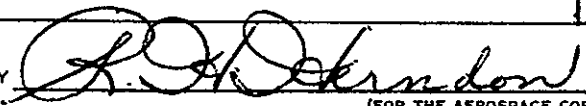
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