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# FINAL REPORT

# LARGE SPACE STRUCTURE EXPERIMENTS

# FOR AAP

## VOLUME II

## ANALYSIS AND EVALUATION OF SPACE STRUCTURE CONCEPTS

REPORT NO. GDC-DCL67-009

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# FOREWORD

The purpose of this report is to present the results of a study of "Large Space Structure Experiments for AAP" conducted by the Convair division of General Dynamics for the Marshall Space Flight Center, NASA. The study was performed during the interval 15 September 1966 to 15 September 1967, at a level of approximately $275,000.00, under Contract NAS 8-18118. The final report is published in five volumes as follows:

Volume I Technical Summary

This volume summarizes the results of the entire study.

Volume II Analysis and Evaluation of Space Structure Concepts

This volume presents the results of the analysis of the 40 space structure concepts analyzed during the first half of the study.

Volume III Crossed H Interferometer for Long Wave Radio Astronomy

This volume contains the design details of the crossed H interferometer that was one of the three concepts selected at mid-term for detailed analysis.

Volume IV Focusing X-Ray Telescope

This volume contains the design details of the focusing x-ray telescope that was one of the three concepts selected at mid-term for detailed analysis.

Volume V 100-Foot Parabolic Antenna

This volume contains the design details of the parabolic antenna that was one of the three concepts selected at mid-term for detailed analysis.

ACKNOWLEDGMENT

Orientation and technical guidance have been provided by the NASA-COR, Mr. W. T. Carey of MSFC. The completeness of the study has been enhanced in the area of crew systems through the coordination of Mr. N. Belasco of MSC. Many companies, listed in Section 1, have submitted concepts for analysis and evaluation. Members of the scientific community, particularly the astronomers, have contributed to the establishment of user requirements and are recognized in Volume I. Key Convair study members are listed below:

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TABLE OF CONTENTS


| Section | | | Page |
|---|---|---|---|
| 1 | | INTRODUCTION | 1-1 |
| | 1.1 | Background | 1-1 |
| | 1.2 | Study Approach | 1-3 |
| | 1.3 | Summary of Results | 1-4 |
| | 1.4 | Introduction to Volume II | 1-5 |
| 2 | | LONG WAVE RADIO ASTRONOMY | 2-1 |
| | 2.1 | User Requirements | 2-1 |
| | 2.2 | Analysis of Provided Concepts | 2-2 |
| | 2.2.1 | Concept 2a — Broadside Dipole (Brown Engineering Co. Technical Note AS-2-16) | 2-2 |
| | 2.2.2 | Concept 2b — Rhomic (Brown Engineering Company Technical Note AS 2-16) | 2-10 |
| | 2.2.3 | Concept 3 — 1500-Meter Paraboloid Antenna Astro-Research Corp., ARC-R-205,224) | 2-17 |
| | 2.2.4 | Concept 6, 600-Meter Rhombus (NASA/MSFC) | 2-22 |
| | 2.2.5 | Concept 7 — Broadside Dipole — (Reference Ryan Report No. 29174-45) | 2-29 |
| | 2.2.6 | Concept 13, ORAO Rhombus Antenna (TRW) | 2-33 |
| | 2.2.7 | Concept 17 — Log Periodic Antenna (Fairchild-Hiller) | 2-37 |
| | 2.2.8 | Concept 20 — Rhombus Antenna (Fairchild-Hiller) | 2-40 |
| | 2.3 | Analysis of New Concepts | 2-44 |
| | 2.3.1 | Concept 51a — Colinear Dipole Antenna | 2-44 |
| | 2.3.2 | Concept 51b, Crossed-H Inteferometer | 2-50 |
| | 2.3.3 | Concept 52a, Rhombic, External Structure | 2-58 |
| | 2.3.4 | Concept 52b, Rhombic Antenna, Internal Structure | 2-61 |
| | 2.3.5 | Concept 53, End-Fire Array | 2-68 |
| | 2.4 | Thermal Analysis with Primary Emphasis on Tubular Frame Structures | 2-71 |
| | 2.5 | Electrical Power Performance Assessment | 2-73 |
| | 2.6 | Radio Astronomy Subsystems | 2-78 |
| 3 | | MILLIMETER WAVE RADIO ASTRONOMY | 3-1 |
| | 3.1 | Millimeter Wave Radio Astronomy User Requirements | 3-1 |
| | 3.2 | Analysis of Concepts | 3-3 |

TABLE OF CONTENTS, Contd


| Section | | | Page |
|---|---|---|---|
| 4 | X-RAY AND GAMMA-RAY ASTRONOMY | | 4-1 |
| | 4.1 | Gamma-Ray Astronomy User Requirements | 4-1 |
| | 4.1.1 | Objectives of Observational Gamma-Ray Astronomy | 4-1 |
| | 4.1.2 | Gamma-Ray Astronomy in the Range 0.1 - 30 Mev | 4-2 |
| | 4.1.3 | Gamma-Ray Astronomy in the Range Above 100 Mev | 4-2 |
| | 4.1.4 | Conclusions | 4-4 |
| | 4.2 | X-Ray Astronomy User Requirements | 4-5 |
| | 4.2.1 | X-Ray Proportional Counter Requirements | 4-5 |
| | 4.2.2 | X-Ray Grazing Incidence Telescope Requirements | 4-7 |
| | 4.3 | Analysis of Concepts | 4-10 |
| | 4.3.1 | Concept 71, X- and Gamma-Ray Detector | 4-11 |
| | 4.3.2 | Concept 72, X-Ray Detector | 4-17 |
| | 4.3.3 | Concept 73, Focusing Telescope for Soft X-Ray | 4-22 |
| 5 | COMMUNICATIONS | | 5-1 |
| | 5.1 | Communications Antennas User Requirements | 5-1 |
| | 5.2 | Analysis of Provided Concepts | 5-6 |
| | 5.2.1 | Concept 1, 60-Ft Parabolic Antenna (NASA Internal Note S-66-4) | 5-6 |
| | 5.2.2 | Concept 4 — Lenticular Satellite (Goodyear Report GER 11893) | 5-20 |
| | 5.2.3 | Concept 9 — Flattenable Truss Parabolic Antenna (Ryan Report 29174-47) | 5-30 |
| | 5.2.4 | Concept 10, 50-Ft Petal Parabolic Antenna (Ryan Report 29174-47) | 5-32 |
| | 5.2.5 | Concept 11 — Petal Parabola (TRW Report No. 5760.000) | 5-35 |
| | 5.2.6 | Concept 12 — TV Broadcast Satellite (RCA) | 5-43 |
| | 5.2.7 | Concept 15 — Expandable Scissor Truss Parabolic Antenna (Fairchild-Hiller) | 5-47 |
| | 5.2.8 | Concept 16 — Phased Array (Fairchild-Hiller) | 5-50 |
| | 5.3 | Analysis of New Concepts | 5-62 |
| | 5.3.1 | Concept 75 — Expandable Truss Parabolic Antenna | 5-62 |
| | 5.3.2 | Additional Concepts | 5-71 |

TABLE OF CONTENTS, Contd


| Section | | | Page |
|---|---|---|---|
| 6 | | SOLAR CELL ARRAYS | 6-1 |
| | 6.1 | Experiment User Requirements | 6-1 |
| | 6.1.1 | Communication Antenna Power | 6-1 |
| | 6.1.2 | Design Requirements Data | 6-5 |
| | 6.2 | Analysis of Provided Concepts | 6-6 |
| | 6.2.1 | Concept 5 - Large Solar Cell Array (Boeing Co.) | 6-6 |
| | 6.2.2 | Concept 8 - Ryan Deployable Solar Array | 6-16 |
| | 6.2.3 | Concept 19 - Fairchild Hiller Roll-Up Solar Cell Array | 6-27 |
| | 6.3 | Analysis of New Concepts | 6-32 |
| | 6.3.1 | Concept 81 - Solar Cell Array - Communications | 6-32 |
| | 6.3.2 | Concept 82 - Solar Cell Array, S-IVB Workshop | 6-36 |
| 7 | | MAGNETOMETERS | 7-1 |
| | 7.1 | Magnetometer User Requirements | 7-1 |
| | 7.2 | Analysis of Concept 91, Magnetometer Boom | 7-3 |
| | 7.2.1 | Structure and Mechanics | 7-3 |
| | 7.2.2 | Stress Analysis | 7-6 |
| | 7.2.3 | Crew and Equipment Functions | 7-6 |
| 8 | | METEOROID DETECTORS AND COLLECTORS | 8-1 |
| | 8.1 | Meteoroid Detector User Requirements | 8-1 |
| | 8.2 | Analysis of Provided Concepts | 8-2 |
| | 8.2.1 | Concept 14, Meteoroid Deep Space Satellite | 8-2 |
| | 8.2.2 | Concept 18, Concertina Meteoroid Detector (Fairchild Hiller) | 8-7 |
| | 8.3 | Concept 85, Meteoroid Detector and Collector Panels | 8-12 |
| | 8.3.1 | Structure and Mechanics | 8-13 |
| | 8.3.2 | Attitude Control and Dynamics | 8-13 |
| | 8.3.3 | Weights | 8-15 |
| | 8.3.4 | Crew and Equipment Functions | 8-15 |
| 9 | | EVALUATION OF CONCEPTS | 9-1 |
| | 9.1 | User vs. Concept Specifications | 9-1 |

TABLE OF CONTENTS, Contd


| Section | | | Page |
|---|---|---|---|
| | 9.1.1 | Bandwidth Concept Rating Method | 9-2 |
| | 9.1.2 | Aperture Concept Rating Method | 9-3 |
| | 9.1.3 | Beamwidth Concept Rating Method | 9-3 |
| | 9.1.4 | Structure Perimeter Deviation Rating Method | 9-3 |
| | 9.2 | Research, Development, Test, and Engineering | 9-31 |
| | 9.2.1 | Manufacturing Research and Development | 9-31 |
| | 9.2.2 | Critical EVA | 9-31 |
| | 9.2.3 | Prerequisite Orbital Experiments | 9-32 |
| | 9.2.4 | Engineering Test and Evaluation | 9-33 |
| | 9.2.5 | Contributions to Advanced Technology | 9-38 |
| | 9.3 | Cost and Schedules | 9-40 |
| | 9.3.1 | Methodology | 9-40 |
| | 9.3.2 | Ground Rules | 9-41 |
| | 9.3.3 | Results | 9-41 |
| | 9.4 | Evaluation of Concepts | 9-42 |
| | 9.4.1 | Key Descriptive Parameters | 9-42 |
| | 9.4.2 | Practicality Evaluation | 9-42 |
| | 9.4.3 | Technology Contribution | 9-42 |
| | 9.4.4 | Deployment Feasibility | 9-43 |
| | 9.4.5 | Schedule | 9-43 |
| | 9.4.6 | Costs | 9-43 |
| | 9.4.7 | Applicability Evaluation | 9-43 |
| | 9.4.8 | Results or Recommendations | 9-43 |
| 10 | | CONCEPT SELECTION | 10-1 |
| | 10.1 | Approach | 10-1 |
| | 10.2 | Commonality | 10-1 |
| | 10.3 | Convair Recommendations | 10-1 |
| | 10.4 | Selection | 10-4 |

| Appendix | | |
|---|---|---|
| A | RELIABILITY ANALYSIS | A-1 |
| B | LAUNCH AND ORBITAL ENVIRONMENTAL CONSTRAINTS | B-1 |
| C | LAUNCH VEHICLE, SPACECRAFT, AND MISSION CONSTRAINTS AND CAPABILITIES | C-1 |
| D | CREW SYSTEMS CONSTRAINTS | D-1 |

# SECTION 1

# INTRODUCTION

1.1 BACKGROUND. The purpose of this study was to identify and define three large space structure experiments through which the following flight objectives could be accomplished: evaluate the role of man in the deployment, assembly, alignment, maintenance, and repair of large structures in space; evaluate the performance and behavior of large structures in space from a technology viewpoint; and provide a useful space structure that can be used to fulfill a "user oriented" requirement such as a radio astronomy antenna or solar cell array.

The logical point of departure for a study such as this is to first determine the most promising areas of science and technology that will probably require large structures in space. In viewing the potential NASA missions throughout the next decade, one can conclude that some of the more prominent requirements will evolve from the areas of astronomy, communications, and, to a lesser but still significant degree, from the requirements for solar cell arrays, micrometeoroid collectors, and magnetometers.

With regard to astronomy, regions of the electromagnetic spectrum that are of interest to astronomers begin with the very long radio waves and continue through the gamma ray region. Although not all astronomers necessarily agree on which areas of the spectrum should receive the highest priority, general concensus on those bands of particular interest and specific recommendations for future astronomy in space are found in "Space Research Directions for the Future, Part 2" (Woods Hole report). This document, together with other appropriate literature, were used to guide this study in its relation to astronomy. A few of the more important conclusions and recommendations contained in the Woods Hole report are summarized in Figure 1-1. The darkened area of the line at the top of the figure signifies those regions of the spectrum in which the atmospheric attenuation is greater than 10 db, and therefore, those regions that are essentially blacked-out from the earth's surface. In these regions, astronomical observations are completely dependent on the ability to go into space. Accordingly, the Woods Hole report has made positive recommendations for space astronomy covering the entire spectrum with the exception of that region shown as radar astronomy on the figure which, according to that report, can be satisfied by ground-based observations.

Several of the bands of interest are particularly challenging to anybody interested in large space structures. First, consider the very long wave (10 m and longer) region. Antennas designed to operate in this region have two dominant characteristics: large physical dimensions and correspondingly large allowable tolerances. The Woods Hole report recommended a "broad band antenna system leading to a 20 km aperture ultimate antenna system." Although such an antenna would not be feasible during the AAP

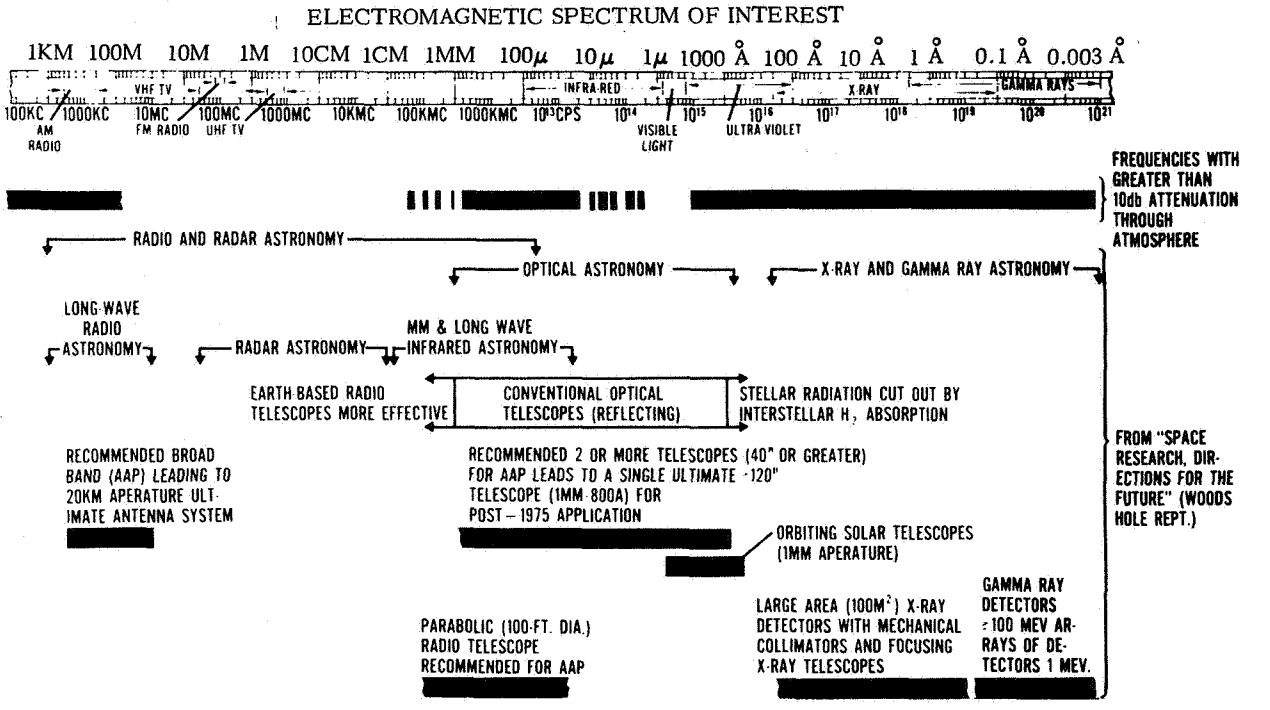



Figure 1-1. Potential Applications of Large Structures in Space

period, antenna types that may prove useful in the long wave region and, therefore, had to be accounted for in this study, include log periodics, rhombics, broadside arrays, and phased arrays. In the sub-millimeter region the primary useful antenna concept is a fairly large parabolic radio telescope with very stringent tolerance requirements. A great deal of emphasis was therefore placed in this region of interest. Additional regions of interest to astronomers, which involves large structures, are the x-ray and gamma-ray end of the spectrum. The Woods Hole report also contains specific large space structure requirements to support these astronomical programs.

In summary, Convair was directed by NASA to include in the study detailed analysis on four types of space structures relating to astronomy, longwave radio, sub-millimeter wave radio, and x- and gamma-ray astronomy. Although optical astronomy including infrared, ultraviolet, and visible regions of the spectrum are extremely important to future space flight, Convair was directed by NASA not to include these types of structures in the study.

In the area of communications many varying types of potential candidate missions exist including TV or voice broadcast type mission (of which there are numerous variations ranging from direct-to-home broadcast to simple point-to-point relay) and deep space relay. Large space structure requirements vary widely with the mission and the potential time frame for application. Although work is continuing to develop high power, long life space power supplies such as radioisotopes and nuclear reactors, there will always be a need for solar cell arrays; thus Convair was directed to include structures of this nature in the study. Additional areas that appeared to have the requirement for structures were magnetometer devices (the structural requirement emanates from the

need to separate the magnetometer a long distance from the mother spacecraft) and micrometeoroid collectors.

In summary, it was directed by NASA that the concepts to be analyzed in the study be centered around those satisfying the following user-oriented applications requirements:

Longwave radio astronomy
Millimeter wave radio astronomy
X- and gamma-ray astronomy
Communications
Solar cell arrays
Magnetometers
Micrometeoroid collectors

1.2 STUDY APPROACH. Figure 1-2 shows the major tasks to be accomplished during the study as directed by NASA during the contract orientation. The study is broken into two parts. The first half deals with the analysis of a large number of candidate space structure concepts and culminates in the selection of three which then undergo more detailed analysis and design during the second half of the study. The only difference between Task 1 and Task 2 is that Task 1 structures were provided to Convair by NASA or industry, whereas concepts analyzed in Task 2 were originated by Convair. It was desired by NASA that the concepts analyzed in Tasks 1 and 2 be representative of all candidate space structures required to fulfill potential NASA mission concepts and ideals throughout NASA and industry. A letter was prepared describing the purpose of the study and soliciting any concepts and ideas that the addressees might wish to submit for inclusion in the Task 1 analysis. Names of those companies solicited and

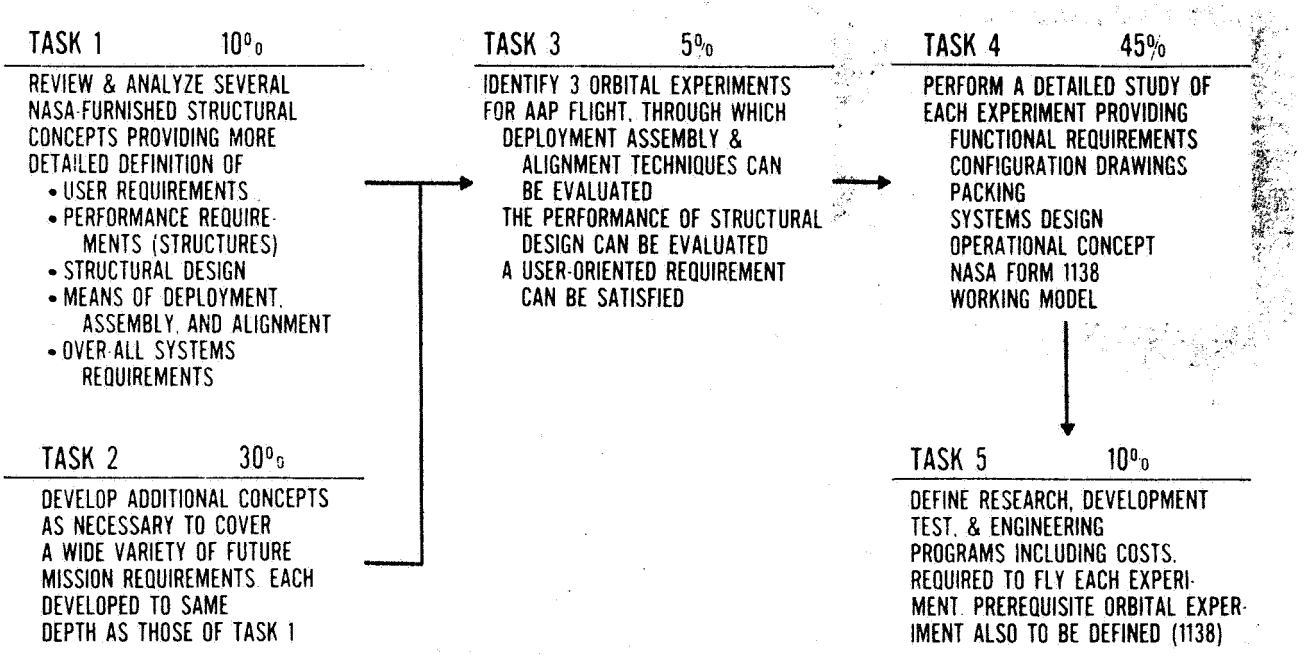



Figure 1-2. Task Areas

the nature of their replies are shown in Table 1-1. It can be seen from this table that an excellent response resulted. The successful performance of this study has been greatly enhanced by the fine cooperation of the many companies responding to this solicitation letter.

Table 1-1. Response to Request for Concepts

| NAME OF COMPANY | NATURE OF REPLY NONE | NEG. | POS. | DATA PROVIDED |
|---|---|---|---|---|
| AEROJET GENERAL CORPORATION | | | X | DATA/RECEIVED LATE |
| AEROSPACE CORPORATION | | | X | DISCUSSION |
| ASTRO RESEARCH CORPORATION | | | X | CONCEPT 3 (NASA PROVIDED) |
| BELL AEROSYSTEMS COMPANY | | | X | PERTINENT REPORT |
| BOEING COMPANY | | X | | CONCEPT 5 (NASA PROVIDED) |
| BROWN ENGINEERING | X | | | CONCEPTS 2A & 2B (NASA PROVIDED) |
| DOUGLAS AIRCRAFT COMPANY | | X | | |
| EMERSON ELECTRIC | | X | | |
| FAIRCHILD HILLER | | | X | CONCEPTS 15-20 & COST ESTIMATES |
| GENERAL ELECTRIC COMPANY | X | | | |
| GOODYEAR | | | X | CONCEPT 4 (NASA PROVIDED REPORTS & DATA) |
| GRUMMAN AIRCRAFT ENGINEERING CORP. | | | X | REFERENCE REPORTS |
| HUGHES | | X | | |
| LITTON INDUSTRIES | X | | | |
| LOCKHEED CALIFORNIA | | | X | REFERENCE REPORTS |
| LOCKHEED MISSILES AND SPACE | X | | | |
| MARQUARDT CORPORATION | | | X | PERTINENT REPORT |
| MARTIN COMPANY | X | | | CONCEPT 14 (NASA PROVIDED) |
| McDONNELL AIRCRAFT | | | X | REFERENCE REPORTS |
| NEOTEC | | | X | DATA WHEN COMPLETED |
| NORTH AMERICAN AVIATION | | X | | |
| NORTHROP SPACE LABORATORIES | X | | | |
| RADIO CORPORATION OF AMERICA | | | X | CONCEPT 12 |
| RYAN ELECTRONICS | | | X | CONCEPTS 7, 8, 9, 10 & ALIGN. TECHNIQUE |
| G. T. SCHJELDAHL | | | X | DATA |
| SANDERS ASSOCIATES | X | | | |
| TRW, INC. | | | X | CONCEPTS 11 & 13 |
| WHITTAKER/NARMCO | | | X | PERTINENT REPORT & DATA |

1.3 SUMMARY OF RESULTS. Tasks 1 and 2 resulted in the preliminary design and analysis of 40 candidate space structure concepts for flight in the 1970-75 time frame. Three of these structures were selected at the mid-term point of the study and were the subject of detailed preliminary design and analysis during the second half of the study. They are: a long wave radio astronomy antenna called a crossed H interferometer, a focusing x-ray telescope, and a 100-ft aperture parabolic antenna. An

overall summary of the entire study can be found in Volume I. Analysis of the 40 candidate structures is contained herein, and detailed analyses of the three selected concepts are contained in Volumes II, IV, and V, respectively.

1.4 INTRODUCTION TO VOLUME II. This volume presents the results of the work performed in Tasks 1, 2, and 3. Figure 1-3 shows a further breakdown of these major task areas. Some of the primary guidelines used during the study are listed below:

a. Concepts were considered for application in the AAP 1968 to 1972 time period and the 1972 to 1975 time period. The 1968 to 1972 baseline astronaut was assumed for 1972 to 1975 but requirements for improved capability was established.

b. Secondary emphasis was placed on solar cell arrays, magnetometers, and meteoriod collectors.

c. A prime contact for the user-oriented requirements in astronomy experiments was through coordination with the Orbital Astronomy Support Facility study during Task A.

d. In development of each experiment and the orbital support equipment concepts, maximum use was made of hardware elements and systems that are either available commerically or have been developed under other related programs.

e. Conventional equipment, procedures, and techniques were utilized whenever possible.

f. Each experiment was designed and packaged such that integration into the launch vehicle system prior to launch can be accomplished simply and easily.

g. Experiments were defined for minimum cost. Although weight, volume, and packaging efficiency were considered, minimum cost was the prevalent consideration.

h. Maximum utilization was made of anticipated experimental results of other foregoing programs such as Gemini and Apollo.

i. To the maximum extent feasible, the experiments were defined such that their performance was independent of any particular spacecraft configuration and mission profile.

In order to provide significant information for the study, extensive coordination with the scientific community, NASA, and industry was required, particularly in the areas of satellite application and crew systems. See Figure 1-4.

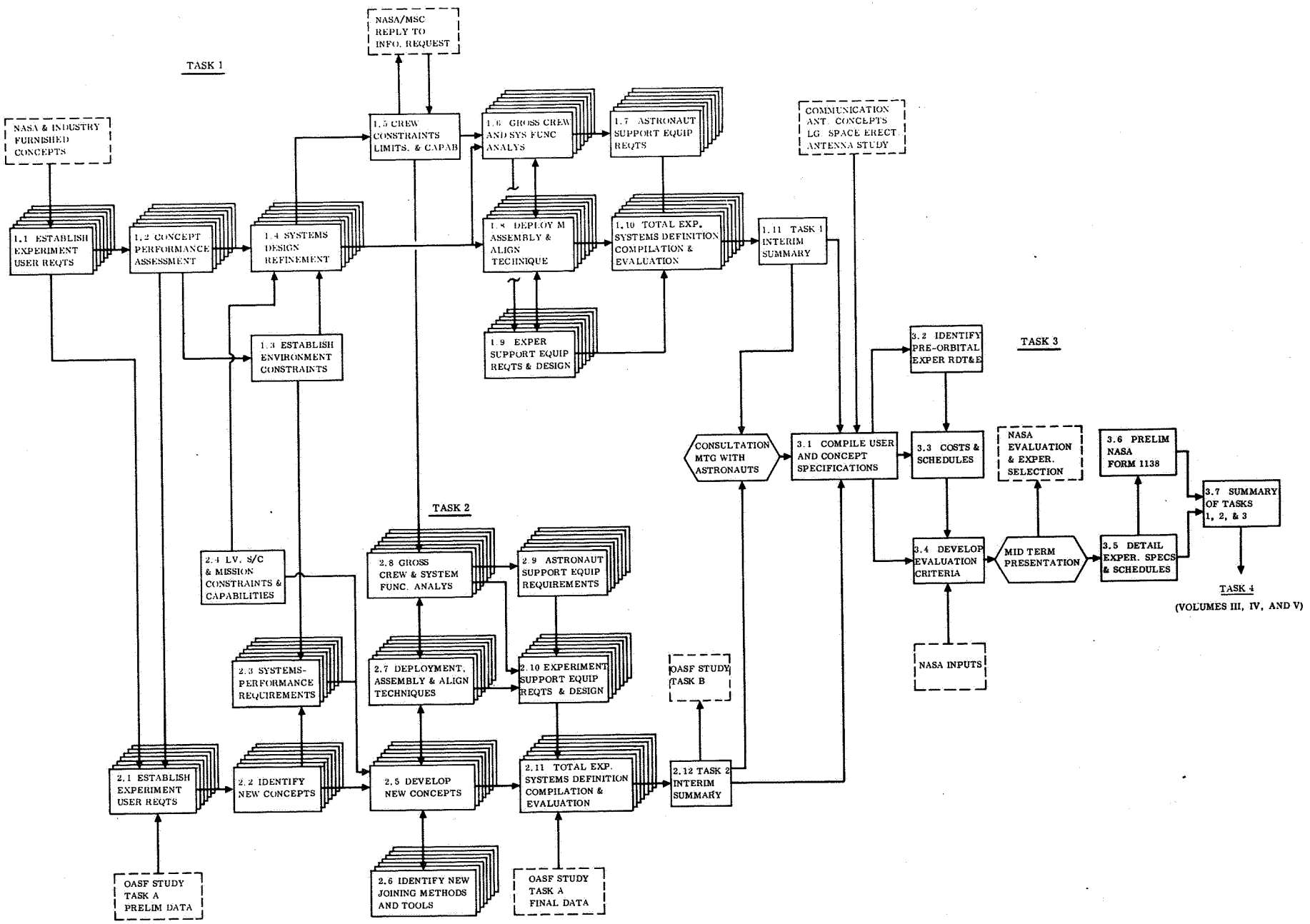



Figure 1-3. Large Space Structures Experiments for AAP Study Plan Block Diagram

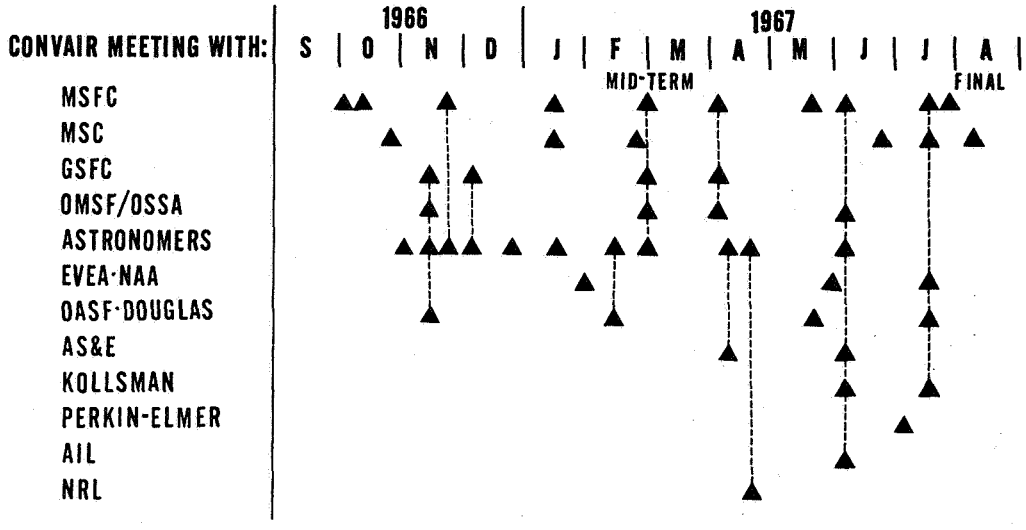



Figure 1-4. LSSE Coordination Meetings

Where the provided concept analysis is confirmed or the analysis has shown that there is no problem, that part of the detailed analysis has been left out in many cases.

The mechanical/structural evaluation criteria concern these specific topics:

a. Fabrication and availability.

b. Deployment and mechanisms.

c. Adaptability of mechanical/structural design to AAP missions (Saturn IB/V).

d. Credibility of mechanical/structural design relative to astronaut assistance.

Gross functional and time line analyses were performed for all of the manned concepts, providing an enormous bulk of data. However, these have been included only for the more promising concepts with respect to feasibility and performance.

The technical analysis also includes dynamics, thermodynamics, stress, mass properties, performance, human factors, and reliability.

## SECTION 2

## LONG WAVE RADIO ASTRONOMY

2.1 USER REQUIREMENTS. Most radio astronomy work that could be accomplished in the 1968-1974 time period is not sensitivity limited, but resolution limited. Hence, the general recommendation is to achieve some type of antenna interferometer array with a 2° beamwidth. For solar and planetary astronomy, far less resolving power is necessary, since these sources are expected to be very strong in comparison to the general noise background. Good resolution will be welcomed by the astronomy community, since to date poor resolution has been obtained with the dipole elements flown by Haddock, Huguenin and others. The RAE to be flown by Stone in 1967 will not provide significant resolving power. About 9° to 10° beamwidth at 1 MHz seems to be acceptable to astronomers providing that some interferometer arrangement could also be provided for additional resolution. The bandwidth should be as large as possible, with primary emphasis on the 0.5 - 10 MHz range. A capability to make polarization measurements should be provided. Specifically, the following parameters were considered:

| | |
|---|---|
| Life Time: | Minimum of 1 year desired. |
| Orbit Altitude: | Minimum of synchronous. |
| Effective Beamwidth: | 100 $deg^2$ at 1 MHz desired — less than 10° in one direction, but could be greater for solar and planetary astronomy. Interferometers should be used, if possible, for improving this resolution to 2°. |
| Pointing Accuracy: | One-half-beamwidth minimum to 1/10 for aspect determinations. In the case of a sweeping-mode or drift-mode antenna, the pointing direction must be known to within 1/10 half-power beamwidth or better. |
| Pointing Stability: | Approximately 1/10 beamwidth or better. |
| Bandwidth: | 500 kHz to 10 MHz desired, with emphasis on lower half. |
| Spectral Resolution: | Good desired, and depends only on electronics for any one antenna. |
| Sensitivity: | Unfilled apertures entirely adequate. |
| Lock-on Time: | One-half second to several hours for time varying phenomena. For mapping observations, however, an antenna arrangement with as slow a drift rate as possible — of up to approximately 1 deg/sec — suffices. |

| | |
|---|---|
| Tolerance: | Prefer 1/20 $\lambda$ , but 1/16 $\lambda$ is adequate. (at 1 MHz, $\lambda$ = 300 m) |
| Orientation: | Eliminate antenna pattern directional ambiguity. |

The 1965 National Academy of Science "Woods Hole" report offered, among other, the following recommendations on long wave radio astronomy:

a. A space radio telescope with an aperture of the order of 20 km appears to be close in size to the ultimate for observations between a few MHz and a few hundred kHz. With larger apertures, effects due to irregularities in the interplanetary medium will, according to our best present knowledge, become the factor that limits the resolving power of the telescope.

b. Since the previous recommendation is for an ultimate space radio telescope, work should be started now that will lead to the use in space, within about 10 years, of a high-resolution broad-band antenna system for radio-astronomical observations over the frequency range 10 MHz to a few hundred kHz.

The following guidelines were suggested:

a. Determine whether or not an antenna covering about 10 MHz to 500 kHz and having a beam area of about 100 $deg^2$ at 1 MHz is feasible and meets the scientific needs.

b. Consider alternative design concepts, among which should be the possible use of a simple or compound interferometer system to give added resolution within the main beam area.

c. Consider and compare locations for such an instrument, particularly as between a high orbit and a lunar base.

d. Consider this antenna as a possible payload for the Apollo Applications Program.

## 2.2 ANALYSIS OF PROVIDED CONCEPTS

2.2.1 Concept 2a — Broadside Dipole (Brown Engineering Co. Technical Note AS-2-16). See Figure 2-1. The purpose of the antenna is to conduct radio astronomy investigations in the 1 to 3 MHz range.

The antenna consits of eight 246-ft dipole elements that are parallel, co-planer, and separated by 246 ft ($\lambda$ /2). An aluminized ground plane is suspended a distance of 123 ft ($\lambda$ /4) from the plane of the dipoles to shield the antenna from earth-generated and/or ionosphere-reflected electromagnetic waves.

A beam structure supports and maintains the relative positions of the dipoles and ground plane. Brown Engineering suggests three rudimentary methods of constructing this beam with three variations of one method.

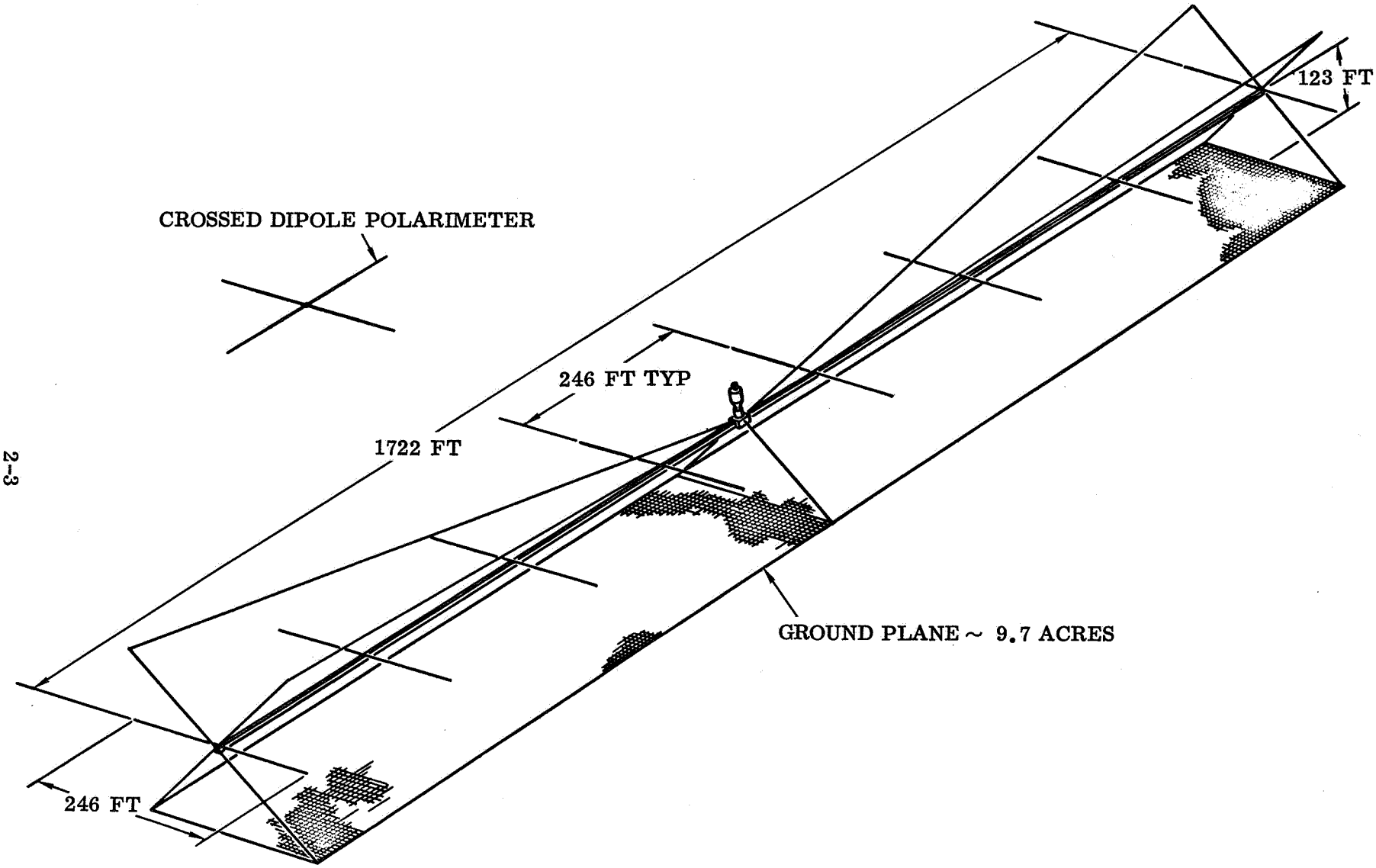



Figure 2-1. Concept 2A, Broadside Dipole Array Antenna (Brown Engineering)

The system also includes a crossed dipole polarimeter flown remotely from the array. In addition, the interferometer principle can be employed to achieve a narrow beamwidth ($\leq 1°$).

### 2.2.1.1 Structure and Mechanics

Extendable Concept (AS 2-16, Figure 39). This concept uses the extendible spring tape principle to construct the boom. The ground plane consists of a 1-ft grid composed of aluminiazed or silver plated net (base material unknown) supported 123 ft from the dipole plane. The method of erection and support is not specified, but could be a difficult problem.

In summary, this concept is not considered feasible in its present form. The rationale for this conclusion is based on the following:

Only the four-corner members and possibly the vertical stiffeners could be pre-assembled onto the drum (considerable doubt exists concerning the vertical stiffeners as they are potential snag items on the drum, which would prevent deployment). The two upper (and lower) corner members are rolled inside of each other onto the drum. This causes a longitudinally out-of-phase relationship of the corner members on the order of at least 7.5 ft. This requires the horizontal stiffeners to be pivoted at each end if attached at the time the joint emerges from the drum. Final securing of this joint cannot occur until after full extension. The same situation is true for guy wires in the upper and lower horizontal planes. The main boom then is not a rigid truss as it emerges from the drum and cannot be rigidized until full extension has occurred. In this disjointed and flailing form it is very unlikely that cinch-up of the joints could ever be realized, nor could stabilized control be accomplished by the CSM.

Previous assumptions permitted pre-assembly of vertical members with the corner members. Due to the out-of-phase relationship, the following number of member components must be space assembled:

| | |
|---|---|
| Horizontal stiffeners | 107 |
| Guy wires | 240 |
| Turnbuckles | 240 |
| Fasteners | 694 |

The concept requires an excessive number of assembly operations and reliance on joint mating alignments. A Manned Extravehicular Maneuvering Unit (a manned work platform studied by Ling Temco Vought for NASA/MSFC) was recommended for assembly of the structure. The relatively light, non-rigidized structure does not allow

a base onto which the MEMU can attach to react manipulator forces. Thus, all reactions constitute a propellant expenditure. To illustrate the consequences, assembly of one 8-foot section of the boom assumes coordinated maneuvers of the two MEMUs in which it is estimated a minimum of 25 stop/start pulses are required for each MEMU. To account for false starts, overshoots and reaction correction, the figure can reasonably be doubled, or 50 pulses. Approximately 56% of the beam requires some build-up via EVA. There would be 129 such sections to build plus the maneuvering to each extending side of the total vehicle to maintain symmetry during construction. Fifty × 120 plus related maneuvers and reserve factor (1.25 × 50 × 120) = 7,500 pulses per MEMU, and a total firing time on the order of 5,540 firing seconds. These values can be doubled again to account for final cinch-down of each joint and adjusting of tension cables. Hence, for each MEMU total pulses = 15,000 and total time = 11,080 sec. Total propellant expenditure is of the order of 3,800 lb. Assuming any one reaction engine totaled half of the use time, it is still beyond present capabilities. (For comparison, the rated duration of the CSM RSC engine is 1,000 sec based on 500 sec continuous operation.) The implication here is that considerable spare or redundancy is required of the MEMU plus maintenance hours.

The adjustment of 1,712 guy wires is an impractical task. At the time of adjustment of any individual guy wire, it would be very time consuming and impractical to determine the required tension in the wire. With the additive effect of 1,712 wires, the tension will vary considerably, will be changing and hence causing undue loads in some members with possible local collapse and/or distortion of the beam as a whole. Due to the great numbers of wires, it would be a monumental task to attempt to "tune" the whole beam or even to determine the point at which to begin.

Basic Shapes Concept (AS 2-16, Figure 40, 41, and 42). A piece-by-piece construction is illustrated using shapes down to the most rudimentary shapes such as tubes, angles, and wires. Figures 41 and 42 differ from 40 in geometric variations but each is constructed from basic shapes.

These three concepts differ from Figure 39, mainly in the manner of constructing the main booms. In this regard, a piece-by-piece construction is illustrated using basic components such as tubing, angles and wires.

Inflatable Concept (AS 2-16, Figure 43). The main support boom consists of seven inflatable tube sections, joined end-to-end and rigidized by a curing process. Mechanically, only the inflatable, rigidized concept is considered feasible, yet it has serious development problems. Distortion of the inflated, rigidized boom due to solar radiation is computed to be 43 ft at a temperature difference of 129° F. Packing ratios on the order of 35:1 to 40:1 are necessary to construct the main boom.

Rigidized structures such as expandable honeycomb have claimed packing ratios better than 1,000:1 in idealized shapes such as spheres. The design geometry of specific applications can vary this ratio by several magnitudes; however, a ratio as high as 40:1 is reasonable in a cylindrical geometry. The rigidized honeycomb has advantages over elastic recovery, or membranes rigidized by straining, because of the greater strengths and rigidity that are obtainable. The rigidizing technique is presently in the development stage by several companies. There are, of course, certain problems to be mastered in this development and include:

a. Reliability of deployment. Assurance that full deployment occurs before rigidization. Rigidizing on command.

b. Susceptibility of materials to space environment deterioration.

c. Thermal instability due to closed tubular shape.

d. Shelf-life of the curing media.

Present materials under investigation for rigidized structures have cure times varying from 10 min to 6 hr but most are less than 1 hr. Bending strengths can be obtained with values between 30,000 and 70,000 psi. A preliminary weight analysis shows these figures:

| | |
|---|---|
| Rigidizer and fabric | 10,600 |
| Dipole attach sections and containers | 640 |
| Clamps and attachments | 125 |
| | 11,365 lb |

2.2.1.2 Antenna Performance. Concept 2a does not satisfy the full range of user requirements; namely, 0.5 to 10 MHz bandwidth and 10° beamwidth. Antenna beamwidth in the H-plane varies from 26° at 1 MHz to 13° at 2 MHz while the beamwidth in the E-plane varies from about 90 to 78° over the same band. Thus, the beamwidth approaches the user requirements in one plane but not the other.

Although the antenna bandwidth is limited primarily by impedance considerations, it is also limited by pattern considerations. Broadside dipole arrays typically have characteristics that vary widely as a function of frequency.

2.2.1.3 Electrical Power. The two basic concepts for large radio astronomy antenna will differ little with respect to power. The erection and activation of these antenna by astronauts implies that the Apollo command and service modules with airlocks will be available to provide power for erection tools. The use of powered erection devices may be limited by the power available, which is not expected to exceed several hundred watts over normal CSM station keeping requirements.

The use of batteries for high-rate power requirement will incur a penalty of approximately 13 lb/kw for periods up to one hour and 225 lb/kw for 24-hour periods. Power for experiment operational periods is composed of scientific electronics plus data and communications requirements. These have been estimated to be approximately 50 watts apiece for a total of 100 watts. This requirement could be accommodated with less than 40 $ft^2$ of unoriented solar cell arrays. Energy storage for dark time operation based upon Ni-Cad secondary batteries at 10 w-hr/lb penalty would be approximately 15 lb for synchronous orbit.

2.2.1.4 Dynamics/ACS. Pertinent concept data are synchronous orbit, one-year lifetime, and pointing accuracy of one degree. It is assumed that the antenna reorients once each hour.

Impulse requirements are dictated by the control maneuvers as they are much larger than external torques. Attitude control by jets is appropriate. Some rationalization must be imposed upon maneuvering in the polarimeter mode due to excessive propellant weight that can accrue.

The impulse requirements are estimated and shown in Table 2-1 for a one-year mission, for jets mounted at the ends of the 1,720-ft. dipole. The propellant weight is estimated using an Isp of 300.

Table 2-1. Dipole Angular Momentum and Propellant

| | PITCH OR YAW | PITCH AND YAW |
|---|---|---|
| Impulse (lb-sec) | | |
| Gravity-Gradient | $1.03 \times 10^4$ | $2.06 \times 10^4$ |
| Maneuvering | $4.28 \times 10^5$ | $8.56 \times 10^5$ |
| TOTAL | $4.38 \times 10^5$ | $8.76 \times 10^5$ |
| Propellant Weight (lb) | | |
| Gravity-Gradient | 34.3 | 68.6 |
| Maneuvering | 1430 | 2860 |
| TOTAL | 1464 | 2929 |

The impulse requirements for maneuvering are much larger than those to overcome external torques mainly because of the assumption of one re-orientation per hour.

The upper limit of the control system frequencies is of the order of a few hundredths of Hz. The structural characteristics of the dipole were calculated for a configuration in which the jets, with their propellant and associated equipment, are located 3/4 out from the center. A beam representation was used for the dipole. The results from the digital program are shown in Table 2-2.

Table 2-2. Dipole Structural Characteristics

| MODE | FREQUENCY (Hz) | GENERALIZED MASS (lb-sec$^2$/in.) | NORMALIZED AT END (in.) | DEFLECTION 3/4 FROM CENTER (in.) |
|---|---|---|---|---|
| 1 | 0.0196 | 9.23 | 0.995 | 0.412 |
| 2 | 0.0592 | 8.44 | ± 1 | ± 0.0319 |
| 3 | 0.0965 | 8.95 | 0.985 | 0.280 |

The maximum deflections of the ends are estimated. Modes 1 and 3 are most favorably excited by two jets firing in the same direction, as would be done for station keeping. Mode 2 is most favorable excited by two jets firing in opposite directions, as in attitude changes. The maximum deflections at the ends of the dipole are 3.56, 0.017, and 0.0976 in. for the three modes. The lower result of the second mode is attributed to the relative ineffectiveness of exciting that mode at the jet location.

Two areas of possible dynamic problems are pinpointed; 1) changes in structural stiffness, and 2) the cross-member vibration.

The basic structural problem of this concept is the structural weakness of the individual members. Once a member begins to buckle, the effective structural stiffness drops and the structural characteristics change. Thus, although no problems are anticipated because of control-system/structure coupling for the configuration, problems could arise from this source should a member begin to buckle.

Vibrations in the cross members could pose problems. Should vibrations occur, the dipole ends will oscillate at twice the cross member frequency.

The concept includes the use of a cross pole polarimeter 9 km away from the dipole to form an interferometer. The interferometer would have a beamwidth of about 1°. The attitude control requirements on both the dipole and the interferometer are of the order of 0.1°. Thus, the interferometer tightens the attitude control requirements and requires control of the 9 km separation distance. The ability of the control system to

hold on 0.1° can be degraded by the coupled control system-structural vibrations. The maximum estimated deflection was 356 in., or 29.7 ft. The resulting effective change of the entire antenna angle can be appreciable.

2.2.1.5 Thermodynamics. For the inflatable concept maximum temperature differences between 120 and 130°F were calculated between the sunlit and dark sides of the boom. The exchange of energy is limited to radiation heat transfer. A discussion of the analysis of tubular members is included in Section 2.4.

2.2.1.6 Stress. In the case of the astronaut-assembled space truss, some consideration of the effects of astronaut impact must be given when sizing members.

The space truss structure proposed does not appear to include diagonal cross bracing in the plane of the cross section. This would be required in order to reduce the effective column length to a reasonable value. Other than this, the space truss or inflatable rigidizing boom concepts appears reasonable. For equivalent weights, the space truss will provide considerably more flexural rigidity but less torsional rigidity than the inflatable rigidizing boom.

2.2.1.7 Crew and Equipment Functions. Gross crew and equipment functions were analyzed and a time line analysis was made to estimate the time required to erect an operational antenna.

There is an excessive reliance upon EVA to construct the antenna. The piecemeal assembly of the many components, multiple attachments, and guy wire adjustments reflects an inadequate mechanical conception on a time and feasibility basis. Consider for example, that the weight of the CSM shown is that for a 14-day mission. This requires assembly, checkout and test within the 14 days. At the very least, the main boom construction and stabilization must be complete. Final assembly, checkout, and test would have to be completed on a subsequent flight. Consider these assembly tasks:

| | | |
|---|---|---|
| 107 stiffeners having 2 attaching ends @ 7 min each | | 749 |
| 240 guy wires having 2 attaching ends @ 7 min each | | 1,680 |
| 1,712 turnbuckle adjustments (initial) @ 2 min each | | 3,424 |
| Final cinch-up of stiffeners @ 3 min per end | | 321 |
| Readjustment of turnbuckles @ 2 min each | | 3,424 |
| | TOTAL | 9,598 min or 160 hr |

Assuming work time of 3 hr per day per astronaut and 2 astronauts working simultaneously; there are four 1-hr working opportunities per day: 160/4 = 40 days. Obviously this is impractical and yet only considers the effort to build the main structural boom.

2.2.2 Concept 2b Rhombic (Brown Engineering Company Technical Note AS 2-16). Concept 2b is a large rhombic antenna conceived by Brown Engineering and consisting of eight 300-meter compression struts supported at mid-span by a nonmetallic 254-meter-diameter toroidal structure as shown in Figure 2-2. The antenna electronic packages and attitude control systems are supported at the extreme vertices of the struts. The assembly facility, air lock and docking port are attached to the torus.

An alternate configuration is included that substitutes a wire-braced rectangular box structure for the torus.

2.2.2.1 Structure and Mechanics. The antenna elements are telescoping tubing or assembled sections of basic structural shapes. The material is aluminum or titanium. One advantage of rhombic geometry is inherent stiffness; this allows use of smaller structural components. Cursory analysis shows antenna element diameters to be approximately 2 in. at terminal points and 6 in. at torus attach points. The element would be 982 ft in length. Approximate weight of the concept is 10,000 lb.

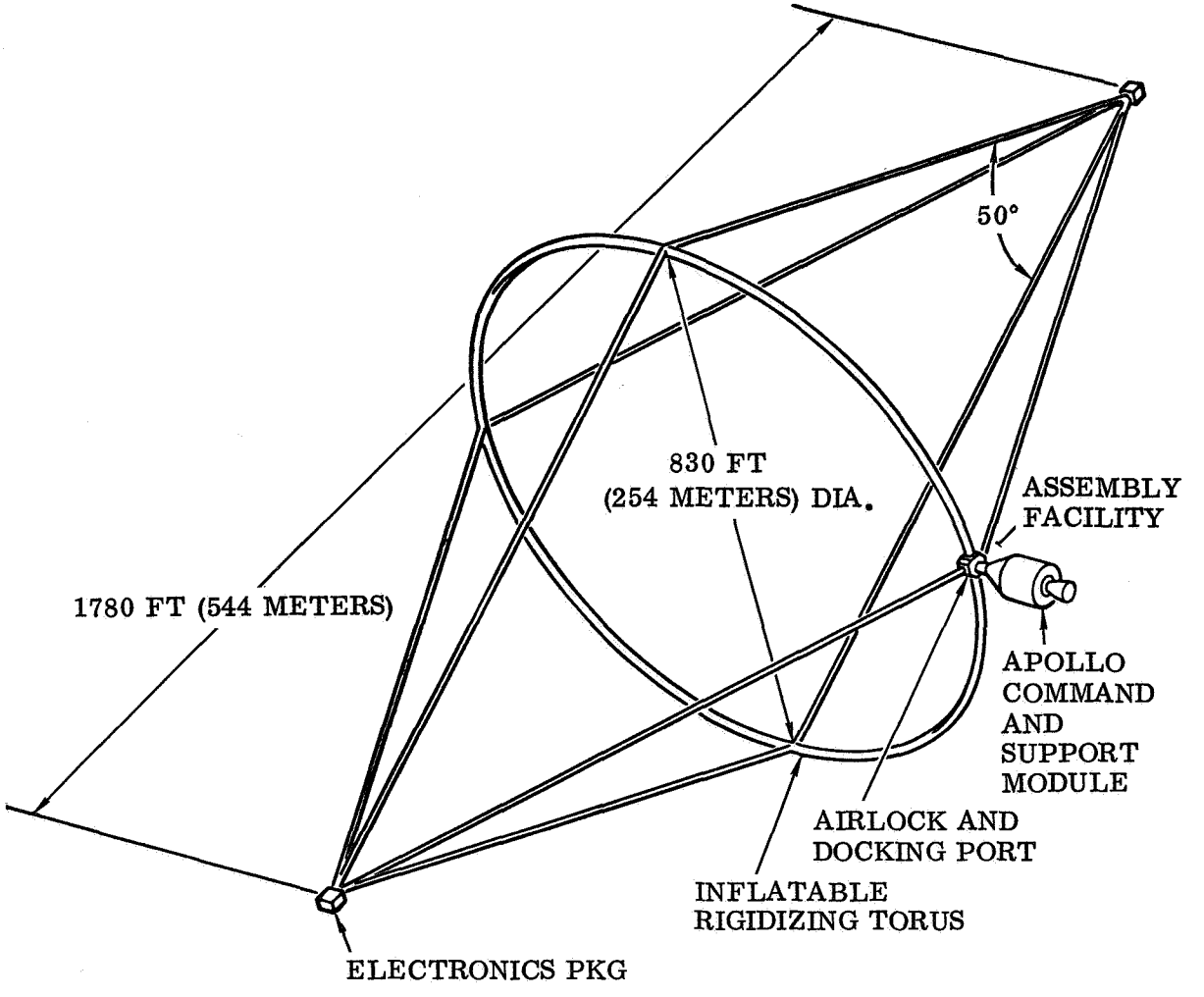
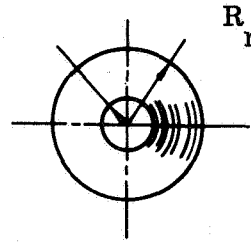



Figure 2-2. Concept 2b, Inflatable-Torus-Center-Section Rhombus Concept (Brown Engineering)

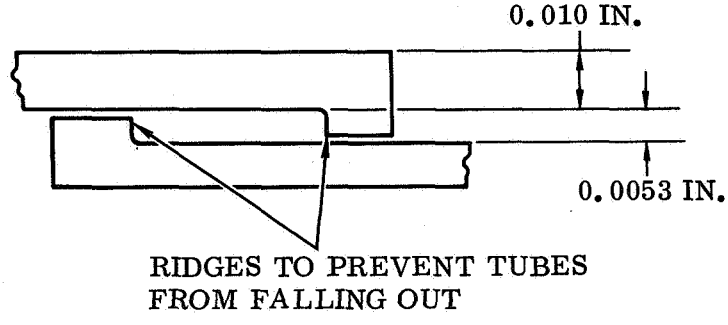
$R_{min}$ = 1 IN.



$R_{max}$ = 3 IN.

Since the maximum telescoping section length is 8 ft, and if there were no final overlap, 982/8 = 123 sections would be required. However to stabilize the joints requires at least 1-1/2 diameter overlap. Therefore, the effective section length would be 8 ft minus 1-1/2 diameter; using an average diameter of 4 in., the section effective length would be 90 in. Therefore, 982/12(90) = 131 sections would be required.

The sections must be contained within the $R_{max} - R_{min}$ = 2 in. The total space for each element wall is then 2/131 = 0.0153 in. Some clearance must be allowed to guarantee sliding, say 0.0053 in. is clearance, then the maximum wall thickness becomes 0.010 in.




It is doubtful that 0.005-in. ridges would be adequate on such thin wall tubing to guarantee that the tubes do not disengage (over extend).

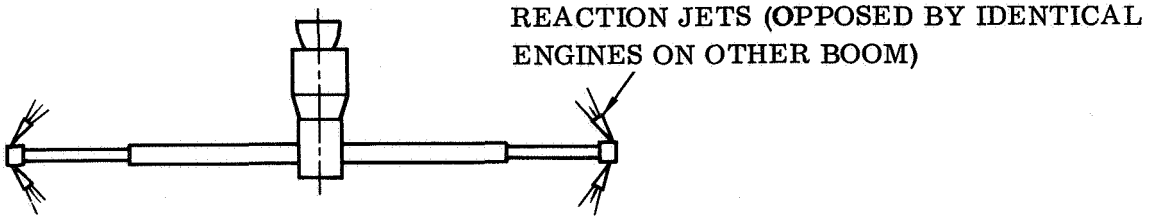
Basic methods of deploying the telescoping tubes are:

a. By internal pressure (restrain extension rate with an internal wire attached to a constant force drum — friction type).

b. Attach rocket motors to the ends and pull the tubes apart (extend).

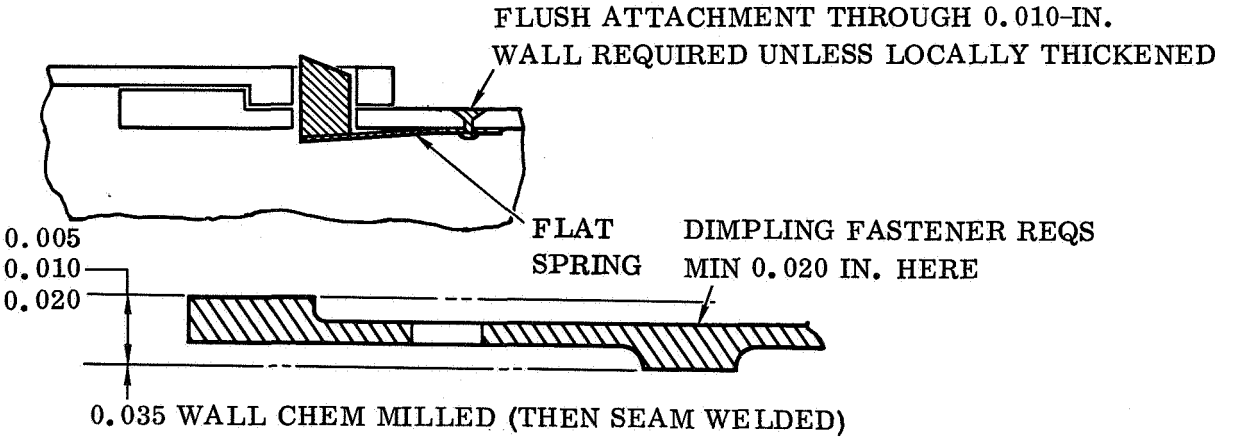
c. Have an astronaut with AMU pull the tubes out.

The second method is probably best.



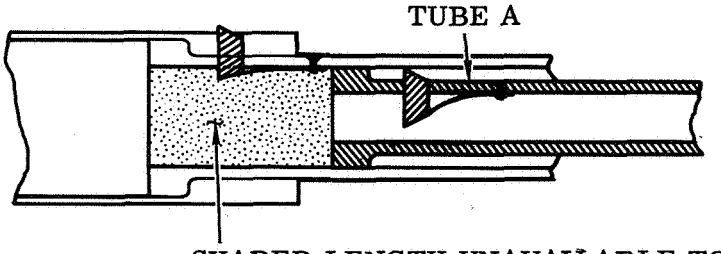

The extended tubes must be locked when extended to react axial compressive loads. At least three fasteners would be required for each joint if mechanical fasteners are used. This represents a total of 130 x 3 x 8 = 3120 fasteners that would require EVA to effect.

One solution to an automatic tube lock might be as shown below.

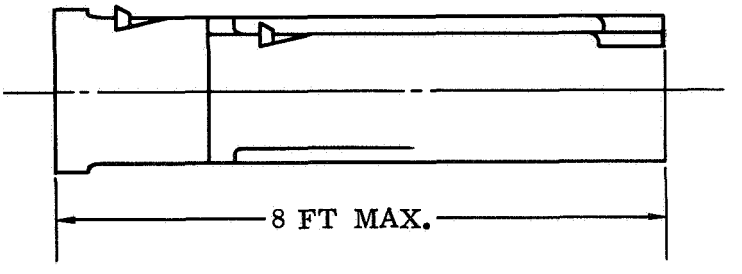



Tube manufacturing would be very expensive since 131 different sizes are required and it appears that they must be made in flat pattern then seam welded into tubes, then possibly stretch stress relieved or pressurized to yield to maintain roundness.

Unfortunately most of the solutions to auto-locking reduce the effective extendible length of a tube.




Therefore the actual maximum practical tube extended length is 8 ft - 1 1/2 diameter -lock mechanism x number of sections nestled.




Assume the lock mechanism occupies a length of 2 in. (.67 ft); the retracted position yields a section length, $L_r = 8 + (8 - 0.67N) + ---$, N = 1,2,3---. The termination is when 0.67N = 8. N = 11.8, but for practical reasons would be 10. The total extended length is $L_e = 8 + (8 - 0.67N - 0.5) + ---$ and when N = 10, $L_e = 37.8$ ft. Obviously, this is far from the necessary 982 ft. It is concluded that insufficient space exists for self locking. If an astronaut could perform a single fastening operation in 1 min, and at 2 hr useful work per day, it would require 26.2 man days to effect lock-up.

Evaluation of the 300-meter-leg rhombus yields the following points:

a. The 300-meter size is not maximum in weight or packaged volume for the manned Saturn-V launch, larger versions should be considered. No maximum size has been determined for the criteria assumed.

b. The structure of the truss should not be solid tubes because of the extremely large thermal deflections, some type of highly perforated screen mesh material is recommended.

c. Telescoping tubes are poor for the large extension ratios required (greater than 120 to 1). They require excessive EVA and would be very expensive to manufacture. Also, the large number of moving parts would imply low reliability.

A modified design using a center compression tube to support wire antenna elements should be studied for feasibility because it will have a minimum of compression members, which have problems of thermal bonding and stowage.

2.2.2.2 Antenna Performance. The orthogonal rhombic antenna pair has leg lengths of 300 meters and acute angles of 50° (Figure 2-2). For optimum RF performance the rhombic acute angle at given leg lengths must be that shown in Figure 2-3. Practical operating frequency limits are such that leg lengths are greater than one wavelength at the operating frequency. Thus, the 300-meter-leg rhombic cannot be used below one MHz.

The design frequency for the given concept (50°) is approximately 4 MHz. The operable bandwidth of this configuration is 3 MHz to 5 MHz. The antenna beamwidth is approximately 30°. To extend the operation of this 300-meter rhombic antenna to the lower frequency of 1 MHz the acute angle must be expanded to approximately 70°. This results in a corresponding limitation on gain and upper operating frequency. The upper frequency region of interest (10 MHz) requires an acute angle of approximately 35° per Figure 2-3.

It can be concluded that the entire frequency band of interest (0.5 to 10 MHz) cannot be covered by the concept presented. To cover frequencies below 1 MHz, leg lengths of at least one wavelength long are required (i.e., for frequency of 0.5 MHz, leg lengths should be greater than 600 meters).

A technique to adjust the angles subtended by the rhombic legs must be developed if one antenna is to operate for the full frequency range from 0.5 MHz to 10 MHz. The required 10° beamwidth is beyond the capability of the 300-meter-leg rhombic.

2.2.2.3 Crew and Equipment Functional Analysis. Gross crew and equipment functions have been analyzed and are indicated in the functional flow diagram, Figure 2-4. A time line analysis has also been made, resulting in an EVA time requirement of 76 hr.

The EV crewman must have the capability of maneuvering around the antenna package out to the end of the telescoping struts (about 300 meters) and around the torus assembly (about 800 meters). Portable handholds would suffice for maneuvering along the telescoping struts as frequent stops every eight feet are required to fasten the strut joints. This assures that the EV crewman has some means of securing the necessary tools to his body and has both hands free for maneuvering. Inspection of the inflatable

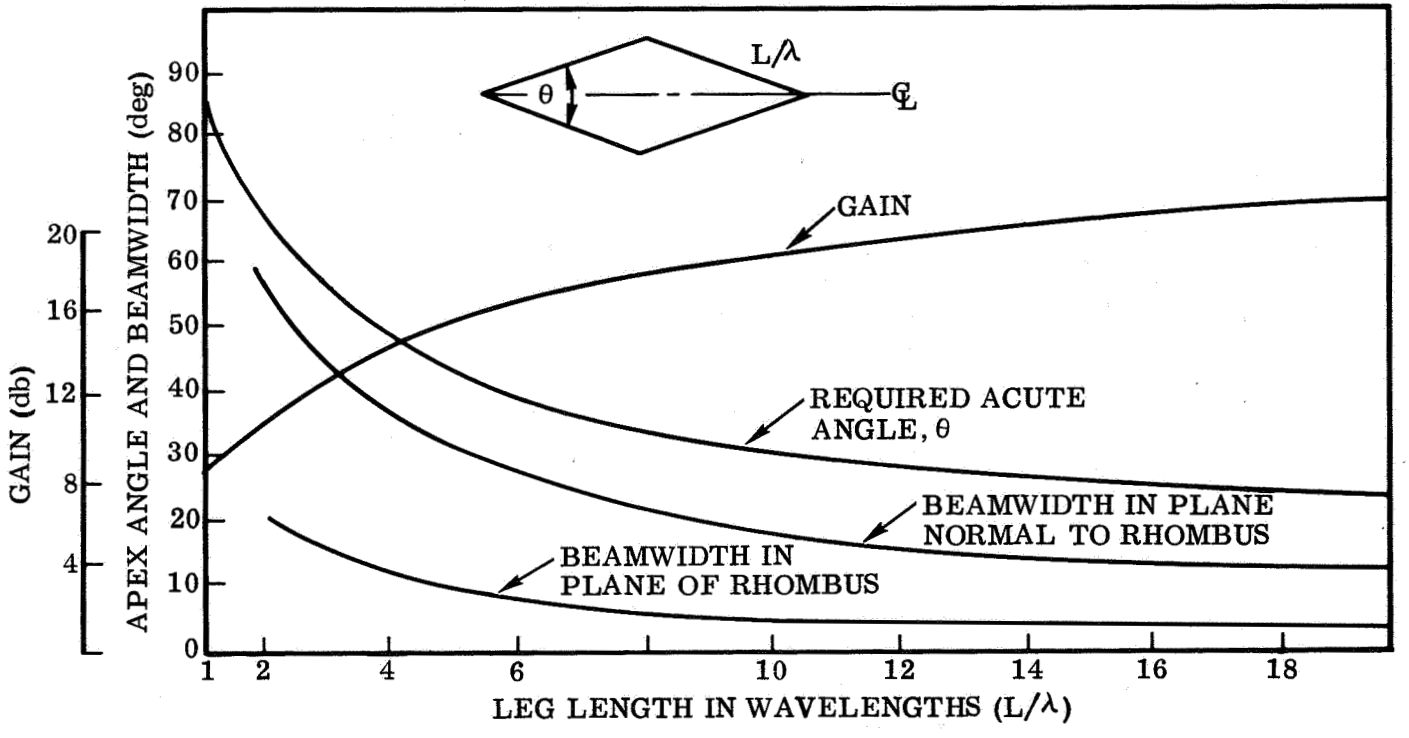



Figure 2-3. Rhombic Antenna Acute Angle Required As a Function of Frequency

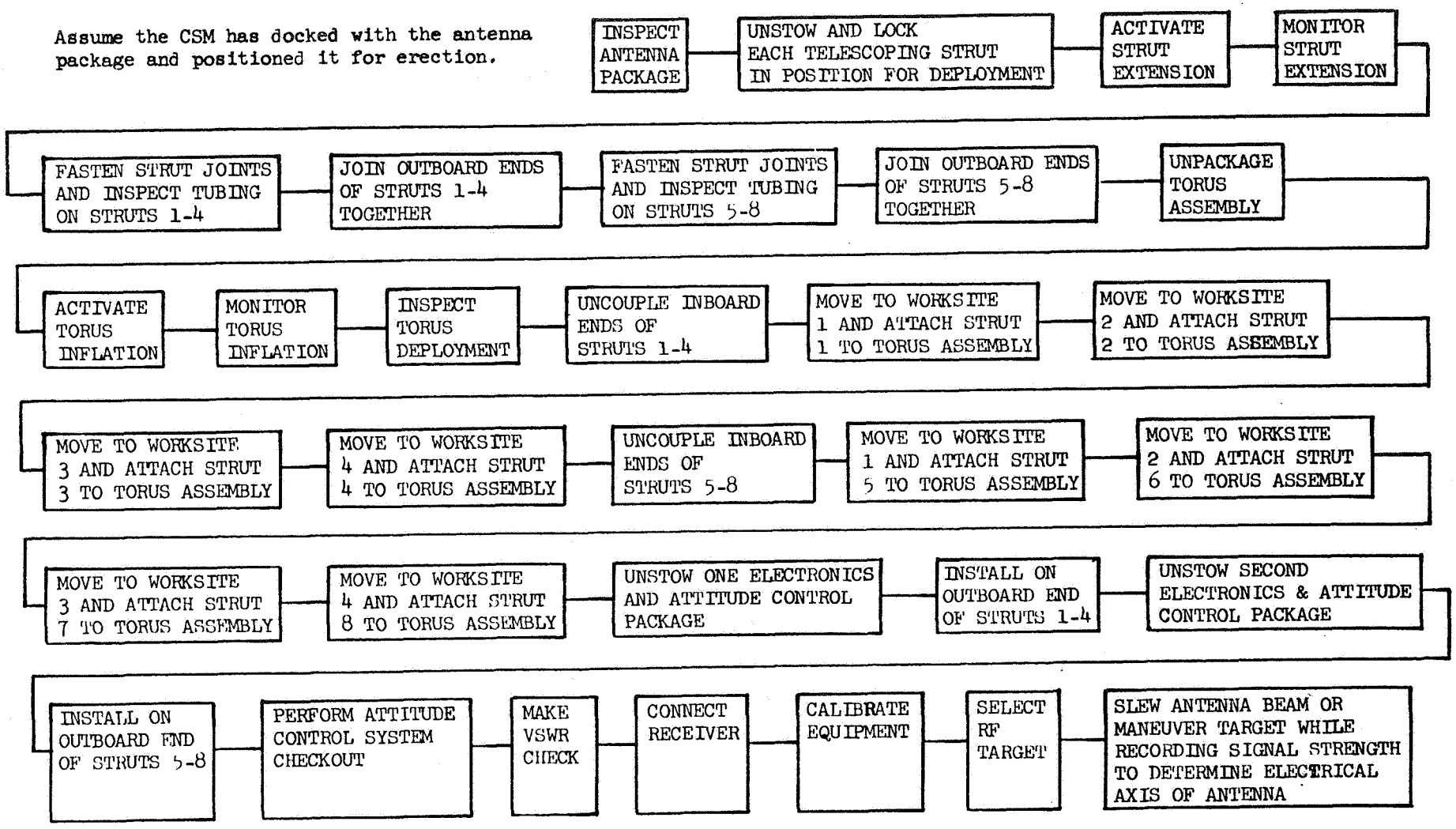



Figure 2-4. Function Flow Diagram for the Erection of a Rhombic Antenna, Inflatable Torus and Rigid Legs (Brown Engineering)

torus assembly would be impractical by this method and a suitable AMU or unstabilized maneuvering device (e.g. hand-held gun) should be used. An AMU should have an operational life time of four hours and have a resupply capability.

2.2.3 Concept 3 — 1500-Meter Paraboloid Antenna Astro-Research Corp., ARC-R-205,224). The orbiting parabolic reflector radio telescope consists of a 1,500-meter-diameter reflector in the form of a truncated paraboloid, Figure 2-5. The reflector surface is formed by a gridwork of metallic strips connected by non-conductive shroud lines to a central mast and held in shape by centrifugal force generated by rotation of the antenna at 1 revolution each 16 min. Because of this rotation and the minimum weight structure, docking and EVA become difficult. Therefore, this concept was considered to be unmanned during all phases of the mission.

The mast supports the RF and electronic equipment and consists of longerons made of unidirectional fiber-glass rod. A special feed is required for the antenna because the 900-meter hole in the center causes a 36% efficiency loss. The antenna will operate over the 0.5 to 10 MHz bandwidth desired, however, at frequencies less than 1 MHz, the beamwidth is greater then 15°. An operating frequency of 4 MHz results in a 5° beamwidth with a 33 db gain. The reflector ribbon thickness of 0.5 mil is 1/3 of the 1.65 mils thickness required for good reflectivity, this results in an additional decrease in efficiency.

The compressive design load on the central mast is approximately 2 newtons for the column length of 820 meters. This yields a load intensity many orders of magnitude below the range commonly encountered in structural design practice.

Critical speed (i.e., bending vibrations resonant with spin rate) and classical Euler buckling enter the design criteria. Deviations from geometrical perfection become a major concern in selection of overall configuration dimensions and design detail. The mast is folded into and deployed from a relatively small payload compartment, allowing a column width of only approximately 3 meters.

A lattice-column design shown in Figure 2-6 is a 1/10 scale model of a partial length; in folded, partially-deployed, and fully-extended conditions. It consists of three continuous longerons made from unidirectional fiber-glass rod connected by swivels to triangular frames. A torsional elastic collapse mode is employed for folding and deployment, actuated (and stabilized) by spring constraints in the diagonal bracings.

2.2.3.1 Structure and Mechanics. The allowable payload volume of 4,500 $ft^3$ for a manned Saturn-V launch vehicle is less than the packaged volume of the experiment, which is approximately 12,000 $ft^3$. This volume assumes that the metallic strip gridwork would be bonded to a thin photolizeable film during manufacturing to facilitate packaging and deployment without snagging. The antenna diameter could be reduced to 400 meters and packaged into 5,720 $ft^3$, which is available for unmanned Saturn -IB launches. A 400-meter paraboloid will provide 30° beamwidth at 2.3 MHz and 5° at 10 MHz.

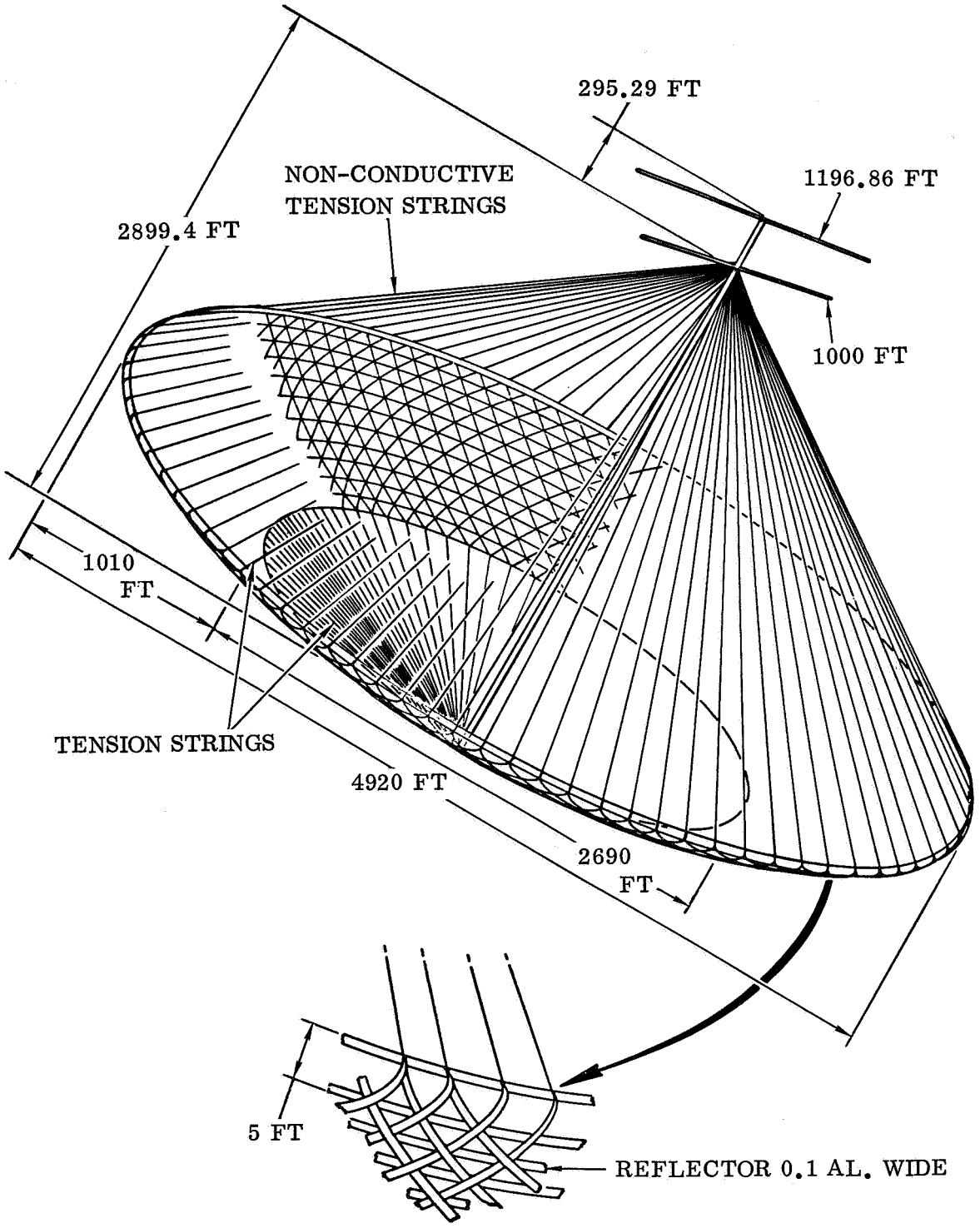
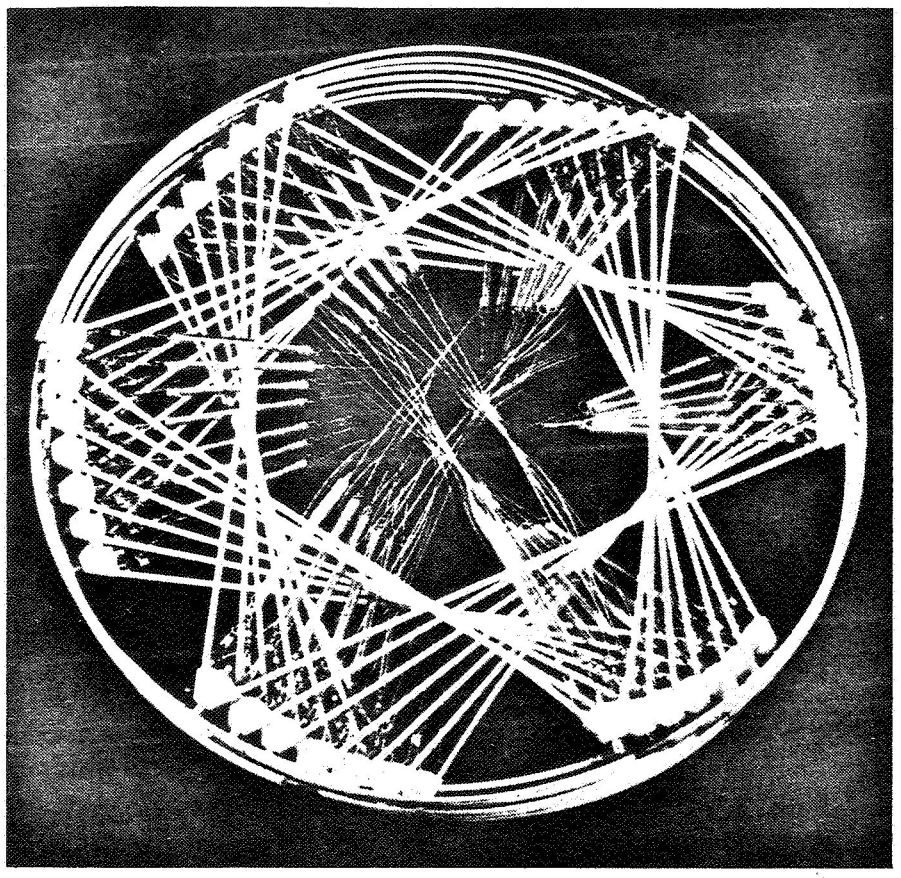
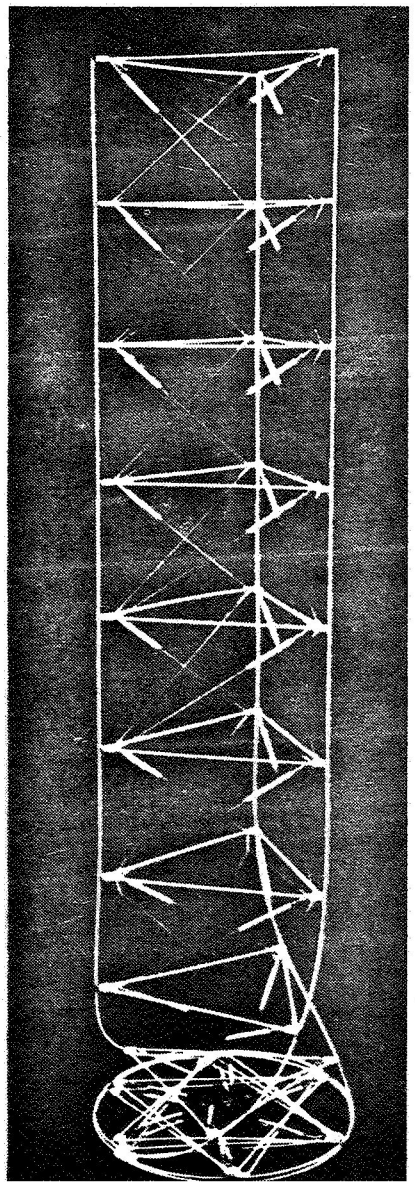


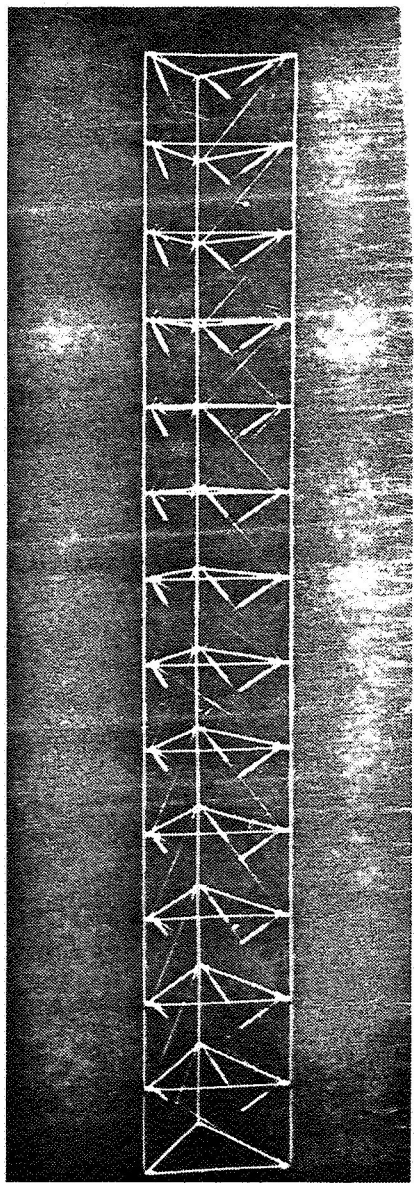

Figure 2-5. Concept 3 — 1500 Meter Parabolic Antenna (Astro Research Corp)



FOLDED



PARTIALLY DEPLOYED



DEPLOYED

Figure 2-6. Deployable Column Folded by Flexure of Longerons

The 400-meter-diameter antenna packaged within the nose fairing and SLA is shown in Figure 2-7. This volume includes package canister and support structure.

2.2.3.2 Antenna Performance. Although a feed design was not supplied, a special feed is required for the antenna having a highly reduced pattern gain over the area of the 900-meter hole in the 1500-meter antenna. The percentage-area loss is equal to $900^2/1,500^2 \cong 36\%$. Spillover will occur both around the outer rim of the paraboloid and through the center making the antenna highly inefficient. A continuous dish at 4 MHz would have a beamwidth of approximately 3.5° and a gain of 33 db. The combined effect of reduced aperture area and increased spillover would enhance the side lobes and reduce the efficiency to 10-20%. Beam angle broadening will occur by a factor of 1-1/2 to 2. An additional consideration for antenna performance is the mesh (ribbon) spacing and thickness. A conductor spacing of 0.8 meter ($\lambda = 75$ meters) is adequate for reflecting energy at 4 MHz. Skin depth in inches for an aluminum conductor is equal to $(3.3/\sqrt{f})$ in. At 4 MHz, the skin depth is 1.65 mils. Ribbon thicknesses of 0.5 mils in the proposed design are only 1/3 of a skin depth rather than a design goal of 2-3 required for good reflectivity.

2.2.3.4 Dynamics/ACS. The rim current attitude control concept will not give control about three axes in its one-loop form and some problems remain with three loops. Substitution of jets solves existing problems but introduces new ones. The weight of each additional loop will be appreciable in order to decrease electrical resistance. Consequently, the basic design of the concept must be changed by the additional structure of the two new loops. Damping must be provided. This can be done by appropriately varying the currents and might be done by damping inherent in the antenna structure. The current loops will not in any case rotate the antenna about the direction of the earth's magnetic field. Since the inertial detection of the field is approximately constant for the mission's equatorial orbits, there is an inertial direction about which only limited torque is available.

The magnitude of the external torques estimated below preclude the possibility of alternate control torquers other than mass expulsion techniques. Therefore, jets are examined. Anticipated thrust level would be tenths of pounds.

The significant external torques are gravity-gradient and solar pressure. The discussion of gravity-gradient torques in the Astro Research reports considers the relation of transverse to longitudinal inertia to vary from 0.5 to 1.2. Recalling that the antenna is to have a low spin rate about the longtidudinal, it does not appear to be desirable to have the transverse larger than the longitudinal inertia.

The maximum gravity-gradient torque is estimated assuming that the ratio of transverse to longitudinal inertia is 0.8. Astro Research estimates the longitudinal inertia as $0.5 \times 10^9$ kg-m$^2$, which is $4.46 \times 10^8$ slug-ft$^2$. This inertia is unusually large for a structure of this weight, and leads to unusually large external torques for the orbital altitude. The physical cause is the concentration of a major fraction of the total system weight out on the rim. The maximum gravity-gradient torque is then $10^6$ ft-lb.

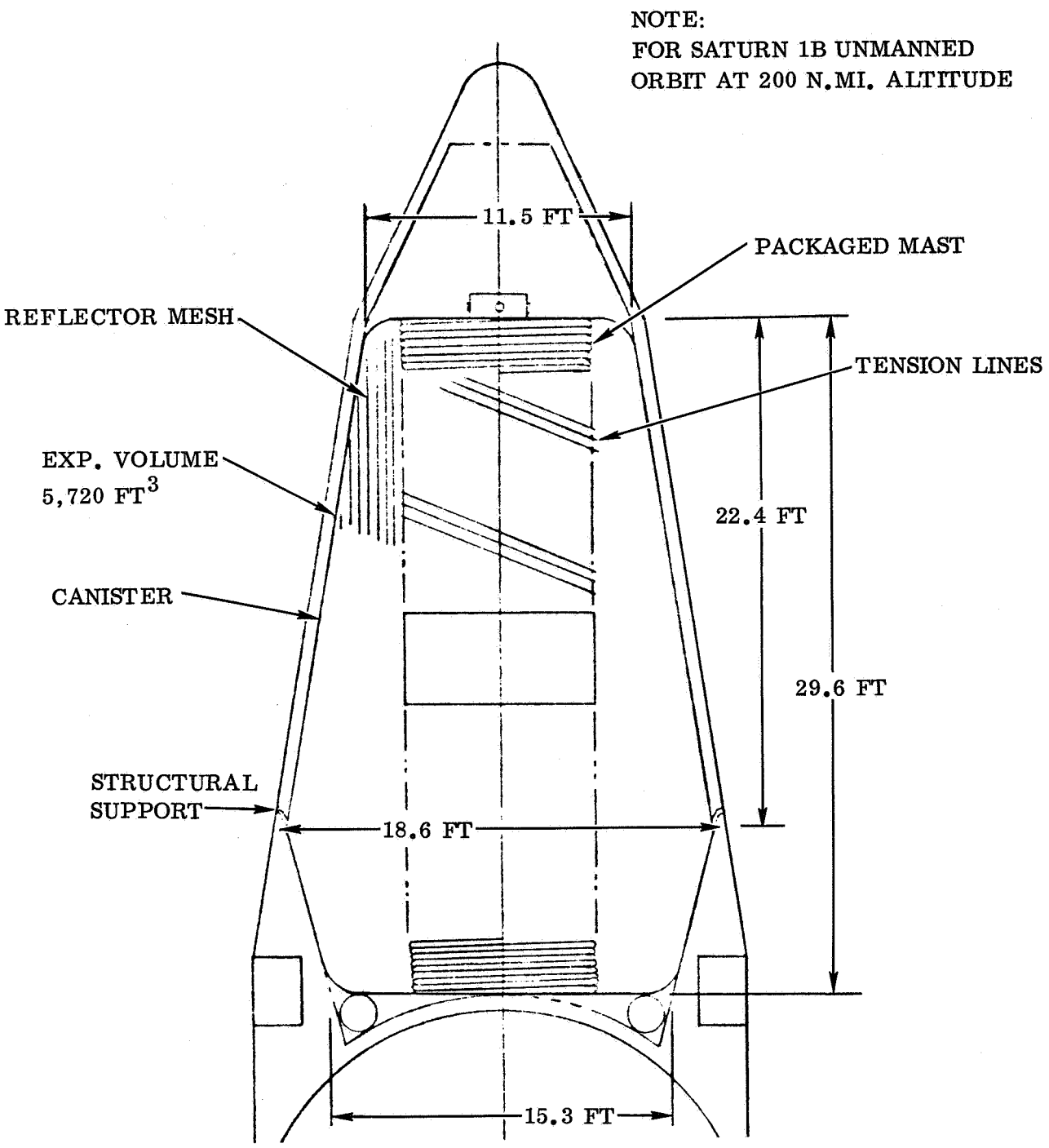



Figure 2-7. 400-Meter-Diameter Antenna Packaged in Nose Fairing and SLA

Solar pressure makes a significant contribution to the external torques on the antenna because the area of the $6.4 \times 10^{10}$ meters total length of 0.1-in.-wide ribbon is $1.75 \times 10^5$ ft$^2$. The solar force is then 0.0169 lb. It is assumed that the maximum solar torque orientation has one-third of the total ribbon area as the collector and that the moment arm is 1,500 ft. The maximum solar torque is then 5.95 ft-lb. The maximum external torque is then 16.2 ft-lb. The angular impulse requirement for one year is estimated to be $2.56 \times 10^8$ ft-lb-sec for each of the pitch and yaw channels. The preferred jet location from the ACS point of view is on the rim. This location is preferred because it increases the spin axis inertia more than the transverse inertias, and the longer moment arm requires less fuel. The radius is 2,460 ft and each channel needs $1.04 \times 10^5$ lb-sec of impulse. An Isp of 300 results in a pitch and yaw propellant weight of 686 lb. This with associated equipment adds about 1,000 lb to the rim. Thus, going to a jet ACS increases the antenna system weight from 2,200 lb to 3,200 lb. Note that no contribution due to maneuvering has been included.

Design considerations would remove this load from the rim and place it on the column end. There is considerable question as to whether the structure can withstand this rim load. However, column mounting is not desirable because of the effects on inertia ratios and an approximate doubling of propellant weight.

2.2.4 Concept 6, 600-Meter Rhombus (NASA/MSFC). This concept, suggested by Mr. Tom Clark of NASA, is a coplanar rhombic antenna with 300- or 600-meter antenna elements. The performance of the 300-meter rhombus was evaluated in Concept 2b; therefore, only the 600-meter size was evaluated further. Mr. Clark did not specify any structural configuration, so a new structural approach was synthesized. The antenna elements are assumed to be wires to eliminate thermal distortion and to reduce the total length of the compression structure required. A center mast supported on rigid cross spreaders is provided to support the wire antenna elements. Thermal deflections of the antenna centerline are minimized by pin-jointing the mast at the cross brace. Equal tension antenna wire reels are used to center the cross brace with respect to the antenna axis. The antenna apex angle can be varied by reeling the wire attach fittings on the cross brace in towards the centerline. See Figure 2-8. The increase in size to 600 meters and variable angle permit operation over the full bandwdith from 0.5 to 10 MHz with a 1 MHz beamwidth of 20 x 57 deg. The wire antenna elements will have first-mode frequencies on the order of 0.047 Hz, which is the range of the control frequencies; therefore, possible control system coupling and the associated large deflections may occur during engine firing. The attitude control propellant increases approximately as the cube of the scale factor (if the center column is designed to a constant $L/\rho$), therefore it is necessary to decrease the scan rate of larger antennas to maintain reasonable propellant weights.

2.2.4.1 Structure and Mechanics. The following structural concepts were considered:

a. Compression Antenna Elements with Braced Cross.

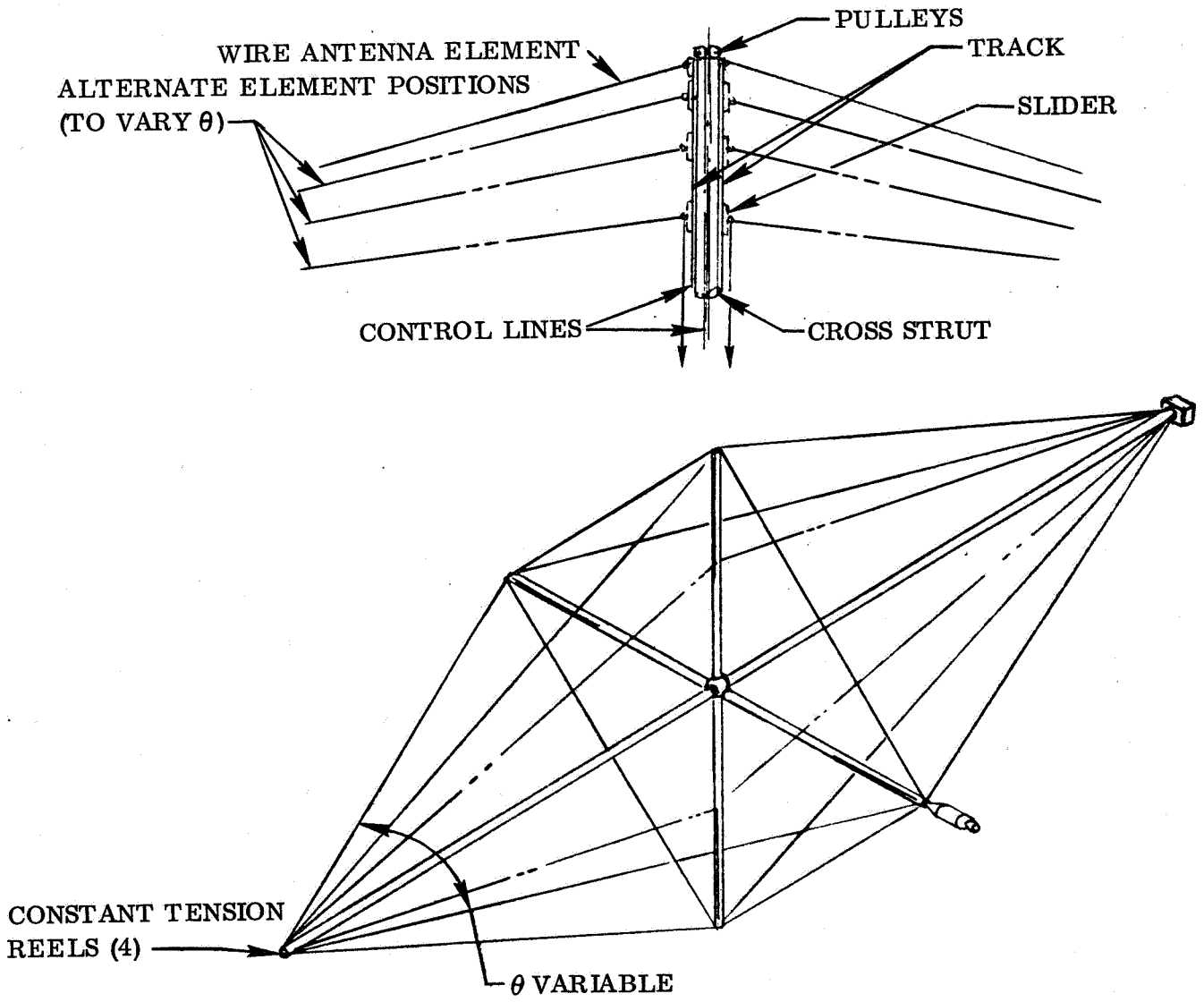



Figure 2-8. Concept 6 — Variable Angle Rhombus, 600 Meter (NASA)

b. Wire Antenna Elements, Rigid Center Tube — Torus.

c. Wire Antenna Elements, Rigid Center Tube — Box.

d. Wire Antenna Elements, Rigid Center Tube – Cross.

The wire antenna elements have increased simplicity and reduced packaged size because only 1040 meters of compressive structure is required for the center tube as compared with 8 x 600 = 4800 meters of compressive structure for the antenna elements in Configuration 1. The least total compression structure, lightest weight, and lowest moment of inertia is provided by Configuration 4. Unfortunately the problem of thermal distortions from solar radiation is not eliminated by the fixed geometry center tube configuration.

It appears that the effect of center tube thermal distortion on the antenna radiating element can be significantly reduced by pin jointing the center tube at the rigid center cross brace. See Figure 2-9.

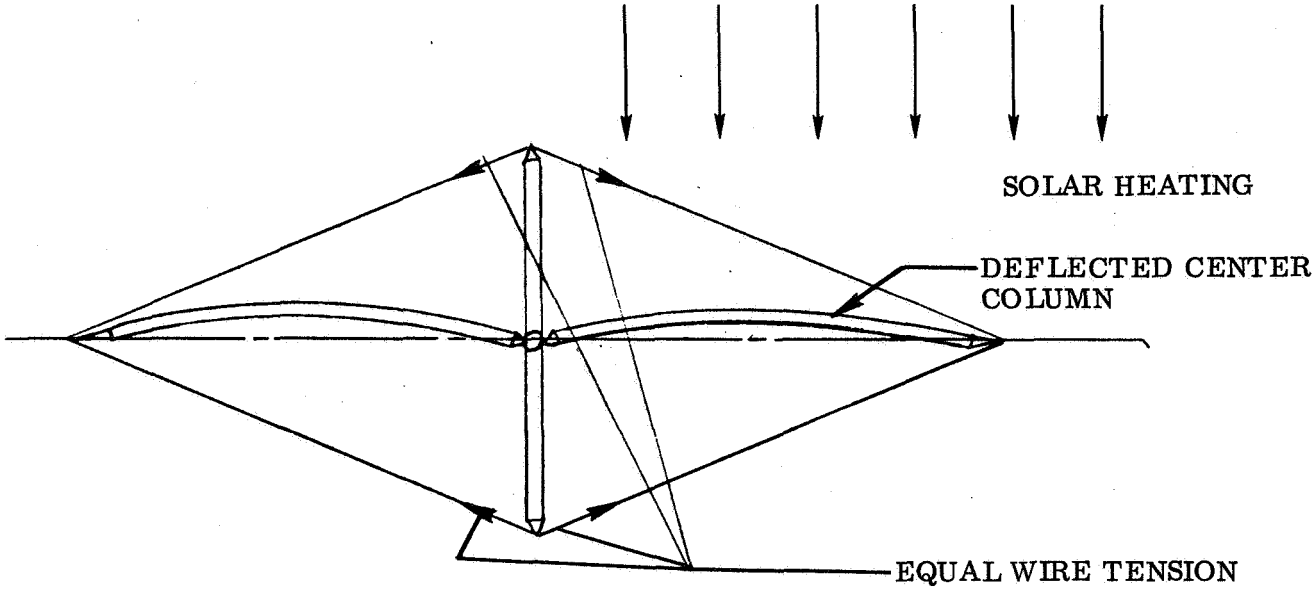



Figure 2-9. Pin-Jointed Center Boom

Symmetry can only be maintained by assuring equal tension in all four antenna elements in each plane of the rhombus, this requires balanced wire reels capable of maintaing force (tension) levels in the 0.01 to 1 lb range. The specific cable design tension must be established by comparing attitude control system thrust requirements against the buckling load of the thermally deflected center column.

The antenna distortion problems can also be improved by using a highly perforated mesh material for the compression members.

2.2.4.2 Antenna Performance. By referring to Figure 2-3 and the discussion of Concept 2b it is readily seen that the frequency range of 100 kHz to 25 MHz given by the user as a desirable frequency band cannot be covered by a rhombic antenna with 300 meter legs.

The 600 meter leg rhombus can be applied to frequencies about 500 kHz, but at 500 kHz the half-power beamwidth would approach 70°, as the acute angle approaches 90°. This system will cover the frequencies above 500 kHz with corresponding beamwidth and gain improvement by decreasing the acute angle in Figure 2-3.

Example: for 10 MHz operating frequency, the acute angle should be adjusted to approximately 26°. This results in a half-power beamwidth of approximately 5 × 15°.

It should be noted that the goal of 10° by 10° beamwidth is not practical for rhombic antennas with legs less than approximately 20 wavelengths. To cover the frequency range from 0.5 MHz to 25 MHz the acute angle defined by the legs of the rhombus must be adjustable from 30 to 90°.

To obtain a beamwidth of 5 by 15° at 0.5 MHz would require rhombic legs 20 $\lambda$ or 12,000 meters long.

2.2.4.3 Dynamics/ACS. The first-mode frequency of the individual legs if 0.047 Hz. The control frequencies are expected to be in the range of hundredths of Hz. The deflection associated with the leg vibration will be relatively large since the damping in the wire leg will be small. The frequency of the complete structure is expected to be significantly above the control frequencies. There is a definite possibility that control system-structural vibration coupling will cause problems.

2.2.4.4 Weights. Although the 600-meter rhombic is double the size of Concept 2b Rhombic, not all component weights double, some quadruple, others remain unchanged. The weight estimate is presented in Table 2-3. The inertias in slug-ft$^2$ are:

| | |
|---|---|
| Pitch | $13.44 \times 10^7$ |
| Roll | $5.71 \times 10^9$ |
| Yaw | $13.44 \times 10^7$ |

2.2.4.5 Gross Crew and Equipment Functional Analysis. Gross crew and equipment functions are indicated in the functional flow diagram, Figure 2-10. A time line analysis (Table 2-4) provides an estimate of the length of time required to erect an operational antenna of this type. Although the analyses were performed for the center-tube torus configuration, they are not expected to differ greatly from the recommended center-tube cross.

Table 2-3. Concept 6 Weight, Volume, and Power

| EXPERIMENT (ONLY) REQUIREMENTS | ELECTRICAL POWER (watts) | PACKAGED VOLUME | WEIGHT (lb) |
|---|---|---|---|
| Basic Experiment Structure | | 1,653 $ft^3$ | 12,053 |
| Docking Port and Air Lock | | 210 $ft^3$ | 400 |
| Experiment Stabilization (ACS Sys) | | 16 $ft^3$ | 1,000 |
| Experiment Stabilization Propellant | | 40 $ft^3$ | 2,011 |
| (Electrical Power System) | Output | | |
| Electronics | (73.5) | (2.76 $ft^3$) | (120.5) |
| Wide Band Amplifier (1) | 1 | 32 $in.^3$ | 2 |
| Filter (6) | 0 | 12 $in.^3$ | 6 |
| Analog to Digital Converter (4) | 3 | 280 $in.^3$ | 5 |
| Sample and Hold Shift Register (4) | | | |
| Clock (1) | 0.5 | 32 $in.^3$ | 1.5 |
| Command Receiver (1) | 1 | 1000 $in.^3$ | 25 |
| Attitude Sensor (1) | 16 | 2400 $in.^3$ | 28 |
| Tape Recorder (2) | 28 | 860 $in.^3$ | 29 |
| Subcarrier Oscillator (4) | 2 | 8 $in.^3$ | 3 |
| FM Modulator (1) | 20 | 80 $in.^3$ | 7.5 |
| FM Transmitter (1) | | | |
| Diplexer (1) | 0 | 12 $in.^3$ | 3.5 |
| Tape Recorder Management (1) | 2 | 64 $in.^3$ | 10 |
| Total Experiment | | | 15,594 |
| Maximum Available | | | (23,992) |

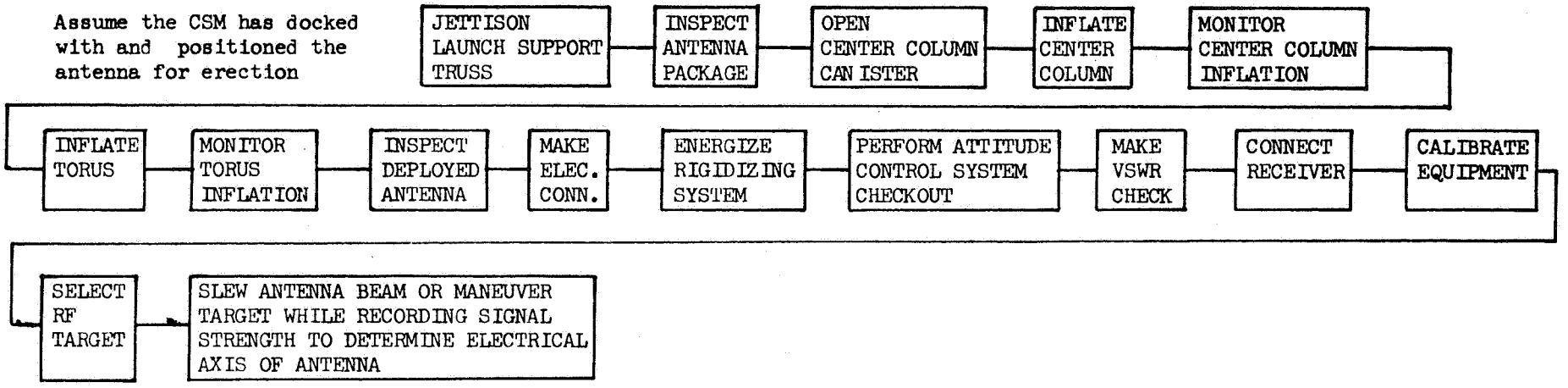



Figure 2-10. Functional Flow Diagram for the Erection of a Rhombic Antenna, Center Column Torus with Wire Antenna Elements (NASA)

Table 2-4. Time Line Analysis for the Erection of a Rhombic Antenna, Center Column Torus with Wire Antenna Elements

| EVENT | ASTRONAUT 1 | TIME (min) | | | ASTRONAUT 2 |
|---|---|---|---|---|---|
| | | EVENT | CUMULATIVE | CUMULATIVE EVA | |
| 1 | Jettison Launch Support Truss | 60 | 60 | | Perform Pre-EVA |
| 2 | | 4 | 64 | | Exit Spacecraft |
| 3 | | 15 | 79 | 15 | Don Maneuvering Device |
| 4 | | 1 | 80 | 16 | Manuever to Antenna Package |
| 5 | | 10 | 90 | 26 | Inspect Antenna Package |
| 6 | | 1 | 91 | 27 | Maneuver to Safety Area or Airlock |
| 7 | | 2 | 93 | 29 | Anchor in Safety Area |
| 8 | Open Center Column Canister | 1 | 94 | 30 | |
| 9 | Inflate Center Column | 30 | 124 | 60 | Monitor Center Column Inflation |
| 10 | Inflate Torus | 20 | 144 | 80 | Monitor Torus Inflation |
| 11 | | 1 | 145 | 81 | Release Worksite Anchors |
| 12 | | 1 | 146 | 82 | Maneuver to Deployed Antenna |
| 13 | | 80 | 226 | 162 | Inspect Deployed Antenna |
| 14 | | 1 | 227 | 163 | Return to Spacecraft |
| 15 | | 10 | 237 | 173 | Doff Maneuvering Device |
| 16 | | 4 | 241 | | Enter Spacecraft |
| 17 | Energize Rigidizing System | 1 | 242 | | Perform Post-EVA |

Table 2-4. Time Line Analysis for the Erection of a Rhombic Antenna, Center Column Torus with Wire Antenna Elements, Contd

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE | CUMULATIVE EVA | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 18 | Perform Attitude Control System Checkout | 30 | 272 | | |
| 19 | Make VSWR Check | 60 | 332 | | |
| 20 | Connect Receiver | 20 | 352 | | |
| 21 | Select RF Target | 10 | 362 | | |
| 22 | Slew Antenna or Maneuver Target While Recording Signal Strength to Find Electrical Axis | 450 | 812 | | |

2.2.5 Concept 7 — Broadside Dipole — (Reference Ryan Report No. 29174-45). The large space structure antenna consists of multiple parallel dipoles arranged along a supporting beam such that the dipole elements are co-planer, and each is of equal length. See Figure 2-11. A box beam maintains the relative positions of the dipoles. The beam and dipoles utilize the principles of the extendible spring tape.

2.2.5.1 Structure and Mechanics. A ground plane is not mentioned in the Ryan report and is not included in this evaluation even though it is considered necessary for proper RF functions. Fully deployed, it is a good structure. It stows within the stated dimensions; the critical aspect is the deployment. The main beam is not a rigid truss as it emerges from the drum and cannot be rigidized in the horizontal plane until full extension has occurred.

The corner members are flattened and rolled inside of each other on the drum. Assuming a 6-ft-diameter drum and a final stowed diameter of approximately 12 ft, an out-of-phase relationship exists longitudinally between these corner members of approximately 6.8 ft. At the outboard end, the diagonal tension ties in the upper and lower planes of each box section must be 16.8 ft, stowed, compared to 13.4 ft deployed (the beam is assumed to be divided into 10-ft incremental lengths). This phase shift discrepancy exists for approximately 57% of the outer length of the beam. See Figure 2-12.

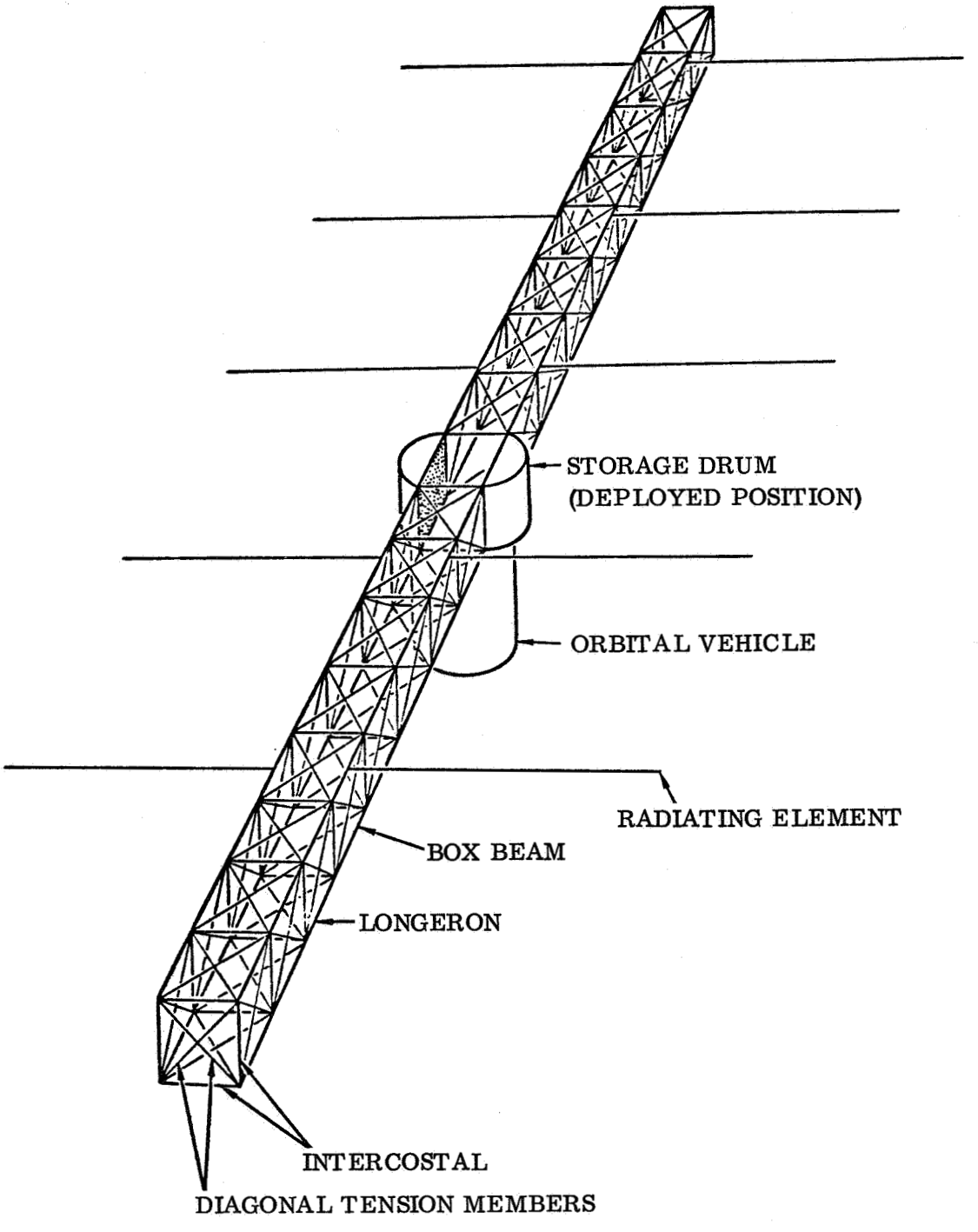



Figure 2-11. Concept 7 — 1500-Foot Dipole Antenna (Ryan)

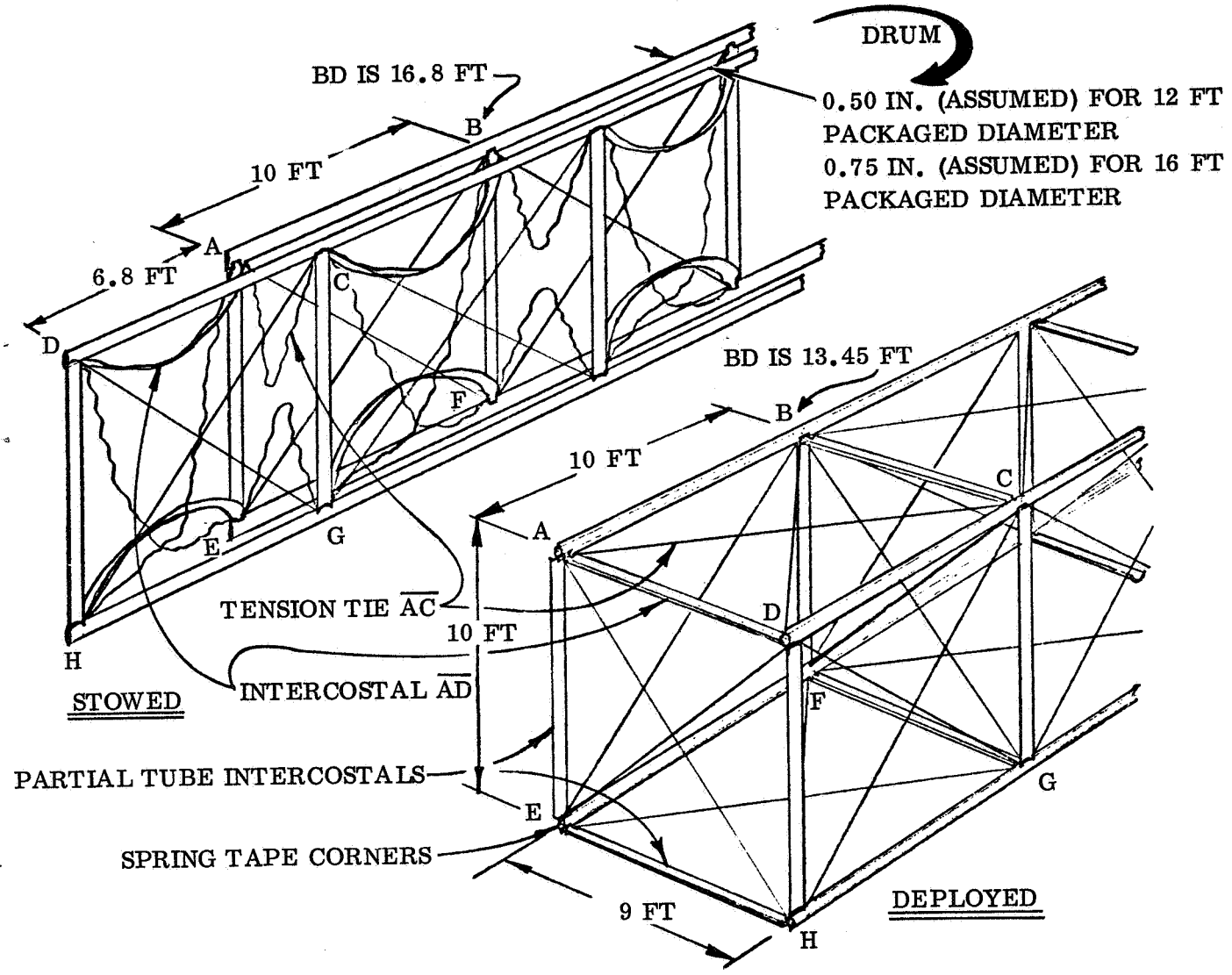



Figure 2-12. Details of 1500 Ft Dipole Antenna

The above out-of-phase relationship also affects the horizontal intercostals. The symmetrical folding and flattening of this intercostal as depicted in Figure 4 of the Ryan report occurs only at the first increment and becomes displaced at the rate of approximately 0.09 ft per 10 ft of running length. This causes a twist in each intercostal, complicating the attaching means and integrity of the member. Yet, this member provides the only horizontal spreading force required to achieve and maintain the beam depth in the horizontal direction.

Another detrimental by-product of this twisted member occurs in the stowed position. Since it no longer forms the convenient flat strip as desired, the consequence is a less compact wrap on the drum. An increase in stowed diameter to 16 ft was postulated with a resulting out-of-phase relationship of 8.5 ft.

In the stowed position the corner members are continuous and each wrap forces the underlying portion to lie flat. Not so with the vertical intercostals. They will not of themselves conform to the drum radius. At positions further outboard along the beam, these verticals overlap the twisted and displaced horizontals.

2.2.5.2 Antenna Performance. Concept 7 consists of a 460-meter boom with 61-meter dipole elements. Thus, the applicable frequency range is approximately 2.94 MHz and above for the broadside array. The lower frequencies, 0.5 to 2.94 MHz, are desirable for long wave radio astronomy, but are beyond the capability of the configuration. The antenna may also be used as an end-fire array with a modified feed system.

Broadside Array. For optimum pattern (minimum beamwidth) the maximum number of half-wave dipole elements should be employed. For this configuration the total length is 460 meters, and corresponds to an operating frequency of 2.94 MHz. The half-wave dipoles should be 167 ft (51 meters). The half power beamwidth is 13° in the plane of the array axis. The beamwidth in a plane normal to the array axis is 78°, the antenna gain is approximately 15 db. The 460-meter array length permits operation at frequencies below 2.94 MHz by compromising array beamwidth, which increases inversely proportional to the frequency. The element length of 200 ft imposes further restrictions on frequency operation below 2.94 MHz because of the impedance mismatch of elements shorter than $\lambda/2$. The element power is at a frequency of about 1.23 MHz.

End-Fire Array. The single lobe pattern produced by an end-fire array has distinct advantages over the front and back patterns of the broadside antenna, thus, the end-fire array eliminates the 180° source location ambiguity problems of broadside arrays. Optimum end-fire antenna performance is achieved when the maximum number of elements are employed at $\lambda/4$ spacing. Ten elements yields a 9/4 boom length. Therefore the maximum is 204 meters for the Ryan concept. This corresponds to an operating frequency of 1.47 MHz and requires (102 meter) 334-ft dipoles, which exceeds the 200 meter length specified in the concept.

Assuming the original 61-meter dipole length is constant, at $\lambda/4$ the wavelength is 244 meters or 1.23 MHz. At this frequency a 10-element array requires $9/4 \times 244 = 550$ meters boom length. The beamwidth of this array is 40° x 40°.

Summary. The concept presented is too small to achieve the desired 10° x 10° beamwidth user requirement. The broadside array can provide narrow beam definition in one axis and has a directional ambiguity problem which could be eliminated by installation of a ground plane behind the array elements. The end-fire array produces a unidirectional beam but cannot approach the desired 10° x 10° beamwidth.

The 200-ft dipole length restriction establishes a minimum operating frequency of 1.23 MHz for both broadside and end-fire arrays. The upper frequency limit is reached when the element spacing exceeds approximately 1 wavelength. At frequencies greater than this, the pattern side lobes become excessive and may cause ambiguities in response. The 460-meter, 10-element array length corresponds to a wavelength of 460/9 = 56 meters or 5.35 MHz.

2.2.6 Concept 13, ORAO Rhombus Antenna (TRW). The Orbiting Radio Astronautical Observatory is a single-plane rhombus antenna deployed in 136 hours to a radius of 13,180 ft by centrifugal force, which rotates once every 6.5 minutes. The aluminum wire antenna and glass filament supporting legs, including tips masses, weigh 32 lb. The 400-lb gross payload weight includes a power supply, communications equipment and structure. A motor at the hub of the antenna is used to alter the diagonals in the antenna to vary the rhombus apex angle. See Figure 2-13.

The 400-lb experiment package shape is adaptable to many manned Saturn V launch configurations that are injected into a synchronous orbit for one year mission without resupply. The satellite is deployed automatically and requires no EVA.

The propulsion system employs two monopropellant hydrazine engines of 0.43-lb thrust and 203-lb of propellant to spin the spacecraft and antenna.

The electrical power system consists of a solar array that provides 30 watts of power, two nickel cadmium batteries, converter and a command distribution unit.

The antenna will operate in an elliptical orbit inclined at 62° (corresponding to the plane of the galaxy), with a perigee altitude of 500 n.mi. and apogee altitude of 25,000 n.mi., 10 hr of the 14-hr period will be spent in the region surrounding the earth termed the plasma trough, where interference from ionospheric or Van Allen electrons is minimized for ORAO orbits. As the spin axis of the observatory is altered by the effects of gravity, the 6° main beam of the antenna (in the plane of the antenna) will sweep out the entire sky, except for small holes at the galactic poles. This survey will take less than two months and thus the sky can be surveyed on six different frequencies in the course of a year. Alternatively, two observatories in orbit whose spin axes are mutually perpendicular will provide complete sky coverage including the galactic polar regions in even less time.

2.2.6.1 Structure and Mechanics. The 400-lb experiment payload is within the allowable payload capability of the Saturn-V manned launch configuration for a synchronous orbit. Sufficient payload volume is afforded in all AAP launch configurations. Figure 2-14 indicates a typical launch configuration with the SSESM. Since the antenna concept is deployed automatically, a launch configuration could also be employed that utilizes a nose fairing without the manned CSM.

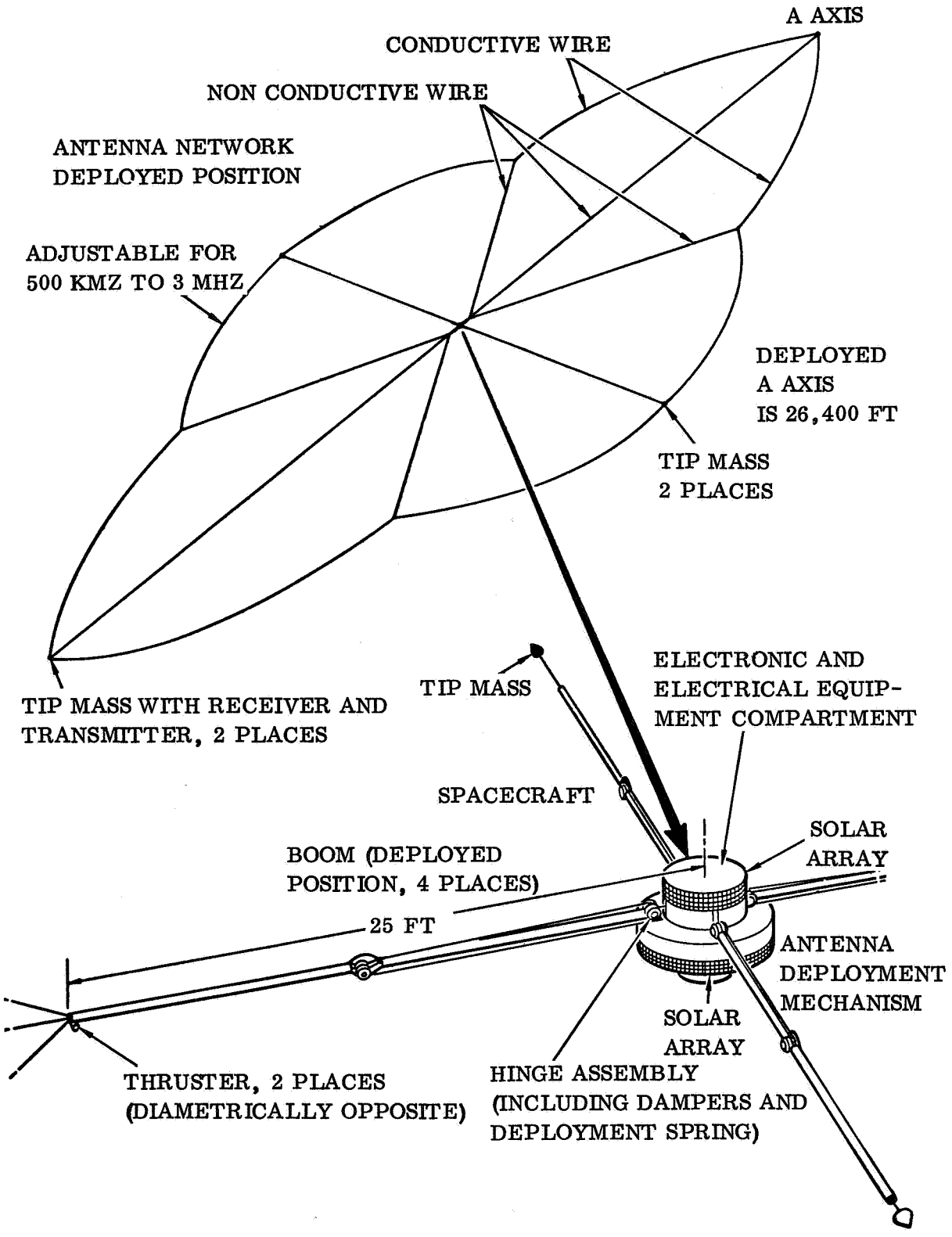



Figure 2-13. Concept 13 ORAO Rhombus Antenna (TRW)

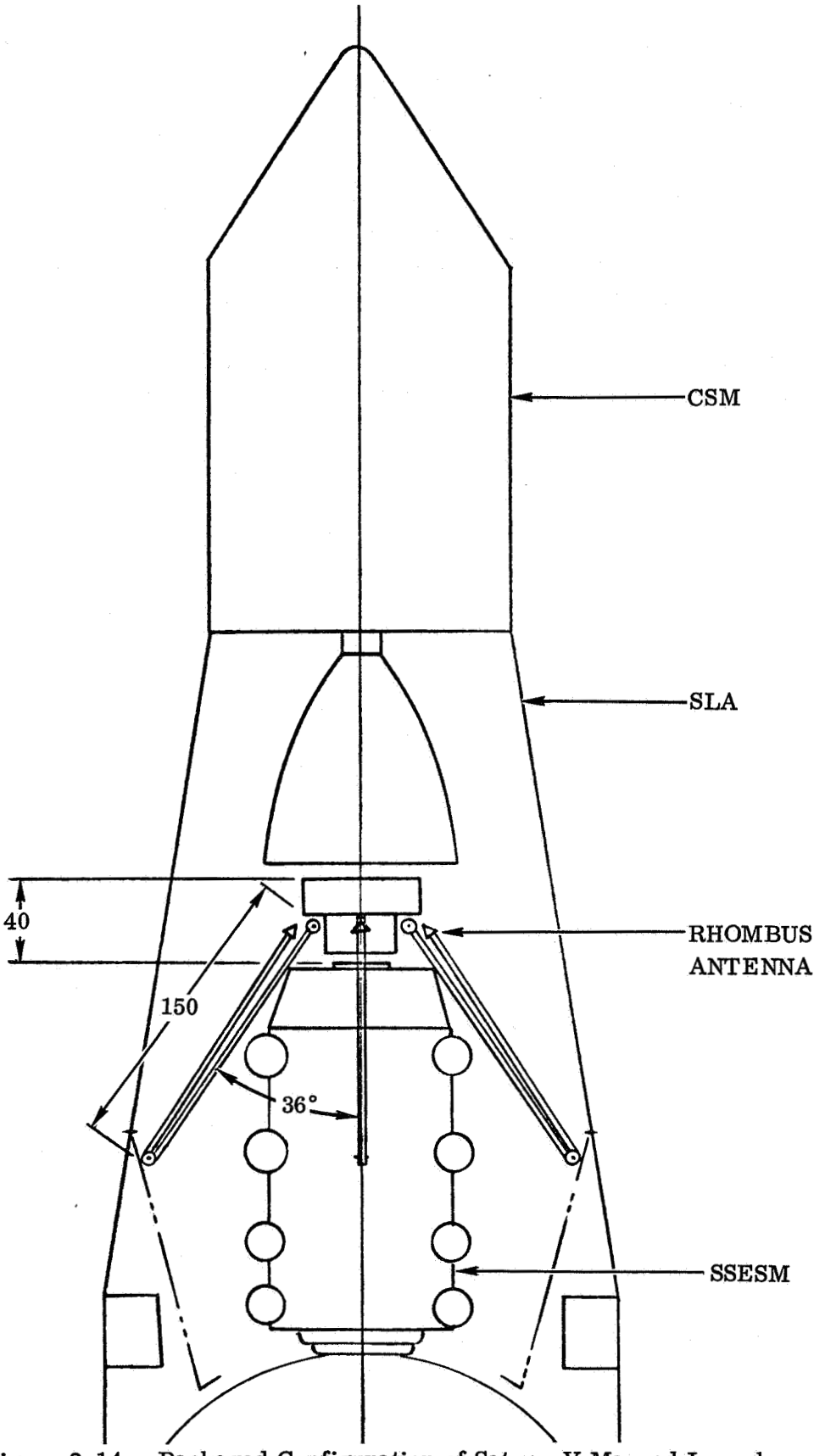



Figure 2-14. Packaged Configuration of Saturn-V Manned Launch Configuration and ORAO Rhombus Antenna

2.2.6.2 Antenna Performance. A summary of electrical characteristics of the concept is shown in Table 2-5 for comparison with user requirements. In general, the concept appears to have considerable potential for long-wave astronomy applications.

Orbital altitude for the concept varies from a 500-mi. perigee to a 32,000-mi. apogee. As a result the antenna is flying part of the time in regions where the ionosphere disrupts the performance of the antenna. About 69% of each 14-hr orbit is spent beyond the effect of the ionosphere; the rest of the time the antenna reception will be affected by the ionosphere. A higher orbit would be desirable.

Table 2-5. Comparison of User Electrical Requirements and Concept Characteristics, Concept 13

| | SPECIFIED USER REQUIREMENT | CONCEPT CHARACTERISTIC |
|---|---|---|
| Lifetime | Minimum of one year | Meets requirement |
| Orbit Altitude | Minimum of synchronous | 1.125 earth radii to 8 earth radii (500 mi. perigee to 25,000 mi. apogee) |
| Effective Beamwidth | 100 $deg^2$ at 1 MHz desired. Interferometers if possible to improve resolution to 2° | 6° in E-plane at 1 MHz. 17° in H-plane at 1 MHz. No interferometer provision |
| Pointing Accuracy | 1/2 beamwidth minimum to 1/10 beamwidth | |
| Bandwidth | 0.5 MHz to 10 MHz | 225 kHz to 10 MHz with adjustment of apex angle |
| Spectral Resolution | Good desired, depends on electronics | Meets requirement |
| Dynamic Range | Unfilled apertures are adequate | Meets requirement |
| Lock-on Time | 1/2 sec to several hours for time varying phenomena | Constant spin at 0.016 radian/sec. Time for scan through half-power beamwidth — 6.5 sec |

A characteristic requirement of the concept is the spinning of the antenna about an axis normal to the plane of the elements. This characteristic is essential to maintaining proper antenna structure deployment. The spin rate of 0.016 radian/sec, across the 6° beamwidth of the antenna, yields a maximum scanning time of about 6.5 sec for any one point in space.

2.2.6.3 Dynamics Attitude Control. This concept is simple, lightweight, and uses passive attitude control techniques. These advantages are attained at the expense of a protion of the user requirements. Target lock-on is not possible by physical orientation. The concept obtains lengthy target looks by attitude sensor outputs and radio signal data integrated by a digital computer, which will require considerable data processing. Celestial coverage is reduced by the holes around galactic poles and by the low altitude of perigee. Pointing accuracy of the concept is designed for attitude tolerances of a few degrees, and this may not be tight enough.

The major dynamics/attitude control feasibility question is net deployment. Possibilities for instabilities can exist in spin up erection techniques. As noted by TRW, additional net stability analysis is needed. However, analytical solutions to this problem will not satisfy all questions of feasibility. Experimental verification is required. This testing must be performed in space, however, the concept size weight is small enough so that a low-cost full-scale net may be deployed for the test in conjunction with some other experiment.

2.2.7 Concept 17 — Log Periodic Antenna (Fairchild-Hiller). The antenna concept proposed by Fairchild Hiller is intended for radio astronomical observations in the low megacycle region. The antenna is pyramdial with two sides serving as log periodic antennas as shown in Figure 2-15. The spacecraft module is supported in the geometrical center of the antenna by four tubular extendible elements and a concertina-type column. The column and the four tubular extendibles restrained by stay wires form the pyramid. The antenna elements are attached to two opposing stay wires.

For long-wave radio astronomy, the user specifies that the antenna satisfy the frequency coverage of 0.5 to 10 MHz, a one-half power beamwidth of 10°, and a resolution of 2°. The concept proposed can achieve a frequency coverage of 2 to 20 MHz, a one-half power beamwidth of 50° × 60°, and a resolution of 10° coupled with an interferometer system. To satisfy the frequency coverage only, the antenna requires scaling up from 500 to 2,000 ft. Despite the size required, it can be packaged in the Saturn-V vehicle. However, a complete analysis of this concept was not performed, since the half-power beam width is considerably greater than desired.

2.2.7.1 Antenna Performance. The log periodic antenna presented is applicable to frequency-independent operation. Gain and beamwidth are sacrificed for wideband performance. The antenna characteristics shown in Figure 2-16 illustrate that beamwidth does not approach the limits requested for long wave radio astronomy. The minimum beamwidth achievable is approximately 50° by 60°.

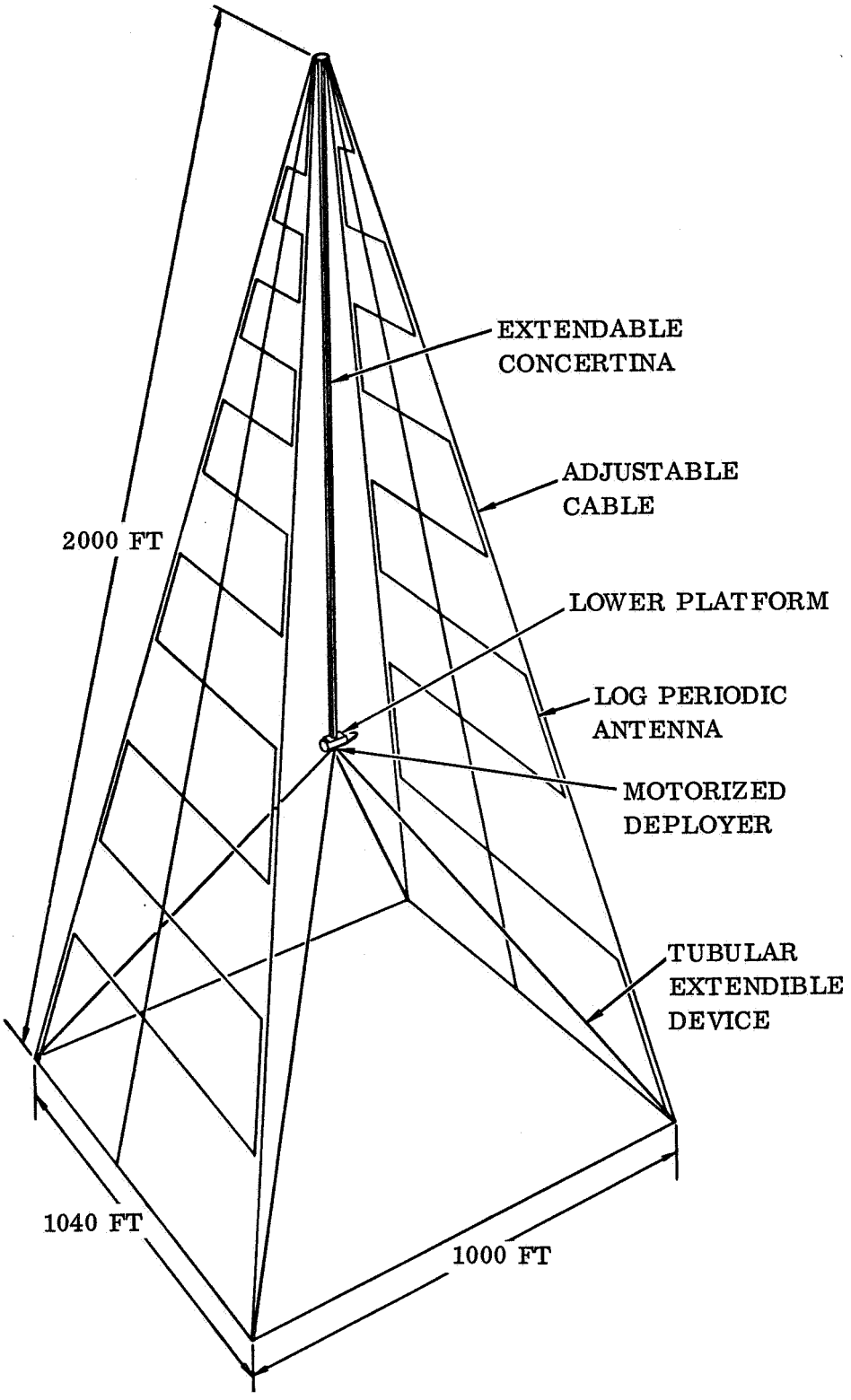



Figure 2-15. Concept 17, Log Periodic Pyramid Antenna (Fairchild-Hiller)

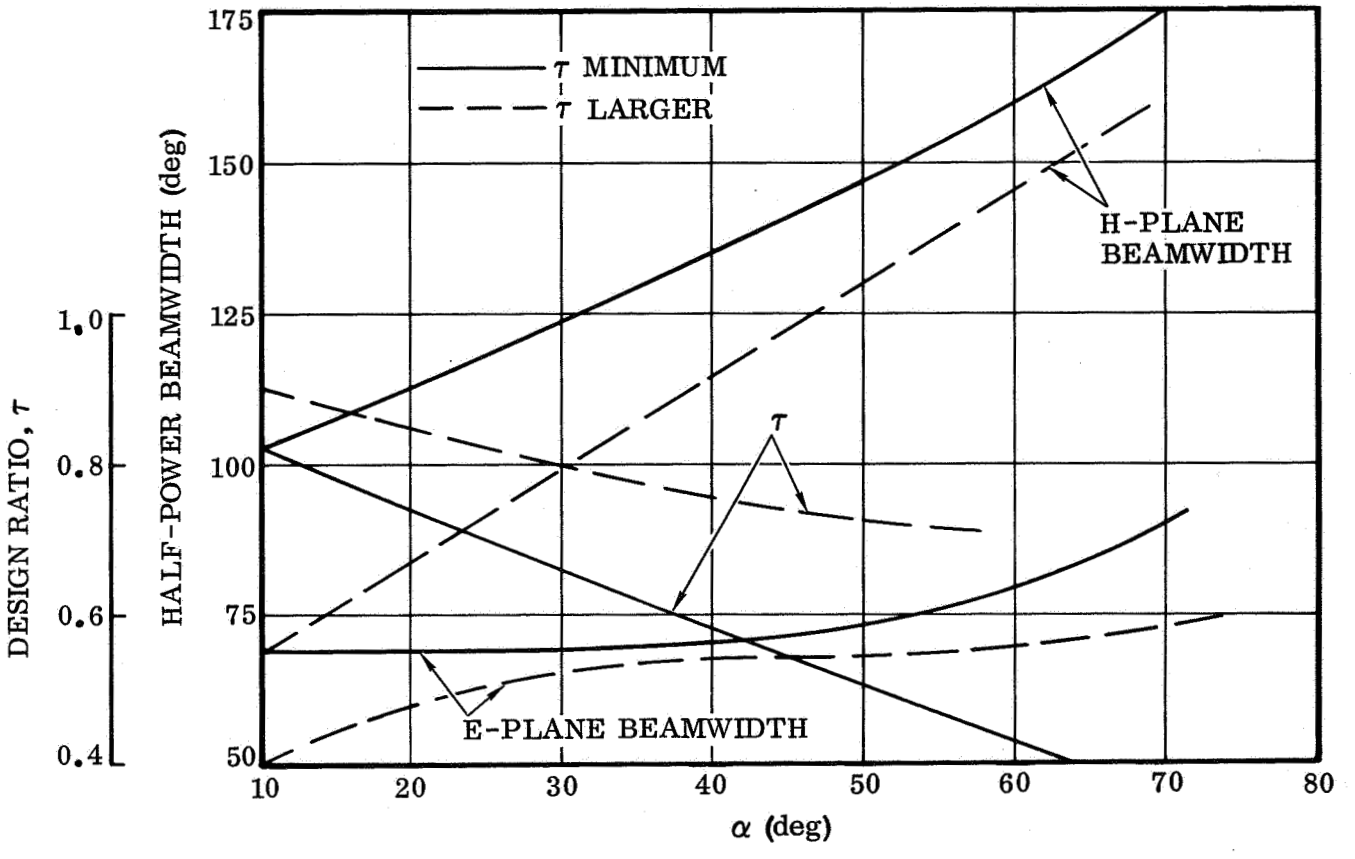



Figure 2-16. Log Periodic Antenna Pattern Characteristics (Wire Trapezoid Tooth Elements)

2.2.8 Concept 20 — Rhombus Antenna (Fairchild-Hiller). The antenna is a single plane structure consisting of two crossed tubular members as columns to support flexible antenna element wires attached at the tips to form the rhombus. See Figure 2-17. The structural members are formed from extendable spring tapes. Application of conventional design criteria for solid-wall wrapped tubes indicates that stress and weight limits restrict the overall length to approximately 2000 to 2500 ft. An advancement in the present spring-tape or extensible-member technology is necessary to obtain stable structures of greater length and at reduced weight.

The concept will not meet the user requirements of 0.5 MHz to 10 MHz frequency at beamwidths approaching 10°. To meet these requirements, rhombic leg lengths on the order of 20 wavelengths are required (12,000 meters at 0.5 MHz).

2.2.8.1 Structure and Mechanics. The extending spring tape device is the critical mechanical/structural item influencing concept feasibility and limitations.

Deployment and Structure Assembly. Two modes of antenna deployment are suggested by Fairchild. One mode deploys the crossed structural members having the outer rhombic active antenna element (flexible cable) attached to the ends of these members. The second mode consists of broadening the operating frequency band by adding elements via EVA. It is also implied that active control of extension and retraction could be used, although the concept description generally refers to a fixed geometry antenna.

The large antennas that are necessary to meet user and RF requirements will require tubular extendible devices with diameters 1 ft or greater to withstand the imposed loads of docking, maneuvers, environment, and crew. This also affects the mechanisms. For instance, the crossed extendible members should be arranged such that their reactive torques balance each other. Typical extendible tubes possess a considerable amount of strain energy when stored, and the reactive force of deployment acts over a fairly long period. In a time-stepped method of deployment to prevent over-stressing, a 1000-ft length can take as long as 29-hours.

The configuration and materials of the tape poses a development problem. As noted in the Fairchild report, analytical studies and development work for the characteristics of tubular spring tapes has been limited to date to diameters generally not greater than 1 in. Of certainty, this knowledge can be applied to larger diameters, but the practical benefit of development hardware is not available.

Another complicating factor lies in the antenna requirement that this spring tape should be a dielectric. Dielectric tapes have been produced; however, they suffer the greatest from the factors most detrimental to their structural use — high thermal distortions and low stiffness. An alternate solution would be to have dielectric interruptions of a metallic tape at intervals of approximately one-tenth of the short end of the operational wavelength.

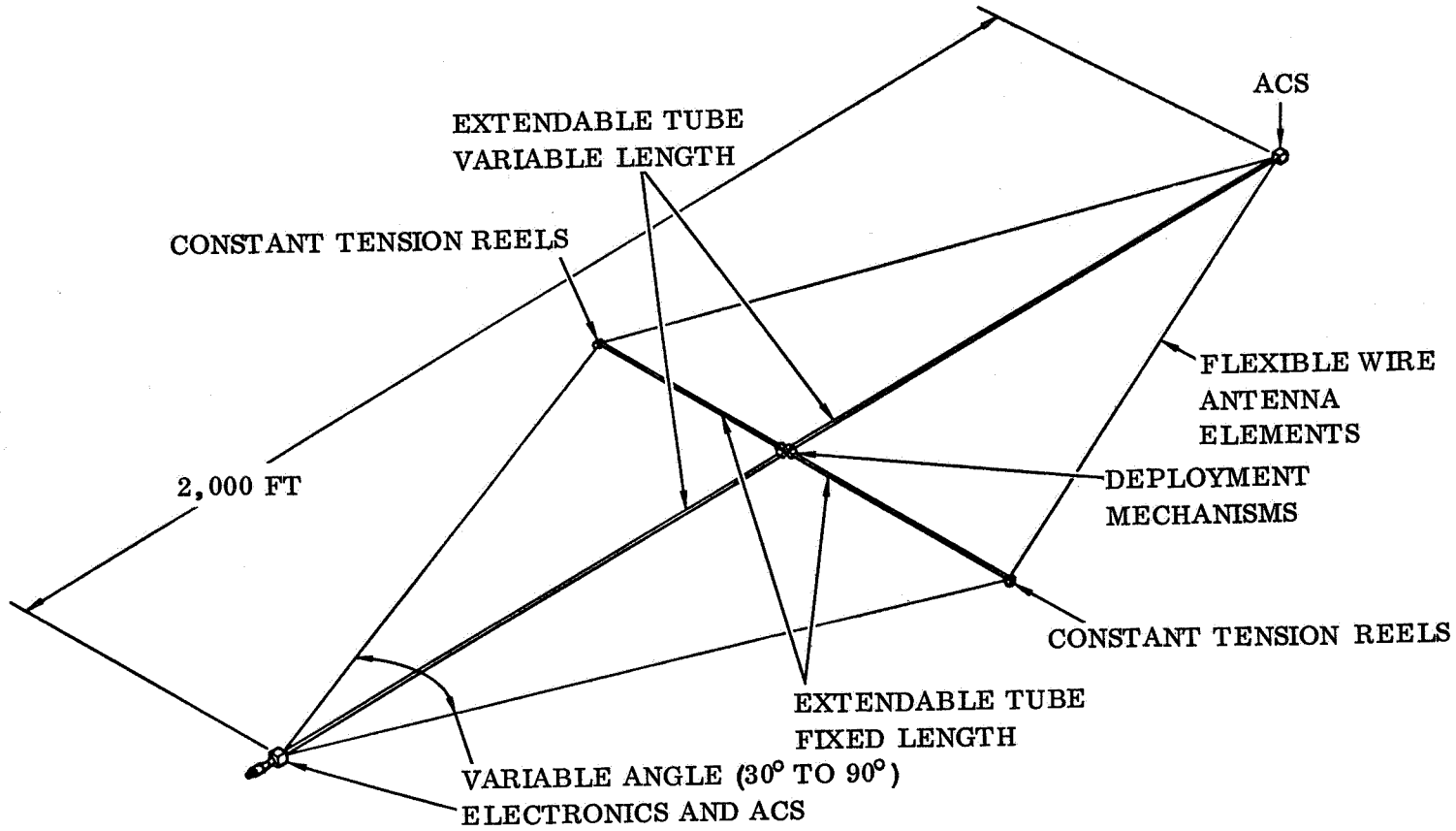



Figure 2-17. Concept 20 — Rhombus Antenna (Fairchild-Hiller)

Solid-wall tubular members are subject to large deflections because of a large thermal gradient. When used as an antenna dipole or as a gravity gradient device, this bending, though undesireable, does not have the ramifications encountered in this closed structural application. Warpage of the crossed members can distort the antenna to impractical properties; secondly, the antenna element tensions can produce loads over-stressing the structure. To counteract this, two design innovations should be considered:

a. The flexible cable antenna elements should be installed on constant tension reels with sufficient reel take-up such that they do not permit over-stressing of the structure, nor themselves, because of geometric deflections.

b. Use of an open-screen type of tape to limit deflections initially.

An ideal material would have high thermal conductivity and small thermal expansion. Use of the open-screen concept substantially reduces solar pressure distortion, permits application of optimum internal and external coatings, and permits the independent selection of materials for conductivity and low thermal expansion. An extendible tape has been developed by Convair that is designed especially to combat the thermal distortions. The tape material is a wire screen having Elgiloy longitudinal elements for low thermal expansion and Be-Cu circumferential elements for good circumferential heat distribution. Being an open screen permits direct solar incidence on the inside as well as the outside of the tube.

The conclusion here is that the entire basic configuration is well suited to automatic deployment. EVA would include initial alignment of deployment components at the support facility, assist where deployment problems might arise, adjustments and/or securing after deployment, and maintenance.

2.2.8.2 Antenna Performance. The maximum rhombus leg length of approximately 500 meters will not provide the performance that is desired in the User Requirements. The primary difficulty is in achieving the 10° beamwidth at 1 MHz. This configuration yields a 25° x 70° beamwidth at an apex angle of 70°. At 6 MHz ($\lambda$ = 50 meters) the beamwidth narrows to 13° x 22° at a 35° apex angle. See Figure 2-3.

Rhombic antennas require a high degree of pointing accuracy. This accuracy is influenced primarily by the factors of attitude control sensitivity and electrical beam pointing. The electrical beam pointing is critical relative to the rhombus apex angle. Extraneous conditions, such as thermal distortions, can vary this angle, yet the angle change must be known to determine the true RF beam direction and shape.

The Fairchild report recommends extending the bandwidth of operation by adding other rhombic elements to the structure. Since this creates rhombus antennas of shorter element lengths than the original, it pushes the operation to higher frequencies whereas the prime area of interest is coverage at the low frequencies (0.5 - 10 MHz). Antenna

performance is limited by the physical size. The single-plane rhombus also suffers from its response being limited to the incident energy polarized in the plane of the rhombus. Another similar structure would be required to respond to the energy polarized in the normal plane.

The major problem areas are listed:

a. The antenna as described cannot respond to the complete range of possible polarizations of incident energy.

b. The beamwidth does not approach the $10^{\circ} \times 10^{\circ}$ desired at 1 MHz.

c. The performance (beamwidth) below 2 MHz is severely degraded.

d. The pointing accuracy capability could be a serious consideration in completing a design for use as an experiment.

2.2.8.3 Dynamics. The dominant dynamic characteristic of this structure is its weakness in the out-of-antenna plane. Modal frequencies may be of the order of thousandths of Hz. The coupling between the control system and structural vibrations and the resulting deflections will probably prohibit maneuvering the structure from target to target by applying jet forces out of the rhombus plane.

Maneuvering can be accomplished using jets thrusting in the structural plane, where the vibrational characteristics are more favorable. This is done by spinning the antenna and reorienting by impulse torquing the spin axis. The propellant weight required is in the range 0.1 to 1.0 lb per maneuver. Maneuvering rates of one per hour are thus weight expensive. Dampers are required.

The spin axis should be the maximum inertia axis. This moves the antenna pattern past radio sources once per spin. Useful data are then obtained from a relatively complex data processing and digital computation procedure.

The spin approach can scan the sky using jet thrust in the antenna plane to slowly precess the spin axis. Pointing accuracy requirements should be easily met, and the complete celestial sphere can be covered. Bowing of the rhombus legs due to centrifugal forces would most likely require an additional radial tension member to the mid-point of each leg.

Gravity gradient stabilization, with active control augmentation, is a promising technique of scanning for the non-spin approach. Pointing accuracy requirements should easily be met. However, a portion of the sky will not be covered.

2.2.8.4 Stress Analysis. A critical stress situation can occur at termination of deployment unless a controlled departure rate is used that is a function of deployed distance. If an assumed velocity of 1 in/sec were suddenly brought to zero, the stress

at the restrained end is 115 psi, the required $L/\rho$ is 1210 and the diameter for a 1000 ft length must be 28 in.

This diameter is excessive since the allowable wall thickness to meet a weight bogey of 15,000 lb would be 0.008 and would have a diameter to wall ratio (d/t) of 3500. It is evident that the departure velocity must be gradually reduced as the tube extends.

If one were to consider an 9 in. diameter, 0.032 in. wall tube and a 2000 ft length, the allowable column load is 1.88 lb. The allowable cable tension therefore must not exceed 0.94 lb. Such a small force imposes severe sensitivity in the tension control mechanism end yet must be ample to react maneuvering loads.

2.3 ANALYSIS OF NEW CONCEPTS. Some of the concepts investigated in this section were found to be unfeasible for various reasons. These concepts are presented in the depth to which they were investigated.

2.3.1 Concept 51a — Colinear Dipole Antenna. This concept is a single structural tube composed of alternately conductive and non-conductive section, which constitute the colinear dipole arrangement. See Figure 2-18. The 5400-meter length allows which dipoles 300-meters long for 600-meter wave length reception. At one end is an array of two crossed rods at right angles, which comprises a polarimeter and completes the antenna configuration. Between each dipole the boom section of non-conductive material is mechanically pivoted to eliminate the accumulation of distortion due to solar deflection.

The antenna leads are insulated wires electrically bonded by the astronaut to the midpoint of each dipole as the boom is extended. Between bond points the wires are also secured to the basic structure during extension by insulated standoffs at maximum spacing. The leads extend to the electronics capsule which is mounted on the bottom end of the boom.

At the top end of the boom the right angle cross of dipoles is mounted a complete assembly by the astronaut and is later extended remotely.

A thruster unit is also mounted in the top of the boom to provide tension in the boom to reduce any tendency to twist and curl as it extends. If extended in the direction of the gravity gradient, that force will be stabilizing as the boom gets longer.

Using earth's gravity gradient to position the axis of the major boom, the celestial sphere can be covered in a matter of months, depending on the orbit inclination. A 60,000-mile orbit is desired, according to Haddock, to avoid magnetosphere influence on the electrical characteristics of the antenna. However, for gravity-gradient stabilizing, forces are greater in a synchronous altitude orbit considered for this study. See Appendix C.

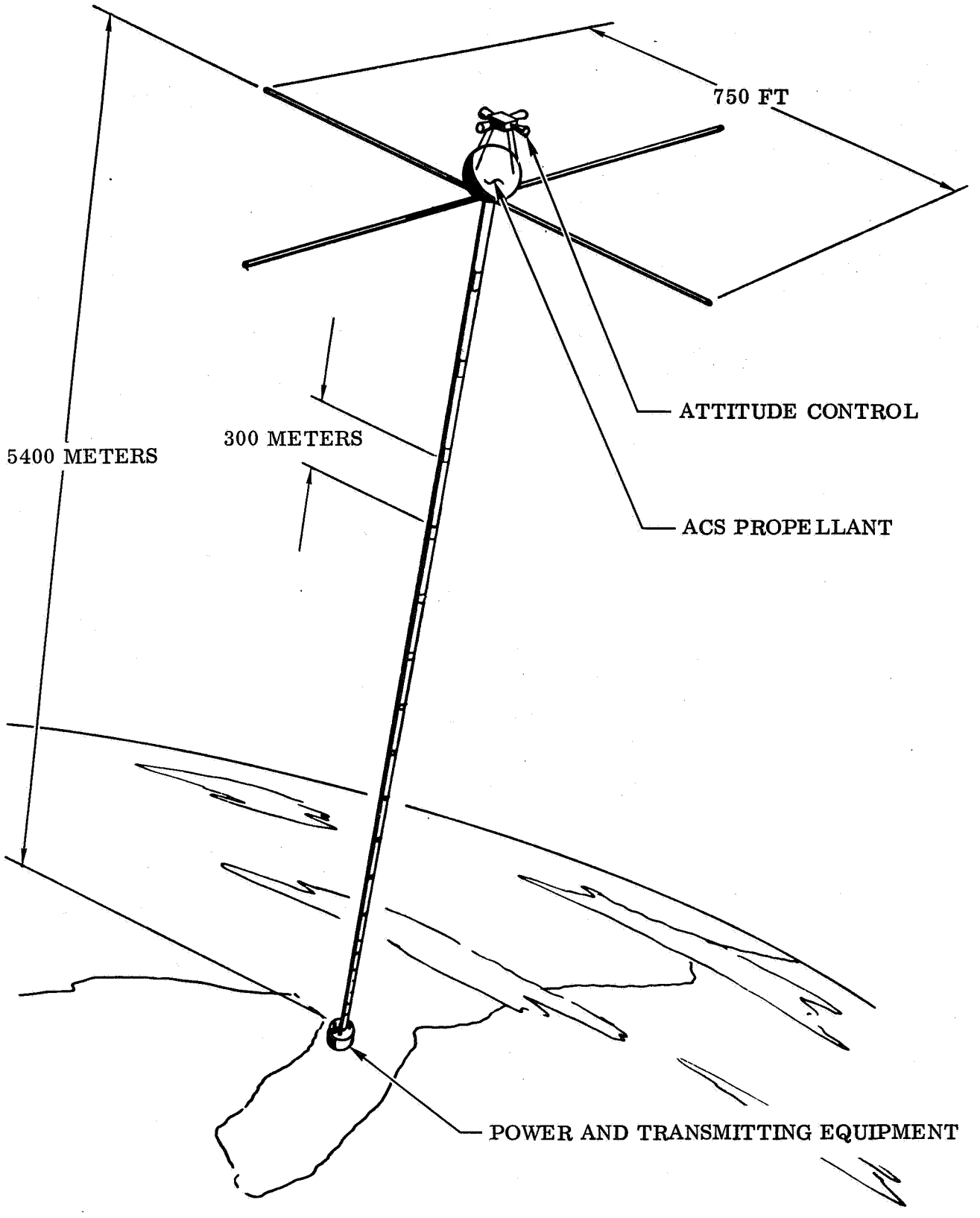



Figure 2-18. The Colinear Dipole Radio Astronomy Antenna

Stabilization about the gravity-gradient position is desired to provide consistent coverage of the celestial sphere. This is accomplished by using the crossed dipole as an oscillation damper. The booms are mounted such that as the primary boom oscillates from the line of gravity, the damper booms tend to oppose the motion and dissipate energy by means of a viscous damper such as the one developed by Convair for OV1 experiments.

2.3.1.1 Structure and Mechanics. The dipole elements are cylinders constructed of a conductive metallic mesh or perforated sheet. The insulating sections between the dipole elements are pairs of plastic drums sized to fit within the ends of the dipole elements. The two drums of each pair are tied together through a universal joint and provide the dipole separation required for electrical isolation. A circumferential toggle clamp placed at the overlap of dipole and drum provides the structural continuity required of the boom.

The elements for stowage on the spacecraft are a reel for the primary boom, a reel for the lead-in wires, 36 pairs of attachment cylinders, one crossed dipole reel unit with viscous damper, one thruster with propellant, controls, etc., and radio instrumentation with its auxiliary equipment such as data storage and telemetry.

The unique items are as follows:

| | | |
|---|---|---|
| a. | Primary boom reel | 10 ft dia. × 8 ft |
| b. | Brakes, guides, rolls, and holding fixtures | 15 ft × 5 ft × 8 ft |
| c. | Lead-in wire reel | 1.5 $ft^3$ |
| d. | Attachment cylinders, 36 required, each | 30 in. dia. × 16 in. |
| e. | Dipole reel units, 2 required, each | 2 $ft^3$ |
| f. | Tools, etc. | 3 $ft^3$ |

The location requirement on the spacecraft is simple that the primary boom can be extended along the earth's gravity gradient. This will probably require the holding fixtures to be transferred from launch position to deployment position.

To simplify deployment, the control unit is stowed complete, including fuel, on the spacecraft. The two crossed dipoles are stowed on a single reel mounted on the damping device, caged, and attached to the base of the control unit.

The metallic cylinders are all stowed consecutively on a single reel mounted on the spacecraft and aligned with the edge-joining device and holding fixture discussed below. The spacecraft is positioned and held in an attitude that aligns the subsequent boom deployment with the gravity gradient and away from the earth. The astronaut starts the tape through the joining rolls and lets it extend just far enough to accept the controls unit attachment plug. With the aid of an AMU he maneuvers the control unit into alignment with the cylinder, and clamps the two together.

As the strip reel unwinds itself (due to stored spring energy), the astronaut manipulates the reel brake and supervises the extending boom, operating the control unit on the end of it if necessary.

When each dipole section is extended its cylindrical end is restrained by the holding fixture while the plastic drum is inserted and clamped. At appropriate points the lead wires are electrically attached to the dipoles and the harness is attached to standoffs clamped to the boom.

Upon extension of the last cylindrical section the capsule containing the electronic gear is aligned and clamped into place. Its antenna is deployed, and the long wave antenna is ready for checkout and separation.

2.3.1.2 Antenna Performance. Bandwidth of 0.25 MHz to MHz and beamwidth of 12° × 360° at 0.25 MHz is significantly below the desired characteristics. The omnidirectional nature of the antenna beam in the plane transverse to the longitudinal axis of the array is a particularly undesirable characteristic, because it creates ambiguities in the response of the antenna that are impossible to resolve, making the value of the antenna relatively low for astronomy use.

The gain of 8 db at 0.25 MHz and 14 db at 1 MHz will satisfy user requirements, since at the long-wave astronomy frequencies the sky brightness is relatively high.

2.3.1.3 Attitude Control and Dynamics. The major attitude control problem associated with this concept is maintaining the local vertical orientation. The dominant disturbance is solar pressure. The area is 41,400 $ft^2$, and the total maximum solar force is 0.00408 lb. The disturbance pitch torque is 0.00408 lb-ft for each foot of vertical cg-cp offset. The gravity-gradient torque is estimated usin $2.37 \times 10^9$ slug-$ft^2$ for the transverse inertia. Pitch gravity torque is 0.653 ft-lb for each degree of pitch error. Thus, an offset of 160 ft can result in 1° error and 400 ft can result in 2.5° error. Noting that 400 ft is 2.44% of the total length, it appears that pitch orientation can be controlled to 2.5°. The cumulative friction in the universal joints between the dipoles could suffice to damp oscillations about the line of gravity. Similar comments apply to maintaining vehicle attitude for roll about the velocity vector.

The solar torque in yaw about the vertical axis is likely to be quite large. The solar force on the tube (and thermal distortion) will cause bowing. This results in a large offset. The passive yaw attitude control is proportional to the difference in the two transverse inertias. The advantage that can be obtained by placing weights at the tips of one of each dipole pair is small. The resulting yaw stiffness is small, and the solar wind will rotate the vehicle. Since the antenna pattern is symmetrical for the colinear dipole there is no requirement for vehicle yaw control. The vehicle can weather vane into the solar wind.

The crossed dipoles lie in planes parallel to the earth's surface. This is a neutrally unstable position for gravity torques. The dipoles are torqued toward the vertical and are restrained by the tube. The gravity torques on the dipoles are $8.01 \times 10^{-5}$ ft-lb per degree of roll about the vehicle velocity vector and $6.01 \times 10^{-5}$ ft-lb per degree of pitch. The primary boom will bend in providing the restoring torque, and the dipole rod will rotate out of the horizontal plane. Active control can be used to hold the dipole rotation to the desired tolerances if necessary.

2.3.1.4 Thermodynamics. Thermal distortion was investigated for both the crossed dipoles and the colinear dipole. If the element deflection is limited to 1/10 wavelength (60 m) the relationship between the element length and the allowable radius of curvature due to distortion can be determined from Figure 2-19. Approximate regions of radius of curvature are indicated for three types of element material. It can be seen that material type selected for the crossed dipoles is not critical due to the allowable radius of curvature of about 350 ft.

Distortion of the colinear dipole, however, requires a much larger radius of curvature to remain within the 1/10 wavelength limit. Thus, the material type chosen for this element must employ special techniques to reduce the thermal gradient to nearly zero.

2.3.1.5 Crew Systems. Functional and time-line analyses were performed. The five days of EVA required for erection is considered reasonable, but failure modes should be studied.

The following activities involved in erecting this antenna require extra attention by the astronaut:

a. The spacecraft should be positioned and held in an attitude that aligns the subsequent boom deployment with the gravity gradient.

b. Establish that the heat and pressure rolls for the tape (which forms the tube) are functioning properly at all times to prevent a tube being improperly formed or a delay in completion of the task.

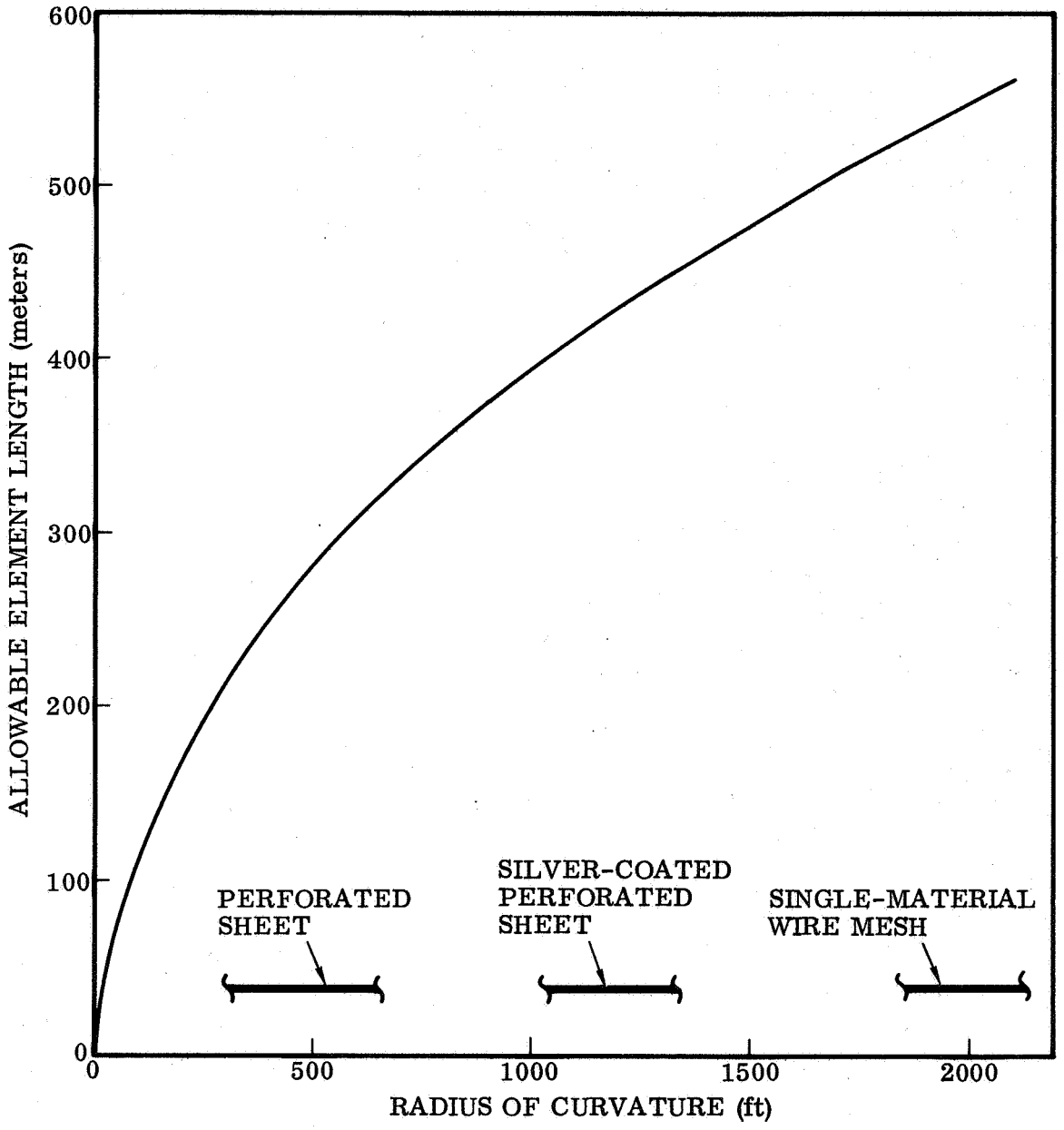



Figure 2-19. Allowable Element Length vs Radius of Curvature for 1/10 Wave Deflection of 600M Antenna

c. Assure that the formed tube is firmly clamped in the holding fixture to preclude damage or loss.

d. Extreme care must be exercised by the astronaut to avoid damage to the wire meshed dipoles (tubes) while fitting insulating drums into the tubes.

2.3.2 Concept 51b, Crossed-H Interferometer. The concept, Figure 2-20, consists of two adjustable end-fire antennas, each composed of two crossed "lazy H" dipole pairs. The end-fire antennas are connected by a tether which is adjustable from a center deployment box which also contains the telemetry and RF electronics equipment. The array separation is adjustable from 0 to 10,000 meters. Power for the antenna is provided by solar cells mounted on the surface of the electronics capsule at each end of the array. The attitude control and deployment thrusters, together with the fuel supply, are also located at each end of the antenna.

The dipoles which are electrically isolated are supported by the dipole heads which are in turn secured to the ends of the booms. Each of the four dipole heads provides the mounting surface for an ACS unit together with laser beam and photo cells, all of which are used for alignment and stabilization. The center section housings contain the electronics, telemetry, data and electrical power units.

The flight package contains all the necessary elements for automatic self erection upon radio command. After the antenna package has been inserted into the desired orbit, the antenna will be deployed; the two boom assemblies still packaged in their center section housings will be released and separated in opposite directions from the cable drum housing. Thruster units mounted on each assembly insure separation. The rate of separation is controlled by the cable, drum, and brake mechanism within the cable drum housing; this restraint puts the cable under tension and prevents it from becoming tangled during deployment.

The second phase of erection takes place when the packaged boom assemblies have been separated by 200 meters and stopped. At this point the support booms are fully extended from their respective center section housings and then the crossed dipole sets are extended from their respective boom ends. Upon extension of all booms and dipoles and controlled testing of all thrusters, the antenna functions are given a preliminary checkout. Upon satisfactory completion of the preliminary checkout, the extended boom assemblies are separated by their thrusters to the full deployment of the interconnecting cable.

Throughout the deployment sequence the astronaut is provided with the capability of manually overriding automatic deployment. Refer to Figure 2-21.

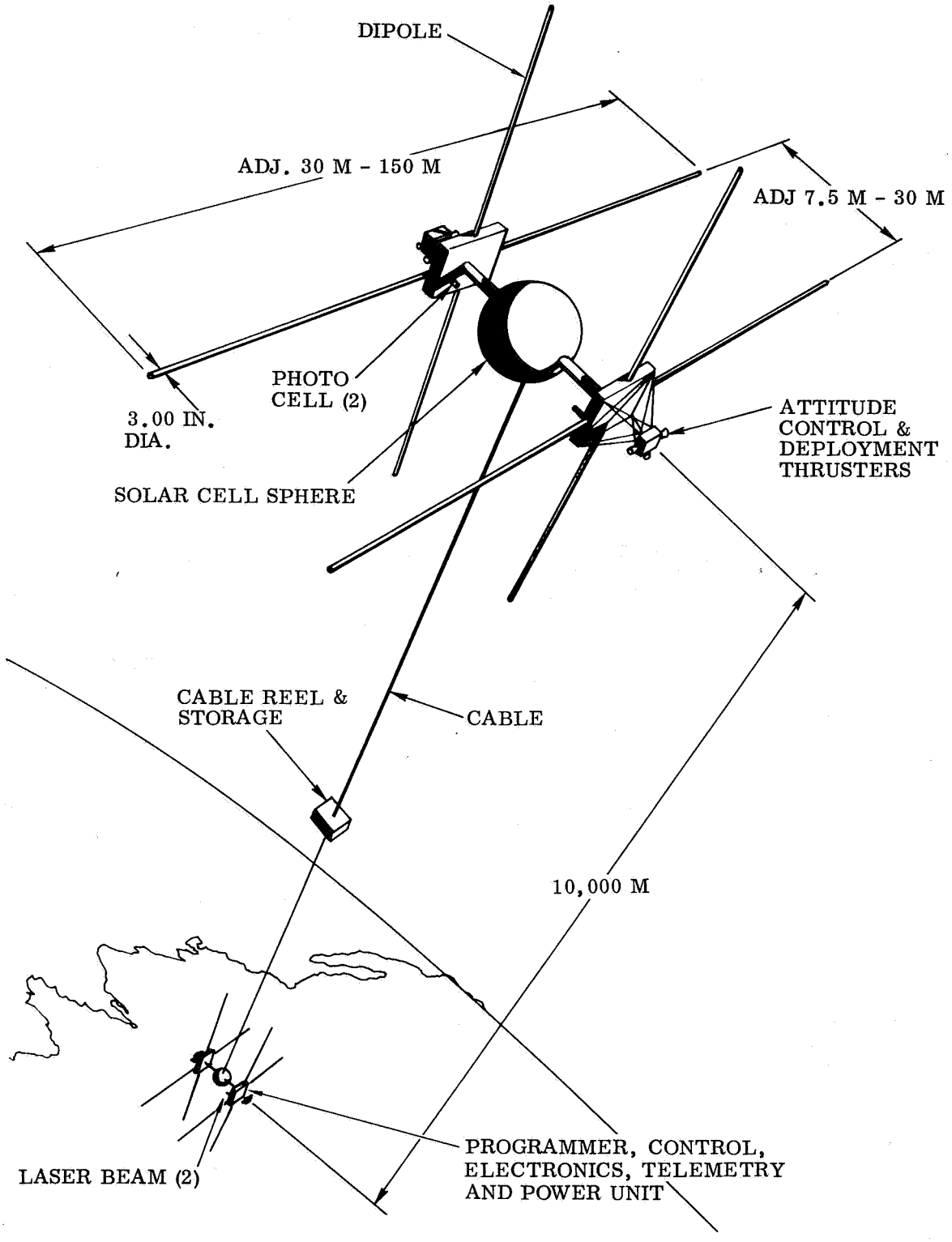



Figure 2-20. Concept 51b, Crossed-H Interferometer Radio Astronomy Antenna

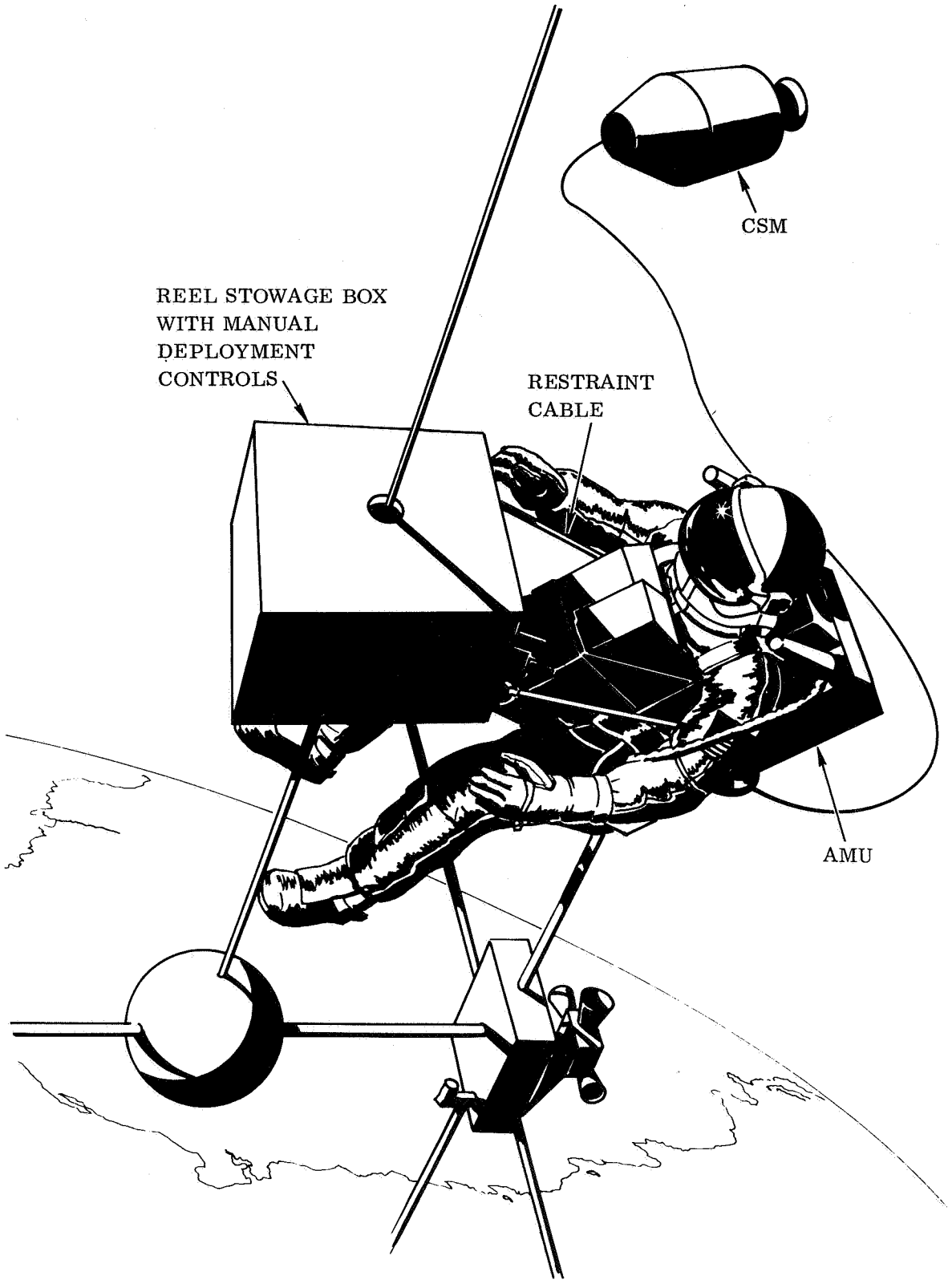



Figure 2-21. Manual Deployment of the Crossed-H Interferometer Radio Astronomy Antenna by EVA

2.3.2.1 Antenna Performance. Principal electrical features of the concept are the interferometer effect, suppresion of reception from one hemisphere, and the possibility of simultaneous reception of both polarization components of signal on separate receivers, and a coverage of the long-wave radio astronomy band from 0.5 MHz to 10 MHz.

Electrical characteristics of the concept are as follows:

a. End-fire radiation pattern envelope normal to the planes of the crossed dipoles.

b. E-plane envelope half-power beamwidth varies from about 90° at 0.5 MHz to about 40° at 2.5 MHz.

c. H-plane envelope half-power beamwidth varies from about 225° at 0.5 MHz to about 180° at 2.5 MHz and above.

d. Interferometer lobes in plane containing long axis of array. Lobe half-power beamwidth is about 1.7° at 0.5 MHz decreasing to 0.10 at 2.5 MHz.

e. Polarization: Receives both polarization components independently.

f. Bandwidth: 0.5 MHz to 10 MHz.

g. Antenna Gain: 6 db at 0.5 MHz, 10 db at 2.5 MHz and above.

h. Antenna Effective Intercept Area: $10^5$ $m^2$ at 0.5 MHz, 450 $m^2$ at 10 MHz

Figure 2-22 illustrates the various dimensions of the crossed-H interferometer array and its electrical characteristics when the array is adjusted for optimum response.

At frequencies below 2.5 MHz, the array is size-limited by structural considerations; in this frequency range, automatic tuning networks would be used to resonate the dipole elements for a given signal frequency. Antenna dimensions would remain constant at their upper limit. The beamwidth of the antenna pattern gradually decreases with an increase in frequency to give a resolution that is about 1.7° at 0.5 MHz and improves to about 0.35° at 2.5 MHz.

At frequences about 2.5 MHz, the array dimensions are reduced to avoid radiation pattern deterioration that would otherwise occur. Above this frequency, the dimensions of the array are adjusted for each signal frequency. Beamwidth characteristics remain essentially constant, with a resolution of about 0.35° above 2.5 MHz.

The E-plane for each dipole is a plane containing the axis of the dipole and a line between the antenna and a source of radio noise. The H-plane for each dipole is a plane perpendicular to the axis of the dipole and containing a line between the antenna and a source of radio noise.

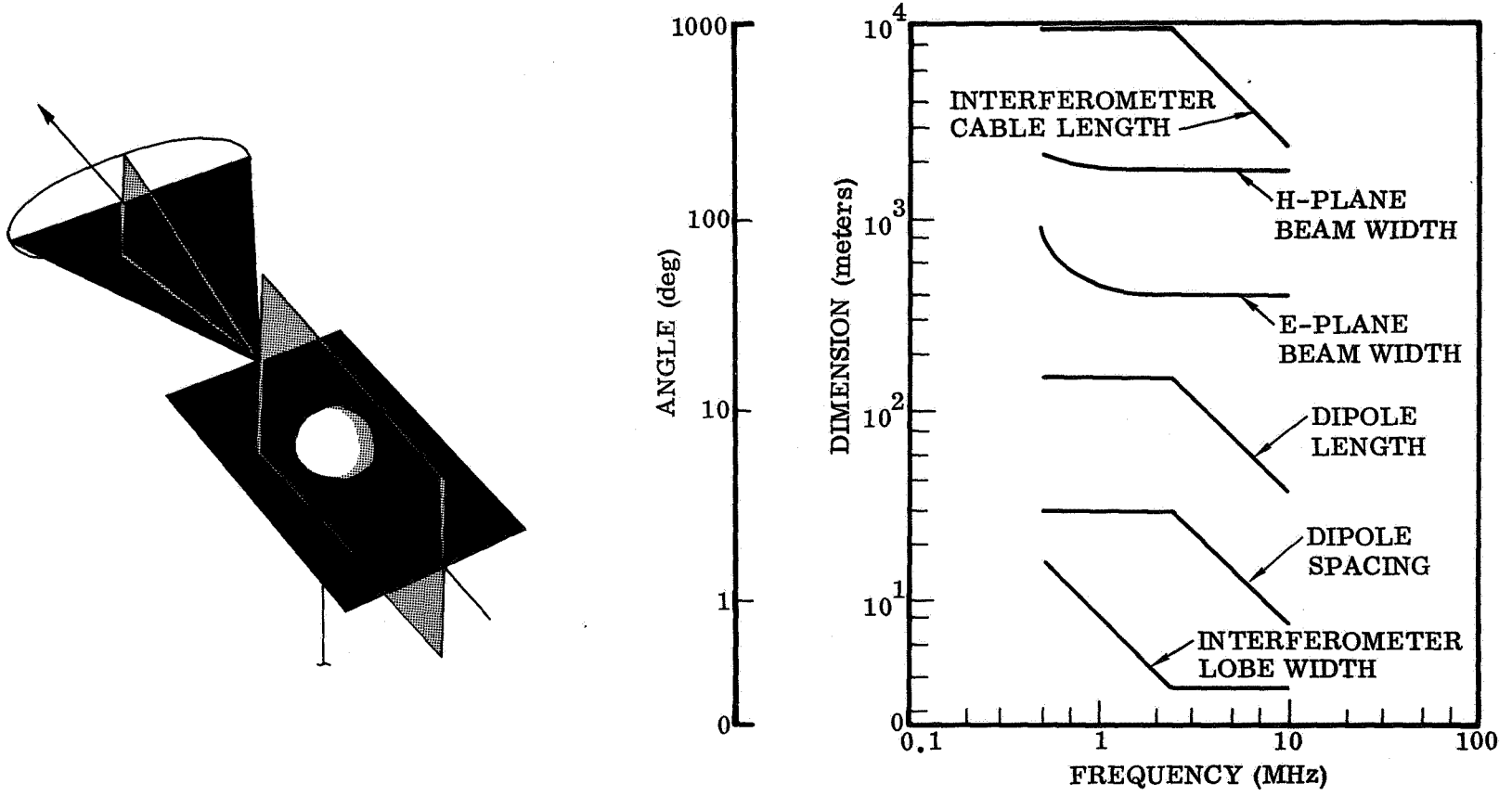



Figure 2-22. Parameters of Crossed-H Interferometer Array

2.3.2.2 Structure and Mechanics. The basic structure is the support boom that carries the crossed dipole and other equipment at each end. It is loaded by thrust and inertia forces at each end and by cable and inertia loads at the center.

Its length is such that thermal distortion is not excessive but must be minimized to reduce variation in dipole attitude. It must also be capable of repeated extension and retraction by remote control or programming.

The crossed dipoles at each end of the support boom are a typical application of the preformed tape technique. The loading, due only to its own inertia and minute solar pressure, does not require joined edges and therefore retraction poses no problem beyond power and control considerations. In this application, however, distortion due to thermal gradient does require additional consideration to maintain the necessary dipole alignment. "Barbecue" rotation of each dipole about its own centerline by the use of solar power is a possible solution.

The dipole head is permanently attached to the support boom head and supports the four dipole reels and thruster unit.

The ACS provides manuevering and stabilization for pointing and dipole alignment. Its controls include laser beams and photo cells to sense the relative position of the dipole pairs. Also included are earth and celestial sensors for gross orientation.

2.3.2.3 Attitude Control and Dynamics. It appears to be feasible to utilize this structure for both celestial sphere scanning and target lock on.

The celestial sphere scanning is done with the aid of the gravity gradient. Active control of each of the two support booms about all three axes is required. The preliminary design uses jets but application of momentum storage devices may be indicated by a more detailed study. An additional attitude control requirement exists in the necessity for keeping the dipole pairs at either end of each 15-meter boom aligned within 5°. The torsional properties of the boom are low, so active control sensing will probably also be required for this orientation.

It should be noted that with this type of attitude control the area subject to solar pressure must be balanced such that the center of solar pressure does not deviate from the center of gravity by more than a limiting distance, perhaps 400 ft at 40,000 n.mi. orbit and 100 ft at 60,000 n.mi. orbit.

Total solar force on this cable connected structure is of the order of 1/100 of the solar force on Concept 51a. There is little possibility that this vehicle will weather vane during the celestial sphere scanning mission and the attitude should be easily held by gravity to a degree or so in pitch and roll. The active three axes plus torsion

control of each of the two structures should meet the requirements, even though the tolerance on motion out of the horizontal plane has been tightened to 0.1°. However, accurate attitude sensors, probably star trackers, will be needed.

The longest element dimension of this concept which is susceptible to thermal distortion is the 150-meter crossed dipole length. Reference to Figure 2-19 shows that this length requires a radius of curvature of less than 170 ft to achieve the deflection limit of 1/10 wavelength. Thus, the material type chosen for the elements of this concept is not critical. However, this requires further detailed analysis. See Volume III.

Using the earth's gravity gradient to position the major axis of the antenna system, the celestial sphere can be covered in a matter of months, depending on the orbit inclination. Stabilization about the gravity gradient position is desired to provide consistent coverage of the celestial sphere. This is accomplished using the attitude control system to damp oscillations. Attitude of the two end-fire arrays is to be controlled to 5° about the cable and to 0.1° in the two attitudes perpendicular to the cable.

In addition to gravity-gradient attitude control this concept has the capability of target search and lock on for definite periods. Length of periods and number of reorientations are determined chiefly by the propellant supply.

2.3.2.4 Functional Analysis. Deployment functions are as follows: 1) Astronaut maneuvers spacecraft to align with gravity gradient; 2) Antenna package is separated from spacecraft; 3) Radio signal and/or separation sensing starts timer and programmer; 4) Programmer starts cable release and, assisted by thrusters, permits packaged boom assemblies to separate; 5) With cable deployed 200 meters as sensed by limit switch, support booms assemblies in center sections are deployed by release of boom reels; 6) Dipole heads are deployed by release by boom reels. Dipole elements stored on reels in dipole heads on extended boom ends are deployed by releasing reels; 7) Attitude control initiated and laser-beam/photo-cell system command thrusters to align dipole heads and aid in stabilization; 8) Magnetometers to verify attitude. Antenna RF operation given preliminary checkout; 9) Adjustments to antenna and equipment made by astronaut in EVA if necessary; 10) End-fire dipole arrays (extended dipoles and support boom assemblies) deployed to the limit of their connecting cable; 11) Final checkout tests made.

2.3.2.5 Crew Systems. Crew requirements are 1) an inspection of the antenna during and after deployment, 2) repair or replacement of any damaged antenna component that the EV crewman has the capability of repairing, 3) unjamming of any fouled equipment, and 4) refurbishment for extended mission.

The inspection task would be broken into two separate tasks due to the 5000-meter spread between the two halves of the antenna. This distance greatly exceeds the EV crewman's range capability.

Therefore each antenna half would be inspected separately with the EV crew returning to their spacecraft before maneuvering to the other antenna half.

The actual inspection of each half might require a minimum of 15 min and could require much more time depending on the extent of antenna repair and replacement. This estimate does not include the necessary pre-EV operations of suit donning, etc., nor spacecraft exit time. The time required for these tasks would total approximately 90 min prior to the inspection task and approximately 60 min after the task.

The astronaut support equipment requirements are limited to that equipment necessary to support the EV crewman during the inspection and repair of the deployed antenna. They consist of the following:

a. An extravehicular life support system (PLSS).

b. The necessary maneuvering devices (HHMU).

c. Portable worksite anchors.

d. Portable illumination systems.

e. Portable power supply systems.

f. A communications system.

g. The necessary repair tools and equipment.

The equipment that is unique for the erection of this particular antenna, such as the necessary repair tools and equipment, depends on the final structural design (e.g., mesh versus perforated sheet for the dipole head extensions) and a definition of its specific nature must await further design detail.

2.3.2.6 Weights. The total weight of the experiment is approximately 3,657 lb. This poses no problem for the required Saturn-V synchronous altitude launch whose available payload capacity is 23,992 lb. Additional weight for more propellant will probably be added to the extent permitted by control dynamics.

2.3.2.7 Electrical Power. The basic power requirement for support of this experiment is not significantly different from Concept 51a, with the exception that additional power will be required to generate the laser beams. The laser generator power is subject to considerable variation depending upon the type of unit employed. Preliminary estimates of laser input power indicate that an allowance of 25 watts will cover any variations due to selection of concepts or equipment. As in Concept 51a, the basic choice lies between solar cells and isotope power. In this instance the problem of disturbing torques is not critical due to the absence of extreme moments in the structure. On this basis, solar cells appear to be a logical approach based upon availability, cost, and design adaptability. A requirement unique to this concept is the need for power on two separate bodies requiring two separate solar cell arrays. It is expected; however,

that the large majority of power can be concentrated on one body with the other body requiring power for only photocells and attitude control system and antenna tuning. Total power penalities for a solar cell system will be 93 lb based upon 0.53 lb/w and 70 $ft^2$ based on 2.5 $w/ft^2$. These values are based on non-oriented paddle mounted cells with energy storage at 10 w-hr/lb.

2.3.3 Concept 52a, Rhombic, External Structure. This antenna has its receiving elements in two rhombic patterns which are in mutually perpendicular planes. This permits shaping the basic structure to conform to the receiving elements with only tension ties of nonconductive material added. See Figure 2-23. Such an approach allows the use of metallic booms without the RF interference of conductive elements extraneous to the fundamental antenna configuration.

A thruster unit is mounted on the tip of the antenna axis to provide tension in the booms as necessary to reduce any tendency of the multiple booms to twist or curl during extension. The extended antenna is expanded by small expanders and take-up of the center non-magnetic tie wire. The expansion is limited by other tie wires serving as tension spokes. Four additional wires, serving as a square rim position the booms circumferentially.

The tie wires are wound on reels controlled remotely. By changing the unreeled length of these ties the apex angle of the rhombus is changed and the wavelength of the antenna is modified.

Although a fairly complete analysis of this concept was performed, it was determined to be similar but inferior to the internal compression structure rhombic Concept 52b. Structural weight and thermal distortion of the legs are greater for the external structured rhombic. Therefore, only an abbreviated summary of Concept 52a results are included.

2.3.3.1 Structure and Mechanics. The four peripheral booms are in themselves the functional antenna as well as the primary structural memebers. To best serve either purpose the legs of all booms should be straight and therefore cannot be subject to distortion from thermal gradient to any significant degree.

The legs are cylinders constructed of a conductive metallic mesh or perforated sheet. The perforations in the wall expose the far side to solar radiation and thereby reduce the circumferential temperature gradient. Inside and outside surface finishes are selected such that each will absorb or reflect solar energy to the degree necessary to maintain a balanced temperature around the tube.

In the extended condition the booms serve as hinged columns that are stabilized by tension ties. One tie is attached to the top adapter and extended along the rhombus centerline to a reel mounted on the bottom adapter. Other ties are attached to the mid-point adapters and extend diameterically and peripherially to drums mounted on the mid-point

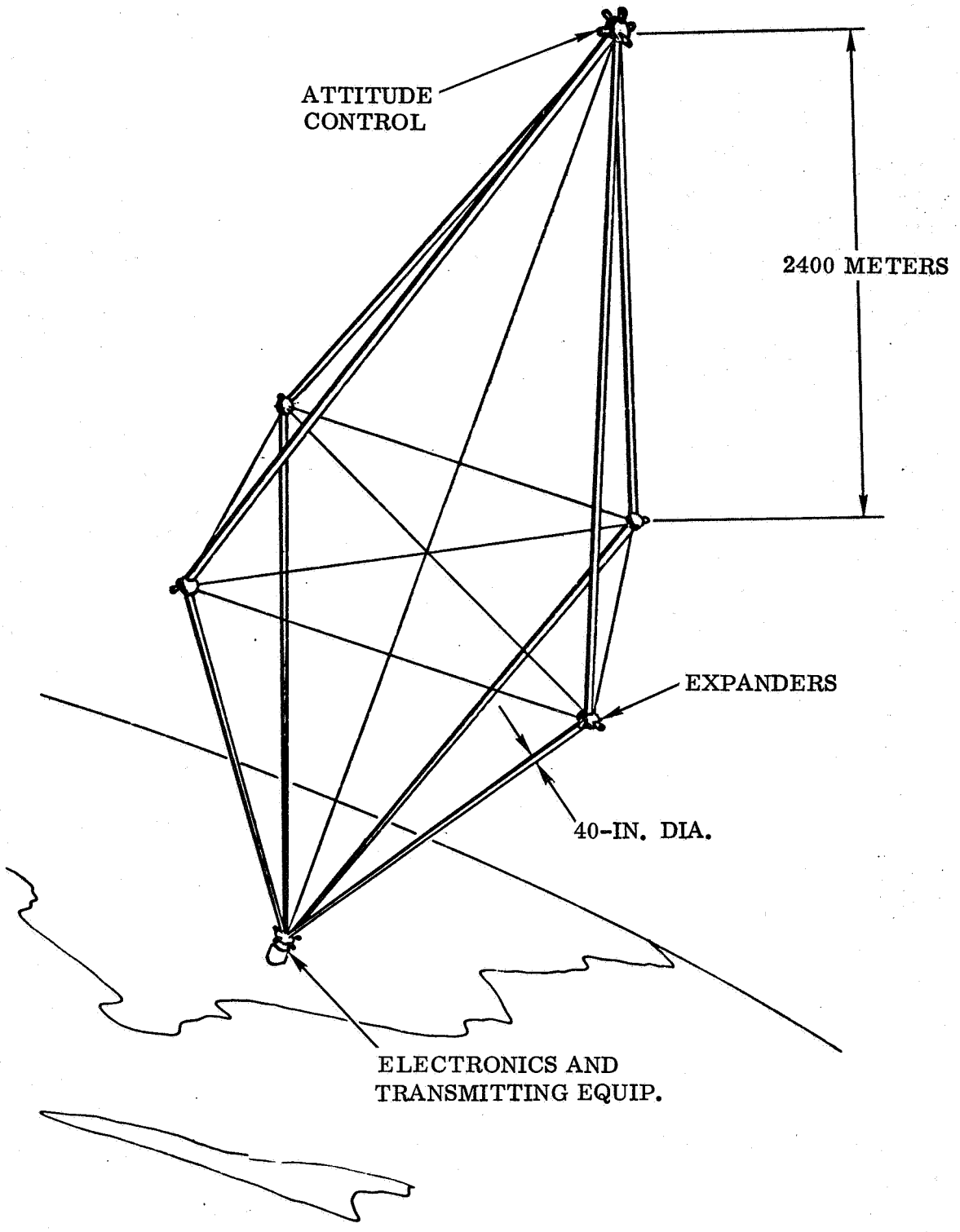



Figure 2-23. Concept 52a, Rhombic Radio Astronomy Antenna

adapters of the other booms. These ties not only stabilize the booms, but provide a means of modifying the rhombus angles. Coordinated rotation of the centerline tie storage reel unwinding and the others winding provides a means of remote control for the antenna apex angles and therefore for RF tuning.

2.3.3.2 Attitude Control and Dynamics. The attitude control is first analyzed with respect to two factors: manuevering rates and structural strength. The analysis was based on a 2400-meter leg length, which has subsequently been reduced to 1800 meters. Manuevering a 2400-meter leg rhombic under the usual assumption of one reorientation per hour for a year requires 12,800 lb propellant for each of two axes. Meanuevering propellant weights of 1000 lb per axis appear reasonable, and they result from reorienting once each 3.6 hr at a vehicle rate of $7.7 \times 10^{-5}$ rad/sec. Two jets of 0.14-lb thrust are needed.

2.3.3.3 Stress Analysis. Assuming each boom is a tube of 40-in. diameter, 2,400 meters in length, and made from 20-grid 7-mil aluminum wire mesh, the allowable column load is only 0.212 lb. Assuming the column is preloaded to 0.1 lb, and thrusters fire symmetrically (0.01 lb each), the net column load is 0.123 lb and f = 250 psi. This is greater than the allowable stress for local instability. Therefore, the applied load must be reduced or the column length shortened. It is estimated that a maximum length of 1,800 meters can be achieved with the given weight. Even this length will require low preload (approximately 0.1 lb).

The above analysis assumes no distortion due to thermal gradient or initial alignment. These distortions must be kept relatively low (approximately 200 in.).

2.3.3.4 Antenna Performance. Referring to Figure 2-3, the advantage of extending the leg length to 2,400 meters from the originally proposed 300 meters becomes obvious. The operable frequency range would be 0.125 MHz with a corresponding beamwidth range from 30° x 70° to 5° x 15°. This can be accomplished by structurally varying the acute angle from 90° to 20°.

The compromise leg length of 1500 meters would give a operable frequency range from 0.2 MHz to 10 MHz with a corresponding beamwidth range from 30° x 70° to 5° x 15°.

2.3.3.5 Thermodynamics. This concept employes mesh elements along each leg. Each element is pinned at the ends and able to bend over the entire 1500-meter length.

If thermal distortion were to be limited to 60 meters, i.e., 1/10 the maximum wavelength, the resulting radius of curvature would be over 20,000 ft. This is an order-of-magnitude greater than available with the present state-of-the-art mesh design.

Advanced mesh materials such as capillary heat pipes may be developed to provide the required large radius of curvature. This concept is not feasible without such a development.

2.3.4 Concept 52b, Rhombic Antenna, Internal Structure. This concept, shown in Figure 2-24, has its receiving elements in two rhombic variable patterns that are in mutually perpendicular planes. These elements are mutually resistance coupled at one apex of the rhombus and lead into an electronic capsule at the other apex where the signal is processed, stored, and subsequently telemetered to earth. The rhombus shape can be varied for a choice of wavelengths.

The structure supporting the antenna wires consists of one major axis longitudinal boom, two minor axis lateral booms and the necessary guy wires to stabilize these booms. The locations of the antenna wire attachments on the booms determines the configuration of the rhombus. They can be varied to select the wave band desired within the limits available.

At the top and at the bottom of the major boom, thrusters are mounted that provide for attitude control during deployment as well as operation.

2.3.4.1 Structure and Mechanics. The longitudinal boom and the two lateral booms of this configuration are the structural truss members that carry compression. The external longitudinal wires that are the functional antenna also serve as truss members in tension together with the internal non-reflective stabilizing wires.

The longitudinal boom is the most heavily loaded member and the longest. For structural integrity it must be straight and therefore cannot be subject to distortion from thermal gradients to any significant degree. It is a cylinder constructed of metallic mesh. The interstices in the wall expose the far side to solar radiation and thereby reduce the circumferential temperature gradient. Both interior and exterior surface finishes are selected such that each will absorb or reflect solar energy to maintain a balanced temperature around the tube.

The diameter of this boom is limited by stowage dimensions and its wall size is limited by weight considerations. However, a satisfactory member appears feasible within these limitations.

The lateral booms are not as long and therefore are not as susceptible to load and distortion problems. However these members must be non-reflective to avoid RF interference, and therefore must be of non-conductive material, or at least isolated electrically into less than 1/10 wavelength sections, about 50 ft.

The acute apex angle of the wires that serve as the functional antenna may be varied to alter the operating waveband of the antenna. This can be accomplished by moving the wire attachment point along the booms.

One such concept would consist of a ring of non-abrasive material, such as Delrin AF, loosely encircling the boom and attaching to the antenna wires at diametrically approved points. The ring would be positioned, and moved, along the boom axis by two

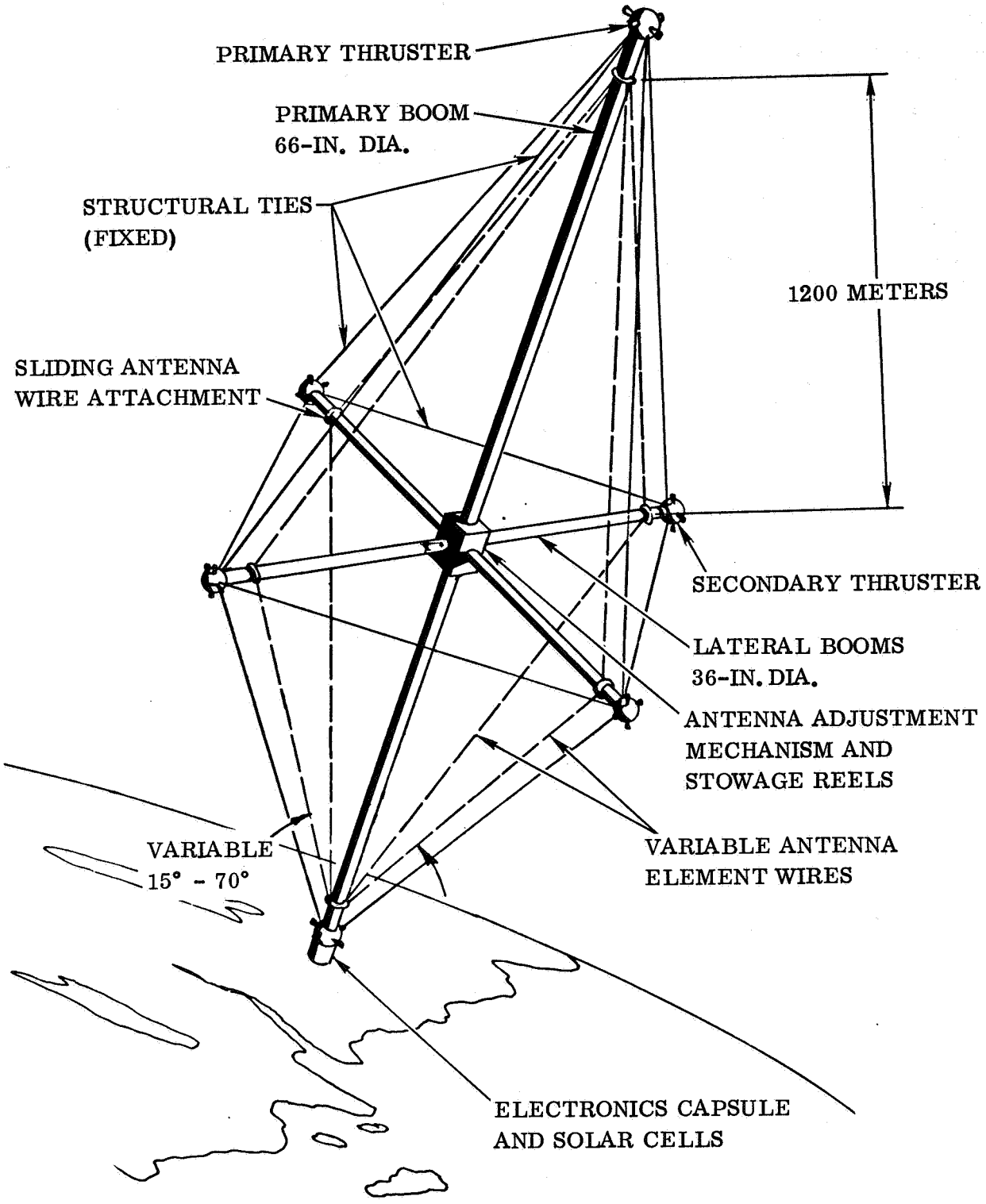



Figure 2-24. Concept 52b, The Rhombic Radio Astronomy Antenna

non-conductive cable circuits. Each cable circuit would be attached to one of the antenna wire attachment points, and would run from a control reel at the antenna center out to a return pulley at the boom tip.

The electronic capsule containing the signal processing and telemetry equipment is mounted on the end of the longitudinal boom and supports its transmitting antenna.

The structure described is stable, assuming the booms to be sufficiently rigid and distortion free. The strength of its primary boom is limited by dimensions and weight compatible with the Saturn-V launch vehicle. Within these limits a satisfactory structure appears feasible, if the length of the sides of the rhombus is limited to 1200 meters.

This concept requires the application to large structures of principles already incorporated in relatively small booms. Preformed tape of metallic mesh is stored on a reel and allowed to resume its tube shape as it is extended. These principles are somewhat further advanced in theory and test under programs currently progressing at Convair.

In long tubes such as required in this concept, it is imperative that the edges of the tape be joined fixedly to provide structural integrity across the longitudinal joints. This joining operation appears feasible if performed as a continuous braze between heat and pressure rolls as the prepared strip edges are lapped and passed through the rolls.

2.3.4.2 Antenna Performance. The operable frequency range is 0.250 MHz to 10 MHz with a corresponding beamwidth range from 30° x 70° to 5° x 15°. This can be accomplished only by structurally varying the acute angle from 90° to 20° corresponding to Figure 2-3.

It is obvious that the performance is critically dependent on the operating frequency (i.e., wavelength). And only at the high frequencies, 10 MHz, can the beamwidth approach the desired 10°.

2.3.4.3 Attitude Control and Dynamics. The orbital attitude is synchronous, the lifetime is one year, and the pointing accuracy is 2.5° with 1° desired. User requirements include scanning the celestial sphere and locking on specific targets.

The three-dimensional structure of this rhombic will support the use jets for active attitude control. The thrust level is structure limited, but maneuvering for target location is feasible. The large antenna dimensions will significantly increase external torque propellant requirements above that required by Concept 2b. Gravity-gradient stabilization could be used for the celestial scan mission to reduce propellant weight, but this approach is not mandatory.

The attitude control was first analyzed with respect to two factors, maneuvering rates and structural strength, and based on a 2400-meter leg length, which subsequently has been reduced to 1200 meters. An assumption of one re-orientation per hour has been previously used as a yardstick. The associated propellant requirements and strength for this large rhombic result in maneuvering once per 18 hr. Jet thrust for this structure is limited to about 0.010 lb, and is located at the ends of the major boom. Lower level thrusters for roll control are located on two of the center apexes.

These attitude control problem areas are tradeoff rather than feasibility considerations. The antenna can be controlled to the desired tolerances and it can be maneuvered at a rate determined by propellant weight and structural strength.

Antenna inertias are estimated at $8.56 \times 10^9$ transverse and $5.60 \times 10^8$ slug-ft$^2$ longitudinal. The maximum gravity torques are 63.5 ft-lb pitch and 21.2 ft-lb yaw. The average per axis is 29.6 ft-lb. Solar torque is sufficiently smaller than gravity torque and it can be ignored in calculating propellant. The gravity torque magnitude precludes the utilization of momentum storage devices. The size of the antenna, 2400-meter legs, and the resulting gravity torques lead to propellant requirements of 428 lb per year for each of two axes. This is not excessive for the payload. Reduction could be achieved by using gravity-gradient stabilization for the celestial scan mission.

Reorientation of the antenna will be slow due to the low thrust levels and high inertia. The quickest way to reorient is to use the engines full on for half the turn and full opposed for the remainder. The nominal turn is 1 rad and acceleration is $1.704 \times 10^{-8}$ rad/sec$^2$. The turn takes 4.25 hr.

Note that momentum storage devices do not have the capacity to reorient structures of this size. A velocity of $1.17 \times 10^{-7}$ rad/sec is imparted by a momentum change 1000 ft-lb-sec. It thus takes 2380 hr to move through 1 rad.

An upper bound to the maneuvering rate is thus once each 4.25 hr. This allots zero time to lock on and requires 2110 lb of propellant for each of two axes. Actual maneuvering capability is estimated to obtain propellant expenditures of 500 lb per axis, using Isp of 300 sec. This expenditure results from one maneuver each 19 hr, or 487 reorientations per 1 year. The antenna is moved in 4.25 hr and has 13.75 hr lock on time. Thus the structural strength limitation of the ACS thrust to 0.01 lb and the self-imposed maneuvering propellant consumption of 500 lb per axis have reduced the maneuvering capability from the usual assumption of one per hour to one each 18 hr.

Operating control system frequencies are needed to discuss coupling, so frequencies are estimated now. The several operational modes considered are maneuvers, settling down from maneuvers, disturbance torque corrections, and limit cycles.

The 1 rad turn maneuver has the jets on full for 2.12 hr, or $1.31 \times 10^{-4}$ Hz. Settling down pulsing is assumed to occur 10 times as fast, or $1.31 \times 10^{-3}$ Hz. The maximum disturbance torque is 63.5 ft-lb. This torque will accelerate the antenna through 2° in 3320 sec, or $3.01 \times 10^{-4}$ Hz. The limit cycle drift velocity is taken as 3% of the maximum attained during thrusting. This maximum is $7.47 \times 10^{-3}$ deg/sec. The antenna drifts through 2° in 8920 sec, so engines fire at $1.17 \times 10^{-4}$ Hz. These numbers are in Table 2-6. The estimates are dependent on mission and vehicle specifics such as thrust level, maneuvering time, turn angle, and attitude tolerance. Consequently, any changes in the preliminary data upon which these frequency estimates are based requires a re-examination of the estimates and their contribution to the coupling considerations discussed below.

Table 2-6. 2400-Meter-Leg Rhombic Control Frequencies

| OPERATION | FREQUENCY (Hz) |
|---|---|
| Maneuver | $1.31 \times 10^{-4}$ |
| Settling down | $1.31 \times 10^{-3}$ |
| Disturbance torque | $3.01 \times 10^{-4}$ |
| Limit cycles | $1.12 \times 10^{-4}$ |

A potential problem area that was examined and found not to be a problem for this specific concept configuration is oscillation of the tension wires. Coupling between the wire vibratory modes and the control system is noteworthy for the extremely low damping of wires. This could cause trouble.

The wire is 0.030-in.-dia aluminum and the tension is 0.4 lb. First mode vibrates at $7.89 \times 10^{-3}$ Hz. This value is sufficiently higher than the control frequency estimates so that coupling will not be a problem. The difference is large enough so that some reasonable changes in mission or vehicle specifics will not be troublesome. For example, if the turn angle is 10° rather than 1 rad, the control frequency increases with the square root of 5.7, and the settling down is $3.14 \times 10^{-3}$ Hz, which is still below that of the wire. Finally, it is observed that the presence of coupling will not necessarily result in unacceptable motions because the antenna is a large energy sink and the thrusters are a relatively small energy source.

2.3.4.4 Functional and Time Line Analysis. A significant characteristic of this concept is the erection in space of deployment equipment. The size of this equipment, consisting of boom reels and edging rolls, is such that it must be stowed in a dismantled state. Once it is reassembled by EVA the deployment of the antenna itself requires little more than supervision from the astronaut. Gross crew and equipment functions are shown in Figure 2-25 and a time line analysis in Table 2-7.

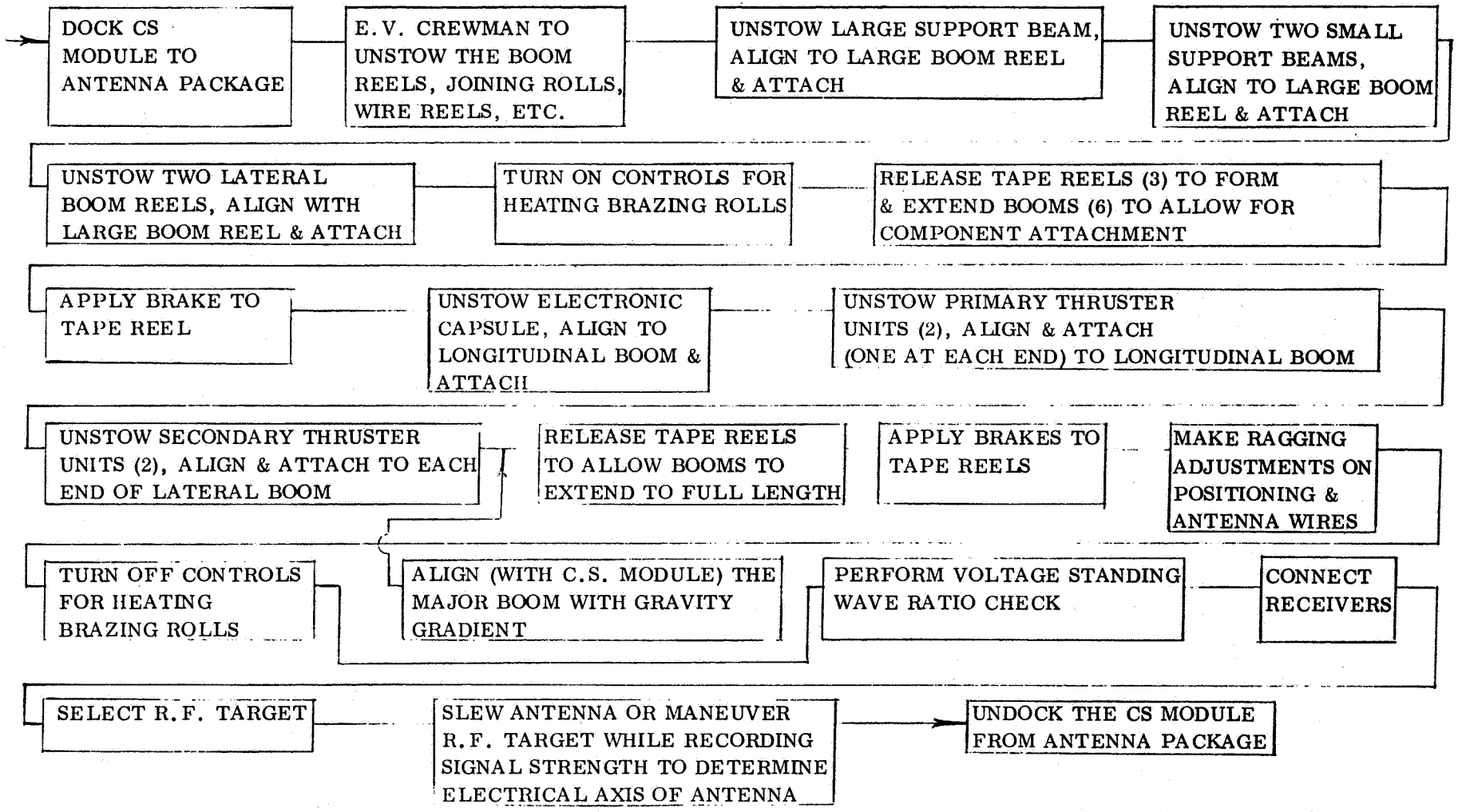



Figure 2-25. Functional Flow Diagram, Concept 52b

Table 2-7. Time Line Analysis for The Erection of The Rhombic Antenna, Concept 52b

| EVENT | ASTRONAUT 1 | TIME (min) | CUMULATIVE TIME (min) | CUMULATIVE EVA TIME (min) |
|---|---|---|---|---|
| 1 | Perform pre-EVA | 30 | 30 | |
| 2 | Cabin depressurization | 25 | 55 | |
| 3 | Hatch opening and exit | 30 | 85 | |
| 4 | Anchor to worksite | 1 | 86 | 1 |
| 5 | Unstow the large support beam for large joining roll | 5 | 91 | 6 |
| 6 | Align large support beam to large boom reel (which was assembled before launch) and attach | 15 | 106 | 21 |
| 7 | Unstow the two small support beams (for joining rolls), align to large boom reel and attach | 15 | 111 | 36 |
| 8 | Unstow the two lateral boom reels, align with large boom reel and attach | 15 | 126 | 51 |
| 9 | Turn on controls for heating brazing rolls | 1 | 127 | 52 |
| 10 | Release tape reels (3) to form and extend booms (6) to allow for component attachment | 5 | 132 | 57 |
| 11 | Apply brake to tape reel | 1 | 133 | 58 |
| 12 | Unstow electronic capsule, align to longitudinal boom and attach | 15 | 148 | 73 |
| 13 | Unstow the primary thruster units (2), align and attach (one at each end) to longitudinal boom | 15 | 163 | 88 |
| 14 | Unstow the secondary thruster units (2), align and attach to each end of lateral boom | 15 | 178 | 103 |
| 15 | Align (with C.S. module) the major boom with gravity gradient | 10 | 188 | 113 |
| 16 | Release tape reels to allow booms to extend to full length | 5 | 193 | 118 |
| 17 | Apply brakes to tape reel | 1 | 194 | 119 |
| 18 | Release reels for wire and attach respective wire to boom ends | 10 | 204 | 129 |

Table 2-7. Time Line Analysis for the Erection of The Rhombic Antenna, Concept 52b, Contd

| EVENT | ASTRONAUT 1 | TIME (min) | CUMULATIVE TIME (min) | CUMULATIVE EVA TIME (min) |
|---|---|---|---|---|
| 19 | Apply brake to wire reels | 1 | 205 | 130 |
| 20 | Make rigging adjustments on positioning and antenna wires | 10 | 215 | 140 |
| 21 | Return to CS module | 5 | 220 | 145 |
| 22 | Enter spcecraft, close hatch, repressurize and perform post-EVA | 60 | 280 | |
| 23 | Turn off controls for heating brazing rolls | 1 | 281 | |
| 24 | Perform voltage standing wave ration check | 60 | 340 | |
| 25 | Connect receivers | 20 | 360 | |
| 26 | Select RF target | 10 | 370 | |
| 27 | Slew antenna or maneuver RF target while recording signal strength to determine electrical axis of antenna | 450 | 820 | |
| 28 | Undock the CS module from the antenna package | 5 | 825 | |

2.3.5 Concept 53, End-Fire Array. This antenna, Figure 2-26, consists of a primary structural boom supporting along its length pairs of crossed dipoles at right angles to each other and perpendicular to the primary boom. The crossed dipoles are extended from their storage by the astronaut and attached to the primary boom and lead-in wires as the primary boom is extended.

The antenna leads are suitable insulated wires electrically bonded by the astronaut to the mid-point of each dipole. Between bond points, the wires are secured to the primary boom, also as it is extended by insulated stand-offs at maximum spacing. A pair of thruster units are mounted on the tip of the primary boom for operational reasons.

Angular alignment of the dipoles along the primary boom is controlled by non-conductive tension ties between adjacent tips — each containing a light tension spring for restoring force and a viscous damper for energy dissipation. Gross torsional

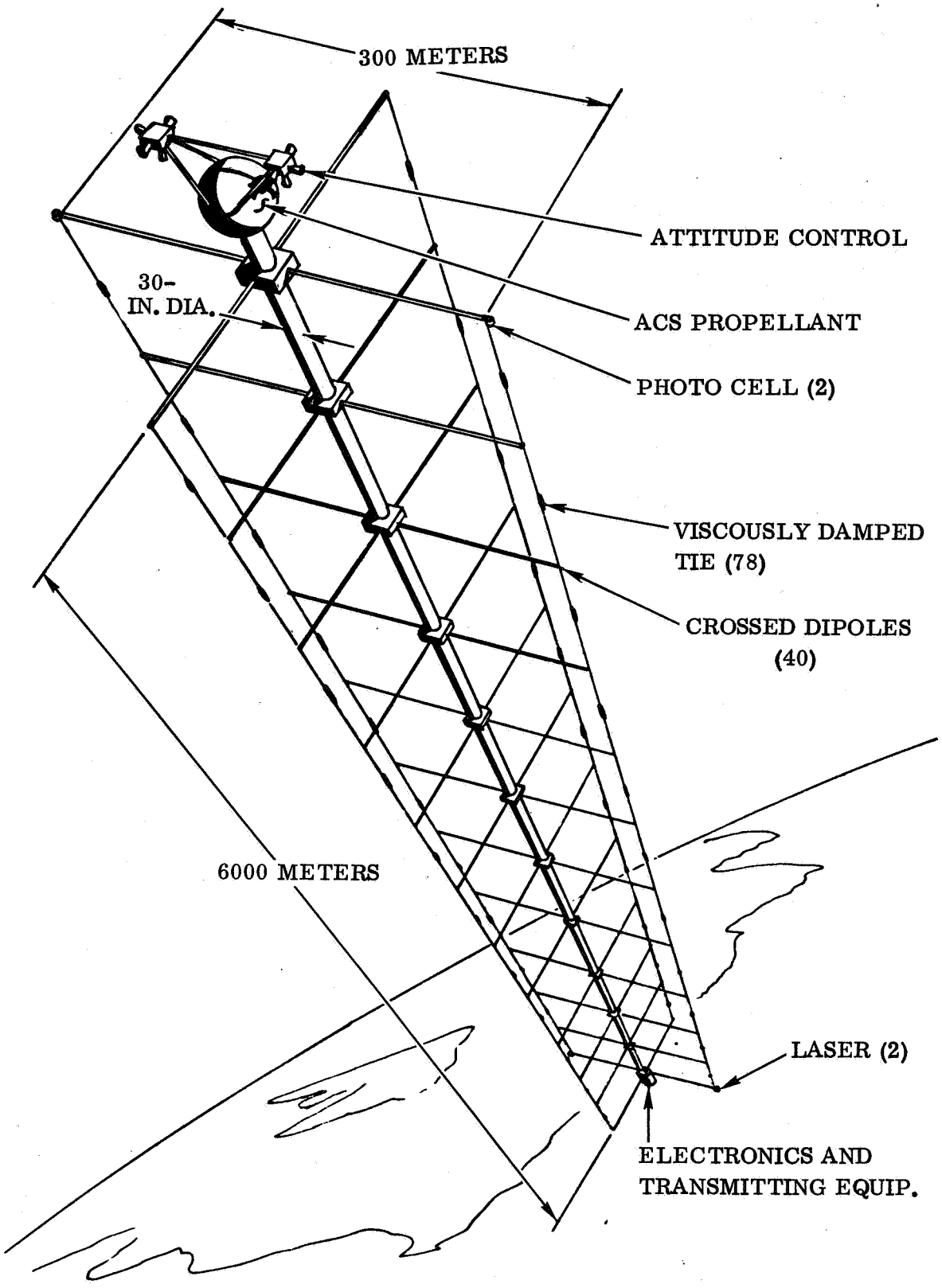



Figure 2-26. Concept 53 Radio Astronomy End Fire Array

distortions from end to end of the primary boom is controlled by lasers located on opposite ends of a bottom dipole and sensed by receivers correspondingly located on a top dipole. The correcting moment is provided by the twin thruster units mounted on the top end of the primary booms.

Although the bandwidth at 1.0 MHz is relatively good, the antenna does not meet the user requirements. In addition, the concept is not recommended because of excessive EVA deployment time, Table 2-8.

Table 2-8. Time Line Analysis for Erection of End Fire Array, Concept 53

| EVENT | ASTRONAUT 1 | CUMULATIVE EVA TIME (min) |
|---|---|---|
| 1 | Unstowing of tape reels and deployment of the major (longitudinal) boom | 210 |
| 2 | Unstowing and installation of U-joint assemblies between each boom section using stand-off clamps | 800 |
| 3 | Unstowing and attachment of lead-in wires to the stand-off clamps | 400 |
| 4 | Deployment of the crossed dipoles (40) | 400 |
| 5 | Unstowing and attachment of the viscously damped ties (78) to the ends of crossed dipoles | 2,400 |
| 6 | Unstowing and attachment of lasers (2) to the ends of the crossed dipoles | 30 |
| 7 | Unstowing and installation of the electronics and transmitting equipment on end of major boom | 30 |
| | | 4,270 |
| | (Approximately 17 days) | |

No method of collapsing the entire assembly seems feasible. The best compromise seems to require the astronaut to make 40 attachments of undeployed dipoles at primary boom joints, and attach the viscously damped ties to deployed dipole ends at 80 places making 20 trips around the 300-meter dipoles in the process.

Dynamic stability could also be a serious problem, even with the damped ties. Consequently, the concept as presently defined is judged not feasible.

2.4 THERMAL ANALYSIS WITH PRIMARY EMPHASIS ON TUBULAR FRAME STRUCTURES. In order to perform a thermal analysis of all the concepts presented in the Task 1 schedule, the large space structures have been divided into three division:

a. Tubular structural members.

b. Flat plate elements.

c. Solar cell flat plate elements.

For the tubular members, equilibrium temperatures are calculated for a solar absorptance/infrared emittance ratio, $\alpha_s/\epsilon$, range of values between 0.2 and 5.0. In addition, at each $\alpha_s/\epsilon$ value, the equilibrium temperature is determined for various angles between the cylinder and the impinging solar radiation.

A special case was considered for the tube to determine the steady state temperatures and the temperature difference between modes when the conductivity of the material is negligible, e.g., fiber-glass cloth and epoxy. A mathematical model was built which accounted only for the radiation exchange of energy. When the mathematical model was changed to include the conduction terms, no change was noted in any of the steady-state temperatures. The low value of conductivity used did not influence the temperature predictions (k = 0.075 Btu/hr-ft-°R).

Three symmetrical booms were analyzed. The internal tube radii were 5, 10, and 60 in. with lengths of 100, 400, and 1200 in. respectively. Each tube was divided into eight nodes; each node was assumed isothermal.

In the first case the sun is being absorbed by only three nodes. Another case considers the sun shining on four nodes; however, with this case no node is normal to the solar radiation.

Table 2-9 gives the temperatures predicted for both Case I and Case II. The assumed values for $\alpha_s$ and $\epsilon$ are:

$$\alpha_s = 0.4$$

$$\epsilon_{ext} = 0.8$$

$$\epsilon_{int} = 0.97$$

It can be concluded that the temperatures for corresponding nodes on each tube are the same. Thus, considering only radiation as a means of heat transfer the size of the boom has no effect upon the temperature.

Table 2-9. Predicted Steady-State Temperatures for an Opaque Boom Considering Only Radiation Heat Transfer ($\alpha_s = 0.4$, $\epsilon_{ext} = 0.8$, $\epsilon_{int} = 0.97$)

| NODE NO. | TEMPERATURE (°F) CASE I | CASE II |
|---|---|---|
| 1 | -76 | 0.4 |
| 2 | 37 | 63 |
| 3 | 67 | 63 |
| 4 | 37 | 0.4 |
| 5 | -76 | -62 |
| 6 | -65 | -55 |
| 7 | -62 | -55 |
| 8 | -65 | -62 |
| 11 | -76 | 0.4 |
| 12 | 37 | 63 |
| 13 | 67 | 63 |
| 14 | 37 | 0.4 |
| 15 | -76 | -62 |
| 16 | -65 | -55 |
| 17 | -62 | -55 |
| 18 | -65 | -62 |
| 21 | -76 | 0.4 |
| 22 | 37 | 63 |
| 23 | 67 | 63 |
| 24 | 37 | 0.4 |
| 25 | -76 | -62 |
| 26 | -65 | -55 |
| 27 | -62 | -55 |
| 28 | -65 | -62 |

$T_{avg} = -14°F$

The maximum temperature difference between nodes 2 and 7, for example is:

$\Delta T_{max} = 37 - (-62) = 129°F$ Case I

$\Delta T_{max} = 63 - (-55) = 118°F$ Case II

Figure 2-27 shows the temperatures as a function of the angle around the tube for Case I and Case II.

Using a value of 0.075 Btu/hr-ft-°R for the thermal conductivity of the fiber-glass cloth and epoxy, there is no effect on the temperature gradients.

Figure 2-28 gives the approximate average temperatures that could be expected for a cylinder under various orbit conditions. All conditions consider solar radiation as the sole source of energy. The temperatures are computed for $\alpha_s/\epsilon$ values between 0.1 and 10.

Equilibrium temperatures are plotted for selected values of solar absorptance to infrared emittance ratios in Figure 2-29.

The purpose of this table and graph is to illustrate the effect of using passive thermal control coatings on the temperatures of large space structure experiments. For lack of more definitive information, equilibrium temperatures were calculated. These temperatures assume the structural member is in a fixed orientation to the sun for an indefinite period of time.

2.5 ELECTRICAL POWER PERFORMANCE ASSESSMENT. While user requirements for experiment support subsystems are not fully defined at this stage in the study, it is possible to make power estimates based on anticipated experiment requirements. Section 6 shows a first estimate against the background of available power system concepts and their areas of application. It may be seen by inspection that requirements for most concepts fall within the application envelope of either solar-photo-voltaic or isotope-generating concepts during operational periods. Experiment activation and checkout functions may be satisfied by batteries or fuel cells depending upon total energy requirements. The selection of a power supply may be somewhat biased by availability, cost, or the nature of associated equipment. For example, for activation and checkout periods up to one week Figure 2-30 shows applicability of three possible power supplies (batteries, chemical dynamic, and fuel cells). This list would logically be reduced to fuel cells and batteries in that these are more likely to be readily available from the CSM, S-IVB stage, LEM descent stage, or other associated modules. The availability of solar cells for battery recharging would further bias selection toward batteries. The use of CSM in the activation function would strongly suggest use of CSM fuel cell power for this portion of the mission. For experiment operational power levels of 30 to 150 watts, and periods of 1 to 2 years the selection of solar cells is indicated.

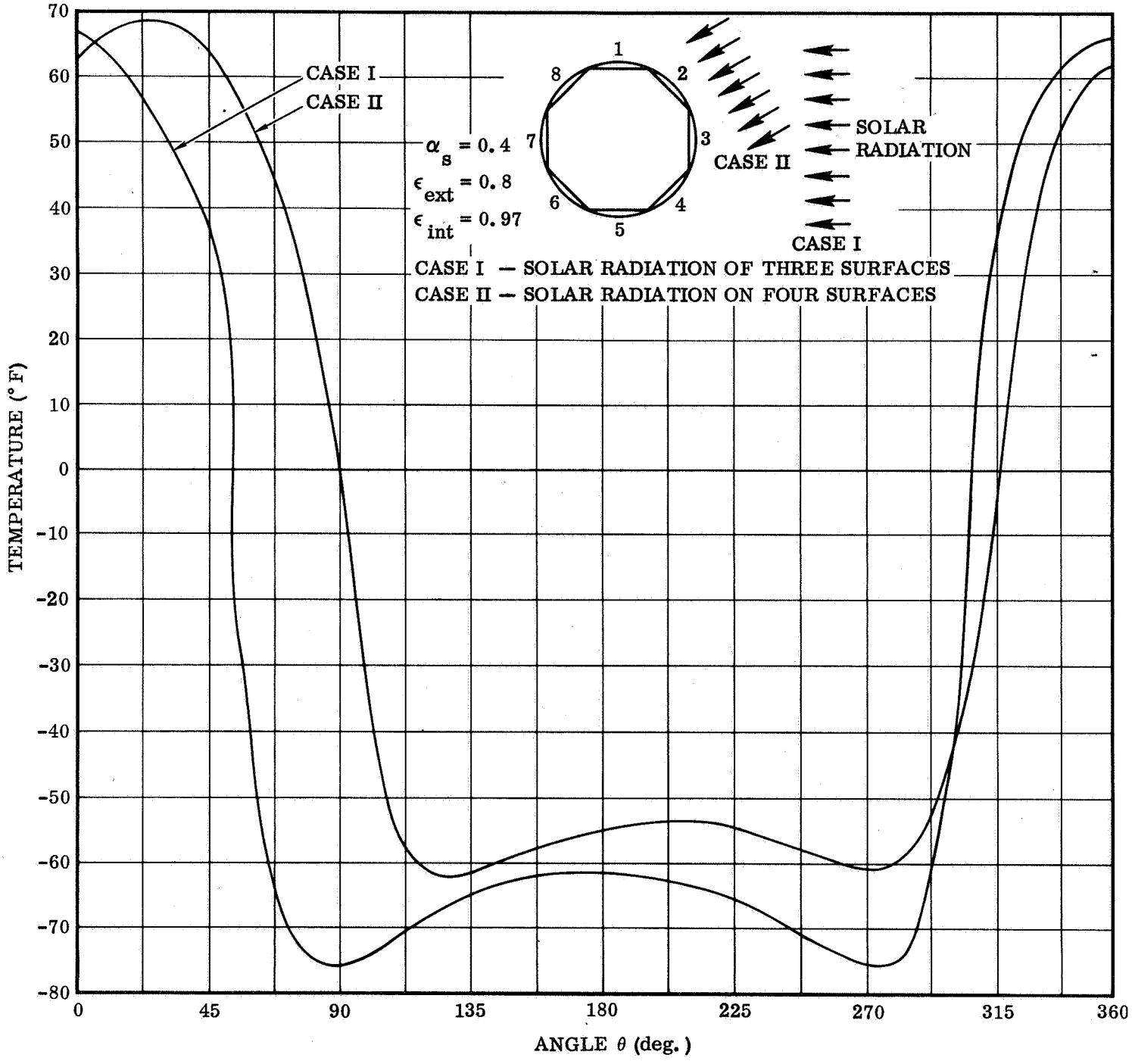



Figure 2-27. Plot of Temperature vs Angle θ for Tubular Structures

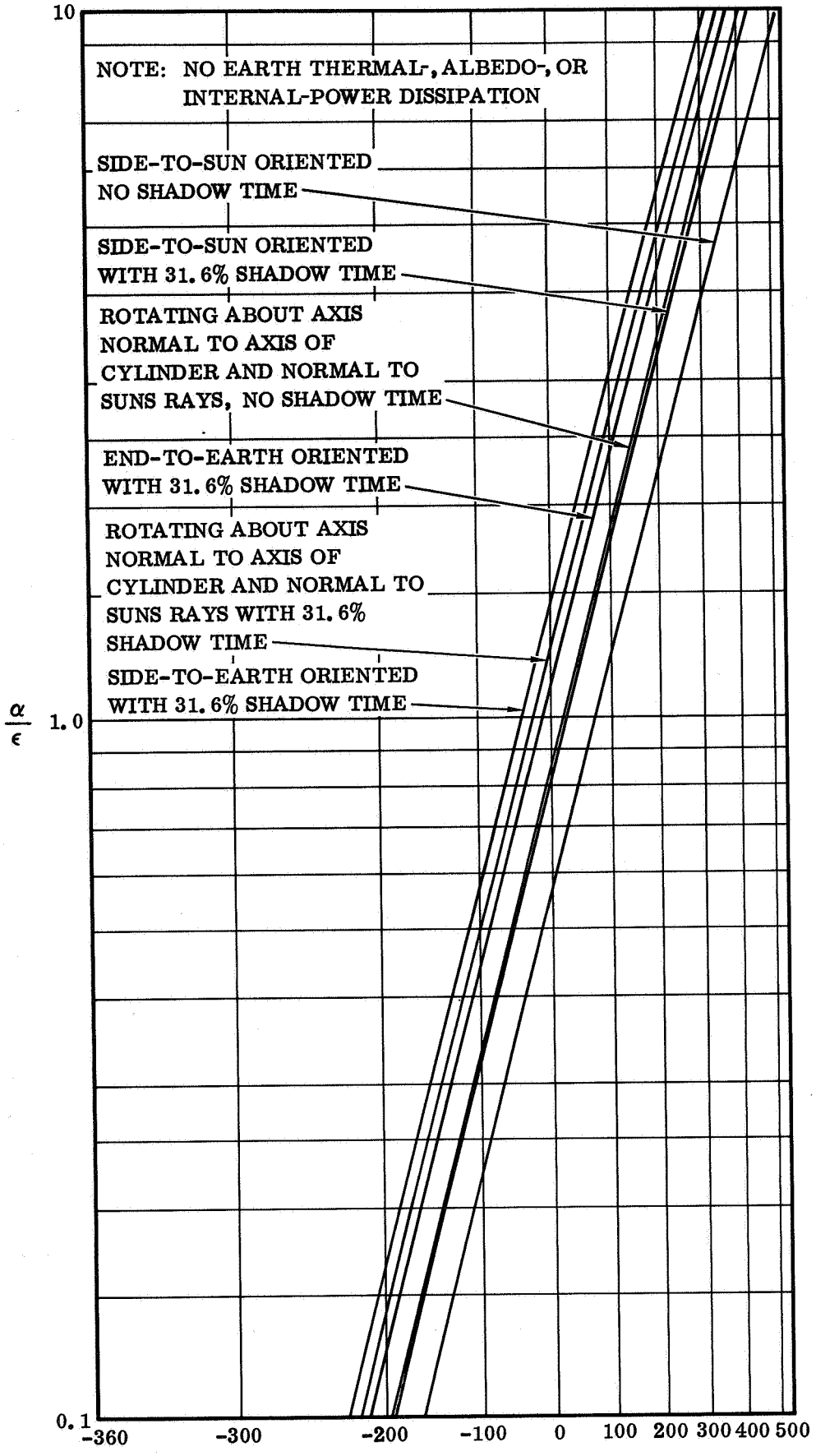



Figure 2-28. Average Temperatures for a Cylinder Under Various Orbit Conditions

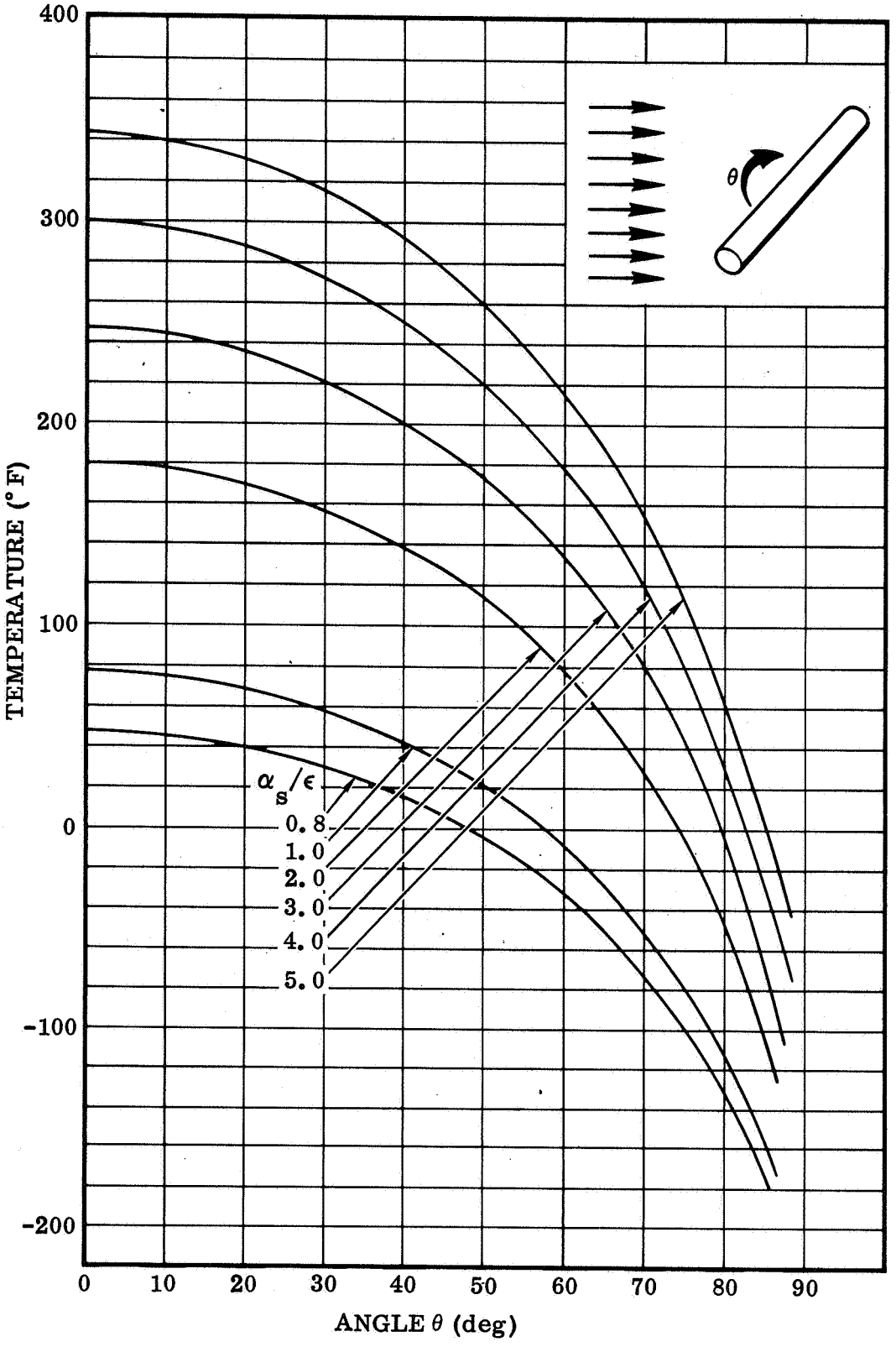



Figure 2-29. Equilibrium Temperatures for a Cylinder at Various Angles with Respect to the Sun

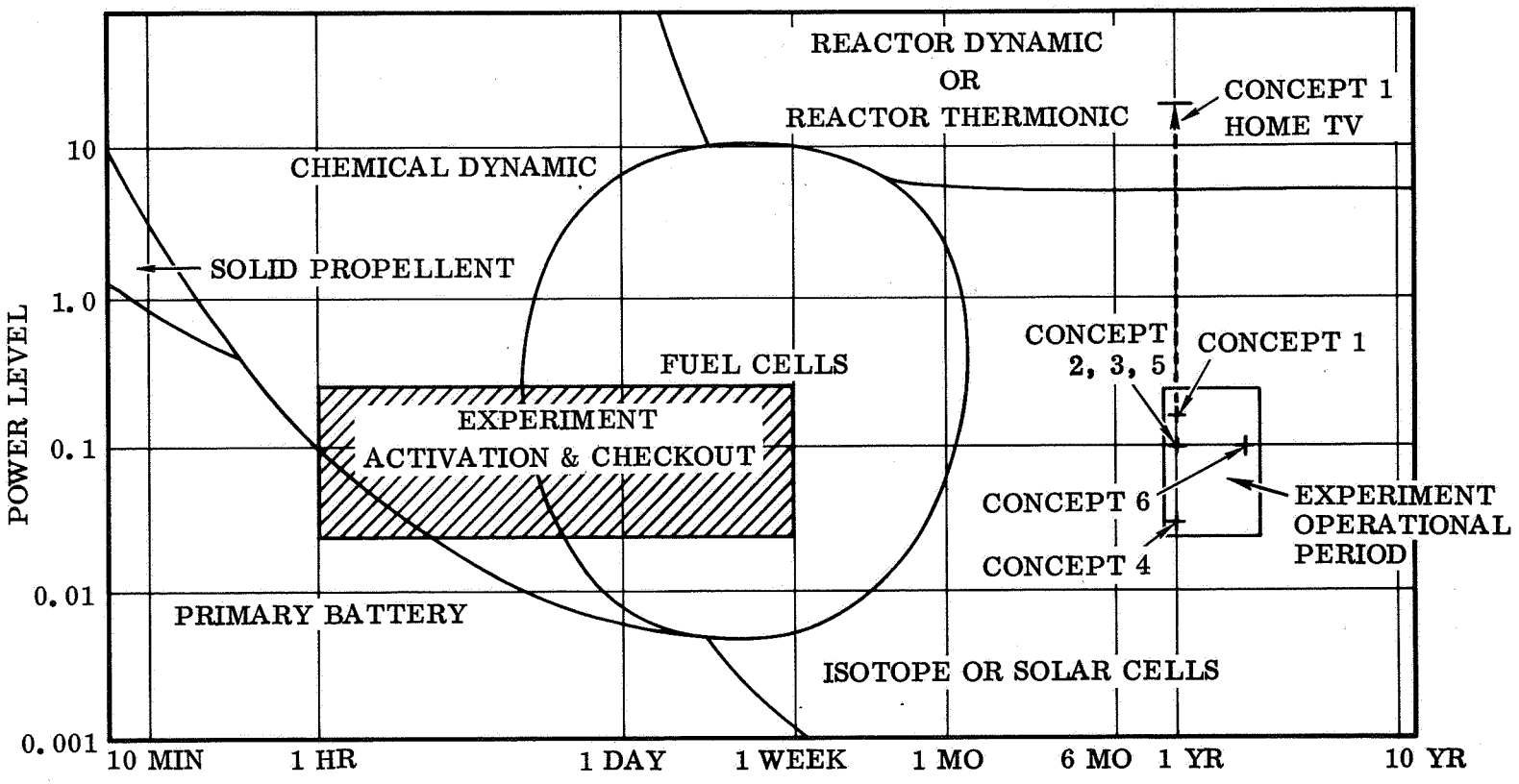



Figure 2-30. Estimated 1972-Plus Power System Applications

2.6 RADIO ASTRONOMY SUBSYSTEMS. One of the primary requirements for orbital radio astronomy antennas is to make spectrum measurements in the frequency range below 10 MHz. These measurements cannot be made from the earth's surface due to the attenuation of these frequencies by the ionosphere.

The desired data are the source direction of radio energy, the frequency spectrum, and intensity of the received signal. Two of these data objectives are compatable (direction and intensity), the third is not (spectrum). The reason for this incompatability is the antenna design required for good direction resolution has very narrow bandwidth or spectrum response. One possible solution would be to make direction and intensity measurements at one frequency until the entire sky is mapped. Then physically change the size and frequency characteristics of the antenna and repeat the mapping at the new frequency. This process could be repeated until the required spectrum resolution is obtained.

Unless active attitude control systems are used, sweeping the entire galatic sphere requires the antenna to have a spin axis inclined to the orbit plane, relying on gravity effects to change the antenna pointing slightly each orbit until the sphere is covered. The spin rate and the beam width of the antenna will determine the rate measurements to be taken:

$$360° \frac{\text{spin rate}}{\text{beamwidth}} = \text{measurements/time}$$

For a resolution of 1 part in 100, 7-bit coding is required. The bit rate for the intensity information is through the measurement rate multiplied by seven.

The attitude reference information will probably be supplied from an inertial platform with one output for each of three gyros. This information will be corrected from phase to analog by a demodulator and then coverted to seven-bit digital by an analog to digital converter.

There are four signals that comprise the required information and two methods that can be used to combine these signals for recording and transmission to earth. One would be to record the four signals on separate tracks, and play these tracks back to modulate four subcarrier oscillators. These four oscillators would then modulate the transmitter. The second method would be to multiplex the four together and record on one track, then use that track to modulate the transmitter. The first of these systems is listed in Table 2-10. The size and power requirements of the two systems are nearly equal.

The tape recorder would operate at 1-7/8 ips (inches per second). With a bit packing density of 550 bpi (bits per inch), this speed will accommodate 1030 bps (bits per second). The playback will be at 22 times record speed or 22,660 bps. The record capacity is 4-1/2 hours with 11-1/2 minutes playback.

Table 2-10. Size Requirements for Attitude Reference Data Transmission Equipment

| QUANTITY | | POUNDS | WATTS | $INCH^3$ |
|---|---|---|---|---|
| (1) | Wide Band Amplifier | 2 | 1 | 32 |
| (6) | Filter | 6 | 0 | 12 |
| (4) | Analog-to-Digital Converter<br>Sample and Hold<br>Shift Register | 11 | 3 | 280 |
| (1) | Clock | 1.5 | .5 | 32 |
| (1) | Command Receiver | 25 | 1 | 1000 |
| (1) | Attitude Sensor Unit | 28 | 16 | 2400 |
| (2) | Tape Recorder | 29 | 28 | 860 |
| (4) | Subcarrier Oscillator | 3 | 2 | 8 |
| (1) | FM Modulator<br>FM Transmitter | 7.5 | 20 | 80 |
| (1) | Diplexer | 3.5 | 0 | 12 |
| (1) | Tape Recorder Management | 10 | 2 | 64 |
| | | 210.5 | 73.5 | 4780 |

## SECTION 3

## MILLIMETER WAVE RADIO ASTRONOMY

3.1 MILLIMETER WAVE RADIO ASTRONOMY USER REQUIREMENTS. The extension of astronomy into the wavelength region between 1 cm and a few microns ( 1 $\mu$ = $10^{-4}$ cm) has in the past been limited by lack of detectors of high sensitivity and by the obscuration of the earth's atmosphere. In the last few years, the devlopment of better detectors has allowed some work to be done in the 1-mm and 11-$\mu$ regions (using a supercooled germanium bolometer) and at 8 mm and 3.5 mm with improved radio techniques. Although the atmosphere of the earth is an absorber throughout the whole wavelength range, there are regions of the spectrum where the absorption is low enough to permit observations from suitably chosen sites on the ground. So far, these observations have been mainly at 8 mm, 3.5 mm, 1 mm, and 11 $\mu$.

No millimeter-wave telescope larger than 16 ft in diameter has yet been proved on earth. However, an uncompensated 36-ft structure is being built. This is probably about the limit of size at which uncompensated structures can be built on earth and still maintain reasonable shape as their orientation with respect to gravity is altered. The thermal gradient problems will also prove to be close to the limit of manageability, even if such a telescope is protected by an astrodome.

A space radio telescope for these wavelengths would still be a parabolic dish. It might be made as a complete surface, or, as is done with the elements of the Hanbury-Brown and Twiss stellar interferometer, as a set of independent reflecting surfaces with a common focus. To give a suitably large step in gain over ground-based instruments, a diameter of as much as 100 ft should be considered.

Problems of gravity deflections in space do not exist. There are problems of making such a dish on earth and erecting it in space, since positional accuracies of the surface of 50 $\mu$ or better (one or two-thousandths of an inch) are needed. Such a dish would be a diffraction-limited antenna at 1 mm and a valuable energy collector in the micron range.

Thermal deflections will present a much more serious problem. Even the best thermal stabilizing paints now known will probably not solve this problem. It may be possible, however, to find techniques whereby thermal equalization may be acceptable.

If such an antenna were built, it could be placed in a low orbit below the ionosphere and above the main atmosphere. Pointing precision of about 1 sec of arc would be needed. The problems of maintaining radiometer performance suggest that cryostats capable of holding the temperature to about 0.25°K would be needed.

The problems of servicing and maintaining equipment in addtion to the task of erection, suggest that such a system would require regular visits by men. The antenna would be used on a variety of programs over a period of several years.

Some comments are in order on **radiometer techniques in the millimeter-wave region.** Thermal detection radiometers, utilizing the germanium bolometer or one of several other possible low-temperature devices, provide high sensitivity and broad bandwidth at millimeter and submillimeter wavelengths. Further improvements in existing designs for ground-based observations and adaptations to the requirements of space telescopes should be undertaken. In addition to the broadband radiometry, it will be necessary to have spectrometers that can provide a wide range in resolution and spectral coverage. Furthermore, the need to obtain the most information possible from the spectrum at $\lambda$ = 0.02 mm to 2.0 mm will require the development of spaceborne cryostats that maintain temperatures as low as 0.25°K.

The following is a summary of the requirements placed upon millimeter wave length and for infrared radio astronomy equipment:

a. Lifetime: 1 year minimum.

b. Orbit altitude: low or high ok, with synchronous desirable.

c. Beamwidth: 1 sec of arc desired, but can be larger with aspect determinations.

d. Pointing accuracy: 1 sec of arc.

e. Aspect determination interval: 1/10 of the beamwidth.

f. Jitter: ± 0.5 beamwidth, when not using aspect determination capability.

g. Sensor or detector: electronic receiver and/or super-cooled bolometer.

h. Bandwidth: desired approximately 50$\mu$ - 10 mm.

i. Spectral resolution: 1°K desired.

j. Collecting area: parabolic dish, as large as possible.

k. Lock-on time: several hours desired, but slow drift allowed for observation of stronger sources, using aspect determinations when beamwidth considerably larger angular diameter than the source.

l. Surface tolerance: approximately 0.1 mm = 0.0025 mm for 50$\mu$ wavelength, and 2 mils for 1 mm wavelength diffraction limited capability.

m. Diffraction limited: down to 1 mm, at least. Preferred to 50$\mu$ = 0.05 mm. Diffraction limited to a certain wave length entails keeping the surface tolerance of those parts of the antenna functioning at that wavelength to within 1/20 of the wavelengths.

n. Physical dimensions: parabolic dish of up to 100 ft diameter.

The suggestion was made by Dr. Ken Hallam of GSFC to design a 50- to 100-ft-dia. dish, utilizing only the inner 10-ft-dia. portion for the shorter wavelengths, which requires the stringent tolerances. The outer portion has a less severe tolerance limit (tolerance is equal to 1/20 of the wavelength). He deemed less than 1 min of arc field of view, with a 10 sec of arc pointing accuracy, as reasonable for such a large dish. The main problems here would appear to be in the control of thermal differences, in achieving pointing control and in finding erection techniques capable of obtaining the high dimensional accuracy needed.

3.2 ANALYSIS OF CONCEPTS. For the time period under consideration there was no significant support from the scientific community for a mm-wave dish larger than 10 ft in diameter. Since this size, for the purpose of this study, was not considered a large structure, no structures were conceived for this application. However, some of the communications concepts were rated against mm-wave astronomy usage criteria. See Section 9.

## SECTION 4

## X-RAY AND GAMMA-RAY ASTRONOMY

4.1 GAMMA-RAY ASTRONOMY USER REQUIREMENTS. Electromagnetic radiation of quantum energy greater than a few tens of kev can traverse the interstellar medium to distances of the order of the Hubble distance. Thus, there is no attenuation problem of such quanta between the Crab Nebula and the earth. However, the earth's atmosphere is opaque to such low energy gammas and any observation must be made at altitudes $\geq$ 70,000 feet. Lower energy x-ray measurements have been successful partially because incident fluxes for $0.1 < E < 10$ kev are relatively high and statistically significant data can be obtained in a few minutes. Measurements at higher energies have been far less successful because the flux falls off rapidly with increasing energy and because background radiation in the upper atmosphere is relatively high due to nuclear weapon debris and cosmic-ray showers.

The future of gamma-ray astronomy (above 1 Mev) is not as clearly defined as x-ray astronomy because no cosmic gamma-ray sources have yet been detected with any certainty. Although the first observations of gamma rays by space vehicles have been sensitive enough to reveal surprises of the magnitude revealed by x-ray astronomy, similar discoveries have not been made. There is some evidence from satellite observations, however, supporting the existence of a diffuse gamma-radiation background from space. Radio-astronomical data provide some evidence that discrete sources of gamma rays exist in the 100-Mev range. Current theories of nucleogenesis indicate that there should be significant amounts of gamma line emission from the decay of heavy radioactive elements, even in a supernova remnant as old as the Crab Nebula. But the gamma-ray flux from these processes is estimated to be two or more orders of magnitude less than would be detectable by instruments already flown.

4.1.1 Objectives of Observational Gamma-Ray Astronomy. With regard to gamma-ray astronomy, the less certain nature of the observations to date suggests a somewhat less specific set of problems:

a. Study of the directional and spectral characteristics of the diffuse gamma-ray flux in both the 1-Mev and 100-Mev regions, with the aim of establishing its real intensity (as distinguished from an upper limit), its origin (whether galactic or extragalactic, or both), and production mechanisms.

b. Study of the gamma-ray flux from strong radio sources with instruments of vastly improved sensitivity and angular resolution.

c. Study of nuclear gamma rays from supernova remnants, such as the Crab Nebula, with instruments capable of improved background rejection.

d. Studies of both diffuse and point sources of the 100-kev to 10-Mev range will require the continued development of actively shielded collimated detectors, as well as the development of techniques using Compton coincidence telescopes or pair spectrometers. Since this is an energy region where line emission is expected, energy resolution of high order will be required. Solid-state detectors should prove useful. Initial observations and background studies may be made with balloons. However, sky surveys at sensitivities of possibly $10^{-5}$ $cm^{-2}$ $sec^{-1}$ and employing detection systems with angular resolution of about 1° will require the long exposure time available only from a satellite.

e. Detection of more energetic gamma rays from discrete sources will require instruments of large collection factor (i.e., area times exposure time), angular resolutions of better than 1°, and, most important, a proven capability for background rejection. Both spark chambers of wide angular sensitivity (but with 1° resolution or better) and gas Cerenkov detectors of narrow angular sensitivity merit and require development. Above 300 Gev, the radiation can be detected with ground-based instruments.

f. Instruments of the type discussed under e above may, if their background rejection properties are adequate, be suitable for further clarification of the diffuse gamma-ray problem. Probably more suitable, however, are instruments especially designed to detect diffuse radiation.

4.1.2 Gamma-Ray Astronomy in the Range 0.1 - 30 Mev. Experiments in the next few years over the energy range 0.1 - 30 Mev will continue to be of an exploratory nature. These will be designed to search for point sources and gamma-ray lines, and to study the diffuse component. Important observational attempts, particularly on known x rays or radio waves will require that continued support be given to the development of detection devices using actively shielded collimators or Compton scattering telescopes, and the development of promising new techniques.

In the 0.1 to 30 Mev energy range, the short wavelength of the photons precludes efficient broadband focusing or interference techniques. Therefore, one must use detectors which depend on scattering or absorption phenomena. At energies above a few Mev, detection devices which utilize the pair production process become practical.

The impressive resolution improvement of solid state gamma detectors, Ge(Li), over the scintillation detectors, NaI (Tl), gives strong incentive for use of these relatively new detectors despite the fact that they must remain cold (225°K) at all times and require considerably more sophisticated electronics. In fact, their intrinsic resolution is so good that the present limiting factor is electronic noise.

4.1.3 Gamma-Ray Astronomy in the Range Above 100 Mev. No definite flow of photons in this energy range has been detected. Because of the low photon intensities and large background radiation, only upper limits to the flux have been determined.

In addition, the ratio of the radiation from point sources to the diffuse radiation may be very small, making the detection of point sources all the more difficult.

In contrast to x rays, ultraviolet light, visible light, and infrared and radio waves, the total primary flux of high-energy gammas, including both the diffuse radiation and all point sources, is known to be no more than a few quanta per square meter per second per steradian. A particularly important quantity to measure — the diffuse flux from interstellar collisions of cosmic rays with gas atoms in the plane of the galaxy — is expected to be about $10^{-4}$ $cm^{-2}$ $sec^{-1}$ $ster^{-1}$ in the direction of the galactic center. A substantial portion of the sky has already been scanned for point sources, with none appearing at intensities as high as $10^{-3}$ $cm^{-2}$ $sec^{-1}$. Optimistic theoretical estimates for the expected flux values yield only $10^{-5}$ $cm^{-2}$ $sec^{-1}$ for the most favorable source (the Crab Nebula), and realistic estimates for the various probable sources are in the range $10^{-7}$ to $10^{-8}$ or less. Surprises may appear, but these figures determine the sensitivity thresholds that are likely to be found necessary for detectors.

The primary flux of charged cosmic rays is at least 1000 times that of the primary high-energy gamma rays. Furthermore, in the atmosphere and in the walls of a vehicle or the detector itself, these charged particles are efficient generators of secondary gamma rays which (except for the directionality of radiation from point sources) are indistinguishable from the primary gammas one hopes to detect. Associated with this secondary production is the likelihood of long-range penetration and large-angle scattering, which rules out simple collimating schemes. In order to detect sources as weak as $10^{-7}$ $cm^{-2}$ $sec^{-1}$, one will need a rejection factor for such general background on the order of ten million; and to reach a sensitivity of $10^{-8}$ the rejection factor must be 100 times higher.

In addition to the radiation from point sources, a diffuse radiation is expected. The measurement of the diffuse flux will be of profound significance, particularly in revealing the composition of intergalactic space, but also for seeing the structure of the galaxy, especially its central part. The ratio of diffuse flux to point source strength will probably be substantially higher than it is for x rays and radio waves, which makes moderately good angular resolution a requisite for distinguishing point sources. In particular, the diffuse flux is expected to be between $10^{-5}$ and $10^{-4}$ $cm^{-2}$ $sec^{-1}$ $ster^{-1}$. For comparison, the current experimental upper limit is $3 \times 10^{-4}$. To observe a point source producing $10^{-7}$ $cm^{-1}$ $sec^{-1}$ against this diffuse radiation, angular resolution of about one degree is needed. Furthermore, in order to resolve the expected structure in the galaxy, the angular resolution of the detectors should be no worse than 0.5°.

It is essential that the detectors provide energy resolution. To do this, the apparatus for measurement of high-energy gammas must be dense, and the dimensions anticipated in meters. Experiments may be expected to grow gradually in size and weight from present levels of 500 lb weight and 1000 $cm^2$ area to future experiments with sensitive areas of at least 10 $m^2$ and a weight of 10,000 lb.

It should be made clear, however, that the attainment of sensitivies as high as $10^{-8}$ requires a fairly large-scale effort (e.g., the product At must then exceed $10^{11}$ $cm^2$ sec), as could be obtained with an apparatus having a sensitive area of 10 $m^2$, directed at single sources for exposure times of about two weeks. This would be pointed only in directions where optical, radio, or x-ray data had indicated the presence of an interesting object to be investigated.

4.1.4 Conclusions. The gamma-ray astronomy user requirements that may be feasibly fulfilled in the period of 1968-72 are:

| | |
|---|---|
| Lifetime | Several months desired. |
| Orbit Altitude | 200 n.mi. or lower, or well above radiation belts. |
| Field of View | Two deg. desired, but larger field is acceptable, depending on synthesis techniques. |
| Resolution | Accomplished by aspect determination or occultation. |
| Pointing Accuracy | <1° or ± 1/2 field of view or better. Alternative target and background observations desirable. |
| Aspect Determination Interval | 1/10 field of view or better. |
| Sensor or Detector | Anticoincidence shielded scintillation counters, spark chambers, or Cerenkov counters. |
| Bandwidth | 40 kev-20Mev using anticoincidence shielded scintillation counters, 30 Mev -300 or 400 Mev using spark chambers. Above 0.5 Bev gas-filled Cerenkov counters. |
| Spectral Resolution | 10 per cent of frequency. |
| Sensitivity | $10^{-3}$ photon/$cm^2$-sec-ster to $10^{-8}$ photon/$cm^2$-sec-ster. |
| Effective Collecting Area | Up to 10 $M^2$, but 1 $M^2$ suffices. |
| Lock-on Time | 6-144 hr; sweep with aspect determination may be employed when possible. |
| Structural Tolerance | The entire array must be mechanically aligned to within the capability provided by aspect pointing knowledge. |

For x-ray and gamma-ray astronomy, there should be an open cone of 30 degrees clear of any equipment obstructions emanating from the arrays.

4.2 X-RAY ASTRONOMY USER REQUIREMENTS. High energy astrophysical phenomena which reveal themselves and their properties through the emission of x and gamma radiation include stellar explosions, nuclear processes, the acceleration of elementary particles, and the interaction of charged particles with fields and matter in space. Systematic sky surveys in the high energy region should result in the discovery of remnants of more supernovas; high angular resolution, high energy resolution, and polarization measurements will provide information on their structure and on the nature of the processes involved. Energetic electrons reveal themselves through their Compton and synchrotron x and gamma radiation. X and gamma rays resulting from the interaction of cosmic rays with interstellar matter can also provide a means of obtaining a fuller understanding of interstellar matter, magnetic fields, and cosmic rays.

The 1965 Woods Hole Report gave the following ultimate objectives of observational x-ray astronomy:

a. Search for weaker discrete sources ($10^{-3}$ to $10^{-6}$ times the strength of those observed so far).

b. Precise determination (to 1 min or better) of the location of the discrete sources in order to make possible identification of the x-ray sources with optical or radio objects.

c. Study of the structure of the discrete x-ray sources with a resolution of better than 5 sec, or establishing an upper limit of this order of magnitude for their size.

d. Study of the spectral distribution of the radiation from the various discrete sources: search for emission lines, absorption edges, and the long wavelength cutoff expected from interstellar absorption.

e. Search for polarization of the x radiation from discrete sources.

f. Directional and spectral study of the diffuse radiation, with the aim of establishing its galactic or extragalactic origin, and investigation of the properties of the media in which it arises and through which it passes.

g. Search for time variation of both long and short duration in the intensity of the discrete sources.

4.2.1 X-Ray Proportional Counter Requirements. X-ray detectors, provided with mechanical collimators (the only type of instrument used so far for nonsolar x rays), will continue to play an important role in the investigation of most of the problems mentioned above, especially if the detector areas can be substantially increased.

Occultation methods have provided some of the most definitive measurements of source structures in radio astronomy. Identifications of x-ray sources with optical or radio objects have been made utilizing this method. Lunar occultations will continue to provide fine resolution of size and accurate positional data, and for these measurements large area detectors are essential.

All of the x-ray instruments flown thus far have employed simple mechanical collimation to define the field of view. Sensitivity has been achieved through the use of large area detectors: Geiger counters, proportional counters, and scintillators.

Various collimators have been used with large area detectors. Hexagonal honeycomb gives a nearly circular field of view and has been used effectively for locating sources in a broad expanse of sky. Rectangular cellular collimators have been employed to increase the resolution in the scan direction, sometimes at the expense of expanding the field of view in the direction perpendicular to the scan. The modulation collimator, consisting of parallel grids, has provided upper limits on source dimensions as small as 20 sec of arc.

Spectral information from large area detectors has been derived by use of pulse height discrimination and by filter photometry. Large area proportional counters achieve 20% energy resolution in practice.

The user requirements for x-ray astronomy proportional counter arrays are:

| | |
|---|---|
| Lifetime | One year. |
| Orbit Altitude | 200 n.mi. or lower preferred. 0° - 28.5° inclination. |
| Field of View | One deg maximum. |
| Resolution | Can be improved to arc seconds by use of ODA type collimator, sweeping across the source, and synthesizing the direction of the incoming radiation. By occultation to about 1' arc. |
| Pointing Accuracy | $<\pm$ 0.50°; prefer $<\pm$ 6' arc. Alternative target and background looks. |
| Aspect Determination | Required as fine as possible; at least 0.5' arc in scan mode. |
| Sensor or Detector | Gas-filled proportional counters, anticoincidence arrangement required. |

| | |
|---|---|
| Bandwidth | 1-20 kev desired. |
| Spectral Resolution | 40 per cent at 1 kev to 20 per cent at 10 kev desired. |
| Sensitivity | 1-10 kev range demands minimum power detectable of about $10^{-2}$ photons/$cm^2$-sec-ster. 10-40 kev range demands about $10^{-4}$ photons/$cm^2$-sec-ster. |
| Effective Collecting Area | As large as possible, up to 100 $m^2$. |
| Lock-on Time | 5-360 minutes, but drift modes may be employed up to 1°/sec. |
| Structural Tolerance | The field of view accomplished by collimators. However, the entire array must be mechanically aligned to within the resolution capability provided by mechanical collimators. |
| Physical Dimensions | Individual cell elements 2 by 2 ft by 8 in. deep, plus electronics and structure. |

4.2.2 X-Ray Grazing Incidence Telescope Requirements. Imaging x-ray optics have already been used successfully for solar observations. Such telescopes will be utilized on the ATM for the observation of cosmic x-ray sources. In principle, they can produce images of cosmic x-ray sources with angular resolutions comparable to what can be achieved in visible light. In addition, they can focus a parallel beam of x rays impinging on a large collecting area onto a small target area, where it can then be detected with a much improved signal-to-noise ratio. The latter application permits the use of dispersive techniques for high-resolution spectroscopic measurements, and affords new possibilities for polarization measurements. The x-ray reflecting telescope should prove to be a powerful tool for the study of galactic and extragalactic sources at wavelengths greater than 1 Å.

The 1965 Woods Hole report and the 1967 President's Scientific Advisory Committee report recommend that plans should be initiated for orbiting an x-ray telescope of long focal length. Lengths in the range of 30 ft to 100 ft are considered necessary to achieve the desired sensitivity and resolution.

The optical devices for focusing x rays consist of reflectors on which x rays impinge at small angles and are totally externally reflected. Total reflection optics are essentially achromatic at wavelengths longer than a certain cutoff value that depends on the atomic number of the reflecting material. The lowest cutoff wavelength that can be achieved at present is at 2 or 3 Å. Reflection optics do not substantially alter the spectral shape and the state of polarization of the impinging radiation. Two or more reflections are necessary to remove first-order aberrations of an imaging focusing system.

For most cases in which imaging systems are used in practice, the lower limit on the intensity of a discrete source that can be detected by an x-ray telescope is set not by the signal-to-noise ratio but by the requirement that a significant number of photons be detected within an image resolution element. This can be understood if one considers that the area of a resolution element in the focal plane is many orders of magnitude smaller than the area of collection. The part of the background that depends on the area of the detector element, such as that produced by cosmic rays, will therefore be reduced by a corresponding factor when compared with detectors with only mechanical collimation, and can be further reduced by increasing the angular resolution of the device. The diffuse cosmic x-ray flux is much weaker than the background from other factors. In view of these considerations, the effective sensitivity of a focusing telescope is proportional to At, the product of the area of collection and the time available for observation. The sensitivity of mechanically collimated detectors, which is determined by the signal-to-noise ratio, is, on the other hand, proportional to $(At)^{1/2}$.

This telescope is intended only for detailed study of previously identified sources or areas of interest in the x-ray sky. The spatial resolution desired is 1" arc for imaging investigations. In addition, polarimetry and spectrometry investigations will be carried out. No attempt has been made to predict the number of sources or areas of the sky which will be of interest by the flight time of this instrument (at present envisaged as 1975), but a safe guess is that the presently known two dozen or so x-ray sources will expand to several hundred. The objective will be to:

a. Study their structural nature (with 1" arc resolution).

b. Locate these sources precisely (within arc seconds).

c. Measure their intensity of x-radiation vs wavelength.

d. Determine their expanse or physical size (to seconds of arc).

e. Determine polarization.

f. Determine time intensity variations, if any.

One-Dimensional Focusing System. To conduct surveys for either discrete or extended x-ray sources in the low-energy ($\geq 8$ kev) range, one may use focusing systems with line-shaped fields of view. A large field of view in one dimension can be obtained with a nest of curved mirrors, arranged somewhat like a simple Venetian blind. Both high sensitivity for detecting discrete sources and precise positional information in one coordinate may be obtained with such a system. Its signal-to-noise ratio lies between that of the large-area detectors with cellular collimators and that of the imaging telescopes discussed above. For medium-sized instruments and with moderate observing times, the sensitivity may be comparable to that obtained in a full imaging system.

Detection Techniques. Although conventional x-ray detectors can be used with focusing x-ray optics in certain applications, the highest resolution will be realized with photographic recording or electronic imaging systems. Electronic image detectors, which are used at present in other branches of astronomy, are more sensitive than photographic emulsions and may be capable of comparable resolutions in the x-ray region. Their principal limitation may be the large rates of data transmission that they require.

We then can state that:

a. Large area (100 square meters) x-ray detectors equipped with mechanical collimators appear to be the optimum system for certain types of surveys and for occultation measurements.

b. A large Venetian blind array supplemented by a small mechanically collimated counter may be used if the survey emphasis is to be placed on detection sensitivity and $\leq 1$ minute of arc position finding accuracy.

c. Imaging x-ray telescopes appear to be the most promising for detailed studies of source structure and for high-resolution surveys.

The design constraints that are applicable to the large imaging grazing incidence x-ray telescope have taken the following form, due in part to the constraint of flying on the Apollo Applications Missions. This constrains the satellite to a 260-n. mi. circular orbit, with a 28.5° inclination, according to current planning.

| | |
|---|---|
| Lifetime | At least a year. |
| Orbit | 260 n.mi. or lower circular, 28.5° inclination. |
| Field of View or Beamwidth | About 10' to 30' arc. |
| Angular Resolution | 2" arc or better; depends on jitter rate and number of exposures per second. |
| Pointing Accuracy | 5' arc offset. |
| Lock-on Capability | Within a 1' arc square. |
| Jitter | The jitter must not exceed the angular resolution desired. |
| Sensors or Detectors | Image intensifier plus film or electronic imaging tube. Spectrometer. Polarimeter. |

| | |
|---|---|
| Bandwidth | Approximately 2 Å to 300 Å. |
| Spectral Resolution | $\lambda/\Delta\lambda$ = 100 to 1,000 or better. |
| Collecting Area | 200 $cm^2$ minimum effective collecting area. |
| Lock-on Time | 10 minutes to several hours. |
| Data | 5 to 10 x $10^6$ bits per orbit. |
| Structural Tolerance | Paraboloidal-hyperboloidal reflecting surfaces must be precision machined and the electro-deposited nickel or flame-sprayed nickel oxide optically polished to reflect grazing incidence x rays down to at least 2 Å wavelength. The alignment of the reflecting mirrors with the focal point must be kept so that the image remains entirely on the detector sensing area. |
| Physical, Unstowed, Dimensions | Up to 10 feet diameter by up to 100 feet long. |

4.3 ANALYSIS OF CONCEPTS. The results of the conception and analysis of structures for x-ray and gamma-ray detectors are presented. Major characteristics are compared in Table 4-1.

Table 4-1. Concept Comparisons

| CONCEPT | GAMMA RAY MODULES | X-RAY MODULES | X-RAY TELESCOPE | WEIGHT | ATTITUDE CONTROL SYSTEM | STOWAGE |
|---|---|---|---|---|---|---|
| 71 | 11 sq ft | 352 sq ft | --- | 13,118 lb | LEM + Moment Wheels | Below Rack |
| 72 | ---- | 1,000 sq ft | --- | 16,576 lb | CSM + Moment Wheels | Below and Above Rack |
| 73 | ---- | ----- | 10 ft Dia | 22,000 lb | Own | Entire SLA |

Common Characteristics: Launch in unmanned Saturn IB SLA. Rendevouz in 200 n.mi. orbit for alignment in zero g and to retrieve and resupply film, etc.

4.3.1 Concept 71, X and Gamma Ray Detector. This concept consists of 352 square feet of x-ray collimators mounted in a single plane and gamma ray detectors with the same orientation. See Figure 4-1. The x-ray detectors are mechanically collimated, gas-filled proportionate counters mounted to eight flat trapezoid skin stringer panels. Each panel is attached to the top octagonal ring of the structural rack with two hinge fittings. The panels are stowed in the LEM adapter. The gamma ray concept utilizes spectrometers using Ge(Li) super cooled detectors, total energy integrating devices using anti-coincidence shielded detectors, spark chambers, or Cerenkov counters. The cooling system is a closed loop refrigerator. The cryogenic system components are housed in the structural rack and anchored to the tubular members.

The attitude of the detector array assembly is controlled by the appropriate RCS system of the CSM or LEM. This concept is stowed under the LEM and launched with a Saturn IB into a 200 n.mi. orbit. Deployment is essentially automatic, but EVA is required for maintenance and resupply of the detectors.

4.3.1.1 Structure and Mechanics. The envelope of the SLA lends itself to movable petals as the support for the x-ray experiment modules. The structural concept for these petals is a trapezoid honeycomb box structure 2 to 4 in. deep x 84 in. x 54 in. x 125 in. long. Each module is attached to the honeycomb with four adjustable fittings. The maximum torque, due to slewing and/or launch loading, that can be applied to each petal for a 0.3° deflection, is:

$$\text{Torque} = T = \frac{\theta 4\, A^2 tG}{LP}$$

$$T = (0.052)\,(4)\,(0.300 \times 10^4)\,(0.020)\,(3.95 \times 10^6)$$

$$T = \frac{2.38 \times 10^8}{1.54 \times 10^4} = 1.54^{10^4} = 15,400 \text{ in.-lb}$$

The petal support structure which attaches to the hinge fittings is a 2219 aluminum welded tubular box truss. Long diagonal tubes in the truss are designed for column loading with a $l/\rho$ not to exceed 150, which keeps the columns in the "short" range. Honeycomb panels (using minimum gages) can be built for 0.5 to 1 lb per sq ft and truss support structure for 0.5 to 1.2 lb per sq ft. Each x-ray module weighs approximately 15 lb. Thermal distortions are minimized by applying a passive thermal control system to the panels and truss, and by providing a sliding joint on the panel-to-truss attachment, allowing the panel or truss to slide.

The x-ray modules are attached to the panels with fittings which can be adjusted to ± 0.005 of their true dimension. This includes thermal expansion and deflection tolerances. The structure therefore uses 0.3° of the required 0.5°, leaving 0.2° for pointing tolerance.

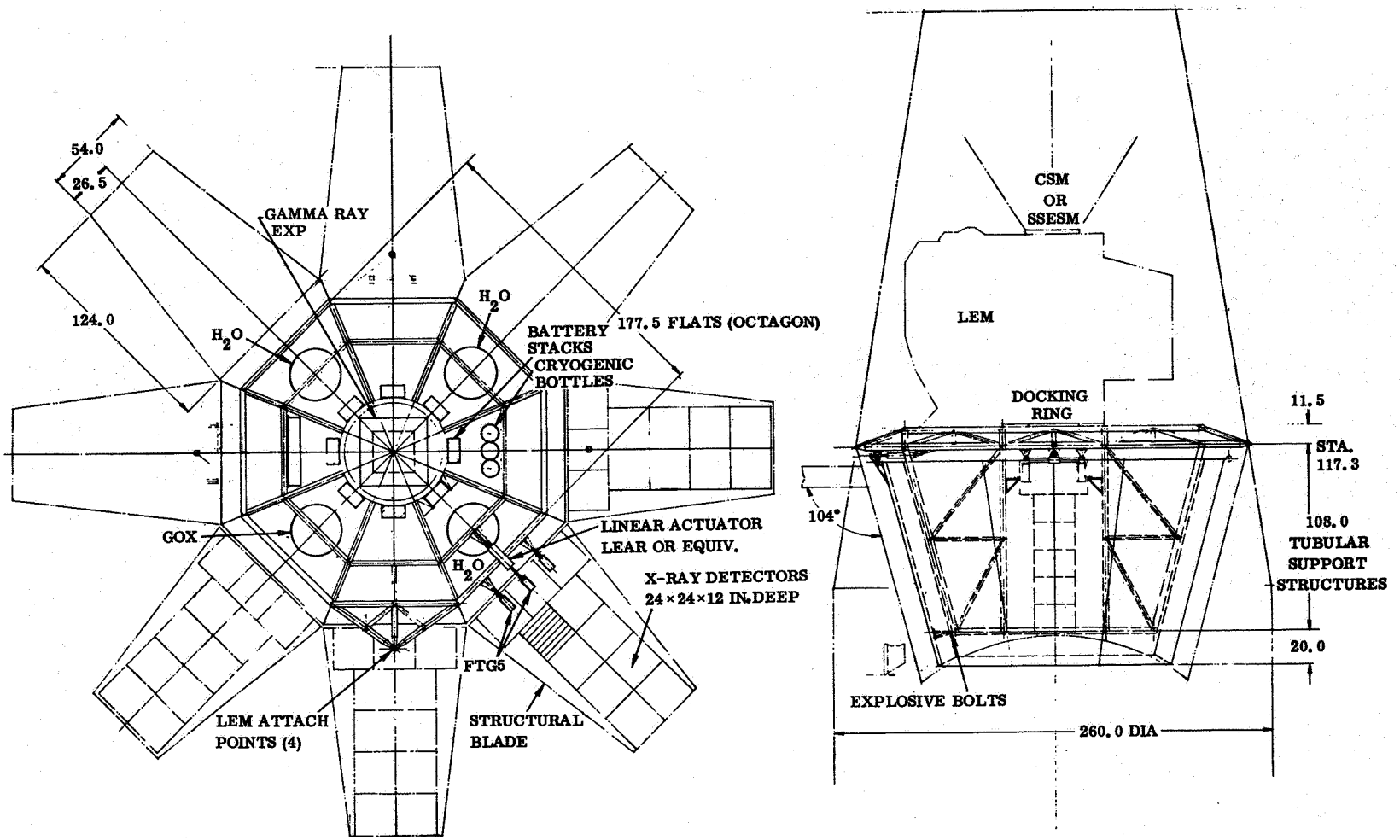



Figure 4-1. X-ray and Gamma-ray Concept 71

An open truss type structure is also used to attach the experiment support structure to the four LEM pickup points at Station 117.3. The components of the support systems are attached to the truss members by use of welded, machined fittings and sheet metal platforms. The electronic components and batteries are attached to the truss members with machined fittings and welded platforms. All system component units are placed about the centerline so that their weight and distance from the center will provide the maximum moment of inertia at each axis.

Each panel is attached to its support structure with two fixed fittings at the bottom, and two hinge fittings at the top. The two fixed fittings are fastened to the truss structure with two explosive bolts. A linear actuator rotates the panel 90° about the two top hinge fittings into a pointing position.

A standard Apollo docking ring is provided at the top and bottom of the panel support structure to accommodate the CSM or LEM. The gamma ray detector instrumentation is mounted directly on the support structure on the centerline of the spacecraft.

4.3.1.2 Attitude Control and Dynamics. This concept appears to be feasible with respect to dynamics and attitude control. The structural frequencies of the petals are anticipated to be separated from the control frequencies so that no coupling will occur. Pointing accuracies of six minutes of arc can be met by vehicles of this type. The orbital altitude of 200 n.mi. or less, combined with the effects of the petals on the drag and inertial properties of the vehicle, result in high propellant requirements. These could be reduced by the utilization of momentum storage devices.

The requirements shown in Table 4-1 are calculated for the CSM-LEM ascent stage vehicle at 200 n.mi., for one year. Reductions in altitude sharply increase the requirements. The roll requirements are much smaller than pitch or yaw and therefore are not estimated.

Gravity-gradient torques are estimated using $1.13 \times 10^6$ slug-ft$^2$ for the longitudinal inertia and $2.79 \times 10^6$ slug-ft$^2$ for the two transverse inertias. The maximums are 0.304 ft-lb pitch and 0.101 ft-lb yaw. The average is 0.142 ft-lb.

The aerodynamic drag is 0.0156 lb. The maximum moment arm is 8.33 ft. The average aerodynamic torque for pitch or yaw is 0.0325 ft-lb.

The maneuvering requirements are estimated assuming the vehicle is reoriented once each hour at a velocity of one radian per hour. The impulse to accelerate the vehicle is 77 ft-lb-sec.

The momentum and propellant requirements are shown in Table 4-2 with a specific impulse of 300 sec. Total propellant weight appears to be high, so shorter mission lifetimes are desirable. Since maneuvering represents only 19.5% of the total requirements, a reduction in the relatively large number of re-orientations which were assumed would not result in significant weight savings.

Table 4-2. Momentum and Propellant Requirements, CSM-LEM Ascent Stage, 200 n.mi.

| | Pitch or Yaw |
|---|---|
| Rotational Impulse, ft-lb-sec | |
| Gravity-gradient | $4.48 \times 10^6$ |
| Aerodynamic | $1.05 \times 10^6$ |
| Maneuvering | $1.35 \times 10^6$ |
| TOTAL | $7.99 \times 10^6$ |
| Linear Impulse, lb-sec | |
| Gravity-gradient | $7.31 \times 10^5$ |
| Aerodynamic | $1.72 \times 10^5$ |
| Maneuvering | $2.21 \times 10^5$ |
| TOTAL | $11.24 \times 10^5$ |
| Propellant Weight, lb | |
| Gravity-gradient | 2,430 |
| Aerodynamic | 580 |
| Maneuvering | 734 |
| TOTAL | 3,744 |

Significant propellant weight savings can be achieved by using momentum storage devices to handle the maneuvering and the oscillatory portion of the external torques. The maneuvering impulse of 77 ft-lb-sec can be obtained from acceleration times up to one minute. The maneuvering torque is in the range from 1.28 to 77 ft-lb. The oscillatory external torques will be somewhat less than the 1,000 ft-lb-sec.

4.3.1.3 Stress Analysis and Weight. The estimated weight of the experiment (approximately 13,118 lb) is considerably less than the design payload (23,200 lb). Therefore, the LEM adapter and payload attaching points will be structurally adequate.

Each of the folding petals weighs approximately 542 lb. For an axial load factor of 4.9 (limit) the tension load in the panel skin is:

$$P = 4.9\ (1.25)\ (542) = 3,320 \text{ lb ult}$$

Assuming 10 in. of each skin working at each hinge:

$$f_t = \frac{13,320}{40(0.005)} = 16,600 \text{ psi (satisfactory)}$$

Similarly, if the panel edges are attached at the aft end of each panel, the bending moment for a 0.65-g limit lateral load factor is:

$$M = \frac{1}{8}(0.65)(1.25)(542)(124)$$

$$= 6,850 \text{ in.-lb}$$

and the face stress is:

$$f = \frac{6,850}{2(70)(0.005)} = 9,800 \text{ psi (satisfactory)}$$

A docking load of approximately 5,000 pounds is assumed, giving a load factor of 5,000/13,118 = 0.38 limit. This is considerably less than the load factor during launch, and docking loads will only be critical for local design of the docking ring and its supporting structure.

If docking of the CSM to the deployed experiment is considered, the limit load factor on the panels is 0.38-g (see above), and the cantilever moment is:

$$M = 0.38\ (1.25)(542)(\frac{124}{2}) = 16,000 \text{ ft-lb}$$

$$f = \frac{16000}{2(80)(0.005)} = 20,000 \text{ psi (adequate)}$$

The total launch weight of this concept should require an unmanned Saturn IB launch to place it at a 200 n.mi. orbital altitude. Under these conditions the experiment would consume about 57% of the available payload. Total attitude control propellant requirement will be partly supplied by tankage to utilize the additional payload available and relieve the CSM, which would be put in orbit with a second launch.

4.3.1.4 Crew Systems. To initiate deployment, the two explosive bolts that attach each x-ray module support panel to the top ring at Station 372.0 are fired, releasing the panels. The linear actuators rotate the panels into pointing position. The actuators are reversible and can move the panels anywhere within 0° to 90°.

Prior to launch each module is bench assembled and attached to the four fittings provided on the panel. The fittings are adjustable and can be aligned with optical tooling and locked in place. Upon deployment, the astronaut checks for full deployment and makes adjustments as necessary.

The astronaut support equipment is limited to what is necessary to support the EV crewman during the inspection, adjustment, and repair of the deployed detector, and replacement of cassettes in each x-ray module.

It consists of the following:

a. An extravehicular life support system (PLSS).

b. The necessary maneuvering devices (HHMU).

c. Portable worksite anchors.

d. Portable illumination systems.

e. Portable power supply systems.

f. A communications system.

g. The necessary repair tools and equipment.

Fixed and/or portable handholds will serve as maneuvering aids, eliminating the requirement for a AMU and saving the checkout and donning time required with such a device.

The anticipated crew functions include:

a. Performance of initial checkout, alignment, and calibration.

b. Maintenance of counters and spark chambers, adjustment of electronic equipment and replacement of defective modules.

c. Weekly recovery and replacement of films or emulsions.

d. Experiment management.

Experiment management includes modifying operating modes; changing logic and geometry of instrumentation; and making experiment decisions such as source selections, exposure times, repeating observations, and changing resolutions.

4.3.1.5 Concept Evaluation. The pointing accuracy of six minutes satisfies all gamma ray and x ray performance requirements. The maximum desired lockon time of 144 hours can be achieved at angles greater than about 36° out of the orbital plane, which avoids occultations by earth.

The structural concept outlined provides suitable packaging within the LEM adapter, present state-of-the-art construction, minimum thermal expansion, and compatibility with Apollo systems and programs.

Stowage, launch, deployment, and maintenance present no significant problems. Maintenance of the detectors is dependent upon frequency of resupply over an extended period.

The orbital altitude of 200 n.mi. might be high if high energy electrons (7 to 8 Mev) are still plentiful from the high altitude nuclear explosion of 9 July 1962. As electrons of this energy can produce annoying secondary x rays, further study of the problem is needed.

4.3.2 Concept 72, X-Ray Detector. This concept consists of 1,000 sq ft of x-ray detectors mounted in two parallel planes with opposite orientation. See Figure 4-2. The x-ray detectors are mechanically collimated gas-filled proportional counters mounted 1) on six flat trapezoid panels, 2) on the top of the modified LEM support structure, and 3) on two semi-circular panels.

The six trapezoidal panels are hinged to and stowed above the modified support structure; the two semi-circular panels are similarily hinged to and stowed below the structure. These eight panels are all deployed by being rotated outward by linear actuators approximately 90° to a position perpendicular to the longitudinal centerline. The assembly is controlled in attitude by the CSM. It requires sensitive attitude sensors to meet the six-minute pointing accuracy. It stows above and below the modified rack for launch by the unmanned Saturn IB to a 200 n.mi. orbit.

4.3.2.1 Structure and Mechanics. The envelope in the adapter lends itself to a movable petal concept. The geometry of a hexagon allows the large panels to be stowed within this envelope. Each of the support panels is a trapezoid 242.50 in. long by 70 in. wide (average) by 2 to 4 in. thick. It will be constructed of aluminum honeycomb box or plate stringer structure. Thirty-six x-ray modules are attached to this panel with four bolts each through adjustable fittings. The x-ray modules are attached to the panels with fittings which can be adjusted to ± 0.005 of their true dimension. This includes thermal expansion and deflection tolerances. The structure therefore uses 0.3° of the required 0.5°, leaving 0.2° for pointing tolerance.

The support for this panel is an open tubular Pratt truss, 24 in. wide by 8 to 10 in. deep by 242.50 in. long. It is a welded tubular (aluminum) box truss. Long diagonal tubes in the truss are designed for column loading with an $l/\rho$ not to exceed 150,

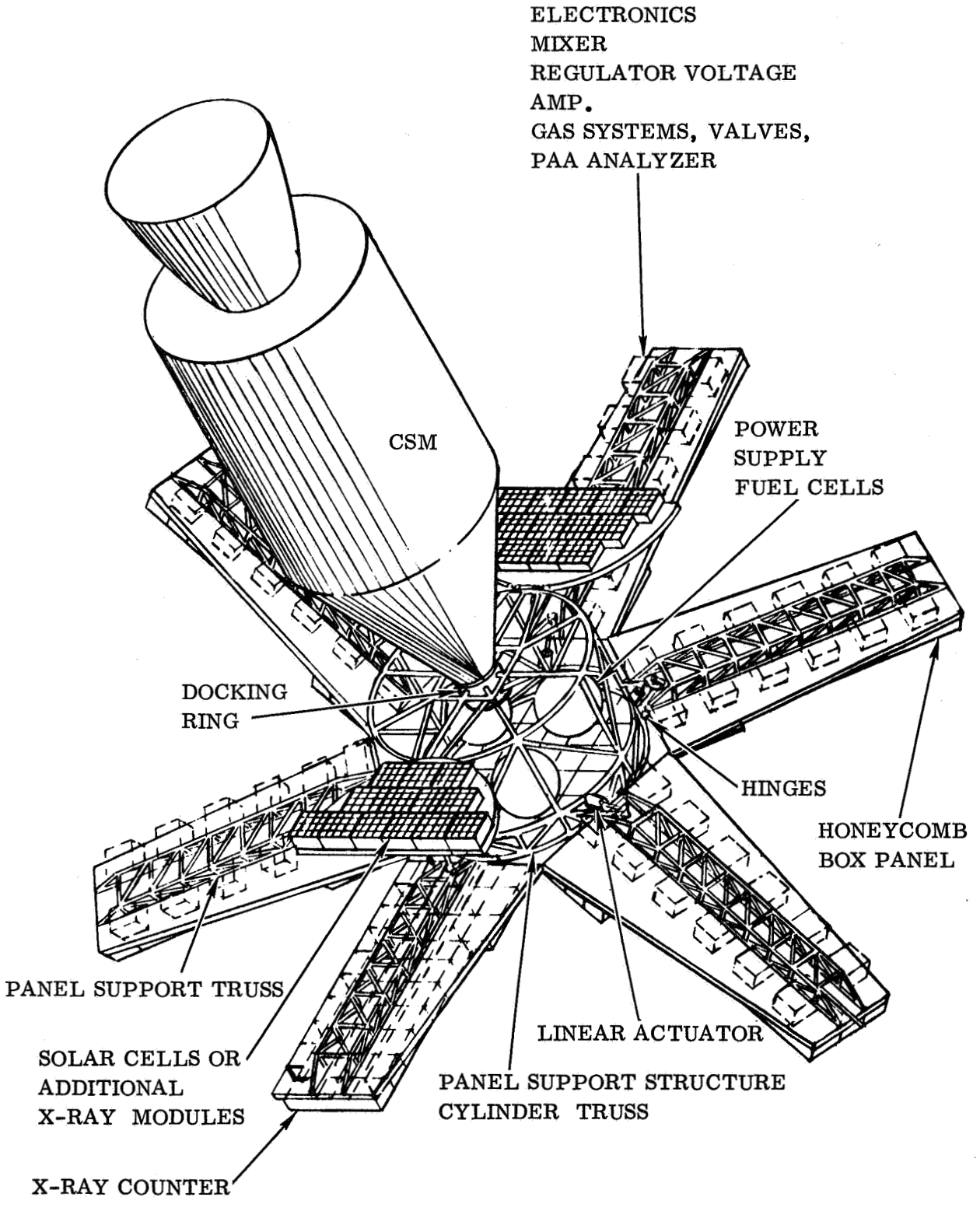



Figure 4-2. Concept 72, X-ray Detector Array

which keeps the columns in the "short" range. Honeycomb panels (using minimum gages) can be built for 0.5 to 1 lb per sq ft, and truss support structures for 0.5 to 1.2 lb per sq ft. Each x-ray module weighs approximately 15 lb. Thermal distortions are minimized by applying a passive thermal control system to the panels and truss and by providing a sliding joint on the panel-to-truss attachment allowing the panel or truss to slide. The lower section of the truss is attached to the basic support structure with two hinge fittings. The top section of the truss is anchored to the top ring of the LEM adapter with two explosive bolts through two fixed fittings.

The basic structure is a tubular circular truss cylinder 54 in. deep by 168 in. in diameter. Diagonal truss members attach this structure to the LEM pick-up points in four places. The subsystem component units are supported by brackets and machined fittings attached to the tubes of the basic structure. The top of this structure is used to support 28 additional x-ray modules in a fixed position on the truss. Additional x-ray modules or solar cells may be installed at the bottom of this structure, and made rotatable by use of a rotary motor.

4.3.2.2 Attitude Control and Dynamics. Orbital altitude is 200 n.mi. or less, operational life is several months to one year, and pointing stability is 6 minutes of arc. Maneuvering is required to orient the x-ray panels.

The environmental and gravity gradient torques are estimated for the x-ray experiment with the CSM. Longitudinal inertia is $1.43 \times 10^5$ and the two transverse inertias are $1.80 \times 10^5$ slug $ft^2$. The two semicircular petals introduce an asymmetry of 6% into the transverse inertias. This 6% difference is ignored here. The calculated torques are proportional to the difference between longitudinal and transverse inertias. Noting that the estimates are nearly the same, an assumed difference of $0.7 \times 10^5$ slug-$ft^2$ was used. The maximum torques are 0.126 pitch and 0.042 ft-lb yaw, and 0.0589 ft-lb is used as the average. Single axis yearly requirements are a rotational impulse of $1.86 \times 10^6$ ft-lb-sec.

The CSM has a 7-ft moment arm and the linear impulse is $2.65 \times 10^6$ lb-sec. A Isp of 300 sec gives 888 lb of propellant for a single axis for one year.

Aerodynamic torque is estimated using 1,317 sq ft of frontal area. The drag is $2.92 \times 10^{-2}$ lb. The vehicle cg is 3.8 ft above the plane of the upper petals, and four ft is taken as the maximum effective moment arm. The maximum torque is then 0.167 ft-lb the average for each of two axes is 0.0292 ft-lb. Single axis yearly requirements are $9.23 \times 10^5$ ft-lb-sec, $1.317 \times 10^5$ lb-sec, and 439 lb of propellant.

The maneuvering requirements are estimated assuming one reorientation per hour at a velocity of one radian per hour. The impulse to accelerate the vehicle is 49.8 ft-lb-sec. The yearly single axis requirements are $8.74 \times 10^5$ ft-lb-sec, $1.248 \times 10^5$ lb-sec, and 416 lb of propellant.

The limit cycle propellant requirements are estimated using the formulas shown for Concept 11, Section 5.

The minimum impulse for one engine is 1.6 lb-sec, the vehicle inertia is $1.80 \times 10^5$ slug $ft^2$, and $\theta$ is 0.2°. The average propellant expenditure for a single axis is $1.9 \times 10^{-4}$ lb/sec, which is 6,000 lb per year.

The yearly single axis propellant requirements for complete jet control are:

| Source | Propellant, lb |
|---|---|
| Gravity | 888 |
| Aerodynamic | 439 |
| Maneuvering | 416 |
| Limit Cycle | 6,000 |
| TOTAL | 7,743 |

This total is assumed to be excessive. Momentum storage devices can be used to provide the torquer requirements for all the maneuvering and limit cycling and the oscillatory portions of the gravity and aerodynamic torques. The momentum storage and torque requirements for this vehicle are met by current devices. This should result in combined single axis yearly propellant and momentum device requirements of approximately 1,000 lb.

4.3.2.3 Stress Analysis and Weight. The estimated weight of the experiment is approximately 17,000 lb, of which 12,000 lb is distributed among the 6 petals. The allowable payload for the attaching points on the LEM adaptor is 23,200 lb; therefore, the existing design is satisfactory. The static reaction at the forward attachments, due to a 0.65 g lateral load factor is 0.65 (6000) = 3900 lb limit. Although this is less than the applied load from the CSM, the method of introducing the load is different, and some redesign of the attachment points may be necessary.

Since the edges of the panels are joined at several places the structure forms a truncated hexagonal pyramid which is relatively rigid and readily analyzed. The trusses stiffen the panels to the point where vibration should not be a problem.

For a docking load of 5,000 lb, the load factor is 0.263 limit. This is considerably less than the load factor during launch; docking loads will only be critical for local design of the docking ring and its supporting structure.

Assuming a temperature gradient of 100° across the panel, the midpoint deflection, assuming ends in contact with the truss, would be 4.8 in.

The weight for this experiment approximates 13,576 lb. However, this only includes a minimum of attitude control propellant (sufficient to provide docking stability) and a 14-day power supply. To maintain attitude control for one year would require an additional 3,000 lb of propellant and momentum control devices. The resultant 16,576 lb is appreciably under the allowable 23,200 lb for the Saturn IB unmanned launch to a 200 n.mi. orbit.

4.3.2.4 Crew Systems. To initiate deployment, two explosive bolts that attach each x-ray module support panel are fired, releasing the panels. The linear actuators rotate the panels into pointing position. The actuators are reversible and can move the panels anywhere within 0° to 90°.

The EV crew functions that are required include:

a. The inspection of the detector after it has been deployed.

b. The repair or replacement of any detector component as required.

c. Recovery and replacement of films and emulsions.

The inspection task requires the visual inspection of each detector panel to insure that it has rotated properly, has not been locked in its deployed position, and has not been damaged.

The above inspection activity probably requires a minimum of 30 minutes. For component repair and/or replacement, additional time is required. The estimate of 30 minutes does not include the time required for pre-EV and post-EV activity such as donning of suit and spacecraft exit. These operations will require approximately 90 minutes prior to the inspection task and 60 minutes after completion.

The astronaut support equipment is limited to what is necessary to support the EV crewman during the inspection, adjustment, and repair of the deployed detector, and recovery of film after exposure. It consists of the following:

a. An extravehicular life support system (PLSS).

b. The necessary maneuvering devices (HHMU).

c. Portable worksite anchors.

d. Portable illumination systems.

e. Portable power supply systems.

f. A communications system.

g. The necessary repair tools and equipment.

Worksite lighting is required in the shadow of the detector panels or other components in addition to the dark side. This lighting could be portable. Helmet-mounted lights are preferred because they do not interfere with the task.

A power supply system is required to provide power for the EV crewman's repair tools, illumination systems, telemetry, voice communication, environmental control, etc.

4.3.2.5 Concept Evaluation. The petal configuration as applied to this concept is compatible with stowage in the adapter without any significant compromise. The tubular truss framework, supporting honeycomb panels with provision for thermal expansion, appears to meet the support requirements for the x-ray modules.

A major portion of the attitude control task could be done with Apollo hardware. Two additions to the CSM flight control system are required. More accurate sensors must be added to attain the desired pointing tolerance of 0.1°. Momentum storage devices are needed to reduce the propellant and, therefore, the system weight from that for an all-jet system. Since the control system goes up with a CSM separately from the experimental payload, the resulting control system weight will not influence the experiment payload considerations. The specifications of this additional hardware are well within current state of the art.

The pointing stability requirements of 0.1° are satisfied. Reduction of orbital altitude below 200 n.mi. rapidly increases aerodynamic torque and hence control torquer requirements, while reduction of the operation life below one year decreases these requirements.

The x-ray structure petal and vehicle structural frequencies are anticipated to be so widely separated from the control frequencies that coupling is precluded. The mission calls for low slewing rates and this insures low control frequencies.

4.3.3 Concept 73, Focusing Telescope for Soft X Ray. This concept uses parabolic/hyperbolic mirrors to collect x rays by grazing incidence according to the principles delineated by H. Wolter*. The telescope has an effective collecting area of 78 sq ft (10 ft dia) and a focal length of 100 ft.

It is noted that this concept is critical with respect to boom deflections from thermal and inertial influences. Also, a telescope of this type and size presents some severe fabrication and finish problems which may dictate a reduction in size for the time considered in this study.

*H. Wolter, Ann. Physik 10, Vol. 286, 1952.

The mirror elements and the image recorder are supported on four telescoping booms extending in both directions from a modified SLA mounting rack. The mirror is mounted by gimbals and vernier adjustments to assure alignment with the image recorder. Four struts extend radially from the mounting rack to support thrusters, solar cells, and batteries. Guy wires from these struts stabilize the telescope booms. Other subsystems, including momentum devices and telemetry, are mounted on the mounting rack.

The image recorder is primarily film which is stored in cassettes to facilitate automatic exposure and manual resupply. However, the target can be monitored by visual observation or by videcon.

The telescope satellite is stowed in the SLA of an unmanned Saturn IB for launch to a 200 n.mi. orbit. The deployment is basically automatic with EVA by one astronaut to attach and rig four guy wires and align the mirrors.

The lens consists of three nested concentric rings, the outer one ten feet in diameter and all of them eight feet long. The forward portion of the inside of each ring collects the soft x rays by means of grazing incidence on its parabolic surface and directs them to the rear hyperbolic section of the ring, where they are redirected into the shortened focal length of 100 ft.

4.3.3.1 Structure and Mechanics. As an outgrowth of an artist's conception of a focusing x-ray telescope depicted in a paper* by R. Giacconi, a structural configuration was evolved as shown in Figure 4-3. It reflects an attempt to balance the components about the three axes for minimal environmental and control torques, and to provide a structural configuration that is inherently stable.

The lens assembly is anticipated to be ribbed aluminum with sprayed nickel surfaces ground to the required tolerances. Shielding and thermal controls are assumed to be necessary.

The lens is mounted on four truss booms by means of vernier-adjusted attachments to permit alignment of the lens to the image recorder. The booms, in turn, are mounted on a truss structure that forms the base of the entire assembly and is located on the c.g. It is constructed of welded aluminum tubes.

The base structure carries the attachments that pick up the hard points on the SLA for launch. It also supports the pivoted radial struts, various subsystems, and the image recorder booms.

*AIAA Third Annual Meeting, 30 November 1966.

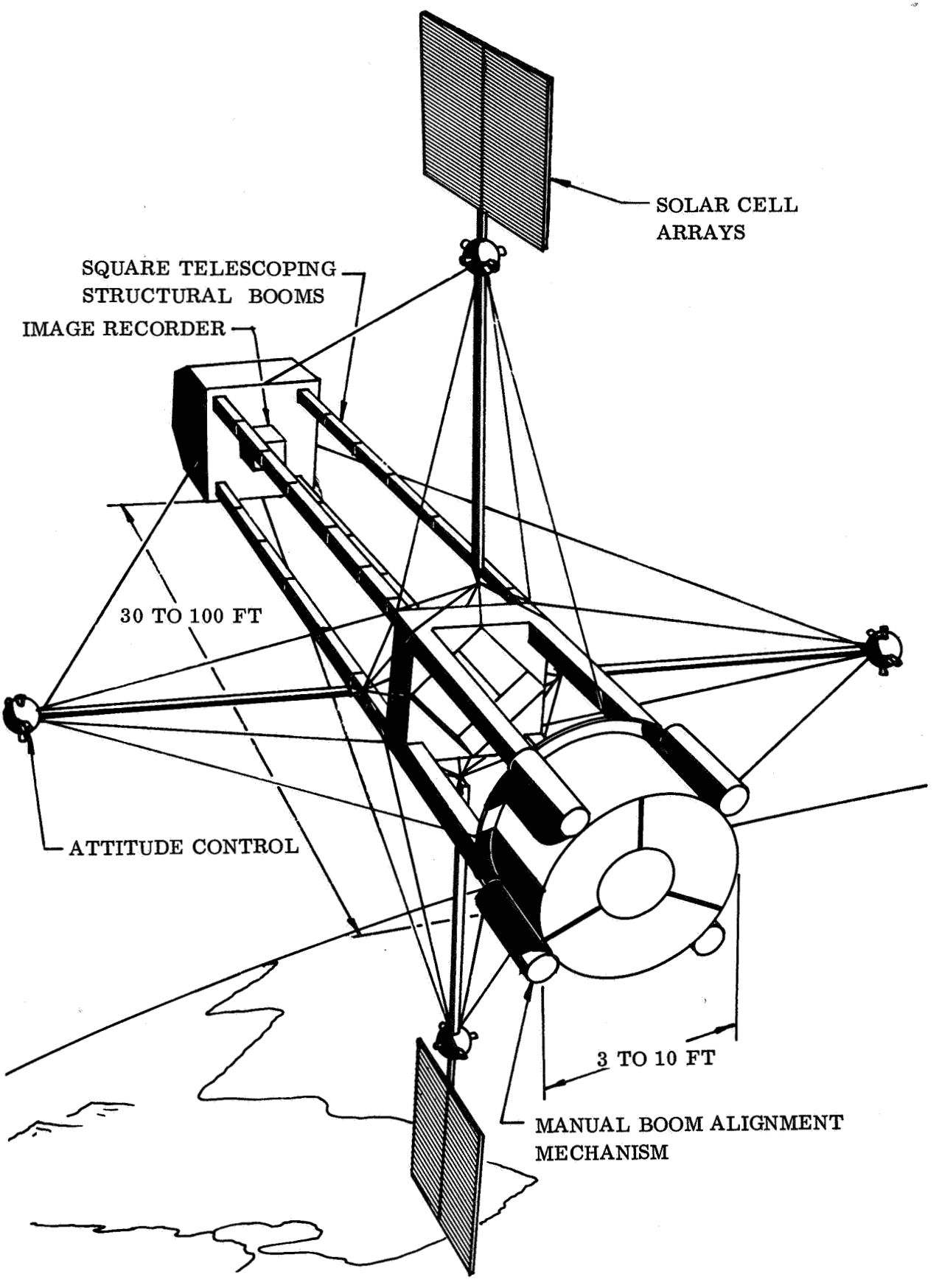



Figure 4-3. Concept 73, Focusing X-ray Telescope

These booms are four-sided, telescoping trusses diminishing in size from 2 ft square at the base to 6 in. square at the image recorder capsule. They telescope to the extent necessary to retract the image recorder capsule for stowage above the S-IVB tank dome.

The image recorder capsule contains the focal plane instrumentation and supplies, a standard docking port, and a monocoque structure suitable for a controlled environment. The film is on cassette rolls for automatic exposure and ease of resupply handling. It is anticipated that passive thermal control will protect the film.

Retractable mirrors can be inserted into the ray path to deflect visable rays into an optical lens or a videcon for visual or remote monitoring of the target.

Atmospheric control within the capsule may be provided by the CSM during resupply to avoid EVA if such is desired by the crew.

The four radial struts, also of welded tube construction, are pivoted on the base structure near the SLA attach points such that they will rotate into a stowed position alongside the telescope lens and within the SLA shrouds. On the outer end of each strut is supported a thruster unit including its propellant and controls, solar cells with their associated batteries, and guy wire attachments.

The subsystems mounted on the base structure include momentum wheels for attitude control, attitude sensors, and electronics for data handling and telemetry.

4.3.3.2 Dynamics and Attitude Control. The focusing x-ray telescope concept was re-examined for the redesigned configuration with relaxed photo imaging drift rate and exposure time requirements. Attitude control during the ten-minute observation period can be supplied by momentum storage devices, but the oscillatory external torques exceed current capacities of such devices. The propellant requirements are reduced to large but acceptable levels. During observation, the telescope structure will be deformed by momentum device torque.

Reviewing current requirements, drift rate for photo imaging is one arc sec per sec, and an exposure can be made in one second. The electronic imaging drift rate is 5 arc sec per sec. Other requirements include 200 n.mi. altitude, several months to one year lifetime, 5 arc min pointing tolerance, and maneuvering to observe specific x-ray sources.

Environmental torques are estimated for 200 n.mi. Maximum gravity torques are 1.13 and 0.376 ft-lb pitch and yaw, and the average is 0.528 ft-lb. Aerodynamic torque, a maximum of 0.1 ft-lb, is not considered further other than to note that it increases rapidly with lower orbits.

The average environment torque, if unopposed, will accelerate the telescope to an angular rate of one arc sec per sec in 7 sec and through an angle of 5 arc min in 65 seconds. Therefore, control torque must be supplied during the 10-min observation period. The momentum change is within the capacity of current momentum storage devices, and no jets need be fired during observations. Maneuvering can also be done by the devices, since the maneuvering momentum change is 212 ft-lb-sec.

However, the oscillatory portion of the environment torques exceeds the current device capacity and jets must be used to oppose environment torques. The yearly one-axis requirement is $1.67 \times 10^7$ ft-lb-sec. Assuming a 40-ft moment arm, the jet impulse is $4.17 \times 10^5$ lb-sec, and a Isp of 300 sec yields 1,396 lb of propellant for each of two axes. Total yearly propellant weight is estimated at 2,792 lb.

4.3.3.3 Stress Analysis. Based on the analysis made for the original concept, it is estimated that, with a reduction in boom length from 100 ft to approximately 75 ft, the addition of guy wires, and a relaxation of torque requirements, the proposed structure is acceptable. Mid-span guys can be added if required.

4.3.3.4 Weights. Preliminary weight estimates are:

| | |
|---|---|
| Lens | 5,000 lb |
| Focal plane capsule | 1,500 |
| Film | 1,500 |
| Booms | 4,000 |
| Thrusters and propellants | 3,200 |
| Solar cells and batteries | 1,660 |
| Momentum wheels | 1,200 |
| Struts and wires | 1,500 |
| Electronics | 500 |
| | 20,060 lb |

4.3.3.5 Crew Systems. After orbit injection and separation from the launch vehicle, the telescope assembly is deployed automatically. The telescoped booms are extended by a winch and cable system, pushing the image capsule ahead. The four radial struts are rotated into position by linear screw jacks. The solar cells are subsequently expanded and unfolded by spring energy. All guy wires, except the four attached to

the image capsule, are pre-attached and unreel, if necessary, as the four struts rotate.

The final steps require EVA. The astronaut unreels the guy wire from the outer end of each strut and attaches it to the capsule. He then rigs the wires for gross alignment of lens to image recorder and adjusts the vernier for fine alignment.

Following are the EV operations required after deployment:

a. Installation and rigging of four guy wires.

b. Adjustment of the mirror to the image recorder.

c. Inspection of the telescope.

d. Repair and/or replacement of any damaged telescope component.

The installation and rigging of the 4 guy wires requires approximately 30 minutes. Another period of 30 minutes is alloted to the adjustment of the mirror to the image recorder. Visual inspection of all components to insure full deployment and detect any damage incurred requires approximately 90 minutes. The foregoing estimates do not include the time required for pre- and post-EV activity, such as donning and removal of suit. These operations require approximately 90 minutes for pre-EV activity and 60 minutes for post-EV activity.

Assuming negligible time for repairs, the above tasks require a total time of approximately 240 minutes for 1 astronaut.

The equipment necessary to support the EV crewman is:

a. An extravehicular life support system (PLSS).

b. A maneuvering unit (HHMU, or AMU for 100 ft telescope).

c. Portable worksite anchors.

d. Fixed and/or portable illumination system.

e. Fixed and/or portable power supply system.

f. A communications system.

g. The necessary rigging and repair tools, and a tool to adjust mirror to image recorder.

The AMU would expedite the inspection activity which requires maneuvering over 100 ft from the CSM.

Fixed and/or portable hand holds on the surface of the structure are used during adjustments, rigging, repairs, and replacements.

Worksite lighting is required in the shadow of the telescope components in addition to the dark side. This lighting could be portable. Helmet-mounted lights are preferred because they do not interfere with the task.

A tool kit is provided for the EV crewman's tools. A temporary holding device attaches the kit to the worksite.

4.3.3.6 Concept Evaluation. This concept promises much desired soft x-ray data. The quantity is limited primarily by film capacity and secondarily by the number of resupply missions. The large multiple mirror provides high sensitivity and, together with the attitude control system, should permit good resolution.

The proposed 10-foot diameter mirror may exceed technical capabilities until beyond the time frame of this study.

Data handling, particularly its retrieval, poses the problems usually associated with large quantities of film. Data return capsules might be the answer.

Changes in the alignment of mirrors and the image recorder due to inertial forces or thermal distortion must be analyzed. A tradeoff analysis of extra film width, with resultant weight and volume, should be compared against more structural rigidity and servo compensation at the vernier mirror mounting. Automatic sensing of alignment by light ray and photo cell is feasible.

Attitude control can be provided to meet the required pointing by a combination of thrusters and momentum wheels. The optimum tradeoff may be affected by propellant stowage limitations near the mirror cluster. Propellant requirements for the concept appear to be minimized by the balance of inertias about the three axes.

The relationship between thruster and structural frequency appears satisfactory, and can be improved, if required, by the addition of more guy wires.

The design is quantitatively based on a 10-ft diameter mirror cluster with a 100-ft focal length. If such an instrument is beyond the next logical technological step the assembly can be scaled down to fit a smaller one with consequent loss of sensitivity but a compensating increase in film capacity.

# SECTION 5

# COMMUNICATIONS

5.1 COMMUNICATIONS ANTENNAS USER REQUIREMENTS. Radio communications utilize propagating electromagnetic waves to transfer information from one point to another. The communications antenna is simply a device to couple free space energy to a transmission line. Antenna gain is a function of antenna size and determines the degree to which energy is directed in a particular direction with respect to an isotropic radiator. The isotropic reference antenna radiates power uniformly in all angular directions.

Antenna gain increases with increasing electrical size which involves either enlarging the antenna physical dimensions and keeping the operating frequency constant or fixing the physical size and increasing the operating frequency.

The frequency bands of most interest for space application are shown in Figure 5-1. A partial tabulation of applications is also given. At the low end of the HF band, the inosphere serves as a reflecting mechanism creating a RF shield between the orbiting spacecraft and ground sites. The magnitude of reflection is statistical in nature with varying cutoff frequencies for different time periods and solar activity. Communication links become more reliable for frequencies about 10 MHz and improve with increasing frequency. Galactic and man-made noises are also problems at the lower frequencies, requiring increased transmitter power and antenna gain to overcome. These noises decrease with increasing frequency. At the upper end of the spectrum in the microwave and millimeter region, atmospheric effects attenuate RF energy totally or partially as it passes through the atmosphere. Absorption and scattering mechanisms; water vapor, oxygen, and precipitation, are also statistically treated phenomena. Energy absorbed is also re-radiated as noise, again degrading the communication link. There is an optimum frequency band between 1 and 10 GHz where ionospheric, galactic, and atmospheric effects are minimal. Radio propagation through the atmosphere allows utilization of frequencies between 10 GHz and 100 GHz for deep space monitoring relay stations.

Many different types of antennas are used for communications that under the definition of conveying information from one point to another include:

a. Broadcast

   1. Radio AM
   2. Radio FM
   3. TV

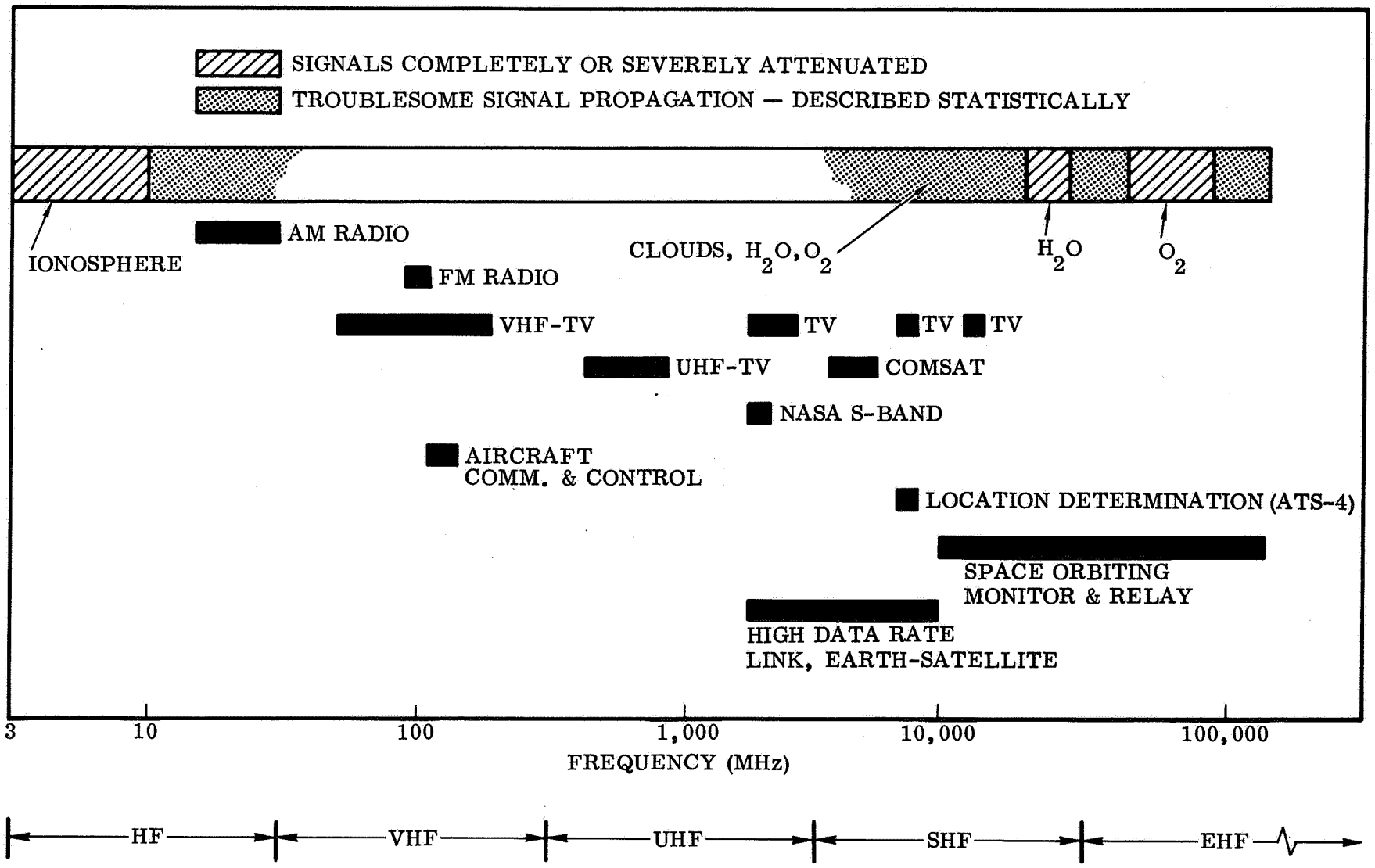



Figure 5-1. Frequency Spectrum for Communications Antennas

b. Space Communications

1. Satellite to earth

2. Satellite to satellite

c. Telephone

1. Personal

2. National

3. Teletype and facsimile

d. Traffice Control, Aircraft

1. Communications

2. Safety

e. Navigation

1. Location

2. Direction Finding

f. Public Safety

1. Law enforcement

2. Fire service

Propagation is essentially along the line of sight for frequencies in the upper HF band and higher; therefore greater broadcast coverage is obtained if the transmitting antenna is elevated. Although orbiting transmitters greatly increase the coverage area, for most orbits many transmitters are required to provide continuous service, but when in a synchronous equatorial orbit the transmitter becomes stationary, providing coverage for more than 30% of the earth's surface. Beamwidths for "hemispherical" coverage require physical antenna sizes and altitudes as shown in Figure 5-2. Large antennas are needed for medium and synchronous altitudes in both HF and VHF frequency bands.

Other requirements such as limited area coverage and point-to-point communications require large-to-maximum size antennas in all frequency bands as illustrated in Figure 5-3. Constant beamwidth values are shown as parameters with several typical earth area coverages. For a given beamwidth, the antenna size is inversely proportional to frequency. Using packaging ratios of 20:1, an antenna of approximately 350 ft can be erected with an unmanned Saturn-V launch. Rigid antennas having diameters nearly equal to that of the launch vehicle require no erection mechanism unless the unit is part of an array of rigid antennas. Erectable antennas receiving the most consideration are parabolic reflectors. These antennas will be gain limited by their achievable surface accuracy. The highest antenna operating frequency (and gain) is

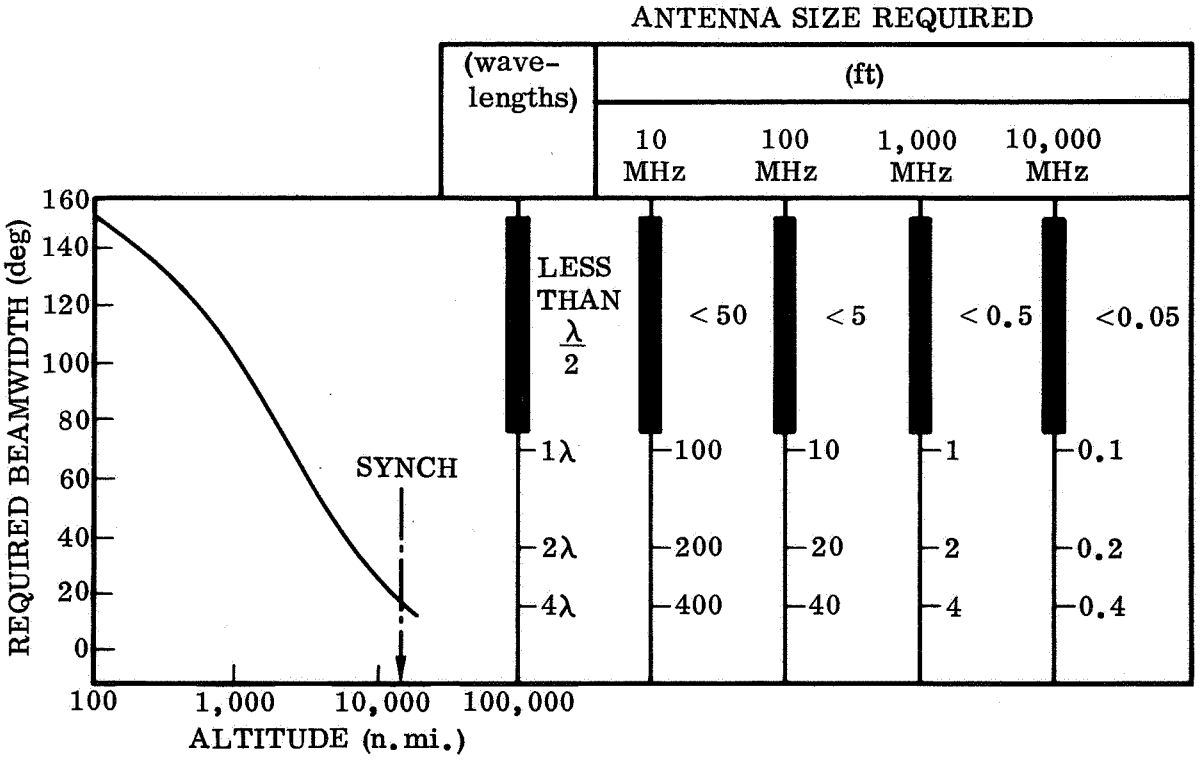



Figure 5-2. Antenna Size Requirements for Hemispherical Coverage

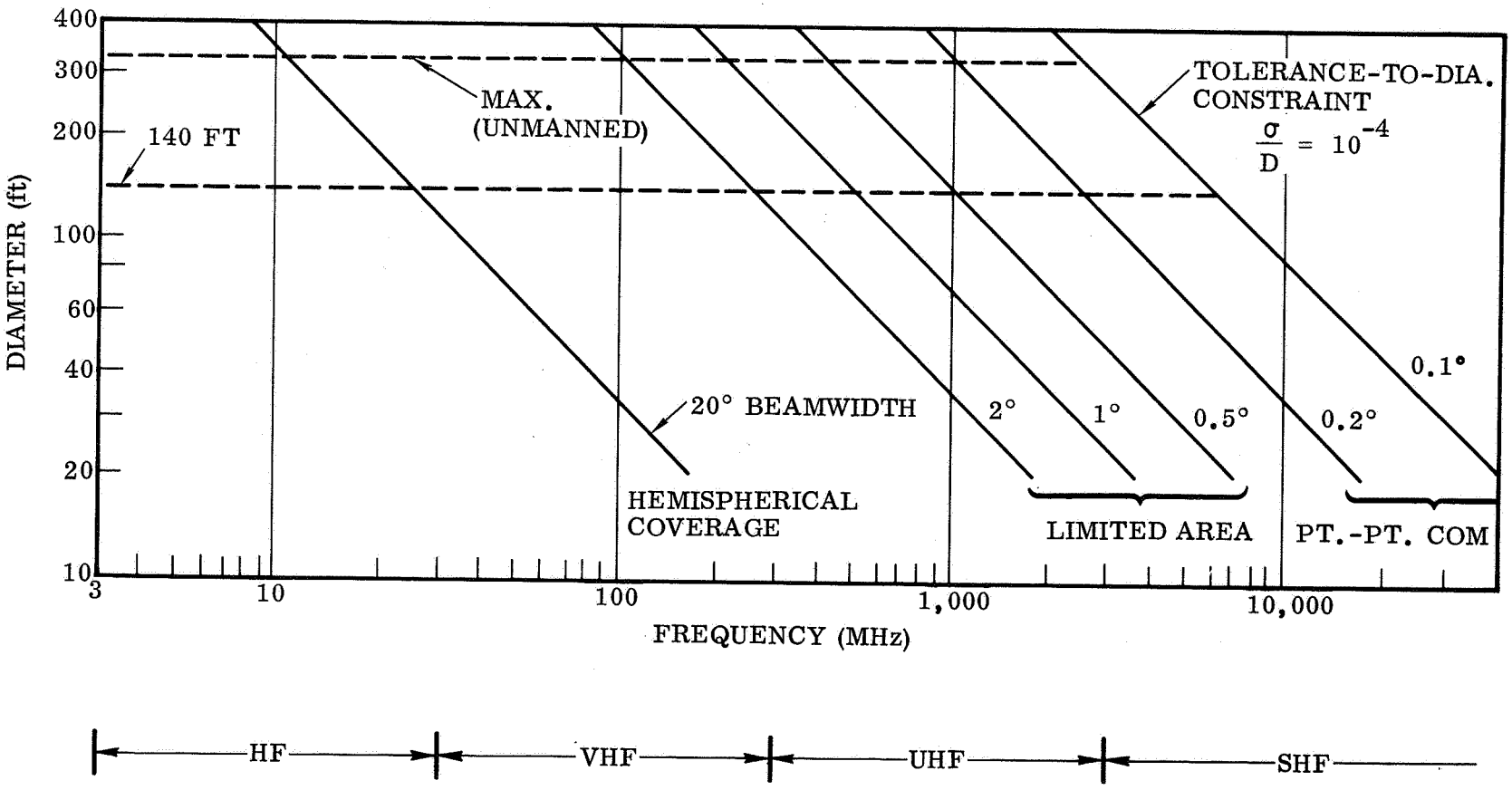



Figure 5-3. Antenna Size Requirements (Paraboloid) for Limited Area Coverage

constrained by tolerances along the constant 0.1° beamwidth line. To achieve this beamwidth, the RMS tolerances must be in the order of 1/10,000 of the antenna diameter. Although this is a difficult design requirement for large erectable antennas, it is readily achieved in smaller rigid dishes usable at the higher frequencies.

Angular pointing requirements are dependent on the antenna beamwidth, usually varying from 1/3 to 1/10 of the RF beam at the minus 3 db levels. A 1/3 beamwidth pointing tolerance corresponds to 1 db loss: 0.1 beamwidth, 0.1 db loss.

Specific antenna applications are considered in detail in Table 5-1.

## 5.2 ANALYSIS OF PROVIDED CONCEPTS

5.2.1 Concept 1, 60-Ft Parabolic Antenna (NASA Internal Note S-66-4). Concept 1, proposed by the P&VE Lab of MSFC, is a rigid, beryllium, Cassegranian, 60-ft parabolic antenna with an F/D ratio of 0.6. The antenna was considered for operation in the millimeter wave lengths, in high or low earth orbits packaged with the LEM descent stage. However, the millimeter surface tolerances of 0.1 to 2 mils cannot be met; therefore, the concept will be considered for communication uses that is will satisfy effectively. Although the attitude control propellant weight varies considerably between 200 n.mi. and synchronous orbit, (4,500 lb vs 22 lb respectively), the total system weight and packaged volumes are suitable for either S-IB or S-V launch vehicles. The deployment/assembly of this concept uses a high degree of astronaut participation. There does exist certain areas that can be more automated, with some penalty to the mechanical and structural simplicity. Figure 5-4 depicts the refined version.

5.2.1.1 Structure and Mechanics. Beryllium is the main structural material. Twenty-four extruded beryllium-tube ribs are the basic load-carrying and contour-supporting member. The reflective surface consists of 96 individual perforated beryllium skin panels held in place by permanent magnets. These skins and the Cassegrain sub-reflector are stowed below the descent stage at launch. The magnets will act as a mechanical damping system to relative movement between magnet-connected items. This is a problem because the magnetic attachments do not recognize a best indexed position and become built-in distortion devices. Also, unequal thermal loads will likely exist during antenna assembly and the magnets will permit a distorted contour to be retained.

The packaging concept is feasible. The antenna and its components can be stowed around and below the descent stage. This arrangement lends itself well to securing the slender segmented ribs to the descent stage. A large 15-ft, 8-in.-dia., fixed high-precision center section of the dish is possible. The feed system package is a permanent installation that does not require removal or alteration from its stowed position except for the provisions to determine alignment.

Table 5-1. Communication System Space Antenna/Transmitter Requirements

| FREQUENCY (Hz) | BEAMWIDTH | GAIN (db) | SIZE (ft) | BANDWIDTH (Hz/Channel) | TRANSMITTER POWER OUTPUT (watt) | ALTITUDE | LIFETIME | REMARKS |
|---|---|---|---|---|---|---|---|---|
| 1. Space Communications to Provide Instant Contact with most of Major Nations | | | | | | | | |
| 1G-10G | 17° | 19 | 0.4 - 4 | 4000 | 100 | Synch | 2-10 yr | 125 Duplex Channels |
| 2. Voice Broadcast Direct to Home | | | | | | | | |
| 15M-30M | 30° | 15 | 104x104x15 | 10k | 5k | Medium | 2-10 yr | |
| 88M-108M | 17° | 19 | 40 | 10k-150k | 5k | Synch | 2-10 yr | 5 Channels |
| 88M-108M | 20° | 18 | 35 | 19k-150k | 1k | Medium | 2-10 yr | 5 Channels |
| 3. Real-Time TV Direct to Home (20 db Ground Antenna, 6 db NF Preamp) | | | | | | | | |
| 900 M (Approx.) | 3.5° x 7.4° | 30 | 22 x 11 | 6M | 57k | Synch | 2-10 yr | U. S. Coverage 3 Channels |
| 900M (Approx.) | 2.6° x 3.5° | 35 | 29 x 22 | 6M | 20k | Synch | 2-10 yr | 1-hr time zone in U. S. (3 Channels) |
| 900M (Approx.) | 0.5° | 50 | 150 | 6M | 1k | Synch | 2-10 yr | 200 x 200 $Mi.^2$ (5 Channels) |
| 4. Real-Time TV to Distribution Center (42 db Ground Antenna, 3 db NF Preamp) | | | | | | | | |
| 0.9G to 4G | 3.5° x 7.4° | 30 | 22 x 11 | 6M | 600 | Synch | 5-10 yr | U. S. Coverage (10 Channels) |
| 0.9G to 4G | 2.6° x 3.5° | 35 | 29 x 22 | 6M | 200 | Synch | 5-10 yr | 1-hr time zone in U. S. (10 Channels) |
| 0.9G to 4G | 0.5° | 50 | 150 | 6M | 6 | Synch | 5-10 yr | 200 x 200 $Mi.^2$ (10 Channels) |
| 5. Telephone Communications, Personal Nature, Global | | | | | | | | |
| 1G-4G | 17° | 19 | 1-4 | 4K | 1.25K | Synch | 10 yr | Hemispherical Coverage (1000 Channels) |
| 1G-4G | 1° | 44 | 20-70 | 4K | 500 | Synch | 10 yr | Multiple Beam Coverage (1000 Channels) |

Table 5-1. Communication System Space Antenna/Transmitter Requirements, Contd

| FREQUENCY (Hz) | BEAMWIDTH | GAIN (db) | SIZE (ft) | BANDWIDTH (Hz/Channel) | TRANSMITTER POWER OUTPUT (watt) | ALTITUDE | LIFETIME | REMARKS |
|---|---|---|---|---|---|---|---|---|
| 6. Spacecraft to Spacecraft Communications | | | | | | | | |
| 12G-100G | 0.5° | 50 | 20 | | 10K | 200-240,000 mi. | | |
| 7. Orbiting Relays for Communications from Deep Space Probes | | | | | | | | |
| 2G-100G | 0.1° | 64 | 170-20 | $10^7$ bps | 10K | 200-22,300 mi. | 3 yr | |
| 8. High Data Rate, Point-to-Point Communications | | | | | | | | |
| 2G-10G | 0.2° | 58 | 170-35 | $10^6$ bps | 10K | 22,300 | 3 yr | |
| 9. Location Determination | | | | | | | | |
| 7G-8G | 0.25° | 56 | 40 | | 0.5K | 22,300 | 2 yr | |

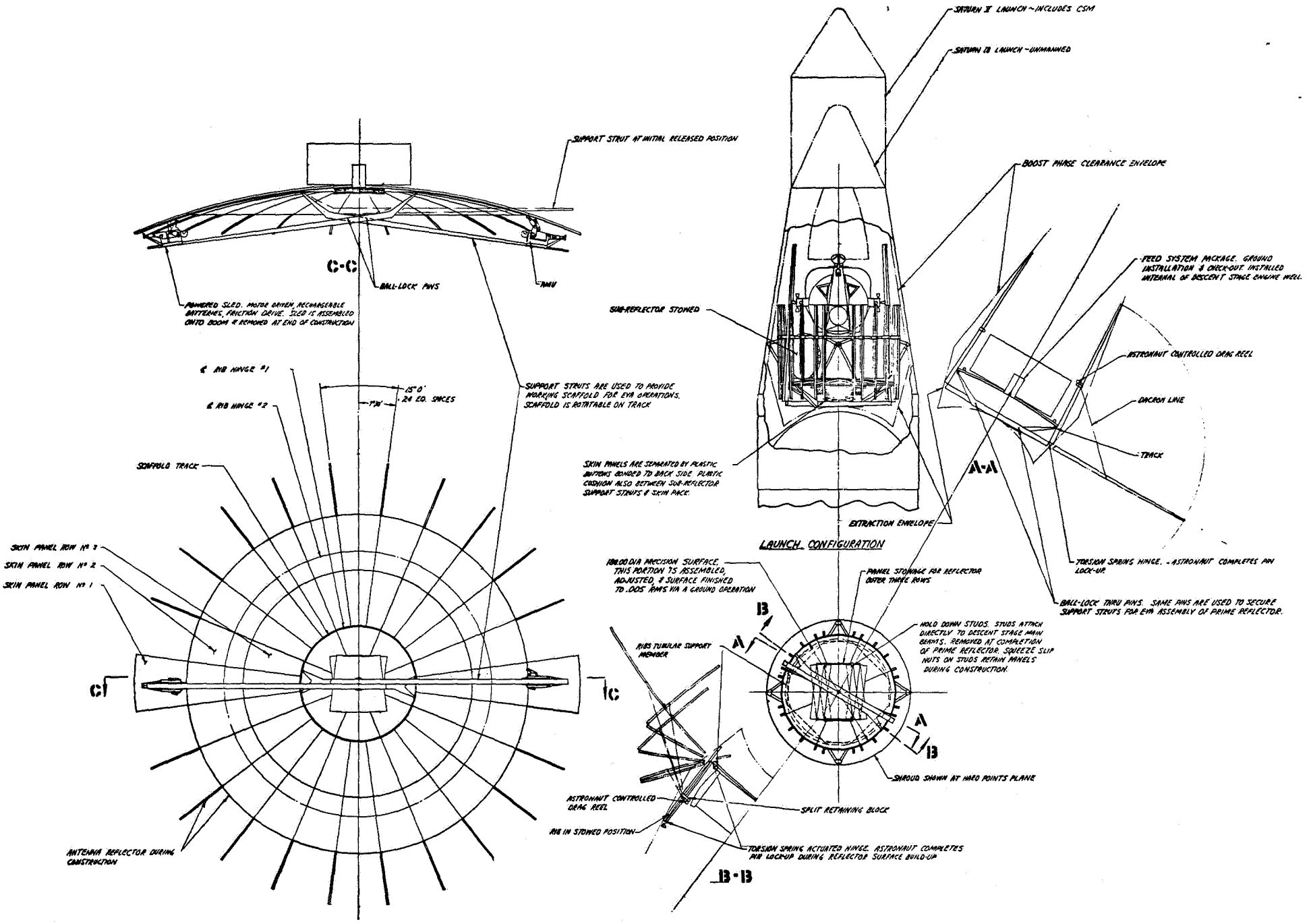



Figure 5-4. Concept 1, 60-ft Parabolic Antenna (NASA)

The initial deployment tasks are simple mechanical actions easily controlled by the astronaut. The more difficult tasks are downstream; progressive crew learning aids in task execution.

Externally powered mechanisms are not used for deployment. Stored energy of torsion springs at the strut and rib hinges is the only motive power beyond the astronauts' action. Much of the assembly task is repetitive. The deployment and assembly can be stopped at practically any point in the event of an emergency.

Panel attachment is a sensitive design problem. The originally proposed magnet system is not adequate. However, no other method has been proposed. The numerous panels with multiple fasteners suggests a need for a semi-automatic system tailored to the crew members control — perhaps a special tool. The fastening system must be reversible to the extent that ground adjustments are not altered. As antenna erection progresses, the work area becomes progressively farther away and more obscure from the control station.

As stated in the NASA document, the design objective of 0.007 in. RMS is not possible. Figure 5-5 compares accuracies of typical antennas.

Values plotted for the several concepts are the RMS surface deviations from the perfect parabola due to characteristic sensitivity of the concepts, to the limitations of manufacturing accuracy, and to mechanical clearances and distorting effects of solar radiation in space. (Strains due to orbital attitude control system forces are not included.) The graph carries three parametric curves: 1) $\sigma/D = 10^{-3}$ is considered an accuracy relatively easy to achieve (e.g., 0.60 in. RMS over a 50-ft-dia. aperture); 2) $\sigma/D = 10^{-4}$ is considered relatively difficult (e.g., 0.060 in. RMS over 50-ft-dia. aperture); 3) $\sigma/D = 10^{-5}$ is beyond the precision capability of present manufacturing practices, even for ground-based antenna structures (e.g., 0.006 in. RMS over a 50-ft aperture).

Of all the space erectable antenna concepts familiar to industry, only the sunflower petaline configuration betters the $\sigma/D = 10^{-3}$ parameter. Further, the graph shows that the concept presented here cannot achieve the 0.007 RMS except perhaps in the inner 15-ft fixed section. The predicted accuracy for this diameter is between 0.20 and 0.30 in. RMS.

A comparison between tolerance requirements and manufacturing cost as a function of antenna diameter is useful since an economic factor will surely be included during the evaluation of the concepts for a representative experiment. Figure 5-6 is indicative of high costs when both large antenna and close tolerances are required. For example, a 50-ft dish with ± 0.010 in. tolerance is 14 times more expensive than a 10-ft dish with ± 0.001 in. tolerance. The information is based on the relative cost required to build antenna tooling and basic forms.

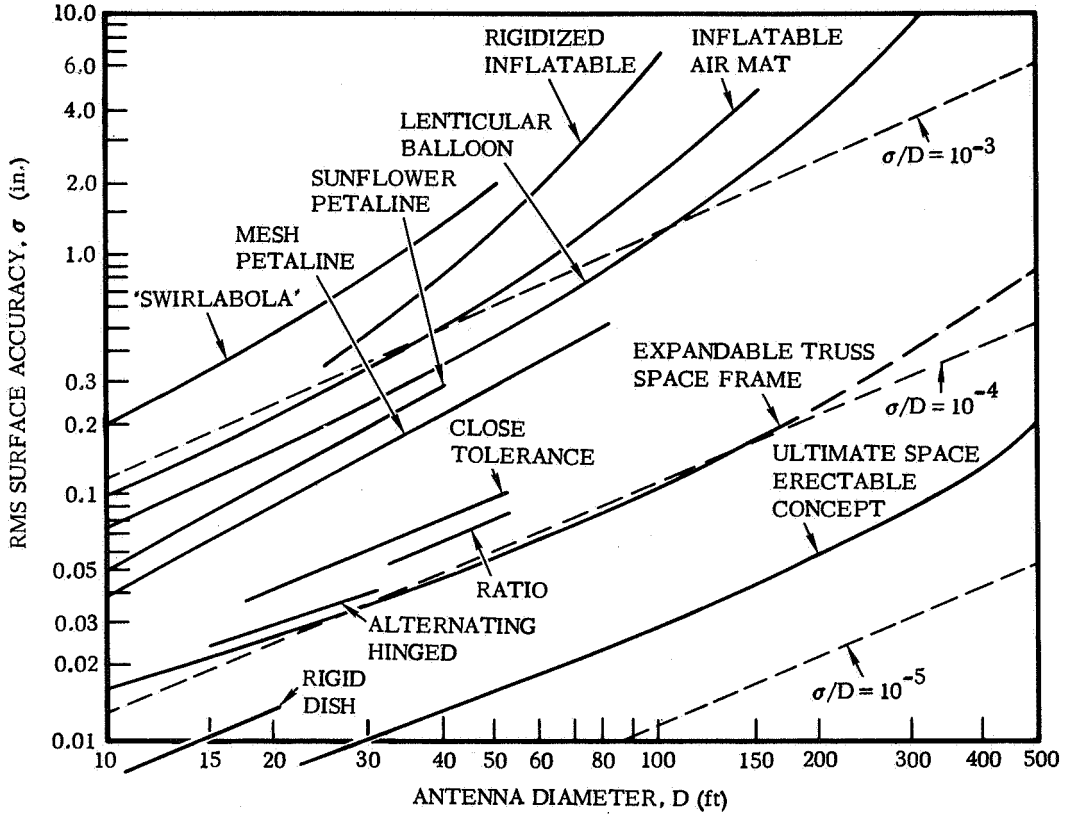



Figure 5-5. Reflective Surface Shape Accuracy Capability of Typical Concepts in the Space Environment

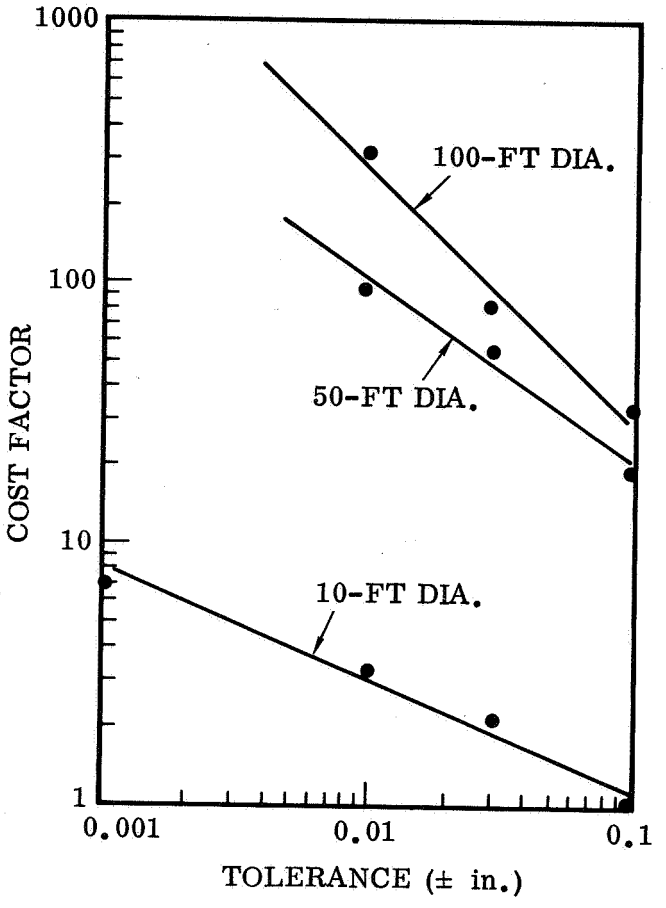



Figure 5-6. Tolerance/CostFactor Comparison

Sub-reflector alignment is a two-phase process. The first is performed by the crew via EVA. The second phase is performed during the RF check whereas the sub-reflector is given vernier adjustments by use of remotely controlled power screws.

The sub-reflector assembly consists of a reflector surface suspended within a circular frame having internal cross beams. During the launch and assembly phases, the hyperbolic surface and circular frame are clamped together. The mechanical adjustment is performed, then the surface is freed from peripheral attachment to the frame.

The final adjustment is remotely controlled and powered because of the frequency and random pattern of RF checkout to optimize the RF qualities. These excursions of the sub-reflector will include ± 1.00 in. in the three orthogonal axes in addition to an angular adjustment of 2° relative to the spacecraft longitudinal centerline. This will be performed by remote stepping motors and indicated by position encoders.

The powered fine-increment system provided for the RF alignment allows the first phase mounting ring adjustment to be a more simplified EVA task. Small sighting scopes, ground installed and calibrated, can be used to boresight onto index targets located on the periphery of the precision inner 15-ft diameter. An additional clamp frame above the sub-reflector adds rigidity. Finally, the strut lower pivots are locked. In event of future EVA check, the equipment is already emplaced for rapid and convenient coarse adjustments.

Other alignment methods are certainly possible and must be considered. The precision inner portion of the dish provides a good reference for relative positioning of the sub-reflector for whatever alignment method is used. The fixed attachment to the rigid descent stage also provides a rigid base to enhance alignment.

An analysis of the mechanical/structural aspects during operation does not reveal any particular point of controversy. There is, however, a basic feature that can be assessed as a reflection upon the mechanical/structural configuration. The rigid attachment of the antenna to the descent stage contributes to the construction feasibility of the concept, but this is not without a penalty — the total spacecraft is subject to the antenna steering mode.

Drag penalties make consideration of an overfly mission impractical for altitudes less than 200 n.mi. unless some overriding requirement exists. To establish a near upper limit on control propellant as a function of antenna size and weight, an equatorial orbit at 200 n.mi. was considered. This orbit permits acquisition and tracking of the ground station once per orbit. When not tracking, the station is put on an earth-oriented attitude hold. This gives a measure of the influence of antenna size on control propellant usage (Figure 5-7).

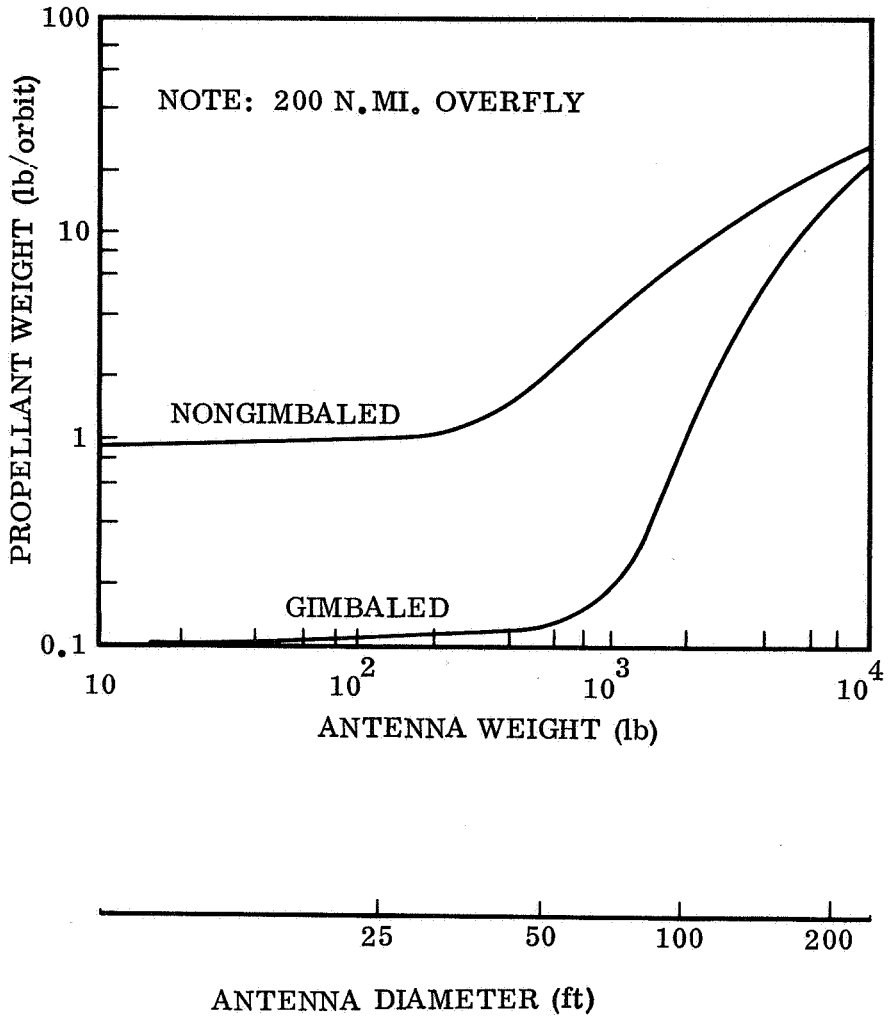



Figure 5-7. Comparison of Propellant Weight Required for Gimbaled and Nongimbaled Antennas of Various Sizes and Weights

Two modes of tracking were considered (Figure 5-7): 1) with the antenna rigidly fixed to the laboratory, and 2) with the antenna gimbaled. The second mode has the advantage of reducing the laboratory contribution to the moment of inertia to be slewed. The curve flattening at lower antenna weights occurs because a small antenna makes practically no contribution to total area or moment of inertia; hence, the required propellant mostly slews or maintains orientation of the laboratory. At the other extreme, the antenna mass properties dwarf those of the laboratory and the gimbaled mode savings shrink toward zero.

Materials. On the basis of physical properties alone, beryllium would seem to be unquestionably the optimum material for consideration (see Table 5-2). The properties listed in the table are those most significant in maintaining a minimum deviation reflector surface.

The original design quotes a 50% perforation of the prime reflector beryllium sheets. The wavelength region dictates hole sizes not greater then 3/16-in. diameter. Hole distribution at this diameter will be 18/in.$^2$; an average panel will contain some 95,000 holes and the entire antenna outer three rows will have 6,800,000 holes. A mesh material seems in order — with a corresponding loss of surface accuracy.

The simple shapes and mild forming should not present undue manufacturing difficulties although manufacturing development costs must be an expected adjunct to use of this material.

5.2.1.2 Antenna Performance. The use of this concept for communications or submillimeter radio astronomy applications depends primarily on the dimensional tolerances that can be maintained in the manufacture, assembly, and employment of the structure.

Beamwidth of the antenna as a function of frequency is shown in Figure 5-8.

5.2.1.3 Electrical Power. Power requirements for the large erectable paraboloid will be for 1) scientific electronics, 2) communications experiments, 3) data and communications. Requirements will significantly exceed those for the more passive "listening" antennas or radio telescopes. Power requirements are shown in Table 5-3.

Power requirements for the communications experiments may vary from a few watts to as high as 25 kw depending upon whether signals are beamed to a large central distribution antenna for subsequent distribution to home receivers or direct to home receivers.

5.2.1.4 Dynamics Attitude Control. Two different configurations were considered in the dynamics/ACS analysis. In one configuration, only the lunar excursion module was attached to the antenna, and for the other, the CSM was coupled to the LEM. Moments of inertia, gravity-gradient torques, and control-torque requirements differ

Table 5-2. Antenna Materials Evaluation

| MATERIAL DESIGNATION | PROPERTY | | | | | | |
|---|---|---|---|---|---|---|---|
| | $\rho$ ($lb/in.^3$) | E ($psi \times 10^6$) | K ($Btu/hr\text{-}ft^2\text{-}°F$) | $\alpha$ ($in./in.\text{-}°F \times 10^6$) | $E/\rho$ ($in. \times 10^6$) | $K/\alpha$ ($Btu/hr\text{-}ft^2 \times 10^6$) | MATERIAL INDEX |
| Aluminum Alloys | | | | | | | |
| 2024-T4 | 0.100 | 10.6 | 70 | 12.6 | 106. | 5.55 | 588 |
| 7075-T6 | 0.101 | 10.3 | 90 | 12.9 | 102. | 6.97 | 725 |
| Magnesium Alloy | | | | | | | |
| AZ 31 B | 0.064 | 6.5 | 56 | 14 | 101.5 | 4.0 | 405 |
| Mag-Thorium | | | | | | | |
| HK 31 A | 0.065 | 6.5 | 61 | 14 | 100. | 4.35 | 435 |
| Mag-Lithium | | | | | | | |
| LA 141 | 0.049 | 6.0 | 25 | 21.8 | 122.5 | 1.148 | 124 |
| Beryllium | 0.067 | 42.0 | 87 | 6.4 | 627. | 13.6 | 8520 |
| Lockalloy BE 36 AL | 0.075 | 32.0 | 123 | 9.2 | 426. | 13.4 | 5700 |
| Titanium 6Al-4V | 0.160 | 16.0 | 3.8 | 4.8 | 100. | 0.792 | 79.2 |

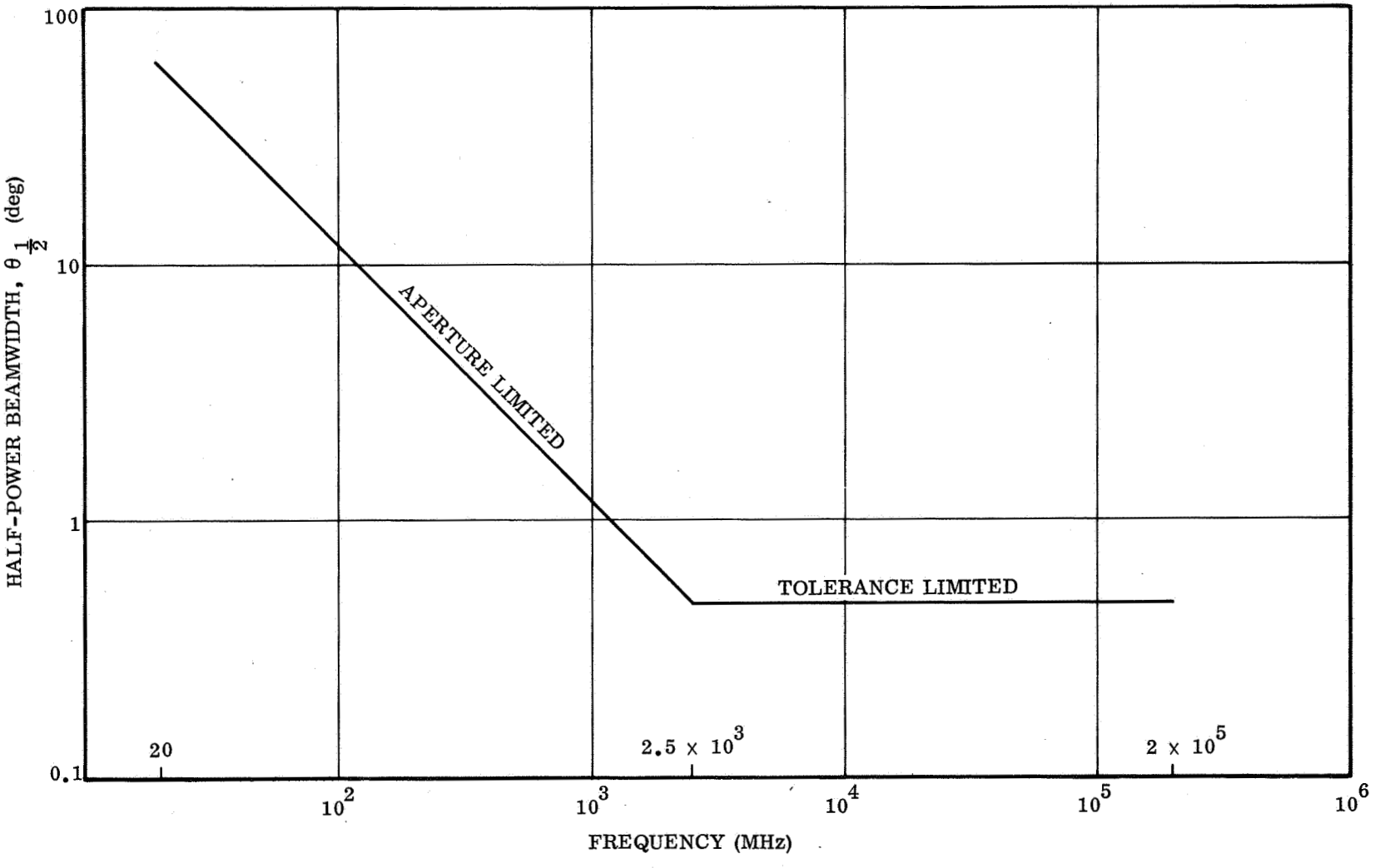



Figure 5-8. Estimated Beamwidth of the 60-Ft-Dia. Paraboloidal Antenna Concept 1

Table 5-3. Estimated Power Requirements for Large Erectable Antenna Experiment

| | | POWER (watts) |
|---|---|---|
| Scientific Electronics | | 50 |
| Broadcast Electronics | Distribution Center | 600 |
| | Direct to Home | $2.5 \times 10^4$ |
| Date and Communications | | 50 |

appreciably for these two different configurations. Rigid-body attitude control for both configurations does not appear to present any problems that would affect the feasibility of the experiment. However, some preliminary analysis of the elastic response of the antenna, using computer program No. 3684, developed by Convair for the dynamic analysis of three-dimensional structures, indicates that a torque impulse of 24 ft-lb-sec will cause a maximum surface deflection of approximately 0.005 in. and an average surface deflection on the order of 0.002 in. Thus, for a 200 n.mi. overfly mission, where control torque requirements are quite high and the minimum torque impulse may be relatively large, the elastic deformation of the antenna surface might exceed the tolerance of 0.007 in. For less demanding missions (such as maintaining a fixed inertial position with a synchronous orbit), it should be possible to keep the elastic deformation of the antenna surface within an acceptable range.

Table 5-4 summarizes the attitude control requirements imposed by environmental torques.

Table 5-4. Attitude Control Propellant Requirements

| CONFIGURATION | ALTITUDE (n.mi.) | AVERAGE TORQUE (ft-lb) | YEARLY CHG. IN MOMENTUM (ft-lb-sec) | JET IMPULSE PER YEAR (lb-sec) | PROP WT. PER YR. (lb) |
|---|---|---|---|---|---|
| Without CSM | 200 | 0.29 | $9.15 \times 10^6$ | $6.53 \times 10^5$ | 2180 |
| With CSM | 200 | 0.60 | $1.89 \times 10^7$ | $1.35 \times 10^6$ | 4500 |
| Without CSM | Sync. | 0.0015 | $4.73 \times 10^4$ | $3.38 \times 10^3$ | 11.3 |
| With CSM | Sync. | 0.0029 | $9.15 \times 10^4$ | $6.53 \times 10^3$ | 21.8 |

It should be noted that the propellant requirements listed do not include the effect of limit cycles on propellant consumption. However, these results do indicate that long-duration missions at altitudes as low as 200 n.mi. are expensive in terms of fuel consumption, which is due to environmental torques.

At higher orbital altitudes, propellant consumption decreases considerably. The Apollo RCS should be capable of providing adequate control if the pointing accuracy requirements are not too stringent (on the order of a few tenths of a degree). For intermediate orbital altitudes, the use of momentum storage devices should be investigated.

5.2.1.5 Stress Analysis. Examination of the structural environment specified in Appendix B indicates that only thermal effects, loads due to astronaut impact, and loads due to post-mission docking of the CSM to the experiment warranted examination.

The maximum temperature difference across an uninsulated rib is predicted to be 1.35. The maximum deviation from the original shape which occurs at the tip, is 0.166 in. This is much greater than the specified allowable deviation of 0.007 in., but may be of the same order of magnitude as distortions due to manufacturing tolerance or vibration. If the ribs are insulated to eliminate this effect, the panel surface temperature will be markedly higher than the tube temperature. Considering the surface to be equivalent to a spherical surface of approximately 100-ft radius and assuming a temperature differential of 100° F, the radial growth of the free surface is 0.77 in. This is a measure of the surface roughness when the panel is constrained to the rib structure by magnets.

5.2.1.6 Weights. A weight of approximately 2,450 lb has been calculated from the data available and the following assumptions:

a. Fiberglass insulation, 0.5 in. thick, is used around rib tubes.

b. Magnets are used at equivalent of 1 in. out of every 4 in. around each panel.

c. Steel strips are used around the entire periphery of reflector panels.

d. Focal point reflector weight is 42 lb.

Inertias in slug-ft$^2$ are:

| | |
|---|---|
| Pitch | 50,000 |
| Roll | 40,000 |
| Yaw | 50,000 |

5.2.1.7 Crew and Equipment Functions. Gross crew and equipment functions have been analyzed for Concept 1 and are indicated in the functional flow diagram, Figure 5-9. A time line analysis (Table 5-5) has also been made to provide an estimate of the length of time required to erect an operational antenna of this type. See Appendix D for basic assumptions.

Astronaut assistance is extensive and critical to assembly, yet the design has no special provisions that must accompany such activity. For instance, the extension of the tubular ribs require the extension of three joints and their lock-up by insertion of a pin. From the design it must be presumed this is done in a free-space maneuver by the astronaut. The only available structure to which he can obtain leverage or restraint is the neighboring ribs. The motion-damping system to counteract the forces of an active astronaut amongst this web of extending ribs could exceed the complexities of a more automated deployment method.

Installation of the reflector panels again presents the same problems associated with astronaut assist as noted above. The magnetic panel attachment presents a very difficult crew task, which may seem contrary to a first observation. The mounting of these panels requires precise approach to the mating magnets in which the panel must essentially be equidistant from each magnet just prior to contact. After contact it is doubtful that corrections can be easily made because of the grip force the magnets possess. Due to the very thin component elements, local prying can likely cause material failure. The surface accuracy will suffer and since progressive building is required, an early misplaced panel could disrupt the remainder of the panel assembly. To this may be added the aggravating problem of the panels being attracted to improper magnet strips during removal from stowage, transit, and positioning maneuvers. If one has ever worked with multiple magnetic catches or contact cement on earth, he will appreciate the pre-positioning difficulties.

The 0.125-diameter guide pins are considered inconsistent with the manipulative sensitivity associated with astronaut maneuvering of large panels.

The support equipment described in Table 5-6 pertains to the concept as shown in Figure 5-4. Astronaut-peculiar items such as tethers are not included.

5.2.2 Concept 4 — Lenticular Satellite (Goodyear Report GER 11893). The 267-ft-diameter lenticular passive communication satellite features an inflatable lens with a radius of curvature of 200 ft. See Figure 5-10. In addition to the lens, the satelltie system consists of two tripod booms, solar sails, a beryllium ring, and a torus, which are all packaged in a 50-in.-dia. canister during launch; the total experiment weight is 1250 lb.

The lenticular lens reflecting surface is fabricated of 0.5-mil photolyzable film cast on a 1-mil copper-wire plain-weave mesh (21 wires per inch); the other half of the lens is photolyzable film to permit initial pressurization. The lenticular lens and

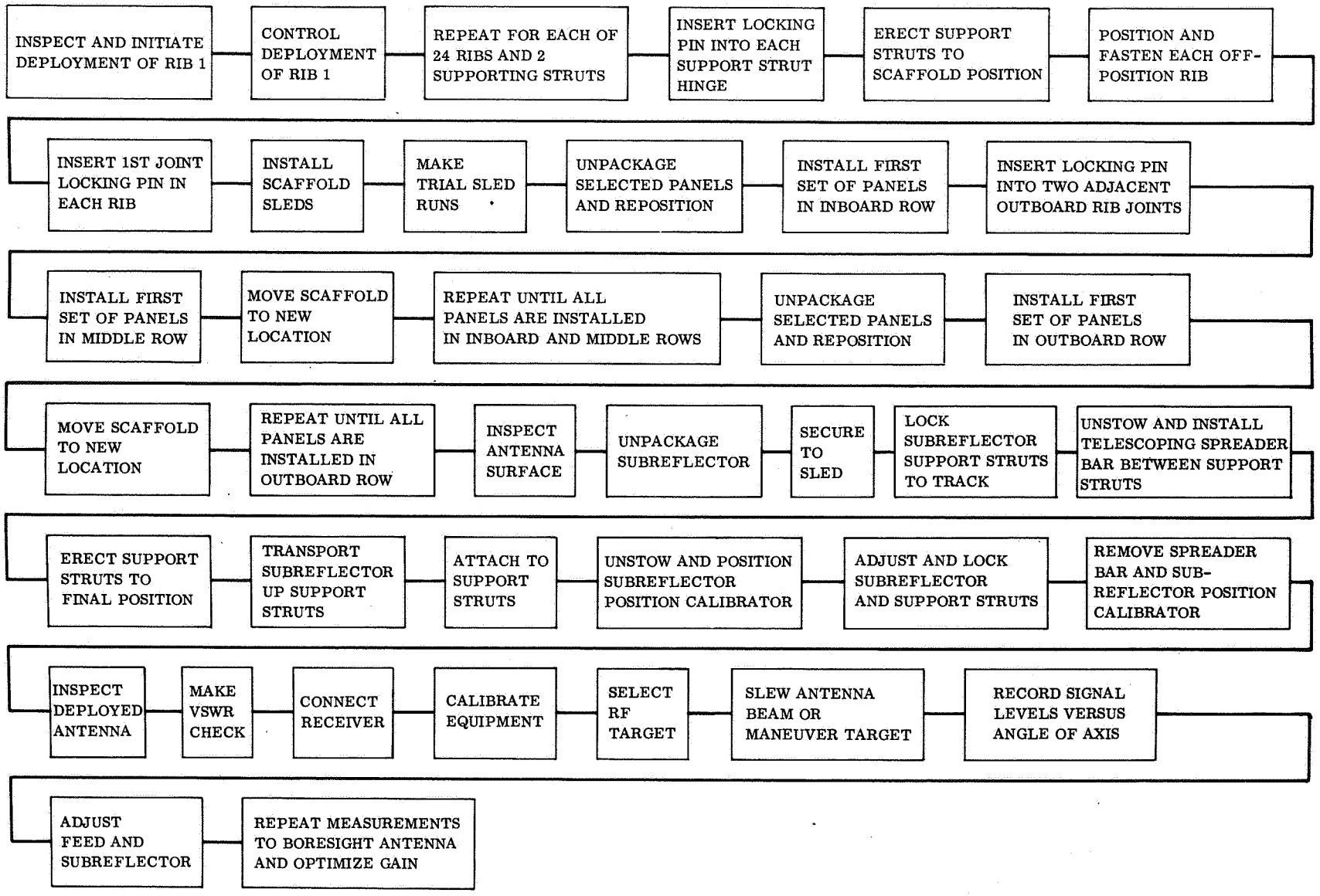



Figure 5-9. Functional Flow Diagram for the Erection of a 60-Ft Paraboloidal Antenna (P&VE Lab, MSFC)

Table 5-5. Time Line Analysis for the Erection of a 60-Ft Paraboloidal Antenna (P&VE Lab, MSFC)

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 1 | Perform pre-EVA | 60 | 60 | | Identical tasks are performed by Astronaut 2 for the opposite side of the antenna |
| 2 | Exit LM | 4 | 64 | | |
| 3 | Maneuver to Rib 1 | 1 | 65 | 1 | |
| 4 | Anchor to worksite | 2 | 67 | 3 | |
| 5 | Release the rib retaining jaw clamp | 1 | 68 | 4 | |
| 6 | Control the deployment of Rib 1 | 1 | 69 | 5 | |
| 7 | Release worksite anchors | 1 | 70 | 6 | |
| 8-67 | Repeat events 3-7 for each of the remaining 11 ribs and 1 support strut on one side of antenna package | 72 | 142 | 78 | |
| 68 | Maneuver to support strut hinge on concave face of antenna | 10 | 152 | 88 | |
| 69 | Anchor to worksite | 2 | 154 | 90 | |
| 70 | Insert locking pin into support strut hinge | 5 | 159 | 95 | |
| 71 | Release worksite anchors | 1 | 160 | 96 | |
| 72 | Maneuver to antenna center | 1 | 161 | 97 | |
| 73 | Anchor to worksite | 2 | 163 | 99 | |
| 74 | Remove support strut position locking bolts | 5 | 168 | 104 | Astronaut 2 assists Astronaut 1 in the following tasks |

Table 5-5. Time Line Analysis for the Erection of a 60-Ft Paraboloidal Antenna (P&VE Lab, MSFC), Contd

| EVENT | ASTRONAUT 1 | TIME (min) | | | ASTRONAUT 2 |
|---|---|---|---|---|---|
| | | EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | |
| 75 | Erect support struts to scaffold position | 5 | 173 | 109 | |
| 76 | Reinsert locking bolts | 5 | 178 | 114 | |
| 77 | Install scaffold position control system | 15 | 193 | 129 | |
| 78 | Unpackage scaffold sled | 10 | 203 | 139 | |
| 79 | Secure sled to scaffold | 10 | 213 | 149 | |
| 80 | Release worksite anchors | 1 | 214 | 150 | |
| 81 | Maneuver to first off-position rib | 1 | 215 | 151 | Astronaut 2 again performs identical tasks on the opposite side of the antenna |
| 82 | Anchor to worksite | 2 | 217 | 153 | |
| 83 | Position the rib | 3 | 220 | 156 | |
| 84 | Fasten the rib | 5 | 225 | 161 | |
| 85 | Install the first joint locking pin | 2 | 227 | 163 | |
| 86 | Release worksite anchors | 1 | 228 | 164 | |
| 87-136 | Repeat events 81-86 for the remaining 11 ribs, deleting events 83 and 84 when not locking off-position ribs | 90 | 318 | 254 | |
| 137 | Return to LM | 1 | 319 | 255 | |
| 138 | Enter spacecraft | 4 | 323 | | |
| 139 | Perform post-EVA | 30 | 353 | | End of Day 1 |
| 140 | Perform pre-EVA | 60 | 413 | | |
| 141 | Exit LM | 4 | 417 | | |
| 142 | Don maneuvering device | 15 | 432 | 270 | |
| 143 | Maneuver to concave face of antenna | 5 | 437 | 275 | |

Table 5-5. Time Line Analysis for the Erection of a 60-Ft Paraboloidal Antenna (P&VE Lab, MSFC), Contd

| EVENT | ASTRONAUT 1 | TIME (min) | | | ASTRONAUT 2 |
|---|---|---|---|---|---|
| | | EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | |
| 144 | Anchor to worksite | 2 | 439 | 277 | |
| 145 | Unpackage selected antenna surface panels | 5 | 444 | 283 | |
| 146 | Position and secure selected panels for access from scaffold | 5 | 449 | 288 | |
| 147 | Release worksite anchors | 1 | 450 | 289 | |
| 148 | Maneuver to scaffold sled | 1 | 451 | 290 | |
| 149 | Anchor to sled | 2 | 453 | 292 | |
| 150 | Make sled trial runs | 2 | 455 | 294 | |
| 151 | Maneuver scaffold to first position | 1 | 456 | 295 | |
| 152 | Install first panel in inboard row | 10 | 466 | 305 | |
| 153 | Pin two adjacent outboard rib joints | 5 | 471 | 310 | |
| 154 | Install first panel in middle row | 10 | 481 | 320 | |
| 155-178 | Repeat events 151-154 for the next six panels in the first two rows | 156 | 637 | 476 | |
| 179 | Release sled anchors | 1 | 638 | 477 | |
| 180 | Return to LM | 5 | 643 | 482 | |
| 181 | Doff maneuvering device | 10 | 653 | 492 | |
| 182 | Enter spacecraft | 4 | 657 | | |
| 183 | Perform post-EVA | 30 | 687 | | End of Day 2 |
| 184-194 | Repeat events 140-150 | 102 | 789 | 530 | |

Table 5-5. Time Line Analysis for the Erection of a 60-Ft Paraboloidal Antenna (P&VE Lab, MSFC), Contd

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 195-214 | Repeat events 151-154 for the next five panels in the first two rows | 130 | 919 | 660 | |
| 215 | Manuever scaffold to new position | 1 | 920 | 661 | |
| 216 | Install first panel in outboard row | 10 | 930 | 671 | |
| 217-226 | Repeat events 215-216 for next five panels in outboard row | 55 | 985 | 726 | |
| 227-231 | Repeat events 179-183 | 50 | 1035 | 742 | End of Day 3 |
| 232-242 | Repeat events 140-150 | 102 | 1137 | 780 | |
| 243-254 | Repeat events 215-216 for remaining six panels in outboard row | 66 | 1203 | 846 | |
| 255 | Inspect total surface area | 45 | 1248 | 891 | |
| 256 | Disengage sled anchors | 1 | 1249 | 892 | |
| 257 | Maneuver to sub-reflector storage area | 5 | 1254 | 897 | Astronauts working together |
| 258 | Anchor to worksite | 2 | 1256 | 899 | |
| 259 | Unpackage sub-reflector | 10 | 1266 | 909 | |
| 260 | Disengage worksite anchors | 1 | 1267 | 910 | |
| 261 | Manuever to concave face of antenna | 10 | 1277 | 920 | |
| 262 | Anchor to worksite | 2 | 1279 | 922 | |
| 263 | Secure sub-reflector | 5 | 1284 | 927 | |

Table 5-5. Time Line Analysis for the Erection of a 60-Ft Paraboloidal Antenna (P&VE Lab, MSFC), Contd

| EVENT | ASTRONAUT 1 | TIME (min) | | | ASTRONAUT 2 |
|---|---|---|---|---|---|
| | | EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | |
| 264 | Lock scaffold (sub-reflector support struts) in final track positions | 5 | 1289 | 932 | |
| 265 | Remove sleds from support struts | 5 | 1294 | 937 | |
| 266 | Unpackage and install temporary spreader bar between support struts | 8 | 1302 | 945 | |
| 267 | Remove locking bolts from support struts | 5 | 1307 | 950 | |
| 268 | Erect support struts to approximate true position | 5 | 1312 | 955 | |
| 269 | Reinstall one sled to one support strut | 10 | 1322 | 965 | |
| 270 | Secure sub-reflector to sled | 5 | 1327 | 970 | |
| 271 | Disengage worksite anchors | 1 | 1328 | 971 | |
| 272-275 | Repeat events 180-183 | 49 | 1377 | 986 | End of Day 4 |
| 276-280 | Repeat events 140-144 | 86 | 1463 | 1008 | |
| 281 | Maneuver sled and sub-reflector to just below attach points | 4 | 1467 | 1012 | |
| 282 | Remove sub-reflector and attach to support struts | 30 | 1497 | 1042 | |
| 283 | Return sled to support strut base and detach it along with other related equipment | 14 | 1511 | 1056 | |
| 284 | Release worksite anchors | 1 | 1512 | 1057 | |

Table 5-5. Time Line Analysis for the Erection of a 60-Ft Paraboloidal Antenna (P&VE Lab, MSFC), Contd

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 285 | Maneuver to LM | 10 | 1522 | 1067 | |
| 286 | Anchor to worksite | 2 | 1524 | 1069 | |
| 287 | Secure above equipment to LM | 10 | 1534 | 1079 | |
| 288 | Unpackage calibrator | 5 | 1539 | 1084 | |
| 289 | Release worksite anchors | 1 | 1540 | 1085 | |
| 290 | Transport calibrator to concave face of antenna | 10 | 1550 | 1095 | |
| 291 | Anchor to worksite | 2 | 1552 | 1097 | |
| 292 | Attach calibrator to the sub-reflector | 20 | 1572 | 1117 | |
| 293 | Adjust sub-reflector and lock in place | 45 | 1617 | 1162 | |
| 294 | Remove spreader bar and calibrator | 10 | 1627 | 1172 | |
| 295 | Disengage worksite anchors | 1 | 1628 | 1173 | |
| 296 | Perform final antenna inspection | 20 | 1648 | 1193 | |
| 297-300 | Repeat events 180-183 | 49 | 1697 | 1208 | |
| 301 | Make VSWR check | 60 | 1757 | | |
| 302 | Connect receiver | 20 | 1777 | | |
| 303 | Calibrate equipment | 30 | 1807 | | |
| 304 | Select RF target | 10 | 1817 | | |
| 305 | Slew antenna or maneuver target while recording signal strength | 60 | 1877 | | |
| 306 | Adjust feed and sub-reflector and repeat measurements to boresight antenna and optimize gain | 480 | 2357 | | |

Table 5-6. Experiment Support Equipment Requirements and Design

| EQUIPMENT DESCRIPTION | DESIGN REQUIREMENT |
|---|---|
| Sled, Scaffold | 1. Interface with astronaut for attachment<br>2. Interface with sub-reflector strut for mounting. Suitable for astronaut landing<br>3. Powered, single lever control<br>4. Rechargeable battery power<br>5. Fourteen-day life<br>6. Controls easily accessible to astronaut by feel<br>7. Foot stirrups, girth band<br>8. Space environment |
| Floodlights | 1. Suitable for positioning by astronaut<br>2. Adaptable to scaffold mounting<br>3. Sufficient intensity that construction is not delayed because of poor lighting<br>4. Automatic on-off control plus astronaut control<br>5. Space environment |
| Panel Fastening Tool | 1. This is subject to the method of panel attachment which is not resolved<br>2. Space environment |
| Panel Retaining Fixture: needed to position several panels in ready position to reduce mounting/dismounting. | 1. Attaches to scaffold<br>2. Mechanical toggle clamp operable by one hand<br>3. Space environment |
| Sub-Reflector Alignment Equipment | 1. It is proposed that this equipment shall be built into the sub-reflector |

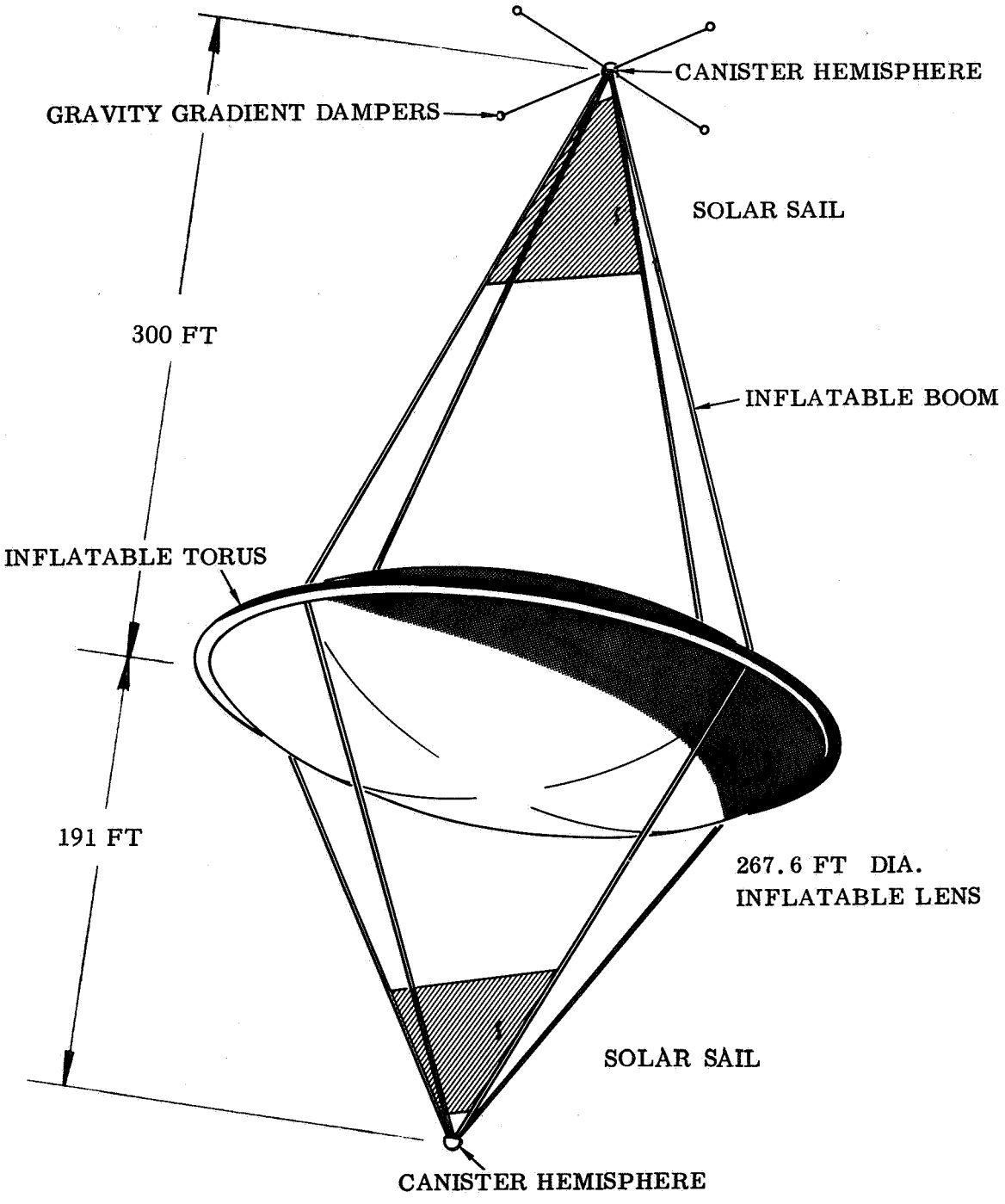



Figure 5-10. Concept 4, Passive Communication Lenticular Satellite (Goodyear)

beryllium rim are deployed with assistance of a torus. Two masses consisting of the canister halves, inflation system, damping system, and control and electronics elements are mounted on the tripods to provide gravity gradient stabilization.

Perforations in the lens and torus surfaces permit evacuation of entrapped air prior to deployment. After deployment, the gas pressure is permitted to decay by exhaust through surface perforations.

The inflatable lens radius of curvature can be held to within ±1% (±2 ft). Forty watts of electrical power are required for instrumentation telemetry and control.

Astronaut participation is restricted to visual checkout of the deployed antenna or minor repair with the aid of an AMU.

Passive reflector satellites are quite restricted in their application to space communications as compared with active relay satellites. The best application is at orbital altitudes well below synchronous altitude and at frequencies in the high VHF or low UHF ranges.

The considerable effort and data generated by the Goodyear Company and NASA agencies has established the feasibility of the concept.

Large expandable structures of the inflatable type employ materials that are very light weight, sensitive, and hence intentionally capable of being launched and deployed automatically without EVA assist.

5.2.3 Concept 9 — Flattenable Truss Parabolic Antenna (Ryan Report 29174-47). This structure is a parabolic antenna of 100-ft-dia. aperature proposed by Ryan Electronics and Space Systems. Applying a typical 0.2 in. RMS surface tolerance, the operational frequency range is 30 MHz to 3.7 GHz.

The featured structures are 24 tapered truss ribs that back up the reflector surface. Tubular compressible spring tape members constitute the top and bottom caps of each truss. These collapsible trusses are intended to be wrapped in the flat around a central drum into a small package. See Figure 5-11. The reflective surface is formed by aluminum honeycomb panels emplaced by the crew onto the deployed truss structure.

5.2.3.1 Structure and Mechanics. Presumably, the rotating, opposed rollers would be used to accomplish stowage of the trusses on the drum. As the two rollers traverse they apply force in a radial direction. As the truss members are rolled onto each other their relative platforms are progressively displaced. After the rollers pass any particular truss location the compressive force causing truss conformance to the drum is relieved. The truss does not remain flattened — it will attempt to expand. The only

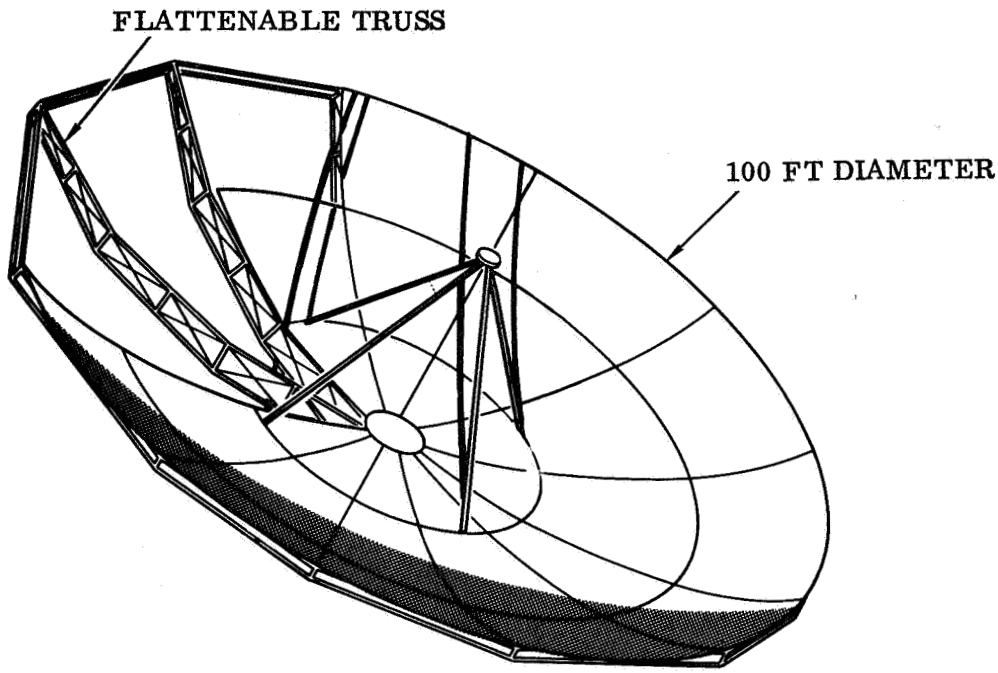



Figure 5-11. Concept 9, Flattenable Truss Parabolic Antenna (Ryan)

restraining media now are the stacked ribs. The strain force in the spring members will produce a following "wave", which is manifested in the truss members lifting off of the drum. See Figure 5-12.

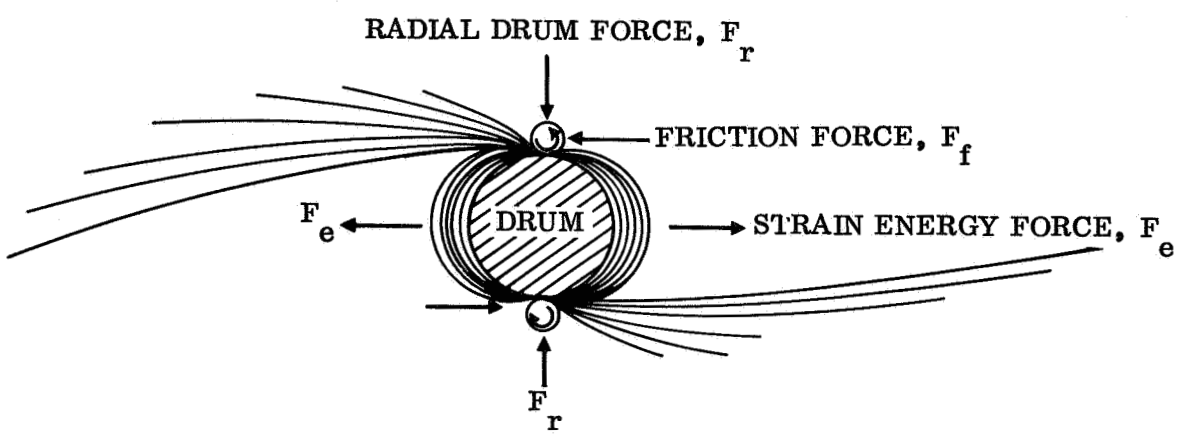



Figure 5-12. Stowing the Trusses

Capture of the rib tips must also be considered. A working model may be required to demonstrate feasibility.

5.2.4 Concept 10, 50-Ft Petal Parabolic Antenna (Ryan Report 29174-47). Ryan Electronics proposes a 50-ft parabolic reflector antenna that is deployed by unfolding 32 petal blades (See Figure 5-13).

It appears best suited for transmission of real-time TV to distribution centers and/or telephone communications. The operating frequency range is 80 MHz to 7.4 MHz, based upon an antenna diameter of 37 ft or less. The proposed 50-ft diameter is too large to stow aboard the manned version of either the Saturn IB or V vehicle and must be reduced to at least 37 ft.

The structural aspects of the antenna petal blades appear feasible. A problem that will likely require Astronaut EVA is the deployment of the associated equipment, such as electronics, ACS, and 780 $ft^2$ of solar cell panels, which are attached to the only hard point available - the parabolic vertex.

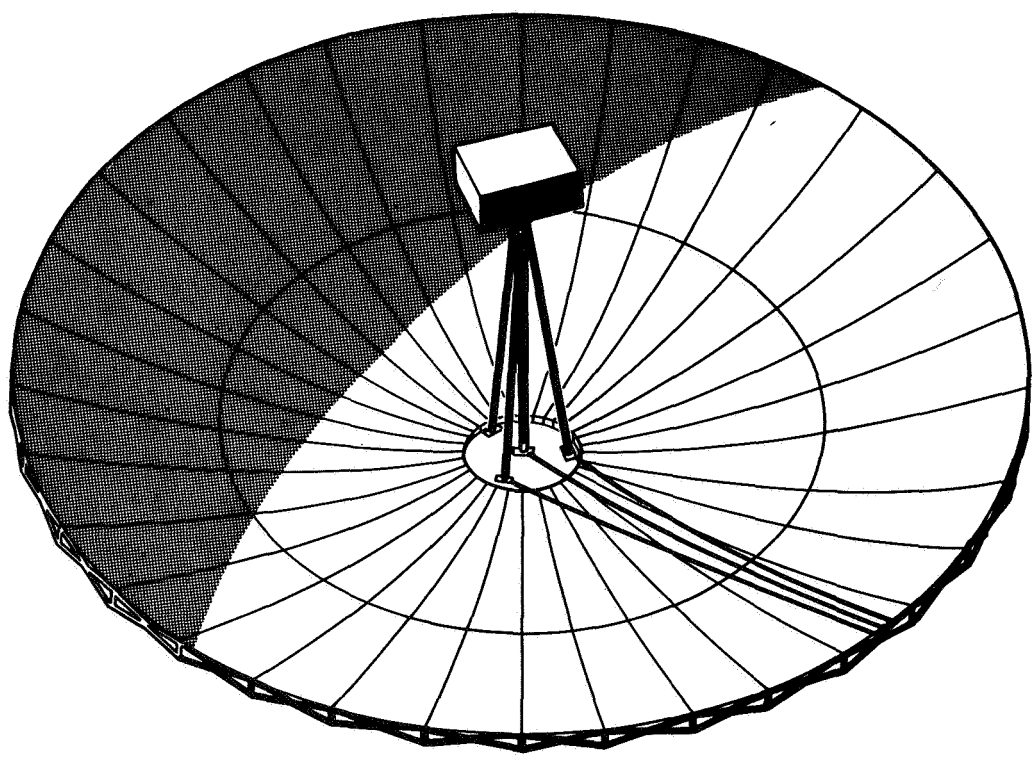


Figure 5-13. Fifty-Foot-Dia. Rigid Petal Parabolic Antenna (Ryan)

The petal structure, consisting of 1/4-in. aluminum honeycomb coupled with the reinforcement of a lazy tong, insures a fairly rigid structure with a possible surface tolerance of 0.1-in. RMS. Frequency coverage for the above configuration is 80 MHz to 7.4 GHz. Thermal distortions can reduce the upper limit to 4.5 GHz or more if not compensated for in the design of the honeycomb structure.

Limiting the antenna function to the user requirements of transmission of real-time TV to distribution centers and/or telephone communication allows the proposed diameter to be reduced to 20- to 30-ft, which may reduce slightly the deployment/structural problem.

5.2.4.1 Structure and Mechanics. The concept as proposed is considered feasible with regard to the mechanism of deployment since the proposed 50-ft-dia. dish is essentially a scaled up version of Ryans 10-ft working model.

Some distortional effects due to solar radiation are to be expected. For a parabolic surface with high reflective properties, the temperature gradient across a 1/4-in. aluminum honeycomb structure is very small but the distortional effects are significant, assuming an absorptivity ($\alpha$) of 0.2 for aluminum and an emissivity ($\epsilon$) of 0.05. For a calculated $\Delta T = 2.4°F$, the blade deflection along its length, where $\ell = 22$ ft, is 1.11 in. However, the lazy-tong mechanism at the periphery and at the mid-section apply enough restraint to prevent any deflection in this plane. The blade warpage across its width, where $\ell = 4$ ft, is 0.037 in. This warpage is relatively significant when compared with the possible fabrication tolerance of 0.1 in. or more.

Deployment is initiated by an electric motor operating through a non-reversible gear box that winds up a gang of reels. The reels in turn unlock the blades and deploy the antenna via the cables that pass through the lazy tongs at the blade tips. When the slack is taken up and the cables pulled tight, the motor is shut off. This phase of the deployment can be done without astronaut assistance.

Figure 5-14 shows most of the equipment stowed around the antenna, which allows the largest possible antenna diameter.

5.2.4.2 Antenna Performance. The 50-ft-dia. parabola could best meet the user requirements if reduced in diameter to accommodate the communications missions:

a. Real-time TV to distribution centers for total U. S. coverage.

b. Personal, global, telephone communications for higher portions of frequency range above 1.5 GHz.

The Ryan report gave only the basic structural member description. This did not include the reflecting surface parameters, such as RMS surface tolerance, necessary for an RF analysis.

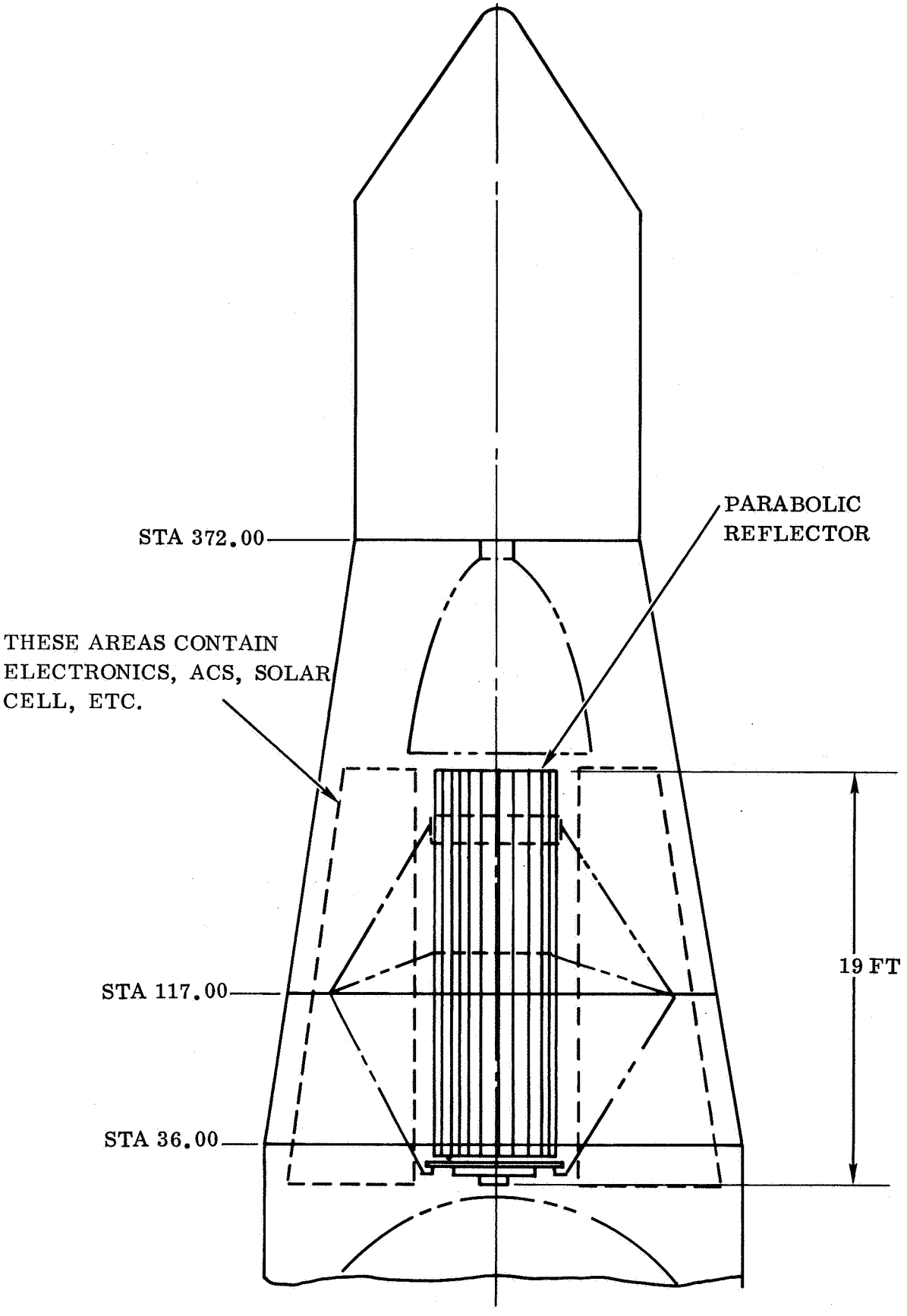



Figure 5-14. Saturn IB/V Manned Launch Configuration

The concept could be considered a "close tolerance" parabola, see Figure 5-5. A reinforced structure of a 50-ft-diameter normally has an RMS surface tolerance, $\sigma$, of approximately 0.1 in.; assuming zero thermal, dynamic, etc., effects.

The upper limit for the range of frequencies of operation is determined by the surface tolerance. This upper limit is established by determining the acceptable wave length, $\lambda = 1.6$ in. The upper frequency is then 7.4 GHz. The lower frequency is the wavelength that is 1/3 the antenna diameter, which is 16.7 ft for the 50 ft concept and 12.7 ft for the reduce diameter.

The lower frequencies are then 60 MHz and 80 MHz, respectively. Figure 5-15 gives the performance characteristics for the proposed 50-ft parabola.

5.2.4.3 Electrical Power. TV broadcast from orbit via local distribution centers reduces power required for transmission by at least an order of magnitude as compared to direct to home broadcast. The transmitted power required for 10 channels at full national coverage via distribution centers has been estimated at 600 watts. Assuming a transmission efficiency of 20%, this will result in a total input power to the TV transmitter of 3.0 kw. Added to this load would be requirements for telemetry and command systems for monitoring and remote operation of the satellite; this load is not expected to exceed 100 watts average. Other nominal electrical loads such as attitude pointing systems and other miscellaneous electronic loads are expected to amount to another 100 watts average, for a total of 3.2 kw.

5.2.4.4 Dynamics/Attitude Control. User requirements for this concept were assumed to be either a low- or high-orbit 37-ft-dia. antenna and low-orbit 50-ft-dia. antennas, to 10 years operating life, and ability to hold non-earth-surface oriented positions as well as earth-surface oriented positions with pointing accuracy tolerances of from 0.1° to 1.7°.

For purposes of computing propellant requirements, rim-mounted jets with thrusts of 0.1 lb and an Isp of 300 were assumed. Propellant consumption due to limit cycling were also computed for the range of pointing accuracies desired and were found to be negligible (less than 1 lb/year per axis). The average yearly propellant requirements for the pitch and yaw axes are listed in Table 5-7.

Although the tighter attitude tolerances do not require significantly more propellant, they will require more accurate attitude sensors. Horizon sensors are adequate for pointing accuracies on the order of 1°, but stellar devices are necessary for the tighter tolerances.

5.2.5 Concept 11 — Petal Parabola (TRW Report No. 5760.000). This antenna concept is suitable for earth communication from a synchronous orbit at a frequency of 2300 MHz with a gain of approximately 44 db. It is made up of 30 rigid petal segments

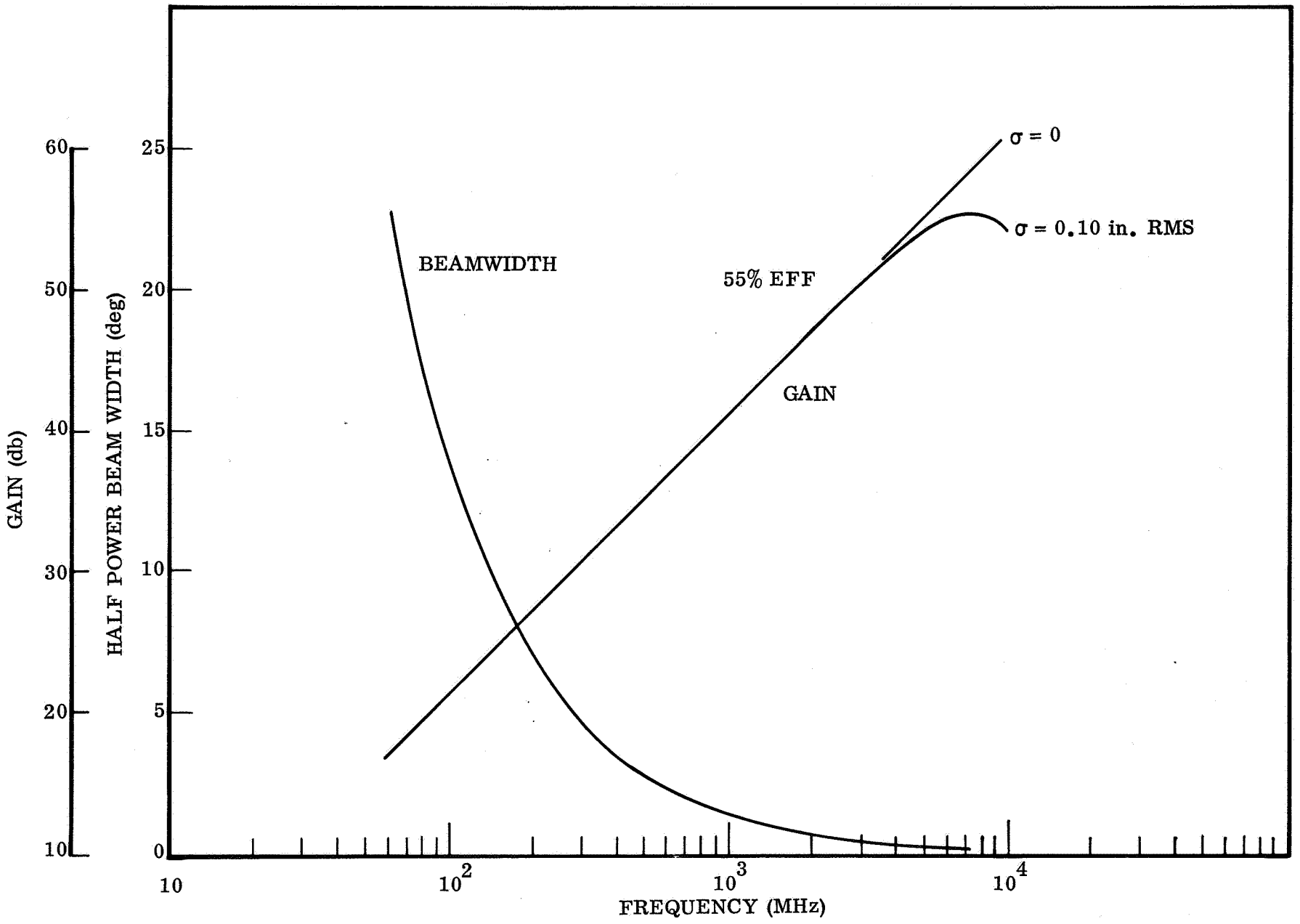



Figure 5-15. RF Performance for a 50-Ft Parabolic Antenna

Table 5-7. Average Yearly Propellant, Concept 10

| ANTENNA DIAMETER (ft) | ORBITAL ALTITUDE (n.mi.) | REQUIRED ANGULAR MOMENTUM (ft-lb-sec/yr) | TOTAL PROPELLANT WEIGHT (lb/yr) |
|---|---|---|---|
| 50 | 200 | $3.1 \times 10^6$ | 4100 |
| 37 | 200 | $1.3 \times 10^6$ | 2270 |
| 37 | Synch | $2.0 \times 10^4$ | 35 |

that are completely stowed during launch within a conventional fairing and subsequently deployed and locked in its paraboloidal configuration. See Figure 5-16.

The folding petal concept allows packaging of the 30-ft antenna and 325 $ft^2$ of solar panels within the limited volume, 9-ft dia. x 16.8 ft, available on the manned Saturn launch vehicles. See Figure 5-17.

The antenna is composed of adhesive-bonded aluminum-honeycomb-sandwich material in sectors around a fixed center dish. Each sector is hinged at a structural support ring attached to the spacecraft. In the stowed position, the segments are folded against the feed support,and hoops around the outside restrain the bundle during launch. At the time of deployment, the restraining hoops release and the springs actuate the deployment. When the open position is reached, locking devices connect adjacent sectors into the final paraboloidal configuration. Deployment is automatic, requiring no EVA.

The RMS height deviations from the ideal paraboloid are measured to predict electrical performance. The maximum tip deflection is 0.20 in. and the maximum random surface deviation is 0.05 in.

The 1,440 lb experiment includes antenna, RF equipment, electrical power supply and an attitude control system that provides for one year mission without resupply.

This concept appears to be structurally and mechanically feasible with good reliability. Initial petal deployment appears excellent. Final petal locking reliability is controlled by the individual mechanical operation of each locking mechanism and must be proof tested in the space environment.

The mechanical design concept and fabrication is similar to the 23-ft-dia. Sunflower solar collector successfully developed by TRW.

5.2.5.1 Structure and Mechanics. Sources of geometric inaccuracy in a specific antenna design must be controlled to within tolerances that insure proper RF performance.

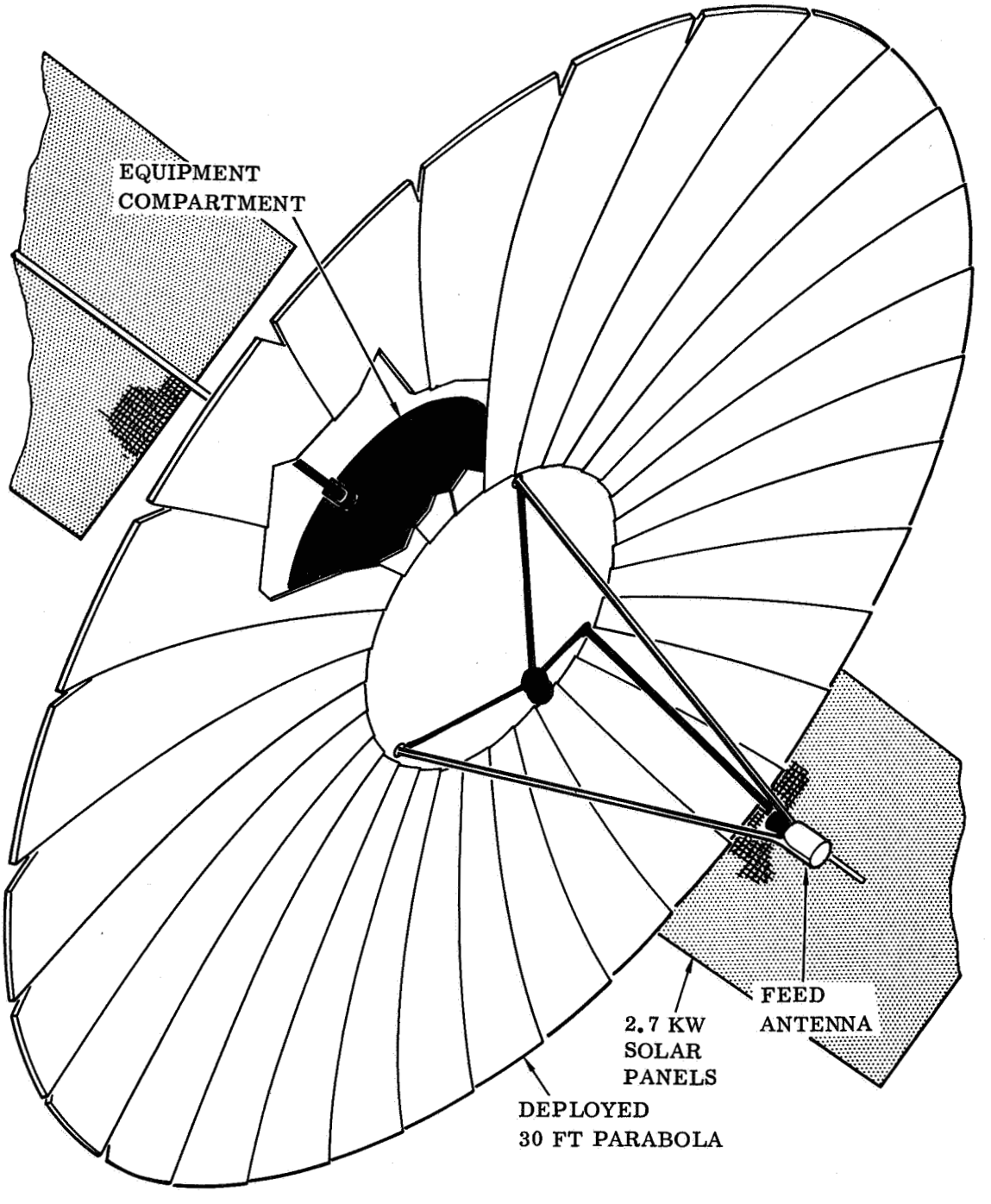



Figure 5-16. Deployed Sunflower Antenna

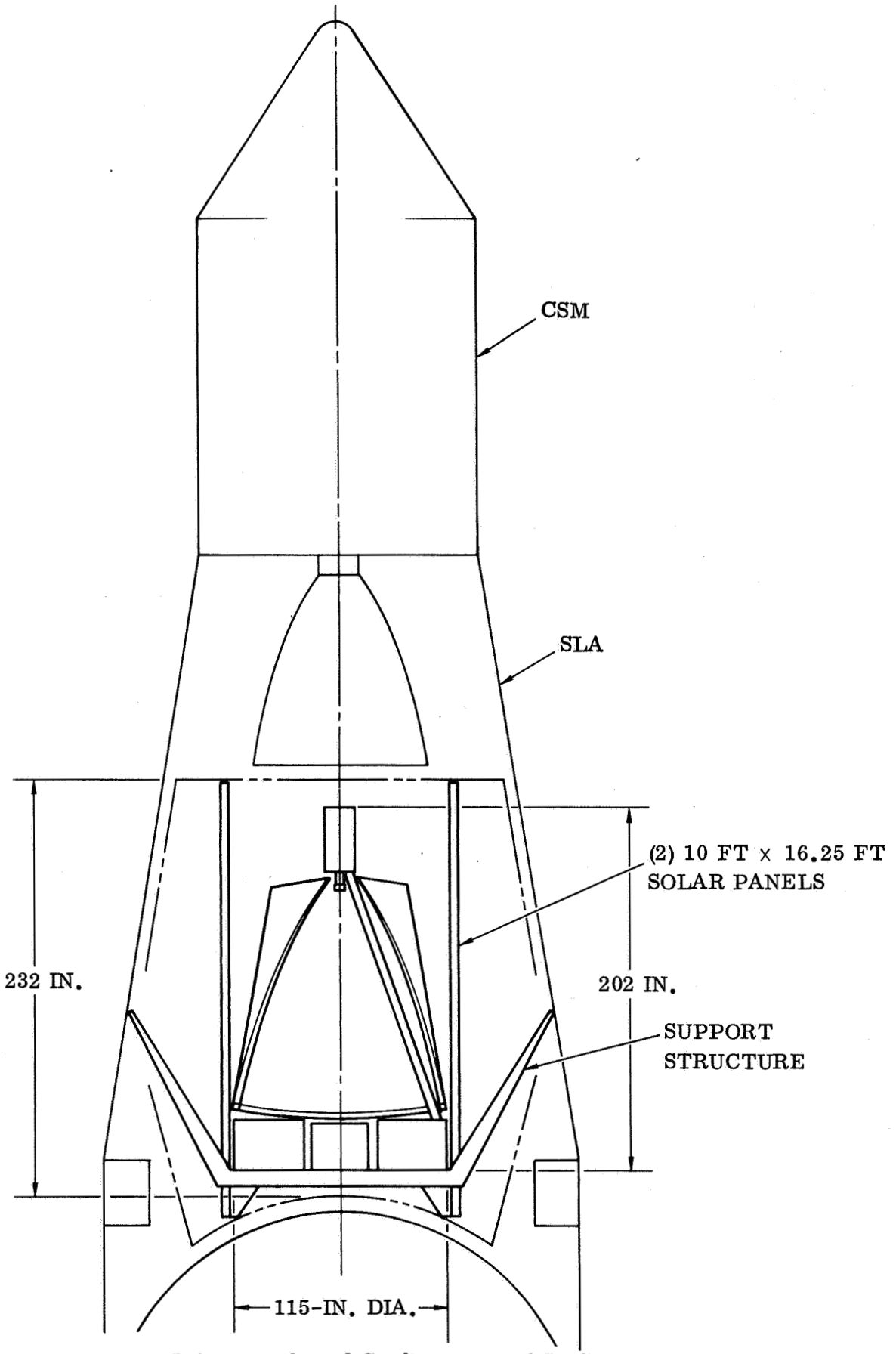



Figure 5-17. Packaged Configuration of Sunflower Antenna (Saturn-V Manned Launch Configuration)

The errors that exist in all large parabolic reflector designs include both mechanical tolerances in manufacturing and phase center problems associated with the feed antenna design.

The mechanical tolerances, which occur as systematic and random surface errors and feed antenna positioning errors, must also include those anticipated from the space environment. For an attitude controlled configuration, thermal gradients existing across the reflector produce distortions in the surface contour and thus increase the systematic phase errors in the aperture. The feed antenna support structure can also be deflected as a result of thermal gradients.

Surface height discontinuities occur due to fabrication tolerances and thermal gradients that exist during operation in the space environment. Since the interlocks between petals provide very good alignment at the point of the interlock, the deviation from a true parabolic survace can be minimized by using a number of interlocks between petals. The following tolerances are obtained for a 30-ft-dia. reflector operating at 2300 MHz:

a. Maximum allowable random surface deviation: 0.050 in. RMS. This results in a reduction in efficiency of approximately 2%.

b. Maximum allowable equivalent systematic surface deviation (given in terms of parabola edge deviation if all deviations were of quadratic form) ± 0.300 in. The associated reduction in efficiency is approximately 2%.

The RMS height deviations from the ideal paraboloid must be measured to predict electrical performance. Actual surface quality data for a large petal paraboloid have been obtained by an optical inspection method. This inspection yields the surface slope deviation at each point along a radius of the paraboloid and includes the net summation of gross waviness, spring-back, deployment positioning, environmental structural loading, and thermal distortion.

The maximum systematic surface deviation (tip deflection) is 0.20 in. and the maximum random surface deviation is 0.05 in. These surface deviations are compatible with the allowables. The surface finish errors are negligible.

5.2.5.2 Antenna Performance. The 30-ft-dia. parabolic reflector provides the basis for a good communications antenna. Reference to Figure 5-18 indicates a gain of 44 db and beamwidth of 1° at the design frequency of 2300 MHz.

The effects of the RMS surface tolerances are noted on the graph. However, the effect of tip displacement on antenna performance is not clearly defined. The tip displacement causes the focal point of the parabola to change and the relative wave phase to distort. The mode of change depends primarily on thermal gradients. These gradients in turn are determined by orientation of the antenna with respect to the sun.

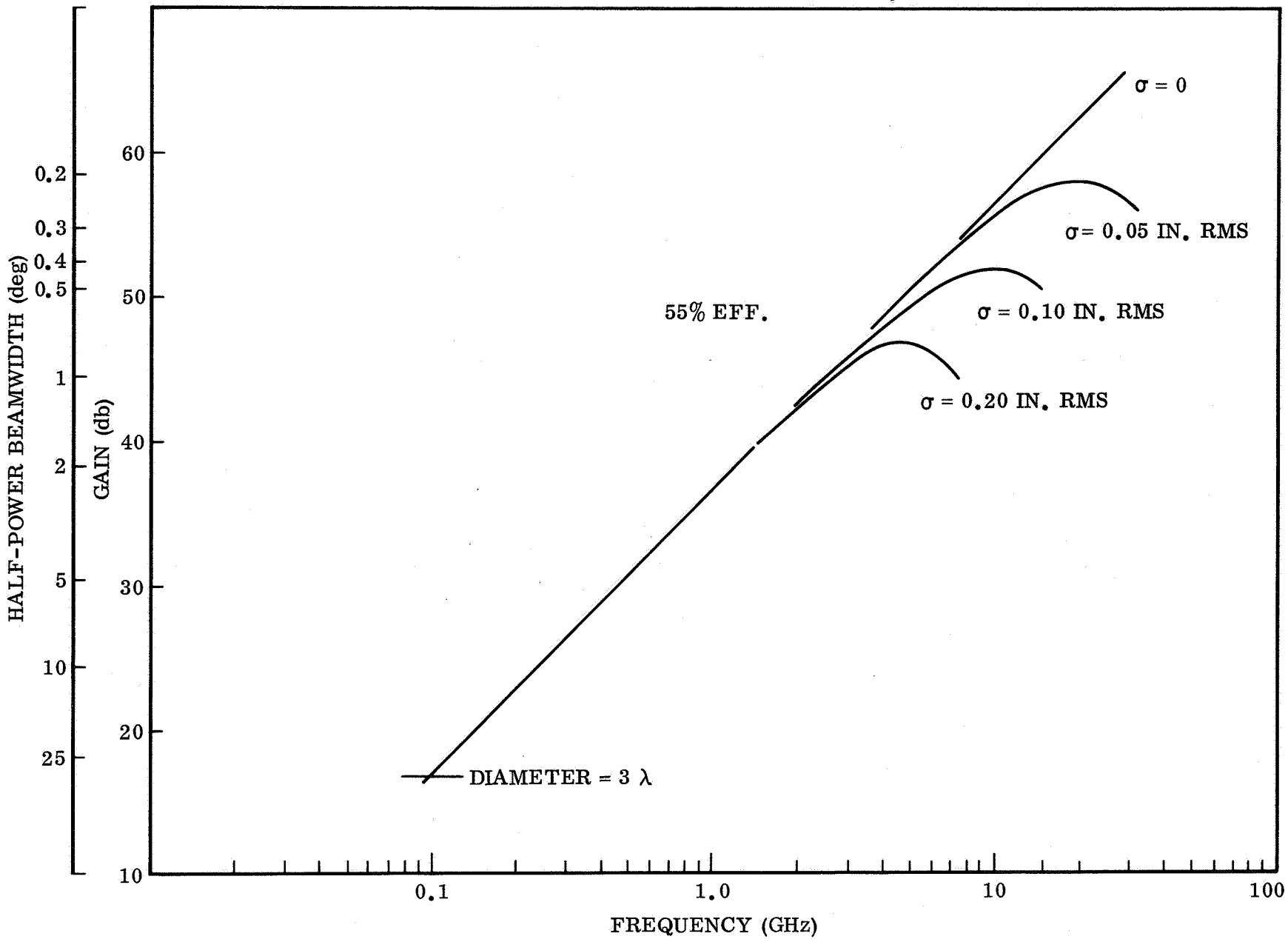



Figure 5-18. RF Performance of 30-Ft Parabolic Antenna

The electrical performance would degrade from that noted on the graph in accordance with the mode of the tip displacement. A method of partial compensation consideration is an adjustable feed. With this capability the feed can be moved to coincide with the optimum focus point for any particular thermal distortion.

The 30-ft parabolic antenna is capable of operation over a wide frequency range. At 1 GHz the antenna gain is 37 db and the half-power beamwidth is 2.3°. At 10 GHz the gain is 57 db and the half-power beamwidth is 0.23°.

The upper frequency performance is limited by the surface tolerance of the reflector.

5.2.5.3 Electrical Power. Electrical power requirements for this concept as based upon 500 w of transmitted power resulting in 2500 w of input plus an estimated 100 w for instrumentation, data and telemetry systems for a maximum load of 2.6 kw. Assuming that the basic load of 2500 w is continuous and the 100 w is of an intermittent nature a design load of approximately 2.6 kw is considered a conservative estimate. A final load determination will require a precise mission analysis, which is outside the scope of this task. It is anticipated that this power requirement can be satisfied using either oriented or non-oriented solar cell panels. Based upon a 2.6 kw average load in synchronous orbit for durations up to 10 years the power requirements in terms of weight, deployed area, and volume are:

| | WEIGHT (lb) | AREA ($ft^2$) | VOLUME ($ft^3$) |
|---|---|---|---|
| Non-Oriented | 428 | 506 | 51 |
| Oriented | 374 | 253 | 26 |

These weights and areas reflect the strong influence exerted by the battery requirements, which tend to even out the comparative weight penalties between paddle-mounted and oriented-panel-mounted systems.

5.2.5.4 Weights. The sunflower antenna weight breakdown is shown in Table 5-8.

Table 5-8. Estimated Weight Summary

| | WEIGHT (lb) |
|---|---|
| Total antenna weight | 247.5 |
| RF equipment | 100.0 |
| Electrical Power Supply | 428 |
| Attitude Control System | 30 |
| Support Structure | 270.0 |
| Total Experiment | 1,075.5 |

5.2.6 Concept 12 — TV Broadcast Satellite (RCA). The television broadcast satellite in synchronous orbit will provide direct service to home receivers over 500,000 $mi.^2$, or 1/6 of the continental U. S. A 50-ft dia. parabolic antenna transmitting at a frequency of 800 MHz with transmitter output power of 5 kw provides an effective radiated power of 30 Mw.

The antenna system consists of an erectable reflector, a conical horn feed, and feed support structure.

The antenna is a fine weave metallic mesh paraboloid containing 30 ribs hinged to a fixed center hub and deployed by torsion springs. Locking devices on each rib hold the mesh in a near parabolic shape. See Figure 5-19.

The satellite power requirements are supplied by an 18 kw Pegasus-type solar cell array. The two 1000-$ft^2$ solar paddles are continuously sun oriented. The required antenna orientation is maintained by stabilization of the spacecraft to within 1° using an infrared horizon sensor. The antenna feed mechanism is canted providing a beam of several degrees and pointing accuracy of 0.2°.

The 3500 lb experiment as proposed cannot be launched on a Saturn-V manned launch vehicle because of insufficient payload volume. The experiment package can be launched in the larger unmanned nose fairing, but there are no current unmanned synchronous orbit missions in the AAP program ground rules for this study. See Appendix C.

The 50-ft antenna that RCA proposes to use is a scaled-up version of the 10-ft model that the company has developed for use by Apollo astronauts on the lunar surface for earth communications. Automatically deploying an unfurlable antenna of the size required for a TV broadcast satellite, with the requirement that it maintain its shape to a reasonable close tolerance, has yet to be done in space. A weight estimate is provided in Table 5-9.

One-third of the proposed RCA spacecraft weight is devoted to a solar cell array to provide electric power and to radiate much of the resulting heat into space. Approximately 16.2 kw of prime electric power must be generated onboard.

Achieving maximum power with minimum paddle area requires a motor drive to keep the solar cell arrays continusouly oriented toward the sun throughout the 24-hr rotation of the earth and spacecraft.

The spacecraft transmitting antenna is oriented by attitude stabilization of the spacecraft to within 1° using an infrared horizon sensor, supplemented by precision aiming of the antenna beam itself. This is accomplished by canting the antenna feed mechanism, providing a range of beam shifting of several degrees, and permitting it to be aimed to within 0.2°.

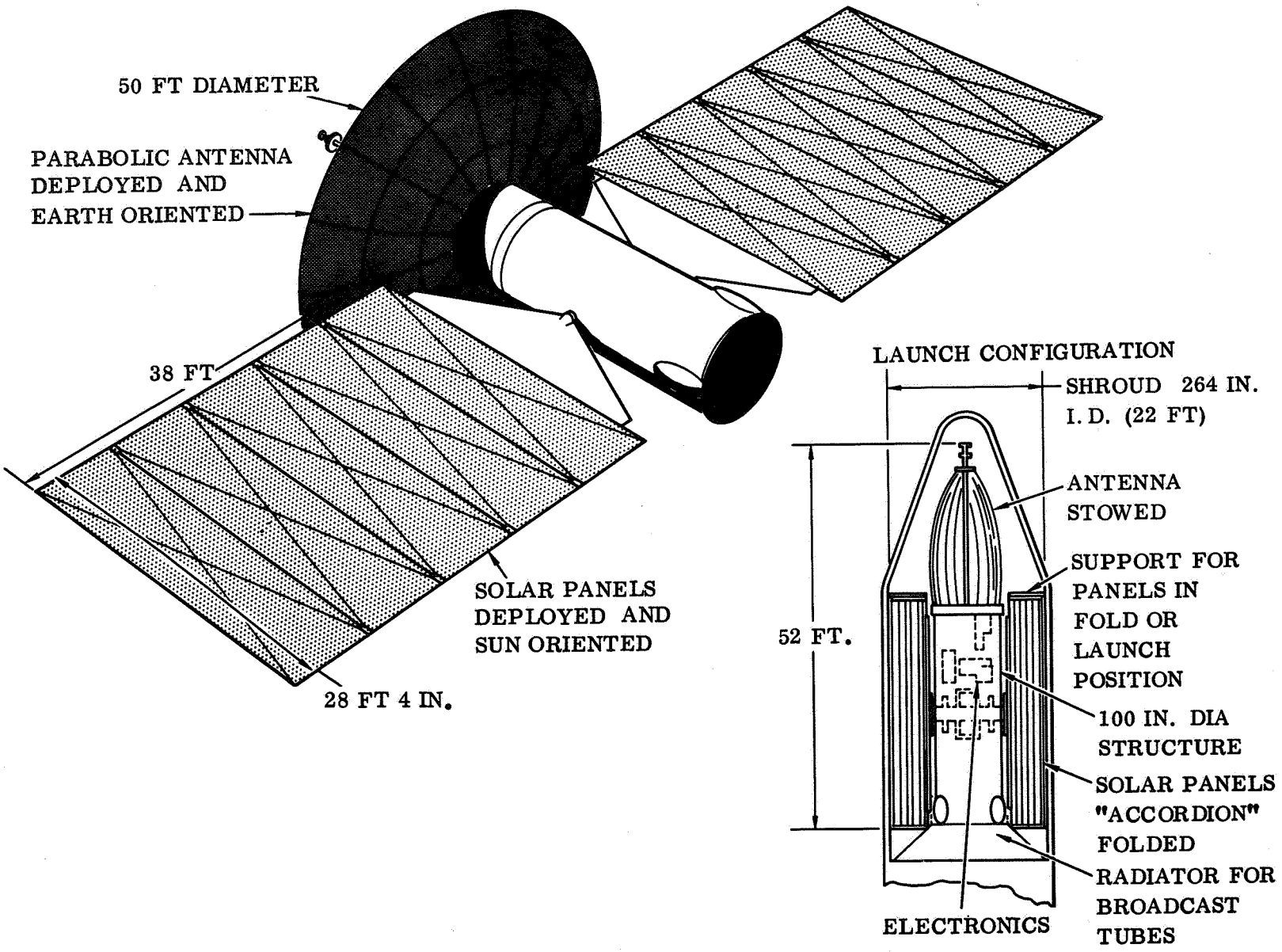



Figure 5-19. Concept 12, TV Broadcast Satellite (RCA)

Table 5-9. Concept 12 Weights

| | WEIGHT (lb) |
|---|---|
| Antenna | 482 |
| Solar panels | 1410 |
| Attitude control | 145 |
| Electronic equipment | 705 |
| Structure | 535 |
| Batteries | 978 |
| Total experiment weight | 4255 |

Precise alignment of the beam centering can be accomplished by installing a tracking beacon on the ground at the desired beam center location and using an antenna servo system to orient to this ground reference.

5.2.6.1 Structure and Mechanics. The antenna is a fine-mesh ribbed paraboloid based on the LEM antenna design intended for erection in space. Weight of the antenna and movable feed subsystem is 482 lb.

Studies of unfurlable, ribbed, parabolic antennas have shown that rib and mesh saddle effect errors are relatively constant over the range of focal ratios (f/d) between 0.35 and 0.6. Deviation from true paraboloidal shape caused by these errors produces a "gain loss" in the unfurlable antenna. The maximum useful gain for a given number of ribs occurs when the contour errors cause a gain loss of 3 db at the operating frequency. More ribs are required to hold the same gain loss as the antenna diameter or operating frequency is increased.

The scissor linkage solar panel deployment has been successfully used for Pegasus; it is a mechanically redundant system, therefore very reliable. The two deployed wing structures, however, are very large compared to those used on Pegasus and interconnected through a bearing supported shaft. A highly rigid system of panel linkage and central supporting shafting is required to provide fundamental vibrational frequencies greater than the input frequencies that are generated by the attitude control system.

5.2.6.2 Antenna Performance. Electrical characteristics of the concept are summarized in Table 5-10 for comparison with communications user requirements. The user requirements listed for comparison are those of real-time TV direct to home, for a single time zone.

Table 5-10. User Requirements vs Concept Characteristics

| FEATURE | SPECIFIED USER REQUIREMENTS | CONCEPT CHARACTERISTIC |
|---|---|---|
| Lifetime | 2 to 10 years | |
| Orbit Altitude | Synchronous | Synchronous |
| Effective beamwidth | 2.6° × 3.5° | 1.75° × 1.75° |
| Pointing accuracy | ± 0.3° | ± 0.2° |
| Frequency | 900 MHz (Approximately) | 758 MHz to 806 MHz |
| Bandwidth | 6 MHz per channel, 3 channels | ≥ 48 MHz |
| Antenna gain | + 35 db | + 39 db |
| Antenna size | 29 ft × 22 ft | 50-ft diameter |
| Transmitter | 6.8 kw per channel | 3 kw average (1 channel) |
| Power output | 20 kw total (3 channels) | |
| Maneuverability | Fixed attitude. Three antenna beams directed continuously each toward a different time zone | Attitude controlled to have fixed attitude. Capable of attitude change to cover a different region of earth |
| Tolerance | Not directly specified | |

The user requirements specified are intended to be typical of those that might be applied to a specific communications function of the type listed. An individual concept may have characteristics quite different in detail than the listed user requirements, and still be entirely applicable to the listed function.

Concept 12 has been devised to perform the specific communications function of a direct-to-home TV relay, to cover a limited user area within the United States. In general the concept appears to be capable of performing that function.

The extent of the study required to make a comprehensive tradeoff analysis of the merits of the concept as opposed to one that would cover the larger area of a complete time zone is outside the scope of the present study, but is a subject for more detailed investigation.

5.2.6.3 Thermodynamics. The temperature gradient problem that occurs as a result of edge effects can be serious. For a vehicle in a synchronous orbit, this condition can exist for possibly two hours. The exact time is a function of the angle between the orbit plane and the ecliptic plane. As the angle tends toward zero, the maximum two hours is approached. Solar heating of the feed is minimized by the earth orientation of this concept.

Surface area required for various power levels to be dissipated was calculated for a cylinder and shown in Figure 5-20. The area sizing was based on a maximum operating temperature of 90°F. A solar absorptance of 0.16 and a thermal emittance of 0.85 were used. Higher power loads than that shown for the housing surface of the cylinder will require auxiliary radiating surfaces.

5.2.7 Concept 15 — Expandable Scissor Truss Parabolic Antenna (Fairchild-Hiller). Fairchild-Hiller proposes a 30-ft-dia. parabolic antenna, Figure 5-21, for FM communications and radio astronomy in the millimeter wavelength region (88 Mhz to

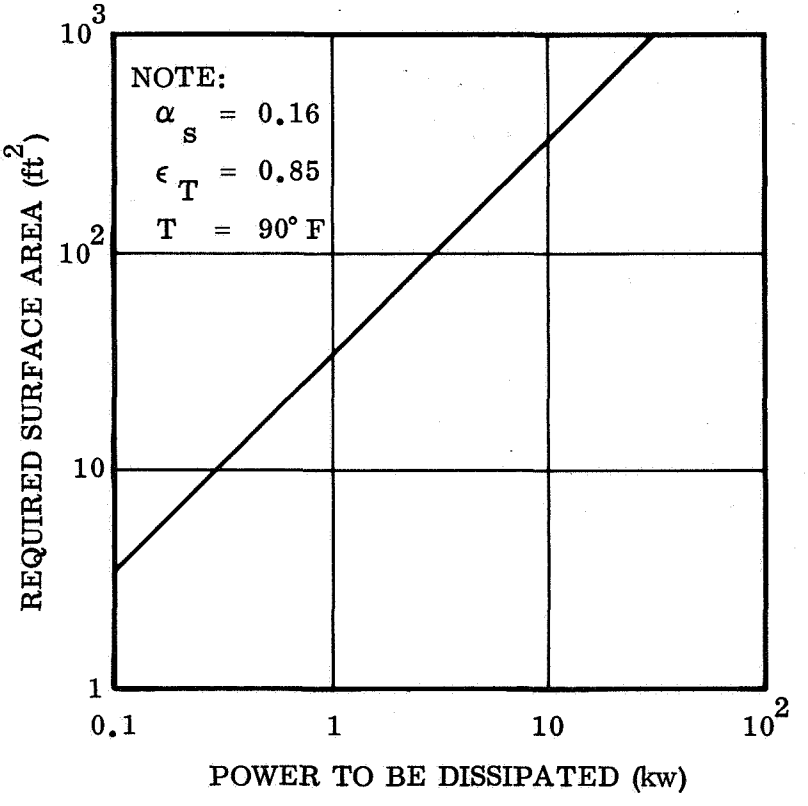



Figure 5-20. Passive Power Dissipation in Synchronous Orbit for a Cylinder

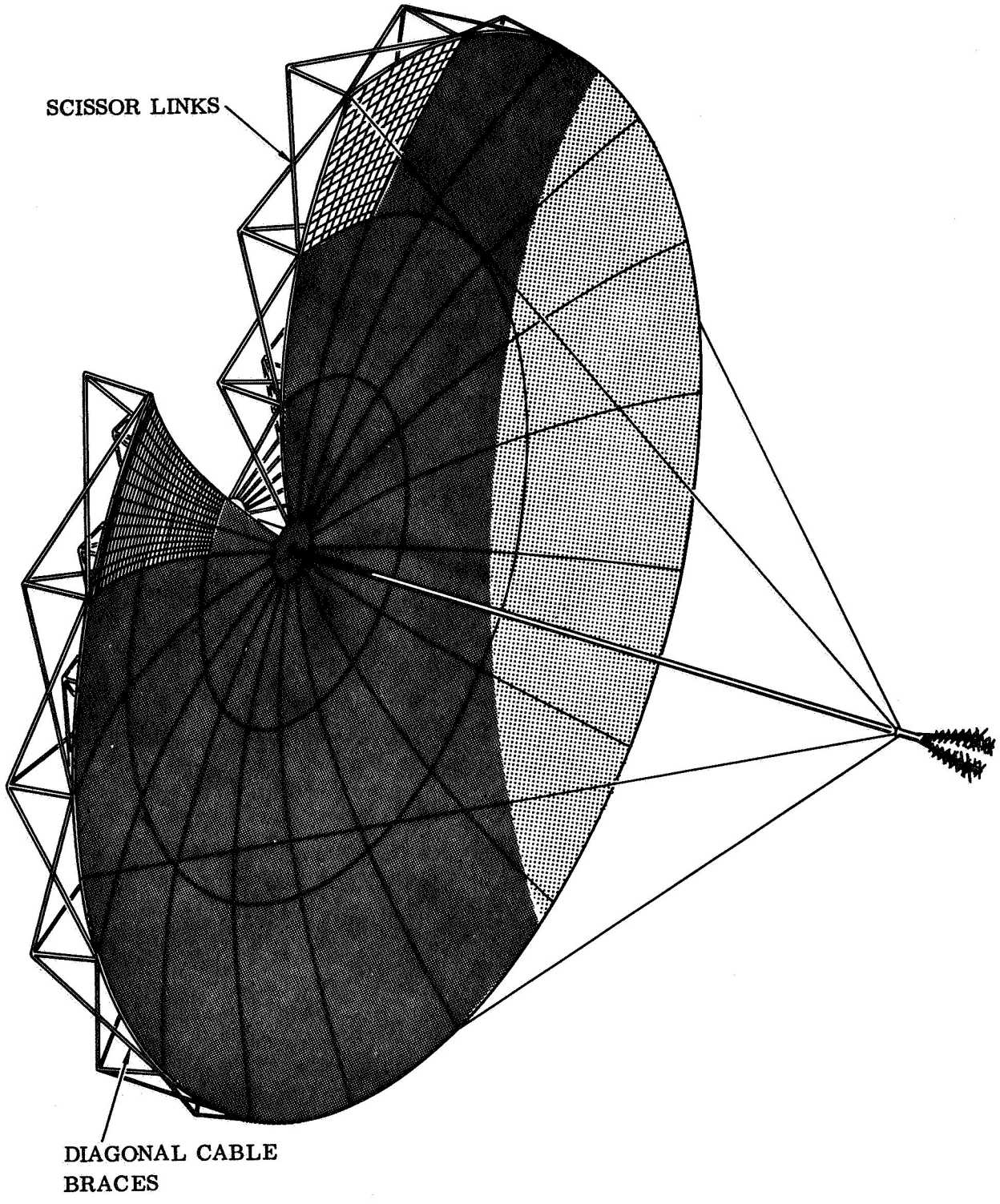



Figure 5-21. Concept 15, 30-Ft-Dia. Communications Antenna (Fairchild-Hiller)

30 GHz). (See Figure 5-21). However, because of the minimum practical structural tolerance and the antenna size, a more realistic operating range is 100 MHz to 4 GHz. Presently, the only user requirement in this frequency range is telephone communications of 1 to 10 GHz. For this application, a manned Saturn-V is required to put the antenna into synchronous orbit.

The support structure appears feasible with regard to rigidity and deployment. Additional studies are required to better define the solar cell arrangement, packaging, docking facility, materials, and methods for aligning the structure while on the ground.

Two alternate types of reflective surface were proposed. It can either be a flexible mesh deployed with the structure or rigid-mesh panels installed by an astronaut. For the flexible mesh surface, a deflection of 0.2 in. can be expected, which limits the antenna upper frequency to 4 GHz. As for the rigid panels, Fairchild-Hiller states that an average surface tolerance of 0.03 in. RMS can be met. Not considering other distortional effects, this tolerance limits the upper frequency to 20 GHz — a definite improvement over the flexible surface with regard to antenna performance, presuming minimal thermal distortion.

The flexible surface deploys with the structure, which reduces the over-all deployment time but also reduces the upper limit of the desired frequency range. Rigid panels, on the other hand, improve antenna capability but increase deployment time since there are 54 panels to install. A problem is anticipated in moving large panels from the launch vehicle to the antenna structure.

The basic antenna weight as given by Fairchild-Hiller is 525 lb and appears to be reasonable. The electrical power supply weight is less than 500 lb and electronic equipment is about 50 lb.

5.2.7.1 Antenna Performance. The 30-ft parabolic antenna provides the basis for a good communications antenna. See Table 5-11. The antenna is capable of operating at frequencies from 0.1 GHz to 4 GHz; the antenna gains and half-power beamwidths are 17 db at 23° and 47 db at 0.57°. See Figure 5-18 for RF performance.

The upper frequency performance is dependent on the reflector surface structural tolerances, surface irregularities, and feed positioning inaccuracies. The primary effects of these random errors is a reduction in gain and increase in side lobe effects. In addition to the random errors systematic errors are also present. These contribute to gain degradation, beam axis tilting, and increase side lobe density. The largest cause of systematic error in surface deformation is the solar induced thermal deflections.

Table 5-11. Summary of Characteristics of the 30-Ft Parabolic Antenna

| FREQUENCY (MHz) | BEAMWIDTH 30-FT APERTURE (deg) | GAIN 55% EFF. (db) | APPLICATION |
|---|---|---|---|
| 100 | 23 | 17 | Broadcast FM aircraft control |
| 800 | 2.9 | 35 | Broadcast UHF TV type signals |
| 1700<br>2100<br>2300 | 1.4<br>1.1<br>1.0 | 41<br>44<br>44.3 | Telemetry and command associated communication experiments (satellite tracking) |

5.2.7.2 Crew Systems. The gross crew functional analysis and time-line analysis are markedly affected by the choice of reflecting surface. The deployment of the entire flexible mesh reflector is a semiautomatic and relatively simple procedure compared to the deployment of the rigid panel reflector as can be seen by their time line analyses, Tables 5-12 and 5-13.

The most difficult task for the EV crew appears to be maneuvering and positioning the solar cell arrays and the rigid surface panels. The degree of difficulty will depend on the exact size and weight of these components, but in any case, the task would be most efficiently handled by two men with maneuvering units or some type of cargo truck, if it can be accomplished at all. The flexible mesh reflector could possibly be deployed by one man, depending on the size and weight of the solar cell arrays.

5.2.8 Concept 16 — Phased Array (Fairchild-Hiller). The Fairchild-Hiller phased array was proposed for applications that include broadband communications over the frequencies from 88 MHz for FM communications to 4 GHz for TV. The best usage of the 78-ft-dia. concept appears to be in the lower VHF range from about 30 MHz to 150 MHz. See Figure 5-22. Increased diameters to 200 ft or larger would increase the application possibilities, particularly in the lower VHF spectrum.

The mechanical and structural concept has considerable merit. Although many joints must function to satisfy deployment, the mechanical simplicity is conducive to reliable performance. The deployed configuration presents a substantially deep truss and its open construction has the advantage of minimum thermal and dynamic distortions. Low solar and aerodynamic torques reduce required ACS propellant in low orbit.

The concept possesses good packaging characteristics, and diameters in excess of 225 ft are conceivable. It is of significance that the 78-ft array can be stowed within all standard launch configurations.

Table 5-12. Time-Line Analysis for Flexible Mesh Reflector

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 1 | | 30 | 30 | | Perform pre-EVA |
| 2 | | 25 | 55 | | Depressurize spacecraft |
| 3 | Initiate and monitor reflector deployment | 30 | 85 | | Open hatch and exit |
| 4 | | 2 | 87 | 2 | Anchor |
| 5 | | 15 | 102 | 17 | Unstow and checkout maneuvering device |
| 6,7 | | 1 | 103 | 18 | Release anchors. Maneuver and inspect deployed reflector, anchoring to worksite and locking flagged joints where necessary |
| 8 | | 90 | 223 | 138 | Unstow, reposition, and attach solar cell arrays to reflector structure and return to equipment stowage area |
| 9 | | 2 | 225 | 140 | Anchor to equipment storage area |
| 10 | | 5 | 230 | 145 | Unstow reflector mast and guy wires |
| 11 | | 1 | 231 | 146 | Release anchors |
| 12 | | 5 | 236 | 151 | Maneuver |
| 13 | | 2 | 238 | 153 | Anchor to worksite |
| 14 | | 15 | 253 | 168 | Install reflector mast and guy wires |
| 15 | | 1 | 254 | 169 | Release anchors |
| 16 | | 5 | 259 | 174 | Maneuver |

Table 5-12. Time-Line Analysis for Flexible Mesh Reflector, Contd .

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 17 | | 2 | 261 | 176 | Anchor to equipment storage area |
| 18 | | 5 | 266 | 181 | Unstow optical checkout equipment |
| 19 | | 1 | 267 | 182 | Release anchors |
| 20 | | 5 | 272 | 187 | Manuever to worksite |
| 21 | | 2 | 274 | 189 | Anchor to worksite |
| 22 | | 15 | 289 | 204 | Perform optical checkout of reflector surface |
| 23 | | 1 | 290 | 205 | Release anchors |
| 24 | | 5 | 295 | 210 | Manuever |
| 25 | | 2 | 297 | 212 | Anchor to equipment storage area |
| 26 | | 5 | 302 | 217 | Stow optical checkout equipment |
| 27 | | 5 | 307 | 222 | Unstow electrical equipment |
| 28 | | 1 | 308 | 223 | Release anchors |
| 29 | | 5 | 313 | 228 | Manuever |
| 30 | | 2 | 315 | 230 | Anchor to worksite |
| 31 | | 5 | 320 | 235 | Make electrical connections |
| 32 | | 1 | 321 | 236 | Release anchors |
| 33 | | 5 | 326 | 241 | Manuever |
| 34 | | 2 | 328 | 243 | Anchor |
| 35 | | 5 | 333 | 248 | Stow maneuvering device |
| 36 | | 1 | 334 | 249 | Release anchors |
| 37 | | 2 | 336 | 251 | Maneuver to spacecraft |
| 38 | Perform electrical and attitude control sys. checkout | 60 | 396 | | Maneuver to spacecraft. Enter and close hatch, repressurize and stow gear |

Table 5-13. Time-Line Analysis for Rigid Panel Reflector

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 1 | Perform pre-EVA | 30 | 30 | | Perform pre-EVA |
| 2 | Initiate and monitor reflector deployment | 30 | 60 | | Depressurize spacecraft |
| 3 | Open hatch and exit | 30 | 90 | | Same as Astronaut 1 |
| 4 | Anchor | 2 | 92 | 2 | Same as Astronaut 1 |
| 5 | Unstow and checkout maneuvering device | 15 | 107 | 17 | Same as Astronaut 1 |
| 6 | Release anchors | 1 | 108 | 18 | Same as Astronaut 1 |
| 7 | Maneuver and inspect deployed reflector, anchoring to worksite and locking flagged joints where necessary | 20 | 128 | 38 | Same as Astronaut 1 |
| 8 | Unstow, reposition, and attach solar cell arrays to reflector structure and return to equipment stowage area | 60 | 188 | 98 | Same as Astronaut 1 |
| 9 | Anchor to equipment stowage | 2 | 190 | 100 | Same as Astronaut 1 |
| 10 | Unstow reflector mast and guy wires | 5 | 195 | 105 | Same as Astronaut 1 |
| 11 | Release Anchors | 1 | 196 | 106 | Same as Astronaut 1 |
| 12 | Maneuver | 5 | 201 | 111 | Same as Astronaut 1 |
| 13 | Anchor to worksite | 2 | 203 | 113 | Same as Astronaut 1 |
| 14 | Install reflector mast and guy wires | 10 | 213 | 123 | Same as Astronaut 1 |
| 15 | Release anchors | 1 | 124 | 124 | Same as Astronaut 1 |
| 16 | Maneuver | 5 | 219 | 129 | Same as Astronaut 1 |
| 17 | Anchor to panel storage area | 2 | 221 | 131 | Same as Astronaut 1 |
| 18 | Unstow first rigid panel | 3 | 224 | 134 | Same as Astronaut 1 |
| 19 | Release anchors | 1 | 225 | 135 | Same as Astronaut 1 |
| 20 | Maneuver | 5 | 230 | 140 | Same as Astronaut 1 |
| 21 | Anchor to worksite | 2 | 232 | 142 | Same as Astronaut 1 |
| 22 | Position panel | 5 | 237 | 147 | Position and fasten panel in place |
| 23 | Release anchors | 1 | 238 | 148 | Same as Astronaut 1 |
| 24 | Maneuver | 5 | 243 | 153 | Same as Astronaut 1 |
| 25-56 | Repeat events 17-24 for next 3 panels | 72 | 315 | 225 | Repeat events 17-24 for next four panels |

Table 5-13. Time-Line Analysis for Rigid Panel Reflector, Contd

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 57 | Anchor | 2 | 317 | 227 | Same as Astronaut 1 |
| 58 | Stow maneuvering device | 5 | 322 | 232 | Same as Astronaut 1 |
| 59 | Release anchors | 1 | 323 | 233 | Same as Astronaut 1 |
| 60 | Maneuver to spacecraft | 2 | 325 | 235 | Same as Astronaut 1 |
| 61 | Enter and close hatch, repressurize and stow gear | 60 | 385 | | END OF DAY 1 |
| 62 | Perform pre-EVA | 30 | 415 | | Same as Astronaut 1 |
| 63 | Depressurize spacecraft | 25 | 440 | | Same as Astronaut 1 |
| 64 | Open hatch and exit | 30 | 470 | | Same as Astronaut 1 |
| 65 | Anchor | 2 | 472 | 237 | Same as Astronaut 1 |
| 66 | Resupply, unstow, and checkout maneuvering device | 15 | 487 | 252 | Same as Astronaut 1 |
| 67 | Release anchors | 1 | 488 | 253 | Same as Astronaut 1 |
| 68 | Maneuver | 1 | 489 | 254 | Same as Astronaut 1 |
| 69-140 | Repeat Events 17-24 for next 9 panels | 216 | 705 | 470 | Same as Astronaut 1 |
| 141-145 | Repeat Events 57-61 | 70 | 775 | 480 | END OF DAY 2 |
| 146-229 | Repeat Day 2 for next 9 panels | 390 | 1165 | 625 | Same as Astronaut 1<br>END OF DAY 3 |
| 230-313 | Repeat Day 2 for next 9 panels | 390 | 1555 | 970 | Same as Astronaut 1<br>END OF DAY 4 |
| 314-397 | Repeat Day 2 for next 9 panels | 390 | 1945 | 1215 | Same as Astronaut 1<br>END OF DAY 5 |
| 398-481 | Repeat Day 2 for next 9 panels | 390 | 2335 | 1460 | Same as Astronaut 1<br>END OF DAY 6 |
| 482-488 | Repeat Events 62-68 | 104 | 2439 | 1479 | Same as Astronaut 1 |
| 489-536 | Repeat Events 17-24 for last 6 panels | 144 | 2583 | 1623 | Same as Astronaut 1 |
| 537 | Anchor to equipment stowage area | 2 | 2585 | 1625 | Same as Astronaut 1 |
| 538 | Unstow optical checkout equipment | 5 | 2590 | 1630 | Unstow electrical equipment |

Table 5-13. Time-Line Analysis for Rigid Panel Reflector, Contd

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 539 | Release anchors | 1 | 2591 | 1631 | Release anchors |
| 540 | Maneuver | 5 | 2596 | 1636 | Manuever |
| 541 | Anchor to worksite | 2 | 2598 | 1638 | Anchor to worksite |
| 542 | Perform optical checkout of reflector surface | 15 | 2613 | 1653 | Make electrical connections |
| 543 | Release anchors | 1 | 2614 | 1654 | Release anchors |
| 544 | Maneuver | 5 | 2619 | 1659 | Manuever |
| 545 | Anchor | 2 | 2621 | 1661 | Anchor |
| 546 | Stow optical checkout equipment | 5 | 2626 | 1666 | Stow maneuvering device |
| 547 | Stow maneuvering device | 5 | 2631 | 1671 | |
| 548 | Release anchors | 1 | 2632 | 1672 | Same as Astronaut 1 |
| 549 | Maneuver to spacecraft | 2 | 2634 | 1674 | Same as Astronaut 1 |
| 550 | Enter and close hatch. Repressurize and stow gear | 60 | 2694 | | |

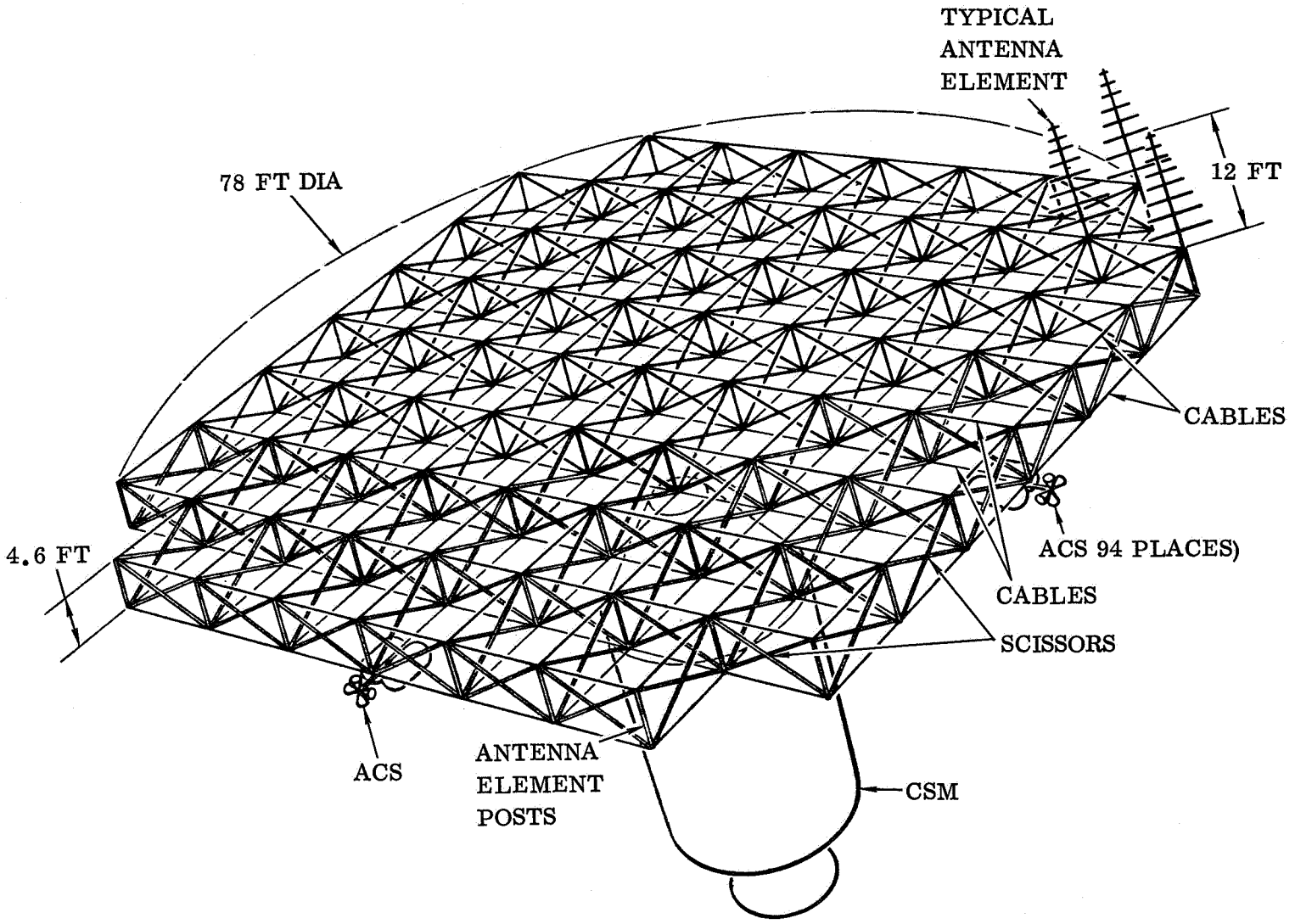



Figure 5-22. Concept 16, Bidirectional Scissors Phased Array (Fairchild-Hiller)

Astronaut participation is required to verify deployment, secure mechanism, insert antenna elements, check element alignment, and make transmission line connections.

The rigidity, and in turn the natural frequency of the array, can be increased by the astronaut adding upper and lower chord members to the truss. These members should be tension cables pre-adjusted for length and tension and stored in reels or equivalent. The same reels and tension cable system can also serve to control the rate of deployment. Resulting lower stress levels elsewhere in the structure will partially compensate for the weight of the cable system.

The concept as originally conceived assumes considerable EVA. The amount seems disproportionate and can be reduced by designing the tension ties as part of the automated deployment. A 200-ft-dia. array would greatly increase the EVA task time.

5.2.8.1 Antenna Performance. The length of the longest dipole in each array is assumed to be 6 ft. With this assumption, the lowest frequency of operation would be one for which the longest dipole element was just slightly greater than a half wavelength, or about 90 MHz. At 90 MHz the 8-ft spacing between elements would be about 0.88 wavelength. The beamwidth for the entire array would be approximately 10° x 10° at 90 MHz.

The upper frequency for practical operation of the array is limited by the fixed physical spacing between elements. This spacing, in terms of electrical wavelengths, increases proportional to frequency. Whereas, at 90 MHz the spacing is about 0.88 $\lambda$, at 150 MHz the spacing amounts to about 5/4 $\lambda$. At frequencies above 150 MHz where the spacing would be greater than 5/4 $\lambda$, the sidelobes would increase to a size that would make the antenna array of doubtful value. Because of this, it appears that 150 MHz is an upper frequency limit for the particular concept configuration assumed for analysis. Thus this configuration would have a ratio of highest to lowest operating frequency of about 1.6.

The concept description indicates that four beams of radiation would be provided by the array. How this is to be achieved was not described. Various methods, however, such as the use of a Butler matrix for feeding the array, are available for accomplishing such beam directing.

The parameters for analysis were chosen to be compatible with the 78-ft-dia. array referenced in the Fairchild concept description. They do not necessarily comprise an optimum design for use in a direct to distribution center function for a television relay system. They do illustrate, however, that the concept usage is limited to applications requiring a bandwidth ratio less than about 1.6. Many of the communications functions do fit within such a restricted range. In general it appears that the best usage of the concept would be in the lower VHF range from about 30 MHz to 150 MHz. In this frequency range, the potential of the concept for a large deployed area could be

best realized because of the large element-to-element spacing requirements. Enlargement of the concept to include array diameters as large as 200 ft or more would add to the number of possible applications for the concept, particularly in the lower VHF spectrum where the wavelength may be as large as 33 ft.

5.2.8.2 Weights. The required electrical power equipment for a communications antenna of this type weighs less than 300 lb. The attitude control requirements for such a configuration are very small. A representative weight breakdown is shown in Table 5-14 for 78- and 200-ft-dia. arrays. Crew support or related facilities are not included.

Table 5-14. Weight Summary

| | ARRAY WEIGHT (lb) | | | |
|---|---|---|---|---|
| | 78-FOOT | | 200-FOOT | |
| ORBIT | 200 N.MI. | SYN | 200 N.MI. | SYN |
| Array structure and mechanisms | 525 | 525 | 2500 | 2500 |
| Feed elements | 155 | 155 | 1872 | 1872 |
| Electrical transmission lines | 50 | 50 | 187 | 187 |
| Electronics | 200 | 200 | 400 | 400 |
| ACS (one year propellant) | 20 | 4 | 81 | 6 |
| Electrical power | 300 | 300 | 300 | 300 |
| TOTAL | 1250 | 1234 | 5340 | 5265 |

The 5340 lb weight of the 200-ft-dia. array indicates that ample margin below the capabilities of the S-1B or S-V launch vehicles exists.

5.2.8.3 Crew Systems. The gross crew and equipment functions in the deployment of the 78-ft bidirectional scissors array were determined, and the time-line analysis is included as Table 5-15.

The majority of the tasks that the EV crewman is required to perform appear to be repetitive, low energy, one-man tasks. The most difficult EV task appears to be maneuvering the modules such as the attitude control system packages from the stowed position to the worksite. The exact workload imposed by this task will depend on the size and weight of the modules and the maneuvering distances involved.

Table 5-15. Time Line Analysis

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 1 | Perform pre-EVA | 30 | 30 | | Perform pre-EVA |
| 2 | Depressurize spacecraft | 25 | 55 | | |
| 3 | Open hatch and exit | 30 | 85 | | |
| 4 | Manuever | 1 | 86 | 1 | Prepare to initiate array deployment |
| 5 | Anchor to worksite | 2 | 88 | 3 | |
| 6 | Release antenna package restraining cables | 4 | 92 | 7 | |
| 7 | Stow cables | 2 | 94 | 9 | |
| 8 | Release package locks | 4 | 98 | 13 | |
| 9 | Extend and lock | 15 | 113 | 28 | |
| 10 | Array in pre-deployment position | | | | |
| 11 | Release worksite anchors | 1 | 114 | 29 | |
| 12 | Manuever | 1 | 115 | 30 | |
| 13 | Enter spacecraft | 2 | 117 | 32 | |
| 14 | | 1 | 118 | 33 | Initiate array deployment |
| 15 | Monitor array deployment | 15 | 133 | 48 | Monitor array deployment |
| 16 | Exit spacecraft | 1 | 134 | 49 | Identical task as Astronaut 1, but for other half of antenna |
| 17 | Manuever | 1 | 135 | 50 | " " " |
| 18 | Anchor to worksite | 2 | 137 | 52 | " " " |
| 19 | Lock telescoping antenna element | 2 | 139 | 54 | " " " |
| 20 | Release anchors | 1 | 140 | 55 | " " " |
| 21 | Maneuver | 1 | 141 | 56 | " " " |
| 22-173 | Repeat Events 18-21 for 38 other elements | 228 | 369 | 284 | Repeat Events 18-21 for 37 other elements |
| 174 | Enter spacecraft, close hatch, repressurize and stow gear | 60 | 429 | | Identical tasks as Astronaut 1<br>END OF DAY 1 |
| 175 | Perform pre-EVA | 30 | 459 | | Identical tasks as Astronaut 1 but for other half of antenna |
| 176 | Depressurize spacecraft | 25 | 474 | | " " " |

Table 5-15. Time Line Analysis, Contd

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 177 | Open hatch and exit | 30 | 504 | | Identical tasks as Astronaut 1 but for other half of antenna |
| 178 | Check alignment of deployed structure | 15 | 519 | 299 | " " " |
| 179 | Maneuver | 1 | 520 | 300 | " " " |
| 180 | Anchor to worksite | 2 | 522 | 302 | " " " |
| 181 | Install active antenna element in proper position on deployed structure | 3 | 525 | 305 | " " " |
| 182 | Release worksite anchors | 1 | 526 | 306 | " " " |
| 183 | Manuever | 1 | 527 | 307 | " " " |
| 184-335 | Repeat Events 180-183 for 31 other elements | 217 | 744 | 524 | " " " |
| 336 | Enter spacecraft, close hatch, repressurize and stow gear | 60 | 804 | | END OF DAY 2 |
| 337 | Perform pre-EVA | 30 | 834 | | Identical tasks as Astronaut 1 |
| 338 | Depressurize spacecraft | 25 | 859 | | Identical tasks as Astronaut 1 |
| 339-366 | Repeat Events 180-183 for 7 remaining elements | 49 | 938 | 573 | Repeat Events 180-183 for Astronaut 2 remaining elements |
| 367 | Maneuver and inspect alignment of active elements | 15 | 953 | 588 | Identical tasks as Astronaut 1 |
| 368 | Maneuver to equipment stowage area | 3 | 956 | 591 | Identical tasks as Astronaut 1 |
| 369 | Anchor to worksite | 2 | 958 | 593 | Identical tasks as Astronaut 1 |
| 370 | Unstow package of antenna element electrical connectors | 5 | 963 | 598 | Identical tasks as Astronaut 1 |
| 371 | Release worksite anchors | 1 | 964 | 599 | Identical tasks as Astronaut 1 |
| 372 | Maneuver with package | 5 | 969 | 604 | Identical tasks as Astronaut 1 |
| 373 | Anchor to worksite | 2 | 971 | 606 | Identical tasks as Astronaut 1 |
| 374 | Anchor to package in general work area | 2 | 973 | 608 | Identical tasks as Astronaut 1 |
| 375 | Unstow electrical connectors | 5 | 978 | 613 | Identical tasks as Astronaut 1 |
| 376 | Release worksite anchor | 1 | 979 | 614 | Identical tasks as Astronaut 1 |

Table 5-15. Time Line Analysis, Contd

| EVENT | ASTRONAUT 1 | TIME (min) EVENT | CUMULATIVE TIME | CUMULATIVE EVA TIME | ASTRONAUT 2 |
|---|---|---|---|---|---|
| 377 | Maneuver, anchor and install electrical connectors to active antenna elements (assume five can be connected on one trip). | 30 | 1009 | 644 | Identical tasks as Astronaut 1 |
| 378 | Return to package, anchor, unstow electrical connectors and release anchors | 8 | 1017 | 652 | Identical tasks as Astronaut 1 |
| 379-384 | Repeat Events 377-378 for three more elements groups, returning to the spacecraft instead of unstowing more connectors after the third group has been connected | 114 | 1131 | 766 | Identical tasks as Astronaut 1 |
| 385 | Enter spacecraft, close hatch, repressurize, and stow gear | 60 | 1191 | | Identical tasks as Astronaut 1<br>END OF DAY 3 |
| 386 | Perform pre-Eva | 40 | 1221 | | Identical tasks as Astronaut 1 |
| 387 | Depressurize spacecraft | 25 | 1246 | | Identical tasks as Astronaut 1 |
| 388 | Open hatch and exit | 30 | 1276 | | Identical tasks as Astronaut 1 |
| 389 | Maneuver to electrical package, anchor, unstow electrical connectors, and release anchors | 10 | 1286 | 776 | Identical tasks as Astronaut 1 |
| 390-397 | Repeat Events 377-378 for remaining connectors, maneuvering back to equipment stowage area after last group | 152 | 1438 | 928 | Identical tasks as Astronaut 1 |
| 398 | Anchor worksite | 2 | 1440 | 930 | Identical tasks as Astronaut 1 |
| 399 | Unstow attitude control module | 5 | 1445 | 935 | Identical tasks as Astronaut 1 |
| 400 | Release anchors | 1 | 1446 | 936 | Identical tasks as Astronaut 1 |
| 401 | Maneuver | 5 | 1451 | 941 | Identical tasks as Astronaut 1 |
| 402 | Anchor to worksite | 2 | 1453 | 943 | Identical tasks as Astronaut 1 |
| 403 | Install ACS module | 10 | 1463 | 953 | Identical tasks as Astronaut 1 |
| 404 | Release anchors | 1 | 1464 | 954 | Identical tasks as Astronaut 1 |
| 405 | Maneuver | 3 | 1467 | 957 | Identical tasks as Astronaut 1 |
| 406 | Anchor | 2 | 1469 | 959 | Identical tasks as Astronaut 1 |
| 407 | Unstow second ACS module | 5 | 1474 | 964 | Identical tasks as Astronaut 1 |
| 408-413 | Repeat events 400-405 for second ACS module | 22 | 1496 | 986 | Identical tasks as Astronaut 1 |
| 414 | Enter spacecraft, close hatch, repressurize, and stow gear | 60 | 1556 | | Identical tasks as Astronaut 1 |
| 415 | ACS tests | 30 | 1586 | | Identical tasks as Astronaut 1 |
| 416 | RF tests | 480+ | 2066+ | | Identical tasks as Astronaut 1 |

A task requiring careful study is the alignment required after the basic structure deployment and after installation of the active antenna elements. The EV crewman cannot use standard optical devices such as a transit to check structural alignment since his helmet will not allow him to properly position his eye against the eyepiece. His efficiency in using optical devices is further limited by the light filtering of his sun visor.

The main safety hazard is entanglement. This hazard could be reduced, as well as the overall EVA time, if the telescoping structural members could be self-locking. The entanglement problem is especially acute for the deployment of the antenna active elements since the EV crewman is required to maneuver within the "forest" of antenna elements to make electrical connections.

Increasing the size of the antenna to 200 ft greatly increases the EVA time required as the number of EV installed active elements is increased from approximately 77 to 500. Although the required EVA time is not directly proportional to the number of active elements, it is apparent that an increase of 6 to 7 times in the number of active elements will cause an excessive amount of EVA.

## 5.3 ANALYSIS OF NEW CONCEPTS

5.3.1 Concept 75 — Expandable Truss Parabolic Antenna. Previous studies have provided an expandable structure technique that permits the stowage of 150-ft antennas (and larger) within the limits of the Saturn-V, and makes stowage, launch and deployment of this antenna entirely feasible under the present state-of-the-art.

The concept shown in Figure 5-23 has outstanding potential in the field of large, rigid, lightweight space structures. Its expansion ratio and unit weight are exceeded only by inflatable structures, though it far exceeds the inflatable concepts in strength, rigidity, and shape accuracy ($\tau = 10^4$D RMS).

The use of the CSM, or equivalent, to provide attitude control is feasible but will require the addition of finer sensors to obtain the 0.05° pointing accuracy required.

Solar cell arrays (Concept 81, Section 6.3) are stowed below the truss structure. They are attached to the truss antenna by rotating struts and are actuated by linear actuators. Up to 4500 $ft^2$ of solar cells can be packaged in the volume below the antenna.

After launch by a Saturn-V, the antenna and solar cells are deployed automatically but require one astronaut for alignment and subsequent maintenance.

5.3.1.1 Structure and Mechanics. This concept is a space expandable parabolic reflector antenna constructed in accordance with Convair's space erectable antenna design (Reference NASW 1438).

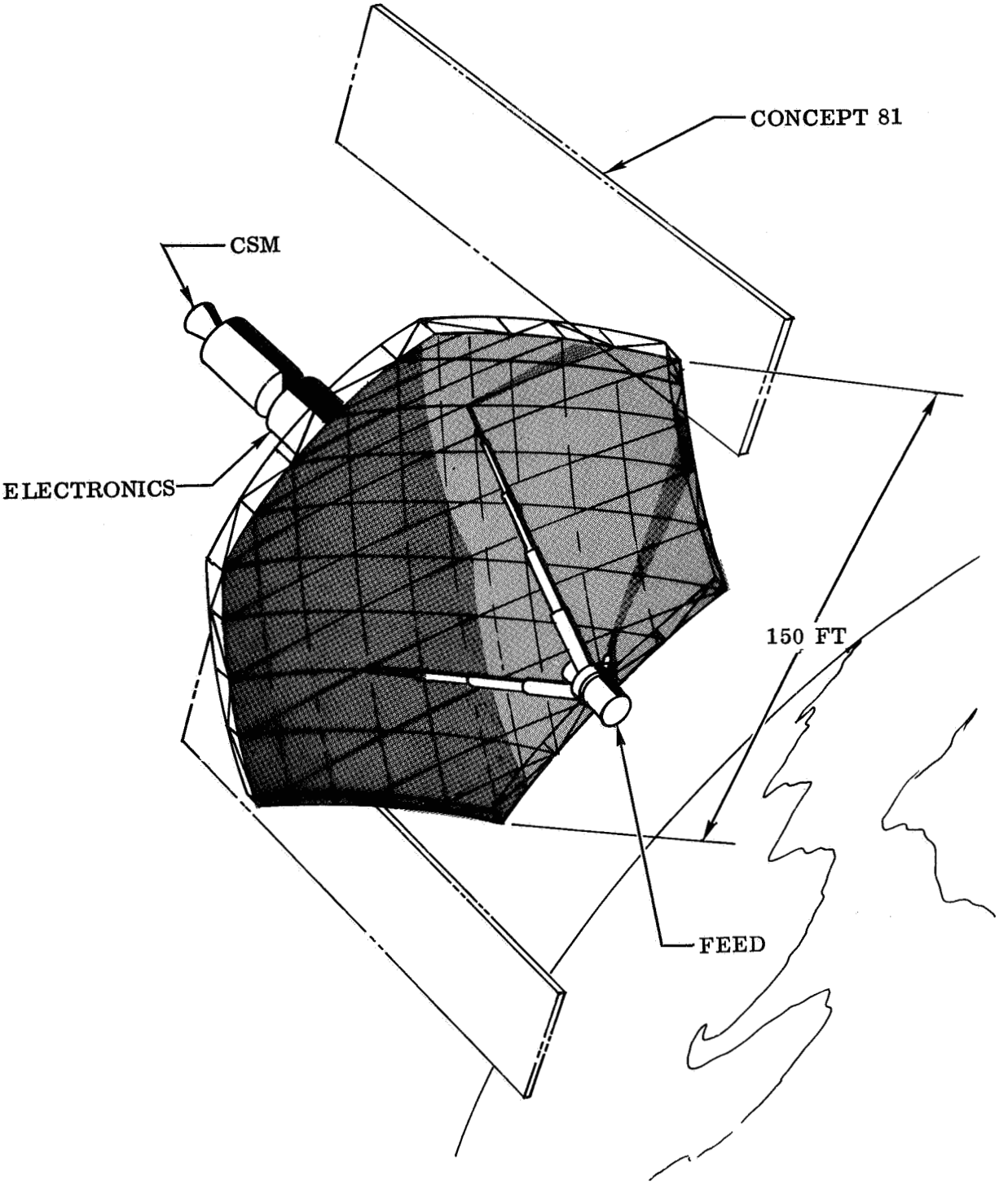



Figure 5-23. Concept 75, Expandable Truss Parabolic Antenna

The truss concept deploys into a three-dimensional, fully integrated, triangular truss structure. Inner and outer truss members are knee-jointed at the midpoints to permit folding. Intermediate (diagonal) members are rigid. All members are hinged at both ends. Built-in springs develop torques at knee joints to effect deployment. The wire mesh surface is installed as an integral part of the stowed package and unfurls with the deploying truss.

The solar cells are mounted on panels that automatically extend into two large paddles, each pivoted about a bearing located at diametrically opposite points on the perimeter of the truss.

5.3.1.2 Antenna Performance. In the "Feasibility Study of Large Space-Erectable Antenna" mid-term presentation (Reference NASW 1438), Convair shows the tolerances for surface accuracy in a 150-ft-dia. antenna to be: manufacturing 0.110 in. RMS, thermal 0.053 in. RMS, and dynamic 0.070 in. RMS — totaling 0.233 in. RMS. Optimistically assuming a tolerance of 0.2 in. RMS and the 150-ft-dia. parabolic, the anticipated performance is shown in Table 5-16.

Table 5-16. Performance of 150-Ft-Dia. Antenna with Surface Accuracy of 0.2 In.RMS

| GAIN | HALF-POWER BEAMWIDTH (deg) | FREQUENCY (Hz) | REMARKS |
|---|---|---|---|
| 58 | 0.16 | 2.8 G | Upper Operating Frequency |
| 57 | 0.21 | 2.2 G | S-Band Telemetry |
| 49 | 0.58 | 0.8 G | TV Broadcast |
| 30 | 5 | 90 M | FM Broadcast |

5.3.1.3 Attitude Control and Dynamics. The important environmental torques at synchronous orbit for this concept are due to solar pressure and gravity gradient. There will be a limited amount of maneuvering. Limit cycle operation is significant if large thrusters are used.

Gravity gradient torques are estimated using $2.38 \times 10^5$ slug-ft$^2$ for the two transverse inertias and $3.38 \times 10^5$ slug-ft$^2$ for the symmetry axis inertia. These values do not include the weight of the electrical supply, but do include a docked CSM. The steady state antenna attitude is assumed to be rotated from the local-vertical/orbital-plane coordinate system 5° in roll from the sun; this position gives maximum effective area and zero moment arm. As the antenna revolves 90° around the earth, the effective surface area gets smaller and the effective moment arm gets larger. The maximum pitch torque occurs between the two positions, and there is a non-zero minimum at 90°.

The maximum is estimated at $2.40 \times 10^{-2}$ ft-lb and the non-zero minimum at $6.78 \times 10^{-3}$ ft-lb. Average pitch torque is taken to be $1.68 \times 10^{-2}$ ft-lb.

The roll and yaw torques are estimated noting that the solar wind is 23° from the earth equatorial plane, which is the orbit plane. The maximum roll and yaw torques are $1.197 \times 10^{-2}$ and the average $5.98 \times 10^{-3}$ ft-lb.

The solar torque momentum change for half an orbit (12 hr) is 735 ft-lb-sec for pitch and 257 ft-lb-sec for roll or yaw. Yearly momentum change requirements are $5.31 \times 10^{5}$ ft-lb-sec for pitch and $1.89 \times 10^{5}$ for roll or yaw.

Attitude control for the duration of the mission is examined for two types of techniques: 1) using an ACS on the antenna itself, and 2) using the flight control system of the CSM. The second technique requires that special provision be made for antenna stabilization during docking. This can be done by docking while the antenna is still fastened to the launch vehicle, or by placing a limited capacity ACS on the antenna. This second approach removes the weight of the flight control system from the antenna launch vehicle.

Maneuvering for the missions associated with this concept is assumed to be limited, and the propellant requirements should be low.

Limit cycle operation is feasible with either control technique, but propellant consumption will be much larger for the CSM because of the high thrust value. Consumption is calculated using $2.38 \times 10^{5}$ slug-ft$^2$ for inertia, 0.1° for angular tolerance, and 300 sec for Isp. The antenna ACS assumes rim mounted jets with 0.1 lb thrust and 0.010 sec on time. The CSM is assumed to have 80 lb thrust and 0.020 sec on time. Single axis control uses $1.207 \times 10^{-10}$ lb of propellant per sec for antenna ACS and $2.87 \times 10^{-4}$ lb per sec for the CSM. Two axes require this control.

The yearly propellant expenditures for each technique is shown in Table 5-17. The limit cycle values clearly show that momentum devices are mandatory with the CSM, but are not required by the antenna mounted ACS. These devies will satisfactorily handle the limit cycle requirements for the CSM and also may be used for the oscillatory portion of the environmental torques.

A potential capacity problem exists for the solar pitch torque. The momentum capacity of current devices is 1000 ft-lb-sec, while the pitch torque integrated over half an orbit is estimated at 735 ft-lb-sec. An increase of one-third in this estimate would preclude using momentum storage devices to reduce the solar torque pitch propellant requirements. Fortunately, feasibility is not dependent upon this propellant reduction.

5.3.1.4 Stress Analysis. Convair has performed several studies of structures of this type with diameters up to 300 ft. In general, it has been concluded that unrealistically high values of $L/\rho$ and D/t of the tubular elements will satisfy the structural requirements.

Table 5-17. Annual Propellant Requirements

| | PROPELLANTS (lb) | |
|---|---|---|
| | ANTENNA MOUNTED | CSM |
| Solar Torque* | 40.5 | 433 |
| Gravity Torque | Significantly smaller than solar | |
| Maneuvering | Assumed to be small | |
| Limit Cycle** | 0.076 | 18,100 |
| Total | 40.6 | 18,533 |

* Can be reduced by momentum devices

** Can be virtually eliminated by momentum devices

Accordingly, a maximum value of $L/\rho = 150$ and minimum wall thickness of 0.020 in. has been used for design of the tubular elements.

The weight of the proposed experiment has been determined by scaling the results shown in Convair report GDC-PIN-66-708 assuming the same wall thickness and increasing linear dimensions by a factor of 3. With these assumptions, effects of thermal distortion are increased as a ratio of aperture diameter to the third power. However, these effects have been proven small and the surface distortion will be satisfactory.

Similarly, the natural frequency will decrease. Studies of a similar design with 100-ft aperture have shown a natural frequency of approximately 1.5 Hz, including coupling effects of the feed structure. Since the effective thickness can be increased with a small weight increase, a 150-ft antenna could have a frequency of approximately 1.0 Hz.

5.3.1.5 Weights. A weight breakdown for Concept 75, including power supply, is shown in Table 5-18.

Table 5-18. Concept 75 Weights

| | WEIGHT (lb) |
|---|---|
| Reflective Surface | 375 |
| Docking Ring | 150 |
| Inner Strut | 1,080 |

Table 5-18. Concept 75 Weights, Contd

| | WEIGHT (lb) |
|---|---|
| Outer Strut | 830 |
| Spiders | 211 |
| Feed | 100 |
| Feed Support Structure | 300 |
| Cables | 40 |
| Diag. Strut | 684 |
| Contingency 5% | 188 |
| Antenna Total | 3,958 |
| Power Supply (Concept 81) | 3,238 |
| Attitude Control | 433 |
| TOTAL | 7,629 |
| Available Payload (Saturn-V to Sync. Orbit) | ( 23,992) |

5.3.1.6 Crew and Equipment Functions. With reference to Figure 5-24, the automatic deployment is as follows. The stacked arrays are deployed by the action of the expanding truss antenna and slide outboard to Position A, where the actuator rotates the stack to Position B, then the stacks unfold into Position C by release of stored spring energy at the hinges. The feed is also positioned by the action of the expanding truss system and telescoped and locked into Position D. All motions of expanding antenna, arrays, and feed are switched controlled, i.e., signals are generated and monitored for positive identification of open, closed, and locked.

All elements of this concept are completely aligned before stowage for launch. After deployment the astronaut inspects and checks out the structure. Alignment and checkout of the reflective surface is also accomplished most feasibly by an astronaut using one of the various methods (Reference Table 5-19) that have been proposed. The following EV tasks are required once the antenna is automatically deployed:

a. Inspection of the antenna components.

b. Adjustments for antenna feed and control.

c. Repair or replacement of any damaged antenna components.

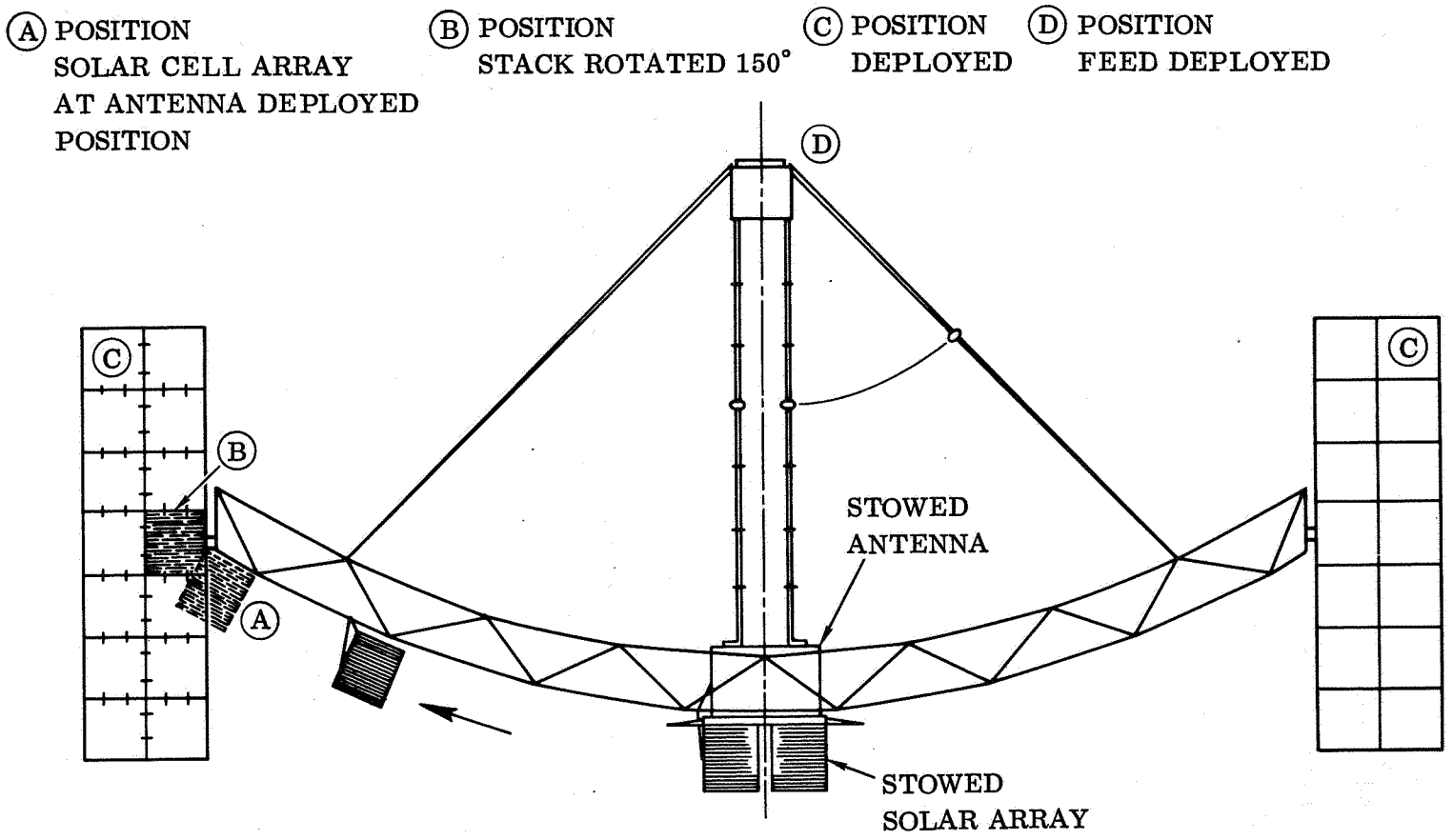
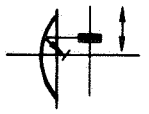
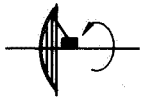
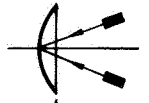
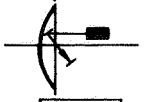
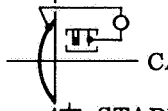
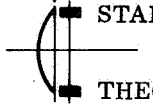
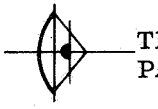
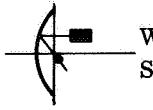



Figure 5-24. Automatic Deployment of Concept 75

Table 5-19. Method of Reflective Surface Checkout

| METHOD | ARRANGEMENT | TYPE OF MEASUREMENT | ADVANTAGE | DISADVANTAGE |
|---|---|---|---|---|
| Optical White-Light Scanning |  RAIL | Analog representation of antenna contour by light intensity difference | Simple principle (reflected light) | Tolerance of mechanical equipment not fine |
| Optical Scan Using Predrawn Reference Lines |  PERISCOPE | Analog | Normal functions of antenna not perturbed during readout | Area resolved is confined to number of reference lines |
| Stereo Camera Scanning |  CAMERA | Film | Fair resolution 0.01 for 50 ft antenna | Time consuming film processes and comparisons with index |
| Laser Interferometer |  ROTATE | Laser beam analog | Highly accurate | High voltages to operate |
| RF Interferometer |  RF SOURCE | Analog | Moderate voltages | Less accuracy. Distortions in arm cause errors in readout |
| Hologram Laser Light |  SEMI-MIRROR | Film | 3-dimension obtained | Extremely slow, one small area can be measured at one time |
| Ionization Capacitance |  CAPACITANCE | Ionization of gas in electric field $I \propto EF$ | High accuracy | High voltages |
| Optical Survey |  STADIA THEODOLITE | Optical Mirror | Simple concept | Requires EVA participation. Each readout must be calculated |
| Moire Pattern |  TRANSPARENT PARABOLA | Optical coupled with pattern inscribed on antenna, film readout | Remote control is easier | Film process is slow, all points must be calculated |
| Optical White Light |  WHITE-LIGHT SOURCE | Photo cells | Simple principal. Can pin-point deformation. Power required is low | Heavy weight requires highly reflective surfaces |

The inspection task is quite extensive due to the numerous components, hinges, joints, etc. A flagging such as a bright paint to indicate that a joint is not locked in place is of great assistance to the astronaut during the inspection. The time allowed for the inspection is approximately 60 min which does not include any repair or replacement of antenna components.

The adjustments for the antenna feed and control is another variable but this task should not take more than 60 min.

The above estimates do not include the necessary pre-EV operations such as suit donning, cabin depressurization, etc. The time required for these tasks would total approximately 90 min prior to the EV task and approximately 60 min for post-EV operations.

To accomplish the inspection, adjustment, and repair task, the following support equipment is required:

a. An extravehicular life support system (ELSS).

b. An astronaut maneuvering unit (AMU).

c. Portable worksite anchors.

d. Fixed and/or portable illumination system.

e. Fixed and/or portable power supply system.

f. A communication system.

g. The necessary adjustment and repair tools.

The ELSS, including an EV space suit should be compatible with an AMU. This support equipment should be able to support the EV crewman for a period of approximately 4 hr, the maximum time that he can function outside of the parent craft for any given day.

Fixed and/or portable hand holds on the surface of the structure are necessary.

Worksite lighting is required not only for darkside operations but whenever work is required in the shadow of the antenna components. This lighting could consist of a helmet-mounted or some other portable lighting system. The helmet-mounted lights are preferred to minimize tool or limb interference with the illumination.

A power supply system is required to provide power for the EV crewman's tools and equipment, illumination system, telemetry, voice communication, environmental control, etc.

A tool kit is provided to retain the tools when transferring between worksites with a temporary holding device at the worksite.

5.3.2 Additional Concepts. The following antennas are furnished from Convair study on TV broadcast satellite systems (PIN No. 66-1044) and the space erectable antenna study (NAS W 1438). Evaluation of these concepts is provided in Section 9.

| CONCEPT | ANTENNA TYPE | DESCRIPTION |
|---|---|---|
| 100A | Trussed Parabolic 50-ft-Dia. | Convair Parabolic Truss Antenna with 2400 $ft^2$ Fixed Solar Array on Inflatable Substrate |
| 100B | Trussed Parabolic 50-ft-Dia. | Convair Portable Truss Antenna with 750 $ft^2$ Oriented Solar Cell Array |
| 100C (Figure 5-25) | Trussed Parabolic 50-ft-Dia. | Convair Parabolic Truss Antenna with 750 $ft^2$ Pegasus-Type Directed Solar Array |
| 101 (Figure 5-26) | Four 9-ft-dia. Octagonal Tubes. 88 ft Length | Kraus and Countrawound Helix |
| 102 (Figure 5-27) | Trussed Parabolic | Convair Parabolic Truss Antenna with 2100 $ft^2$ Fixed Solar Array and LEM Docking Cone |
| 103 (Figure 5-28) | Horn Phased Array | 3276 Horns on 18 Hexagonal Panels. 2500 $ft^2$ Fixed Cell Array with SSESM |
| 104 (Figure 5-29 | Petalated Parabolic | Array of Seven 301 Petalated Antennas with RTG Power and LEM Docking Cone |
| 105 (Figure 5-30) | 7-Dish Parabaloid | Single Dish Tolerance with Petaline Packaging |

Concepts 100A, B, and C utilize the same 50-ft antenna but differ in the solar cell array design. Concept 100A uses a drum type of array that is not oriented to the sun's rays but exposes 750 $ft^2$ of its 2400 $ft^2$ of cells to the sun at all time. Concept 100B is a 750 ft flat panel array that must be pointed to the sun. Concept 100C uses a Pegasus-type array.

Detailed descriptions of these concepts can be found in Convair PIN 67-108 and 67-341.

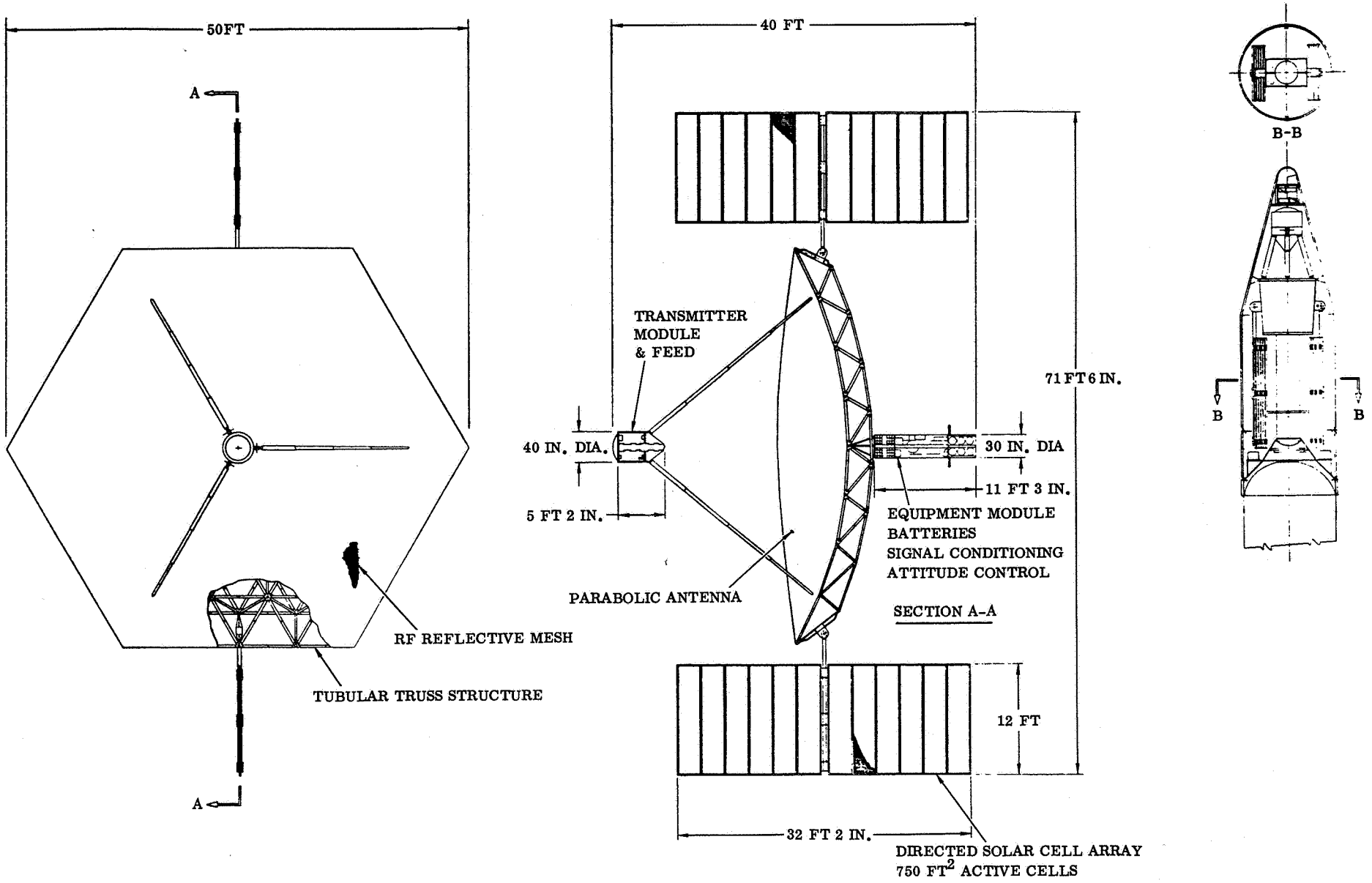
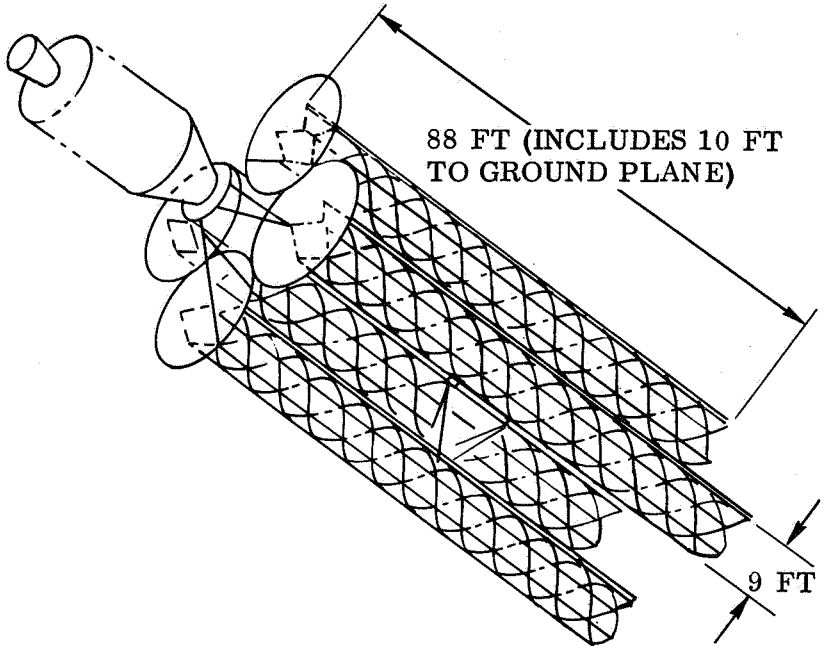



Figure 5-25. Concept 100C, TV Broadcast Satellite




NOTES:

| | |
|---|---|
| WEIGHT | 7,870 LB (INCLUDING MODIFIED SSESM) |
| STOWED VOL. | 3,600 $FT^3$ |
| EVA | INSPECTION |
| RF CAPABILITY | |
| BAND | 15-75 MHz |
| BEAM | $32^\circ$ -$14.5^\circ$ |
| GAIN | 15-22 db |

Figure 5-26. Concept 101, 88-Ft Helical Array Antenna (Convair)

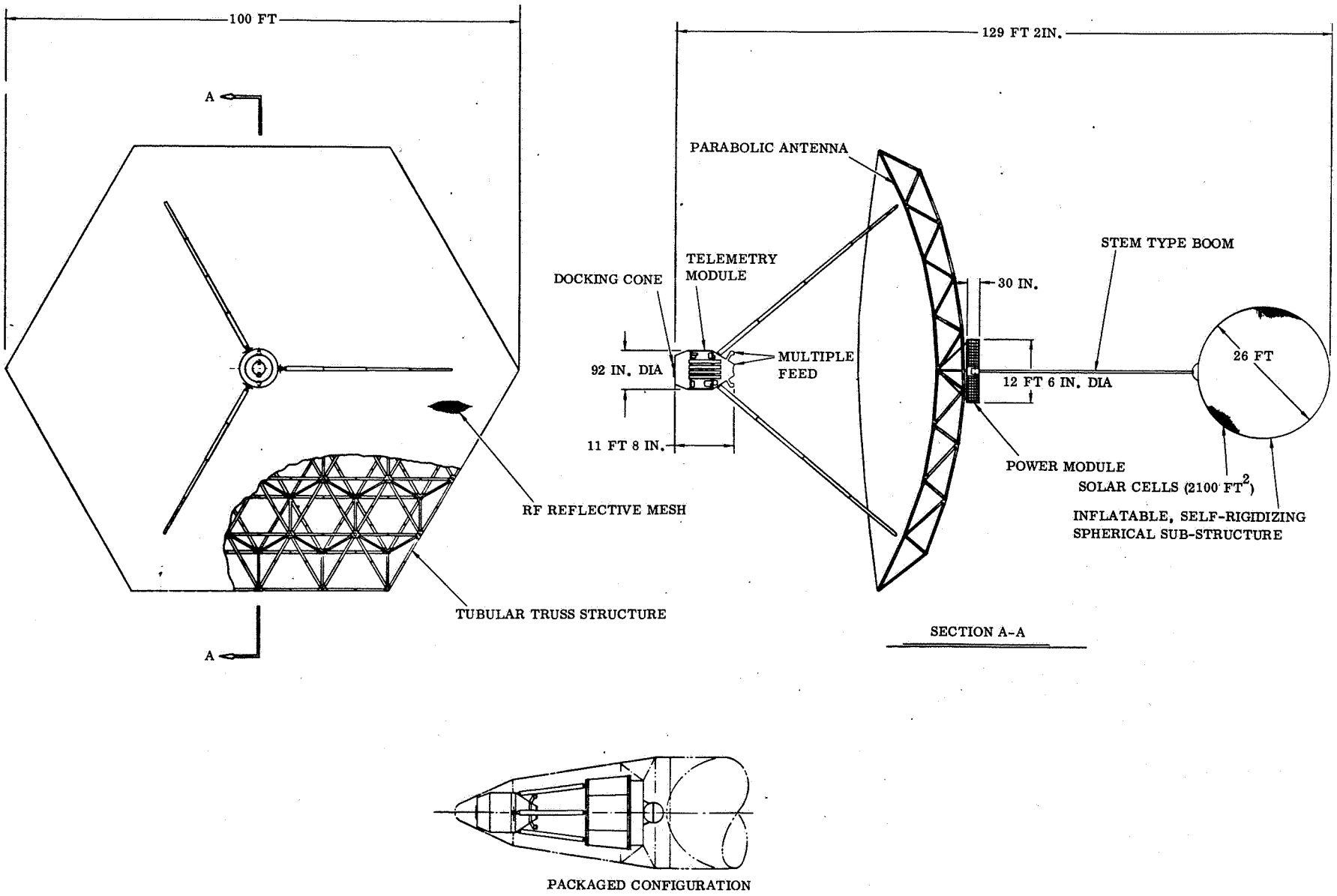
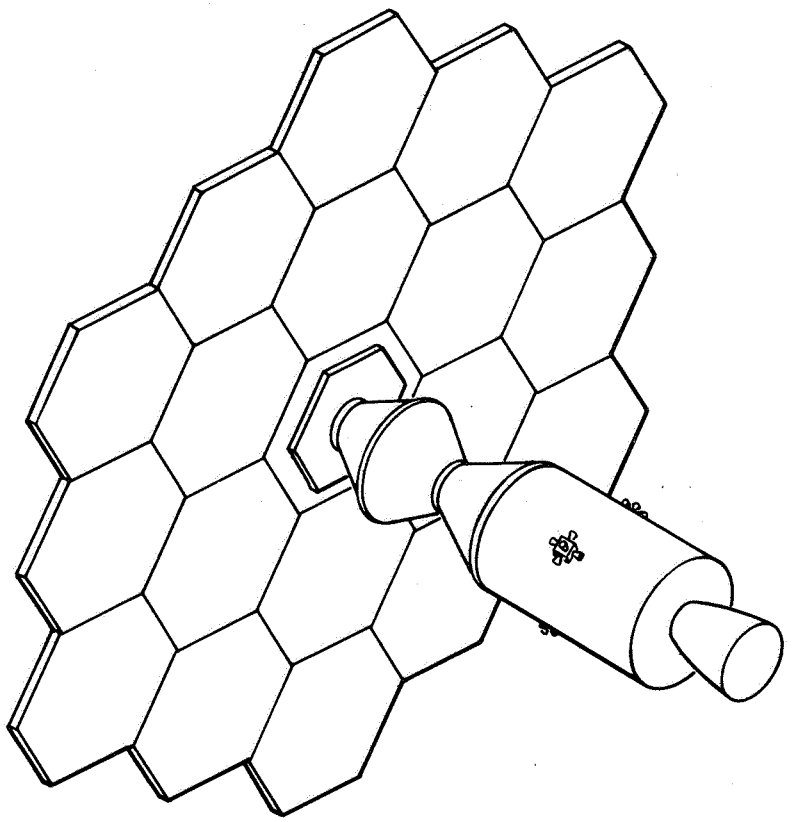


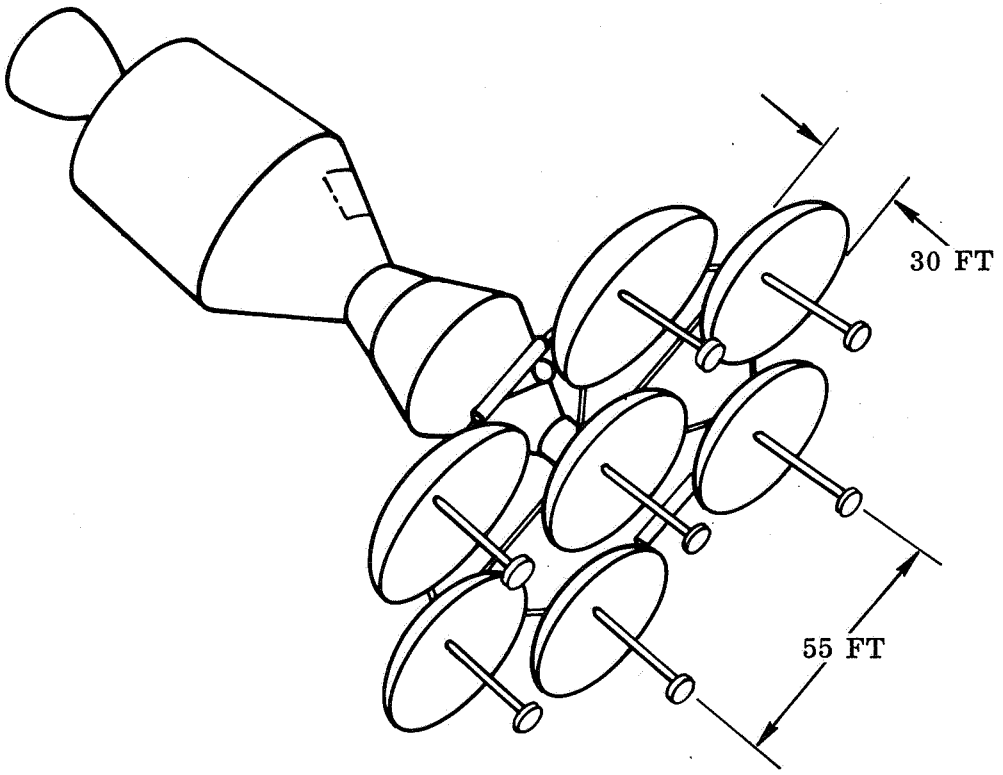

Figure 5-27. Concept 102, TV Broadcast Satellite for Saturn-V



NOTES:

| | |
|---|---|
| WEIGHT | 27,572 LB (INCLUDES SSESM) |
| STOWED VOL. | 5,450 $FT^3$ |
| EVA | INSPECTION |
| RF CAPABILITY | |
| BAND | 800-900 MHz |
| BEAM | $1.5^\circ \times 1.5^\circ$ |
| GAIN | 42.8 db |

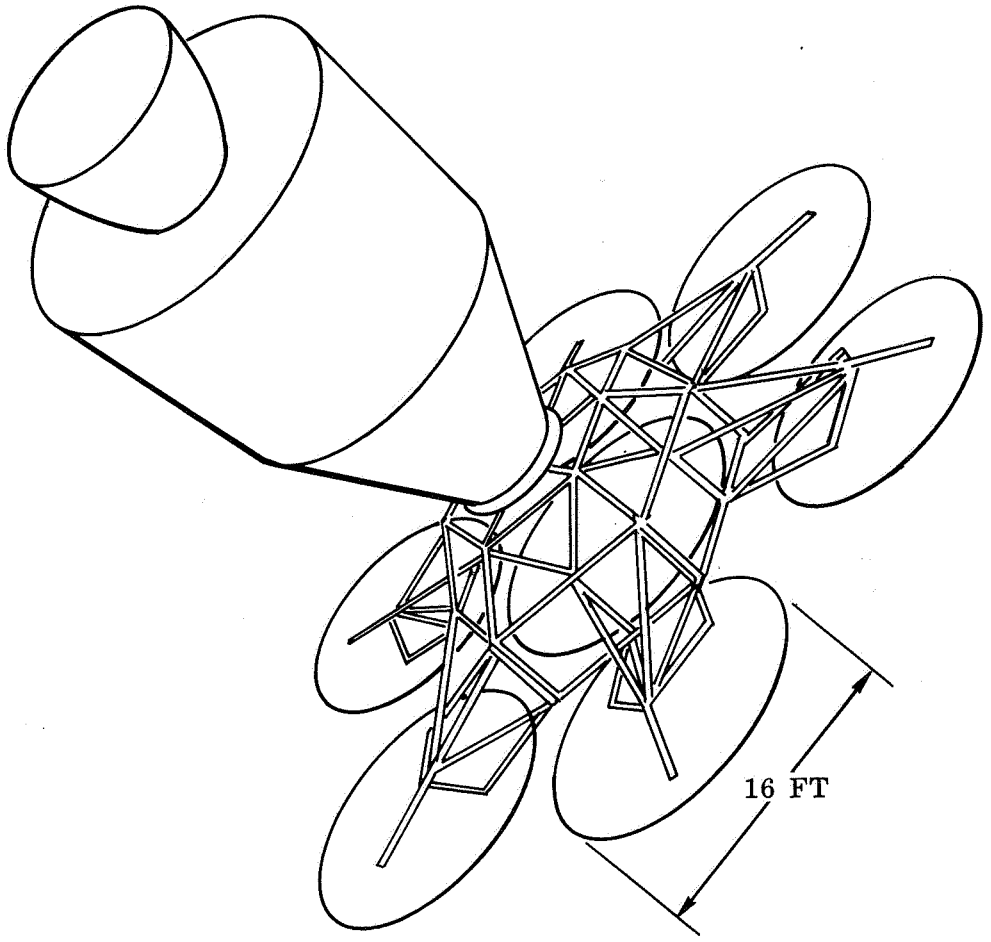
Figure 5-28. Concept 103, 18-Module Phased Array with 2500 $Ft^2$ Solar Array (Convair)




NOTES:

| | |
|---|---|
| WEIGHT | 12,625 LB (INCLUDES SSESM) |
| STOWED VOL. | 5,400 $FT^3$ (INCLUDES SSESM) |
| RF | |
| BEAM | 3-1/2° |
| BAND | 450-890 MHz |
| GAIN | 38.5 db AT 450 MHz<br>45 db AT 890 MHz |
| EVA | INSPECTION |

Figure 5-29. Concept 104, Seven Petalated Dish with Isotope Power (Convair)




NOTES:

| | |
|---|---|
| WEIGHT | 1,236 LB |
| STOWED VOL. | 4,500 $FT^3$ |
| FREQUENCY | 0.8-0.9 GHz |
| BEAM | 1.8° -1.9° |
| GAIN | 38.5 db AT 800 MHz<br>39.5 db AT 900 MHz |

Figure 5-30. Concept 105, Paraboloid Dish Array (Convair)

# SECTION 6

# SOLAR CELL ARRAYS

The results of the solar cell array study are summarized in this section. Deployed size and weights for provided and new concepts are summarized in Table 6-1. Table 6-2 summarizes concept performance.

6.1 EXPERIMENT USER REQUIREMENTS. Electrical power is required for 1) scientific electronics and 2) data and communications equipment. Scientific electronics includes experimental devices (receivers, sensors, actuators, signal generators) and associated instrumentation. Data and communications equipment consists of telemetry, programmers, command receivers, and recorders necessary to conduct experiments and collect and transmit scientific data to ground stations. Table 6-3 gives the best estimates of power requirements for Concepts 1 through 6. These requirements are based upon correlations of each concept with requirements for similar proposed AAP experiments. Also indicated are weight and approximate physical dimension.

System weights and areas shown in Table 6-3 must be derated for certain concepts to account for additional shadowing effects of large area structures. These concepts include earth- or lunar-oriented antennas where the geometric relationship of the satellite and the sun is varying throughout the orbit as opposed to those pointed toward distant objects beyond the solar system. Trade studies must be conducted to evaluate orientation penalties versus added solar cells required for non-oriented systems.

6.1.1 Communication Antenna Power. Power requirements for communications satellites may vary from a few watts to a megawatt or more. The power level depends to a great extent upon whether the mission is experimental or operational in nature. An experimental mission to test the feasibility of a concept can, with few exceptions, be supported with a few hundred watts supplied by a small (unoriented) solar cell array or radio isotope thermoelectric generator. On the other hand, if the same concept is upgraded to a fully operational system operating on a multichannel basis, the power requirement may be increased to the limits imposed by availability of power systems for the time period considered. This is demonstrated in Figure 6-1. These upper limits for power generation systems appear to be approximately 50 kw based upon the largest system now under study or development. These high-powered systems would take the form of either large solar cell arrays in excess of 5000 $ft^2$ of deployed area or Snap 8 reactor systems. The development of both reactors and high power solar cell arrays to produce this level of power for operational systems is still somewhat in question.

Solar cells for communication satellites have the advantages, over other candidate systems, of availability, life, and weight-assuming high orbit operation. Existing solar

Table 6-1. Task 1 Provided Concepts and Task 2 New Concepts

| CONCEPT | DESCRIPTION | ORIGINATOR | BOOSTER* CONFIGURATION | DEPLOYED DIMENSIONS (ft) | WEIGHT** (lb) |
|---|---|---|---|---|---|
| 5 | 5,000 $ft^2$ Array | Boeing | S-IB, C, & D<br>S-VA | (4) Panels<br>25 x 28 | 1960 |
| 8 | 5,000 $ft^2$ Array | Ryan | S-IB, A, C, & D<br>S-VA & B | (6) Panels<br>16 x 53 | 3060 |
| 19 | 160 $ft^2$ Array | Fairchild-Hiller | S-IB, A, B, C, & D<br>S-VA & B | (4) Panels<br>2 x 20 | 140 |
| 81 | 3,000 $ft^2$ Array | Convair | S-IB, A & C<br>S-VA | (2) Panels<br>28 x 54 | 3238 |
| 82 | 3,000 $ft^2$ Array | Convair | S-IB, A, C, & D<br>S-VA, & B | (4) Panels<br>17 x 47 | 4312 |

*A Manned/SLA

B Manned/SSESM/SLA

C Unmanned/SLA/Nosecone

D Unmanned/SSESM/Nosecone

**Includes cell components, substructure, and mechanisms. Does not include batteries, conditioning equipment, and cabling.

Table 6-2. Summary of Solar Array Concept Performance

| FACTOR | CONCEPT NO. 5 | 8 | 19 | 81 | 82 |
|---|---|---|---|---|---|
| Gross Area, $ft^2$ | 5,000 | 5,000 | 160 | 3,000 | 3,080 |
| Net Area, $ft^2$ | 4,433 | 3,500 | 147 | 2,700 | 2,770 |
| Gross Generated Power, kw | 47.7 | 15.8 | 1.7 | 30 | 27 |
| Specific Output, watts/$ft^2$ | 10.8 | 4.5 | 10 | 11.1 | 10.1 |
| watts/lb | 24.3 | 5.2 | 12.1 | 9.3 | 6.3 |
| *Gross Weight | 1,960 | 3,060 | 140 | 3,238 | 4,312 |
| Stowed Volume, $ft^3$ | 3,832 | 1,400 | 150 | 1,800 | 1,400 |
| Orbit Lifetime, yr | >1 | >1 | >1 | >1 | >1 |
| Orbit Altitude | 200 n.mi. Sync | 200 n.mi. Sync | 200 n.mi. Sync | 200 n.mi. Sync | 200 n.mi. Sync |
| Surface Error Tolerance | | | ±0.03 | | |
| Target Locked on Time | Cont | Cont | Cont | Cont | Cont |
| Structural Feasibility | Good | Good | Good | Good | Good |
| Mechanical Feasibility | Satis-factory | Good | Good | Good | Good |
| Required Pointing Accuracy | <±15° | <±15° | <±15° | <±15° | <±15° |
| EVA Hours to Position/ Deploy | N.A. | 9 | N.A. | Not Req'd | Not Req'd |
| Reliability Rating | 0.99700 | 0.99681 | 0.99489 | | 0.99600 |

*Does not include batteries, conditioning equipment, or cabling.

Table 6-3. Summary of Estimated Power Penalties for Experiment Operations Assuming Solar Cell Power Supply

| CONCEPT | POWER (watts) | DURATION (years) | SYSTEM WEIGHT PENALTY (lb) | ARRAY AREA ($ft^2$) |
|---|---|---|---|---|
| 1 | 160 | 1+ | 80 | 59 |
| 2 | 100 | 1+ | 53 | 37 |
| 3 | 100 | 1+ | 53 | 37 |
| 4 | 30 | 1+ | 16 | 12 |
| 5 | | | | |
| 6 | 100 | 2 | 53 | 37 |

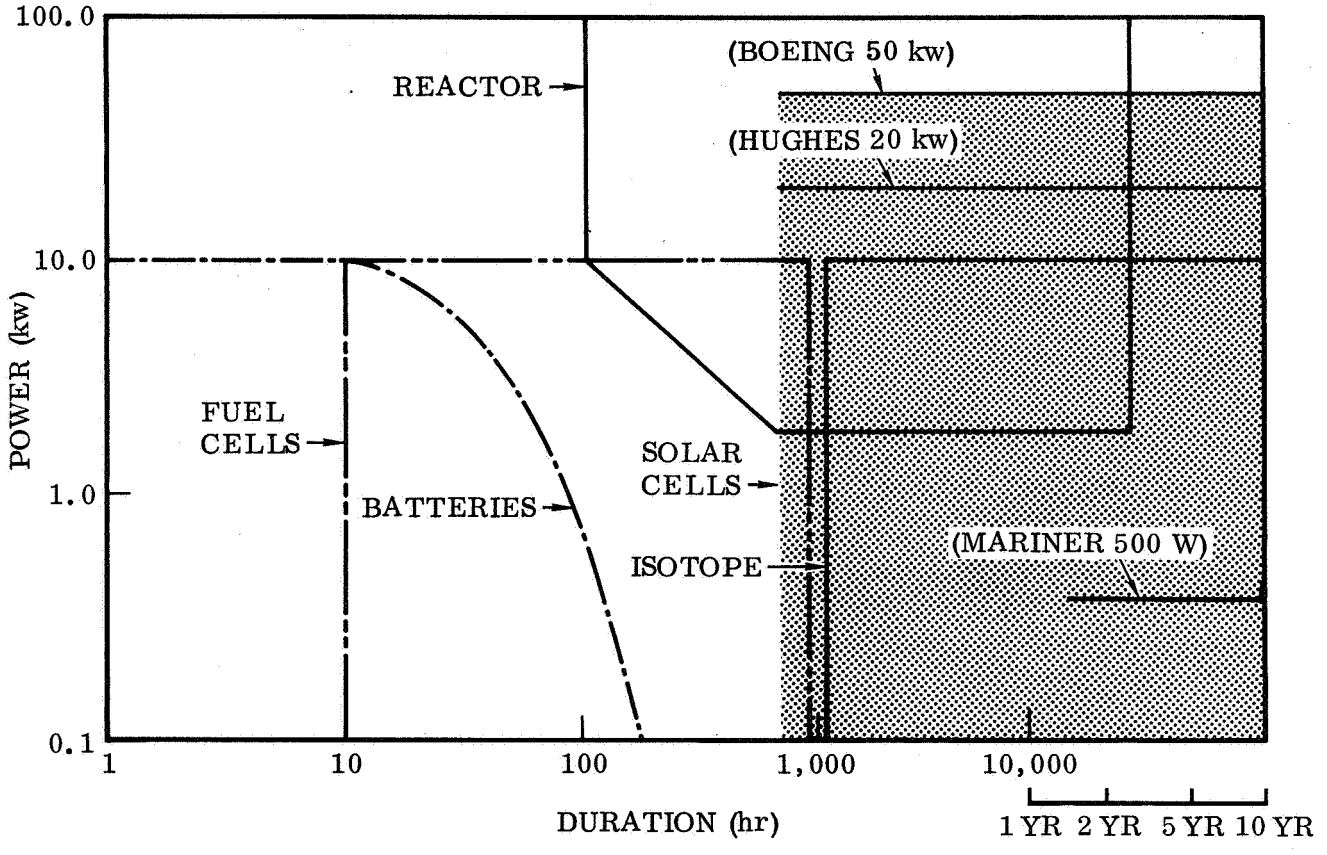



Figure 6-1. Power System Application

cells are being manufactured in large quantities with efficiencies better than 11%, yielding overall power-to-area ratios approaching 10 watts/ft$^2$.

With expected improvements in substrate structural designs using 8 to 10 mil silicon cells and 1 to 3 mil cover glasses, a 15-25 watt/lb solar array is feasible. Thin film single crystal systems are expected to yield an ultimate of 50 watt/lb in an optimum system.

### 6.1.2 Design Requirements Data

a. The mission profile and objectives determine the characteristics of gross power, size, weight, and deployment configuration.

b. Use of 1967 state-of-the-art components, materials, and processes for structures and mechanisms.

c. The deployed array shall be designed to stiffness requirements which preclude coupling with the spacecraft guidance and control system.

d. Structural enter-ties between the spacecraft and array canisters must be held to a minimum to keep arrays isolated as far as practical from amplified structural dynamic loadings during launch.

e. Structural degradation shall not occur due to conditions of transient thermal stresses due to repeated entry into, dwell time within, and emergence from, earth or spacecraft shadow; also, the thermal gradients that develop between the sun and shade sides of the array.

f. The array shall be capable of withstanding the accelerations due to docking impacts, astronaut impacts, and spacecraft maneuvers. Specific values are related to mission requirements.

g. In the deployed configuration, under steady-state conditions, the solar array must have sufficient rigidity to maintain sun plane orientation within $\pm 10°$. This tolerance shall include deflections from static spacecraft load inputs and thermal gradients, but shall not include deflections due to dynamic load inputs.

h. All mechanisms shall function structurally for the duration of the mission.

i. Lifetime: one year.

j. The deployment mechanism must be capable of operation in the 1g earth environment for a minimum of 10 operating cycles without failure or replacement. Assist fixtures to negate 1g effects on the extending members is permissible.

k. Replacement capability of panels is subject to specific definition of the mission requirements.

l. Power levels anticipated: over 10 kw.

m. Orbit altitudes: 200 n.mi. to synchronous.

## 6.2 ANALYSIS OF PROVIDED CONCEPTS

6.2.1 Concept 5 - Large Solar Cell Array (Boeing Co.). Two solar cell array concepts were matched to two different boosters for the primary purpose of supplying direct power to electric propulsion engines on unmanned spacecraft enroute to a Mars orbit. They also supplied all other electric power. The performance goal was 20 watts of electrical power per pound of array weight at a sun probe distance of one astronomical unit.

The primary purpose of the program was to evaluate the feasibility of fabrication for large, deployable, solar voltaic arrays in the range of 3 to 50 kilowatts power based on 1966 technology. Operational time period is 1971 to 1975. The selected array characteristics are given in Table 6-4.

Table 6-4. Array Characteristics

| | 50 kw SYSTEM | 10 kw SYSTEM |
|---|---|---|
| Booster | Saturn IB/Centaur | Atlas/Centaur |
| Gross Area | 4,944 $ft^2$ | 1,004 $ft^2$ |
| Cell Area | 4,433 $ft^2$ | 930 $ft^2$ |
| Array Weight | 1,960 lb | 518 lb |
| Electrical Yield @ 1 AU | 47.7 kw | 10.18 kw |
| Power per Unit Area | 10.8 watts/$ft^2$ | 10.8 watts/$ft^2$ |

When deployed, the array consists of four solar panel assemblies in a cruciform pattern. Each solar panel consists of a series of hinged sub-assemblies. Each sub-panel assembly contains a main sub-panel and two hinged auxiliary sub-panels. See Figure 6-2.

The reference document is Boeing Co. Report D2-23942-5.

6.2.1.1 Structure and Mechanics. The basic structure of each panel consists of rectangular frames of bonded beryllium with stretched and woven fiberglass tape mats over one face of the frame. The tapes directly support 8-mil solar cells having 4-mil cover glasses.

The deployment scheme consists of a motor-driven, two-stage harmonic drive high reduction (21,333:1) winch employing a Dacron cable routed to Delrin sectors at the main panel hinges. This system deploys the six radially aligned main panels of any one quadrant. Each main panel from 2 through 5 contains two auxiliary panels each. These

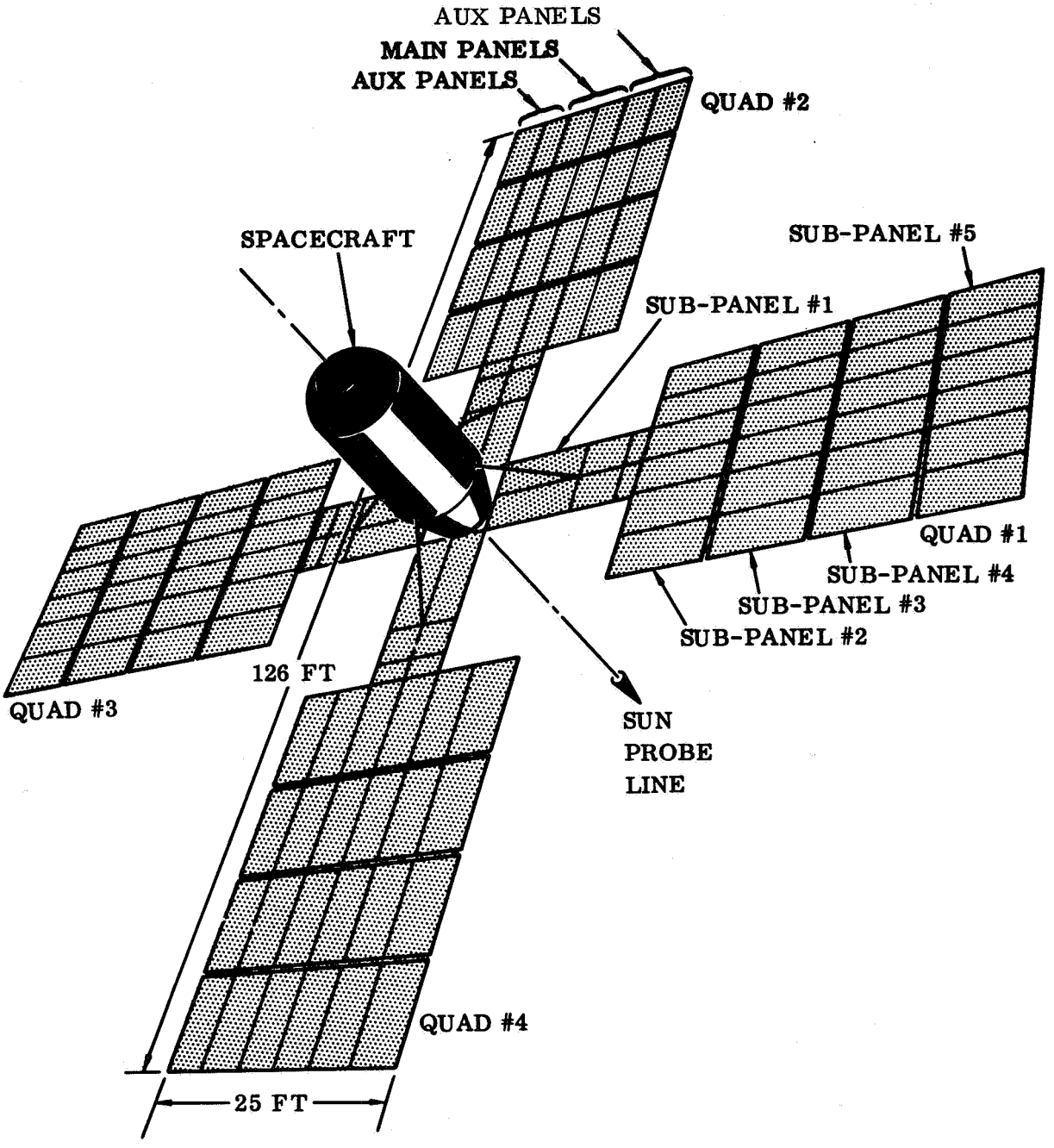



Figure 6-2. Boeing Large Solar Cell Array

are deployed by torsion springs. The main panels are snubbed and secured by combination viscous dampers and bayonet locks. Over-center, self-energizing locks secure each hinged set of auxiliary panels.

The deployment sequence takes 14 minutes and each of the 4 quadrants of the cruciform deploy in an identical and concurrent fashion. The deployment events within a quadrant follow a rather peculiar sequence pattern in that some auxiliary sub-panels deploy ahead of further outboard main panels. This is due to the progressive release sequence wherein releases are a result of the prior locking action of an adjacent sub-panel. The characteristics of the presented design are the particular product of the original mission, design constraints, and ground rules. The primary factors that fashioned the configuration are:

a. 20 watt/lb design goal - direct power to ion engine

b. 1966 technology - 1971-75 operational flight period

c. Sun oriented spacecraft

d. Unmanned spacecraft

e. Launch envelope set by booster shroud and spacecraft

f. Very low attitude correction moments.

The Boeing design concept is an optimized system designed to requirements of a specific mission. It is unique in that the generated electricity is the fuel for an ion engine. This mission is so totally removed from the manned earth orbit AAP missions that a useful evaluation is difficult.

The very high power-to-weight ratio is well beyond 1966 technology earth orbit systems (2 to 4 watts/lb for total system weight including batteries, converters, and cabling). Omitting the batteries, converters, and cabling, the present conventional systems still have a yield considerably less at about 6.5 watts/lb. This discrepancy can be attributed largely to the specific end use.

In view of the above observation, it can be expected that the earth-orbiting arrays as a system can vary grossly from the configurations conceived by the Boeing Co.

Fabrication and Availability. The structural fabrication techniques are of the present state of the art, but at the extreme end. Beryllium fabrication is in its infancy and it must be expected that unforeseen pitfalls will develop. The Boeing design philosophy seems rather optimistic in this regard. The frame construction is adaptable to modules of varying sizes, a definite merit in applying these construction principles to configurations peculiar to the AAP missions. Cell availability and cost, however, are likely to favor the concentrator type of construction at a sacrifice in array size, weight, and package volume. It is considered that if specific applications are delineated for earth orbit, the concentrator construction must be reviewed.

The composite materials also warrant consideration in material tradeoffs. Such composites as aluminum/boron have an $E_c$ value of 83% of beryllium and $E_c$/density ratio of 0.51 that of beryllium.

Deployment and Mechanisms. The deployment scheme is an application of similar techniques used and proven in aircraft design. Again, when applied to a specific mission where the configuration can be grossly affected, an entirely different mechanism may be warranted. The system in principle is applicable for earth orbital missions, although some exceptions or alternatives may exist in detail. It is rather trite to suggest that simplication should be a design objective; nevertheless, the deployment can be intuitively assessed as complicated and needing simplification.

There are some areas of detail that warrant investigation and it is felt that improvement of the mechanical system does exist in detail. For instance, Mylar cable should be considered where Dacron is now used. Mylar, of course, has been used extensively in space applications and is now available in rope or cable forms. The Mylar cable would have considerably less stretch than Dacron with even better pliable characteristics. See Figure 6-3.

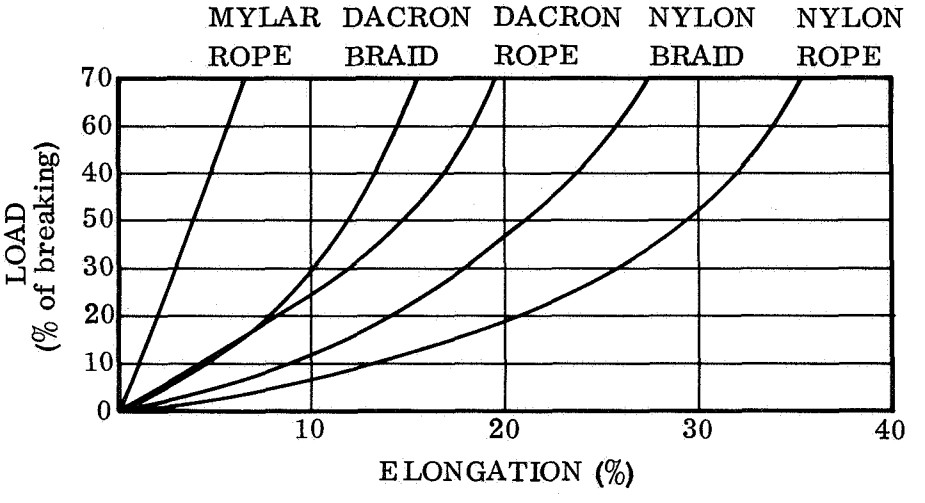



Figure 6-3. Cable Elongation Comparisons

Overcenter latches are used as the retaining latches for the auxiliary panels. In this application the principle is appropriate since they are essentially self tripping. However, the design in detail requires refinement. The existing geometry is very sensitive to accumulative tolerances at the stowed position. On release of the panel lock, initial motion can result in carrying the tension link past overcenter position before the compression link begins travel. Toggle action will occur in reverse and the panels will be

locked at about 20° open in lieu of the necessary 90°. The moving joints employ a drag joint to negate this tendency but a geometric refinement is necessary to better the margin. See Figure 6-4.

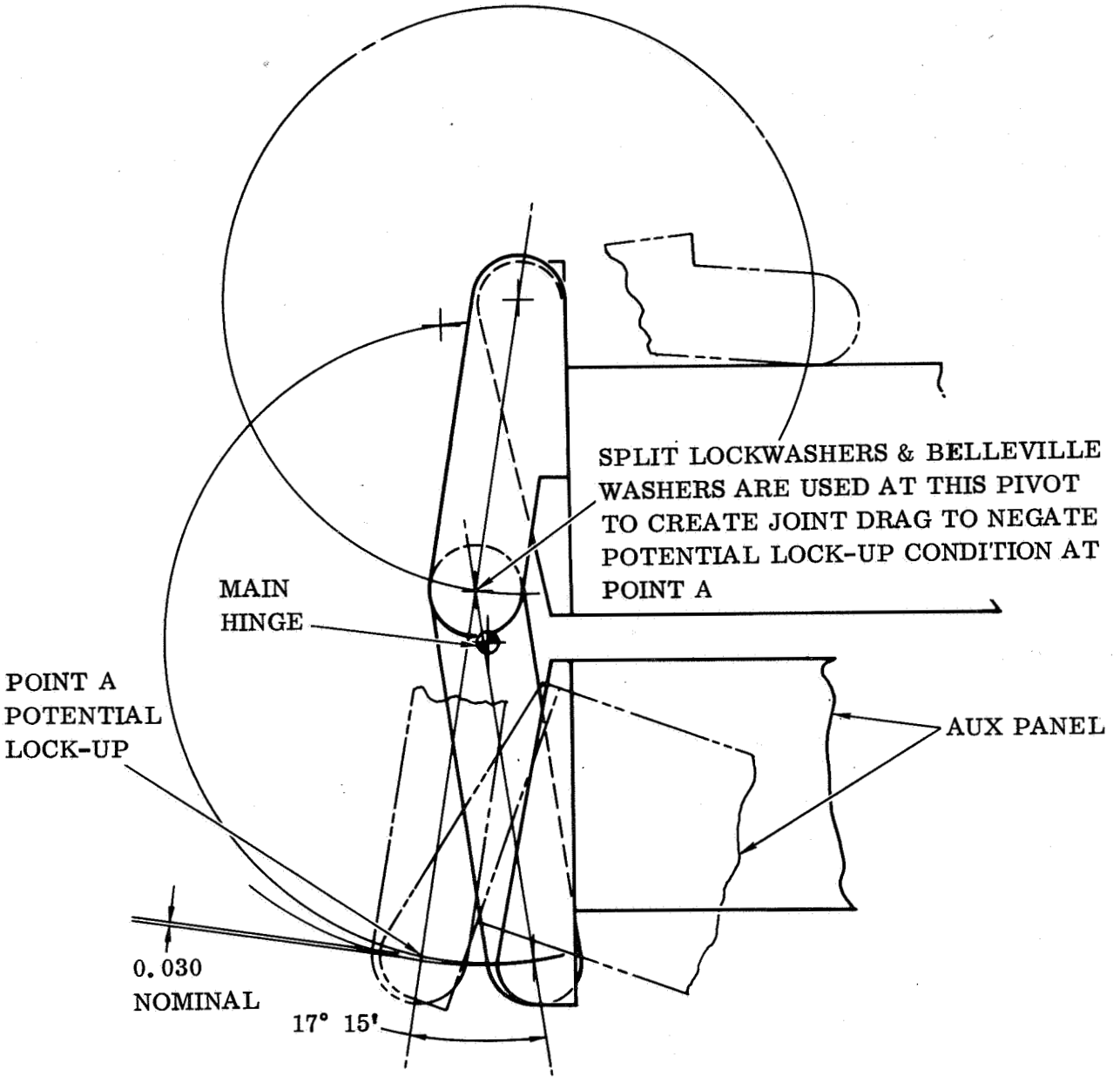



Figure 6-4. Overcenter Latch

The Boeing concept naturally does not include a requirement for EVA assist. Because of the flimsy nature of solar arrays, it is advisable that the system be nearly 100% automated, except that post deployment bracing could be applied via EVA in order to withstand the more rigorous slew rates and spacecraft maneuvers.

The Boeing concept per se is not appropriate for the AAP missions except as an experiment in conjunction with an unmanned configuration. This configuration need not be a sole experiment of itself but can be associated with and supply power to other experiments. These experiments must not be of a type requiring precise pointing nor slew rates exceeding the very low design rates such as the $2 \times 10^{-5}$ rad/sec$^2$ in pitch. Structural beef-up is always a possibility to extend the capabilities of the array. These arrays are generally unsuited for applications to astronomy experiments with the exception of solar astronomy.

When considering arrays used as appendages to manned spacecraft, a specific configuration must be generated in which the principles only can be extracted from the Boeing design. Such systems will not perform at 20 watts per pound but more realistically at perhaps 10 to 15 watts per pound. Mechanism simplication can be expected although astronaut assist does not present significant advantages to increase the power-to-weight ratio.

The Boeing reliance upon beryllium construction was a necessity to meet the power/weight ratio necessary for the Mars type mission. This philosophy is extravagant for earth orbit missions and cost should be given a greater emphasis.

Recommendations

a. Consider application of the Boeing concept in its present form only to unmanned vehicles or experiments in which the prime pointing requirement is dictated by the solar array.

b. For missions requiring the arrays mounted to manned spacecraft or experiments requiring gimbaling, consideration must be given to complete redesign although the principles of construction and deployment of the Boeing concept are a point of beginning. It is advisable to include further investigation of the concentrator concepts and structural materials to reduce costs.

c. A final concept evaluation for a specific application should compare other concepts and principles as advanced and developed by other manufacturers.

Design Requirements Derived from the Evaluation

a. The mission profile and objectives determine the characteristics of gross power, size, weight, and deployment configuration. The specific requirements for structure and mechanisms cannot be defined until related to mission requirements.

b. Use of 1966 state-of-the-art components, materials, and processes.

c. The deployed array shall be designed to stiffness requirements which preclude coupling with the spacecraft guidance and control system.

d. Structural ties between the spacecraft and solar arrays must be held to a minimum to keep arrays isolated as far as practical from amplified spacecraft dynamic loadings.

e. Structural degradation shall not occur due to conditions of transient thermal stresses due to repeated entry into, dwell time within, and emergence from earth or spacecraft shadow; also, the thermal gradients that develop between the sun and shade sides of the array.

f. The array shall be capable of withstanding the accelerations due to docking impacts, astronaut impacts, and maneuvers. Specific values are related to mission requirements.

g. In the deployed configuration, under steady-state conditions, the solar array must have sufficient rigidity to maintain sun plane orientation within $\pm 10°$. This tolerance shall include deflections from static spacecraft load inputs and thermal gradients, but shall not include deflections due to dynamic load inputs.

h. All mechanisms shall function structurally for the duration of the mission after deployment.

i. The deployment mechanism must be capable of operating in the 1g earth environment for a minimum of 10 operating cycles without failure or replacement. Assist fixtures to negate 1g effects are permissible.

j. Modular replacement of panels is subject to specific definition of the mission requirements. Replacement is to be avoided.

6.2.1.2 Electrical Power. Power requirements for the large solar cell array experiment are unique in that the experiment itself is a large power generator. This suggests that this concept should be tested in conjunction with another experiment concept requiring high power levels. On the other hand, other power consuming experiments should not be totally dependent upon an experimental solar cell system for mission success. On this basis a redundant power supply must be considered which would satisfy the minimum requirements of the power consuming experiments in the event of unsatisfactory performance of the solar cell array.

It is assumed that the user has two basic objectives: proof of the feasibility of 1) fabricating, deploying, and orienting a large lightweight structure, and 2) assembly of silicon solar cells into an array with favorable electrical characteristics at very high power levels. These experiment requirements would require instrumentation, telemetry, and communications. It is anticipated that after initial deployment and checkout this power could be provided by the solar cell array itself. Instrumentation and telemetering requirements are a very small percentage of total array capacity. Power for initial activation could well be provided with batteries or fuel cells associated with other supporting spacecraft modules such as Apollo CSM or S-IVB workshop.

6.2.1.3 Attitude Control and Dynamics. Complexity of the attitude control is highly dependent on the maneuvering and pointing requirements of the space vehicle mission. Coupling between the solar panel vibration modes and control system frequencies becomes an increasing problem as the vehicle moves oftener and at higher vehicle rates, and as the pointing tolerances decrease. This concept was originally intended for a Mars mission. Interplanetary missions have slow low-rate maneuvers and coupling problems can be easily avoided by designing the solar panel nodal frequencies separated from control frequencies. Utilization of solar panels for earth orbit missions has led to serious coupling problems in several space station studies. It is assumed here that the vehicle maneuvers and pointing tolerances will be of such magnitudes that coupling can be precluded via frequency separation.

The external torques are greatly different for the two orbits considered for AAP. At 200 n.mi. aerodynamic drag torque and gravity-gradient are significant. Solar pressure and gravity-gradient are important at synchronous orbits. The configuration considered has the LEM Lab and CSM and the larger 4944 sq ft solar panels. Estimated inertias are 92,900 slug-ft$^2$ longitudinal and 140,000 slug-ft$^2$ transverse. The maximum pitch gravity-gradient torque is $7.75 \times 10^{-2}$ ft-lb at 200 n.mi. and $3.42 \times 10^{-4}$ ft-lb at synchronous. The aerodynamic drag at 200 n.mi. is 0.109 lb. The effective moment arm is 10 feet for both drag and solar force. The solar force is $4.77 \times 10^{-4}$ lb. The rotational impulse due to external torques for a one-year mission is shown in Table 6-5. Also shown are the impulse, assuming jet arm of 15 ft, and propellant weight, assuming $I_{sp}$ of 300.

Table 6-5. Solar Panel Vehicle Momentum and Propellant, External Torques

| | 200 n.mi. ORBIT | |
|---|---|---|
| | PITCH OR YAW | PITCH AND YAW |
| Rotational Impulse, ft-lb-sec | $9.75 \times 10^{-6}$ | $1.95 \times 10^{-7}$ |
| Impulse, lb-sec | $6.51 \times 10^{-5}$ | $1.30 \times 10^{-6}$ |
| Propellant Weight, lb | $2.17 \times 10^{-3}$ | $4.34 \times 10^{-3}$ |
| | SYNCHRONOUS ORBIT | |
| | PITCH OR YAW | PITCH AND YAW |
| Rotational Impulse, ft-lb-sec | $1.08 \times 10^{-5}$ | $2.16 \times 10^{-5}$ |
| Impulse, lb-sec | $7.17 \times 10^{-3}$ | $1.43 \times 10^{-4}$ |
| Propellant Weight, lb | 23.9 | 47.8 |

Vehicle maneuvering places additional requirements on the control system. They can be met by providing additional propellant or by introducing momentum storage devices. The tradeoffs involved will not be examined here. The feasibility of ACS of this type of vehicle is within the state of the art, provided coupling is precluded.

The problem of jet exhaust impingement on the solar cells noted by Boeing is an area which needs further study.

6.2.1.4 Stress. As noted in Boeing Report D2-23942-5, the critical structural environment is the resonant vibration encountered during launch. It would appear that the magnitude of the loading is excessive, since the effective g is:

$$g = \frac{23580}{1.121(960)} = 22g \text{ ultimate}$$

and the amplitude is:

$$z = \frac{22 \times 386}{(6.2)^2 (10.6)^2} = 2 \text{ in.}$$

However, the analysis presented is conventional and reasonable and shows the concept to be structurally feasible.

Due to the relatively long times used to deploy, inertia loads during deployment should be minimal. Inertia loads after deployment for the Mars mission are much smaller than predicted for earth orbit as previously discussed. Examination of the results of the thermal analysis presented on Page 192 of D2-23962-5 leads to the conclusion that thermal distortion is not a problem.

6.2.1.5 Weights. The weights published for both the 50 kw (1960 lb) and the 10 kw (518 lb) versions appear to be reasonable with no unusual assumptions.

With an orbital altitude of 200 n.mi., approximately 2/3 of the total launch weight of 6300 lb is comprised of attitude control propellant. This is within the payload capability of a Saturn IB providing a Hohmann transfer is used. At synchronous orbit the ACS propellant requirements drop to about 50 lb. This reduces the total launch weight to slightly over 2000 lb, or well within the payload acceptability of a Saturn V synchronous launch.

6.2.1.6 Materials. Beryllium, as recommended by Boeing, seems to be the obvious structural material to meet performance requirements for the Mars mission. However, the high cost of beryllium metal, and the cost and problems associated with fabricating beryllium in eight-foot lengths, makes the substitution of another material for beryllium desirable for the Earth orbit missions of this study.

One material that has good formability and a high modulus (near that of beryllium) is a composite of boron fibers in an aluminum matrix. A composite of 2024 aluminum alloy plus $50^{vol}/0$ of undirectionally aligned boron fibers (4 mil diameter) has a tensile yield strength of 175 ksi (parallel to the fiber axis) and 20 ksi (perpendicular to the fiber axis). The modulus for the composite is $35 \times 10^6$ psi, which is only slightly less than that of beryllium. An aluminum $50^{vol}/0$ composite material is now being fabricated in sheet form by the Harvey Aluminum Co.

The aluminum-boron composite material should be considered as a substitute for beryllium since it is doubtful that fabrication of large pieces of beryllium is feasible in the near future.

6.2.1.7 Evaluation Summary. The Boeing Company solar cell array concept is designed for a very high power to weight ratio (20 watts/lb). Two similar designs are presented for two different boosters with the primary purpose of supplying raw electrical energy to propulsion engines on unmanned spacecraft enroute to a Mars orbit. The program objective was to evaluate the feasibility of fabricating large deployable arrays of 3 to 50 kilowatts power based on 1966 technology.

The specific concept is unique in that the experiment itself is a large power generator; this suggests that the concept should be tested in conjunction with another experiment requiring high power levels.

The low design stress levels inherent with the original purpose requires structural beef-up to extend the capabilities of the arrays for Earth orbit purposes. The Boeing Company reliance upon extensive beryllium construction was a necessity to meet the power/weight ratio required for the Mars type mission. This philosophy is extravagant for manned Earth orbital missions unless the feasibility proof of beryllium construction is a paramount consideration.

Aerodynamic drag and gravity gradient dominate the 200 n.mi. orbit; torques are of the order of tenths of ft-lb and mass expulsion torquers are indicated. Solar pressure and gravity-gradient are dominant at synchronous orbit; torques are of the order of $10^{-4}$ ft-lb and the use of momentum storage devices with jet desaturation is indicated.

The concept is considered feasible and is an appropriate large space structure although considerable revision is warranted to adapt it to Earth-orbiting manned spacecraft. Astronaut assist does not present significant advantages to increase the power to weight ratio. The specific application will have a great influence upon the actual configuration, but in its present form this concept is best suited to unmanned vehicles or experiments in which the prime pointing requirement is dictated by the solar array.

6.2.2 Concept 8 - Ryan Deployable Solar Array. The large space structure is a deployable solar cell array concept which is applicable to arrays of very large size. Ryan's work on this concept is performed under contract to JPL. One prototype unit 3 ft x 18.5 ft is being fabricated.

The solar cells are supported on a flexible fiberglass substrate stowed on a "wrap-cylinder." A small electric motor and gear system rotate the wrap-cylinder to effect deployment and retraction. A supporting frame for the substrate consists of flattenable spring tape circular beams which serve as the positive drive transfer medium. See Figure 6-5.

The fragile solar cells are protected against dynamic launch environment and induced wrap forces by means of a flexible foam medium between the adjacent layers of the wrapped substrate.

The reference document is Ryan Report No. 29174-46.

6.2.2.1 Structure and Mechanics. The deployment and structural features of this concept are considered feasible and of current technology. The mechanisms and extendible structural members have been proven in prototype in similar forms by several aerospace companies.

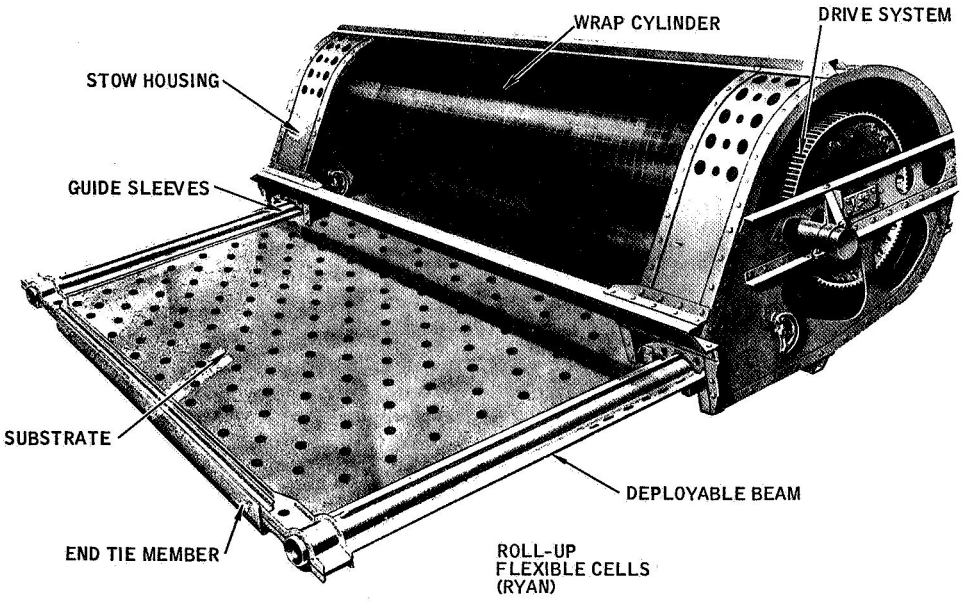



Figure 6-5. Ryan Deployable Solar Array Concept

The operational advantage of retractible arrays is obvious for spacecraft which require large maneuvering impulses. Flexible solar cell arrays, in conjunction with the wrap-cylinder stowage, produce a system much more tolerant to shock, acceleration, and vibrations when compared to the conventional silicon cell arrays.

Although the mechanism and structure have been built in several prototype forms, the cell/substrate assembly development is not as far advanced. The particular wrap-drum concept is generally applicable to the new technology of the flexible type cells, although some work is being done to adapt conventional cells to roll-up dispensers. The characteristic advantages, disadvantages, etc., of cell detail will not be discussed here; however, the following comparisons are typical:

| | Flexible Cells | Conventional Cells |
|---|---|---|
| Watts/lb (Cells Only) | 75-100 | 35 |
| Watts/lb (Cells + Covers + Substructure) | 15 | 6.5 |
| Watts/$ft^2$ (Active Area) | 4.5 | 9.8 |
| lb/$ft^2$ (Active Area) | 0.3 | 1.5 |
| Cell Conversion Efficiency | 5% | 12% |
| Space Exposure Efficiency (2 Years) | 80% | 50% |
| Storage Degradation (Earth) | Poor (Due to Moisture) | Good |
| Radiation Resistance | Good | Poor |
| Shock, Acceleration, Vibration | Very Good | Poor |
| Dollars/Watt (Cells Only) | 20-30 | 200 |
| Net/Gross Area Efficiency | 75% | 90% |

It is reasonable to speculate that thin film cell array power-area ratios will be available at approximately 10-15 watts per $ft^2$ and with area efficiencies near 90% in the operational period 1968-1972.

The pecularities of specific applications are the determining factors when choosing between the flexible arrays and the conventional rigid arrays. The flexible array offers a simpler deployment mechanism, and it is such arrays that hold the promise to advance solar cell systems into the region of high power outputs (up to perhaps 50 kw) at reasonable cost. The rigid arrays offer proven capabilities. If one considers the array to be primarily an experiment of its own, then the subject concept is appropriate to move the laboratory development of flexible solar cells into a space operational system. It is worthy of note that the cell/substrate assembly can be adapted to a concept having removable subsections. The retractable feature lends itself to simplifying the EVA task and keeping such tasks near the spacecraft.

Increasing the flexible array to a power level comparable to the conventional array by increasing the length will result in a length for each array unit of 146 feet (was 52.8 ft). See Figure 6-6.

Thermal distortions effectively tighten the control requirements to maintain acceptable solar incidence angles. The total mass moment of inertia increase would perhaps improve vehicle stability but it would also increase maneuvering propellants. The Ryan selection of titanium produces serious consequences in regards to the thermal bending. Assuming the arrays act as cantilever structures, and at a 90° solar incidence at the root, the front-to-back temperature difference of the structural edge member will be:

$$\Delta T_s = \frac{d^2}{4\, t k_{T_i}} \alpha_s S = \frac{(2)^2(0.2)(3.12)}{(4)(0.007)(0.833)} = 107^\circ F$$

where

$d$ = tube dia = 2 in.

$t$ = wall thickness = 0.007 in.

$\alpha_s$ = solar absorptance = 0.2

$S$ = solar constant = 3.12 Btu/hr/in.$^2$

$k_{T_i}$ = thermal conductivity for titanium = 0.833 Btu/hr/in./° F

A simplified expression for tip deflection gives:

$$\delta = \frac{\alpha\ \Delta T_s\ \ell^2}{2d} = \frac{(4.8)(10^{-6})(107)(1750)^2}{(2)(2)(12)} = 32.8 \text{ ft}$$

where

$\alpha$ = coefficient of thermal expansion = $4.8 \times 10^{-6}$ in./in./° F

$\ell$ = length = 1750 in.

This deflection results in a 27-degree deflection angle at the tip. This poor exposure can be bettered by coating the titanium to reduce solar absorptance to 0.1 and by canting the array as depicted in Figure 6-6. Another solution would be a complete material change - perhaps an aluminum alloy, shown in Figure 6-7 for comparison.

6.2.2.2 Packaging. One advantage claimed of the flexible cell array is its reduced packaged volume. However, large arrays are gained at the expense of high aspect ratios. The width dimension of the array sets the practical design limit on total array size. In manned flights it is conceivable to aid this type of concept by stowing the array canisters for maximum packaging efficiency, then positioning the canisters via the EVA mode, making electrical connections, and remotely initiating the automatic deployment.

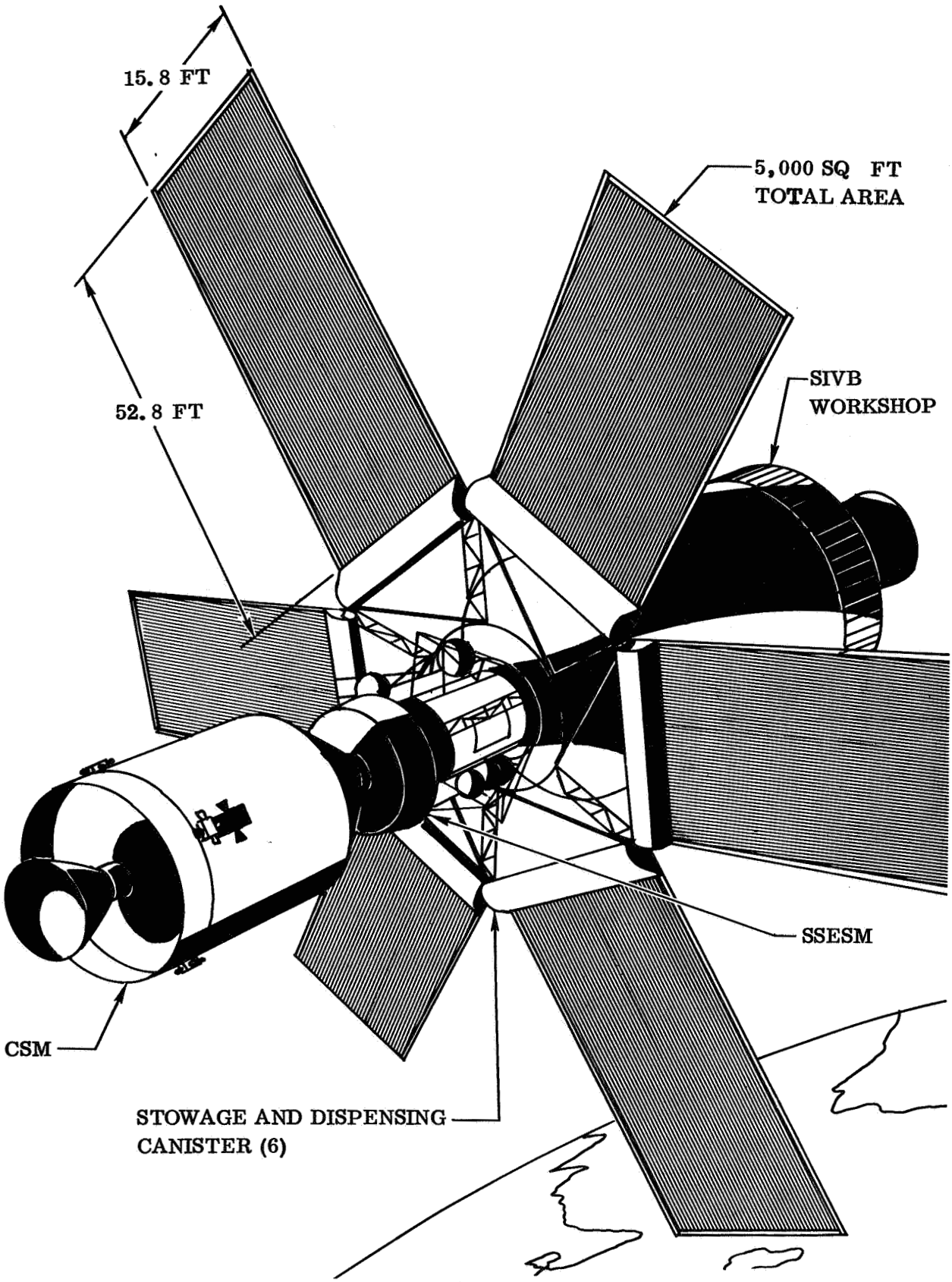



Figure 6-6. Concept 8, The Large Solar Cell Array (Ryan)

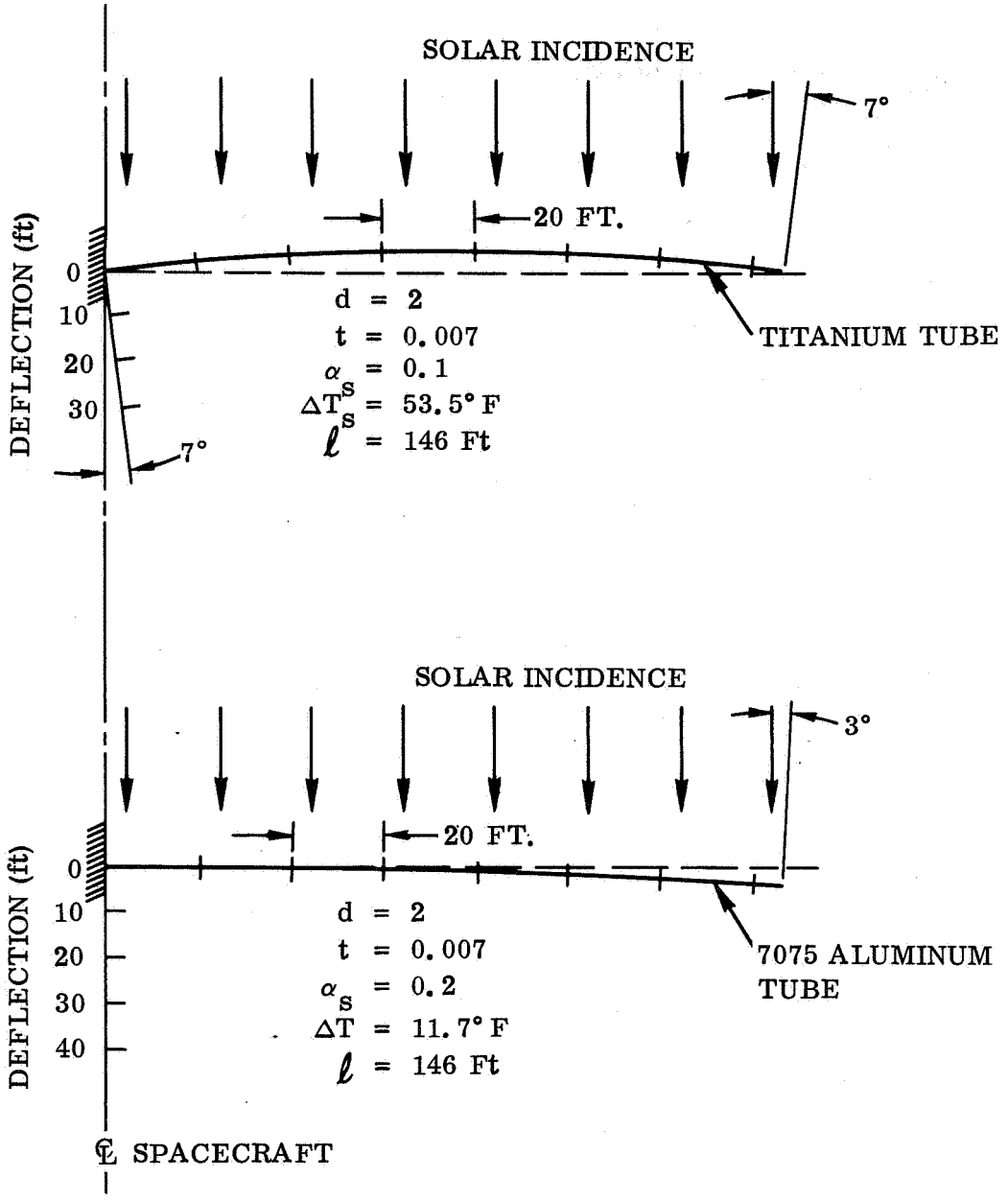



Figure 6-8. Array Thermal Deflections

Figure 6-8 illustrates the stowage of a 5,000 $ft^2$ solar array.

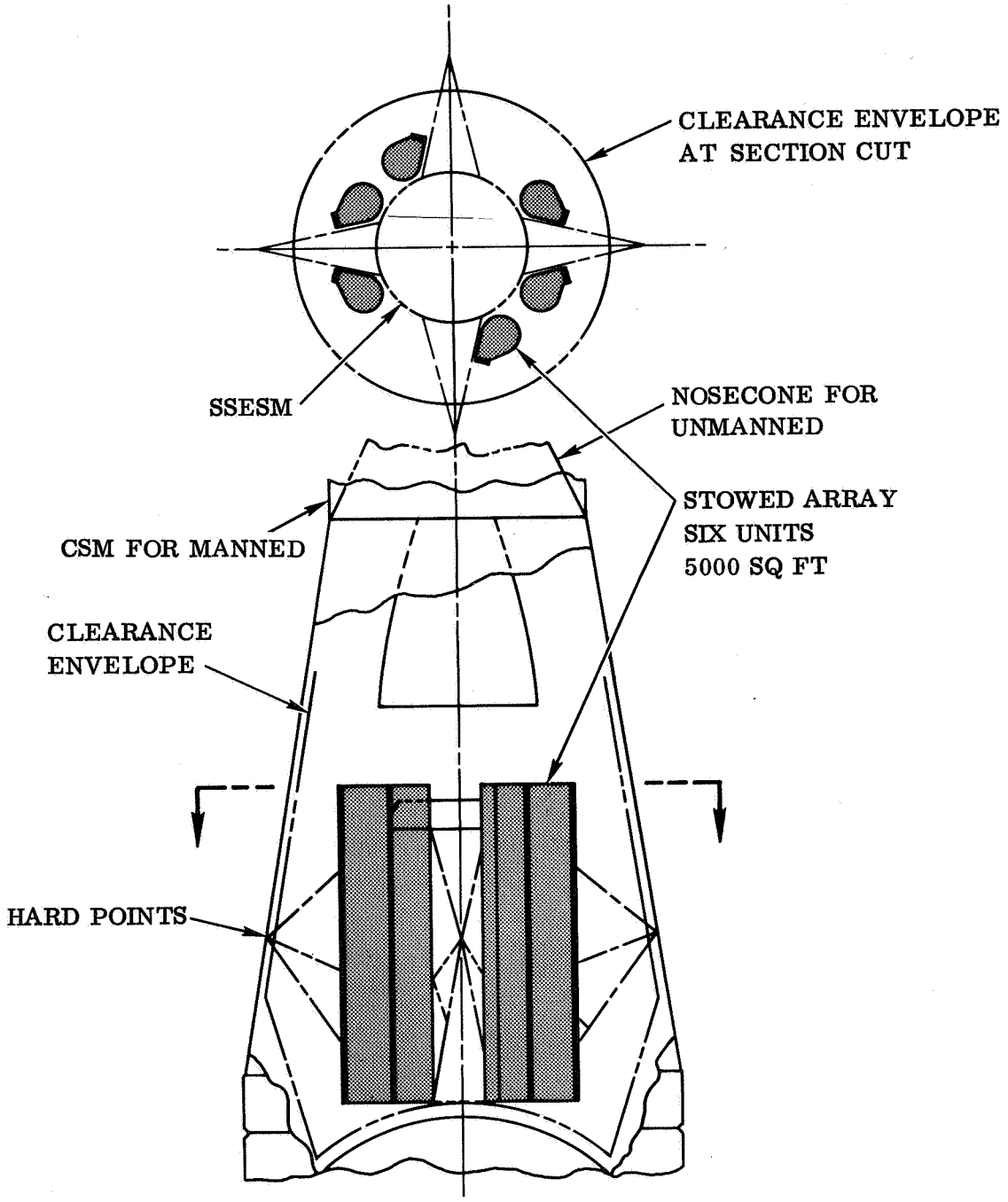



Figure 6-8. Stowage of the Solar Array Concept

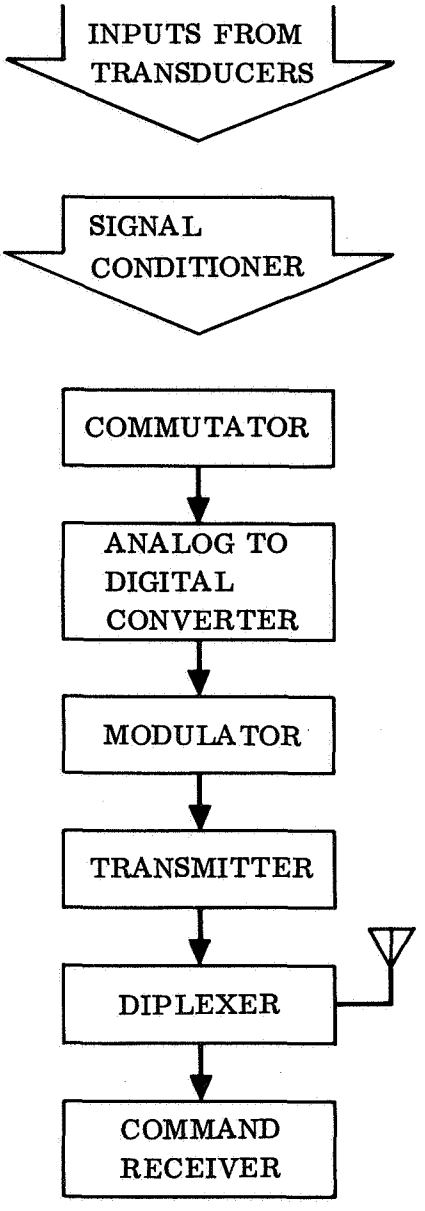



Figure 6-9. Telemetry Command System

6.2.2.3 Electronics. The telemetry system, desirable for monitoring satellite performance, can have some common equipment with the command link. It is also possible to incorporate a transponder type of tracking beacon into this composite system.

The telemetry system is used to monitor voltages, pressures, temperatures, and other low information rate sources. The bandwidth required to telemeter sixty of these measurements is approximately one KHz. The type of modulation used would be either frequency modulation (FM) or Pulse Code Modulation (PCM). The current trend is with PCM and the system described below will be of this type.

The frequency of the system will be in the lower VHF, around 300-400 MHz because equipment is more available in this range. A block diagram of the system is shown in Figure 6-9. The transmitter power requirements are easily met with a one-watt output. This assumes zero gain at the satellite antenna and 35 db gain at the ground antenna. The narrow information bandwidth (1000 bits per second) allows the noise improvement factor to be 16 db while maintaining a 15 db signal-to-noise ratio at the receiver. Size, power, and weight of this system is shown in Table 6-6.

6.2.2.4 Dynamics/Attitude Control. Configuration 8 (Ryan solar array) is notable in that the arrays are large and flexible. The concept has a high watt-to-weight ratio, but this is purchased at the expense of using large areas and large spans.

Minimum orbital altitude is raised by the larger areas, and maneuvering (control system operating frequencies) is restricted by the structural weaknesses. The location of minimum orbital altitude and the maneuvering limitations must be investigated for specific satellite-solar cell array configurations. However, satisfactory behavior is anticipated for the typical synchronous orbit missions, which usually have slow maneuvering rates.

Table 6-6. Telemetry System Characteristics

| | WEIGHT (lb) | POWER (w) | VOLUME ($in.^3$) |
|---|---|---|---|
| Signal Conditioner | 1 | 0.5 | 24 |
| Commutator | 1 | 0.5 | 16 |
| A to D Converter | 2 | 1 | 32 |
| Modulator | 2 | 3 | 14 |
| Transmitter | 5 | 5 | 20 |
| Diplexer | 2 | 0 | 12 |
| Receiver | 25 | 1 | 200 |
| Transducers and Cables | 12 | 5 | 210 |
| Total | 50 | 16 | 528 |

The necessity for placing a lower bound on structural frequencies in order to avoid coupling with the control system has been discussed for various concepts and will not be belabored further here. This minimum was examined in detail for this concept, since it is the weakest of the solar cell arrays. The analysis of the others has included the assumption that the first mode frequency (i.e., structural strength) is high enough so that the load-induced deflections are acceptable.

The analysis concludes that the minimum modal frequency established by stiffness and deflection requirements is of the order of 0.05 Hz. This minimum is feasible. Should the attitude control be supplied by large engines, such as those of the CSM, the minimum would be increased.

The deflection was calculated for the first mode, which is most of the total deflection. Table 6-7 shows the tip deflection, for several values of torque and frequency, for a satellite inertia of 3,000 slug-$ft^2$.

Table 6-7. Tip Deflection due to Jets (feet)

| FREQUENCY (Hz) | TORQUE (ft-lb) | | |
|---|---|---|---|
| | 1 | 10 | 25 |
| 0.1 | 0.01095 | 0.1095 | 0.274 |
| 0.05 | 0.0437 | 0.4370 | 1.076 |
| 0.0317 | 0.1095 | 1.0950 | 2.740 |
| 0.01 | 1.0950 | 10.9500 | 27.400 |

Deflection values in the table can be used to estimate the minimum first modal frequency. This is linear with assumed torque. A minimum of 0.05 Hz is a reasonable value.

The largest environmental disturbance will be the aerodynamic drag. At 200 n.mi., the drag on the 833 sq ft of one array, $F_d$, can be 0.0185 lb. Most solar cell array applications are anticipated to be at higher altitudes than 200 n.mi.

The tip deflection due to a given jet torque will decrease for satellite inertias above 3,000 slug-ft$^2$. But the use of Apollo hardware such as the CSM involves higher torque levels as well as higher inertias. A tip deflection of 3.78 ft is calculated at 0.1 Hz using CSM parameters. Thus the utilization of the ACS of the CSM (or LEM or S-IVB) significantly increases the minimum allowable array frequency.

6.2.2.5 Weights. The Ryan report indicates a weight of 22 lb for their 51 ft$^2$ prototype model, not including cells. This prototype was designed to accommodate 2 x 2 cm silicon solar cells on a 0.003-in. fiberglas substrate. On this basis, a six-arm array totaling 5000 ft$^2$, assuming an area utilization of 0.75, would weigh:

| | | | |
|---|---|---|---|
| Structure + substrate | = 5000 (0.38) | = | 1900 lb |
| Cells | = 5000 (0.75)(0.50) | = | 1870 lb |
| Dispenser unit + truss | = | | 2000 lb |
| | | | 5670 lb |

This assumes the deployable titanium side beams would be the same size as the 51 ft$^2$ model.

Applying this concept to the thin film flexible arrays, the weight for the same area becomes:

| | | |
|---|---|---|
| Side members | = | 93 lb |
| Cell + substrate | = | 967 lb |
| Dispenser + truss | = | 2000 lb |
| | | 3060 lb |

6.2.2.6 Human Engineering. The gross crew functions envisioned for the deployment of the Ryan large solar cell array structure are illustrated in Figure 6-10. An initial estimate of the time it would take to complete these functions is given in Table 6-8. This table assumes the Concept 8 arrays are used with the SSESM.

Crew EV functions appear to be within the capabilities of the EV crewman. He is not required to maneuver large distances, his basic worksites are expected to be adequate for attaching anchoring devices, and his primary task — changing the array canisters from the stowed to the pre-deployment position — should not involve functions that exceed his capacities. However, the exact nature of this basic task and the degree to which it taxes the EV crewman depends on the final size and mass required for the array canisters.

The inspection of the lightweight panels would normally create maneuvering and anchoring difficulties, but if the panels can be extended and retracted at will, this task could be easily accomplished with the EV crewman anchored to the supporting structure as the panels are extended. However, if the panel extension operation is considered hazardous to an EV crewman, the inspection task will have to be performed after panel deployment. If it can be limited to visual inspections only, it may not require any EVA, but might be accomplished simply by CSM maneuver and inspection through the spacecraft windows or by the use of a spacecraft-mounted TV system.

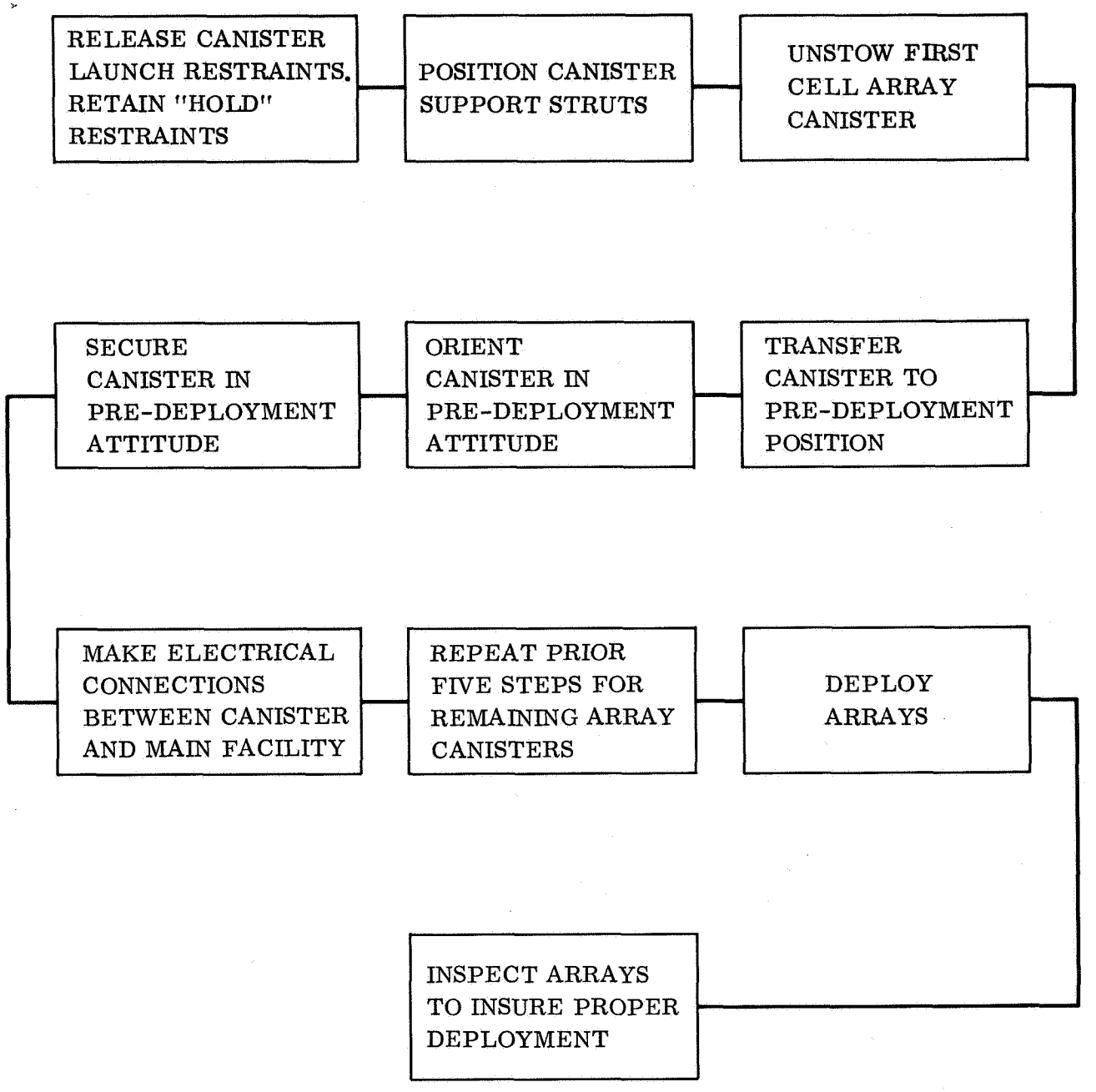



Figure 6-10. Gross Crew Functions

If the panels only partially deploy, the EV crew can be useful in correcting the malfunction. This would again involve anchoring at the main facility and adjustment of the array canisters and should not require maneuvering at or anchoring to the ends of the lightweight panels.

To allow the EV crewman to perform his required tasks, the following EV support equipment will be required: PLSS or AMU, communications, power supply, illumination, tether, and anchoring systems. In addition, special support equipment will be required to assist the EV crewman in changing the array canisters from the stowed to

Table 6-8. Time Line Analysis

| EVENT | ASTRONAUTS (2) | TIME IN MINUTES | | |
|---|---|---|---|---|
| | | EVENT* | CUMUL | EVA CUMUL |
| 1 | Perform pre-EVA | 30 | 30 | |
| 2 | Transfer to SSESM | 5 | 35 | |
| 3 | Depressurize airlock | 20 | 55 | |
| 4 | Open hatch and exit | 10 | 65 | |
| 5 | Don and check out maneuvering device | 20 | 85 | 20 |
| 6 | Release canister primary package restraints | 10 | 95 | 30 |
| 7 | Exit from SSESM thermal shroud | 10 | 105 | 40 |
| 8 | Unstow and lock in place the canister support struts | 15 | 120 | 55 |
| 9 | Remove thermal shroud hole cover and unstow first array canister | 10 | 130 | 65 |
| 10 | Transfer canister to pre-deployment position | 5 | 135 | 70 |
| 11 | Orient canister in pre-deployment attitude | 5 | 140 | 75 |
| 12 | Secure canister in pre-deployment attitude | 15 | 155 | 90 |
| 13 | Replace thermal shroud canister hole covers | 5 | 160 | 95 |
| 14 | Unstow electrical equipment | 5 | 165 | 100 |
| 15 | Make electrical connections | 5 | 170 | 105 |
| 16-50 | Repeat events 8-15 for remaining canisters | 300 | 470 | 405 |
| 51 | Maneuver to safety area | 3 | 473 | 408 |
| 52 | Anchor in safety area | 2 | 475 | 410 |
| 53 | Deploy flexible cell arrays | 30 | 505 | 440 |
| 54 | Release anchors | 1 | 506 | 441 |
| 55 | Inspect arrays | 30 | 536 | 471 |
| 56 | Doff and stow maneuvering device | 15 | 551 | 486** |
| 57 | Enter spacecraft, close hatch, repressurize, and stow gear | 60 | 611 | |

* Includes time to maneuver, and attach and release worksite anchors where appropriate.

** Approximately 2 EVA days.

the pre-deployment position. The specific type of equipment could range from properly located handles on the canisters and the necessary fastening tools to a spacecraft-mounted rotation and securing device actuated by control inputs from the EV crewman.

6.2.2.7 Evaluation Summary. The Ryan concept as a deployable and functioning large space structure is mechanically and structurally feasible and is of current technology. The cell/substrate assembly is the subject of several development programs but is not as far advanced as dispensing mechanisms. The wrap-drum concept is generally applicable to the new technology of the thin film flexible solar voltaic cells although some work is being done to adapt conventional cells to the roll-up configurations. The Ryan development model is to be used with conventional rigid cells, 2 cm × 2 cm.

The thin-film array system offers a simple and reversible deployment mechanism and operational flexibility. It is from such arrays that potential exists to advance solar cell systems into regions of high power outputs (perhaps 50 kw) at reasonable costs. Thin film cell costs are approximately one-tenth that of rigid cells. Present form thin-film flexible arrays suffer primarily in two respects when compared to conventional rigid cells. These are low cell conversion and area utilization efficiencies, resulting in gross areas exceeding twice that for rigid cells.

The geometric nature of the wrap-cylinder will configure concepts generally having undesirable high aspect ratios, especially as the area increases. Large aspect ratios contribute to control problems and array deflections. The Ryan concept using titanium is especially prone to thermal deflections; these deflections exceed sun-line pointing tolerances.

The flexible cell system is especially amenable to compact stowage and ability to withstand severe vibration and high acceleration. Deployed areas greater than 15, 000 $ft^2$ could be stowed in the AAP launch configurations, although this size may exceed the total weight limitations of the booster.

Since the deployment cycle is entirely automated, astronaut assist is ideally relegated to positional tasks in the near vicinity of the spacecraft. Large packages are handled but are not complicated and the total array is assembled by separate increments (example: the 5000 $ft^2$ array consists of six independently positioned and deployed canisters).

This concept is considered feasible and justifiable. It contributes to the technology of large space structures and performs as a useful power plant. It is especially adaptable for use in conjunction with other large structures.

6.2.3 Concept 19 - Fairchild Hiller Roll-Up Solar Cell Array. The Fairchild Hiller roll-up solar cell concept consists of an extendable panel 20-foot long by 2-foot wide with cells mounted on a flexible substrate that is wound on two spindles. The array is

deployed by extension of a scissors mechanism between the two spindles. The total weight of each panel and deployment mechanism is 35 pounds, and each can be packaged into a size 25×25×8 inches. An array would consist of any number of these panels arranged in fixed, pivotal, or gimbaled configurations.

A model of a partially deployed solar array structure is depicted in Figure 6-11. Solar cells, cover glass, and interconnections are bonded to the flexible substrate that is rolled on two spindles, similar in manner to film in a camera. The array is deployed by extension of a scissor mechanism between the two spindles. Deployment is controlled by centrifugal brake and synchronizing gear train. A thin layer of polyurethane foam is bonded to the back of the flexible substrate to provide a shock resistant buffer between the face of the solar cell and the back of the next outer layer when in a stowed (undeployed) configuration. This provides sufficient protection and considerable damping. A spring motor and screw jack provde energy to deploy the scissors linkage. A release mechanism provides 3-axis support for the extension linkage during the boost phase of the flight and keeps the rollers from deploying the solar cells prematurely.

The concept design goal is for an electrical power to weight ratio of 20 watts per pounds based on a kilowatt size power supply.

Figure 6-12 shows a satellite and four packaged 20-foot long by 2-foot wide cell arrays. The cells and belt substrate and extension mechanism can be rolled into small units whose dimensions are 25×25×8 inches. The weight of each array and deployment mechanism is 35 pounds, for a total weight of 140 pounds. The panels can be deployed from a satellite spinning at rates up to 160 rpm.

6.2.3.1 Structure and Mechanics. A typical packaged experiment, shown in Figure 6-12, has a maximum height of 3 feet. The configuration shown assumes a Saturn-V launch vehicle for a synchronous orbit mission. The arrays supply power to an ejectable satellite, which is stowed in conjunction with the SSESM within the SLA envelope. The deployed solar panels will provide 1 kw of electrical power to a satellite of 530 cu ft maximum volume, including support structure to the SSESM. The satellite would contain an experiment, RF data communication, and attitude control equipment.

The 20 ft × 2 ft solar panel shown in Figure 6-11 contains solar cells on only one side of the deployed belt. In this case, the deployed panel is continuously sun oriented by an electric motor located in the base of the panel. Configurations of crossed arrays having one controlled axis can maintain all panels sun oriented only if the satellite is sun oriented. At other attitudes, the derived power output will be some fraction of full capability.

The solar cells, including substrate, are wound on a 5-in. drum. This small radius of curvature will allow the sharp edge of a cell on one layer to coincide with the center of the cell in the next layer. Sufficient tension on the substrate during deployment will cause bending stresses within the cells and produce cracking. The proof of reliability must, therefore, be substantiated by test.

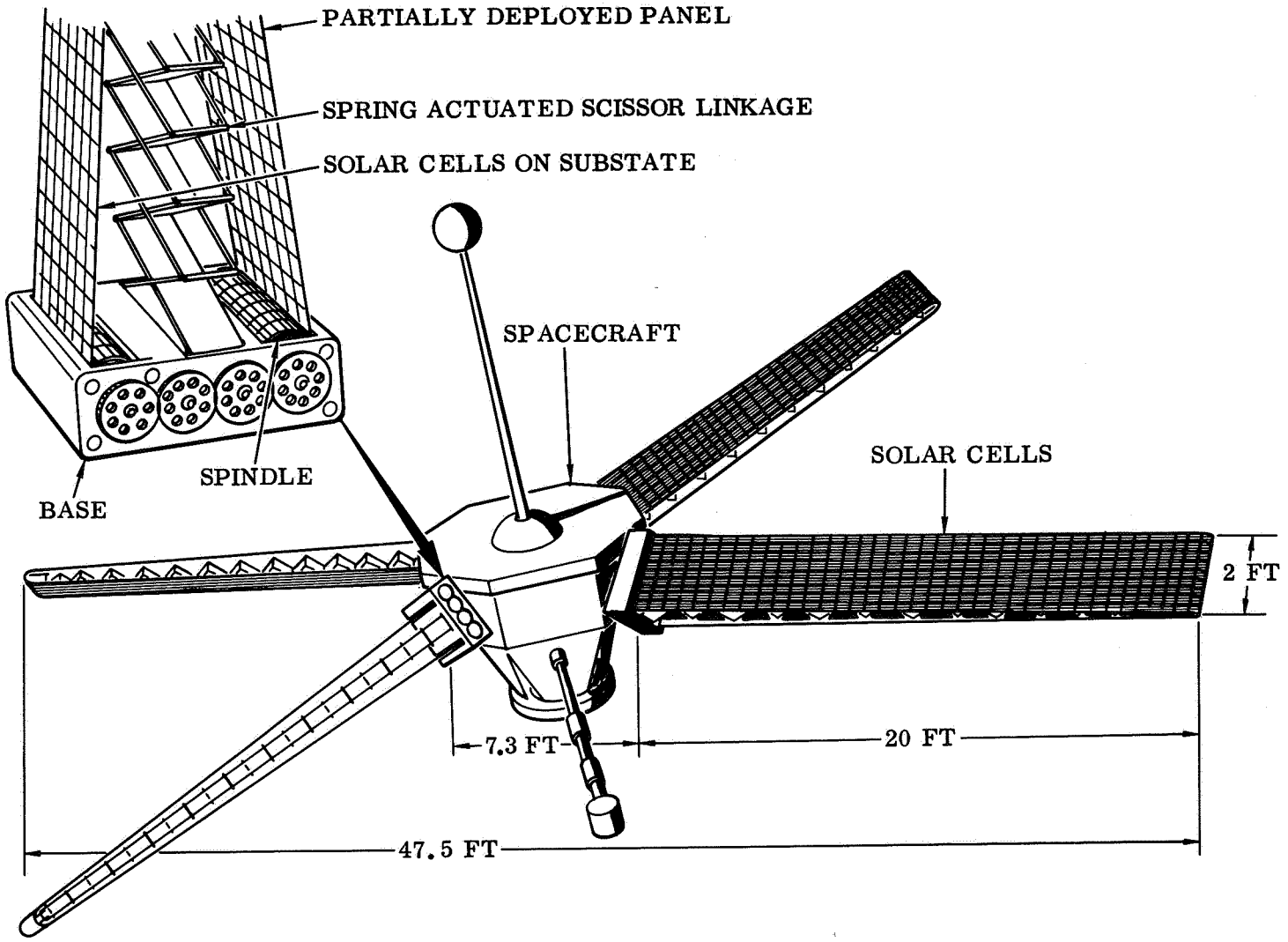



Figure 6-11. Concept 19, Rolled Solar Cell Array (Fairchild Hiller)

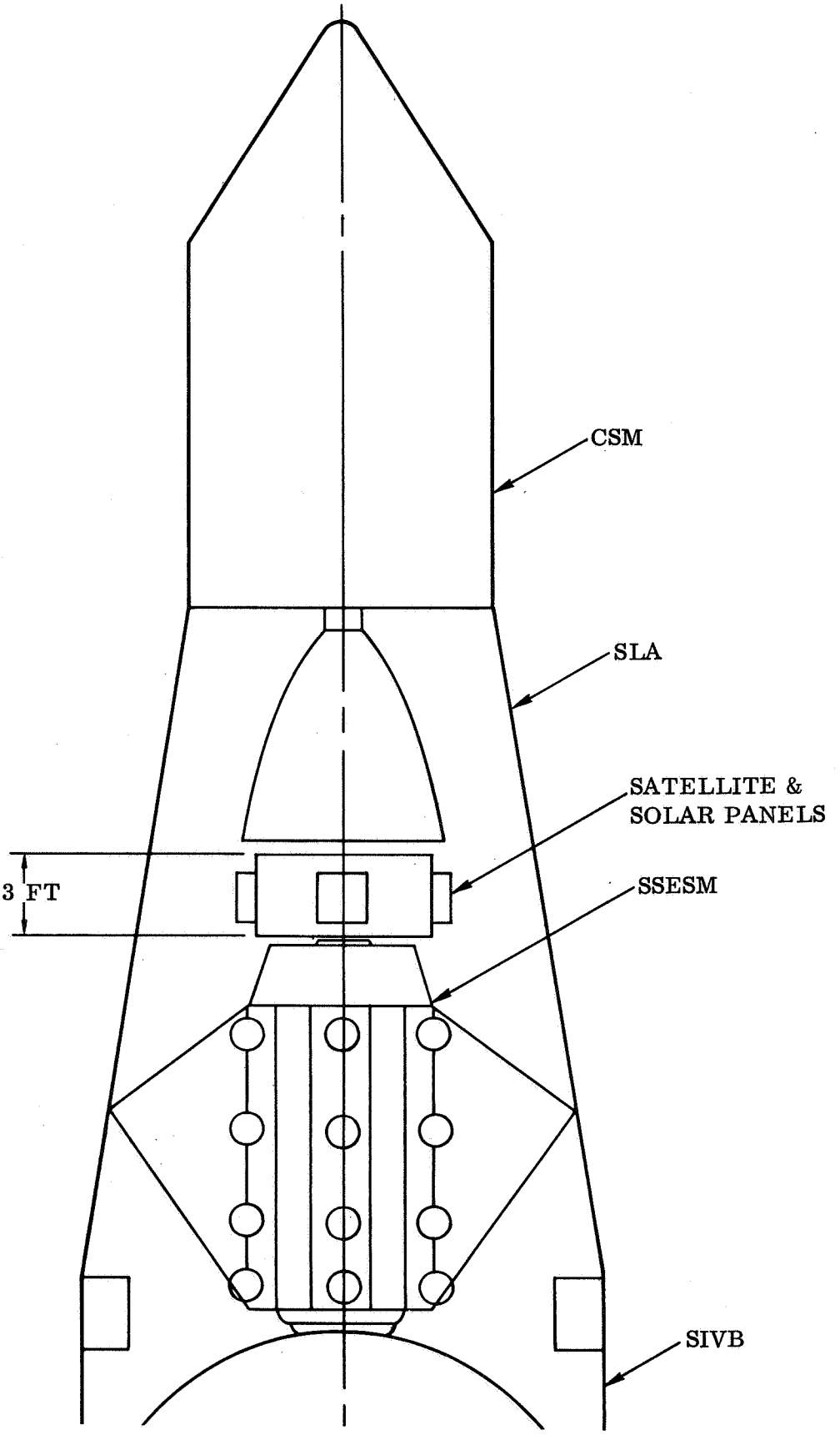



Figure 6-12. Typical Packaged Experiment

The array system specific power output is 12.5 watts/pound. The 20 watts/pound noted in the Fairchild-Hiller report is derived by consideration of cell output only. This cell weight breakdown is given in Section 6.2.3.3. At 0.545 lb/ft$^2$ of cell components only, the output is 20 watts/pound. This figure is misleading as it does not include the substrate or mechanism weight of 13.2 pounds per panel. The array system specific power output of 12.5 watts/pound does not include the batteries, conditioning equipment nor cabling for a total power system, but is indicative of the array system for comparative purposes.

6.2.3.2 Attitude Control and Dynamics. This specific concept is notable in that the solar cell area is only 160 sq ft, which is smaller than the other solar array concepts. As a result, most of the typical solar array problems are relatively unimportant.

6.2.3.3 Weights. Table 6-9 presents a weight breakdown of the solar cell and array components. A total electrical power system including batteries, conversion equipment, and cabling is not presented. The weight values, therefore, are representative of only the array and its deployment subsystem.

Table 6-9. Solar Array Weight Summary

| | | |
|---|---|---|
| Cell Components | | |
| Solar Cells | 0.159 lb/ft$^2$ | |
| Cover Glass | 0.190 | |
| Adhesive | 0.080 | |
| Connections | 0.070 | |
| Solder | 0.046 | |
| Total | 0.545 lb/ft$^2$ | |
| Total Cell Installation Weight: (0.545)(160) | | 87.2 lb |
| Structure and Mechanisms | | |
| Belt Substrate | 1.66 lb | |
| Spindles | 3.20 | |
| Scissors Mech | 3.12 | |
| Gear Train and Brake | 0.57 | |
| Support Structure | 2.66 | |
| Screwjack and Springs | 1.00 | |
| Orientation Motor | 1.00 | |
| Total | 13.21 lb | |
| Total Weight, Structure and Mechanism: (13.1)(4) | | 52.4 lb |
| Total Array Weight | | 139.6 lb |

6.2.3.4 Human Engineering. Extravehicular crew functions are not required in the deployment of the subject array as it is designed for automatic deployment. The EV crew can best be utilized in an inspection and repair role after panel deployment or to correct equipment malfunctions should the panels not properly deploy. However, if the panels are deployed from a spinning satellite, the EV crew cannot be utilized unless the satellite spin can be stopped and restarted without jeopardizing mission success.

In an inspection and repair role, the EV crew could perform such functions as an inspection of the panel deployment mechanisms to ensure proper alignment and complete deployment, and inspection of the solar cells for damage during launch or deployment, and an inspection of the electrical connections for continuity. If the inspection tasks can be limited to visual inspection only, they may not require any EVA, but might be accomplished simply by CSM maneuver and inspection through the spacecraft windows. If EVA is required, the provision of a dock on the satellite must be compared with CSM stand-off EVA.

To allow the astronaut to perform the EV inspection and repair tasks, the normal EV support equipment will be required: PLSS or AMU, communications, power supply, illumination, tether, and anchoring systems. The lightweight nature of the solar cell panels (35 pounds/panel) requires maneuvering and anchoring systems that minimize the possibility of damage due to EVA.

6.2.3.5 Evaluation Summary. The gross size is hardly indicative of a large space erectable structure, although it can certainly be scaled to larger sizes. The array specific weight of 12.5 watts/lb is higher than most conventional rigid panels using silicon cells (9 watts/lb).

The deployment mechanism and structure are current technology. The application of rigid type solar cells to the flexible substrate and on a small diameter wrap drum suggests rather difficult detail problems in this area.

## 6.3 ANALYSIS OF NEW CONCEPTS

6.3.1 Concept 81 - Solar Cell Array - Communications. This concept, shown in Figure 6-12, will provide electrical power for the 150-ft-diameter antenna discussed in Section 5.3. It consists of Concept 75 plus two solar cell arrays.

The solar cell array of 3000 sq ft is supported by two large paddles mounted diametrically opposite on the rim of the antenna parabolic dish. Each is pointed and power driven to obtain optimum solar exposure as determined by its sensors. The antenna ACS provides gross pointing and stabilization of the structure on which the solar panels are pivoted.

The solar panels are stowed in the LEM adapter below Station 117.3 supported by a tubular truss structure as shown in Figure 5-61. The antenna, constructed in

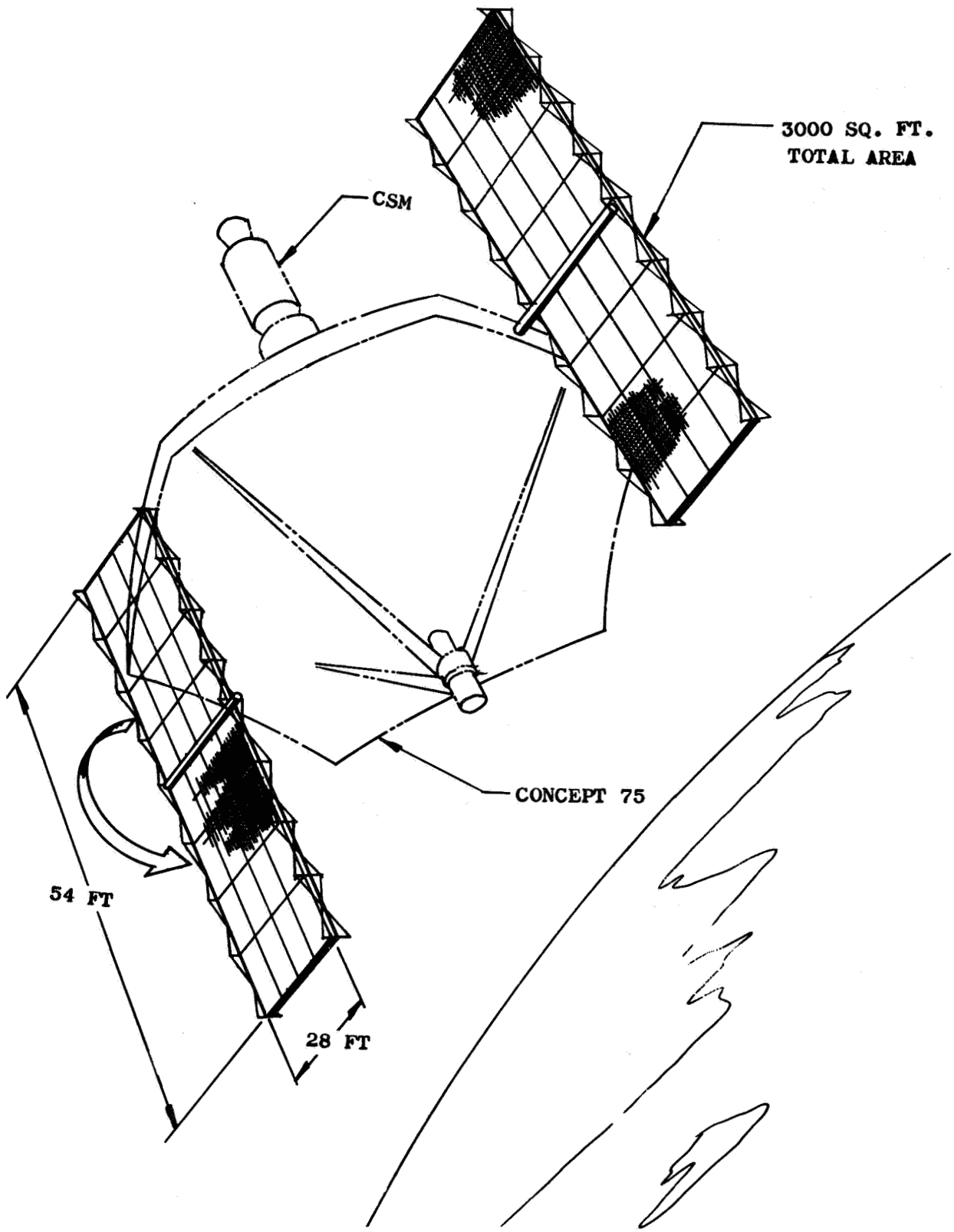



Figure 6-12. Concept 81, Solar Cell Array, Communication Antenna Application

accordance with Convair's Space Erectable Antenna design, is stowed on the same truss but above Station 117.3.

The combination is launched by a Saturn V and automatically deployed. Since antenna inspection is accomplished by EVA, the solar array is similarly inspected after deployment.

6.3.1.1 Structure and Mechanics. The pre-deployed configuration is a package 100 in. high by 179 in. wide. It is supported by an aluminum tubular truss structure 10 in. deep, that attaches to the 4 major Saturn (SLA) adapter points at Station 117.3. The solar cell arrays are stowed below this truss structure and anchored to it at four points with explosive bolts. See Figure 5-62.

All motions of expanding antenna, arrays, and feed are switch-controlled; i.e., signals are generated and monitored for positive identification of open, closed, and locked.

6.3.1.2 Attitude Control and Dynamics. The large surfaces of the solar cell arrays result in large aerodynamic torques at lower orbital altitudes. Depending on surface area and on mission details, the minimum acceptable altitude is expected to range from 100 to 300 n.mi.

Structural characteristics of solar panels place an upper-bound on control system frequencies. These bounds restrict vehicle maneuvering and slewing rates. Low maneuvering missions, such as interplanetary probes or earth pointing communication satellites, are not likely to be degraded by these restrictions.

The solar panel arrays must be continuously pointed at the sun with tolerances of ±15 deg, in order to be efficient. Slower maneuvering missions allow this requirement to be satisfied.

With the antenna pointed and stabilized by its own ACS, the solar panel attitude control consists of sun sensors and rotation drive mechanisms. However, the possible advantages of using the CSM sensors and comparing paddle position by computer must be considered.

6.3.1.3 Stress Analysis. The addition of the solar cell areas complicates the deployment analysis. Inertia loads associated with sudden stoppage at final deployment will certainly require heavy tubes at the attaching points and may affect the design of all antenna members.

Loads due to solar pressure, gravity gradient, and maneuvering are small and no structural problem is envisioned. If docking of the CSM to the deployed experiment is necessary, the impact load could be approximately 5000 lb and the inertia load factor is approximately 0.5 limit. The loads at the attachment of the solar cell are:

$$P = 0.5(400) = 200 \text{ lb}$$

$$M = 200\left(\frac{28 \times 12}{2}\right) = 33,600 \text{ in.-lb}$$

These loads may require some reinforcement of the antenna structure at the docking ring and near the solar cell array attachments.

Assuming a temperature differential of 100° F, 4 in. deep panels, the tip deflection and slope are:

$$\zeta = \frac{\alpha(\Delta T)(L^2)}{2h} = \frac{13(10^{-6})(100)(324)^2}{8} = 17 \text{ in.}$$

$$\theta = \frac{2(17)}{324} = 0.105 \text{ or } 6°$$

Although this value is within the allowable tolerance for solar cells, it is apparent that the temperature differential must be considered.

6.3.1.4 Weights. The following weight statement details the weights for the solar cell array. This in itself is not a complete experiment. Therefore, Concept 75 has been added and attitude control requirements established.

| | | |
|---|---|---|
| Solar Cells | 1560 | |
| Honeycomb Core 2 in. Deep | 800 | |
| Skin, 0.008 Al | 161 | |
| Bonding | 350 | |
| Frame, 0.04 Al 2 × 1 × 1 in. "C" | 260 | |
| Misc Hardware | 20 | |
| Contingency 10% | 87 | |
| Solar Cell Array | 3238 | 3238 |
| Antenna (Concept 75) | | 3958 |
| Attitude Control | | 433 |
| Total | | 7629 |
| Available Payload, Saturn V to Synchronous Orbit | | 23,992 |

6.3.1.5 Crew and Equipment. See Section 5.3.

6.3.1.6 Concept Evaluation. Materials used for structure and components of the arrays are within the present state of the art. The method of solar panel stowage and deployment utilizes only a portion of the volume available and requires a minimum of EVA.

The concept of solar cell arrays on panels located out from the satellite as an electrical power source is feasible for many missions and vehicles. Lower orbital altitudes and more rapid vehicle maneuvering degrade the feasibility. The synchronous orbits and the low slewing rates of communications satellites are expected to minimize dynamic problems.

The general conclusions about Concept 75 are unchanged by the addition of these solar panels. The solar torques are increased only slightly. The inertias, and hence the gravity gradient torques are changed significantly. Solar torques are much larger than the gravity torques and hence the latter are ignored. Consequently, propellant expenditures are approximately the same. The anticipated low slewing rates preclude coupling.

Stowage, launch, and deployment appear to present no significant problem. EVA is minimal and consists essentially of inspection; however, this does require traversing the deployed panels.

Maintenance problems should be minor and consist only of replacement of solar cells that might be damaged by meteoroids.

Deployment of the solar cell arrays are reversible (together with the antenna) so they can be operated on Earth for test and alignment.

6.3.2 Concept 82 - Solar Cell Array, S-IVB Workshop. This concept is for application to the S-IVB workshop. To satisfy power requirements in excess of the Apollo CSM fuel cell capacity, this solar cell array will supply approximately 27 kw from a gross area of 3000 sq ft.

Four large folding panels are extended from the SSESM as depicted in Figure 6-13. Each panel is rotated about its truss support by a drive mechanism for continuous solar orientation within ±10° . The CSM or S-IVB provides gross attitude control of the spacecraft.

Extended duration versions of the S-IVB workshop may produce loads of over 10 kw, requiring array areas of several thousand sq ft. Solar intensities at synchronous orbit make it possible to derive 10 watts/$ft^2$, while at the low altitudes the average output would be 8 watts/$ft^2$. Preliminary studies of a combined S-IVB workshop, air lock, CSM system indicates that approximately 3000 $ft^2$ of solar cells may be accommodated in the volume surrounding and above the SSESM (airlock), representing 27 kw of power.

The solar cells for this concept are mounted on four paddles, each supported by a tubular shaft mounted with two degrees of rotational motion to a truss that in turn is attached to the SSESM structural rack.

Each paddle consists of seven panels 80 in. wide and 200 in. long, hinged together along their sides to permit accordian stacking for stowing. In operation, the paddle is

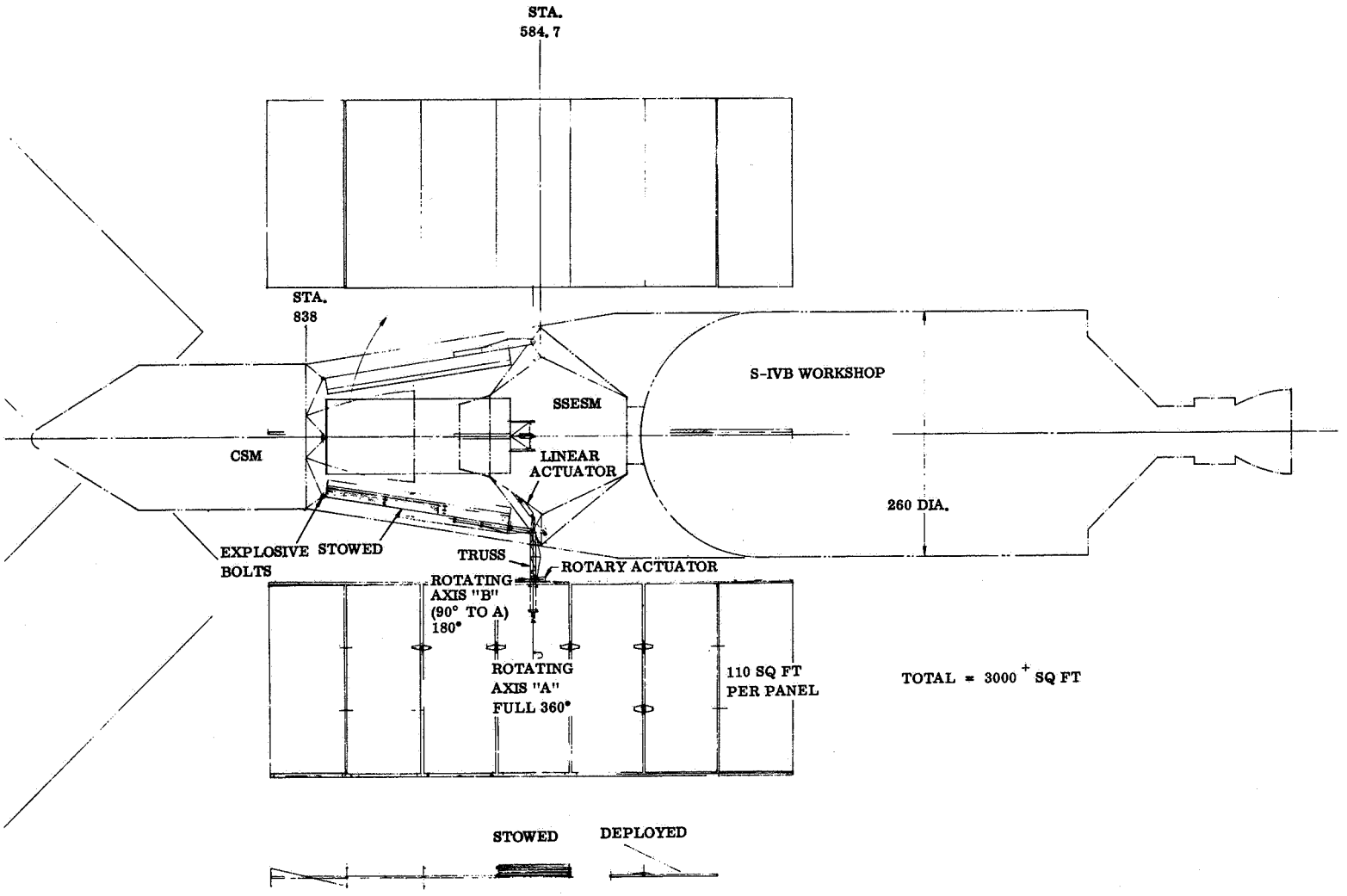



Figure 6-13. Stowage and Deployment

rotated by its shaft which is driven by a sealed electric motor geared to orient the paddle continually toward the sun.

The tubular shaft rotates in a housing which is hinged about an axis at 90° to the shaft and driven by a rotary actuator with worm and gear. The housing bearings are supported on a trussed boom extending into the SSESM support structure where it is hinged for deployment motion.

When stowed the panels are stacked and the paddle assembly is stowed in the volume between the SSESM and the LEM adapter, where it is retained against deployment by explosive bolt fittings on the top ring of the LEM adapter and the top of the stacked panels.

In the stowed condition each blade is attached to the top ring at Station 838 with two 1/2 in. explosive nuts (Hi Shear or equivalent). When signaled, these bolts separate from the nut and free the blade to be deployed by the linear actuator (Lear Inc. or equivalent) located at Station 584. The actuator rotates the blade about the hinge line at Station 580 through approximately 97° and the blade locks in place. Explosive nuts again release the blade panels for deployment. Deployment energy is obtained by releasing energy stored in torsion springs at each panel hinge fitting. The panels deploy with an accordian motion and lock in flat position. Pin-pullers (Hi-Shear or equivalent) release the blade from support tube attachment, allowing 360° of rotation about the centerline of the support tube. This motion is controlled by a sealed electric motor and gear housed within the support tube. Deployment of the complete blade to a position 90° to the centerline of the tube is initiated by two pin-pullers effecting release and another sealed electric motor and gear housed in the tube outboard of the hinge center, powering the blade to the final position. This hinge centerline is 90° to the main deployment hinge centerline.

Prior to launch, the blades and panels are checked in the test area under 1 g. The technique for testing is by use of the standard optical tooling methods. All blades have adjustment screws to be set and locked when the vernier optical data reads zero on the scales.

The materials used in the construction of the panels and support structure are state of the art and will be updated as significant changes take place. The use of aluminum (2219) is practical from the aspects of weight, cost, and availability. Magnesium by itself has a disadvantage for use in space structures in that it evaporates (sublimes) approximately $4 \times 10^{-3}$ in. per year under vacuum. However, suitable surface treatments such as Alodine protect the bare metal and extend its useful space life.

Recent Convair comparative studies of compression panels constructed of plate/stringers vs honeycomb show the minimum weight of a panel to be used as a base for a solar cell array can be obtained by constructing it of aluminum plate/stringers. See Figure 6-14.

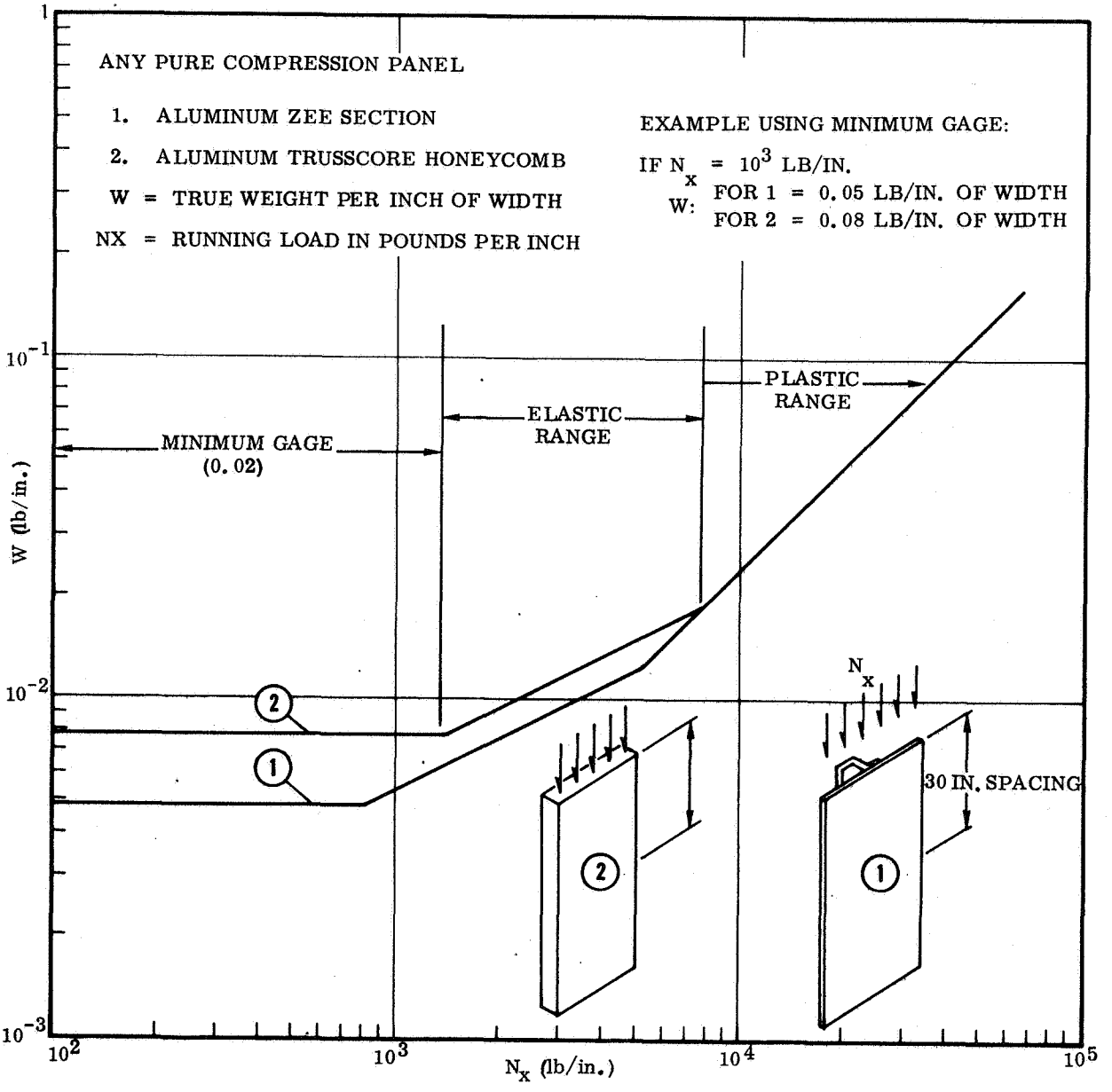



Figure 6-14. Relative Panel Weights

For this concept the panels are constructed of 0.020 aluminum (type 2219). The shape is a rectangle 80 in. wide by 200 in. long. Channel sections 2 in. deep × 1.0 legs × 0.05 thick form a frame around the perimeter of the panel. Six zee stringers 200 in. long and 2 in. deep × 1.0 in. legs × 0.05 in. thick attach to the skin with rivets and to the frame with rivets at both ends. Sheet metal bulkheads 2 in. deep × 0.6 in. legs × 0.04 in. thick spaced every 20 in. provide shear carrying capability and are attached with rivets to skin, stringer, and frame. The skin, 0.02 in. thick, is used as the base to attach the solar cells. Machined aluminum fittings are shaped to attach the two hinges at 1/3 span, and to house the release springs and locks.

6.3.2.1 Attitude Control and Dynamics. The array area is large compared to the spacecraft and the arrays have relatively low modal frequencies. The associated spacecraft is either the CSM/SSESM/S-IVB or the SSESM/S-IVB.

Mission characteristics include 200 n.mi. and synchronous orbital altitudes. Pointing accuracy and maneuvering requirements will vary. Low altitude and fast maneuvering can degrade the performance to the unacceptable point. Detailed examination of specific missions which approach these conditions must be done to establish actual feasibility. For example, numerical results presented below show that operation at 200 n. mi. with the SSESM/S-IVB is not feasible.

It is assumed that the frequency separation is satisfactory. This will impose structural requirements and maneuvering restrictions. In particular, the maximum maneuvering and duty cycles available with the CSM or S-IVB ACS cannot be utilized. The vibrational behavior of this specific design has not been examined. These considerations merit further study.

The control torque and propellant requirements are investigated for both 200 n.mi. and synchronous orbits; the requirements are higher for the SSESM/S-IVB configuration, which is used for the calculations. A pointing tolerance of one degree is assumed. The torques are estimated for inertial hold missions. This gives results which may be higher than true for missions that fix the vehicle to local vertical.

Maneuvering requirements are estimated for the often used assumption of one reorientation per hour for one year, at a velocity of one radian per hour. Many missions would use less maneuvering. A synchronous TV satellite, for example, has virtually none. Limit cycle estimates assume one degree tolerance, an S-IVB ACS, and an unmanned version (without CSM).

Gravity gradient torques are estimated using $7.39 \times 10^5$ slug-ft$^2$ for the two transverse inertias and $1.25 \times 10^5$ for longitudinal. At 200 n.mi., the maximum torque is 1.100 ft-lb and the average is 0.514 ft-lb. At synchronous altitude the maximum is 0.00486 ft-lb and the average for arbitrary inertial orientations is 0.00227 ft-lb. The annual momentum and propellant requirements are summarized in Table 6-10. The all-axes values are approximately double that for single axis.

The aerodynamic and solar torques (of importance at 200 n.mi. and synchronous, respectively) for this configuration have a maximum for the effective moment arm of 17.2 ft. This is unusually large. Total area is 4490 sq ft.

Aerodynamic torque maximum is 1.72 ft-lb and the average for arbitrary inertial hold orientations is 0.429 ft-lb. Solar torque maximum is 0.00746 and the average is 0.00187.

Table 6-10. Momentum and Propellant Requirements, Concept 82 (S-IVB, ACS, All Jet, One Year)

| | 200 N. MI. ORBIT | | SYNCHRONOUS ORBIT | |
|---|---|---|---|---|
| | ONE AXIS | ALL AXES | ONE AXIS | ALL AXES |
| Maneuvering | | | | |
| Momentum, ft-lb-sec | $3.68\times10^6$ | $7.36\times10^6$ | $3.68\times10^6$ | $7.36\times10^6$ |
| Propellant, lb | 613 | 1216 | 613 | 1216 |
| Limit Cycle | | | | |
| Momentum, ft-lb-sec | $8.82\times10^5$ | $1.76\times10^6$ | $8.82\times10^5$ | $1.76\times10^6$ |
| Propellant, lb | 147 | 294 | 147 | 294 |
| Aerodynamic Environ. | | | | |
| Momentum, ft-lb-sec | $1.36\times10^7$ | $2.72\times10^7$ | negligible | negligible |
| Propellant, lb | 2270 | 4540 | negligible | negligible |
| Solar Environ. | | | | |
| Momentum, ft-lb-sec | negligible | negligible | $5.91\times10^4$ | $1.18\times10^5$ |
| Propellant, lb | negligible | negligible | 9.9 | 19.8 |
| Gravity Gradient | | | | |
| Momentum, ft-lb-sec | $1.67\times10^7$ | $3.25\times10^7$ | $7.17\times10^4$ | $1.43\times10^4$ |
| Propellant, lb | 2680 | 5360 | 12.0 | 24.0 |
| Total Propellant | 5710 | 11,410 | 782 | 1554 |

Maneuvering the vehicle requires 210 ft-lb-sec to start or stop about each of two axes. The limit cycle propellant consumption is $4.64\times10^{-6}$ lb per sec. Note that maneuvering and limit cycle estimates depend on mission criteria but are independent of altitude.

The propellant requirements can be reduced by the utilization of momentum storage devices to supply control torque for maneuvering, limit cycling, and the oscillatory portion of environmental torques. The maneuvering and limit cycle momentum changes are within the capacity of planned devices. The oscillatory portions of the environmental torques at 200 n.mi. for inertial hold orientations exceed capacity.

The gravity gradient torques are sharply reduced for selected local vertical vehicle orientations. The maximum aerodynamic torque is not so substantially reduced because the solar arrays are sun oriented, and hence have an arbitrary relationship to local vertical and the drag. This appears to preclude operation of this concept with the SSESM/S-IVB as low as 200 n.mi.

Since aerodynamic drag drops quickly with altitude, the minimum orbital altitude with local vertical vehicle orientation for low gravity torque is not far above 200 n.mi. Gravity gradient torques drop more slowly. Minimum orbital altitude for arbitrary or inertial hold vehicle orientations will be somewhat further above 200 n.mi.

6.3.2.2 Stress Analysis. Each of the folded arrays weighs approximately 1100 lb and the bending moment due to 0.65 g lateral load factor is:

$$M = \frac{1100(200)(0.65)(1.25)}{4} = 44,600 \text{ in.-lb Ult}$$

Assuming no shear ties between panels, each panel carries 1/7 of this moment, or 6400 in.-lb. Eight longitudinals carry this load in bending, each having I = 0.13. The bending stress is then:

$$f = \frac{6400\,(1)}{8\,(0.13)} = 6150 \text{ psi (satisfactory)}$$

All of the axial load is carried by one panel and truss structure. This load, at 4.9 load factor, is:

$$P = 1100\,(4.9)(1.25) = 6750 \text{ lb Ult}$$

The total axial area, neglecting truss rails, is approximately 1.0 sq in. and the stress is approximately 6750 psi, which is satisfactory.

Since the CSM and S-IVB workshop have relatively high moments of inertia, inertia forces on the deployed panels will be low. Arbitrarily using 0.1 g, the load/panel is approximately 110 lb and the bending moment in the rib direction is:

$$M = \frac{55\,(280)}{2} = 7700 \text{ in.-lb}$$

Assuming 2 ribs at each hinge carry this moment, each with I = 0.09:

$$f = \frac{7700\,(1)}{4\,(0.09)} = 21,400 \text{ psi (satisfactory)}$$

The loads at the end of the truss are:

$$P = 110 \text{ lb}$$

$$M = 110\left(\frac{200}{2}\right) = 11,000 \text{ in.-lb}$$

$$\text{Maximum Moment in Truss} = 11,000 + 110(200) = 33,000 \text{ in.-lb}$$

Assuming a truss depth of 10 in., the load in each rail is 1650 lb and the stress for a 0.1 rail area = 16,500 psi.

Assuming a temperature differential across the panel of 100° F, the tip deflection and slope of the panels are:

$$y = \frac{\alpha(\Delta T) L^2}{2D} = \frac{13(10^{-6})(100)(280)^2}{4} = 33.2 \text{ in.}$$

$$\theta = \frac{2(33.2)}{280} = 0.237 \text{ or } 13.6°$$

6.3.2.3 Weights. The weight breakdown of Concept 82 is:

| | |
|---|---|
| Solar Cells | 1590 lb |
| Honeycomb | 2160 |
| Arms | 120 |
| Miscellaneous Hardware | 50 |
| Contingency, 10% | 392 |
| Total | 4312 lb |

Since this concept is to provide power for the S-IVB workshop, it will be launched in conjunction with a CSM, SLA, and possibly the SSESM. Under these conditions, the manned Saturn IB to 200 n.mi. altitude payload capability would be sufficient only for 12- and 14-day missions.

However, the Saturn V payload to synchronous orbit using a 14-day CSM is 15,992 and can carry this concept easily. Also, unmanned Saturn IB launch configurations have excess capability.

6.3.2.4 Crew and Equipment. The following tasks are required of the EV crewman after automatic deployment of the solar cell array.

a. An inspection of the solar cell array and other components.

b. Release of any components that may have jammed, preventing a full deployment of solar cells.

c. Adjustment, repair, and/or replacement of solar cell, array, and/or other components.

The inspection task after deployment consists of checking for damage to cells during launch or deployment, that extension of solar cell panel hinges is complete, that all joints are secure, etc. The use of a bright paint, which is visible if a joint is not completely locked in place, is of great assitance to the EV crewman during the inspection.

The approximate time allotted for the inspection is 90 minutes. This estimate does not include any required repairs, replacement, or correction of malfunctioned components.

The above time estimate for the inspection task does not include the necessary pre-EV operations such as suit donning, cabin depressurization. The time required for these tasks totals approximately 90 minutes prior to the EV task and approximately 60 minutes for post-EV operations.

6.3.2.5 Concept Evaluation. The principal advantages for application of solar cells are long life, relative light weight, and minimum power conditioning. Solar voltaic power systems are limited by these characteristics: the need for energy storage for dark time operations and peak power, large extendable areas resulting in drag penalties under 200 n.mi. orbit, and requirements for orientation in orbit with limitations on maneuverability.

The cp-cg offset is unusually large due to the array location ahead of the spacecraft. Consequently, the environmental torques, especially with the S-IVB, will be high. The propellant requirements, and thus the minimum orbital altitude, will be increased. The use of momentum storage devices is indicated. However, the capacity of these devices may not be sufficient for the environmental torques at lower altitudes.

Coupling is precluded by requiring separation of control system frequencies and structural mode frequencies. This imposes restrictions on the array structural characteristics and on the vehicle maneuvering. Additional restrictions are placed on the structural frequencies by array deflection considerations. Additional restrictions are placed on vehicle maneuvering by the array solar orientation requirement. ACS exhaust impingement on the solar cells is a possible problem area. The attitude control system used is that of the CSM or the S-IVB. Momentum storage devices may be introduced, as noted above. Attitude sensors compatible with specific mission pointing requirements may also be added.

The deployment of this system is by positive actuators with redundancy and high reliability. The complete subsystem is fully automatic and within the state-of-the-art for space equipment.

EVA required during deployment consists primarily of inspection and possibly repair or replacement. Maintenance is chiefly replacement of solar cells as they become significantly degraded by the space environment. This replacement is facilitated by the modular construction of solar cell mountings involving four clips and a disconnect plug. The automatic deployment can be reversed making alignment and test entirely feasible on Earth, thereby minimizing the EVA task.

This concept indicates that a suitable array of solar cells can be launched and deployed for use in conjunction with the S-IVB workshop such that it will provide a reasonable power source for a year or more.

# SECTION 7

# MAGNETOMETERS

7.1 MAGNETOMETER USER REQUIREMENTS. Magnetometers must be isolated from large structures and mounted on non-ferromagnetic supports in such a position that disturbances in the surrounding plasma due to the spacecraft do not interfere with the measurements. Usually the magnetometer should be separated from the spacecraft by about three times the largest spacecraft dimension to escape large loops of current. When a large structure is deployed it is advantageous to have the magnetometer extended further beyond it on a boom. Each configuration requires separate consideration. It is suggested that ferromagnetic materials, such as a polished nickel solar collector, be kept distant from magnetometers in order not to influence the field.

Magnetometers are not large, and no more than two units per vehicle are necessary. Scalar instruments can be employed at low altitudes and vector (orthogonal arrays) instruments at high altitudes. It is necessary to know the magnetometer pointing attitude at high altitudes to about 1°, and at low altitudes to between 3° and 5°.

Thermal control is required for some types of magnetometers and for their preamplifiers. The rubidium-vapor magnetometer is probably the most temperature sensitive type, with the flux gate type somewhat less sensitive. Both types will probably require small heaters for temperature control.

A scalar instrument, for use at low altitudes where earth's magnetic field may vary from 20,000 to 100,000 gamma, requires a data storage system with a capacity for 100 bits/sec. A vector instrument for use at high altitudes, where weak fields on the order of 100 gamma exist, may require kilobits/sec depending on the type. Here measurement sensitivity as low as 1 to 5 gamma may be feasible.

In addition to the measurement of the earth's magnetic field it is of interest to measure the modulations of this field brought about by solar wind interactions at the magnetosphere boundary. This will require additional magnetometers capable of responding to frequencies ranging from 0.01 Hz to 1000 Hz similar to those on OGO.

A major magnetometer requirement is to periodically establish the offset. The output voltage, V, varies fairly linearly with the magnetic flux density, B, as shown in Figure 7-1.

In laboratory practice the offset is determined by inverting the magnetometer, since this cannot be done expediently by the crew if the magnetometer is on a long boom, some remote mechanical means of inverting the magnetometer is required.

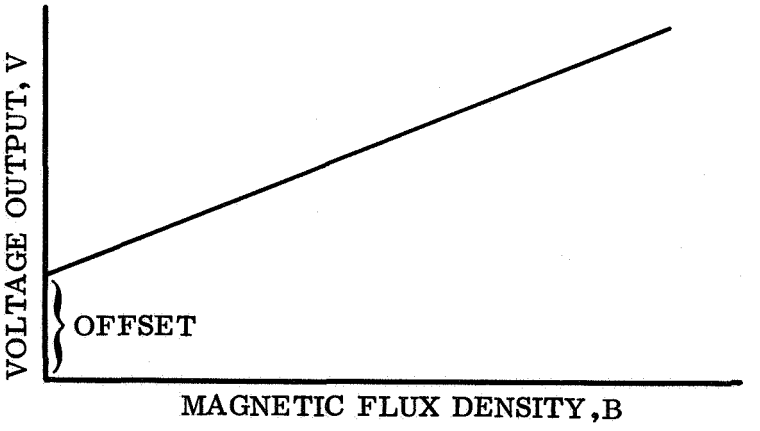



Figure 7-1. Magnetometer Voltage Offset

The magnetometer support boom must be compatible with the characteristics of the magnetometer listed in Table 7-1.

Table 7-1. Magnetometer Characteristics

| | |
|---|---|
| Lifetime: | Commensurate with that of spacecraft |
| Orbit Altitude: | Within the magnetosphere |
| Orbit Inclination: | Perpendicular and parallel to the earth-sun line |
| Orbit Eccentricity: | High for magnetic lapse rate measurements, low for magnetic isodistance lines |
| Field of Influence: | Magnetosphere first priority |
| Pointing Accuracy: | Approximately 1° at high altitudes, 3° — 5° at low altitudes |
| Sensor or Detector: | Rubidium-vapor or flux gate magnetometer |
| Field Strength: | 20,000 — 100,000 gamma at lower altitudes to approximately 100 gamma at high altitudes |
| Sensitivity: | < 1000 gamma for low altitudes to 1-5 gamma variations at high altitudes |
| Flux Directions Measured: | 4 $\pi$ steradians |
| Size: | Approximately 1 ft diameter by 6 in. high |
| Structure Tolerance: | Only in instrument itself |
| Deployment: | Extend on long boom, about three times as long as the largest spacecraft dimension, to avoid magnetic materials |

Table 7-1. Magnetometer Characteristics, Contd

| | |
|---|---|
| Operation: | Continuous |
| Important Measurements: | Solar flare magnetosphere modulations, lunar and terrestrial magnetic tail or wake studies |
| Date Rate: | 100 bps for low altitudes — scalar magnetometer. 1000 bps for high altitudes — vector magnetometer |
| Power Requirement: | Approximately 5 w for each scalar axis |
| Thermal Constraint: | Preamplifier heaters required, or preamplifier redesign necessary to withstand -54°C to +65°C. |

7.2 ANALYSIS OF CONCEPT 91, MAGNETOMETER BOOM. This concept consists of a magnetometer boom attached to a petal of a Concept 72 spacecraft. The boom is composed of eleven telescoping sections that extend 150 ft from the perimeter of the spacecraft. Each of the sections is triangular in cross section and is fabricated from sheet metal. (See Figure 7-2).

Attitude control of the magnetometer mounted at the end of the boom consists of keeping it aligned with the supporting spacecraft. Attitude control is provided to the extent required by boom deflections by vernier servo adjustments on two axes. In its launch configuration the boom is stowed in a package approximately 12 in. by 14 in. by 200 in., and weighs approximately 200 lb.

7.2.1 Structure and Mechanics. The individual parts of the magnetometer boom are formed from sheet metal after the truss work has been stamped or punched out leaving a beaded edge, which provides added stiffness to the finished welded-seam truss. There are eleven sections ranging from 0.86 in. on a side to 13.9 in. on a side.

Each movable section is provided with teflon coated shear surfaces which are adjustable to limit excessive side play, but allow proper movement for deployment. As the boom extends to its full length, cam operated latches lock each section in place and prevent free play in the boom assembly.

The electrical leads from the magnetometer and attitude and thermal control systems to the spacecraft are located in the center of the boom. As the boom is deployed, the electrical cable is carried along and is supported at regular intervals by standoffs located on the inside of the boom.

Deployment is accomplished with a system of pulleys and cables. The cable is wound on a winch, which provides the force to extend the various sections of the boom. The deployment system has been designed to use a minimum length of cable which in turn minimizes the drum size.

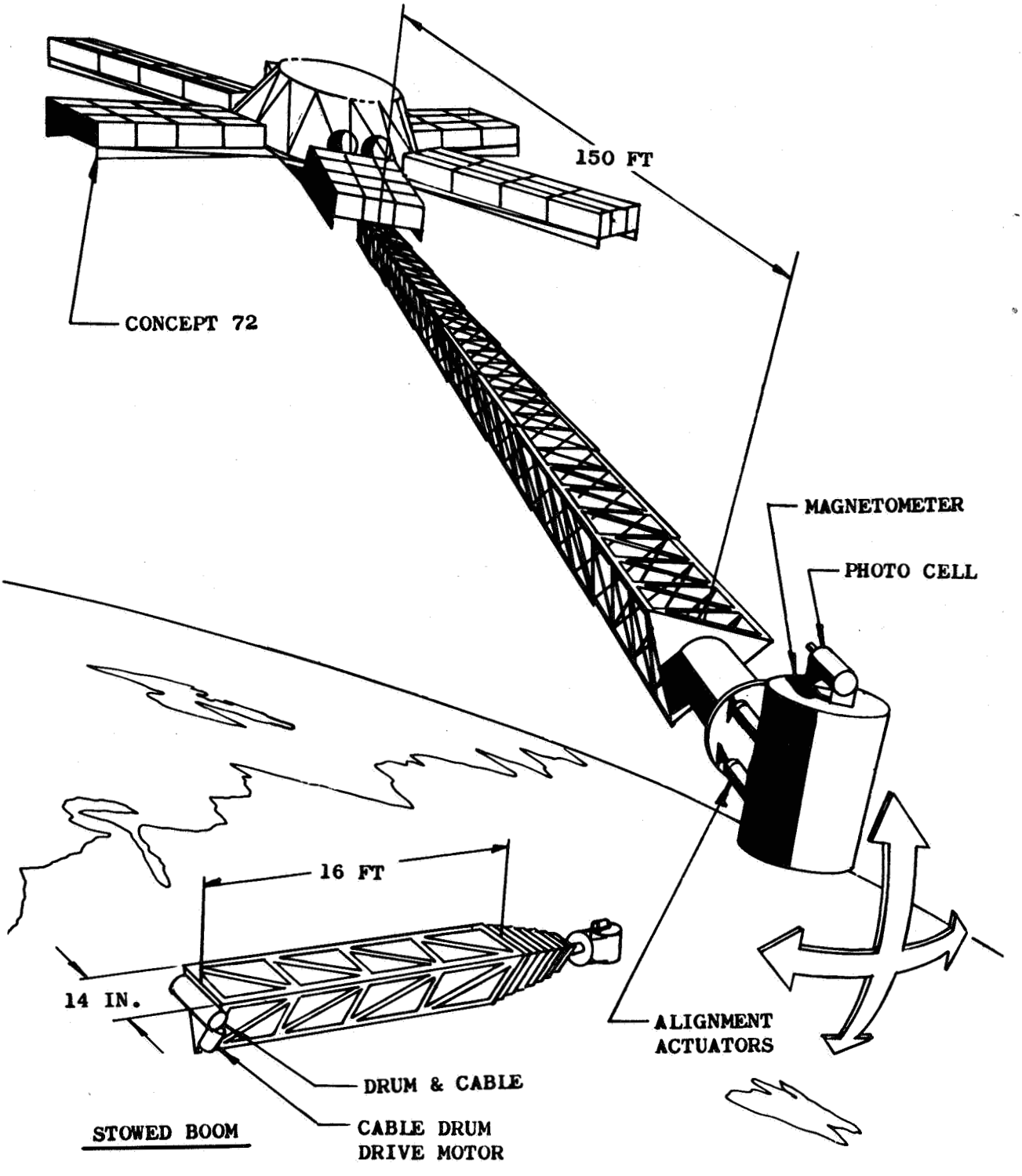



Figure 7-2. Concept 91, Magnetometer

At the end of the boom there is a control system to maintain alignment of the magnetometer. A photo cell senses boom deflection and corresponding misalignment of the magnetometer in relation to the spacecraft and causes the proper actuator to correct for the misalignment.

Since the boom is a truss it will partially expose its far side to solar radiation and thereby reduce the structural distortion that would result from unequal thermal expansion. The use of inside and outside surface finishes that will absorb or reflect solar energy to the degree necessary to maintain a balanced temperature around the truss will reduce thermal distortion to an acceptable degree.

Surface area and inertia of the boom will increase the environmental control torque requirements of the vehicle to which the boom is added. The inertial increase due to the boom will increase the maneuvering and decrease the limit cycle requirements. The percentage increase should be a fraction of that without the boom. Associated propellant increase should not result in unacceptable payloads unless the original configuration was marginal. Annual momentum requirements for the boom are listed in Table 7-2. Propellant weights are included for all jet control using the CSM.

Table 7-2. Propellant Requirements — Additional Momentum and Propellant Requirements, One Boom, 200 n.mi., One Year

| ADDITIONAL MOMENTUM REQUIRED (ft-lb-sec) | | |
|---|---|---|
| | One Axis | All Axes |
| Gravity Gradient | $1.32 \times 10^6$ | $2.66 \times 10^6$ |
| Aerodynamic Drag | $6.32 \times 10^5$ | $1.26 \times 10^6$ |
| Maneuver* | Increased | |
| Limit Cycles* | Decreased | |

| ADDITIONAL PROPELLANT REQUIRED BY ALL JET SYSTEM, CSM (lb) | | |
|---|---|---|
| | One Axis | All Axes |
| Gravity Gradient | 635 | 1270 |
| Aerodynamic Drag | 301 | 602 |
| Maneuvers* | Increased | |
| Limit Cycles* | Decreased | |

*Independent of orbital altitude

Utilization of a second boom for balancing environmental torques could be helpful for vehicle missions with orientation aligned to desirable local vertical positions.

The jet torques of the CSM are likely to cause tip motions of several feet, but this should not degrade the 1° pointing tolerance. Problems could occur if the boom were attached to a vehicle having experiments with tighter tolerances. One degree at the tip is three feet of motion, and the anticipated deflection is of this order. Since the modal frequency is low, the actuators at the end of the boom can probably hold the magnetometer within the tolerance. A mechanical damper could be introduced.

7.2.2 Stress Analysis. The boom structure was examined for launch and docking loads and found to be satisfactory. If the load factor due to vibration induced by maneuvering can be maintained at 0.1 g or less, the tip deflection is 4.2 in. and the slope at the tip is at least 0.8°.

This approaches the allowable deviation for magnetometer use, and it is apparent that maneuvering accelerations must be kept very low. Preliminary estimates indicate that effective load factors of 0.01 to 0.02 g will result from full application of CSM control. Following such application the deviation will exceed limits until structural damping reduces the vibration.

7.2.3 Crew and Equipment Functions. The erection of the Concept 91 magnetometer is fully automatic. The following EV tasks are required after deployment:

a. Inspect magnetometer components, including an alignment check. (In orbit, alignment consists of coordinating the sensing device and the vernier servo to the magnetometer null. This is accomplished by one astronaut in EVA.)

b. Repair or replace any damaged component.

c. Release any fouled equipment that does not completely deploy.

The inspection task would also require the visual inspection of each boom section latch to insure proper deployment of the automatic system. A portable, battery powered drive motor carried by the astronaut could be plugged in to serve as a back-up to the automatic boom extension system.

## SECTION 8

## METEOROID DETECTORS AND COLLECTORS

8.1 METEOROID DETECTOR USER REQUIREMENTS. Three basic types of meteoroid detectors are of interest: gelatin, wire mesh, and crystal. These detectors permit determination of mass and velocity, allow crater characteristics to be studied, and enable recovery of undamaged particles from emulsions. The gelatin type is not expected to be a large structure but must be examined periodically for possible particle retrieval. Most types require periodic retrieval and replacement of detector surfaces or trapping mechanisms for onboard studies.

There is also a need to expose surfaces designed for meteoroid protection in order to evaluate various impact phenomena and provide realistic calibrations.

After deployment, the detectors should be inspected and tested. Crystal detectors may be sensitive to thermal changes and might require superinsulation over some surfaces.

During meteor showers it would be desirable to face the detectors toward incoming particles. Attitude control and pointing capability is therefore necessary.

User requirement data are provided in Table 8-1.

Table 8-1. Micrometeoroid Collector User Requirements

| | |
|---|---|
| Life Time | 1 year. |
| Orbits | Altitude: various, depending on investigation desired. Inclination: normal to earth's trajectory. Eccentricity: preferably circular (zero). |
| Field of View | $2\pi$ steradians. |
| Directional Resolution | Desired to ± 10 deg, but not necessary. |
| Pointing Accuracy | Within 10-deg array should point perpendicular to earth's trajectory. |
| Detector | Gelatin emulsions, thin plates, membranes, wire grids, or crystals. |
| Sensitivity | Depends on individual investigation. |
| Collecting Area | As large as possible (up to 15,000 sq ft for some investigations). |
| Physical Dimensions | In case of wire grids and emulsions, individual cells of 2 by 2 ft. |
| Structural Tolerance | To synthesize the incoming directions of several particles, individual cells must be aligned within resolution capability. |

Table 8-1. Micrometeoroid Collector User Requirements, Contd

| | |
|---|---|
| Significant Measurements | Meteor stream peaks; fuel tanks and wall panel penetration. |
| EVA | Post-micrometeoroid detector replacement, handling, and maintenance, removal and replacement of particle collectors possibly weekly or monthly. |
| Materials | For some applications, gelatin, which may have to be kept warm to remain gelatinous. |
| Electrical Power | 10 watts/sq meter. |

## 8.2 ANALYSIS OF PROVIDED CONCEPTS

8.2.1 Concept 14, Meteoroid Deep Space Satellite. The Z blade meteoroid satellite concept is designed to be launched by a Saturn IB/Centaur vehicle into an interplanetary mission. For this purpose it appears, by a limited analysis of a four volume report (Martin Company ER 13964), to be well conceived.

To convert the concept to the user requirement established for this study requires some sacrifice and modification. With minor redesign the satellite can be stowed for launch in a Saturn IB unmanned configuration for a 200 n.mi. orbit. An unmanned launch to synchronous orbit was not considered in this study. See Appendix C. A reduction in size sufficient to permit stowing in a manned configuration would be a severe penalty to this concept.

To obtain the desirable feature of selective pointing, an active ACS is both required and feasible, but such a modification defeats some of the advantages of spin stabilization and the resultant concept configuration.

However, eliminating spin has three additional advantages: it permits recognition of target direction in the plane perpendicular to the axis, it facilitates EVA resupply and service of sensors, and it eliminates the rotation that might not be compatible with additional experiments.

The meteoroid satellite is designed for deep space missions, and to provide interplanetary micrometeoroid data relating to flux density, velocity, direction, mass, and impact.

The sensors, arranged in four panels of "Z" cross-section, provide three dimensional space coverage. See Figure 8-1. The four main panels are independently attached to the central support structure, and eight auxiliary planes (two per arm) are hinge-mounted on the main planes in a symetrical arrangement that provides stability against the effects of solar pressure.

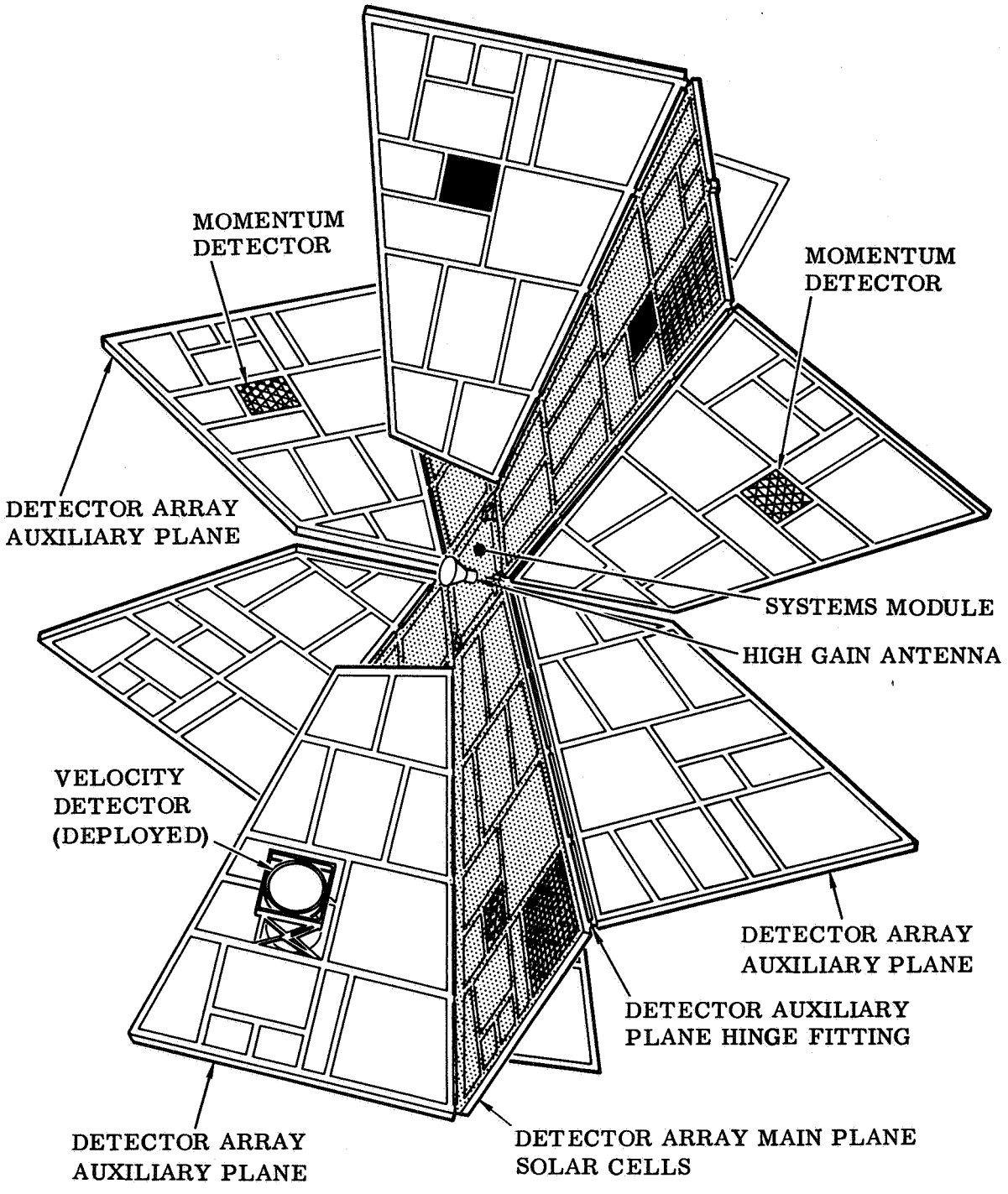



Figure 8-1. Concept 14, Deep Space Meteorite Satellite (Martin)

The spacecraft is spin stabilized with its axis parallel to the plane of the ecliptic. The side of the spacecraft adjacent to the booster during launch is oriented to the earth and the sun, and has the high gain antenna and solar cells installed on it. In addition to the high gain antenna, the vehicle has two omnidirectional turnstiles, one on the earth's side, the other on the opposite side. Attitude sensing equipment consists of a sun sensor and two single-axis magnetometers.

8.2.1.1 Structure and Mechanics. The four "Z" arms are mounted from the sides of the square central support structure, independently of each other, and are deployed from the stowed position approximately 78° where they are locked in a horizontal plane before spinup and separation. The eight auxiliary planes (two per arm) are hinge-mounted on the main planes and locked. They are deployed independently (120°) by torsion springs dampened by dashpots and locked in position by a linkage.

The total solar cell area on the spacecraft is 146.8 sq ft: 19 on each of the main planes, 4.8 on the long edge of each auxiliary plane facing the sun, and 12.9 on the outboard edges of the main auxiliary planes.

The primary onboard sensors are the micrometeoroid penetration detectors. These can be provided as either capacitors or pressure cells of annealed stainless steel type 347 in the following thicknesses and areas:

| THICKNESS (mil) | CAPACITOR SENSOR AREA (sq ft) | PRESSURE SENSOR AREA (sq ft) |
|---|---|---|
| 1 | 77.8 | 82.56 half tube |
| 3 | 410.2 | 407.28 tube |
| 8 | 916.9 | 997.68 quilted panel |
| 16 | 1221.3 | 1406.56 quilted panel |
| | 2626 | 2894 |

Six velocity gages and six momentum gages are provided, each having a sensor area of 3 sq ft for a total of 36 sq ft.

The onboard systems are contained in an equipment module, removable as a unit from the spacecraft central support structure. Individual systems are separately removable from the module, even on the launch pad.

A separation motor (NASA-LRC Cygnus 5 solid rocket) is located on the centerline of the spacecraft utilizing the antenna mount. The spacecraft spinup system is a mono-propellant ($H_2O_2$) system with two motors mounted on the detector array support arms.

For launching, the folded satellite is contained within the payload envelope for the Saturn IB unmanned launch vehicle. See Figure 8-2. It is mounted on a support structure installed on the SLA.

In orbit, the satellite weighs 5,042 pounds with capacitor sensors or 4,818 pounds with pressure cell sensors. It has a span of 540 inches and a height of 289 inches.

The original concept relies on the initial spin to maintain its attitude in deep space. An active attitude control is considered below for a synchronous altitude orbit and a 200 n.mi. orbit.

8.2.1.2 Dynamics Attitude Control. User requirements for this concept were assumed to be for a one-year life in either a low or high orbit. Only 200 n.mi. and synchronous orbits are considered. It is assumed that the vehicle will be primarily operated in the unmanned state with perhaps occasional operation while docked to the Apollo CSM. Maneuver capability and the capability of maintaining a fixed inertial position to within $\pm 1^{\circ}$ are assumed to be the attitude control requirements.

Analysis is based on using a 3-axis attitude control system utilizing rim mounted jets with thrusts of 0.1 lb. An Isp of 300 sec is assumed and a lever arm of 22.4 ft is used. At the lower altitudes, where environmental torques are most severe, momentum storage devices might be used to handle oscillatory torques.

The Martin Company reports that the expected structural frequency of the panels is 3 Hz which will be sufficiently removed from the control system operating frequencies to preclude coupling. The vibratory motion that does occur should be of such a level that it will not degrade the mission.

The average yearly momentum and propellant requirements per axis for each of two axes to overcome environmental torques are:

UNMANNED

| ORBITAL ALTITUDE | MOMENTUM | PROPELLANT |
|---|---|---|
| 200 n.mi. | $1.4 \times 10^6$ ft-lb-sec | 210 lb |

MANNED

| ORBITAL ALTITUDE | MOMENTUM | PROPELLANT |
|---|---|---|
| 200 n.mi. | $3.8 \times 10^6$ ft-lb-sec | 570 lb |

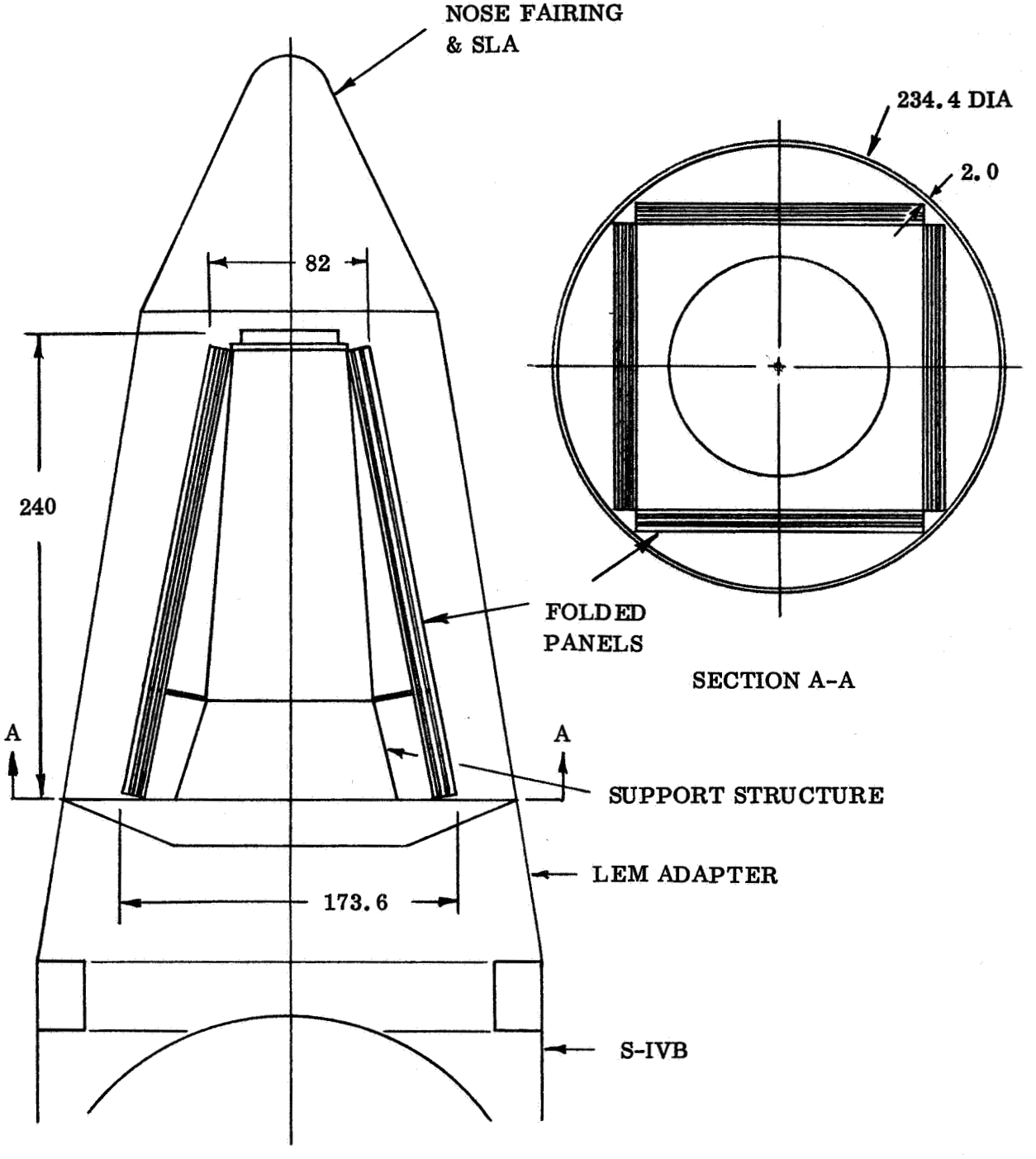



Figure 8-2. Packaged Configuration

Use of a momentum storage device at 200 n.mi. orbits could reduce propellant requirements greatly. A momentum storage requirement on the order of 150 ft-lb-sec for the unmanned structure is indicated.

With the CSM docked to the meteoroid collector, the momentum storage requirement is 350 ft-lb-sec. Propellant requirements for the third axis were not calculated. However, they should be lower than for the other two axes due to vehicle symmetry about the roll axis which causes environmental torques to be very low.

The angular accelerations the ACS is capable of providing about the pitch and yaw axes are:

| | |
|---|---|
| Unmanned | $3 \times 10^{-4}$ rad/sec$^2$ |
| Manned | $4.5 \times 10^{-5}$ rad/sec$^2$ |

These should be sufficient to meet any expected maneuvers.

8.2.2 Concept 18, Concertina Meteoroid Detector (Fairchild Hiller). This concept is a meteoroid detector and/or collector. It consists of three radial booms, each of concertina construction and deployment wherein the hinged panels contain or support the detectors or the collectors. See Figure 8-3.

Assuming the stated gross launch weight of 2,000 lb for 2,000 sq ft of detector panels does not include subsystems, an estimated weight allowance indicates the satellite can be launched by a manned Saturn IB into a 200 n.mi. or a manned Saturn V to synchronous altitude orbit.

In the stowed position the hinged side panels fold inward and the lower and upper panels fold outward. The panels not only function as part of the structure, they are also utilized as capacitor type sensor cells for meteoroid detection. Springs located at the hinge joints provide the deployment energy. Stowed volume is 100 cubic feet. Solar cell panels are mounted at the ends of the three trusses.

8.2.2.1 Structure and Mechanics. Each truss consists of 667 square feet of cells. Assuming that the trusses are square shape with three feet on a side, truss length is 55.5 feet. Adding the solar cell panels and the center structure forms a three-arm truss structure approximately 117 feet in diameter at the periphery of the trusses. The trusses, assuming cells one-foot wide and an inch thick, can be retracted into stowed lengths of about 9 feet for a stowed diameter of 18 to 19 feet.

Truss configurations other than square, such as rectangular or triangular, provide almost unlimited variations in panel display in combination with two, three, four, or five-arm configurations. Increasing the number of arms and cell sizes could possibly increase the surface area to 4000 sq ft or more.

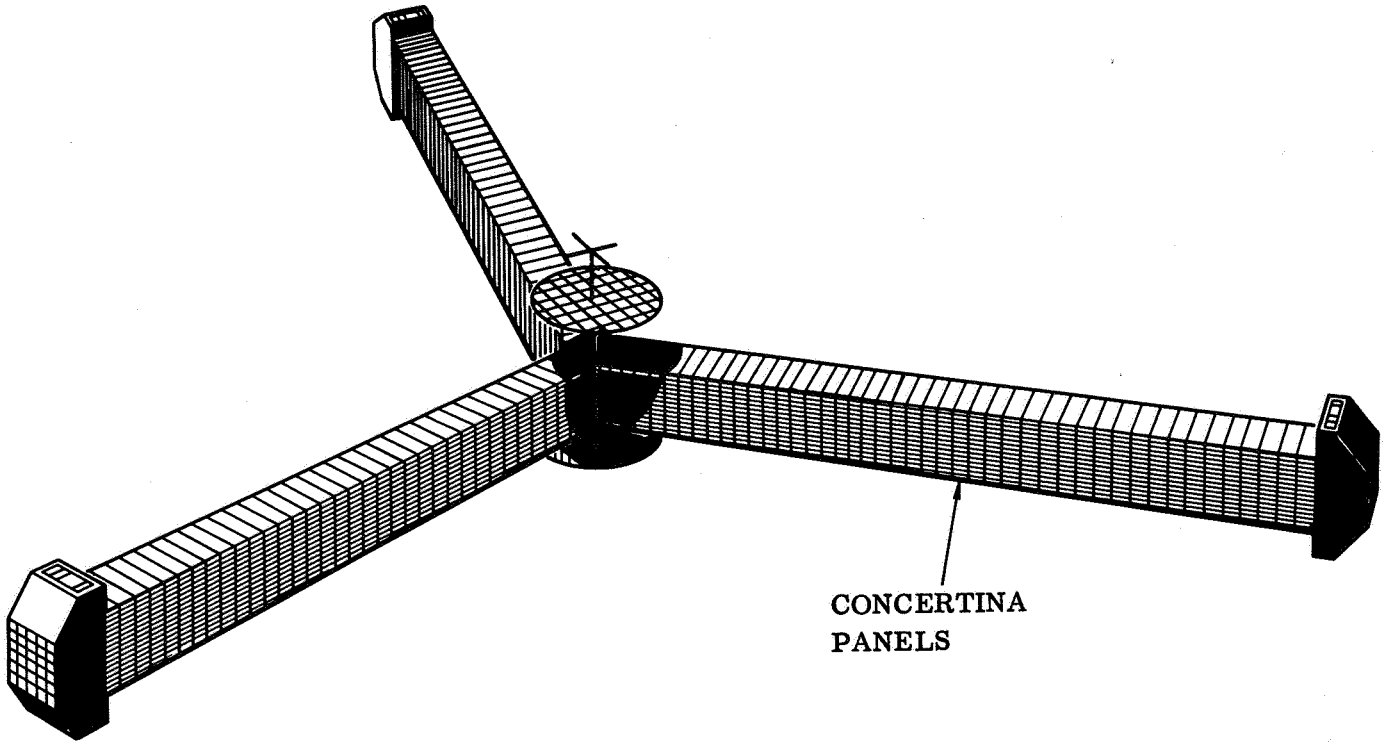



Figure 8-3. Concept 18, Concertina Meteoroid Detector/Collector (Fairchild Hiller)

The concertina-type truss with the spring loaded joints is a fairly rigid structure. The springs tend to keep the truss from buckling when subjected to bending loads. This is a desirable feature when the sensor cells require replacing by an astronaut.

Use of the device as a meteoroid collector is difficult since the dimensional differences between capacitor-type sensor cells and collector panels panels practically prohibit their being installed in concertina-type trusses.

Deployment is accomplished by torque springs located at the panel hinge joints. To prevent the panels from rotating at excessively high angular velocities, the deployment of the trusses requires restraining, such as by a cable system, wherein the trusses are tied together at the governing drum to insure controlled deployment in unison. A considerable amount of design effort is required to resolve some phases of this concept, such as feeding out the electrical wiring along with the trusses. However, the deployment is considered feasible and has the attractive feature of being essentially fully automatic. Simplicity is derived in part from the simple and repetitive pivot joint mechanism. See Figure 8-4.

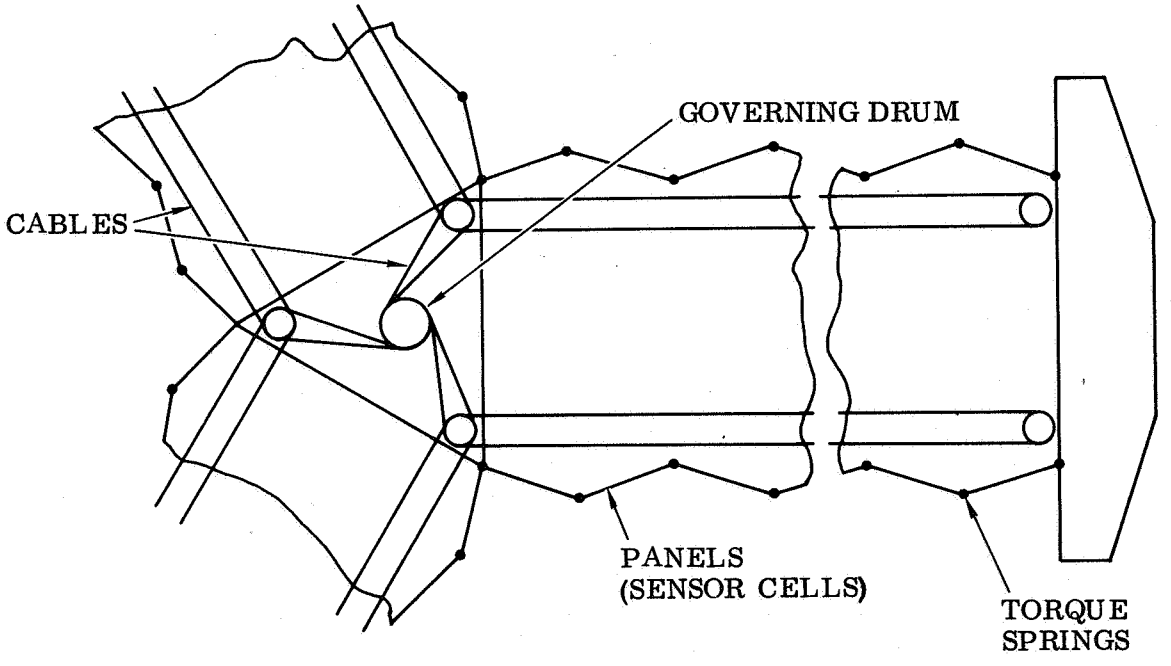



Figure 8-4. Deployment Restraint System

The subject concept can be packaged for launch aboard manned or unmanned versions of the S-IB for low orbit missions or the S-V for manned synchronous orbit mission. To withstand launch induced loads, the packaged truss requires extensive support since the concertina trusses are sensitive to acceleration and vibration when collapsed.

8.2.2.2 Dynamics/Attitude Control. User requirements for this concept were assumed to be for a one year life in either a low or high orbit. Only 200 n.mi. and synchronous orbits were considered. It was assumed that the vehicle would be primarily operated in the unmanned state with occasional operation while docked to the Apollo CSM. The capability of maintaining a fixed inertial position to within $\pm 1^{\circ}$ was taken to be the primary attitude control system requirement. Requirements on maneuver rate were assumed to be low.

Analysis was based on using a 3-axis attitude control system utilizing rim mounted jets with thrusts of 0.1 lb. An Isp of 300 sec was assumed. At the lower altitudes, where enviromental torques are most severe, momentum storage devices might be used to handle oscillatory torques and reduce the propellant requirements.

The average yearly momentum and propellant requirements per axis for the unmanned vehicle to overcome environmental torques are, for a 200 n.mi. orbit:

| | TRANSVERSE AXES | LONGITUDINAL AXIS |
|---|---|---|
| Momentum | $2.0 \times 10^6$ ft-lb-sec | $1.3 \times 10^6$ ft-lb-sec |
| Propellant | 125 lb | 72 lb |

8.2.2.3 Electronics. The telemetry system, desirable for monitoring satellite performance, can have some common equipment with the command link. It is also possible to incorporate a transponder type of a tracking beacon into this composite system.

The telemetry system would be used to monitor voltages, pressures, temperatures, and other low information rate sources. The band width required to telemeter 60 of these measurements is approximately 1 kHz. The type of modulation used would be either frequency modulation (FM) or pulse code modulation (PCM). The current trend is with PCM which is described below.

The system will be in the lower VHF range, around 300-400 MHz, because this equipment is more available. A block diagram of the system is shown in Figure 8-5.

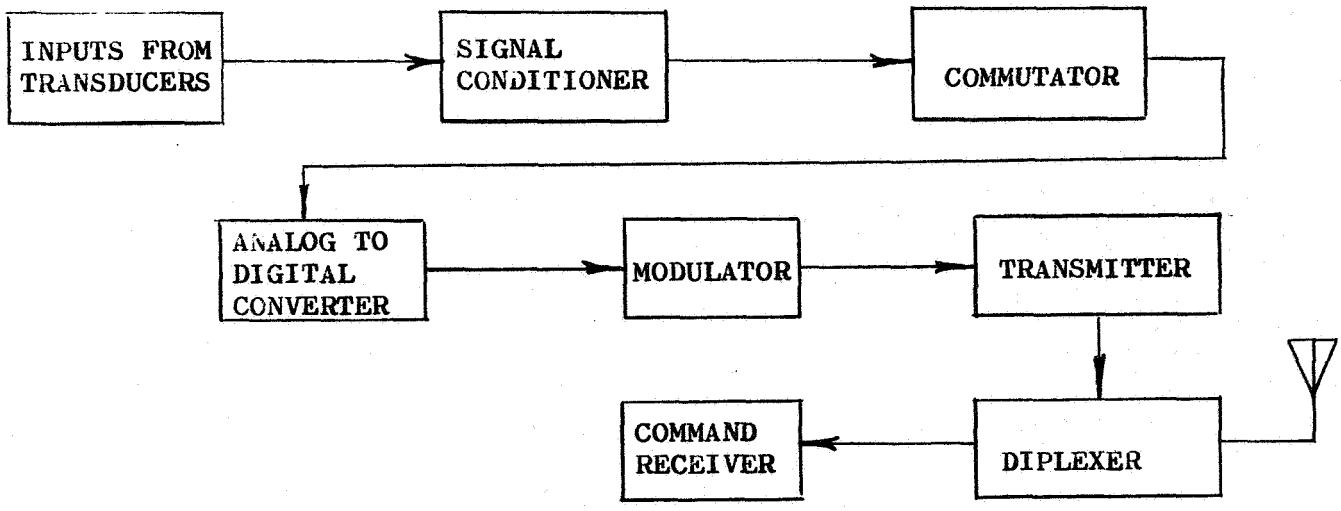



Figure 8-5. Telemetry-Command System

The transmitter power requirements are easily met with a one watt output. This assures zero gain at the satellite antenna and 35 db gain at the ground antenna. The narrow information band width (1000 bits per second) allows the noise improvement factor to be 16 db while maintaining a 15 db signal-to-noise ratio at the receiver. Size, power, and weight of this system is shown in Table 8-2.

Table 8-2. Electronics System Weight, Power, and Volume

| | WEIGHT (lb) | POWER (watts) | VOLUME ($in.^3$) |
|---|---|---|---|
| Signal Conditioner | 1 | 0.5 | 24 |
| Commutator | 1 | 0.5 | 16 |
| A to D Converter | 2 | 1.0 | 32 |
| Modulator | 2 | 3.0 | 14 |
| Transmitter | 5 | 5.0 | 20 |
| Diplexer | 2 | 0.0 | 12 |
| Receiver | 25 | 1.0 | 200 |
| Transducers and cables | 12 | 5.0 | 210 |
| Total | 50 | 16.0 | 528 |

8.2.2.4 Crew and Equipment Functions. The concept is capable of deploying automatically. Once in space, it may require periodic maintenance and replacement of sensor cells.

In the event the panels only partially deploy, the EV crew can be useful in correcting the malfunction and avoiding failure of the mission. Due to the nature of the spring-loaded hinge structure, however, the type of correction tasks should be carefully evaluated from the crew safety standpoint. A restraining device to control deployment speed is considered a necessity for crew participation in the malfunction-correction role. Deployment control must also be subject to over-ride by the crew.

8.3 - CONCEPT 85, METEOROID DETECTOR AND COLLECTOR PANELS. This concept, developed by the Convair division of General Dynamics, is a meteoroid collector and detector capable of launch by a manned Saturn IB vehicle to 200 n.mi. or by a manned Saturn V to a synchronous altitude orbit. It has eight panels that fold out from under the LEM support rack and support 2 ft x 2 ft x 2 ft collector or detector modules. The exposed area is 320 square feet and the entire experiment weighs 9,900 lb.

This amount of collection surface is sufficient to obtain substantial data on the smaller particles, but is insufficient to obtain practical data on the larger materials because of their infrequent occurrence.

The concept provides a practical means of obtaining meteoroid data in orbits of from 200 n.mi. to synchronous altitudes. Its weight, about half the available payload of a Saturn manned launch vehicle, makes it an appropriate candidate for launch in combination with other compatible experiments. It meets all the user requirements, as determined by this study, including the desired $10^{\circ}$ direction resolution and accommodation of 2-ft deep detector/collector modules.

This concept utilizes the volume below Station 117.3 of the LEM adapter for storage of the complete system. The eight panels that support the meteoroid collector modules form the sides of an octagon and are attached with two hinges each to the octagonal tubular support rack. Linear actuators deploy the panels to pointing position. Power to operate the systems can be supplied by solar cell arrays stowed in the tapered LEM adapter above Station 117.3.

Deployment is automatic, but requires EVA for initial inspection and subsequent film collection.

The meteoroid collecting modules employed in this concept are 2 ft x 2 ft x 2 ft deep and consist of 6 banks of wire mesh in a square 6 in. x 6 in. grid. Charge, velocity, direction, and location are timed by an electronic timer as the meteoroids pass through the grids to the movable roll of foam or film where they are collected. The meteoroid size and flux is from $10^{-9}$ to $10^{-12}$ grams and about $10^{-11}$ gram/sq meter. The electronics equipment (amplifiers, etc.) is mounted on the sides of the box.

8.3.1 Structure and Mechanics. The structural tolerance will be minimized by machining stops on the eight blades to ±0.002 between mating surfaces. Adjustable screws will be provided to align the modules (on assembly). The panel support structure (Figure 8-6) is a welded tube space structure, octagonal in shape, and 135 in. diameter by 120 in. deep. Its tubes are 2219 aluminum 2 in. in diameter and with an L/P less than 150. The geometry of the octagon allows eight panels to be folded into the LEM adapter volume below Station 117.3 with 12 in. clearance all around. The panels themselves are trapazoids (55 in. x 80 in. x 135 in. x 3 in. thick) of plate stringer construction. The 80 meteoroid modules 2 ft x 2 ft x 2 ft are attached to the inside face of the panel with four adjustable bolts for each module. Choice of the plate stringer construction will provide a conductive path for quickly radiating the accumulative heat to space.

Each panel is locked in the stowed position to the truss support structure with two explosive nuts (Hi-Shear or equivalent) at the tip end. The deployment mechanism rotates the panel into pointing position after the explosive bolts are fired to free the tip end of the panel.

The deployment mechanism is a simple linear (screw jack) actuator operated by a sealed motor with worm and gear. Two screw jacks per blade will be used for redundancy. Either one or both will rotate the blade to pointing position ($0^\circ$ to $90^\circ$). EVA participation will consist of inspection, repair, and replacement of the modules.

The components of the various systems are attached to the panel support structure with sheet metal brackets or machined aluminum fittings. They are located in a position to provide maximum moment of inertia about the three axes.

A standard docking ring is provided on the panel support structure for the CSM. The attitude control system of the CSM is used to maintain pointing position on target during data gathering times.

8.3.2 Attitude Control and Dynamics. The experimental concepts pertinent to dynamics/attitude control are as follows. Both high and low orbits are considered as possible environments for the experiment. The desired lifetime is one year. The attitude control system should be capable of controlling the vehicle attitude to $\pm 1^\circ$.

Active control of the structure about all three axes is required. The preliminary analysis uses the Apollo CSM attitude control system, which is capable of the desired pointing accuracy. The use of momentum storage devices may be required to reduce propellant requirements. Since propellant usage due to limit cycling is high and at 200 n.mi., so is the propellant usage.

The maneuvering during reorientation of the structure is accomplished with the CSM jets or by momentum devices. Since reorientation of the structure should not be necessary more often than once per several hours of operation, propellant weight requirements should not be excessive.

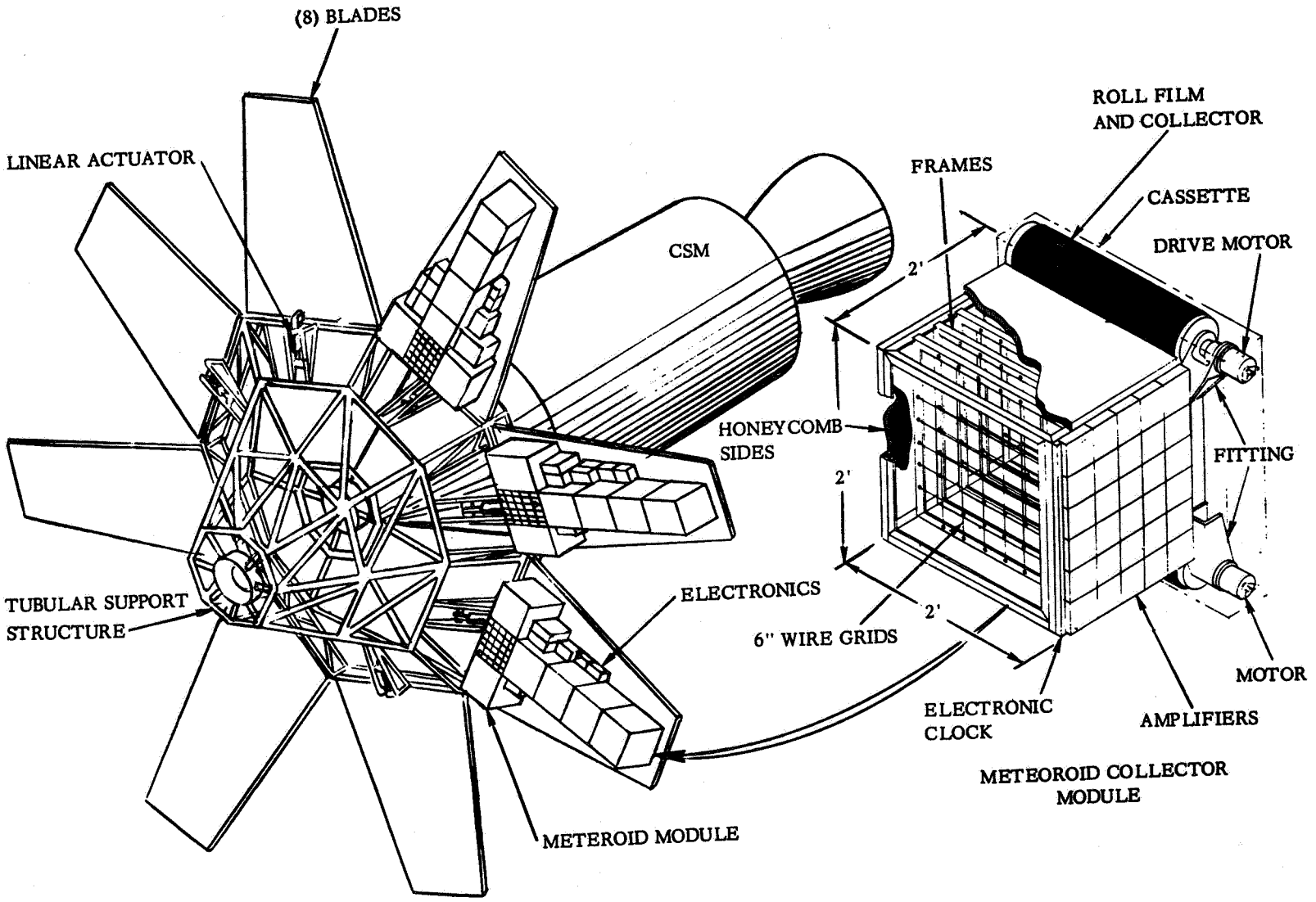



Figure 8-6. Concept 85, Meteoroid Detector and Collector

No structural dynamics problems are expected.

Principal moments of inertia for the structure when coupled to the CSM are: about the transverse axes, 26,000 slug-ft$^2$; about the longitudinal axis, 100,000 slug-ft$^2$.

Propellant usage due to limit cycling is quite high. About the transverse axes, 4,100 lb per axis per year is required. About the longitudinal axis, 1100 lb per year is required.

The CSM jets will give angular accelerations about the transverse axes of $4.9 \times 10^{-2}$ rad/sec$^2$ and $1.1 \times 10^{-2}$ rad/sec$^2$ about the longitudinal axis.

8.3.3 Weights. The total weight for this concept is:

| | |
|---|---|
| Experiment (542 lb × 8) | 4,336 |
| Rack | 1,500 |
| Electronics | 200 |
| Instr., Comm. & Wiring | 844 |
| Fuel Cell System | 1,736 |
| Thermal Control | 1,342 |
| | 9,958 |

The total launch weight of this concept is 9,958 lb. This weight permits an unmanned Saturn IB launch to place it at a 200 n.mi. orbital altitude. Under these conditions the experiment consumes about 43% of the available payload. Total attitude control propellant requirement will be partly supplied by tankage to utilize the additional payload available and relieve the CSM which is put in orbit with a second launch.

8.3.4 Crew and Equipment Functions. The automatic erection of this meteoroid collector is monitored by the crew. Following are tasks that are required after deployment.

a. Inspection of the detector.

b. Repair and/or replacement of any collector component that may be required.

c. Removal and replacement of the meteoroid film cassettes when required.

The visual inspection of the eight panels or blades insures they have deployed fully into the desired position. Damage incurred during deployment is detected by visual inspection of the additional components, such as the modules, tubular support structure, and linear actuator.

This inspection requires a minimum of 90 minutes. Approximately five minutes is required for removal and replacement of each of the cassetts (80). Although this would take approximately 6-1/2 hours, it may not be necessary to remove and replace all 80 cassetts at the same time. In the event that other components require repair and/or replacement, additional time is allotted. The estimate of 90 minutes for inspection and 5 minutes for replacement of each cassett does not include the time required for pre- and post-EV activity such as donning and removal of suit. These operations will require approximately 90 minutes for pre-EV activity and 60 minutes for post-EV activity.

Barring extensive requirements for repairs or replacement (other than the cassetts) the total task of pre-EV preparation, inspection, removal, and replacement of 80 cassetts, and post-EV porcedures will require a total time of approximately 11 hours for one astronaut.

# SECTION 9

# EVALUATION OF CONCEPTS

This section presents the results of the evaluation of over 40 concepts previously analyzed. A method of rating concept performance when measured against user requirements in each application area is developed. Significant research and technology requirements for each concept, including prerequisite orbital experiments, are identified. From these data preliminary costs and schedules have been assembled for each concept. Finally, the results of the concept evaluation, based on criteria developed in coordination with the NASA/MSFC project monitor, are presented.

9.1 USER VS. CONCEPT SPECIFICATIONS. A user requirements/concept comparison and rating form was derived (by consideration of critical parameters of each user requirement category) to determine what factors are common to most of the concepts. The format identifies the concept experiment, enables a choice of critical parameters for evaluation, gives an estimate of maneuverability, and gives a calculation of a relative concept rating against the identified user requirements. The top line in each table shows a summary of the ideal user requirements utilized in this study; e.g., R-1 for long wave radio astronomy in Table 9-1.

The critical performance parameters for most of the concepts considered are: effective aperture area, the beamwidths at lowest practicable frequency, beamwidths at the highest practicable frequency, the basic bandwidth of the device (if applicable), and an estimate of the largest critical structural deviation. (Since most of the large-structure candidates evaluated were electromagnetic radiation energy collectors, these particular performance parameters apply. However, many of the concepts include provisions for universally adapting structure depolyment to a particular frequency.)

During the study, effort was applied to obtain pointing, stability, and maneuvering characteristics compatible with the effective beamwidths for each of the large structures considered; if the performance could be attained, it is reported as "yes"; if not, then "no" is entered, and the precession on other natural characteristics of the structure will be used to sweep the device over the desired observation areas.

The following procedures are used in the evaluation. A table is prepared for each user requirement category. The ideal set of requirements is entered in the top data line in accordance with the user requirement study results. Then candidate concepts that have some probability of some performance in the particular user requirement category are entered in the table. For this study only four of the critical parameters are evaluated.

The rating terms are as follows:

(A) = ratio of aperture areas (indication of energy or particle collection efficiency)

(F) = inverse ratio of products of beamwidths (fields), for lowest frequency (gives an indication of interference rejection capability of the device, since a small solid angle is usually desired)

(B) = ratio of bandwidths (implies a basic capability of receiving signals, energy, or particles with respect to some ideal portion of the spectrum)

(D) = inverse ratio of total structural deviations (implies how well the critical portion of the structures meet ideal requirements)

When these four terms are multiplied together, they tend to give a mutually compensated figure of merit for the electromagnetic radiation or particle receiving device evaluated. Where the idealized requirements are met or exceeded for the particular device, the ratio value is given as "one".

The following mathematical approximations show how the ratios have been computed in Tables 9-1 through 9-14:

9.1.1 Bandwidth Concept Rating Method

$f_{C_1}$ = lowest concept frequency

$f_{C_2}$ = highest concept frequency

$f_{R_1}$ = lowest required frequency

$f_{R_2}$ = highest required frequency

$$\text{(B)} \quad \text{concept rating} = \frac{f_{C_2} - f_{C_1}}{f_{R_2} - f_{R_1}} \quad \text{for } f_{C_2} \leq f_{R_2} \text{ and } f_{C_1} \geq f_{R_1}$$

$$\frac{f_{R_2} - f_{C_1}}{f_{R_2} - f_{R_1}} \quad \text{for } f_{C_2} > f_{R_2} \text{ and } f_{C_1} \geq f_{R_1}$$

$$\frac{f_{C_2} - f_{R_1}}{f_{R_2} - f_{R_1}} \quad \text{for } f_{C_2} \leq f_{R_2} \text{ and } f_{C_1} < f_{R_1}$$

9.1.2 Aperture Concept Rating Method

$$(A)\ \underset{\lim \rightarrow 1.0}{\text{concept rating}} = \frac{\text{concept aperture}}{\text{required aperture}}$$

9.1.3 Beamwidth Concept Rating Method

$$(F)\ \underset{\lim \rightarrow 1.0}{\text{concept rating}} = \frac{\text{required beamwidth @ } f_{R_1}}{\text{concept beamwidth @ } f_{C_1}}$$

9.1.4 Structure Perimeter Deviation Rating Method

$$(D)\ \underset{\lim \rightarrow 1.0}{\text{concept rating}} = \frac{\text{required deviation}}{\text{concept deviation}}$$

# Table 9-1. R-1, Long Wave Radio Astronomy

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEUV-ERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | *(F) DEG | $f_1$ MHz | DEG | $f_2$ MHz | | SUR-FACE | (D) PERIM. DEFL. | | | |
| R-1 | Long Wave Radio Astronomy | 24,000 | 4,500 | $2(10^4)$ | $3.2(10^7)$ | <10x10 | 0.5 | <<10x10 | 10 | 0.5-10 MHz | 4.8 FT | 48 FT | YES | | <2° For Interferometer |
| C-2a | Broadside Dipole Antenna | | | | | | | | | | | | | | Inflatable Version; Struct. & |
| | Array | 18,133 | 3,400 | $>2(10^4)$ | $1(10^6)$ | 26x90 | 1.0 | 9x47 | 3.0 | 1.0-3.0 MHz | | 43 FT | YES | | Mech. Feasible; 26x90 at 1 MHz |
| | Concept Rating | | | | 0.031 | 0.043 | | | | 0.210 | | 1.0 | | $2.8(10^{-4})$ | |
| C-2b | 300 Meter Rhombic Antenna | 8,370 | 1,890 | $>2(10^4)$ | $1.1(10^5)$ | 15x45 | 3.0 | 11x35 | 5.0 | 3-5 MHz | | 150 FT | YES | | 30° x30° at 4 MHz; 300M used |
| | Concept Rating | | | | 0.003 | 0.15 | | | | 0.210 | | 0.32 | | $3.03(10^{-5})$ | instead of 2400M; Requires |
| | | | | | | | | | | | | | | | 7000M for 10° at 1 MHz |
| C-3 | 1500 Meter Paraboloidal | 4,530 | 12,000 | $>2(10^4)$ | $5(10^5)$ | 68x68 | 0.5 | 2x2 | 10.0 | 0.5-10 MHz | | 24 FT | NO | | 3.5° at 4 MHz. If Reduced |
| | Antenna | | | | | | | | | | | | | | to 400M Dia, Vol=5720, Beam |
| | Concept Rating | | | | 0.0156 | $2.2(10^{-2})$ | | | | 1.00 | | 1.0 | | $3.4(10^{-4})$ | 30° at 2.3 MHz |
| C-6 | 600 Meter Rhombic Antenna | 15,600 | 2,000 | $>2(10^4)$ | $6.3(10^5)$ | 30x90 | 0.5 | 5x15 | 10.0 | 0.5-10 MHz | | 10 FT | YES | | *Does not include Att. Control |
| | Concept Rating | | | | 0.020 | 0.037 | | | | 1.0 | | 1.0 | | $7.4(10^{-4})$ | Prop; 30° x30° at 2 MHz |
| C-7 | Member Structure for Large | | | | | | | | | | | | | | Not feasible due to box beam |
| | Astronomy Antenna | 2,600 | 2,300 | $>2(10^4)$ | $7.8(10^4)$ | 20x90 | 1.23 | 7x40 | 5.35 | 1.23-5.35 MHz | | 30 FT | YES | | struct. hazard |
| | Concept Rating | | | | $2.4(10^{-3})$ | $5.5(10^{-2})$ | | - | | 0.434 | | 1.0 | | $5.7(10^{-5})$ | 13° x78° at 3 MHz |
| C-13 | Orbiting Radio Astronomical | | | | | | | | | | | | | | |
| | Observatory | 400 | 30 | 500 to 32,500 | $1.2(10^7)$ | 9x28 | 0.5 | 5x18 | 10 | 0.225-10 | | 9.6 | NO | | |
| | Concept Rating | | | | 0.38 | $4(10^{-1})$ | | | | 1.0 | | 1.0 | | $1.4(10^{-1})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R-()}{C-()}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-1. R-1, Long Wave Radio Astronomy, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH *(F) DEG | $f_1$ MHz | DEG | $f_2$ MHz | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | (D) PERIM. DEFL. | | | |
| C-17 | Log Periodic Pyramidal | (2,084) | 100 | $>2(10^4)$ | $3(10^6)$ | 50×60 | 0.5 | 50×60 | 10 | 0.5-10 MHz | | 30 | YES | | *10° Resolution in Interfer- |
| | Concept Rating | | | | 0.094 | 0.033 | | | | 1.0 | | 1.0 | | $3.1(10^{-3})$ | ometer Configuration |
| C-20 | Extensible Rhombic | 18,000 | 1000 | $2(10^4)$ | $1.7(10^6)$ | 30×90 | 0.5 | 7×21 | 10 | 0.5-10 MHz | | (200) | YES | $4.2(10^{-3})$ | |
| | Concept Rating | | | | 0.053 | 0.33 | | | | 1.0 | | 0.24 | | | |
| C-51a | Colinear Dipole Array, 5 Km | 2,084 | 3,500 | $2(10^4)$ | $7.49(10^7)$ | 13×360 | 0.25 | 3×360 | 1.25 | 0.25-1.25 MHz | | 45 FT | | | |
| | Concept Rating | | | | 1.0 | $2.14(10^{-2})$ | | | | $7.9(10^{-2})$ | | 1.0 | | $1.7(10^{-3})$ | |
| C-51b | Crossed-H | | | | | | | | | | | | | | Interferometer Lobes 3.3° at |
| | Interferometer | 3,657 | 1280 | $2(10^4)$ | $1.07(10^6)$ | 90×225** | 0.5 | 40×80** | 10 | 0.5-10 MHz | | 10 FT | | | 0.5 MHz and 0.7° at 4 MHz |
| | Concept Rating | | | | $3.34(10^{-2})$ | $5(10^{-3})$ | | | | 1.0 | | 1.0 | | $1.7(10^{-4})$ | |
| C-52a | 1500 Meter Rhombic Antenna | | | | | | | | | | | | | | |
| | Ext Struct | 23,866 | 4,100 | $2(10^4)$ | $5.45(10^6)$ | 17×50 | 0.5 | 5×15 | 10 | 0.5-10 MHz | | 50 FT | | | |
| | Concept Rating | | | | $1.70(10^{-1})$ | 0.118 | | | | 1.0 | | 0.96 | | $1.9(10^{-2})$ | |
| C-52b | 1200 Meter Rhombic Antenna, | | | | | | | | | | | | | | |
| | Int Struct | (22,000) | 4,100 | $2(10^4)$ | $3.84(10^6)$ | 20×55 | 0.5 | 5×15 | 10 | 0.5-10 MHz | | 50 FT | | | |
| | Concept Rating | | | | $1.2(10^{-1})$ | $9.1(10^{-2})$ | | | | 1.0 | | 0.96 | | $1.1(10^{-2})$ | |
| C-53 | End Fire Antenna 3 Km Array | 4,000 | 3,800 | $2(10^4)$ | $4.5(10^7)$ | 30×30 | 0.25 | 14×14 | 1.25 | 0.25-1.25 MHz | | | | | Rejected for Excessive EVA |
| | Concept Rating | | | | 1.0 | 0.11 | | | | 0.105 | | | | | and Poor Dynamics, Work |
| | | | | | | | | | | | | | | | Discontinued on Concept |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R-()}{C-()}$

** PLUS INTERFEROMETER RESOLUTION TO 1.7° AT 0.5 MHz AND 0.35° AT 2.5 MHz AND ABOVE

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-2. R-2 UHF-UHF-SHF Radio Astronomy

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | $f_1$ MHz | DEG | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM. DEFL. | | |
| R-2 | VHF-UHF-SHF | | | | | | | | | | | | | |
| | Radio Astronomy | 24,000 | 4,500 | $2(10^4)$ | $1.4(10^5)$ | $10^2$ | 2 x 2 | $2(10^4)$ | 2 x 2 | $10^2$-$2(10^4)$MHz | 0.03 IN. | 0.3 IN. | YES | | |
| C-1 | 60 Ft Paraboloid Antenna | 9,100 | 1,980 | $2(10^4)$ | $1.4(10^3)$ | 50 | 23 x 23 | $2.5(10^3)$ | 0.466 | 50-$25(10^3)$MHz | 0.03 IN. | 0.3 IN. | YES | | |
| | Concept Rating | | | | 0.010 | | $7.55(10^{-3})$ | | | 0.123 | | 1.0 | | $9.3(10^{-6})$ | |
| C-9 | Parabolic Truss (Ryan) | 6,527 | 1,800 | $2(10^4)$ | 4,250 | 30 | 25 x 25 | $3.7(10^3)$ | 0.19 x 0.19 | 30-$3.7(10^3)$MHz | | 0.2 IN. | YES | | Electrical Pwr supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | $3.03(10^{-2})$ | | $6.4(10^{-3})$ | | | 0.181 | | 1.0 | | | |
| C-10 | Folding Petal Antenna (Ryan) | 3,015 | 1,000 | $2(10^4)$ | 1,000 | 60 | 23 x 23 | $7.4(10^3)$ | 0.19 x 0.19 | 60-$7.4(10^3)$MHz | 0.1 IN. | 0.1 IN. | YES | | Electrical Pwr supply can vary from 500 to 2,200 lb. |
| | Concept Rating | | | | $7.15(10^{-3})$ | | $7.55(10^{-3})$ | | | $3.67(10^{-1})$ | | 1.0 | | $2.0(10^{-5})$ | |
| C-12 | 40 Ft Dia Ant., 18 kw Solar Vista Ant (RCA) | 4,255 | 14,730 | $2(10^4)$ | 1,000 | 60 | 23 x 23 | 810 | 2 x 2 | 60-810 MHz | | 0.33 IN. | YES | | |
| | Concept Rating | | | | $7.5(10^{-3})$ | | $7.55(10^{-3})$ | | | $3.55(10^{-2})$ | | 0.91 | | $1.9(10^{-6})$ | |
| C-15 | Conic Scissors Antenna | <2,000 | 1,000 | $2(10^4)$ | 400 | $10^2$ | 23 x 23 | $3.7(10^3)$ | 0.63 x 0.63 | $1(10^2)$ - $3.7(10^3)$ MHz | 0.03 IN. | 0.2 IN. | YES | | |
| | Concept Rating | | | | $2.86(10^{-3})$ | | $7.55(10^{-3})$ | | | 0.15 | | 1.0 | | $3.2(10^{-6})$ | |
| C-16 | Bidirectional Scissors Array (Fairchild) | 1,250 | 200 | $2(10^4)$ | $2.8(10^3)$ | 90 | 10 x 10 | 150 | 6 x 6 | 90-150 MHz | | 1.5 IN. | YES | | Not enough study time to extrapolate to large size |
| | Concept Rating | | | | $2.0(10^{-2})$ | | 0.04 | | | $2.5(10^{-3})$ | | 0.2 | | $4.0(10^{-7})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-2. R-2, UHF-UHF-SHF Radio Astronomy, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| RFQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU- VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) | BEAMWIDTH | | | | * TOTAL STRUCTURE DEVIATION | | | | | |
| | | WEIGHT LBS | VOLUME CU FT | ALT. NM | APERTURE AREA SQ FT | *(F) DEG | $f_1$ MHz | DEG | $f_2$ MHz | * (B) BANDWIDTH | SUR- FACE | (D) PERIM. DEFL. | | | |
| C-75 | Commercial TV, 150 Ft Dia | | | | | | | | | | | | | | |
| | (Fager) | 7,629 | 3,040 | $2(10^4)$ | 9,500 | 20 | 23 x 23 | 2.4 $(10^3)$ | 0.2 x 0.2 | 20-2.4 $(10^3)$ MHz | | 0.3 IN. | YES | | |
| | Concept Rating | | | | $6.78(10^{-2})$ | | $7.55(10^{-3})$ | | | 0.116 | | 1.0 | | $5.9(10^{-5})$ | |
| | | | | | | | | | | | | | | | |
| C-100a | 50 Ft Trussed Parabolic | | | | | | | | | | | | | | T III-C Booster. Beryllium |
| | Ant. With 750 $Ft^2$ Solar Array | 2,000 | 570 | $2(10^4)$ | 1,000 | 60 | 23×23 | 13.5 $(10^3)$ | 0.1×0.1 | 60-1.35$(10^4)$ | | 0.055 IN. | YES | | Struct |
| | Concept Rating | | | | $7.15(10^{-3})$ | | $7.55(10^{-3})$ | | | 0.673 | | 1.0 | | $3.6(10^{-5})$ | |
| | | | | | | | | | | | | | | | |
| C-100b | 50 Ft Trussed Parabolic Ant. | | | | | | | | | | | | | | |
| | With 2400 $Ft^2$ Solar Array | 2,600 | 500 | $2(10^4)$ | 1,000 | 60 | 23×23 | 13.5 $(10^3)$ | 0.1×0.1 | 60-1.35$(10^4)$ | | 0.055 IN. | YES | | T III-C Centaur Booster |
| | Concept Rating | | | | $7.50(10^{-3})$ | | $7.55(10^{-3})$ | | | 0.673 | | 1.0 | | $3.6(10^{-5})$ | |
| | | | | | | | | | | | | | | | |
| C-100c | 50 Ft Trussed Parabolic | | | | | | | | | | | | | | Beryllium Struct. Atlas- |
| | With Directed Solar Array | 1,615 | 570 | $2(10^4)$ | 1,000 | 60 | 23×23 | 13.5 $(10^3)$ | 0.1×0.1 | 60-13.5$(10^3)$MHz | | 0.055 IN. | YES | | Advanced Centaur Booster |
| | Concept Rating | | | | $7.15(10^{-3})$ | | $7.55(10^{-3})$ | | | 0.674 | | 1.0 | | $3.6(10^{-5})$ | |
| | | | | | | | | | | | | | | | |
| C-101 | 78 Ft Countrawound Helix | | | | $1.1(10^4)$ | | | | | | | | | | |
| | (Fager) | 7,870 | 1,440 | $2(10^4)$ | at 15 MHz | 15 | 32×32 | 75 | 14.5x 14.5 | 15-75 MHz | | 0.22 IN. | YES | | |
| | Concept Rating | | | | $7.85(10^{-2})$ | | $3.91(10^{-3})$ | | | 0 | | 0.733 | | 0 | |
| | | | | | | | | | | | | | | | |
| C-102 | 100 Ft Dia Trussed Parabolic | | | | | | | | | | | | | | |
| | Ant. With 2100 $Ft^2$ Fixed | | | | | | | | | | | | | | |
| | Cell Solar Array | 7,570 | 5,450 | $2(10^4)$ | 4000 | 30 | 23×23 | 7.4 $(10^3)$ | 0.1× 0.1 | 30-7.4 $(10^3)$ MHz | | 0.1 IN. | YES | | Saturn V Booster |
| | Concept Rating | | | | $2.86(10^{-2})$ | | $7.55(10^{-3})$ | | | 0.367 | | 1.0 | | $7.9(10^{-5})$ | |
| | | | | | | | | | | | | | | | |
| | | | | | | | | | | | | | | | |
| | | | | | | | | | | | | | | | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R-()}{C-()}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-2. R-2, VHF-UHF-SHF Radio Astronomy, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| RFQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | BEAMWIDTH *(F) DEG | BEAMWIDTH $f_2$ MHz | BEAMWIDTH DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | * TOTAL STRUCTURE DEVIATION (D) PERIM. DEFL. | | | |
| C-103 | Horn Phased Array With | | | | | | | | | | | | | | |
| | 2500 $Ft^2$ Fixed Cell Solar | | | | | | | | | | | | | | |
| | Array | 14,800 | 5,450 | $2(10^4)$ | 1250 | 800 | 1.5×1.5 | 900 | 1.5×1.5 | 800-900 MHz | | 0.19 IN. | YES | | Saturn V Booster |
| | Concept Rating | | | | 8.93 $(10^{-3})$ | | 1.0 | | | 0.00502 | | 1.0 | | $4.5^{(10-5)}$ | |
| C-105 | Seven Dish Paraboloid TV | | | | | | | | | | | | | | |
| | Broadcast Antenna With | | | | | | | | | | | | | | |
| | Solar Array | 1,236 | 4,500 | $2(10^4)$ | 800 | 185 | 10×10 | $6.3_{(10^3)}$ | 0.25×0.25 | 185-6.3 $(10^3)$ MHz | 0.006 IN. | 0.116 IN. | YES | | |
| | Concept Rating | | | | $5.72(10^{-3})$ | | $4(10^{-2})$ | | | $3.11(10^{-1})$ | | 1.0 | | $7.1(10^{-5})$ | |
| C-11 | Paraboloid (30 Ft) Petal | | | | | | | | | | | | | | |
| | Antenna | 3,018 | 1,200 | $2(10^4)$ | 400 | $1(10^2)$ | 2.3×2.3 | $3(10^3)$ | 0.8×0.8 | $10^2$-$3(10^3)$ MHz | 0.05 IN. | 0.20 IN. | YES | | |
| | Concept Rating | | | | 2.86 $(10^{-3})$ | | 0.8 | | | 0.146 | | 1.0 | | $3.34(10^{-4})$ | |
| C-104 | 7-Petalated 30 Ft Dia | | | | | | | | | | | | | | |
| | Antennas With RTG Power | 12,625 | 5,450 | $2(10^4)$ | 2240 | 100 | 9×9 | $6.3_{(10^3)}$ | 0.15×0.15 | 100-6.3 $(10^3)$ MHz | | 0.125 IN. | YES | | |
| | Concept Rating | | | | $1.6(10^{-2})$ | | $4.94_{(10^{-2})}$ | | | 0.312 | | 1.0 | | $2.5(10^{-4})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-3. R-3 Submillimeter Radio Astronomy

TABLE USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | BEAMWIDTH *(F) DEG | BEAMWIDTH $f_2$ MHz | BEAMWIDTH DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | * TOTAL STRUCTURE DEVIATION (D) PERIM. DEFL. | | | |
| R-3 | Submillimeter Radio | | | | | | | | ** $2.3(10^{-3})$ | $3(10^4)$-$6(10^6)$ MHz | ** 0.002 | | | | ** Not diffraction limited below |
| | Astronomy | 24,000 | 4,500 | $2(10^4)$ | $7.75(10^3)$ | $3(10^4)$ | <2×2 | $3(10^5)$ | × $2.3(10^{-3})$ | 10 mm-50μ | IN. | 0.02 IN. | YES | | one millimeter in wavelength. |
| C-1 | 60 Ft Paraboloid Antenna | 9,100 | 1,980 | $2(10^4)$ | 96 | $3(10^4)$ | 0.15× 0.15 | $7.4(10^4)$ | 0.06× 0.067 | $3(10^4)$-$7.4(10^4)$ MHz | 0.005 IN. | 0.01 IN. | YES | | Values shown for center 15.6 ft |
| | Concept Rating | | | | $1.24(10^{-2})$ | | 1.0 | | | $7.37(10^{-3})$ | | 1.0 | | $9.1(10^{-5})$ | edge Defl = 0.01 in. |
| C-9 | Parabolic Truss (Ryan) | 6,527 | 1,800 | $2(10^4)$ | 3600 | 30 | 25×25 | $3.7(10^3)$ | 0.19× 0.19 | 30-$3.7(10^3)$ MHz | | 0.2 IN. | YES | | Reject: Frequency does not lie |
| | Concept Rating | | | | 0.465 | | $6.4(10^{-3})$ | | | 0 | | 0.1 | | | within required band |
| C-10 | Folding Petal Antenna (Ryan) | 3,015 | 1,000 | $2(10^4)$ | 600 | 80 | 25×25 | $7.4(10^3)$ | 0.19× 0.19 | 60-$7.4(10^3)$ MHz | | 0.1 IN. | YES | | Reject: Frequency does not lie |
| | Concept Rating | | | | $7.75(10^{-2})$ | | $6.4(10^{-3})$ | | | | | 0.20 | | 0 | within required band. |
| C-11 | 30 Ft. Dia. Folding Petal | | | | | | | | | | | | | | |
| | Antenna (TRW) | 3,018 | 1,200 | $2(10^4)$ | 707 | $1(10^3)$ | 2.3×2.3 | $4(10^3)$ | 0.23× 0.23 | $1(10^2)$-$3(10^3)$ MHz | 0.05 IN. | 0.20 IN. | YES | | Reject: Frequency does not lie |
| | Concept Rating | | | | $9.13(10^{-2})$ | | 0.756 | | | | | 0.1 | | 0 | within required band |
| C-15 | Conic Scissors Antenna | | | | | | | | | | | | | | |
| | (Fairchild) | <2,000 | 1,000 | $2(10^4)$ | 390 | $1(10^3)$ | 2.9×2.9 | $4(10^3)$ | 0.23× 0.23 | $1(10^2)$-$3.7(10^3)$ MHz | 0.03 IN. | 0.02 IN | YES | | Reject: Frequency does not lie |
| | Concept Rating | | | | $5.03(10^{-2})$ | | 0.476 | | | | | 0.1 | | 0 | within required band. |
| C-16 | Bi-Directional Scissors | 1,250 | | | | | | | | | | | | | |
| | Array (Fairchild) | | | $2(10^4)$ | 2930 | 90 | 10×10 | 150 | 1×1 | 90-150 MHz | | 6.0 IN. | YES | | Reject: Frequency does not lie |
| | Concept Rating | | | | $3.78(10^{-1})$ | | 0.04 | | | | | $3.3(10^{-3})$ | | 0 | within required band |
| C-105 | Seven Dish Paraboloid | 1,236 | 4,500 | $2(10^4)$ | 100 | $3(10^4)$ | 0.15× 0.15 | $1.23(10^5)$ | 0.4× 0.4 | $3(10^4)$-$1.23(10^5)$ | 0.006 IN. | 0.006 IN. | YES | | Using only one of the |
| | Concept Rating | | | | $1.29(10^{-2})$ | | 1.0 | | | $1.56(10^{-2})$ | | 1.0 | | $2.0(10^{-4})$ | paraboloids |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R-()}{C-()}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-4. R-4 Longwave Earth Atmosphere Absorption Measurement

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | * TOTAL STRUCTURE DEVIATION | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | * (F) DEG | BEAMWIDTH $f_2$ MHz | BEAMWIDTH DEG | * (B) BANDWIDTH | SUR-FACE | (D) PERIM. DEFL. | | | |
| R-4 | Longwave Earth Atmosphere | | | | | | | | | | | | | | |
| | Absorption Measurements | 24,000 | 4500 | $2(10^4)$ | $3.2(10^7)$ | 0.5 | <10×10 | 10 | <<10×10 | 0.5-10 MHz | 4.8 FT | 48 FT | YES | | |
| | | | | | | | | | | | | | | | |
| C-2a | Broadside Dipole Antenna | | | | | | | | | | | | | | Inflatable version struct and |
| | Array | 18,133 | 3400 | $2(10^4)$ | $1(10^6)$ | 1.0 | 26×90 | 3.0 | 9×47 | 1.0-3 MHz | | 43 FT | YES | | mech feasible |
| | Concept Rating | | | | 0.031 | | 0.043 | | | 0.210 | | 1.0 | | $2.8(10^{-4})$ | |
| | | | | | | | | | | | | | | | |
| C-2b | 300 Meter Rhombic Antenna | 8,370 | 1890 | $2(10^4)$ | $1.1(10^5)$ | 3.0 | 15×45 | 5.0 | 11×35 | 3-5 MHz | | 150 FT | YES | | 300M used instead of 2400M. |
| | Concept Rating | | | | 0.003 | | 0.15 | | | 0.210 | | 0.32 | | $3.03(10^{-5})$ | Req 7000M for 10° at 1 MHz |
| | | | | | | | | | | | | | | | |
| C-3 | 1500 Meter Paraboloidal | | | | | | | | | | | | | | If reduced to 400M dia, |
| | Antenna | 4,530 | 12,000 | $>2(10^4)$ | $5(10^5)$ | 0.5 | 68×68 | 10.0 | 2×2 | 0.5-10 MHz | | 24 FT | NO | | Vol = 5720, Beam 30° at |
| | Concept Rating | | | | 0.0156 | | 0.022 | | | 1.00 | | 1.00 | | $3.4(10^{-4})$ | 2.3 MHz |
| | | | | | | | | | | | | | | | |
| C-6 | 600 Meter Rhombic Antenna | 15,600** | 2,000 | $>2(10^4)$ | $6.3(10^5)$ | 0.5 | 30×90 | 10.0 | 5×15 | 0.5-10 MHz | | 10 FT | YES | | **Does not include att control prop |
| | Concept Rating | | | | 0.020 | | 0.037 | | | 1.00 | | 1.0 | | $7.4(10^{-4})$ | 30° ×30° at 2 MHz |
| | | | | | | | | | | | | | | | |
| C-7 | Member Structure for Large | | | | | | | | | | | | | | Not feasible due to box beam |
| | Astronomy Antenna | 2,600 | 2,300 | $>2(10^4)$ | $7.8(10^4)$ | 1.23 | 20×90 | 5.35 | 7×40 | 1.23-5.35 MHz | | 30 FT | YES | | struct hazard |
| | Concept Rating | | | | $2.4(10^{-3})$ | | $5.5(10^{-2})$ | | | 0.434 | | 1.0 | | $5.7(10^{-5})$ | |
| | | | | | | | | | | | | | | | |
| C-13 | Orbiting Radio Astronomical | | | | | | | | | | | | | | |
| | Observatory | 400 | 30 | 500 to 32,500 | $1.2(10^7)$ | 0.5 | 9×28 | 10 | 5×18 | 0.5-10 MHz | | 9.6 | NO | | |
| | Concept Rating | | | | 0.38 | | 0.397 | | | 1.0 | | 1.0 | | $1.4(10^{-1})$ | |
| | | | | | | | | | | | | | | | |
| | | | | | | | | | | | | | | | |
| | | | | | | | | | | | | | | | |

* RELATIVE CONCEPT RATING - (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-4. R-4 Longwave Earth Atmosphere Absorption Measurement, Contd

TABLE USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | $f_1$ MHz | * (F) DEG | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM. DEFL. | | |
| C-17 | Log Periodic Pyramidal | 2,084 | 100 | $>2(10^4)$ | $3(10^6)$ | 0.5 | 50×60 | 10 | 50×60 | 0.5-10 MHz | | 3.0 | YES | | 10° Resolution in interfer- |
| | Concept Rating | | | | 0.094 | | 0.033 | | | 1.0 | | 1.0 | | $3.1(10^{-3})$ | ometer configuration |
| C-20 | Extensible Rhombic | 18,000 | 1000 | $2(10^4)$ | $1.7(10^6)$ | 0.5 | 30×90 | 10 | 7×21 | 0.5-10 MHz | | 200 | YES | | |
| | Concept Rating | | | | 0.053 | | 0.33 | | | 1.0 | | 0.24 | | $4.2(10^{-3})$ | |
| C-51a | Colinear Dipole Array, 5 Km | 2,084 | 3500 | $2(10^4)$ | $7.49(10^7)$ | 0.25 | 13×360 | 1.25 | 3×360 | 0.25-1.25 MHz | | 45 FT | | | |
| | Concept Rating | | | | 1.0 | | $2.14(10^{-2})$ | | | $7.9\ (10^{-2})$ | | 1.0 | | $1.7(10^{-3})$ | |
| C-51b | Crossed-H Interferometer | 3,657 | 1280 | $2(10^4)$ | $1.07(10^6)$ | 0.5 | 90×225 | 10 | 40×80 | 0.5-10 MHz | | 10 FT | | | |
| | Concept Rating | | | | 0.0334 | | $5(10^{-3})$ | | | 1.0 | | 1.0 | | $1.7(10^{-4})$ | |
| C-52a | 1500 Meter Rhombic Antenna | | | | | | | | | | | | | | |
| | Ext. Struct. | 23,866 | 4100 | $2(10^4)$ | $5.45(10^6)$ | 0.5 | 17×50 | 10 | 5×15 | 0.5-10 MHz | | 50 FT | | | |
| | Concept Rating | | | | $1.70(10^{-1})$ | | 0.118 | | | 1.0 | | 0.96 | | $1.9(10^{-2})$ | |
| C-52b | 1200 Meter Rhombic Antenna | | | | | | | | | | | | | | |
| | Int. Struct. | 22,000 | 4100 | $2(10^4)$ | $3.84(10^6)$ at 0.2 MHz | 0.5 | 20×55 | 10 | 5×15 | 0.5-10 MHz | | 50 FT | | | |
| | Concept Rating | | | | 0.12 | | $9.1(10^{-2})$ | | | 1.0 | | 0.96 | | $1.1(10^{-2})$ | |
| C-53 | End Fire Antenna 3 Km Array | 4000 | 3800 | $2(10^4)$ | $4.5(10^7)$ | 0.25 | 30×30 | 1.25 | 14×14 | 0.25-1.25 MHz | | | | | Rej. for EVA hours and poor |
| | Concept Rating | | | | 1.0 | | 0.11 | | | 0.105 | | | | | dynamics |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-5. R-5 Point-To-Point Communication

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | $f_1$ MHz | *(F) DEG | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM DEFL | | | |
| R-5 | Point-to-Point Communication | 24,000 | 4,500 | $2(10^4)$ | 1,264 | $2(10^3)$ | 0.5×0.5 | $2(10^4)$ | 0.05× 0.05 | $2(10^3)$-$2(10^4)$MHz | 0.03 IN. | 0.3 IN. | YES | | Single beam for inter-planetary work |
| C-1 | 60 Ft Paraboloid Antenna | 9,100 | 1,980 | $2(10^4)$ | $1.4(10^3)$ | $2(10^3)$ | 0.6×0.6 | $2.5(10^3)$ | 0.466× 0.466 | $2(10^3)$ - $2.5(10^3)$ MHz | 0.03 IN. | 0.3 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 0.695 | | | $2.78(10^{-2})$ | | 1.0 | | $1.93(10^{-2})$ | |
| C-4 | Lenticular Communications Satellite | 1,484 | 78 | $2(10^4)$ | ← NOT APPLICABLE → | | | | | | | 2 Ft | NO | | Passive reflector not completely evaluated |
| | Concept Rating | | | | | | | | | | | | | -- | |
| C-9 | Large Compressible Member Truss Struct for Space Antenna | 6,527 | 1,800 | $2(10^4)$ | 4,250 | $2(10^3)$ | 0.34× 0.34 | $3.7(10^3)$ | 0.19× 0.19 | $2(10^3)$ - $3.7(10^3)$ MHz | | 0.2 IN. | YES | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 1.0 | | 1.0 | | | $9.45(10^{-2})$ | | 1.0 | | $9.45(10^{-2})$ | |
| C-10 | Folding Petal Struct for Large Space Antenna | 3,015 | 1,000 | $2(10^4)$ | 1,000 | $2(10^3)$ | 0.7×0.7 | $7.4(10^3)$ | 0.19× 0.19 | $2(10^3)$ - $7.4(10^3)$ MHz | 0.1 IN. | 0.1 IN. | YES | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.79 | | 0.51 | | | 0.3 | | 1.0 | | $1.21(10^{-1})$ | |
| C-11 | 30 Ft Petal Paraboloid Antenna | 3,018 | 1,200 | $2(10^4)$ | 400 | $2(10^3)$ | 1.2×1.2 | $3(10^3)$ | 0.8×0.8 | $2(10^3)$ - $3(10^3)$ MHz | 0.05 IN. | 0.2 | YES | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.317 | | 0.174 | | | $5.56(10^{-2})$ | | 1.0 | | $3.06(10^{-3})$ | |
| C-12 | Vista Direct Home TV Broadcast | 4,255 | 14,730 | $2(10^4)$ | 900 | 60 | 1.75× 1.75 | 810 | 1.75× 1.75 | 60-810 MHz | | 0.33 | YES | | Reject. Frequency does not lie within required band |
| | Concept Rating | | | | 0.71 | | 8.17 $(10^{-2})$ | | | 0 | | 0.91 | | 0 | |
| C-15 | Conic Scissors Parabola Array | <2,000 | 1,000 | $2(10^4)$ | 400 | $2(10^3)$ | 1.2×1.2 | $3.7(10^3)$ | 0.68× 0.68 | $2(10^3)$ - $3.7(10^3)$ MHz | 0.03 IN. | 0.2 | YES | | Electrical power supply variable from 500 to 2,200 lb |
| | Concept Rating | | | | 0.317 | | 0.174 | | | $9.45(10^{-2})$ | | 1.0 | | $5.2(10^{-3})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

$$\text{CONCEPT RATING} = \frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{\text{R - ( )}}{\text{C - ( )}}$$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-5. R-5 Point-To-Point Communication, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | *(A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | $f_1$ MHz | *(F) DEG | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM DEFL | | | |
| C-16 | Bi-Directional Scissors Array | 1,250 | 200 | $2(10^4)$ | 2,930 | 90 | 10×10 | 150 | 6×6 | 90-150 MHz | | 1.5 | YES | | Lacks sizing Reject. Frequency does not lie within required band |
| | Concept Rating | | | | 1.0 | | $2.5(10^{-3})$ | | | 0 | | 0.2 | | 0 | |
| C-75 | Commercial TV, 150 Ft Dia (Fager) | 7,629 | 3,040 | $2(10^4)$ | 9,500 | $2(10^3)$ | 0.23× 0.23 | 2.4 $(10^3)$ | 0.2× 0.2 | $2(10^3)$-$2.4(10^3)$MHz | | 0.3 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 4.73 $(10^{-4})$ | | | $2.22(10^{-2})$ | | 1.0 | | $1.05(10^{-5})$ | |
| C-100a | 50 Ft Trussed Parabolic Ant. With 750 $Ft^2$ Solar Array | 2,000 | 570 | $2(10^4)$ | 1000 | $2(10^3)$ | 0.7×0.7 | $1.35(10^4)$ | 0.1×0.1 | $2(10^3)$ -$1.35(10^4)$ MHz | | 0.055 IN. | YES | | Beryllium Struct |
| | Concept Rating | | | | 0.792 | | 0.51 | | | 0.638 | | 1.0 | | $2.58(10^{-1})$ | |
| C-100b | 50 Ft Trussed Parabolic Ant. With 2400 $Ft^2$ Solar Array | 2,600 | 500 | $2(10^4)$ | 1000 | $2(10^3)$ | 0.7×0.7 | $1.35(10^4)$ | 0.1×0.1 | $2(10^3)$-$1.35(10^4)$ MHz | | 0.055 IN. | YES | | |
| | Concept Rating | | | | 0.792 | | 0.51 | | | 0.638 | | 1.0 | | $2.58(10^{-1})$ | |
| C-100c | 50 Ft Trussed Parabolic With Directed Solar Array | 1,615 | 570 | $2(10^4)$ | 1000 | $2(10^3)$ | 0.7×0.7 | $1.35(10^4)$ | 0.15× 0.15 | $2(10^3)$ -$1.35(10^4)$ MHz | | 0.055 IN. | YES | | Beryllium Struct |
| | Concept Rating | | | | 0.792 | | 0.51 | | | 0.638 | | 1.0 | | $2.58(10^{-1})$ | |
| C-101 | 78 Ft Countrawound Helix (Fager) | 7,870 | 1,440 | $2(10^4)$ | $1.1(10^4)$ at 15 MHz | 15 | 32×32 | 75 | 14.5× 14.5 | 15-75 MHz | | | | | Reject. Frequency does not lie within required band |
| | Concept Rating | | | | 1.0 | | $2.4(10^{-4})$ | | | 0 | | | | 0 | |
| C-102 | 100 Ft Dia Trussed Parabolic Ant. With 2100 $Ft^2$ Fixed Cell Solar Array | 7,570 | 5,450 | $2(10^4)$ | 4000 | $2(10^3)$ | 0.35× 0.35 | $7.4(10^3)$ | 0.1×0.1 | $2(10^3)$ -$7.4(10^3)$ MHz | | 0.1 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | 0.3 | | 1.0 | | $3.0(10^{-1})$ | |

*RELATIVE CONCEPT RATING = (A) (F) (B) (D)

$$\text{CONCEPT RATING} = \frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-5. R-5 Point-To-Point Communication, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU- | * RELATIVE | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | ALT NM | *(A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | *(F) DEG | $f_2$ MHz | DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | (D) PERIM DEFL | VERABLE | CONCEPT RATING | |
| C-103 | Horn Phased Array With 2500 $Ft^2$ Fixed Cell Solar Array | 14,800 | 5,450 | $2(10^4)$ | 2410 | 800 | 1.5×1.5 | 900 | 1.5×1.5 | 800-900 MHz | | 0.19 IN. | YES | | Reject. Frequency does not lie within required band |
| | Concept Rating | | | | 1.0 | | 0.111 | | | 0 | | 1.0 | | 0 | |
| C-104 | 7-Petalated 30 Ft Dia. Antennas with RTG Power | 12,625 | 5,450 | $2(10^4)$ | 2240 | $2(10^3)$ | 0.5×0.5 | $6(10^3)$ | 0.15× 0.15 | $2(10^3)$ -$6(10^3)$ MHz | | 0.125 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | 0.222 | | 1.0 | | $2.2(10^{-1})$ | |
| C-105 | Seven Dish Paraboloid TV Broadcast Antenna With Solar Array | 1,236 | 4,500 | $2(10^4)$ | 850 | $2(10^3)$ | 0.8×0.8 | $6.3(10^3)$ | 0.25× 0.25 | $2(10^3)$ -$6.3(10^3)$ MHz | 0.006 | 0.116 | | | |
| | Concept Rating | | | | 0.67 | | 0.39 | | | 0.239 | | 1.0 | | $6.24(10^{-2})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R-()}{C-()}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-6. R-6 Multiple Access Radio Communication

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | BEAMWIDTH *(F) DEG | BEAMWIDTH $f_2$ MHz | BEAMWIDTH DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | * TOTAL STRUCTURE DEVIATION (D) PERIM DEFL | | | |
| R-6 | Multiple Access Radio Communication | 24,000 | 4,500 | $2(10^4)$ | 1,264 | 1.5(10) | 17×17 | $2(10^4)$ | 17×17 | 1.5(10) -$2(10^4)$ MHz | 0.03 IN. | <0.3 IN. | YES | | Multiple Beams |
| C-1 | 60 Ft Paraboloid Antenna | 9,100 | 1,980 | $2(10^4)$ | $1.4(10^3)$ | 50 | 23×23 | $2.5(10^3)$ | 0.47× 0.47 | 50-$2.5(10^3)$ MHz | 0.03 IN. | 0.3 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 0.546 | | | $1.23(10^{-1})$ | | 1.0 | | $6.7(10^{-2})$ | |
| C-4 | Lenticular Communications Satellite | 1,484 | 78 | $2(10^4)$ | ← NOT APPLICABLE → | | | | | | | | | | This is a passive reflector concept. The analysis used for other concepts does not apply. |
| | Concept Rating | | | | | | | | | | | | | | |
| C-9 | Large Compressible Member Truss Struct for Space Antenna | 6,527 | 1,800 | $2(10^4)$ | 4,250 | 30 | 23×23 | $3.7(10^3)$ | 0.19× 0.19 | 30-$3.7(10^3)$ MHz | | 0.2 IN. | YES | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 1.0 | | 0.546 | | | $1.84\ (10^{-1})$ | | 1.0 | | $1.0(10^{-1})$ | |
| C-10 | Folding Petal Struct for Large Space Antenna | 3,015 | 1,000 | $2(10^4)$ | 1,000 | 60 | 23×23 | $7.4(10^3)$ | 0.19× 0.19 | 60-$7.4(10^3)$ MHz | 0.1 IN. | 0.1 IN. | YES | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.792 | | 0.546 | | | $3.67(10^{-1})$ | | 1.0 | | $1.6(10^{-1})$ | |
| C-11 | 30 Ft Petal Paraboloid Antenna | 3,018 | 1,200 | $2(10^4)$ | 400 | 100 | 23×23 | $3(10^3)$ | 0.8×0.8 | 100-$3(10^3)$ MHz | 0.05 IN. | 0.2 IN. | YES | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.317 | | 0.546 | | | $1.45\ (10^{-1})$ | | 1.0 | | $2.5(10^{-2})$ | |
| C-12 | Vista Direct Home TV Broadcast | 4,255 | 14,730 | $2(10^4)$ | 900 | 60 | 23×23 | 810 | 1.75× 1.75 | 60-810 MHz | | 0.33 IN. | YES | | |
| | Concept Rating | | | | 0.713 | | 0.546 | | | $3.76\ (10^{-2})$ | | 0.91 | | $1.3(10^{-2})$ | |
| C-15 | Conic Scissors Parabola Array | <2,000 | 1,000 | $2(10^4)$ | 400 | 100 | 23×23 | $3.7(10^3)$ | 0.63× 0.63 | 100-$3.7(10^3)$MHz | 0.03 IN. | 0.2 IN. | | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.317 | | 0.546 | | | $1.84(10^{-1})$ | | 1.0 | | $3.18(10^{-2})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{\text{R - ( )}}{\text{C - ( )}}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-6. R-6 Multiple Access Radio Communication, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | $f_1$ MHz | *(F) DEG | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM DEFL | | | |
| C-16 | Bi-Directional Scissors Array | 1,250 | 200 | $2(10^4)$ | 2,800 | 90 | 10×10 | 150 | 6×6 | 90-150 MHz | | 1.5 IN. | YES | | Lacks sizing |
| | Concept Rating | | | | 1.0 | | 1.0 | | | 3.0 $(10^{-3})$ | | 0.2 | | $6.0(10^{-4})$ | |
| C-75 | Commercial TV, 150 Ft Dia (Fager) | 7,629 | 3,040 | $2(10^4)$ | 9,500 | 20 | 23×23 | $2.4(10^3)$ | 0.2×0.2 | 20-$2.4(10^3)$MHz | | 0.3 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 0.546 | | | $1.19(10^{-1})$ | | 1.0 | | $6.5(10^{-2})$ | |
| C-100a | 50 Ft Trussed Parabolic Ant. With 750 $Ft^2$ Solar Array | 2,000 | 570 | $2(10^4)$ | 1,000 | 60 | 23×23 | $1.35(10^4)$ | 0.1×0.1 | 60-$1.35(10^4)$ MHz | | 0.055 IN. | YES | | Beryllium Struct |
| | Concept Rating | | | | 0.792 | | 0.546 | | | $6.73(10^{-1})$ | | 1.0 | | $2.92(10^{-1})$ | |
| C-100b | 50 Ft Trussed Parabolic Ant. With 2400 $Ft^2$ Solar Array | 2,600 | 500 | $2(10^4)$ | 1,000 | 60 | 23×23 | $1.35(10^4)$ | 0.1×0.1 | 60-$1.35(10^4)$MHz | | 0.055 IN. | YES | | |
| | Concept Rating | | | | 0.792 | | 0.546 | | | $6.73(10^{-1})$ | | 1.0 | | $2.92(10^{-1})$ | |
| C-100c | 50 Ft Trussed Parabolic With Directed Solar Array | 1,615 | 570 | $2(10^4)$ | 1,000 | 60 | 23×23 | $1.35(10^4)$ | 0.1×0.1 | 60-$1.35(10^4)$ MHz | | 0.055 IN. | YES | | Beryllium Struct |
| | Concept Rating | | | | 0.792 | | 0.546 | | | $6.73(10^{-1})$ | | 1.0 | | $2.92(10^{-1})$ | |
| C-101 | 78 Ft Countrawound Helix (Fager) | 7,870 | 1,440 | $2(10^4)$ | $11(10^3)$ at 15 MHz | 15 | 32×32 | 75 | 14.5× 14.5 | 15-75 MHz | | 0.22 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 0.282 | | | $3.0(10^{-3})$ | | 1.0 | | $8.46(10^{-4})$ | |
| C-102 | 100 Ft Dia Trussed Parabolic Ant. With 2100 $Ft^2$ Fixed Cell Solar Array | 7,570 | 5,450 | $2(10^4)$ | 4,000 | 30 | 23×23 | $7.4(10^3)$ | 0.1×0.1 | 30-$7.4(10^3)$MHz | | 0.1 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 0.546 | | | $3.69(10^{-1})$ | | 1.0 | | $2.02(10^{-1})$ | |

* RELATIVE CONCEPT RATING (A) (F) (B) (D)

$$\text{CONCEPT RATING} = \frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-6. R-6 Multiple Access Radio Communication, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | * (F) DEG | $f_2$ MHz | DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | (D) PERIM. DEFL | | | |
| C-103 | Horn Phased Array With 2500 $Ft^2$ Fixed Cell Solar Array | 14,8000 | 5,450 | $2(10^4)$ | 2,410 | 800 | 1.5×1.5 | 900 | 1.5×1.5 | 800-900 MHz | | 0.19 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | 5.0 $(10^{-3})$ | | 1.0 | | 5.0$(10^{-3})$ | |
| C-104 | 7-Petalated 30 Ft Dia Antennas With RTG Power | 12,625 | 5,450 | $2(10^4)$ | 2,240 | 100 | 9×9 | $6(10^3)$ | 0.15× 0.15 | 100-$6(10^3)$ MHz | | 0.125 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | 2.95 $(10^{-1})$ | | 1.0 | | 2.95$(10^{-1})$ | |
| C-105 | Seven Dish Paraboloid TV Broadcast Antenna With Solar Array | 1,236 | 4,500 | $2(10^4)$ | 800 | 185 | 10×10 | $6.3(10^3)$ | 0.25× 0.25 | 185-$6.3(10^3)$ MHz | 0.006 IN. | 0.116 IN. | YES | | |
| | Concept Rating | | | | 0.63 | | 1.0 | | | 3.06$(10^{-1})$ | | 1.0 | | 1.93$(10^{-1})$ | |

* RELATIVE CONCEPT RATING - (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-7. R-7 TV Broadcast

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | BEAMWIDTH * (F) DEG | BEAMWIDTH $f_2$ MHz | BEAMWIDTH DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | * TOTAL STRUCTURE DEVIATION (D) PERIM. DEFL. | | | |
| R-7 | TV Broadcast | 24,000 | 4,500 | $2(10^4)$ | $6(10^3)$ | $8(10^2)$ | 3.5×7.4 | $9(10^2)$ | 3.5×7.4 | 800-900 MHz | 0.6 IN. | 6 IN. | YES | | Multiple beams, one per broadcast area |
| C-1 | 60 Ft Paraboloid Antenna | 9,100 | 1,980 | $2(10^4)$ | $1.4(10^3)$ | $8(10^2)$ | 1.4×1.4 | $9(10^2)$ | 1.3×1.3 | 800-900 MHz | 0.03 IN. | 0.3 IN. | YES | | |
| | Concept Rating | | | | 0.233 | | 1.0 | | | 1.0 | | 1.0 | | $2.33(10^{-1})$ | |
| C-4 | Lenticular Communications Satellite | 1,484 | 78 | $2(10^4)$ | ← NOT APPLICABLE ** → | | | | | | | | | | |
| | Concept Rating | | | | | | | | | | | | | | |
| C-9 | Large Compressible Member Truss Struct for Space Antenna | 6,527 | 1,800 | $2(10^4)$ | 4,250 | $8(10^2)$ | 0.86× 0.86 | 900 | 0.77× 0.77 | 800-900 MHz | | 0.2 IN. | YES | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.708 | | 1.0 | | | 1.0 | | 1.0 | | $7.08(10^{-1})$ | |
| C-10 | Folding Petal Struct for Large Space Antenna | 3,015 | 1,000 | $2(10^4)$ | 1000 | 800 | 1.7×1.7 | 900 | 1.55× 1.55 | 800-900 MHz | 0.1 IN. | 0.1 IN. | YES | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 1.0 | | 0.1 | | $1.67(10^{-1})$ | |
| C-11 | 30 Ft Petal Paraboloid Antenna | 3,018 | 1,200 | $2(10^4)$ | 400 | 800 | 2.9×2.9 | 900 | 2.6×2.6 | 800-900 MHz | 0.05 IN. | 0.2 IN. | | | Electrical power supply can vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.066 | | 1.0 | | | 1.0 | | 1.0 | | $6.6(10^{-2})$ | |
| C-12 | Vista Direct Home TV Broadcast | 4,255 | 14,730 | $2(10^4)$ | 1,000 | 800 | 1.7×1.7 | 810 | 1.7×1.7 | 800-810 MHz | | 0.33 IN. | YES | | |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 0.1 | | 1.0 | | $1.67(10^{-2})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

** This is a passive reflector concept. The analysis used for other concepts does not apply for this concept.

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R-()}{C-()}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-7. R-7 TV Broadcast, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | * TOTAL STRUCTURE DEVIATION | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | * (F) DEG | $f_2$ MHz | DEG | * (B) BANDWIDTH | SUR-FACE | (D) PERIM. DEFL. | | | |
| C-15 | Conic Scissors Parabola | | | | | | | | | | | | | | Electrical power supply can |
| | Array | <2,000 | 1,000 | $2(10^4)$ | 400 | 800 | 2.9×2.9 | 900 | 2.6×2.6 | 800-900 MHz | 0.03 IN. | 0.2 IN. | YES | | vary from 500 to 2,200 lb. |
| | Concept Rating | | | | 0.066 | | 1.0 | | | 1.0 | | 1.0 | | $6.6(10^{-2})$ | |
| C-16 | Bi-Directional Scissors Array | 1,250 | 200 | $2(10^4)$ | 2,930 | 90 | 10×10 | 150 | 6×6 | 90-150 MHz | | 1.5 IN. | YES | | Lacks Sizing |
| | Concept Rating | | | | 0.49 | | 0.259 | | | 0 | | 1.0 | | 0 | Reject: Frequency does not lie within required band |
| C-75 | Commercial TV, 150 Ft Dia | | | | | | | | | | | | | | |
| | (Fager) | 7,629 | 3,040 | $2(10^4)$ | 9,500 | 800 | 0.58×0.58 | 900 | 0.51×0.51 | 800-900 MHz | | 0.3 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | 1.0 | | 1.0 | | 1.0 | |
| C-100a | 50 Ft Trussed Parabolic | | | | | | | | | | | | | | |
| | Antenna with 750 $Ft^2$ Solar | | | | | | | | | | | | | | |
| | Array | 2,000 | 570 | $2(10^4)$ | 1,000 | 800 | 1.7×1.7 | 900 | 1.55×1.55 | 800-900 MHz | | 0.055 IN. | YES | | Beryllium Struct |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 1.0 | | 1.0 | | $1.67(10^{-1})$ | |
| C-100b | 50 Ft Trussed Parabolic | | | | | | | | | | | | | | |
| | Antenna with 2,400 $Ft^2$ Solar | | | | | | | | | | | | | | |
| | Array | 2,600 | 500 | $2(10^4)$ | 1,000 | 800 | 1.7×1.7 | 900 | 1.55×1.55 | 800-900 MHz | | 0.055 IN. | YES | | Beryllium struct |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 1.0 | | 1.0 | | $1.67(10^{-1})$ | |
| C-100c | 50 Ft Trussed Parabolic With | | | | | | | | | | | | | | Beryllium struct |
| | Directed Solar Array | 1,615 | 570 | $2(10^4)$ | 1,000 | 800 | 1.7×1.7 | 900 | 1.55×1.55 | 800-900 MHz | | 0.055 IN. | YES | | |
| | Concept Rating | | | | 0.167 | | 1.0 | | | 1.0 | | 1.0 | | $1.67(10^{-1})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

$$\text{CONCEPT RATING} = \frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-7. R-7 TV Broadcast, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | *(F) DEG | $f_2$ MHz | DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | (D) PERIM. DEFL. | | | |
| C-101 | 78 Ft Countrawound Helix | | | | | | | | | | | | | | |
| | (Fager) | 7,870 | 1,440 | $2(10^4)$ | 1.1 $(10^4)$ | 15 | 32×32 | 75 | 14.5× 14.5 | 15-75 MHz | | 0.22 IN. | YES | | Reject: Frequency does not lie |
| | Concept Rating | | | | 1.0 | | $2.53(10^{-2})$ | | | 0 | | 1.0 | | 0 | within required band |
| C-102 | 100 Ft Dia Trussed Para- | | | | | | | | | | | | | | |
| | bolic Antenna With 2,100 $Ft^2$ | | | | | | | | | | | | | | |
| | Fixed Cell Solar Array | 7,570 | 5,450 | $2(10^4)$ | 4,000 | 800 | 0.86× 0.86 | 900 | 0.77× 0.77 | 800-900 MHz | | 0.1 IN. | YES | | |
| | Concept Rating | | | | 0.666 | | 1.0 | | | 1.0 | | 1.0 | | $6.66(10^{-1})$ | |
| C-103 | Horn Phased Array With | | | | | | | | | | | | | | |
| | 2,500 $Ft^2$ Fixed Cell Solar | | | | | | | | | | | | | | |
| | Array | 14,800 | 5,450 | $2(10^4)$ | 2,410 | 800 | 1.5×1.5 | 900 | 1.5×1.5 | 800-900 MHz | | 0.19 IN. | YES | | |
| | Concept Rating | | | | 0.402 | | 1.0 | | | 1.0 | | 1.0 | | $4.02(10^{-1})$ | |
| C-104 | 7-Petalated 30 Ft Dia | | | | | | | | | | | | | | |
| | Antennas With RTG Power | 12,625 | 5,450 | $2(10^4)$ | 2,240 | 800 | 1.2×1.2 | 900 | 1.0×1.0 | 800-900 MHz | | 0.125 IN. | YES | | |
| | Concept Rating | | | | 0.373 | | 1.0 | | | 1.0 | | 1.0 | | $3.73(10^{-1})$ | |
| C-105 | Seven Dish Paraboloid TV | | | | | | | | | | | | | | |
| | Broadcast Antenna With Solar | | | | | | | | | | | | | | |
| | Array | 1,236 | 4,500 | $2(10^4)$ | 850 | 800 | 1.9×1.9 | 900 | 1.7×1.7 | 800-900 MHz | 0.006 IN. | 0.116 IN. | YES | | |
| | Concept Rating | | | | 0.142 | | 1.0 | | | 1.0 | | 1.0 | | $1.4(10^{-1})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-8. R-8 X-Ray Astronomy (Collimation)

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | $f_1$ MHz | * (F) DEG | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM. DEFL. | | |
| R-8 | X-Ray Astronomy | | | | | | | | | | | | | |
| | (Collimation) | 9,900 | 1,980 | $2(10^2)$ | $1.0(10^3)$ | 12Å | 1×1 | 0.3Å | 1×1 | 12Å-0.3Å | - | <0.1° | YES | | 6 min of arc resolution |
| | | | | | | | | | | | | | | |
| C-71 | Flat Panel X-Ray Detector | | | | | | | | | | | | | |
| | Array | 15,118 | 1,580 | $2(10^2)$ | 352 | 12Å | 0.7×0.7 | 0.3Å | 0.7×0.7 | 12Å-0.3Å | | 1° | YES | | |
| | Concept Rating | | | | $3.5(10^{-1})$ | | 1.0 | | | 1.0 | | $1.0(10^{-1})$ | | $3.5(10^{-2})$ | |
| | | | | | | | | | | | | | | | |
| C-73 | Focusing X-Ray Grazing | | | | | | | | | | | | | | System may be used to produce |
| | Incidence Reflecting | | | | | | | | | | | | | | collimation data. |
| | Telescope (Imaging) | 14,000** | 1,580** | $2(10^2)$ | 78** | 12Å | 0.15× 0.16 | 2Å | 0.5×0.5 | 12Å-2Å | | <0.1** | YES | | Added 100 lb for gimbals |
| | Concept Rating | | | | $7.8(10^{-2})$ | | 1.0 | | | 0.855 | | 1.0 | | $6.7(10^{-2})$ | |
| | | | | | | | | | | | | | | | |
| C-72 | X-Ray Det. Array | 13,576 | 1,580 | $2(10^2)$ | 1000 | 12Å | 0.7×0.7 | 0.3Å | 0.7×0.7 | 12Å-0.3Å | | 10 | | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | 1.0 | | $1.0(10^{-1})$ | | $1.0(10^{-1})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

** Present concept dynamically unsuitable. To be redesigned and re-evaluated.

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R-()}{C-()}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-9. R-8A X-Ray Astronomy Imaging

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | | |
| | | | | | | $f_1$ MHz | *(F) DEG | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM. DEFL. | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | | | | | | | | | | |
| R-8A | X-Ray Astronomy Imaging | 9,900 | 1,980 | $2(10^2)$ | 21.5 | 12Å | 0.5×0.5 | 2Å | 0.5×0.5 | 12Å-2Å | - | 1.6 | YES | | 2 sec of arc resolution |
| | | | | | | | | | | | | $(10^{-1})^\circ$ | | | |
| C-71 | Flat Panel X-Ray Detector | | | | | | | | | | | | | | |
| | Array | 15,118 | 1,580 | $2(10^2)$ | 352 | 12Å | 0.7×0.7 | 2Å | 0.7×0.7 | 12Å-2Å | | 1° | YES | | Array type output can be synthesized to produce imaging data |
| | Concept Rating | | | | 1.0 | | 0.51 | | | 1.0 | | $1.6\ (10^{-1})$ | | $8.16(10^{-2})$ | |
| C-73 | Focusing X-Ray Grazing | | | | | | | | | | | | | | ** |
| | Incidence Reflecting | | | | | | | | | | | | | | |
| | Telescopes | 14,000** | 1,580** | $2(10^2)$ | 21.5** | 12Å | 0.16×0.16 | 2Å | 0.5×0.5 | 12Å-2Å | | **<0.1° | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | 1.0 | | 1.0 | | 1.0 | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

** Present concept dynamically unsuitable. To be redesigned and re-evaluated.

$$\text{CONCEPT RATING} = \frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-10. R-9 Gamma-Ray Astronomy

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | BEAMWIDTH *(F) DEG | BEAMWIDTH $f_2$ MHz | BEAMWIDTH DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | * TOTAL STRUCTURE DEVIATION (D) PERIM. DEFL. | | | |
| R-9 | Gamma-Ray Astronomy | 9,900 | 1,980 | $2(10^2)$ | 100 | 0.3Å | 1×1 | 0.03 mÅ | 1×1 | $3(10^{-1})$ Å- $3(10^{-5})$ Å | - | <0.1° | YES | | |
| C-71 | X-Ray + Gamma-Ray Flat Panels | 15,118 | 1,580 | $2(10^2)$ | 352 | 0.3Å | 0.7×0.7 | $3(10^{-2})$ Å | 0.7×0.7 | $3(10^{-1})$ Å- $3(10^{-2})$ Å | | 1° | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | $1.0(10^{-3})$ | | $10^{-1}$ | | $1.0(10^{-4})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-11. R-10 50 KW Solar Energy Conversion

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | BEAMWIDTH | | | | | * TOTAL STRUCTURE DEVIATION | | | | |
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | $f_1$ MHz | *(F) DEG | $f_2$ MHz | DEG | * (B) BANDWIDTH | SUR-FACE | (D) PERIM. DEFL. | | | |
| R-10 | 50 KW Solar Energy | | | | | | | | | | | | | | Angular coverage indicates |
| | Conversion | 24,000 | 4,500 | $2(10^4)$ | $4.9(10^3)$ | 2.0μ | 15×15 | 0.3μ | 5×5 | 2.0-0.3μ | -- | <5° | YES | | deviation from ideal aspect |
| | | | | | | | | | | | | | | | angle |
| | | | | | | | | | | | | | | | |
| C-5 | Solar Cell Array | | | | | | | | | | | | | | Principles of Boeing design |
| | (Boeing/JPL) | 1,960 | 3,850 | $2(10^4)$ | $4.4(10^3)$ | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | <5° | YES | | included but array modified |
| | Concept Rating | | | | $9.0(10^{-1})$ | | 1.0 | | | 0.412 | | 1.0 | | $3.7(10^{-1})$ | for earth orbit |
| | | | | | | | | | | | | | | | |
| C-8 | Large Solar Cell Array | | | | | | | | | | | | | | 10Kw using Ryan concept |
| | Struct (Ryan) | 5,500 | 1,400 | $2(10^4)$ | $3(10^3)$ | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | 30° | YES | | without modification |
| | Concept Rating | | | | $6.1(10^{-1})$ | | 1.0 | | | 0.412 | | 0.167 | | $2.5(10^{-1})$ | |
| | | | | | | | | | | | | | | | |
| C-19 | Solar Array (Roll-Out or | | | | | | | | | | | | | | |
| | Scissors) (Fairchild) | 140 | 11.6 | $2(10^4)$ | 160 | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | 0.03 | <5° | YES | | 1 Kw output |
| | Concept Rating | | | | $3.3(10^{-2})$ | | 1.0 | | | 0.412 | | 1.0 | | $1.4(10^{-2})$ | |
| | | | | | | | | | | | | | | | |
| C-75 | Commercial TV, 150 ft Dia | | | | | | | | | | | | | | |
| | (Fager) | 7,629 | 3,040 | $2(10^4)$ | $3.0(10^3)$ | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | 1° | YES | | |
| | Concept Rating | | | | $6.1(10^{-1})$ | | 1.0 | | | 0.412 | | 1.0 | | $2.5(10^{-1})$ | |
| | | | | | | | | | | | | | | | |
| C-81 | TV or Communication | 3,238 | | | | | | | | | | | | | |
| | Support Solar Cell Array | | 1,600 | $2(10^4)$ | $3(10^3)$ | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | 1° | YES | | |
| | Concept Rating | | | | $6.1(10^{-1})$ | | 1.0 | | | 0.412 | | 1.0 | | $2.5(10^{-1})$ | |
| | | | | | | | | | | | | | | | |
| C-82 | S-IVB Lab Solar Cell Array | 4,312 | 740 | $2(10^4)$ | $3.08(10^3)$ | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | <5° | YES | | |
| | Concept Rating | | | | $6.3(10^{-1})$ | | 1.0 | | | 0.412 | | 1.0 | | $2.6(10^{-1})$ | |
| | | | | | | | | | | | | | | | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

$$\text{CONCEPT RATING} = \frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-11. R-10 50 KW Solar Energy Conversion, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | $f_1$ MHz | $f_1$ MHz | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM. DEFL. | | | |
| C-100a | 50 Ft Trussed Parabolic With | | | | | | | | | | | | | | |
| | 750 $Ft^2$ Solar Array | 2,000 | 570 | $2(10^4)$ | 750 | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | <5° | YES | | |
| | Concept Rating | | | | $1.53(10^{-1})$ | | 1.0 | | | 0.412 | | 1.0 | | $6.3(10^{-2})$ | |
| C-100b | 50 Ft Trussed Parabolic | | | | | | | | | | | | | | |
| | Antenna With 2,400 $Ft^2$ | | | | | | | | | | | | | | |
| | Solar Array | 2,600 | 500 | $2(10^4)$ | 2,400 | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | <5° | NO | | |
| | Concept Rating | | | | $4.9(10^{-1})$ | | 1.0 | | | 0.412 | | 1.0 | | $2.0(10^{-1})$ | |
| C-100c | 50 Ft Trussed Parabolic With | | | | | | | | | | | | | | |
| | Directed Solar Array | 1,615 | 570 | $2(10^4)$ | 750 | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | <5° | YES | | Beryllium struct |
| | Concept Rating | | | | $1.53(10^{-1})$ | | 1.0 | | | 0.412 | | 1.0 | | $6.3(10^{-2})$ | |
| C-102 | 100 Ft Dia Trussed Para- | | | | | | | | | | | | | | **Since solar array is not |
| | bolic With 2,100 $Ft^2$ Fixed | | | | | | | | | | | | | | oriented to sun the effective |
| | Cell Solar Array | 7,570 | 5,450 | $2(10^4)$ | 2,100** | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | <5° | NO | | area is variable |
| | Concept Rating | | | | $4.29(10^{-1})$ | | 1.0 | | | 0.412 | | 1.0 | | $1.8(10^{-1})$ | |
| C-103 | Horn Phased Array With | | | | | | | | | | | | | | **Since solar array is not |
| | 2,500 $Ft^2$ Fixed Cell Solar | | | | | | | | | | | | | | oriented to sun the effective |
| | Array | 14,800 | 5,450 | $2(10^4)$ | 2,500** | 1.0 | 15×15 | 0.3 | | 1.0-0.3μ | | <5° | NO | | area is variable |
| | Concept Rating | | | | $5.1(10^{-1})$ | | 1.0 | | | 0.412 | | 1.0 | | $2.1(10^{-1})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R-()}{C-()}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-12. R-11 Meteoroid Collection

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | $f_1$ MHz | * (F) DEG | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM. DEFL. | | | |
| R-11 | Meteoroid Collection | 24,000 | 4,500 | $2(10^4)$ | $1.0(10^4)$ | ~3 | 360×360 | ~3 | 360×360 | $3(10^{-2})$ MHz | 0.03 | <12 IN. | Optional | | Maximum micrometeoroid |
| | | | | | | $(10^{-2})$ | | $(10^{-2})$ | | | IN. | | | | collision rate shown |
| C-14 | Deep Space Meteoroid | | | | | | | | | | | | | | |
| | Satellite (Martin) | 5,800 | 2,380 | >1AU | 2,894** | -- | 360×360 | -- | 360×360 | | | | | | |
| | Concept Rating | | | | $2.89(10^{-1})$ | | 1.0 | | | | | | | $2.89(10^{-1})$ | |
| C-18 | Concertina Micrometeoroid | | | | | | | | | | | | | | |
| | Panel (Fairchild) | 2,000 | 1,000 | $2(10^4)$ | 500 | -- | 360×360 | -- | | | | 10 IN. | YES | | |
| | Concept Rating | | | | $5.0(10^{-2})$ | | 1.0 | | | | | 1.0 | | $5.0(10^{-2})$ | |
| C-85 | (S-V and S-1B) Meteoroid | | | | | | | | | | | | | | |
| | Collector | 3,192 | 1,600 | $2(10^2)$ | $2.6(10^2)$ | -- | 90×90 | -- | 90×90 | | | 2.2 IN. | YES | | |
| | Concept Rating | | | | $2.6(10^{-2})$ | | $6.2(10^{-2})$ | | | | | 1.0 | | $1.6\ (10^{-3})$ | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

** Area for pressure sensors. If capacitor sensors used area = 2,626 $ft^2$

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-13. R-12 Magnetic Field Measurement

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | BEAMWIDTH *(F) DEG | BEAMWIDTH $f_2$ MHz | BEAMWIDTH DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | * TOTAL STRUCTURE DEVIATION (D) PERIM. DEFL. | | | |
| R-12 | Magnetic Field Measurement | 9,900 | 4,500 | $2(10^2)$ | 1 | ~0.03 | 360×360 | ~0.03 | 360×360 | 0.03 MHz | -- | 12 IN. | Optional | | Deviation shown for 150-ft boom |
| C-91 | Magnetometer and Support Beam | 200 | 23.3 | $>2(10^2)$ | 1<br>1.0 | -- | 360×360<br>1.0 | -- | 360×360 | 1.0 MHz | | <1°<br>1.0 | YES | 1.0 | Boom total = 165 ft from center |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-14. R-13 Meteorology: Atmospheric RF Radiometry

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEU-VERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | BEAMWIDTH *(F) DEG | BEAMWIDTH $f_2$ MHz | BEAMWIDTH DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | * TOTAL STRUCTURE DEVIATION (D) PERIM. DEFL. | | | |
| R-13 | Meteorology: Atmospheric | | | | | | | | | $2(10^3)$ - | 0.002 | 0.02 | Spins about | | Assumed to revolve about tan- |
| | RF Radiometry | 9,900 | 4,500 | $2(10^2)$ | 1,264 | $2(10^3)$ | 0.9×0.9 | $3(10^5)$ | 0.5×0.5 | $3(10^5)$ MHz | IN. | IN. | tangent to orbit | | gent to orbit at 1 rev/sec to |
| | | | | | | | | | | | | | | | map atmos from polar orbit |
| C-1 | 60 Ft Paraboloid Antenna | 9,100 | 1,980 | $2(10^2)$ | $1.4(10^3)$ | $2(10^3)$ | 0.58× 0.58 | $2.5(10^3)$ | 0.47× 0.47 | $2(10^3)$ - $2.5(10^3)$ MHz | 0.03 IN. | 0.3 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | $1.68(10^{-3})$ | | $6.7(10^{-2})$ | | $1.12(10^{-4})$ | |
| C-4 | Lenticular Communications | | | | | | | | | | | | | | |
| | Satellite | 1,484 | 78 | $2(10^4)$ | ← | | NOT APPLICABLE | | | | | | → | NO | |
| | Concept Rating | | | | | | | | | | | | | | |
| C-9 | Large Compressible Member | | | | | | | | | | | | | | Electrical power supply can |
| | Truss Struct for Space | | | | | | | | | | | | | | vary from 500 to 2,200 lb |
| | Antenna | 6,527 | 1,800 | $2(10^4)$ | 4,250 | $2(10^3)$ | 0.34 × 0.34 | $3.7(10^3)$ | 0.19 × 0.19 | $2(10^3)$ - $3.7(10^3)$ MHz | | 0.2 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | $5.7(10^{-3})$ | | 0.1 | | $5.7(10^{-4})$ | |
| C-10 | Folding Petal Struct for | | | | | | | | | | | | | | Electrical power supply can |
| | Large Space Antenna | 3,015 | 1,000 | $2(10^4)$ | 1,000 | $2(10^3)$ | 0.7 × 0.7 | $7.4(10^3)$ | 0.19 × 0.19 | $2(10^3)$ - $7.4(10^3)$ MHz | 0.1 IN. | 0.1 IN. | YES | | vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.792 | | 1.0 | | | $1.81(10^{-2})$ | | 0.2 | | $2.87(10^{-2})$ | |
| C-11 | 30 Ft. Petal Paraboloid | | | | | | | | | | | | | | Electrical power supply can |
| | Antenna | 3,018 | 1,200 | $2(10^4)$ | 400 | $2(10^3)$ | 1.2 × 1.2 | $3(10^3)$ | 0.8 × 0.8 | $2(10^3)$ - $3(10^3)$ MHz | 0.05 IN. | 0.2 | | | vary from 500 to 2,200 lb |
| | Concept Rating | | | | 0.317 | | 0.58 | | | $3.36(10^{-3})$ | | 0.1 | | $6.18(10^{-5})$ | |
| C-12 | Vista Direct Home TV | | | | | | | | | | | | | | Tolerance limitations do not |
| | Broadcast | 4,255 | 14,730 | $2(10^4)$ | | | | | | 60-810 MHz | | 0.33 IN. | YES | | permit this use |
| | Concept Rating | | | | | | | | | 0 | | $6.06(10^{-2})$ | | 0 | |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

# Table 9-14. R-13 Meteorology: Atmospheric RF Radiometry, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| REQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | MANEUV-ERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | WEIGHT LBS | VOLUME CU FT | | * (A) APERTURE AREA SQ FT | BEAMWIDTH $f_1$ MHz | * (F) DEG | $f_2$ MHz | DEG | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION SUR-FACE | (D) PERM. DEFL. | | |
| C-15 | Conic Scissors Parabola | | | | | | | | | | | | | EPS variable from 500 to |
| | Array | <2,000 | 1,000 | $2(10^4)$ | 400 | $2(10^3)$ | 1.2 x 1.2 | $3.7(10^3)$ | 0.63 x 0.63 | $2(10^3)$ - $3.7(10^3)$ MHz | 0.03 IN. | 0.2 IN. | YES | 2,200 lb |
| | Concept Rating | | | | 0.317 | | 0.58 | | | $5.7(10^{-3})$ | | 0.1 | | $1.05(10^{-4})$ |
| | | | | | | | | | | | | | | |
| C-16 | Bidirectional Scissors Array | 1,250 | | | | | | | | 90-150 | | | YES | Lacks sizing |
| | Concept Rating | | | | | | | | | 0 | | | | 0 |
| | | | | | | | | | | | | | | |
| C-75 | Commercial TV, 150 Ft. | | | | | | | | | | | | | |
| | Dia (Fager) | 7,629 | 3,040 | $2(10^4)$ | 9.500 | $2(10^3)$ | 0.23 x 0.23 | $2.4(10^3)$ | 0.2 x 0.2 | $2(10^3)$ - $2.4(10^3)$ MHz | | 0.3 IN. | YES | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | $1.34(10^{-3})$ | | $6.7 \times 10^{-2}$ | | $8.98(10^{-5})$ |
| | | | | | | | | | | | | | | |
| C-100a | 50 Ft. Trussed Parabolic | | | | | | | | | | | | | |
| | Ant, with 750 $Ft.^2$ Solar Array | 2,000 | 570 | $2(10^4)$ | 1,000 | $2(10^3)$ | 0.7 x 0.7 | $1.35(10^4)$ | 0.1 x 0.1 | $2(10^3)$ - $1.35(10^4)$ MHz | | 0.055 IN. | YES | |
| | Concept Rating | | | | 0.792 | | 1.0 | | | $3.86(10^{-2})$ | | $3.66(10^{-1})$ | | $1.12(10^{-2})$ |
| | | | | | | | | | | | | | | |
| C-100b | 50 Ft. Trussed Parabolic | | | | | | | | | | | | | |
| | Ant. with 2,400 $Ft.^2$ Solar | | | | | | | | | | | | | |
| | Array | 2,600 | 500 | $2(10^4)$ | 1,000 | $2(10^3)$ | 0.7 x 0.7 | $1.35(10^4)$ | 0.1 x 0.1 | $2(10^3)$ - $1.35(10^4)$ MHz | | 0.055 IN. | YES | |
| | Concept Rating | | | | 0.792 | | 1.0 | | | $3.86(10^{-2})$ | | $3.66(10^{-1})$ | | $1.12(10^{-2})$ |
| | | | | | | | | | | | | | | |
| C-100c | 50 Ft. Trussed Parabolic | | | | | | | | | | | | | |
| | with Directed Solar Array | 1,615 | 570 | $2(10^4)$ | 1,000 | $2(10^3)$ | 0.7x0.7 | $1.35(10^4)$ | 0.1x1.0 | $2(10^3)$ - $1.35(10^4)$ MHz | | 0.055 IN. | YES | |
| | Concept Rating | | | | 0.792 | | 1.0 | | | $3.86(10^{-2})$ | | $3.66(10^{-1})$ | | $1.12(10^{-2})$ |
| | | | | | | | | | | | | | | |
| C-101 | 88 Ft. Countrawound Helix | | | | | | | | | | | | | Does not operate in desired |
| | (Fager) | 7,870 | 1,440 | $2(10^4)$ | $1.1(10^4)$ @ 15 MHz | | | | | 15-75 MHz | | 0.22 IN. | | frequency range |
| | Concept Rating | | | | 0.87 | | | | | 0 | | $9.1(10^{-2})$ | | 0 |

* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

## Table 9-14. R-13 Meteorology: Atmospheric RF Radiometry, Contd

USER REQUIREMENT/CONCEPT COMPARISON AND RATING

| RFQ OR CONCEPT NUMBER | IDENTITY | EXPERIMENT TOTAL | | ORBIT ALT. NM | CRITICAL PERFORMANCE PARAMETERS | | | | | | | | MANEUV-ERABLE | * RELATIVE CONCEPT RATING | REMARKS |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | * (A) APERTURE AREA SQ FT | BEAMWIDTH | | | | * (B) BANDWIDTH | * TOTAL STRUCTURE DEVIATION | | | | |
| | | WEIGHT LBS | VOLUME CU FT | | | $f_1$ MHz | * (F) DEG | $f_2$ MHz | DEG | | SUR-FACE | (D) PERIM. DEFL. | | | |
| C-102 | 100 Ft. Dia. Trussed Para- | | | | | | | | | | | | | | |
| | bolic Ant. with 2,100 Ft.$^2$ | | | | | | | | | | | | | | |
| | Fixed Cell Solar Array | 7,570 | 5,450 | $2(10^4)$ | 4,000 | $2(10^3)$ | 0.35 x 0.35 | $7.4(10^3)$ | 0.1x0.1 | $2(10^3) - 7.4(10^3)$ MHz | | 0.1 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | $1.48(10^{-2})$ | | 0.2 | | $2.96(10^{-3})$ | |
| | | | | | | | | | | | | | | | |
| C-103 | Horn Phased Array with | | | | | | | | | | | | | | |
| | 2,500 Ft.$^2$ Fixed Cell | | | | | | | | | | | | | | Does not operate in desired |
| | Solar Array | 27,572 | 5,450 | $2(10^4)$ | 2,410 | 800 | 1.5x1.5 | 900 | 1.5x1.5 | 800-900 MHz | | 0.19 IN. | YES | | frequency range |
| | Concept Rating | | | | 1.0 | | 0.36 | | | 0 | | $1.05(10^{-1})$ | | 0 | |
| | | | | | | | | | | | | | | | |
| C-104 | 7-Petalated 30 Ft. Dia. | | | | | | | | | | | | | | |
| | Antennas with RTG Power | 12,625 | 5,450 | $2(10^4)$ | 2,240 | $2(10^3)$ | 0.5x0.5 | $6(10^3)$ | 0.15 x 0.15 | $2(10^3) - 6(10^3)$ MHz | | 0.125 IN. | YES | | |
| | Concept Rating | | | | 1.0 | | 1.0 | | | $1.34(10^{-2})$ | | $1.6(10^{-1})$ | | $2.14(10^{-3})$ | |
| | | | | | | | | | | | | | | | |
| C-105 | Seven Dish Paraboloid TV | | | | | | | | | | | | | | |
| | Broadcast Antenna with | | | | | | | | | | | | | | |
| | Solar Array | 1,236 | 4,500 | $2(10^4)$ | 850 | $2(10^3)$ | 0.8x0.8 | $6.3(10^3)$ | 0.25 x 0.25 | $2(10^3) - 6.3(10^3)$ MHz | | 0.116 IN. | | | |
| | Concept Rating | | | | 0.67 | | 1.0 | | | $1.44(10^{-2})$ | | $1.72(10^{-1})$ | | $1.66(10^{-3})$ | |
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* RELATIVE CONCEPT RATING = (A) (F) (B) (D)

CONCEPT RATING = $\frac{\text{CONCEPT PERFORMANCE}}{\text{USER REQUIREMENT}} = \frac{R - ( )}{C - ( )}$

OR BY RECIPROCAL ACCORDING TO DESIRED PERF BOUNDARIES

9.2 RESEARCH, DEVELOPMENT, TEST, AND ENGINEERING. This section of the study is to identify the significant problems associated with the development of large light-weight structures relative to:

a. Manufacturing research and development.

b. EVA and the prerequisite orbital experiments unnecessary to evaluate the required astronaut assist.

c. Engineering test problems as related to the development of materials, requirements for full scale test methods simulating EVA.

The development of an acceptable configuration in the desired form would not be difficult if it were not for the inverse problem of reducing the structure to a practical packaged volume for subsequent deployment in orbit with an acceptable level of EVA assisted joining and assembly.

Most of the problems are related to the extreme size of the experiment concepts and the mechanical tolerances required for satisfactory operational performance. Some of these are:

a. Packaging and deployment.

b. Joining and assembly with EVA.

c. Alignment.

d. Thermal distortions.

e. Rigidity and maneuverability.

f. Ground testing and checkout.

9.2.1 Manufacturing Research and Development. The preponderance of required manufacturing R&D is peculiar to the rhombics, dipole arrays, and log periodics. Major areas are new or unique materials, manufacture, automatic deployment equipment for use in space, and EVA handling, assembly, and alignment.

Considerably less manufacturing R&D is required for the other concepts. Generally these have common materials and/or composites. An exception is the use of beryllium structure; manufacturing development is required for its fabrication. The degree of automatic deployment or erection is integral to the particular design. These range from fully automatic to a building block concept wherein the pieces are manually assembled in space.

9.2.2 Critical EVA. The extravehicular activity required in the erection of the large space structures proposed in the present study varies from simple astronaut monitoring and inspection tasks (e.g. inspecting an automatically deployed beam to

insure that its joints have properly locked) to gross maneuvering and fine alignment of large structural members of a size and mass clearly beyond the capabilities of the individual astronaut (e.g. maneuvering and aligning component sections 8 ft long, 5-1/2 ft in diameter and weighing approximately 1600 lb). The astronaut's ability or lack of ability to accomplish these and similar tasks must be determined before the primary large-structure experiments can be flown. This capability would be more clearly defined if the unknowns in some critical areas could be resolved from orbital experience prior to the primary large-structure experiment.

As a result of this study, the most critical areas in terms of establishing the constraints on astronaut participation in large structure erection are the following:

a. Maneuvering techniques.

b. Anchoring techniques.

c. Alignment techniques.

d. Fastening and joining techniques.

e. Inspection techniques.

f. Illumination techniques.

g. Safety and rescue techniques.

h. Communications techniques.

It is clear that many of the unknowns associated with astronaut EV capabilities must be experimentally determined to aid the engineering analysis and more detailed design of large space structures.

9.2.3 Prerequisite Orbital Experiments. The general nature and extent of all of the experiments for the Apollo Applications Program (AAP) has resulted in a trend to very large overall dimensions, extending up to 5 mi. in overall length. Even more ambitious experiments are suggested by some users. This, naturally, raises many questions regarding packaging, deployment and EVA functions definitions. If we accept the fact that a manned launch is committed to a particular experiment it is reasonable to believe that a suitable "pre-experiment" may be economically justified to insure success of the final experiment. In any event, many of the large experiments do require satisfactory demonstration of extended EVA tasks to provide essential design data for engineering.

It is, therefore, proposed that after initial ground test results of EVA simulation etc., some typical, relatively simple experiment is then desirable to check out certain basic problems such as:

a. Extended astronaut and AMU capabilities.

b. Communication and tether problems.

c. Maneuvering and aligning large components.

d. Joining and tool development.

e. Lighting requirements.

f. Equipment functioning.

g. Structural and thermal effects.

h. Gravity gradient characteristics.

Most of the concepts require some advances in various technologies and a nice balance between EVA requirements and automatic deployment in orbit. It is desirable, therefore, to devise a relatively simple experiment to introduce most, if not all, of the problems listed above.

9.2.4 Engineering Test and Evaluation. To provide the most effective and orderly development program it is necessary to conduct ground tests designed to demonstrate the ability of the experiment concept to:

a. Withstand ground handling, checkout, launch, and orbital environments.

b. Deploy in orbit to theoretical shapes and contours within specified tolerances.

c. Respond to prescribed attitude control dynamic parameters.

d. Meet specified operational requirements.

Whenever possible, full scale testing is prescribed. In the case of the extremely long concepts, structural modules or breadboards are utilized. These are defined as sections of full-scale structure assembled from full-scale structural elements and members.

9.2.4.1 Materials Development. Many of the materials selected for use in the various concepts are not new to space structure applications. The present technology provides a broad base of information to draw from. Some new materials, such as metallic and non-metallic meshes that show great promise in eliminating distortion due to thermal gradients in long beams, will require extensive testing and development. Also included, but not limited to, are inflatable materials such as fiber-glass sheeting with mylar lining, and photolyzable materials over metallic screens. Development of these materials should include investigation of structural properties, thermal properties, thermo-vacuum deterioration, emissivity, and radiation effects. A more detailed enumeration of significant materials development will only be possible after the experiment concept designs are further defined.

9.2.4.2 Joining Techniques and Tool Development. All of the concepts that do not rely on completely automatic deployment, require extensive astronaut assist during deployment and assembly. Although the experiment concepts evaluated in Task I and Task II do not include the selection of particular and unique tools, it is apparent

that a new generation of joining techniques, tools, and support equipment must be developed. Power tools presently under development, such as the Martin-NASA zero-reaction tool with interchangeable attachments, permits the astronaut to cut, drill, hammer, or torque fasteners. Special devices under development by NASA that allow capturing of the tool in the head of the fastener and provides easier tool-to-fastener alignment could possibly be utilized (see Figure 9-1). Tools utilizing self contained chemicals for gas pressure generation as shown in Figure 9-2, are another possible development. Although universal type tools that would reduce the number of tools required appears desirable, it is equally important that such tools not be complex, requiring excessive adjustment or difficult manipulation. Other examples of processes that require further development are: exothermic brazing; electron beam welding; application of laser or maser principles; also cold welding processes and adhesive bonding.

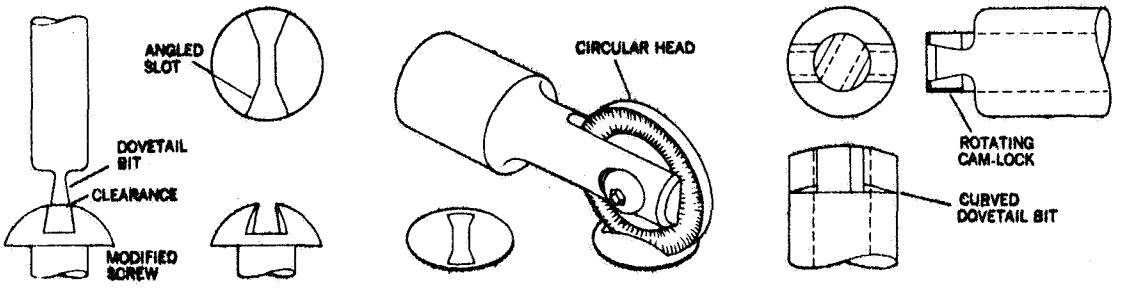



Figure 9-1. Tools Under Development in NASA Studies

To allow effective utilization of these tools, stabilization, in the form of some type of positive attachment, must be developed. A non-rigid tether is inadequate since it affords only one-degree of restriction. A possible development would be an attachment or harness with two or three telescoping rigid legs adaptive to the space structure, giving the astronaut complete stability while still retaining the use of the arms and legs (see Figure 9-3). Suitable hard points for attachment would have to be provided in the concept designs that have large open trusses or fragile members fabricated from mesh materials.

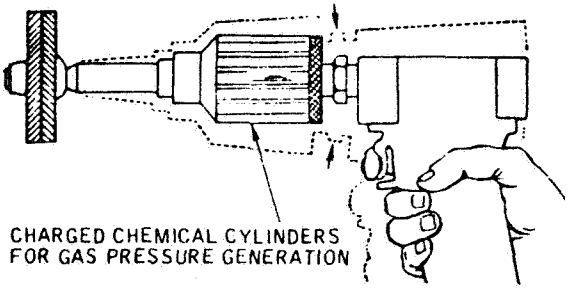



Figure 9-2. Portable Tool for Locking Fasteners in Space

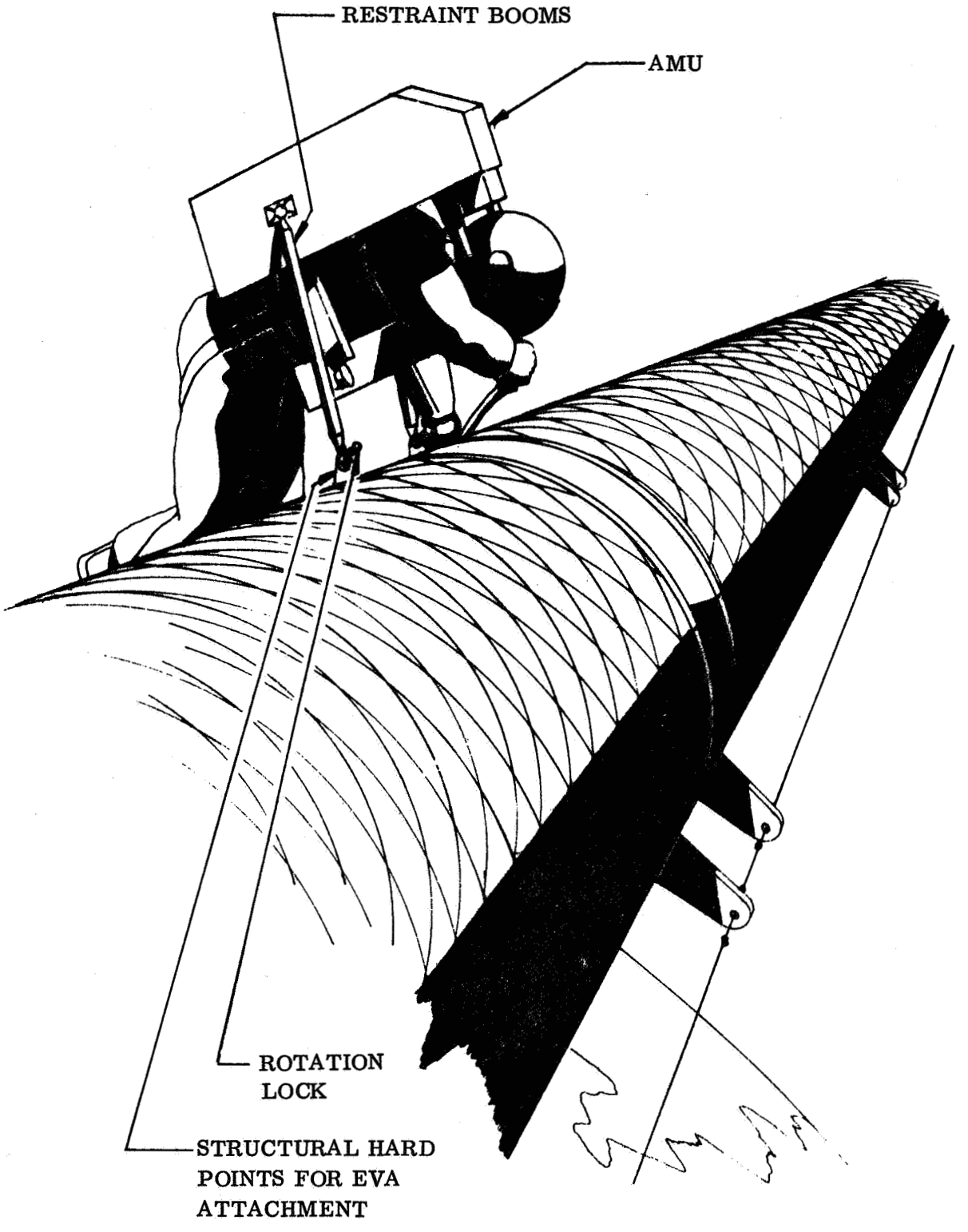



Figure 9-3. Joint Fabrication on Concept 51A

Equipment unique to a particular concept design, such as the assembly sled in Concept 1 must be developed, and the expense related to the quality and performance required.

9.2.4.3 Simulated EVA. All of the tools, maneuvering units, communications devices, and lighting equipment concerned with joining, stabilizing, inspecting or other unique tasks associated with the experiment concept EVA requirements must first be evaluated in a simulated environment. Neutral buoyancy techniques, frictionless-motion (air bearing) devices, six-degree-of-freedom simulators and man rated space chambers would be required. Simulated EVA would also be required in the group deployment and assembly tests of the concepts that do not utilize fully automatic deployment.

9.2.4.4 Structural Sampling Tests. The basic structural components and elements such as but not limited to, truss sections, boom sections, extendible tube sections, spring joints, lock mechanisms, and spiders appropriate to a particular concept would be tested as follows:

a. Design Limit Load — The basic design loads and ultimate strength in an ambient atmosphere would be established.

b. Thermal Cycling — Distortion would be evaluated under simulated thermal conditions in a vacuum environment.

9.2.4.5 Component Testing. Operational testing under simulated environments should be conducted on the following components and accessory systems:

a. Deployment Mechanisms — Components such as typical joints incorporating bearings, electric motors and gears would be operated under load in an environment simulating the thermovacuum conditions of space. Operation would follow exposure to the environment for periods equal to the scheduled orbital plan, in order to evaluate lubrication and binding problems.

b. Electronics and Instrumentation Packages — Specimens would be subjected to temperature/humidity and handling shock simulating the prelaunch environment; vibration, shock, acceleration, and acoustic levels simulating the launch environment; and operational performance evaluation in a thermovacuum simulation of the space environment.

9.2.4.6 Deployment and Assembly Evaluation. For the Class I and III concepts a full-scale structural model would be utilized. The Class II concepts would utilize a structural module of approximately 100 ft in length. The concepts that rely on astronaut assist during the deployment and assembly would utilize simulated EVA wherever possible to provide the most realistic evaluation. A balanced force system of numerous suspension tapes attached to the deploying structure is a possible method of providing a simulated zero-g environment (see Figure 9-4). At full deployment the truss

struts and joints would be relatively stress free.

All tests should be performed in a temporarily recontrolled area with minimum extraneous air motion.

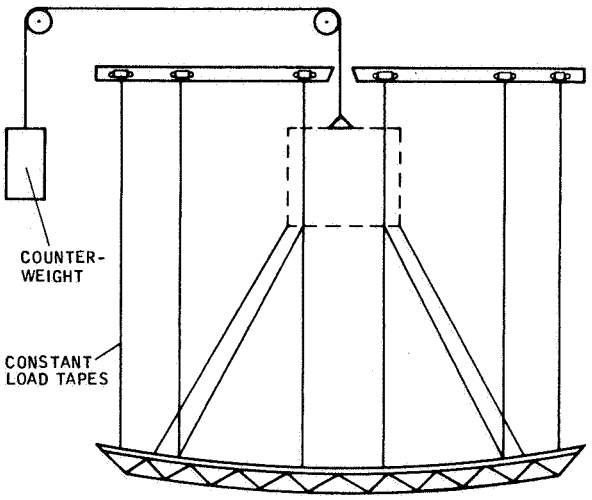



Figure 9-4. Deployment in a Zero-g Field

9.2.4.7 Distortion Evaluation. This test is a primary requirement for the rhombics, dipole array, and log periodic concepts only. After the full-scale model has been deployed in a simulated zero-g environment, the distortion of the contoured reflective surface could be measured by the tracing beam method.

9.2.4.8 Dynamic Modal Survey. A full-scale structural model should be used in this test for the Class I concepts. For the dipoles and magnetometers of the very large Class II concepts a structural module of approximately 100-ft length should be used. Scale models with an overall length of 100 ft or less should be used for the rhombic and log periodic concepts in Class II. The same simulated zero-g balanced force system and recontrolled area used in the deployment test would be utilized with the addition of very soft springs attached to the antenna concentrated masses (see Figure 9-5. Torsional forces about one axis at a time would be introduced through load transducers attached in the area of the attitude control thrusters by low frequency exciters. After modes have been defined from acceleration response, damping could be determined from the logarithmic decrement technique from response decay records obtained after the exciter armature cricuit is opened.

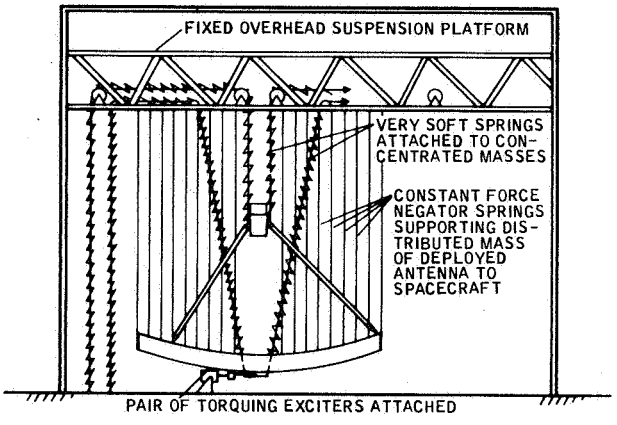



Figure 9-5. Suspension System for Zero-g Free-Free Torsional Mode Survey

9.2.4.9 Packaged Vibration. The packaged experiment should be subjected to both pre-launch and launch environments. Full scale structural models with all accessory equipment or mass simulation are required. The test conditions would be as follows:

a. Pre-launch — The packaged experiment would be subjected to a simulation of handling and transportation shock levels and temperature-humidity tests.

b. Launch — The packaged assembly would be attached to a fixture simulating the launch vehicle/adapter interface. The package excited in translation along the longitudinal axis and each of the major transverse axes. Equivalent viscous damping factors for modes in the packaged configuration could be calculated from the ratios of acceleration transmissibility. Following this the package would be attached to a rigid vibration fixture constrainted to rotate about the thrust axis. Phase-locked coupled forces would be applied. Significant torsional mode shapes and modal frequencies from 5 to 150 Hz would be measured. Modal damping could be determined from empirical acceleration transmissibility ratios.

9.2.4.10 RF Evaluation. The RF characteristics of the antenna concepts would be evaluated. Scale models that duplicate final deployed geometry could be used, and full scale tests would also be used where practical. To test the reflective qualities of the surface, the feed match, and the feed positioning mechanism of the Class I concepts, the reflective surface and support structure must be assembled around a trusswork frame stiff enough to maintain the required contour when positioned on edge. The feed support would also be reinforced. After checking for proper surface curvature and all necessary adjustments made, the assembly would be installed on the elevation and azimuth mount for complete RF testing. Testing would include evaluation of patterns, gain, efficiency, beam scanning range, beam deviation factor, beam pointing accuracy, and failure modes.

9.2.4.11 Summary. A summary of the typical engineering tests required for each experiment concept is shown in Table 9-15. This table provides a comparison of each of the concepts with respect to the engineering tests required. Some tasks are flagged to indicate an above normal degree of difficulty. For example Concept 3 being 1500 ft in dimaeter and utilizing fully automatic deployment presents some very difficult test problems even when scaled down to a more convenient size. The rhombic antennas because of size and geometry would also be very difficult to test.

9.2.5 Contributions to Advanced Technology. Review of the many concepts showed that the trend was toward very large structural sizes of experiments to satisfy the user. As a consequence, most of the proposals exhibit considerable ingenuity in methods of packaging and deployment in order to fit within the available volume and weight allowance in the manned launch Saturn vehicles.

The three most significant and fundamental ideas having wide application potential to advanced technology and the future space programs were:

a. Collapsible, extending tubular sections.

b. Expanding "rigid" space frame structures.

c. Wire mesh structures (gas deployed).

# Table 9-15. Summary of Engineering Tests

| | CONCEPTS | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | PARABOLOIDS AND HORN PHASED ARRAYS | | | | | | | | | | | | | | | | | RHOMBICS, DIPOLES, LOG PERIODS, AND MAGNETOMETERS | | | | | | | | | | | | | | SOLAR CELLS, X-RAY DETECT. AND METE-OROID COLLECTORS | | | | | | | | | | |
| | 1 | 3 | 9 | 10 | 11 | 12 | 15 | 16 | 75 | 100a | 100b | 100c | 101 | 102 | 103 | 104 | 105 | 2b | 6 | 13 | 20 | 52a | 52b | 2a | 7 | 17 | 51a | 51b | 53 | 54 | 91 | 5 | 8 | 14 | 18 | 19 | 71 | 72 | 73 | 81 | 82 | 85 |
| 1. Materials Investigation | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| 2. Joints, Tools and Process Development | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | x | X | X | X | X | X | X | X | X | X | X | X | X | X |
| 3. Simulated EVA | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| A. Observation and Inspection Techniques | X | ● | X | X | X | X | X | X | ● | X | X | X | X | ● | X | X | X | ● | X | ● | ● | ● | ● | X | ● | X | ● | ● | ● | X | X | X | X | X | X | X | X | X | X | X | X | ● |
| B. Assembly Operations | X | | X | | | | X | X | | | | | | | | | | ● | X | ● | ● | ● | ● | X | ● | X | ● | ● | ● | X | X | | | | | | | | | | | |
| 4. Structural Sampling | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| A. Design Limit | X | ● | X | X | X | X | X | X | X | X | X | X | X | ● | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| B. Thermal Cycling | X | ● | X | X | X | X | X | X | X | X | X | X | X | ● | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| 5. Component Evaluation | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| A. Deployment Mechanism | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | ● | X | ● | ● | ● | ● | X | ● | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| B. Electronics and Instrumentation | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| 6. Deployment (Simulated Zero-g) | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| A. Full Scale | X | | X | X | X | X | X | X | ● | X | X | X | X | ● | X | X | X | | | | | | | | | | | | | | | X | X | X | X | X | X | X | X | ● | X | X |
| B. Module (Up to 100-ft) | | | | | | | | | | | | | | | | | | | | | | | | X | X | | | X | | X | X | | | | | | | | | | | |
| C. Scale Model (Up to 100-ft) | | ● | | | | | | | | | | | | | | | | X | X | X | X | X | X | | | X | X | | X | | | | | | | | | | | | | |
| 7. Geometry Accuracy (Zero-g) | X | | X | X | X | X | X | X | X | X | X | X | X | ● | X | X | X | | | | | | | | | | | | | | | | | | | | | | | | | |
| 8. Dynamic Modal Survey (Zero-g) | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| A. Full Scale | X | | X | X | X | X | X | X | ● | X | X | X | X | ● | X | X | X | | | | | | | | | | | | | | | X | X | X | X | X | X | X | X | X | X | X |
| B. Module (Up to 100-ft) | | | | | | | | | | | | | | | | | | | | | | | | X | X | | | X | | X | X | | | | | | | | | | | |
| C. Scale Model (Up to 100-ft) | | X | | | | | | | | | | | | | | | | X | X | X | X | ● | ● | | | X | X | | X | | | | | | | | | | | | | |
| 9. Packaging - Models and Full Scale | X | ● | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | ● | X | X | X | ● | ● | X | ● | X | ● | ● | ● | X | X | X | X | X | X | X | X | X | X | X | X | X |
| 10. Packaged Vibration - Full Scale | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| 11. RF Evaluation | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | | |
| A. Model | | ● | | | | | | | X | | | | | X | | | | X | X | X | X | X | X | X | X | X | X | X | ● | X | X | | | | | | | | | | | |
| B. Full Scale | X | | X | X | X | X | X | X | | X | X | X | X | | X | X | X | | | | | | | | | | | | | | | | | | | | | | | | | |

● Task Complexity Above Average

Practically all of the structures considered depend, in varying degrees, on one or more of the three stated areas of development. This can also be taken as an indication that future space programs are somewhat dependent on defining the practicality of these basic ideas. In this context, therefore, the NASA experiments should be considered as essential for advanced technology.

From the foregoing the method of rating technological value to future programs would be in degree of contribution, as:

a. Excellent — Having broad application to all programs.

b. Good — General advances in space structures.

c. Normal — Detail contributions in all areas.

9.3 COST AND SCHEDULES. An abbreviated cost and schedule analysis was undertaken for each of the 42 concepts considered in Task I and Task II of this study. One of the primary objectives of this portion of Task III was to identify major differences in the R&D program costs, the unit costs, and the schedule requirements that might be expected among the various structure concepts studied. The data generated provide an input to the subsequent evaluation and selection process.

A simplified methodology for estimating costs and development schedules was required because of the number of concepts involved. Further, because of the lack of detailed definition and information in some cases, substantial assumptions were necessary concerning these individual concepts. Both the methodology used and the assumptions that were made are discussed below.

9.3.1 Methodology. Nonrecurring and recurring cost and a preliminary development schedule were generated for each of the concepts considered. The nonrecurring costs include all research and development engineering, prototypes and other test hardware, and the test programs necessary for the production of the flight article. The recurring costs are those uniquely associated with the single flight article and may be thought of as the difference in total cost for a program with one flight and a program with two flights. The recurring cost includes all material and labor for fabrication and assembly, acceptance testing, and launch integration and operations of the single flight article.

Initially, a brief development plan and schedule were generated for each of the concepts based on the complexity of the concept, state-of-the-art of the technology involved, material research and development required, and other associated influences. In the case of some of the concepts these schedules were based on a more detailed development plan investigation available from previous work. For others, where no analog was readily available, comparisons with the more defined concepts were made to assess the relative complexity and overall task difficulty.

The recurring program cost is the sum of the engineering costs and hardware costs charged to the single flight unit. The engineering task includes system/subsystem tests (acceptance testing) and launch integration and operations, and the hardware cost includes the single article unit cost.

9.3.2 Ground Rules. In order to facilitate and simplify the cost and schedule estimates, a series of general and specific assumptions were made concerning the overall program and the individual concepts studies.

The general program ground rules were as follows:

a. The space structure concepts are to be flown as part of the AAP program on Saturn IB or Saturn V in the 1969 to 1972 time period (or as the experiment R&D program permits).

b. All costs are in 1967 dollars.

c. The program hardware will consist of one flight article, and ground test articles and components as required.

d. The program is assumed to be relatively austere and the schedule is assumed to be a tight, nominally paced development, but in no sense a crash program.

e. It is assumed that there will be a separate AAP experiment integration contractor involved with overall responsibility for that effort.

f. It is assumed that one company will handle the entire experiment task with the exception of the AAP integration.

g. No costs were included for basic extravehicular activity capability development that may be required.

h. No costs were included for precursor experiments.

i. GSE costs were not included.

j. Capital facilities for assembly or testing are assumed to be available and no costs were included for unique facilities that may be required.

k. Costs for launch vehicles, for the AAP spacecraft and the SSESM (when used) are not included.

9.3.3 Results. The results of the cost and schedules analysis are summarized in Table 9-16. This table includes nonrecurring, recurring, and total program costs and the simplified milestone schedule. Also included is the unit cost of the structural subsystem for a direct comparison of the structural aspects of each concept. The costs presented in this table are believed to be representative and compatible in all cases with the level of definition available.

9.4 EVALUATION OF CONCEPTS. Final Convair evaluation presentation of study results is expected to cover concept descriptive parameters, a practicality evaluation, an applicability evaluation, and a means for recording the combined results and recommendations, was prepared in accordance with the results of a conference with the NASA/MSFC project monitor. Evaluation criteria headings from the table are defined below. Performance rating numbers are those derived in Tables 9-1 through 9-14.

It should be particularly noted that rejection of a concept in the terms of the evaluation summary does not necessarily mean that the concept could not be modified in size of configuration to provide an acceptable user concept. It only means that the concept, as defined, is considered either not feasible or not applicable during the AAP time period.

9.4.1 Key Descriptive Parameters. The descriptive parameters provide a means for identifying the concept, the originator, expected launch vehicle, recommended operating orbit, and the undeployed concept experiment weight and volume. One of the key decisions possible from the description is that of rejection of the concept if a single launch vehicle is employed and the concept experiment weight and volume exceed allowable payload weight and payload volume of the vehicle.

9.4.2 Practicality Evaluation. The practicality evaluation consists of combining estimates of feasibility, technology contribution, schedule, and costs to form a summary rating of: good, fair, poor, or unacceptable. Under "feasibility," the more detailed parameters uncovered in tasks 1 and 2 of the study activity are used by qualified designers to rate structural feasibility, EVA requirements, and deployment feasibility. The EVA requirements are shown in hours, with a large number regarded as undesirable. In structural feasibility evaluation, consideration is given to material properties, thermal stability, attitude control reactions (dynamics), rigidity, packaging ratios (deployed/undeployed dimensions), space environmental deterioration versus time, deployment loads, interacting deflections, and manufacturing problems.

9.4.3 Technology Contribution. Technology contribution is estimated by considering the probable gain over current techniques in assembly of large structures in space if the candidate concept experiment is implemented and accomplished. The technology gains are reflected indirectly by an increase in extravehicular activity effectiveness (quality of work or accuracy), a reduction in extravehicular activity time, and a consequent gain in the degree of automation without degradation in deployment probability. Consideration also should be given to probable structural techniques that will be required for assembly of large manned or unmanned stations or spacecraft in space for interplanetary exploration, enduring astronomical observatories, communication stations (broadcast, high-density traffic, earth airspace traffic monitoring and control) and manufacturing in space. The necessary activities in planning, design manufacturing, testing, assembly, adjustment, and operation of a candidate concept would develop a capability to accomplish other space tasks. The technology contribution rating needs to consider the importance of the key techniques that are likely to result if a given candidate concept is chosen for one of the structure experiments.

9.4.4 Deployment Feasibility. Deployment feasibility is estimated by qualified designers in terms of the number of operations required, the size of the largest piece, special tool requirements, the basic mechanical design feasibility, and probability of successful deployment. The number of EVA hours also tends to indicate complexity of deployment. In fact, if the number of EVA hours for deployment of a device exceeds some limit, such as 60 hours, it may be rejected, and the reason for the rejection will be stated in the remarks column.

9.4.5 Schedule. The scheduled flight availability date is estimated by assuming a go-ahead date of 1 January 1967 for each candidate concept and adding to it the total time in years required for continuing design, development, fabrication, test, qualification, and integration with the carrier vehicle. If the flight availability date exceeds 1972 or some other desired date, the concept can be rejected.

9.4.6 Costs. Costs are shown in terms of a fixed cost for each candidate concept development effort and the recurring cost per additional unit. Some limit may be placed to reject by cost is so desired.

9.4.7 Applicability Evaluation. Applicability evaluation rates each candidate concept against 13 potential user areas in terms of the performance desired by each user requirement of concept performance and user requirements as shown in the user requirement/concept comparison and rating sheet. The relative concept ratings, obtained by comparison of concept critical performance values against ideal user requirements, are assembled into an applicability evaluation matrix. The relative concept ratings (against the 13 user requirements) per concept are combined to give the combined user rating number. The user rating number can be converted to the qualitative scale by evaluating "good" as between 1 and $10^{-2}$, "fair" between $10^{-2}$ and $10^{-4}$, "poor" between $10^{-4}$ and $10^{-6}$, and "unacceptable" less than $10^{-6}$.

9.4.8 Results or Recommendations. The results or recommendations portion of the concept evaluation and recommendation table now consists of three terms per concept. The practicality term indicates: 1) the combined estimate obtained under the practicality evaluation (good, fair, poor, or unacceptable), 2) the combined user rating transformed into qualitative terms (good, fair, poor, or unacceptable), and 3) a final conclusion which indicates rejection as a candidate or selection for further evaluation. A "weak link" criterion is used in accumulating the ratings in the first preliminary evaluation i.e., the final rating is the poorest of the reviewed ratings.

One can then scan the remaining candidate concepts as indicated by the rating rejection or selection column. Again, the weak-link criterion is applied. If more than three different kinds of candidates survive in the good or fair category, the combined user rating numbers can be written next to the good or fair entry to indicate user requirement/concept rating influence. Continuing rejection of lower performance or applicability candidates then occurs until only the desired three remain.

In each case where a concept is rejected, the basis for rejection is stated in the adjacent remarks column. Likewise, for the surviving three candidates in three differing concept areas, the basis for selection is stated. The remarks column will also contain additional information, such as interferometer angular resolution, so that these additional considerations can be used to rate a concept.

Table 9-16. Concept Ev

| CON-CEPT NO. | CONCEPT IDENTITY | LAUNCH VEHICLE | ORBIT ALT. N. MI. | ORBIT INCL (deg) | PACKAGED EXPERIMENT WEIGHT (lb) | PACKAGED EXPERIMENT VOLUME ($ft^3$) | FEASIBILITY STRUCTURE | FEASIBILITY EVA HRS | FEASIBILITY DEPLOYMENT | STRUCTURES TECHNOLOGY CONTRIBUTION | SCHEDULE FLIGHT DATE | COSTS ($10^6) FIXED | COSTS ($10^6) RECURRING | 1 LONGWAVE RADIO ASTRONOMY | 2 VHF-UHF-SHF RADIO ASTRONOMY | 3 MILLIMETER + SUBMILLIMETER RADIO ASTRONOMY | 4 EARTH IONOSPHERE ABSORPTION | 5 POINT-TO-POINT RADIO COMMUNICATION | CO |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 1 | 60 Ft Paraboloid Antenna (NASA/MSFC) | SAT-V | $2(10^2)$ / $2(10^4)$ | 28.5 | 9,100 | 1,980 | Fair | 40 | Good | Normal | 1970.5 | 7.3 | 2.0 | 0 | $9.3(10^{-6})$ | $9.1(10^{-5})$ | 0 | $1.93(10^{-2})$ | |
| 2a | Broadside Dipole Antenna Array (Brown Engr.) | SAT-V | $>2(10^4)$ | 28.5 | 18,133 | 3,400 | Good | 25 | Unaccept. | Good | 1969.6 | 8.0 | 2.9 | $2.8(10^{-4})$ | 0 | 0 | $2.8(10^{-4})$ | 0 | |
| 2b | 300M Rhombic Antenna (Brown Engr.) | SAT-V | $>2(10^4)$ | 28.5 | 8,370 | 1,890 | Fair | 76 | Poor | Normal | 1970.5 | 13.9 | 5.1 | $3.03(10^{-5})$ | 0 | 0 | $3.03(10^{-5})$ | 0 | |
| 3 | 1500M Paraboloid Antenna (Astro Research) | SAT-V | $>2(10^4)$ | 28.5 | 4,530 | 12,000 | Poor | 8 | Poor | Good | 1972 | 9.5 | 1.6 | $3.4(10^{-4})$ | 0 | 0 | $3.4(10^{-4})$ | 0 | |
| 4 | Lenticular Communications Satellite (Goodyear) | SAT-V | $>2(10^4)$ | 28.5 | 1,484 | 78 | Fair | 1 | Fair | Excellent | | | | 0 | 0 | 0 | 0 | Not Completely Evaluated | N E |
| 5 | Solar Cell Arrays (Boeing) | SAT-IB/ Centaur | $>2(10^4)$ | 28.5 | 1,960 | 3,850 | Fair | 0 | Fair | Good | 1970 | 6.9 | 15.3 | 0 | 0 | 0 | 0 | 0 | |
| 6 | 600M Rhombic Antenna (NASA/MSFC) | SAT-V | $>2(10^4)$ | 28.5 | 15,600 | 2,000 | (No Data Presented) (Fair) | 3 | (No Data Presented) (Good) | Good | 1970.5 | 9.8 | 2.7 | $7.4(10^{-4})$ | 0 | 0 | $7.4(10^{-4})$ | 0 | |
| 7 | Member Structure Antenna (Ryan) | SAT-V | $>2(10^4)$ | 28.5 | 2,600 | 2,300 | Good | 80 | Unaccept. | Good | 1970.5 | 12.0 | 2.3 | $5.7(10^{-5})$ | 0 | 0 | $5.7(10^{-5})$ | 0 | |
| 8 | Large Solar Cell Array (Ryan) | SAT-IB/V | $2(10^2)$/ $2(10^4)$ | 50/ 28.5 | 5,500 | 1,400 | Good | | Good | Good | 1970 | 4.0 | 15.3 | 0 | 0 | 0 | 0 | 0 | |
| 9 | Large Compressible Truss Antenna (Ryan) | SAT-V | $2(10^4)$ | 28.5 | 6,527 | 1,800 | Unaccept. | 53 | Unaccept. | Good | 1970 | 12.0 | 2.9 | 0 | $3.5(10^{-5})$ | 0 | 0 | $9.45(10^{-2})$ | |
| 10 | Folding Petal Space Antenna (Ryan) | SAT-IB/V | $2(10^4)$ | 50/ 28.5 | 3,015 | 1,000 | Poor | | Poor | Normal | 1970 | 10.5 | 2.7 | 0 | $2.0(10^{-5})$ | 0 | 0 | $1.21(10^{-1})$ | |
| 11 | Paraboloid Antenna (30 Ft Petal) (TRW/STL) | SAT-V | $2(10^4)$ | 28.5 | 3,018 | 1,200 | Good | 3 | Fair | Normal | 1969 | 6.6 | 2.0 | 0 | $3.3(10^{-4})$ | 0 | 0 | $3.06(10^{-3})$ | |
| 12 | "Vista" Direct Home Broadcast Antenna Solar Array (RCA) | SAT-V | $2(10^4)$ | 28.5 | 4,255 | 14,730 | Fair | 0 | Fair | Excellent | 1971 | 17.2 | 10.7 | 0 | $1.9(10^{-6})$ | 0 | 0 | 0 | |
| 13 | Orbiting Radio Astronomy Observatory (TRW/STL) | SAT-V | 500 to 32,500 | 62 | 400 | 30 | Good | 0 | Good | Excellent | 1970 | 5.0 | 0.5 | $1.4(10^{-1})$ | 0 | 0 | $1.4(10^{-1})$ | 0 | |
| 14 | Deep Space Meteoroid Satellite** (Martin) | SAT-V | >1 AU** | 28.5 | 5,800 | 2,380 | Good | 0 | Good | Normal | 1970 | 13.1 | 2.3 | 0 | 0 | 0 | 0 | 0 | |
| 15 | Conic Scissors Parabola Antenna (Fairchild) | SAT-V | $2(10^4)$ | 28.5 | <2,000 | 1,000 | Good | 4/28 | Fair | Normal | 1970 | 10.4 | 2.3 | 0 | $3.2(10^{-6})$ | 0 | 0 | $5.2(10^{-3})$ | |
| 16 | Bidirectional Scissors Antenna Array (Fairchild) | SAT-V | $2(10^4)$ | 28.5 | 1,250 | 200 | Good | 33 | Good | Normal | 1970 | 11.6 | 2.5 | 0 | $4.0(10^{-7})$ | 0 | 0 | 0 | |
| 17 | Log Periodic Pyramidal Antenna (Fairchild) | SAT-V | $>2(10^4)$ | 28.5 | 2,084 | 100 | Fair | | Poor | Normal | | | | $3.1(10^{-3})$ | 0 | 0 | $3.1(10^{-3})$ | 0 | |
| 18 | Concertina Micrometeoroid Panel (Fairchild) | SAT-V | $2(10^4)$ | 28.5 | 2,000 | 1,000 | Good | 0 | Good | Normal | 1969.75 | 8.5 | 2.5 | 0 | 0 | 0 | 0 | 0 | |
| 19 | Solar Array, Rollout or Scissors (Fairchild) | SAT-IB/V | >200 | 50/ 28.5 | 140* | 11.6 | Unaccept. | | Fair | Normal | 1970 | 4.0 | 1.1 | 0 | 0 | 0 | 0 | 0 | |
| 20 | Extensible Rhombic Antenna (Fairchild) | SAT-V | $>2(10^4)$ | 28.5 | 18,000 | 1,000 | Fair | 2.5 | Fair | Good | 1970.5 | 12.9 | 4.6 | $4.2(10^{-3})$ | 0 | 0 | $4.2(10^{-3})$ | 0 | |

DESCRIPTIVE PARAMETERS | EXPERIMENT PRACTICALITY EVALUATION

9-45-1 FOLDOUT FRAME 1

9-45-2 FOLDOUT FRAME 2

luation Summary

| USER APPLICABILITY EVALUATION RATING PER USER REQUIREMENT CATEGORY | | | | | | | | | | RESULTS OR RECOMMENDATIONS | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 6 | 7 | 8 | 8a | 9 | 10 | 11 | 12 | 13 | Σ | | | | |
| ULTIPLE-ACCESS RADIO MUNICATION | TV BROADCAST COMMUNICATION | X-RAY ASTRONOMY | X-RAY IMAGING | GAMMA RAY ASTRONOMY | SOLAR POWER CONVERSION | METEOROID COLLECTION | MAGNETIC FIELD MEASUREMENT | RF METEOROLOGY OBSERVATIONS | COMBINED USER RATING | EXPERIMENT PRACTICALITY | USER APPLICABILITY | RATING, REJECTION, OR SELECTION | REMARKS |
| $.7(10^{-2})$ | $2.33(10^{-1})$ | 0 | 0 | 0 | 0 | 0 | 0 | $1.12(10^{-4})$ | $3.2(10^{-1})$ | Fair | Good | Fair | |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $5.6(10^{-4})$ | Unacceptable | Fair | Rejected | Evaluate as an interferometer. |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $6.1(10^{-5})$ | Poor | Poor | Poor | Evaluate also as an interferometer. |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $6.8(10^{-4})$ | Poor | Fair | Poor | Volume greater than adapter volume. |
| Completely luated | 0 | 0 | 0 | 0 | 0 | ? | 0 | 0 | -- | Fair | Not Known | -- | |
| | 0 | 0 | 0 | 0 | $3.7(10^{-1})$ | ? | 0 | 0 | $3.7(10^{-1})$ | Fair | Good | Fair | |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $1.5(10^{-3})$ | Fair | Fair | Fair | |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $1.1(10^{-4})$ | Unacceptable | Fair | Rejected | |
| | 0 | 0 | 0 | 0 | $2.5(10^{-1})$ | ? | 0 | 0 | $2.5(10^{-1})$ | Good | Good | Good | |
| $.0(10^{-1})$ | $7.08(10^{-1})$ | 0 | 0 | 0 | 0 | 0 | 0 | $5.7(10^{-4})$ | $9.0(10^{-1})$ | Unacceptable | Good | Rejected | |
| $.6(10^{-1})$ | $1.67(10^{-1})$ | 0 | 0 | 0 | 0 | 0 | 0 | $2.87(10^{-2})$ | $4.8(10^{-1})$ | Poor | Good | Poor | |
| $.5(10^{-2})$ | $6.6(10^{-2})$ | 0 | 0 | 0 | 0 | 0 | 0 | $6.18(10^{-5})$ | $9.4(10^{-2})$ | Fair | Good | Fair | |
| $.3(10^{-2})$ | $1.67(10^{-2})$ | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $3.0(10^{-2})$ | Fair | Good | Fair | Volume greater than adapter volume. |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $2.8(10^{-1})$ | Good | Good | Good | |
| | 0 | 0 | 0 | 0 | 0 | $2.9(10^{-1})$ | 0 | 0 | $2.9(10^{-1})$ | Fair | Good | Fair | **Stowed length exceeds length of stowage space for synchronous orbit. |
| $.2(10^{-2})$ | $6.6(10^{-2})$ | 0 | 0 | 0 | 0 | 0 | 0 | $1.05(10^{-4})$ | $1.0(10^{-1})$ | Fair | Good | Fair | |
| $.0(10^{-4})$ | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $6.0(10^{-4})$ | Fair | Fair | Fair | |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $6.2(10^{-3})$ | Poor | Fair | Poor | Evaluate as an interferometer. |
| | 0 | 0 | 0 | 0 | 0 | $5.0(10^{-2})$ | 0 | 0 | $5.0(10^{-2})$ | Fair | Good | Fair | |
| | 0 | 0 | 0 | 0 | $1.4(10^{-2})$ | 0 | 0 | 0 | $1.4(10^{-2})$ | Unacceptable | Good | Rejected | |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $8.4(10^{-3})$ | Fair | Fair | Fair | |

| CON-CEPT NO. | CONCEPT IDENTITY | LAUNCH VEHICLE | ORBIT ALT. N. MI. | ORBIT INCL (deg) | PACKAGED EXPERIMENT WEIGHT (lb) | PACKAGED EXPERIMENT VOLUME ($ft^3$) | FEASIBILITY STRUCTURE | FEASIBILITY EVA HRS | FEASIBILITY DEPLOYMENT | STRUCTURES TECHNOLOGY CONTRIBUTION | SCHEDULE FLIGHT DATE | COSTS ($10^6) FIXED | COSTS ($10^6) RECURRING | 1 LONGWAVE RADIO ASTRONOMY | 2 VHF-UHF-SHF RADIO ASTRONOMY | 3 MILLIMETER + SUBMILLIMETER RADIO ASTRONOMY | IO AI |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | DESCRIPTIVE PARAMETERS | | | | | | EXPERIMENT PRACTICALITY EVALUATION | | | | | | | | | | |
| 51a | 5KM/300M Sect. Colinear Dipole Antenna | SAT-V | $2(10^4)$ | 28.5 | 2,084 | 3,500 | Good | 18 | Poor | Fair | 1970.5 | 9.3 | 1.6 | $1.7(10^{-3})$ | 0 | 0 | |
| 51b | Crossed-H Interferometer | SAT-V | $2(10^4)$ | 28.5 | 3,657 | 1,280 | Good | 2 | Good | Excellent | 1970.5 | 19.8 | 3.4 | $1.7(10^{-4})$ | 0 | 0 | |
| 52a | Rhombic Antenna - External Structure | SAT-V | $2(10^4)$ | 28.5 | 23,866 | 4,100 | Poor | 4 | Fair | Good | 1970.5 | 13.2 | 4.7 | $1.9(10^{-2})$ | 0 | 0 | |
| 52b | Rhombic Antenna - Internal Structure | SAT-V | $2(10^4)$ | 28.5 | 22,000 | 4,100 | Fair | 4 | Fair | Good | 1970.75 | 13.0 | 3.6 | $1.1(10^{-2})$ | 0 | 0 | |
| 53 | 3KM/300 Dipoles End Fire Antenna | SAT-V | $2(10^4)$ | 28.5 | 4,000 | 800 | Good | 71 | Poor | Poor | 1970 | -- | -- | ? | 0 | 0 | |
| 54 | Broadside Array - Antenna | Not Estimated | Not Estimated | -- | Not Estimated | Not Estimated | Fair | >100 | Poor | Poor | -- | -- | -- | ? | 0 | 0 | |
| 71 | X-Ray Detector Array (352 $ft^2$) (Gamma Ray) | SAT-IB | $2(10^2)$ | 50 | 15,118 | 1,580 | Good | 3 | Good | Normal | 1970 | 8.8 | 1.9 | 0 | 0 | 0 | |
| 72 | X-Ray Detector Array (1000 $ft^2$) | SAT-IB | $2(10^2)$ | 50 | 13,576 | 1,580 | Good | 3 | Good | Good | 1970 | 8.6 | 2.0 | 0 | 0 | 0 | |
| 73 | X-Ray Telescope, Reflecting, Grazing Incidence* | SAT-IB | $2(10^2)$ | 50 | 14,000* | 1,580* | Fair* | 4* | Fair | Excellent | 1972 | 12.2* | 11.3 | 0 | 0 | 0 | |
| 75 | 150 ft Diameter Commercial Antenna With 3000 $ft^2$ Solar Array #81 | SAT-V | $2(10^4)$ | 28.5 | 7,629* | 3,040 | Good | 4 | Good | Excellent | 1970.5 | 18.9 | 12.8 | 0 | $5.9(10^{-5})$ | 0 | |
| 81 | 3000 $ft^2$ Solar Cell Array (For Comm) Used With Various Antennas | SAT-V | $2(10^4)$ | 28.5 | 3,238* | 1,600 | Fair | 4 | Fair | Normal | 1969 | 4.7 | 9.1 | 0 | 0 | 0 | |
| 82 | S-IVB Solar Cell Array | SAT-IB-V | $2(10^2)$ $2(10^4)$ | 50/ 28.5 | 4,312 | 740 | Good | 5 | Good | Excellent | 1969 | 3.3 | 9.3 | 0 | 0 | 0 | |
| 85 | Meteoroid Collector | SAT-IB | $2(10^2)$ | 50 | 3,192 | 1,600 | Fair | 4 | Good | Normal | 1970 | 5.0 | 1.3 | 0 | 0 | 0 | |
| 91 | Magnetometer | SAT-IB | $2(10^2)$ | 50 | 200 | 23.3 | Good | 4 | Good | Excellent | 1969 | 1.8 | 0.2 | 0 | 0 | 0 | |
| 100a | 50 ft Diameter TV Broadcast Antenna With Pegasus Solar Cell Array | T-III-C | $2(10^4)$ | 0 | 2,000 | 570 | Good | 0 | Good | Normal | 1970.5 | 14.0 | 9.9 | 0 | $3.6(10^{-5})$ | 0 | |
| 100b | 50 ft Diameter TV Broadcast Antenna With Inflatable Solar Cell Array | T-III-C/ Centaur Boost | $2(10^4)$ | 0 | 2,600 | 500 | Fair | 0 | Good | Excellent | 1970.5 | 14.0 | 5.0 | 0 | $3.6(10^{-5})$ | 0 | |
| 100c | 50 ft Diameter TV Broadcast Antenna With Directed Solar Cell Array | Atlas/Adv. Centaur Boost | $2(10^4)$ | 0 | 1,615 | 570 | Good | 0 | Good | Good | 1970.5 | 14.0 | 6.1 | 0 | $3.6(10^{-5})$ | 0 | |
| 101 | 78 ft Contrawound Helix Voice Broadcast Antenna Array | SAT-V | $2(10^4)$ | 0 | 7,870 | 3,600 | Excellent | 3 | Good | Excellent | 1970 | 7.8 | 1.9 | 0 | 0 | 0 | |
| 102 | 100 ft Diameter Broadcast Array With Fixed Cell Solar Array | SAT-V | $2(10^4)$ | 0 | 7,570 | 5,450 | Good | | Good | Good | 1970.5 | 18.3 | 11.1 | 0 | $7.9(10^{-5})$ | 0 | |
| 103 | Broadcast Horn Array With Fixed Cell Solar Array | SAT-V | $2(10^4)$ | 0 | 27,572 | 5,450 | Good | | Good | Excellent | 1970.5 | 20.7 | 18.7 | 0 | $4.5(10^{-5})$ | 0 | |
| 104 | Seven Petalated TV Broadcast Antenna With Isotope Power | SAT-V | $2(10^4)$ | 0 | 12,625 | 5,450 | Fair | | Fair | Excellent | 1970 | 16.7 | 3.1 | 0 | $2.5(10^{-4})$ | 0 | |
| 105 | Seven Dish Paraboloid Antenna, With Solar Array | SAT-V | $2(10^4)$ | 0 | 1,236 | 4,500 | Good | 0 | Good | Normal | 1969.5 | 3.4 | 1.2 | 0 | $7.1(10^{-5})$ | $2.0(10^{-4})$ | 0 |

For EVA 60 hrs = Rejection Level

6. Concept Evaluation Summary, Contd

| USER APPLICABILITY EVALUATION RATING PER USER REQUIREMENT CATEGORY | | | | | | | | | | | | RESULTS OR RECOMMENDATIONS | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 4 | 5 | 6 | 7 | 8 | 8a | 9 | 10 | 11 | 12 | 13 | | | | | |
| EARTH NOSPHERE SORPTION | POINT-TO-POINT RADIO COMMUNICATION | MULTIPLE-ACCESS RADIO COMMUNICATION | TV BROADCAST COMMUNICATION | X-RAY ASTRONOMY | X-RAY IMAGING | GAMMA RAY ASTRONOMY | SOLAR POWER CONVERSION | METEOROID COLLECTION | MAGNETIC FIELD MEASUREMENT | RF METEOROLOGY OBSERVATIONS | $\Sigma$ COMBINED USER RATING | EXPERIMENT PRACTICALITY | USER APPLICABILITY | RATING, REJECTION, OR SELECTION | REMARKS |
| $1.7(10^{-3})$ | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $3.4(10^{-3})$ | Poor | Fair | Poor | |
| $1.7(10^{-4})$ | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $3.4(10^{-4})$ | Good | Fair | Fair | Rate as an interferometer in final selection ($1.7^{\circ}$ at 0.5 MHz) |
| $1.9(10^{-2})$ | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $3.8(10^{-2})$ | Poor | Good | Poor | |
| $1.1(10^{-2})$ | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $2.2(10^{-2})$ | Fair | Good | Fair | |
| ? | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | ? | Poor | ? | Rejected | Excessive EVA hours |
| ? | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | ? | Poor | ? | Rejected | Excessive EVA does not allow viewing of all angles due to solar pressure |
| ) | 0 | 0 | 0 | $3.5(10^{-2})$ | $8.2(10^{-2})$ | $1.0(10^{-4})$ | 0 | 0 | 0 | 0 | $3.5(10^{-2})$ | Fair | Good | Fair | Requires SAT-V or unmanned SAT-1B launch. |
| ) | 0 | 0 | 0 | $1.0(10^{-1})$ | 0 | 0 | 0 | 0 | 0 | 0 | $1.0(10^{-1})$ | Good | Good | Good | Requires SAT-V or unmanned SAT-1B launch. |
| ) | 0 | 0 | 0 | $6.7(10^{-2})$* | 1.0* | 0 | 0 | 0 | 0 | 0 | $1.07(10^{0})$* | Fair* | Good* | Fair* | *Present concept dynamically unsuitable. To be redesigned and re-evaluated. |
| ) | $1.05(10^{-5})$ | $6.5(10^{-2})$ | 1.0 | 0 | 0 | 0 | $2.5(10^{-1})$ | 0 | 0 | $8.98(10^{-5})$ | $1.3(10^{0})$ | Good | Good | Good | *Weight includes 433 lbs. for additional attitude control. |
| | 0 | 0 | 0 | 0 | 0 | 0 | $2.5(10^{-1})$ | 0 | 0 | 0 | $2.5(10^{-1})$ | Fair | Good | Fair | *Additional attitude control weight included with antenna. |
| | 0 | 0 | 0 | 0 | 0 | 0 | $2.6(10^{-1})$ | 0 | 0 | 0 | $2.6(10^{-1})$ | Good | Good | Good | |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $1.6(10^{-3})$ | 0 | 0 | $1.6(10^{-3})$ | Fair | Fair | Fair | |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 1.0 | 0 | $1.0(10^{0})$ | Good | Good | Good | Do not accept; poor user requirement definition, "weight"/user area. |
| | $2.58(10^{-1})$ | $2.92(10^{-1})$ | $1.67(10^{-1})$ | 0 | 0 | 0 | $6.3(10^{-2})$ | 0 | 0 | $1.12(10^{-2})$ | $7.9(10^{-1})$ | Good | Good | Good | |
| | $2.58(10^{-1})$ | $2.92(10^{-1})$ | $1.67(10^{-1})$ | 0 | 0 | 0 | $2.0(10^{-1})$ | 0 | 0 | $1.12(10^{-2})$ | $9.3(10^{-1})$ | Fair | Good | Fair | |
| | $2.58(10^{-1})$ | $2.92(10^{-1})$ | $1.67(10^{-1})$ | 0 | 0 | 0 | $6.3(10^{-2})$ | 0 | 0 | $1.12(10^{-2})$ | $7.9(10^{-1})$ | Good | Good | Good | |
| | 0 | $8.46(10^{-4})$ | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | $8.5(10^{-4})$ | Good | Fair | Fair | |
| | $3.0(10^{-1})$ | $2.02(10^{-1})$ | $6.66(10^{-1})$ | 0 | 0 | 0 | $1.8(10^{-1})$ | 0 | 0 | $2.96(10^{-3})$ | $1.4(10^{0})$ | Good | Good | Good | Volume greater than adapter volume. Requires unmanned launch or redesign. |
| | 0 | $5.0(10^{-3})$ | $4.02(10^{-1})$ | 0 | 0 | 0 | $2.1(10^{-1})$ | 0 | 0 | 0 | $6.2(10^{-1})$ | Good | Good | Good | Volume greater than adapter volume. Requires unmanned launch or redesign. |
| | $2.2(10^{-1})$ | $2.95(10^{-1})$ | $3.73(10^{-1})$ | 0 | 0 | 0 | 0 | 0 | 0 | $2.14(10^{-3})$ | $8.9(10^{-1})$ | Fair | Good | Fair | |
| | $6.24(10^{-2})$ | $1.93(10^{-1})$ | $1.4(10^{-1})$ | 0 | 0 | 0 | 0 | 0 | 0 | $1.66(10^{-3})$ | $4.0(10^{-1})$ | Fair | Good | Fair | Volume greater than adapter volume. |

Good = (1.0) > $(10^{-2})$
Fair = $(10^{-2})$ → $(10^{-4})$
Poor = $(10^{-4})$ → $(10^{-6})$
Unacceptable < $(10^{-6})$

9-46-5 FOLDOUT FRAME 3

9-46-4 FOLDOUT FRAME 4

# SECTION 10

# CONCEPT SELECTION

10.1 APPROACH. An important part of this study is the selection of three large-structure experiments for detailed analysis during the second half of the study. Evaluation criteria was discussed and agreed upon with the NASA-COR early in Task 3. These criteria included performance, feasibility, technology contribution, schedule, and costs. The results of the concept analysis and evaluation were presented in Section 9. Since some of the criteria are subjective, it is obvious that a final numerical rating is difficult, and it would then become necessary to assign a value to each user requirement area.

10.2 COMMONALITY. Although it is conceivable that a large structure could be made adaptable to "unrelated" user applications, a detailed study reveals an ever increasing number of incompatibilities and compromises. An example would be a solar cell array combined with X-ray detectors. Incompatibilities would exist with respect to pointing requirements, heat rejection, etc.

The only extensive commonality (or adaptability of concepts to alternative user applications) occurs for communication antennas. Table 10-1 provides a ranking of concepts in the order of Convair's predicted performance for each user requirement application. Critical performance characteristics are as previously defined: aperture, beamwidth, bandwidth, and structural deviation. The 100-ft-dia. expandable truss (Concept 102) performs well in many applications. If sub-millimeter wavelength operation is desired, Concept 105 (an array of seven 16-ft dishes) should be considered.

10.3 CONVAIR RECOMMENDATIONS. From the analysis and evaluation of approximately 40 large structures, the concepts considered to be most promising for application during the AAP time frame are listed in Table 10-2. Three specific concepts were recommended to NASA at the Mid-term Presentation on 1 March 1967. Concept 51b was recommended primarily because of the excellent performance potential for long wave radio astronomy. Concept 73 was recommended to fulfill a projected strong user requirement. A parabolic antenna, in the size range of 50- to 150-ft diameter, was recommended as a third concept.

An added reason for the recommendation of these three concepts is that they provide three uniquely different structures with respect to man's role and cover a broad range of structure technology. It is desirable to study the spectrum of problems associated with future space structures and their deployment, alignment, and operation.

Table 10-1. Communication Concept Ranking by Performance

## COMMUNICATION CONCEPT RANKING BY PERFORMANCE

| VHF–UHF–SHF RADIO ASTRONOMY | MM–SUB MM RADIO ASTRONOMY | POINT-TO-POINT COMMUNICATION | MULTIPLE-ACCESS RADIO COMMUNICATION | TV BROADCAST COMMUNICATION | RF METEOROLOGY OBSERVATION |
|---|---|---|---|---|---|
| 11 | 105 | 102 | 104 | 75 | 10 |
| 104 | 1 | 100 | 100 | 9 | 100 |
| 102 | – | 104 | 102 | 102 | 102 |
| 105 | – | 10 | 105 | 103 | 104 |
| 75 | – | 9 | 10 | 104 | 105 |
| 103 | – | 105 | 9 | 1 | 9 |
| 100 | – | 1 | 1 | 100 | 1 |
| 10 | – | 15 | 75 | 10 | 15 |

1 60 FT. PARABOLOID (P&VE, MSFC)
9 COMPRESSIBLE TRUSS PARABOLOID (RYAN)
10 FOLDING PETAL PARABOLOID (RYAN)
11 30 FT. PETAL PARABOLOID (TRW)
15 CONIC SCISSORS PARABOLOID (FAIRCHILD)
75 150 FT. EXPANDABLE TRUSS PARABOLOID
100 50 FT. EXPANDABLE TRUSS PARABOLOID
102 100 FT. EXPANDABLE TRUSS PARABOLOID
103 HORN ARRAY
104 SEVEN PETULATED DISH ARRAY
105 SEVEN DISH ARRAY

Table 10-2. Candidate Large Structures Concepts

# CANDIDATE LARGE STRUCTURES CONCEPTS

**LONG-WAVE RADIO ASTRONOMY**

- 6 600-METER RHOMBIC (NASA)
- 13 ORBITING RADIO ASTRONOMY OBSERVATORY (TRW)
- * 51b CROSSED H INTERFEROMETER
- 52b 1,200—RHOMBIC-INTERNAL STRUCTURE

**X & GAMMA RAY DETECTORS**

- 71 352-FT² X-RAY & GAMMA RAY DETECTOR
- * 73 FOCUSING X-RAY TELESCOPE

**MICROMETEOROID COLLECTORS**

- 85 MICROMETEOROID COLLECTORS

**MAGNETOMETER**

- 91 MAGNETOMETER EXTENDABLE TRUSS

**COMMUNICATIONS**

- * 75 150-FT EXPANDABLE TRUSS PARABOLOID
- * 100 50-FT EXPANDABLE TRUSS PARABOLOID
- * 102 100-FT EXPANDABLE TRUSS PARABOLOID
- 104 SEVEN-PETULATED-DISH/ISOTOPE POWER
- 105 SEVEN-DISH PARABOLOID

**SOLAR CELL ARRAYS**

- 8 SOLAR CELL ARRAY (RYAN)
- 81 3,000-FT² SOLAR CELL ARRAY
- 82 3,000-FT² SOLAR CELL ARRAY— S-IVB WORKSHOP

* RECOMMENDED

10.4 SELECTION. Subsequent to the mid-term presentation, NASA selected the following three large space structures for more detailed analysis and design during the second half of the study.

Crossed H Interferometer. This concept (essentially the same as Concept 51b) was to be capable of accomplishing a wide variety of long-wave radio astronomy observations in the 0.5 to 10 MHz range, and is assumed to be launched by a manned Saturn V launch vehicle into a synchronous orbit. Results of the detailed preliminary design and analysis of this structure is contained in Volume III of this report.

Focusing X-Ray Telescope. The second selected concept was a focusing x-ray telescope very similar to Concept 73; however, much smaller in size. It was NASA's assessment that the appropriate size of such an instrument for flight in the mid-1970's should be in the neighborhood of 20-40 inches. Thus, the preliminary design and analysis of a "nominal 30-inch aperture" device was established as a guideline for the second half of the study. It was further directed by NASA that the x-ray telescope should be capable of accommodating a wide variety of soft x-ray observations throughout its useful lifetime. Detailed results of the subsequent study effort on the x-ray telescope are found in Volume IV.

100-Foot Parabolic Antenna. The third concept selected by NASA is a nominal 100-foot-diameter parabolic antenna similar to Concept 102. Although no operational mission is directly associated with this selected concept, the parabolic dish is considered a technology step leading to the development of the structures technology required to accomplish a broad spectrum of potential future missions such as point-to-point communications, TV broadcast, etc. Results of the preliminary design of this concept are contained in Volume V.

It was further directed that the preliminary design and analysis of these three concepts be pursued with the three primary flight objectives in mind: namely, the first objective having to do with the evaluation of man's role in the assembly, alignment, checkout, etc., of a large structure in space; the second directed toward structures technology; and the third as appropriate to suit the particular applicable mission.

# APPENDIX A

## RELIABILITY ANALYSIS

A.1 CONCEPT ANALYSIS COMMON ASSUMPTIONS. The concept reliability and the gross crew functions each have a commonality of philosophy that is applicable for all concepts. These philosophies or assumptions are related here to avoid repetition within each concept evaluation.

A.2 RELIABLITY ANALYSIS. The assessment of the system reliability requirement is accomplished based on the reliability goal of 0.997 for Concept 5 (20 w/lb solar cell array), which was derived in the previous studies by the Boeing Company. This method of analysis is found to be most appropriate and effective in allocation of reliability goals to various subsystems based on the use of weighting factors. The factors that are considered are system complexity, mission time, operational profile (the environment stresses that are being imposed on the equipment at the time the function is being performed), tolerance requirements, and maintainability (astronaut's activity for deployment).

Each of the factors are evaluated for the proposed system concept and listed in a table. The factors are assigned a value of 1 to 10. In the instance of system complexity, each system is evaluated by considering the probable quantity and type of components making up the system and also judged by the assembled intricacy of these components. The most complex system is assigned a ten level, the least complex a one.

The level of environmental exposure and the operational time requirements are considered. Systems operating for maximum time are rated at ten, the minimum time at one.

Dimensional tolerances of RF reflecting or radiating surfaces and elements are considered. Systems with high tolerance requirements are rated at ten; the systems requiring minimum tolerance requirements are rated one.

Complexity, difficulty, and duration of total scheduled deployment EVA tasks were also considered. Systems with high maintainability requirements are rated ten, and those systems with the least maintainability requirements are rated one.

Based on reliability goal of 0.997 for Concept 5, the probability of deployment and mission success for the proposed concepts are evaluated and shown in another table. A third table shows the apportioned reliability and expected failures for the proposed concepts for fixed structure, movable mechanical joints, beam pointing, and deployment control.

Concept 82, solar cell array for SIVB workshop, was selected as an example to illustrate the reliability analysis approach used during the evaluation of all provided concepts. Table A-1 depicts the weighting factors for subsystems. Table A-2 shows the reliability allocations for subsystems for Concept 82. Finally, the reliability allocation for the selected example is shown in Table A-3.

The summary of the first 20 evaluated concepts with respect to reliability allocations is shown in Table A-4.

Table A-1. Determination of Weighting Factors for Subsystems — Concept 82

| SYSTEM | SYSTEM COMPLEXITY (10 = System Most Complex) | DEPLOYMENT TIME (10 = System Operating Most) | ENVIRONMENT EXPOSURE (10 = System Extremes) | TOLERANCE REQUIREMENTS (10 = System Requiring High Tolerance During Deployment | MAINTAINABILITY (Astronaut's Activity for Deployment, 10 = Most Complex) |
|---|---|---|---|---|---|
| Fixed Structure | 10 | 8 | 4 | 8 | 4 |
| Movable Mechanical Joints | 8 | 4 | 5 | 8 | 4 |
| Beam Pointing Including Spacecraft | 5 | 3 | 4 | 6 | 3 |
| Deployment Control and Computation | 5 | 2 | 3 | 5 | 2 |

Table A-2. Reliability Allocation for Subsystems — Concept 82

| SYSTEM | PRODUCT (Lowest Number Most Reliable) | RATIO = $\alpha_i$ | $R_1 = R^{\alpha_i}$ | EXPECTED FAILURES PER $10^5$ MISSION HR |
|---|---|---|---|---|
| Fixed Structure | 10240 | 0.61171 | $R_1 = 0.99600^{0.61171}$ <br> $= 0.99755$ | 245 |
| Movable Mechanical Joints | 5120 | 0.30585 | $R_1 = 0.99600^{0.30585}$ <br> $= 0.99878$ | 122 |
| Beam Pointing | 1080 | 0.06452 | $R_1 = 0.99600^{0.06452}$ <br> $= 0.99974$ | 26 |
| Deployment Control | 300 | 0.01792 | $R_1 = 0.99600^{0.01792}$ <br> $= 0.99993$ | 7 |

$$\sum_{i=1}^{n} \Pi_i = 16740$$

Table A-3. Reliability Allocation for Concept 82

| CONCEPT | SUM OF PRODUCTS $\sum_{i=1}^{n} \Pi_i$ | RATIO = $\beta_i$ $\left(\sum_{i=1}^{n} \Pi_i\right) \Big/ \left(\sum_{i=1}^{n} \Pi_i\right)_m = \beta_i$ | $R = R_G^{\beta_i}$ | EXPECTED FAILURES PER $10^5$ MISSION HR |
|---|---|---|---|---|
| 82 | 16740 | $\frac{16740}{12564} = 1.33238$ | $Ra = 0.997^{1.33238}$ $= 0.99600$ | 400 |

1 Yr

$$MTBF = \frac{8760}{100400} = 2,190,000$$

Table A-4. Summary of Concept Reliability Allocations

| CONCEPT NO. | CONCEPT IDENTITY | RELIABILITY ALLOCATION |
|---|---|---|
| 1 | 60-Ft Paraboloid Antenna (NASA/MSFC) | 0.99299 |
| 2a | Broadside Dipole Antenna Array (Brown Engr.) | 0.98706 |
| 2b | 300M Rhombic Antenna (Brown Engr.) | 0.99708 |
| 3 | 1500M Paraboloid Antenna (Astro Research) | 0.99893 |
| 4 | Lenticular Communications Satellite (Goodyear) | 0.99865 |
| 5 | Solar Cell Arrays (Boeing) | 0.99700 |
| 6 | 600M Rhombic Antenna (NASA/MSFC) | 0.99793 |
| 7 | Member Structure Antenna (Ryan) | 0.99601 |
| 8 | Large Solar Cell Array (Ryan) | 0.99681 |
| 9 | Large Compressible Truss Antenna (Ryan) | 0.99540 |
| 10 | Folding Petal Space Antenna (Ryan) | 0.99642 |
| 11 | Paraboloid Antenna (30-Ft Petal) (TRW/STL) | 0.99846 |
| 12 | "Vista" Direct Home Broadcast Antenna Solar Array (RCA) | 0.99553 |
| 13 | Orbiting Radio Astronomy Observatory (TRW/STL) | 0.99995 |
| 14 | Deep Space Meteoroid Satellite (Martin) | 0.99910 |
| 15 | Conic Scissors Parabola Antenna (Fairchild) | 0.99568 |
| 16 | Bidirectional Scissors Antenna Array (Fairchild) | 0.99338 |
| 18 | Concertina Micrometeoroid Panel (Fairchild) | 0.99818 |
| 19 | Solar Array, Rollout or Scissors (Fairchild) | 0.99489 |
| 20 | Extensible Rhombic Antenna (Fairchild) | 0.99749 |

# APPENDIX B

## LAUNCH AND ORBITAL ENVIRONMENT CONSTRAINTS

This section provides specific and general information for design and analysis of the various experiment concepts and missions. The areas covered are:

a. Structural Environment.

b. Vibration.

c. Docking Maneuvers.

d. Thermal Environment and Control.

e. Meteoroid Environment.

f. Radiation.

The crew systems constraints (Appendix D) are related to the environment constraints and were considered jointly in the study.

### B.1 STRUCTURAL ENVIRONMENT

B.1.1 Saturn V Launch and Boost Loadings. Studies by Grumman Aircraft Engineering Corporation under NASA Contract NAS9-4983-Addendum, December 1965 summarize the structural environment for an integrated LEM laboratory weighing 32,000 lb at launch. The data generated are reproduced here.

| LIMIT ACCELERATIONS | | X (g) | Y (g) | Z (g) |
|---|---|---|---|---|
| Liftoff | | +1.60 | ± 0.65 | ± 0.65 |
| Max | (S-1C) | +2.07 | ± 0.30 | ± 0.30 |
| Boost | (S-1C) | +4.90 | ± 0.10 | ± 0.10 |
| Cut-off | (S-1C) | -1.70 | ± 0.10 | ± 0.10 |
| Engine Hardover | | +2.15 | ± 0.40 | —— |
| S-II | | | | |
| Earth Orbit | | 0 | 0 | 0 |

Note: Angular Acc. = 0 rad/sec$^2$

Typical loadings sustained by the experiments during translunar space flight with the propulsion system operating are as follows:

| ACCELERATIONS | X (g) | X ($rad/sec^2$) | Y (g) | Y ($rad/sec^2$) | Z (g) | Z ($rad/sec^2$) |
|---|---|---|---|---|---|---|
| SM Prop. Syst. Operating | -0.36 | — | ±0.062 | ± 1.99 | ± 0.062 | ± 1.99 |
| Midcourse Correction | -0.36 | 0 | ±0.087 | ± 2.82 | ± 0.087 | ± 2.82 |

B.1.2 Vibration. The vibration environment envelope can be represented by a superposition of random and sinusoidal envelopes given in the following:

Inputs to equipment supports from exterior primary structure:

| RANDOM (Hz) | |
|---|---|
| 10 to 23 | 12 db/octave rise to |
| 23 to 80 | $0.0148g^2$/Hz |
| 80 to 105 | 12 db/octave rise to |
| 105 to 950 | $0.0444g^2$/Hz |
| 950 to 1250 | 12 db/octave decrease to |
| 1250 to 2000 | $0.0148g^2$/Hz |
| SINUSOIDAL (Hz) | |
| 5 to 18.5 | 0.154 in. D.A. |
| 18.5 to 100 | $0.0148g^2$/Hz |

Inputs to equipment supports from interior primary structures:

| RANDOM (Hz) | |
|---|---|
| 10 to 23 | 12 db/octave rise to |
| 23 to 80 | $0.0148g^2$/Hz |
| 80 to 100 | 12 db/octave rise to |
| 100 to 1000 | $0.0355g^2$/Hz |
| 1000 to 1200 | 12 db/octave decrease to |
| 1200 to 2000 | $0.0148g^2$/Hz |

| SINUSOIDAL (Hz) | |
|---|---|
| 5 to 16 | 0.154 in. D.A. |
| 16 to 100 | 1.92 peak |

For design purposes, the environment can be adequately represented by the above random spectrum applied for 5 min along each of the three mutually perpendicular axes (X, Y, and Z) in addition to the corresponding sinusoidal spectrum acting for 5 sec at the natural frequency of the equipment being designed.

B.1.3 Acoustics. Sound pressure levels external to LEM in decibels, and referenced to 0.0002 dynes/cm$^2$, are presented below:

| OCTAVE BAND (Hz) | C-5 AT MAX LEVEL (db) |
|---|---|
| 9 to 18.8 | 136 |
| 18.8 to 37.5 | 142 |
| 37.5 to 75 | 146 |
| 75 to 150 | 143 |
| 150 to 300 | 139 |
| 300 to 600 | 135 |
| 600 to 1200 | 130 |
| 1200 to 2400 | 125 |
| 2400 to 4800 | 119 |
| 4800 to 9600 | 113 |
| Overall | 150 |

B.1.4 LEM Laboratory and CSM Docking Data. The following criteria describe the docking maneuver between the LEM Lab and CSM.

| PROBE CONTACT CRITERIA | |
|---|---|
| Axial Velocity | 0.1 to 1.0 fps |
| Radial Velocity | 0.0 to ± 0.5 fps |
| Angular Misalignment | ± 10° |
| Angular Velocity | ± 1.0 deg/sec |
| Miss distance at probe + 1 ft | |

LIMIT LOADS AT PROBE CONTACT

$F_x$ = 2235 lb

$F_Z$ = 3118 lb

M = 8134 ft/lb

MASS PROPERTIES

| | | | |
|---|---|---|---|
| Active Vehicle | M | 2,000 slugs | C/M + S/M |
| | I | 74,000 slug-ft$^2$ | |
| Target Vehicle | M | 2,200 slugs | S-IVB + LEM |
| | I | 1,150,000 slug-ft$^2$ | |

At final hard surface contact, the axial velocity will be 0.15 fps and the load ($F_x$) will be 5000 lb.

B.2 THERMAL ENVIRONMENT. Experiments that are located external to the vehicle are exposed to various natural and man-made environments, depending upon their location on the vehicle and the mission on which they are used. Thermal radiant interchanges can be present with the sun, the earth (or moon), the vehicle itself, and deep space. Solar heating (440 Btu/hr/ft$^2$) can impinge directly upon the experiment, be reflected off the earth or moon (i.e., albedo), or be reflected off the vehicle thermal control skins. The earth will radiate to the experiment with an effective black body temperature of 450°R. Deep space will appear to have an energy level equivalent to a black body at 50°R. If insulation and thermal control mechanisms are not added to an experiment which is exposed to these environments, it would tend to seek a temperature in the range of -250°F to 250°F for an $\epsilon = 1.0$.

The principal man-made environment that can be thermally damaging to experiments is the impingement of exhaust gases from the reaction control engines. Items that are located in the path of these gases can be heated to temperatures in excess of 2000°F during firings, while items that are even slightly impinged upon may reach temperatures of 200°F to 300°F for a few seconds. The exhaust products of these engines can also present problems. Heavy insulation and ablative coating are needed if an experiment must be located in these plumes.

Thermal Design Requirements. The thermal control of experiments for the LEM laboratory is accomplished by both active and passive control systems. Heat generated by experiments inside the vehicle is absorbed by radiation to the structure, convection to the cabin atmosphere, cold plates attached to the LAB ECS loops, or other special control means. Experiments external to the vehicle are exposed directly to the environments discussed above. These experiments are controlled by either the use of

thermal control coatings supplemented with heaters or by a combination of super-insulation and a cold plate attached to the LAB ECS. In order to be compatible with the LAB thermal design, these external experiments must not be dependent upon LAB orientation relative to the earth, sun, or moon for survival.

B.3 METEOROID ENVIRONMENT. The composition, characteristics, occurrence, and origin of interplanetary dust particles are of crucial importance in the planning of successful manned space missions because these parameters relate to the successful prediction of meteoroid shielding effectiveness. Furthermore, exposure of sample meteoroid shielding to the space environment is of comparable importance and goes hand in hand with the basic experiments.

Recent work (The Near Earth Environment, Robert J. Naumann, Research Projects Laboratory, MSFC, 12 October 1966 (Review Copy) indicates that the smaller dust particles believed capable of eroding spacecraft glass windows and reflector materials are not as numerous as previously thought. Tarnishing rather than erosion may be more accurate. The larger particles, on the contrary, are more numerous than extrapolated distribution curves indicate and are closer to past pessimistic estimates.

Since the latter particles are hazardous when encountered, but much less abundant, it will be necessary to increase the size of meteoroid detectors to the order of square meters. Otherwise, an encounter with a larger, more hazardous particle, may not be experienced until the first manned interplanetary journey.

Two possible fits to data obtained by numerous satellite experiments were derived by Naumann. Tabulations of pertinent measurements are presented in Table B-1 and Table B-2. Plots of the distributions are shown in Figure B-1.

B.4 CHARGED PARTICLE RADIATIONS. Radiation shielding criteria will be based on NASA General Working Paper No. 10,008, "A Further Amplification of Radiation Protection Criteria for the Exposure of Man to Space Radiations on Missions of Short Duration", William L. Gill and Stanley C. White, NASA, MSC, 1 August 1966.

B.4.1 Excerpt of Criteria. In the establishment of exposure criteria for short missions, the following conditions were applied:

a. The philosophy of radiation exposure for early space missions was one of balancing the risk of the deleterious effects of radiation against other mission risks.

b. In the consideration of this problem for a limited number of personnel, only the somatic effects of radiation were considered; and genetic damage to the population as a whole is neglected.

c. For personnel who demonstrate no clinical indications of radiation effects, previous radiation exposure history was ignored in the evaluation of the acute effects of radiation.

Table B-1. Micrometeoroid Data Measurements, Fit I

| Measurement | $\phi$ Obs. no./$m^2$ day | $\lambda$ | $F(\lambda)$ | N no./$m^2$ day sterad | log d | Est. Mass (gm) | Mass From Fit (gm) |
|---|---|---|---|---|---|---|---|
| .04 cm. Peg. | $10^{-2.16}$ | .773 | $10^{.230}$ | $10^{-2.39}$ | -2.230 | $10^{-5.91}$ | $10^{-6.03}$ |
| .02 cm. Peg. | $10^{-1.53}$ | .601 | $10^{.264}$ | $10^{-1.79}$ | -2.211 | $10^{-6.76}$ | $10^{-6.90}$ |
| .0050 cm. Exp. XXIII | $10^{-.516}$ | .292 | $10^{.360}$ | $10^{-.876}$ | -1.673 | $10^{-8.33}$ | $10^{-9.01}$ |
| .0025 cm. Exp. XXIII | $10^{-.278}$ | .215 | $10^{.392}$ | $10^{-.670}$ | -1.709 | $10^{-9.19}$ | $10^{-9.83}$ |
| .0050 cm. Exp. XVI | $10^{-.638}$ | .331 | $10^{.345}$ | $10^{-.983}$ | -2.017 | $10^{-8.33}$ | $10^{-8.67}$ |
| .0025 cm. Exp. XVI | $10^{-.352}$ | .239 | $10^{.382}$ | $10^{-.734}$ | -1.990 | $10^{-9.19}$ | $10^{-9.55}$ |
| Low Power Radar | $10^{-4.12}$ | 1.270 | $10^{1.092}$ | $10^{-5.21}$ | -3.209 | $10^{-4.18}$ | $10^{-3.32}$ |
| High Power Radar | $10^{-3.04}$ | 1.109 | $10^{.888}$ | $10^{-3.93}$ | -3.210 | $10^{-5.25}$ | $10^{-4.40}$ |
| -2 Photo. Mag. | $10^{-10.02}$ | .760 | $10^{.597}$ | $10^{-10.62}$ | -2.686 | $10^{1.05}$ | $10^{1.38}$ |
| 2.5 Photo. Mag. | $10^{-7.61}$ | 1.176 | $10^{.981}$ | $10^{-8.59}$ | -2.608 | $10^{-.948}$ | $10^{-.694}$ |

Table B-2. Micrometeoroid Data Measurements, Fit II

| Measurement | $\phi$ Obs. no./$m^2$ day | $\lambda$ | $F(\lambda)$ | N no./$m^2$ day sterad | log d | Est. Mass (gm) | Mass From Fit (gm) |
|---|---|---|---|---|---|---|---|
| .04 cm. Peg. | $10^{-2.16}$ | .833 | $10^{.221}$ | $10^{-2.38}$ | -8.116 | $10^{-5.91}$ | $10^{-6.24}$ |
| .02 cm. Peg. | $10^{-1.53}$ | .690 | $10^{.244}$ | $10^{-1.77}$ | -8.175 | $10^{-6.76}$ | $10^{-7.03}$ |
| .0025 cm. Exp. XXIII | $10^{-.516}$ | .273 | $10^{.368}$ | $10^{-.884}$ | -7.911 | $10^{-8.33}$ | $10^{-8.87}$ |
| .0050 cm. Exp. XXIII | $10^{-.278}$ | .125 | $10^{.433}$ | $10^{-.711}$ | -7.867 | $10^{-9.19}$ | $10^{-9.77}$ |
| .0025 cm. Exp. XVI | $10^{-.638}$ | .341 | $10^{.342}$ | $10^{-.980}$ | -8.224 | $10^{-8.33}$ | $10^{-8.55}$ |
| .0050 cm. Exp. XVI | $10^{-.352}$ | .173 | $10^{.410}$ | $10^{-.762}$ | -8.215 | $10^{-9.19}$ | $10^{-9.42}$ |
| Low Power Radar | $10^{-4.12}$ | 1.09 | $10^{.869}$ | $10^{-4.99}$ | -9.042 | $10^{-4.18}$ | $10^{-3.58}$ |
| High Power Radar | $10^{-3.04}$ | 1.02 | $10^{.785}$ | $10^{-3.83}$ | -9.014 | $10^{-5.25}$ | $10^{-4.68}$ |
| -2 Photo. Mag. | $10^{-10.02}$ | 1.25 | $10^{1.07}$ | $10^{-11.09}$ | -8.948 | $10^{1.85}$ | $10^{1.55}$ |
| 2.5 Photo. Mag. | $10^{-7.61}$ | 1.21 | $10^{1.02}$ | $10^{-8.63}$ | -8.950 | $10^{-.948}$ | $10^{-.44}$ |

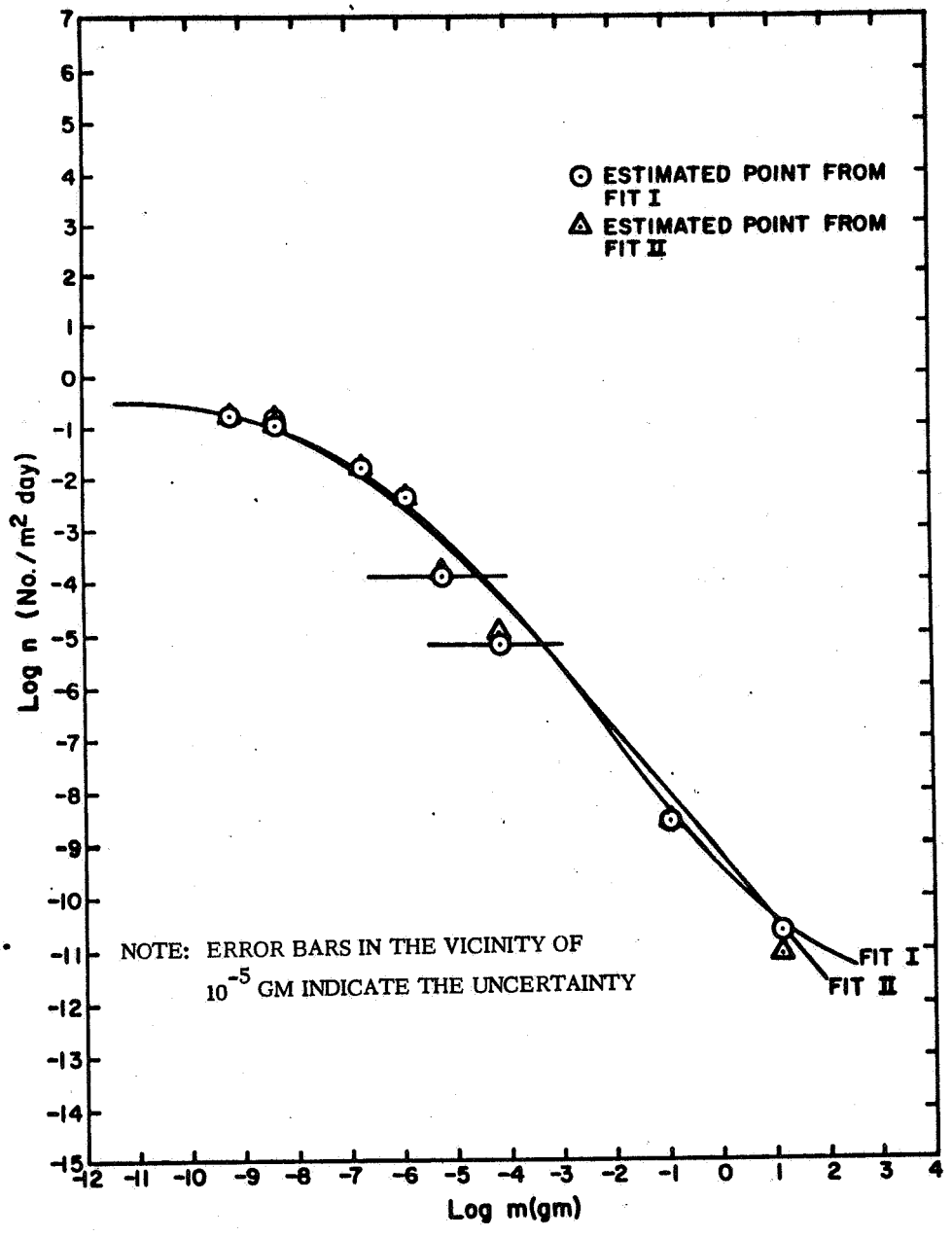



Figure B-1. Cumulative Mass Flux Distribution that Best Fits Estimated Characteristic Mass for Each Measurement

d. The mechanisms of chronic space radiation effects are little understood, and it does not appear that the effects of such variables as dose rate, periods between exposure, age at exposure, and total dose can be brought to a single, coherent, quantitative model in any organ or organ system in the near future. Therefore, a conservative approach to this phase of the problem has been applied. The exposure limits during a flying career, which was estimated to be no shorter than 5 years, were the same as the permissible limits over a 30-year period for an atomic industry radiation worker. A flying career was considered to start only after an appreciable dose of radiation had been received, as will be described in a subsequent section. The 30-year limit has not yet been shown to produce deleterious effects, although some minor manifestations of radiation changes may be noted near the end of life.

e. The radiation dose pattern in the body due to isotropically incident corpuscular radiation is very much higher on the surface than at the midline, and the arrangement of equipment in the vicinity of the body can cause considerable variation in this pattern. The body itself considerably attenuates the incident radiation; thus, the effect of radiation exposure is to deliver markedly different doses to different organs or different sites in the same organ systems. This type of exposure is very much different from that delivered by plane monodirectional gamma rays or neutrons, where the dose delivered to all portions of the exposed volume of a body seldom varies by more than a factor of 3. The measure of exposure dose — for example, the incident body dose measured in air — can be correlated with biological effect in the latter case, but is meaningless in the isotropic exposure. To evaluate the biological effect for space radiations, the dose at the site of interest must be measured or calculated. Hence, both the dose and the point in the body where it is measured must be specified in a shield design. The shield designer must describe the pattern of radiation in the case of actual exposure to provide the attending physician with insight as to the possible effects of exposure. With the incorporation of the above factors, Table B-3 was prepared for shield design criteria and was considered to cover all the organs that would influence shield design.

Table B-3. Radiation Exposure Dose Limits

| BODY LOCATION | MAXIMUM PERMISSIBLE INTEGRATED DOSE (rem) | RBE (rem/rad) | AVERAGE YEARLY DOSE (rad) | MAXIMUM PERMISSIBLE SINGLE ACUTE EMERGENCY EXPOSURE (rad) | LOCATION OF DOSE POINT |
|---|---|---|---|---|---|
| Skin of whole body | 1,600 | 1.4 (approx.) | 250 | 500[a] | 0.07mm depth from surface of Cylinder 2 at highest dose-rate point. |
| Blood-forming | 270 | 1.0 | 55 | 200 | 5cm depth from surface of Cylinder 2. |
| Feet, ankles, and hands | 4,000 | 1.4 | 550 | 700[b] | 0.07mm depth from surface of Cylinder 3 at highest dose point. |
| Eyes | 270 | 2[c] | 27 | 100 | 3mm depth from surface on Cylinder 1 along eye-line. |

(a) Based on skin erythema level.

(b) Based on skin erythema level, but these appendages are believed to be less radio-sensitive.

(c) Slightly higher RBE assumed since eyes are believed more radio-sensitive.

# APPENDIX C

## LAUNCH VEHICLE, SPACECRAFT, AND MISSION CONSTRAINTS AND CAPABILITIES

C.1 CONFIGURATIONS. The various launch vehicle envelopes considered in the LSSE study are outlined in Figures C-1 through C-5. In general the particular configuration will depend upon the payload requirements of the selected experiments, for either low altitude or synchronous orbit. Table C-1 shows the weight limitations for each launch vehicle configuration.

Table C-2 indicates some of the specific constraints for low altitudes and synchronous orbit in the 1968-1972 time frame. (Appendix B covers the overall environment constraint problems.)

The selected experiments will make maximum utilization of the spent S-IVB Stage as an orbital laboratory and workshop. The LEM is not being considered.

Each experiment will introduce its own interface requirements. In general these will be:

a. Support structure for the experiment to the S-IVB stage, CMS, or SLA.

b. Special adaptor provisions.

c. Power systems.

d. Environment control.

e. Controls and instrumentation.

C.2 CSM WEIGHT ASSUMPTIONS

a. $O_2$ leakage = 4.78 lb/day (@ 5 psi).

b. Electrical power @ 1.6 kw (housekeeping only) — fuel cells.

c. One gas system @ 5 psi.

d. Life support and crew systems based on 3 men.

e. Repressurize 3 times for 12 to 14 days.
6 times for 28 days.
10 times for 45 to 90 days.

f. Two portable life support systems recharged for each repressurization.

g. Reaction control system full capacity (840 lb) for 12- through 45-day missions; 90 days = double capacity.

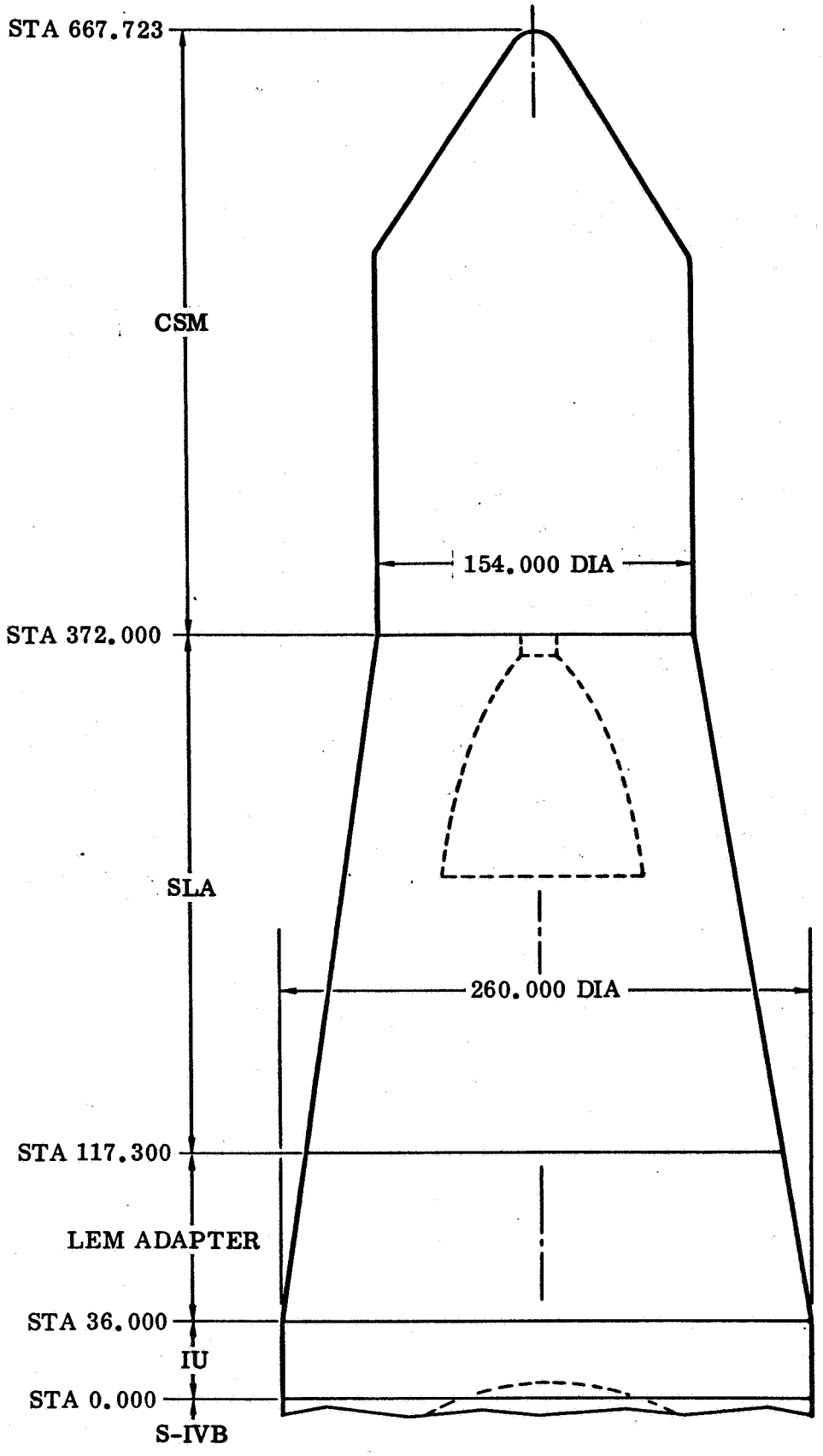



Figure C-1. Configuration A, Saturn IB/V Manned Launch

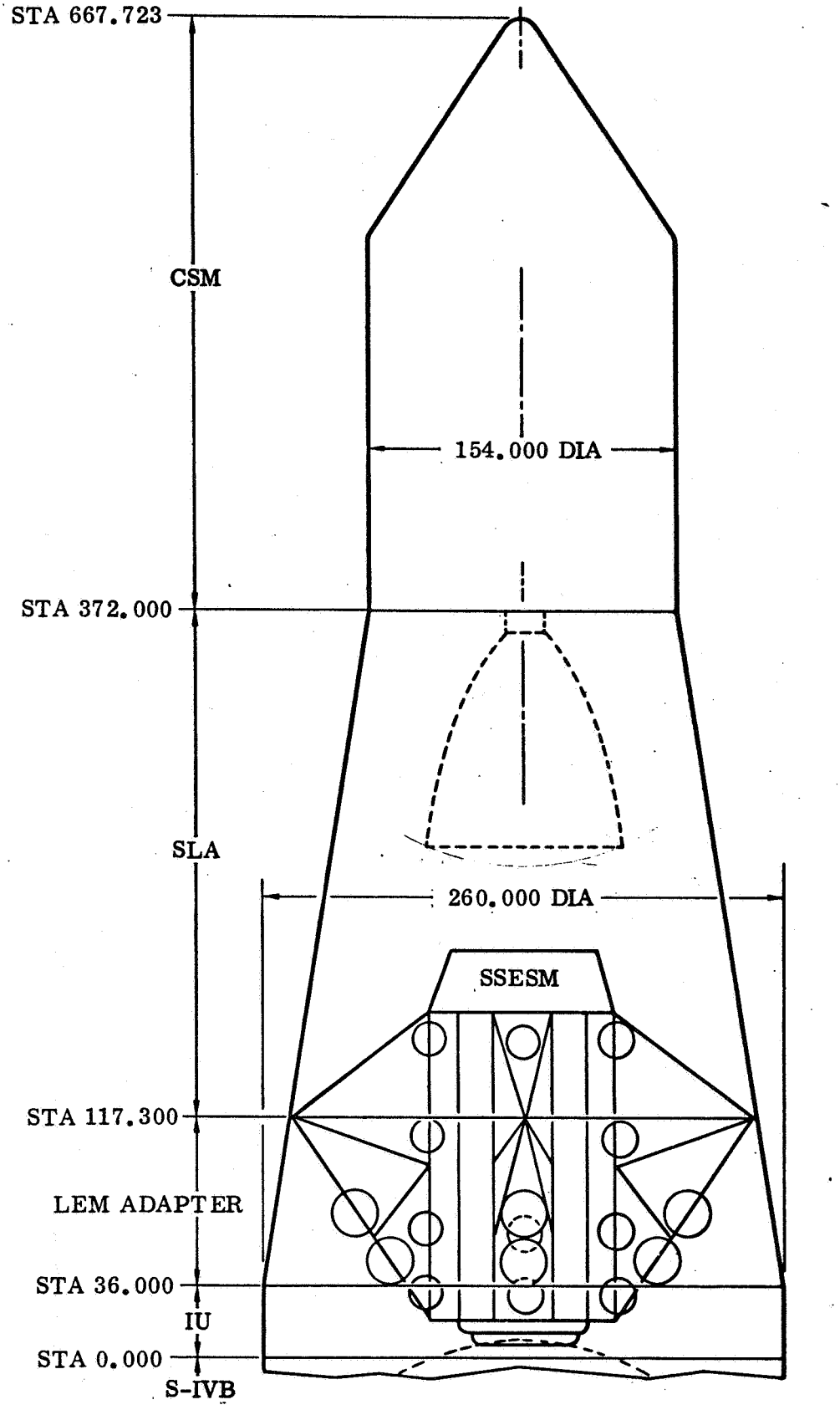



Figure C-2. Configuration B, Saturn IB/V Manned Launch Workshop

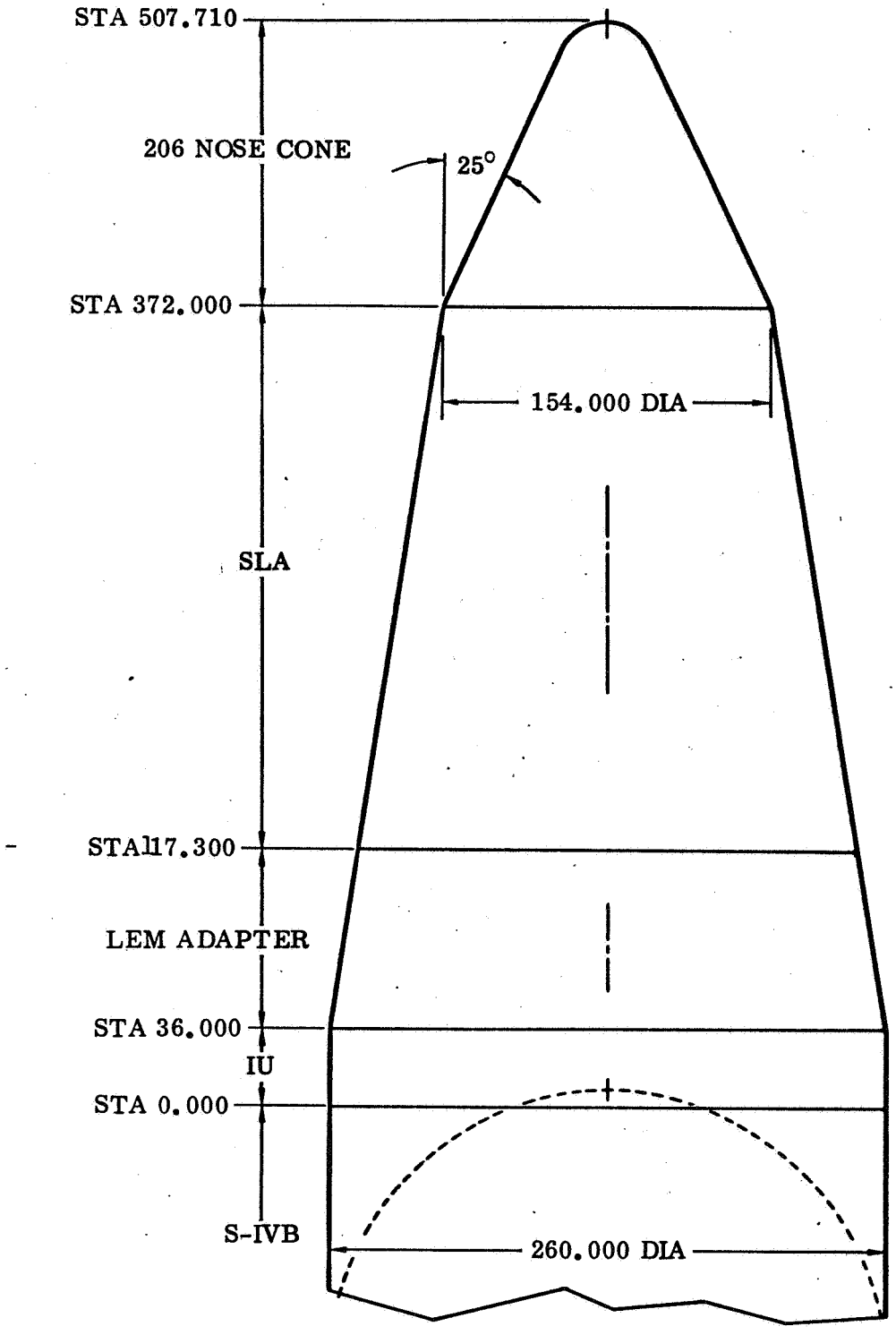



Figure C-3. Configuration $C_1$, Saturn IB Unmanned Launch

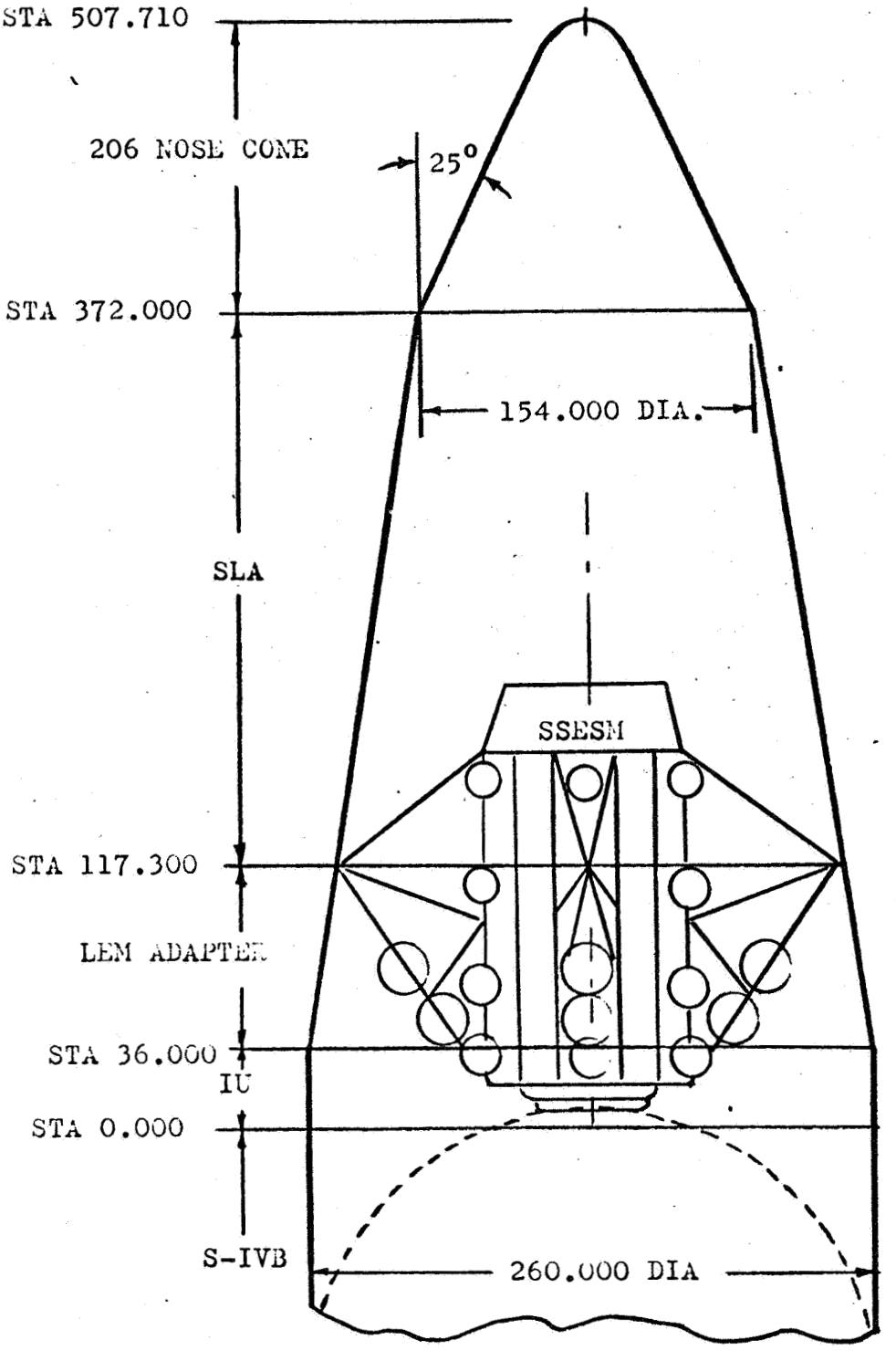



Figure C-4. Configuration $C_2$, Saturn IB Unmanned Launch Workshop

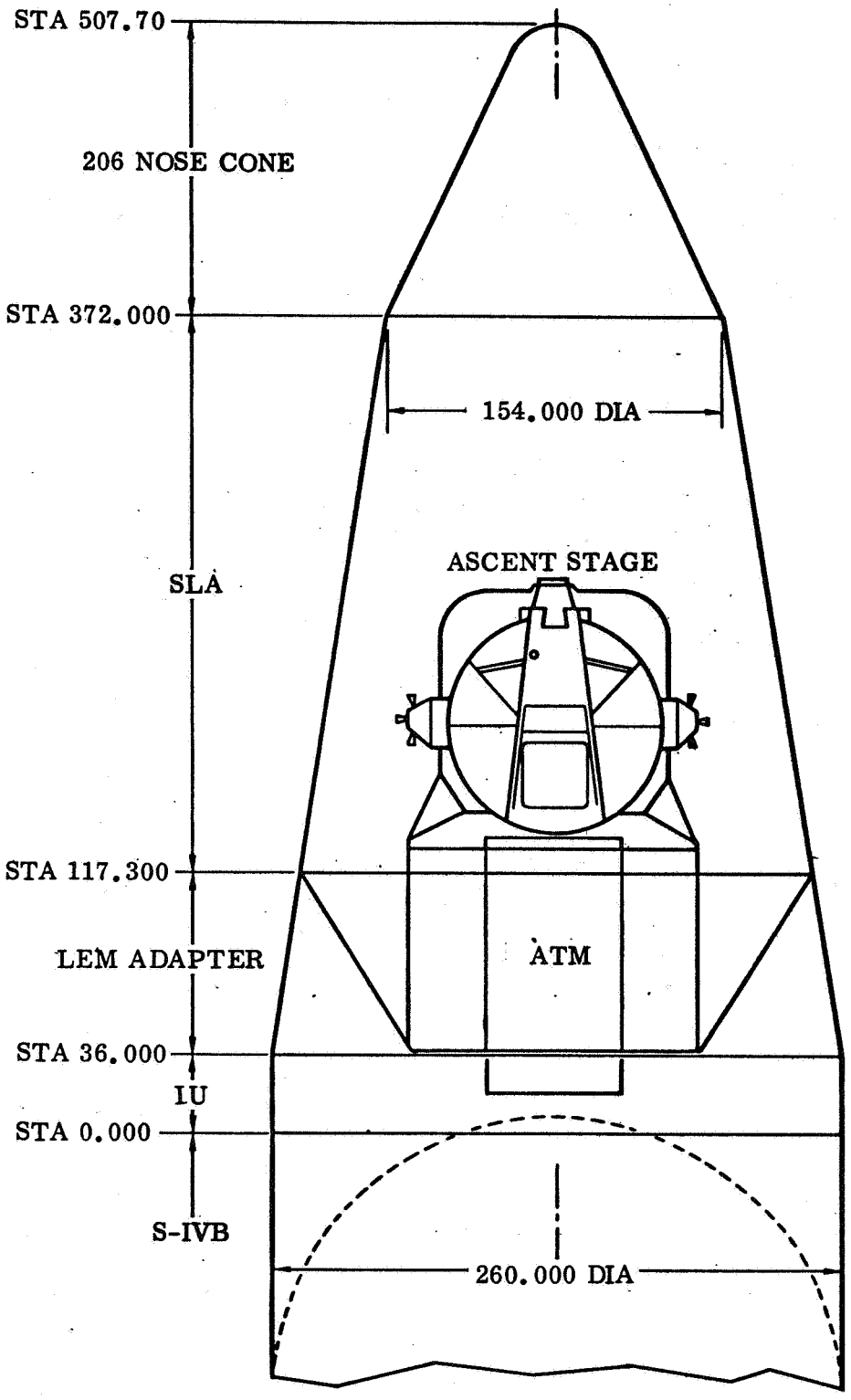



Figure C-5. Configuration D, Unmanned ATM Launch

Table C-1. Launch Vehicle Available Payloads and Weight

| AAP FLIGHT | TIME (days) | ALTITUDE (n.mi.) | INCLINATION | MAXIMUM PAYLOAD | REQUIRED PAYLOAD | AVAILABLE PAYLOAD (DIRECT) | AVAILABLE PAYLOAD (HOHMANN TRANSFER) | LAUNCH CONFIGURATION |
|---|---|---|---|---|---|---|---|---|
| (215) | 12 | 200 | 50° | 29,800 | CSM,SLA = 24,632 | 5,168 | 10,068 | A |
| (215) | 14 | 200 | 50° | 29,800 | CSM,SLA = 24,800 | 5,000 | 9,900 | A |
| (215) | 28 | 200 | 50° | 29,800 | CSM,SLA = 26,025 | 3,775 | 8,675 | A |
| (218/220) | 45 | 200 | 50° | 29,800 | CSM,SLA = 27,624 | 2,176 | 7,076 | A |
| (221/222 223/224) | 90 | 200 | 50° | 29,800 | CSM,SLA = 31,828 | — | 2,872 | A |
| (Not in AAP) | 14 | Sync. | 0° | 57,020 | CSM,SLA = 43,608 | — | 13,412 | A |
| | 28 | Sync. | 0° | 57,020 | CSM,SLA = 45,535 | — | 11,485 | A |
| (510*) | 14 | Sync. | 28.5° | 67,600 | CSM,SLA = 43,608 | — | 23,992 | A |
| (515*) | 28 | Sync. | 28.5° | 67,600 | CSM,SLA = 45,535 | — | 22,065 | A |
| (510) | 14 | Sync. | 28.5° | 67,600 | CSM,SLA,A/L=51,608 | — | 15,992 | B |
| (515) | 28 | Sync. | 28.5° | 67,600 | CSM,SLA,A/L=53,535 | — | 14,065 | B |
| (Not in AAP) | Unmanned Launch | 200 | 50° | 29,800 | SLA,Nose Cone=6,600 | 23,200 | 28,100 | $C_1$ |
| (Not in AAP) | Unmanned Launch | 200 | 50° | 30,100 | SLA,AL, Nose Cone = 15,500 | 14,600 | 19,500 | $C_2$ |
| (216/219) | 14 | 200 | 50° | 29,800 | CSM,SLA,A/L=32,800 | — | 1,900 | B |
| (214/217) | 45 | 200 | 50° | 30,100 | LEM A/S, SLA, Nose Cone = 12,600 | 17,500 | 22,400 | D |

Notes:
1. A/L = Air Lock Slice of S-IVB Workshop: WT = 8,000 lb.
2. SLA = Folding pedals of LEM adapter: WT = 3,600 lb.
3. CSM weights to be subtracted vary, depending on duration and SM propellant requirements for deorbit.
4. For general information: S-IVB dry weight = 24,500 lb., IU weight = 4,150 lb.
5. For 260 n.mi. orbit at 50° inclination decrease 200 NM payloads by 18%.

*For missions like 510 and 515 without workshop.

Table C-2. Environment Constraints

| EXPERIMENT CONSTRAINTS | LAUNCH AND BOOST | ORBITAL | |
|---|---|---|---|
| | | LOW ALTITUDE | SYNCHRONOUS |
| SOLAR FLARES | Occurrence of a major solar flare could delay a launching. | None constraining except for EVA over earth's polar caps. | A major solar flare could be lethal. Radiation shielding will be required. See solar flare discussion in Appendix B. |
| METEOROIDS | None constraining. | Spacecraft components that must be designed for meteoroid protection include optical surfaces, thermal coatings, liquid filled containers, expandable structures and vehicle hulls. See meteoroid discussion in Appendix B. | |
| COMETOIDS | None constraining. | A hazard may exist around 1 January 1969 in the form of ice chunks up to 200 ft in size. See cometoid discussion in Appendix B. | Chances of encounter are only 4% of that for low altitude orbits. |
| TRAPPED PARTICLES | None constraining. | None constraining except possible artificial trapped elections. See enclosure on trapped particles. | None constraining hazards exist at lower altitudes. See enclosed chart. |
| TEMPERATURE (° F) | Internal laboratory temperature: 0° F during ascent. See enclosure. | Experiments mounted outside the spacecraft, if not insulated, will range from +250° F to -250° F, for $\alpha/\epsilon \leq 1.0$. Ascent stage structure will vary in temperature from 0° F to 130° F and can be used as a heat sink of 5 to 6 w/ft$^2$ of radiating area. By removing vehicle thermal insulation and replacing skin by Alzak, or a low ($\alpha/\epsilon$) material, heat may be radiated directly to space (15 w/ft$^2$) for 200 n. mi. orbit. | |
| ACCELERATION (g) | SATURN V: Liftoff X +1.6, Y ±0.65, Z ±0.65; Maximum +2.1, ±0.30, ±0.30; Boost +4.9, ±0.10, ±0.10; Cut-off -1.7, ±0.10, ±0.10; Engine hard over +2.2, ±0.40, — | | |
| ACOUSTICS | Typical loadings and sound pressure levels are given in Appendix B. | | |
| VIBRATION | Environment can be represented by the random spectrum, acting for 5 min along each axis, given in Appendix B. Sinusoidal component must also be added. | | |
| CABIN ATMOSPHERE | 5.8 psi oxygen atmosphere maintained between 50° F and 95° F. Gas is circulated by a fan with resulting convective heat transfer coefficient of ~0.5 Btu/hr/ft$^2$/° F. Cabin humidity will vary from 0% to 100%. Small heaters are | | |

h. Total water requirements supplied by fuel cells except for 60 lb emergency reserve.

i. 200 lb spares allowance for missions through 28 days, 300 lb through 45 days, and 450 lb through 90 days.

j. $CO_2$ removal by LiOH through 45 days; 90 day mission by Hamilton Standard regenerative system.

k. Radiation shielding for the synchronous orbit missions was based upon a maximum dose of 55 rads for missions of 30 or fewer days. The exact shape and distribution of the 4000 lb has not been determined.

C.3 POWER CAPABILITIES OF LAUNCH VEHICLE AND SPACECRAFT. Electrical energy for experiment activation and checkout can probably be best supplied from one of the associated mission modules such as the Apollo command and service module. This system has a capability of providing 634 kw-hr in the Block II configuration at a power level of 2720 watts with an overall life of 14 days (336 hr). At an average housekeeping load of 1.67 kw a total of 560 kw-hr is consumed during the 14 day mission leaving 74 kw-hr for contingencies or possible for experiment power. A shorter mission (less than 14 days) for erection and activation could make a proportionately greater quantity of CSM energy available for experiments.

Modification of the Block II CSM to AAP configuration would make significantly greater quantities of energy available at power levels up to 4 kw. By modification to extend fuel cell life to 1200 hr and providing added cryogenic storage for reactants, the total energy may be extended to provide approximately 2400 hr for mission durations up to 45 days.

Another source of energy in the form of LEM-type primary batteries may be provided at a much increased weight penalty of approximately 80 w-hr/lb. These batteries represent existing Apollo state-of-the-art and may be readily incorporated as add-ons in missions that are not weight limited. Battery power is available to a limited extent (1200 amp-hr) from the S-IVB instrumentation unit for a limited time period or from a possible LEM lab configuration in 300 or 400 amp-hr at 131 lb and 140.5 lb increments respectively. Existing LEM configuration incorporates approximately 2200 amp-hr (62 kw-hr) of primary batteries.

Extended mission power requirements for experiment operational period over 45 days should consider use of other power sources such as solar cells or isotope generating systems.

C.4 DATA RECORDING AND COMMUNICATIONS. The requirements in these areas for instrumentation and equipment will depend upon the specific experiments selected in Tasks 3 and 4. In general, no severe constraints are anticipated for the concepts being reviewed at this state of the study.

## C.5 STRUCTURAL ENVIRONMENT

C.5.1 Factor of Safety. The following factors of safety shall be applied to the limit loads given in this section.

a. The general factor of safety of 1.25 shall be used. If the magnitude of the load is known to close tolerance and if the simplicity of the analysis warrants the assumption, lower factors of safety may be used after approval by the contracting agency.

b. A factor of safety of 1.5 shall be used where safety of personnel is involved.

c. A factor of safety of 2.0 shall be used for design of pressure vessels.

Loads on Packaged Experiment. These loads are due to the following effects:

a. Transportation and handling — Table C-3 gives limit load factors which are representative for transportation and handling operations.

b. Launch and boost — Limit load factors obtained during launch, boost and translunar space flight are given in Appendix B, Section B.1.

c. Docking — Loads associated with the docking maneuver between the lab and CSM are given in Appendix B, Section B.1.4.

Table C-3. Transportation and Handling Operations

| | | MAXIMUM VERTICAL CONDITION | MAXIMUM LATERAL CONDITION | MINIMUM LONGITUDINAL CONDITION |
|---|---|---|---|---|
| Aircraft and Highway Transportation | $N_{Vert}$ | +2.5 (Down) -0.5 (Up) | +1.0 (Down | +1.0 (Down |
| | $N_{Lat}$ | 0.0 | ±1.0 | 0.0 |
| | $N_{Long}$ | 0.0 | 0.0 | ±1.5 |
| Jacking Condition | $N_{Vert}$ | +2.0 (Down) | +1.0 (Down) | +1.0 (Down) |
| | $N_{Lat}$ | 0.0 | ±0.5 | 0.0 |
| | $N_{Long}$ | 0.0 | 0.0 | ±0.5 |
| Hoisting and Positioning | $N_{Vert}$ | +2.0 (Down) | +1.0 (Down) | — |
| | $N_{Lat}$ | 0.0 | ±0.5 | — |

Loads During Deployment. Since the methods of deploying the self-erectable experiments vary, loads associated with deployment will be develped in the stress analysis section. Some of the loads which may be needed are:

a. Inflation pressures, including impact loads on membranes during unfolding.

b. Thrust from erecting motors.

c. Inertia and snubbing loads on spring-erected structures.

d. Loads required to produce strain-stabilized members.

Loads on Deployed Experiment

a. Loads associated with attitude control and station keeping are given in Section C.5.2.

b. When the mission requires docking of the CSM to the experiment, the docking loads are determined in Appendix B, Section B.1.4. Loads due to inadvertent impact of the astronaut with more fragile portions of the experiment will be determined assuming a closure rate of 1 fps.

c. Miscellaneous loads and effects which should be considered are:

   1. Thermal stresses and distortions — These effects may be calculated from the thermal environment specified in Appendix B, Section B.2.

   2. Gravity gradient forces and torques — These effects are given in Section C.5.2.

   3. Solar pressure — Solar pressure varies with surface reflectivity and may be taken as $10^{-7}$ lb/ft$^2$ (reflectivity 1.0).

   4. Meteoroid impact — Effects of meteoroid impact are given in Appendix B, Section B.3.

   5. Loads associated with disposal of the experiment after mission completion will not be considered. The requirement for post-mission disposal will be a subject of negotiation with the contracting agency.

   6. Vibration and acoustic environment are given in Appendix B, Sections B.1.2 and B.1.3.

   7. Air drag at low altitude.

C.5.2 Attitude Control Constraints. The attitude control constraints of the LEM and the CSM are mission oriented. The attitude control does not constrain the launch vehicle, except for weight contributions. The attitude control constraints considered are due to the engines, the sensors, and the autopilot software.

Pertinent engine data are shown in Table C-4. The structure for the particular mission must be able to withstand the thrusts and torques. The resulting vibrations in the structure must not cause motions which exceed the mission requirements. Minimum attitude tolerance is dependent on the minimum engine bit. The amount of propellant which must be carried depends on the specific impulse.

Table C-4. LEM/LSM Attitude Control Engine Data

| | LEM | COMMAND MODULE | SERVICE MODULE |
|---|---|---|---|
| Thrust, lb | 100 | 93 | 100 |
| Torque, one pair, ft-lb | 870 | | 1200 |
| Minimum Impulse lb | 0.5 | 2.0 | 0.5 |
| Specific Impulse | 150-280 | 150-266 | 150-280 |
| Operating Life | 500 | 70-160 | 500 |
| Propellant capacity | | 838 | |

Additional propellant tanks may be required by some missions. The engine operating life and number of engine firing cycles must also be considered.

Pertinent sensor data are shown in Table C-5. The sensors are not adequate for mission requirements of the order of tenths of degrees, or less.

Therefore the missions with tighter tolerances will require additional more accurate sensors. For some experiments, such as radio astronomy, it may be possible to use the experiment itself as the sensor.

Table C-5. Sensor Data

| | LEM | SERVICE MODULE |
|---|---|---|
| Attitude Gyro | ± 0.25° | Threshold ± 0.1° |
| Rate Gyro | 5% | 0.1 deg/sec |
| | | Threshold 0.02 deg/sec |
| | | Resolution 0.06 deg/sec |

The autopilot software can be used for a variety of missions by changing gains, dead zone and hysterises values. A single degree of freedom analog simulation was conducted under Convair ERR in support of the large space erectable antenna study. The

control system operated in a satisfactory manner with the simple software changes indicated above. Generalizations are dangerous, but it appears likely that for those missions in which the engines are satisfactory, the autopilot software will need only these simple changes.

# APPENDIX D

## CREW SYSTEMS CONSTRAINTS

This appendix is from the NASA/MSC Summary Description of Baseline Extravehicular Astronaut for 1968-1972, Report CSD-S-012, dated 21 February 1967. It is the intention of MSC to update the document periodically.

Purpose. The purpose of the document is to provide a baseline description of an extravehicular astronaut's performance capabilities as a function of the type of equipment available for extravehicular operations. The report was prepared from information and experience compiled by personnel of the Crew Systems Division of the Manned Spacecraft Center. It is anticipated that the information will be of use to cognizant personnel from NASA, DOD, and industry who are involved in planning and defining future extravehicular mission requirements.

Scope. The document provides a description of an extravehicular astronaut and his equipment for the time period of 1968 to 1972 or mainline Apollo and early Apollo Applications Program extravehicular activity missions. The baseline astronaut's capability is based on utilization of equipment that is presently available or now under development for use during the 1968-1972 time period.

BASELINE ASTRONAUT SUMMARY FOR EXTRAVEHICULAR ACTIVITY, 1968-1972

| CONSTRAINTS | REQUIRED DATA | TENTATIVE ASSUMPTIONS | REMARKS |
|---|---|---|---|
| 1. OPERATIONAL CONSTRAINTS | | | |
| 1.1 Available experimental time | Crew time available for experimental work | 12 hr/man/day | Remaining 12 hr required for daily activities such as sleeping, eating, personal hygiene, station monitoring, etc. Gemini results indicate that only 75% of experiment time was used in handling spacecraft problems. Therefore, an open day should be programmed to account for this contingency time. |
| 1.2 Available EVA time | Crew time available for EVA | 4 hr/man/day, plus 4 hr/man/day preparation and post-EVA activities | Limited by EVA support equipment, astronaut, capabilities and medical considerations. |
| 1.3 Pre-EVA operations | Time required to prepare EVA equipment | 2 hr/man/day | Includes unstowing, donning, adjusting, and checkout of suit and PLSS or other life support system utilized, and biomedical instrumentation as required. |
| | Time required to depressurize cabin | 4 min (CM) | The required time will be proportionately increased or decreased as a function of spacecraft volume and depressurization valve orifice sizing. |
| | Time required to depressurize Airlock module EV lock | 45 sec minimum | Assumes full opening of depressurization valve with no attempt to throttle flow. |
| | Time required to open hatch and exit CM | 30 min | |
| | Time required to open Airlock module EV hatch and exit | 2 min | Use Gemini spacecraft hatch. |
| 1.4 Post-EV Operations | Time required to enter spacecraft and close hatch: | | Hatch open assumes that cabin hatch remains open during EV activity. Closed hatch assumes spacecraft must be depressurized by EVA astronaut. |
| | Hatch open | 30 min (CM) | |
| | Hatch closed | 60 min (CM) | |
| | Time required to enter Airlock module and close hatch: | | |
| | Hatch open | 5 min (EV lock) | |
| | Hatch closed | 15 min (EV lock) | |
| | Time required to repressurize CM | 3 min to 2.8 psia<br>5 min to 4.5 psia | |
| | Time required to repressurize Airlock module (EV lock) | 15 sec | By opening pressure equalization valves interconnecting Airlock module EV lock with either the S-IVB lab or MDA. |
| | | 15 min | Utilizing Airlock ECS umbilical connections. |
| | Time required for removing EVA equipment | 60 min/man | Includes suit doffing, suit drying, stowing gear, and recharge of PLSS if used, which includes $O_2$, $H_2O$, LiOH, and battery (60 min/man for 100% recharge, 5 min/man for 90% recharge, 50 min additional required for $O_2$ recharge to 100%). |
| 1.5 Task Performance time | Change in task performance time due to EV conditions | Same as sea-level established timelines | With whole task training and proper simulation, accurate mission time-lines can be established; i.e., use of neutral bouyancy training. However, contingency times for ancillary problems (dumping data to ground stations, etc.), should be included in mission time-lines. |
| 1.6 EV rest periods | Length and frequency of rest periods | 2 min after every min of moderate to moderately heavy activity and 2 min after each period of heavy activity | To reduce metabolic expenditures. Allowable heart rates must be established for each task and these heart rates utilized to verify adequacy of scheduled rest periods or to terminate a task if the level of astronaut effort warrants. During rest periods after each heavy exercise, there will be no work performed until acceptable heart rates are attained. |

| CONSTRAINTS | REQUIRED DATA | TENTATIVE ASSUMPTIONS | | REMARKS |
|---|---|---|---|---|
| 2. EQUIPMENT CONSTRAINTS | | | | |
| 2.1 Life Support Systems | | | | |
| 2.1.1 Space Suit | | ILC A-5L | RX-3 | |
| Force/Torque Capability | Maximum exertable push force | 25 lb | Unknown | Requires good body anchoring. Both hands free. |
| | Maximum exertable pull force | 25 lb | Unknown | Requires good body anchoring. Both hands free. |
| | Maximum exertable torquing force | 600 in.-lb | Unknown | Requires good body anchoring, and both hands free. |
| | Maximum exertable one-hand push force | 25 lb | Unknown | Assumes one-hand holding while other is pushing and good body positioning. |
| | Maximum exertable one-hand pull force | 25 lb | Unknown | Assumes one-hand holding while other is pulling and good body positioning. |
| Joint Mobility | Humerus abduction | 74.8 | 80.0 | Percent of nude joint mobility range retained at 3.7 psig, (average values are not available). ILC-A-5L information is based on utilization of Block II thermal micrometeoroid garment. Data available for A-5L suit taken from early testing performed with competition Block II A-5L suit. |
| | Humerus flexion | 71.9 | 87.0 | |
| | Humerus extension | 56.6 | 90.0 | |
| | Elbow extension | 100.0 | 100.0 | |
| | Elbow flexion | 89.5 | 96.0 | |
| | Hand/Wrist, Abduction | 100.0 | 97.0 | |
| | Hand/Wrist, Abduction | 100.0 | 100.0 | |
| | Hand/Wrist, dorsi-flexion | 90.4 | 87.0 | |
| | Hand/Wrist, Palmar-flexion | 93.3 | 100.0 | |
| | Femur flexion | 60.7 | 90.0 | |
| | Knee extension | 100.0 | 100.0 | |
| | Knee flexion | 93.8 | 90.0 | |
| | Angle/Foot dorsi-flexion | 88.1 | 92.5 | |
| | Angle/Foot Plantar-flexion | 94.4 | 95.8 | |
| Visibility | Operational visual field | 105°upward 95°down-ward | 130°upward 100°down-ward | -5L: Head is permitted to move within helmet. -3: Minimum requirements. With fixed bubble or dome helmet and head free to rotate, visibility can be increased. |
| | Peripherial vision | 120° side — left and right | | |
| | Optical characteris-itcs of sun visor | 16 to 20% transmit-tance in visible range | | Sun visor only. |
| | | 10% transmittance in visible range 0.39 to 0.75 $\mu$ | | Total transmittance of sun visor, impact, and pressure visors combined. |
| | | 0.5% transmittance in UV range 0.25 to 0.39 $\mu$ | | Total transmittance of sun visor, impact, and pressure visors combined. |
| | | 2.0% transmittance in IR range 0.75 to 2.5$\mu$ | | Total transmittance of sun visor, impact, and pressure visors combined. |
| | | ILC A-5L | RX-3 | |
| Suit Dimensions w/o TMG & PLSS | Helmet width | 24.6 cm | 30.4 cm | -5L: Pressurized to 3.7 psig on a 5-ft, 10-1/2-in., 168-lb subject. |
| | Helmet depth | 31.3 cm | 30.4 cm | |
| | Shoulder width | 50.7 cm | 58.0 cm | |
| | Chest depth | 34.6 cm | 34.5 cm | -3: Hard suit has no growth factor due to pressurization. |
| | Hip width | 51.4 cm | 40.7 cm | |
| | Hip depth | 33.7 cm | 33.85 cm | |
| | Overall height | 187 cm | 181.3 cm | |
| Suit Dimensions with TMG & PLSS | Elbow width | 76.2 cm | 70.0 cm | |
| | Chest to back of PLSS | 66.1 cm | 60.16 cm | |
| | Elbow width arms crossed | 64.8 cm | | |
| | Arm width both hands over shoulders | 77.5 cm | | |
| | Elbow to outer arm width one hand above shoulder | 75.0 cm | | |

| CONSTRAINTS | REQUIRED DATA | TENTATIVE ASSUMPTIONS | REMARKS |
|---|---|---|---|
| Suit Dimensions with TMG & PLSS, Continued | | ILC-A-5L RX-3 | |
| | Hand to PLSS | 84.0 cm | Hand in front of body at mouth level. |
| | Hand to back | 56.0 cm | Hand in front of body at mouth level. |
| Glove Function | Performance capability | To rotate a 0.375-diameter knob in the pressurized condition | Design specifications. Minimum clearance to operate knob is 3-in. circle with knob centered in circle. Whole hand grasp requirements should be avoided. A 1-in.-dia. bar can be held for about 5 min. |
| Waste Management | Urine collection capacity during EVA | Sufficient for 4-hr EVA | |
| | Feces collection capacity during EVA | Part of Block II suit assembly used for Lunar EVA, i.e., Block II diaper | Not recommended for normal mode of EVA. |
| 2.1.2 PLSS | Oxygen supply capacity | 3 to 4 hr | Dependent on EV metabolic load. For purposes of mission planning approximately 1 hr of the total PLSS operating time should be designated as contingency reserve and not be considered available for normal mission performance. In addition, PLSS operating time must include system start-up, shut-down, as well as spacecraft egress/ingress times, which will require a total of 30 to 60 min. Therefore, the remaining useful work time for an average mission of 1600 Btu/hr will be from 1 hr to 1-1/2 hr and for an average mission of 1200 Btu/hr is 2 hr to 2-1/2 hr. |
| | Heat removal | 1200 Btu/hr for 4 hr<br>1600 Btu/hr for 3 hr | Considering total heat rejection, $CO_2$ produced, and $O_2$ consumed, the PLSS is limited to a metabolic and environmental mission total of 6000 Btu. Advanced system development is considering the following metabolic heat rejection capabilities. 2000 Btu/hr mission average; 2500 Btu/hr sustained, suit loop environmental parameters remain in specification; 3500 to 4000 Btu/hr mission peaks. The mission peaks will be of very short duration and will not be imposed on the life support system instantaneously. Metabolic heat rejection is presently limited to rates of 2000 Btu/hr due to limitation of LCG and thermal response for conduction of metabolic heat from body core to body surface. By LCG optimization to increase body coverage, it is estimated that metabolic rates (MR's) could be increased to 2500 Btu/hr without resultant body heat storage. |
| | | 2400 Btu/hr for 5-10 min | The time limitation on the 2400 Btu/hr MR is based on utilization of LCG's of present design. Work at this level or greater would result in significant body heat storage. |
| | | 3500 Btu/hr peak loads | The PLSS has been tested at rates up to 3500 Btu/hr and all environmental parameters out of the PLSS have remained in spec. However, present ventilation flow levels provided by the PLSS (6 acfm) with the present Block II helmet design limits metabolic activity to approximately 2000 Btu/hr for oro-nasal $PCO_2$ levels not in excess of 7.6 mmHg. |
| | Oxygen | 4800 Btu total plus 200cc suit leak | |
| | LiOH Battery | 6700 Btu total (max)<br>5 hr or 240 watt-hr<br>300 watt-hr max | |
| | Oxygen | 0.92 hr | |
| | LiOH | 4.43 lb, cannister not rechargeable | |
| | Battery | 5.14 lb; not rechargeable | |
| | Water | 7.33 lb + approximately 0.6 lb at reduced performance | |
| | E.O.S | 3.07 lb, not rechargeable | |

BASELINE ASTRONAUT SUMMARY FOR EXTRAVEHICULAR ACTIVITY, 1968-1972, Contd

| CONSTRAINTS | REQUIRED DATA | TENTATIVE ASSUMPTIONS | REMARKS |
|---|---|---|---|
| 2.1.3 ELSS + AMU | Oxygen supply capacity | Peak flow: 50 min | Cooling and $CO_2$ control for MR's up to 2000 Btu/hr. |
| | | Normal flow: 90 min | Cooling and $CO_2$ control for MR's up to 1400 Btu/hr. |
| 2.1.4 Umbilical | Life support capacity | Limited by spacecraft supplies, capability of spacecraft heat sink to reject heat, and LCG ability to absorb body heat as indicated in remarks of 4.2.1.2 above | Umbilical will provide oxygen, $H_2O$, biomedical/communications, and tethers. |
| 2.2 Anchoring Systems | Available anchoring and restraining devices | Handholds:<br>Foot Stirrups<br>Dutch Shoes | Preferred type are those that allow the EV crewman to work freely with both hands; specifically rectangular configurations. |
| | | Restraint systems:<br>Variable flexible<br>Flexible<br>Rigid Tubular | Gemini XII indicated that the waist or upper torso is the desired attach point for a restraint system. |
| 2.3 Maneuvering Systems | Available maneuvering aids | Handholds<br>Handrails<br>Hand Gun (HHMU)<br>Telescoping "fishing pole" devices<br>AMU<br>CMG<br>Jet shoes | AMU, CMG, and jet shoes are planned for evaluation in S-IVB lab. |
| 2.3.1 Handrails | Available types | Continuous<br>Spaced | Translation along handrails requires both hands. Max rate is 2 ft/sec and nominal rate 1 ft/sec. |
| 2.3.2 Special Shoes | Available types | Velcro<br>Dutch shoes (GT-XII) | Velcro shoes are adequate for body stability only; not for maneuvering or translation. |
| 2.3.3 Hand Gun (HHMU) | Possible delta V | 6 fps continuous firing<br>nominal burst firing<br>2 ft/sec | Gemini IV and X propellant usage rate approximately 2 lb/hr. |
| | Propellant | $N_2$, $O_2$, Hydrazene | |
| 2.3.4 AMU | Operational Lifetime | 1 hr | Limited by oxygen and electrical power supplies. Propellant utilization dependent on total mission delta V or translation requirements; total delta V = 250 ft/sec. Effective translation capability is roughly halved when translation is performed with the AMU operated in the stabilization and translation modes combined. |
| | Resupply capability | None | Using hot gas propellant. |
| | | Possible from S/C | Using cold gas as propellant. |
| | Range | 1000-ft radius of action | Design specification — practical range is undetermined to date. Translation up to 100 ft would be tethered and distances greater would be subject to further investigation. |
| | Maneuverability | Manual translation control in the fore/aft and up/down directions | |
| | Stabilization Capability | Within deadband of ± 2.4 | Manual and automatic three-axis attitude and rotational control. |
| | Nominal thrust/ thrust chamber | 2.3 ± 0.2 lb | Two thrusters/direction of translation. |
| | Total impulse provided | 300-3500 lb-sec | 90% hydrogen peroxide fuel. |
| | Required time for checkout and donning | 25 min | |
| | Storage location | Installed on outside of S/C | |
| | Hazards presented by AMU use | Thrustor plume impingement on suit or other support equipment | Modified coverlayer for EVA suit and gloves plus nozzle extensions near shoulder area minimize hazard. |

BASELINE ASTRONAUT SUMMARY FOR EXTRAVEHICULAR ACTIVITY, 1969-1972, Contd

| CONSTRAINTS | REQUIRED DATA | TENTATIVE ASSUMPTIONS | REMARKS |
|---|---|---|---|
| 2.3.5 CMG | Operational life time | To be determined | Limited by oxygen and electrical power supplies. |
| | Resupply capability | Additional bottles | Replaceable oxygen bottles for cold gas propellant. |
| | Range | To be determined | |
| | Maneuverability | Manual translation control in the fore/aft, up/down, left/right directions | |
| | Stabilization capability | Completely automated in all axes. Zero deadband | Manual and automatic stabilization with rotation control in all 3 axes. Rotation derived from momentum exchange within the CMG with RCS override. |
| | Nominal thrust | To be determined | |
| | Total impulse provided | 500 lb-sec/bottle | Additional bottles can be added with gaseous oxygen at 7500 psia. |
| | Required time for check-out and donning | To be determined | |
| | Hazards presented by CMG use | None | |
| | Storage location | MDA | Can be stowed in CSM. |
| 2.3.6 Navigational Requirements | Maximum distance for use of "line-of-sight" navigation | 500 ft | Without range-range rate radar. |
| 2.4 Reactionless Tools | Basic tool function available | Torquing — 65 ft/lb in 2 sec | NASA prototype tool. No present mantenance tasks require this high a magnitude of torque. A new tool is being developed for AAP. Maximum torque 250 in-lb. |
| | | Hammering | |
| | | Drilling - 1/4 to 21/32 in. | Information on D-16, DOD reactionless power tool is available from Propulsion Laboratory of WPAFB, Dayton, Ohio, Capt. Dan Seiger. |
| | | Sawing — 4 ft/min in thin materials (0.016 in.) 4 in./min in 0.100 in. 7075-T6 | |
| | Handle reaction torque | 5.6 in.-oz | |
| | Power supply | 12 vdc, 5-lb Ag-Zn battery | Housed in tool box. |
| | Method for securing and transporting tools | Tool kit | A temporary storage for tools, parts, fasteners is provided. |
| | Requirements for reactionless tools | | If EV crewman is properly restrained, modified hand holds should suffice where only low torque is required. |
| 2.5 Illumination Systems | | | |
| 2.5.1 Intensity of Illumination | Illumination for gross positioning of structural members | 5 ft-candles | Human engineering standards. |
| | Illumination for adjustment, maintenance and repair of strucutral members | To be determined | |
| 2.5.2 Place of Illumination | Available location of lighting devices | Chest mounted<br>Wrist mounted<br>Spacecraft mounted or mounted on assembled structure | At least two lights are required for redundancy and shadow fill. The chest mounted location is preferred to minimize possible interference by tool or limb movement, or restricted access openings. Location of light should be accessible to astronaut in a pressurized suit. |
| 2.6 Communications System | | | |
| 2.6.1 PLSS | Available systems | Voice plus 7 biomed TM channels | Monitors backpack and biomedical parameters. |
| 2.6.2 Umbilical | Available systems | Hardline | Requirements will impact design of small sized umbilicals. |

BASELINE ASTRONAUT SUMMARY FOR EXTRAVEHICULAR ACTIVITY, 1968-1972, Contd

| CONSTRAINTS | REQUIRED DATA | TENTATIVE ASSUMPTIONS | REMARKS |
|---|---|---|---|
| 2.7 Power Supply Systems | | | |
| 2.7.1 Spacecraft | Available types | Umbilical supply | Limited by spacecraft power supplies. |
| 2.7.2 Self-contained Battery | Available types | AMU, ELSS, PLSS | Not sufficient to power other equipment. |
| 3. SAFETY CONSTRAINTS | | | |
| 3.1 Safety Guidelines | Number of astronauts per EVA | Two | Two man EV crew necessary for mission success to reduce the overall danger to crew safety. (2nd man may perform only as a "lifeguard") AAP ground rule. |
| | Use of safety tethers | Required | Tethers required for all operations at present time. |
| 3.2 Hazards | Operational hazards | Loss of thermal and pressure controlled environment<br>Loss of life support<br>Loss of communications<br>Loss of visibility<br>Loss of contact w/S/C<br>Loss of body control<br>Impact with structures<br>Entanglement<br>Etc. | For example, damage to space suit. |
| 4. ENVIRONMENTAL CONSTRAINTS | | | |
| 4.1 Radiation | Maximum acceptable | 54 rads | Dosage to blood forming organs of body (bone marrow) is limiting factor. Adequate eye protection assumed. |
| | Maximum permissible single acute emergency exposure | 200 rads | Altitudes considered safe for extravehicular activity are below 300 mi. and above 18,000 mi. for mission durations considered for AAP. |
| | Equivalent shielding of EVA suit | | |
| | Type of personal monitoring devices | Dosimeters | Device mounted on wrist. |