

LMSC-A990558

DESIGN GUIDE FOR

SPACE SHUTTLE LOW-COST PAYLOADS

30 June 1971

Contract NAS W-2156

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SPACE SYSTEMS DIVISION • SUNNYVALE, CALIFORNIA

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F O R
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GLOSSARY OF TERMS USED IN THIS DOCUMENT

Although generally accepted in a technical sense, many of the terms used herein have various connotations within the aerospace community. Thus, as a guide, some of the basic terms are explained below.

PAYLOAD SYSTEM	is used to collectively describe the payload, the payload/launch vehicle adapter, and any deployment or separation devices required to effect a separation of the payload from the launch vehicle.
PAYLOAD	is a collective word used to describe the total operating entity, such as a satellite, that is launched into orbit by the launch vehicle; it comprises spacecraft and experiments but excludes launch vehicle related elements - such as adapters - that are non-functional in orbit.
BASELINE PAYLOAD	describes a representative example of a current unmanned payload used to provide a basis for the development of low-cost approaches and cost comparisons; examples are: OAO-B, SRS, Synchronous Equatorial Orbiter. The term Initial-Design is used interchangeably with Baseline.
LOW-COST PAYLOAD	refers to payload designs which were developed using low-cost approaches and techniques that are compatible with the cost-saving potential arising from use of the new launch vehicles.
LAUNCH SYSTEM	is the system used to inject or place the payload into its specified orbit; specifically in the Payload Effects Study, three launch vehicle types are considered - current alternate expendable, new low cost expendable, and reusable Space Shuttle and Space Tug.
SUBSYSTEM	refers to the major functional elements of a payload, covering prime equipment categories: ● Experiments ● Structures & Mechanisms ● Electrical Power ● Stabilization & Control ● Attitude Control & Propulsion ● Communications, Data Processing & Instrumentation ● Environmental Control

MODULE refers to a complete functional portion of a subsystem; a module comprises several components and interconnect electrical harnesses housed within a single structural box.

COMPONENT an assembly such as a star tracker, transmitter, or similar. Components are assemblies of parts.

PART a piece of hardware, a quantity of which are assembled into a single component; examples are: transistor, lens, shaft, etc. Parts categories considered are:

- High-reliability
- MIL-Spec
- Aircraft
- Commercial

Section 1
INTRODUCTION

This document, "Design Guide for Space Shuttle Low-Cost Payloads*", has been prepared by Lockheed Missiles & Space Company, Space Systems Division, as part of the total effort under NASA Contract NAS W-2156, covering the Payload Effects Analysis Study. A separate document, LMSC-A990556, dtd 30 June 1971, is the Final Report which describes the study and its results.

The principal objectives of the overall study were:

- Create new low-cost designs for representative future NASA unmanned payloads
- Prepare program plans and costs for these low-cost payloads, establishing the dollar savings relative to current equivalent (baseline) payloads
- Establish the effect of the new low-cost payloads upon the future launch/transportation systems and ground systems
- Prepare a designers handbook which will delineate for future unmanned payload designers the principles of low cost design developed during the course of the study.

This document is the result of effort on the last item listed. It was prepared with the intent to summarize the principal results of the Payload Effects Study, with emphasis on data which is important to the designers and engineers

* The word "Payload" as used in this document, refers to a flight vehicle, usually comprising a spacecraft and an experiment package, which is carried to orbit in the cargo bay of the Space Shuttle. For additional definitions, consult the Glossary.

who will create the unmanned payloads to be flown in the Space Shuttle era, and who will bear the principle responsibility for implementing low-cost hardware programs and associated development and flight programs.

Preliminary economic analyses indicate that payload design approach has a dramatic influence on the overall program costs for missions in the 1978 to 1990 period. So strong is this influence, that without the implementation of the low-cost payload approaches, the Space Shuttle itself cannot be shown to be cost-effective. The importance of applying the low-cost methodology provided herein (or equivalent) cannot be over-emphasized if the NASA space programs of the future are to be attained within reasonable (and austere) national budget constraints.

Note: The English system of units has been used in analysis and design. Factors for conversion of English units to International units are included as required.

Section 2 INTENDED USAGE OF GUIDE

a. Basic Objective

The initial objective of this document is to communicate to the future space program managers, planners, designers, engineers, and scientists; (1) the basic relationships between payload designs and program cost effects, and (2) some concepts for designing low-cost payloads and implementing low-cost programs.

b. Need for Implementation of Concept at all Levels

Because much of the low-cost methodology proposed is at variance with past practice, there must be a determined effort to apply the changed concept. This will require the full support of leaders in both Government and industry if the very significant economic advantages are to accrue. The cost savings potentially realizable in the design of payloads will not be attained if traditional methods of spacecraft development and production are unchanged.

c. Limitations in Scope of Payload Coverage

The total NASA/USAF Mission Model, which lists all of the Space Shuttle traffic for the 1978-1990 period, includes many types of payloads which will be carried by the Space Shuttle; these include, as principal categories:

● Unmanned Programs

- Earth Satellites
- Planetary Orbiters
- Space Probes
- Propulsion Stages

- Manned Programs

- Space Station Segments
- Capsules for Shuttle Sortie Missions
- Large Experiment Capsules
- Logistic Supply Capsules for Man Support

The aforementioned Payload Effects Study was limited to development of low-cost payload concepts for three unmanned earth satellites: namely, an orbiting astronomical observatory (OAO), a synec orbiter (SEO), and a small research satellite (SRS). Although the low-cost concepts included herein are applicable in at least a general sense to all unmanned payloads, and possibly to certain of the manned-program payloads; specific attention is given to the earth-satellite type.

The low-cost payloads designed during the Payload Effects Study had on-orbit lifetime requirements as follows:

- SRS 6 months
- OAO 1 year
- SEO 2 years

However, the low-cost design concepts implemented in their design and presented herein are equally applicable to payloads having longer on-orbit lifetime requirements. Costs savings comparable to those attained in the designs of the low-cost SRS, OAO and SEO should be realizable in the design of long-life spacecraft, through the application of low-cost design concepts.

The lifetime of any spacecraft can be extended by refurbishment. Therefore, detailed cost analyses should be performed to determine the optimum nominal on-orbit lifetime requirement (refurbishment cycle time) of a new spacecraft. Such analyses may establish that very long on-orbit lifetimes, requiring costly equipment redundancy and backup modes, may not be cost effective.

d. Transportation System Data & Interfaces

This document includes brief summaries of Space Shuttle and Space Tug performance, environmental, and interface data as they relate to the low-cost payload concepts. For more specific data, the reader is referred to a NASA document, "Space Shuttle Design Handbook".

e. Emphasis on Low-Cost Approaches

This document is not intended as a design handbook to instruct designers in the specifics of hardware design and analysis. Rather, it is devoted to data which will:

- (1) provide orientation to the effects of payload design upon program cost
- (2) identify those payload design areas in which cost savings can accrue as a result of the use of the Space Shuttle
- (3) describe the special design considerations necessary in adapting a payload for transportation and support by the Shuttle system.

Section 3

GENERAL RELATIONSHIP OF PAYLOAD DESIGN TO PROGRAM COST

3.1 TYPICAL PROGRAM COST BREAKDOWNS

It is important to identify all of the cost categories so that none are omitted when cost-effectiveness tradeoffs are performed in deriving design-approach decisions. The principal categories are: (1) Research, Development, Test and Evaluation (RDT&E); (2) Unit Recurring; and (3) Operations (also recurring-type costs). A breakdown of these into sub-elements is shown on Fig. 3-1.

3.2 EFFECT OF RELATIVE RDT&E AND RECURRING COSTS3.2.1 Discounted \$ Effects

In making cost tradeoffs to determine which of alternative designs to select, situations occur which show that lower costs may be obtained in either RDT&E or in recurring costs. If there is this option, it is usually better to select the approach which accrues the higher dollar saving in RDT&E. Most funding and expenditures are now based upon a discounting principle wherein money not spent today allows spending of the same money later plus the interest that would accrue on the money during the time interval. In other words, \$1.00 not spent today allows spending of \$1.10 one year from now (using a 10 percent discount rate). Similarly, \$1.00 saved in RDT&E in 1971 allows spending \$1.25 in 1973 on recurring costs (10 percent discount rate). Conversely, \$1.00 saved on recurring costs in 1973 would have an absolute worth in 1971 (if used for RDT&E) of only \$0.83. In summary, if options exist, savings in RDT&E should be pursued more strongly than savings in recurring costs because RDT&E spending usually precedes other expenditures by 2 to 3 years and dollar discounting will play an important part in deriving true total savings.

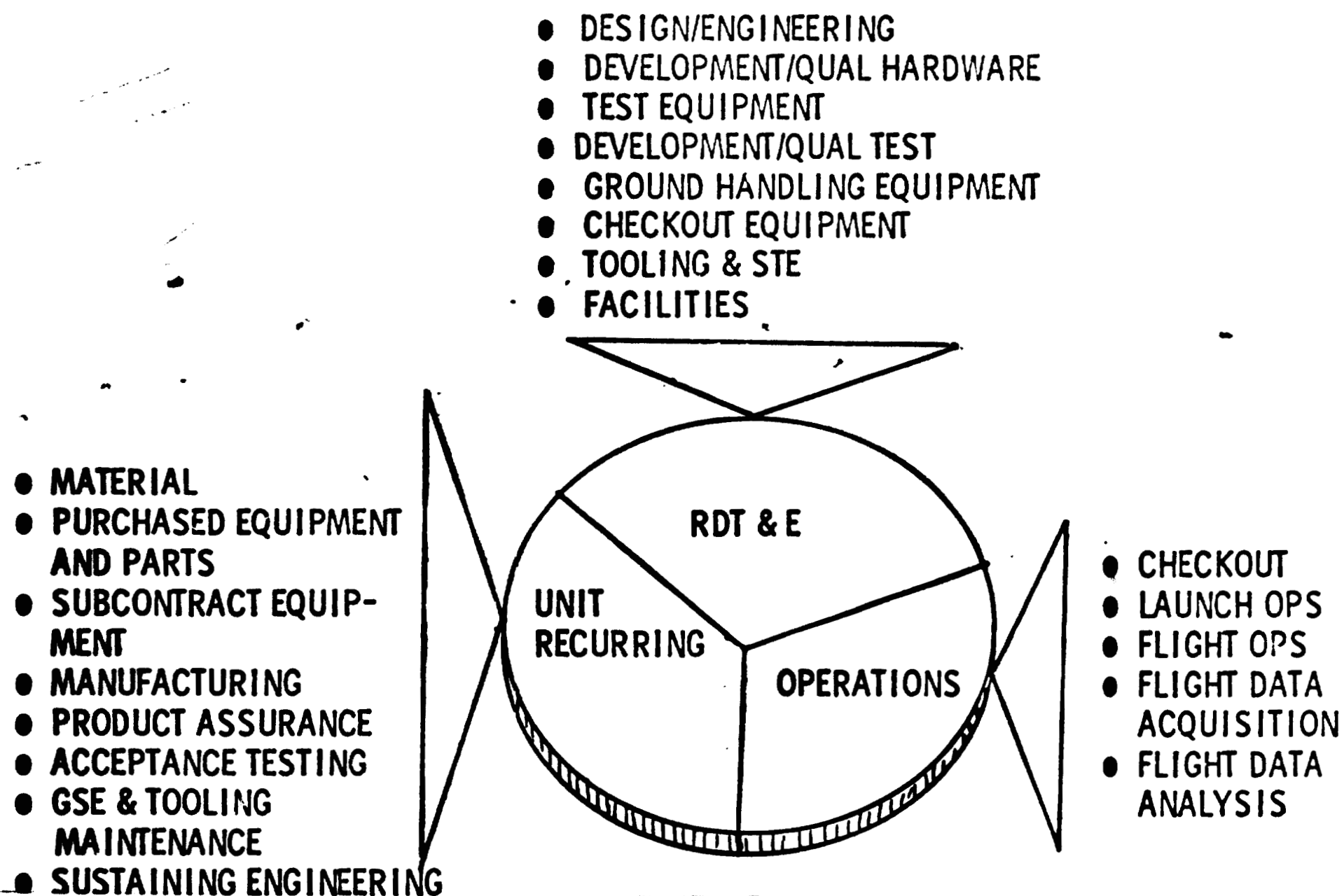


Fig. 3-1 Cost Breakdown for Visibility of Payload Effects

3.2.2 Unit Recurring Cost Influence on RDT&E Costs

For typical development and qualification test programs, even at the reduced-level planned for low-cost payloads, up to 1.5 equivalent payloads (varies depending on complexity of payload) are used as test hardware. These costs accrue to the RDT&E total for testing. This fact indicates that multiple savings can be accomplished by reduction of payload unit recurring cost; a saving in both unit cost and RDT&E will result. The aforementioned discounting (par. 3.2.1) provides additional benefits to the portion of cost allocatable to RDT&E.

3.3 IMPACT OF PAYLOAD DESIGN UPON PROGRAM COSTS

Payload design at all levels; system, subsystem, component, or part; significantly affects all categories of program costs. Payload designers must consider the cost impact of all design decisions if program costs are to be minimized. This requires an awareness among designers of the cost consequences of their decisions; an awareness that will arise only through study of the design-cost relationships.

3.3.1 Potential Cost-Reduction Areas

There are a large number of potential cost-reduction areas which come into being as a result of using the Space Shuttle. These have been reduced to a reasonably brief list as shown on Fig. 3-2. The "X" entered in each block indicates to which cost categories each cost reduction approach is applicable. This basic form, or a similar one, can be used as an initial checklist to determine if all cost-reduction possibilities are being pursued in the design of a particular payload.

A similar matrix chart may be used during preliminary design evaluation and tradeoff as shown in Fig. 3-3. On this chart, the proposed hardware approach is listed in the left-hand columns; designations of "high", "moderate", or "low" cost impact are entered in each cost category. One chart may be made for each subsystem of the payload.

COST REDUCTION AREA	Launch Applicability	RDT&E								UNIT PRODUCTION					OPERATIONS				
		Program Mgt.	Payload Integr.	Devel/Des. Engr	Devel/Qual Hdwr	Devel/Qual Test	Test Equipment	Tooling	Checkout Equip.	Facilities	Matl./Parts	Purch. Equip.	Manufacturing	Product Assur.	Accep. Test	Checkout	Launch Ops	Flight Ops	Flt Data Acq/Anal
SIMPLIFIED CONTRACT/DOCUMENT REQUIREMENTS	E/S	X	X	X		X					X	X	X	X	X				
SIMPLIFIED CONFIGURATION MANAGEMENT (TRACEABILITY)	S	X	X	X		X					X	X	X	X	X				
USE OF PROVEN TECHNOLOGY-OFF SHELF	E/S			X		X									X				X
USE OF LOW-COST MATERIALS	E/S				X	X		X			X		X	X	X				
DECREASE STRESS LEVEL ON PARTS	E/S				X	X					X	X			X	X			
USE LOWER-QUALITY PARTS	S			X	X	X		X		X	X	X	X	X	X	X			
INCREASE STRUCTURE SAFETY MARGIN	E/S			X	X	X	X	X		X	X	X	X	X	X	X			
USE LOWER RELIABILITY GOALS (W/MAINTENANCE)	S			X	X	X	X		X		X	X		X	X	X			
INCREASE VOLUME OF PACKAGES	E/S			X				X				X	X	X					
INCREASE HARDWARE WEIGHT ALLOWANCE	E/S			X	X			X			X	X	X	X					
SIMPLIFY/MODULARIZE HARDWARE	E/S			X	X	X	X		X		X	X	X	X	X	X			X
SIMPLIFY/REDUCE GROUND TESTING	S	X	X	X	X	X	X			X									
USE ORBIT MAINTENANCE/REFURBISHMENT AND REUSE	S	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
EMPLOY HARDWARE UPDATE (VS NEW HARDWARE)	S	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
USE PRE-DEPLOYMENT ORBIT CHECKOUT	S			X	X	X	X		X	X	X	X		X	X	X		X	X

* E = EXPENDABLE; S = SHUTTLE

Fig. 3-2 Potential Cost Reduction Areas

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LOW-COST PAYLOAD -COST REDUCTION		MODEL OAO-B (1 YR.) SUBSYSTEM CDP & I																
AFFECTED COMPONENT(S)	PROPOSED DESIGN APPROACH	COST CATEGORY AFFECTED																
		RDT&E						Recur. Prod.				Ops.						
		Design/Engineering	Develop. Hardware	Develop. Test	Grnd. Hand. Equip.	Checkout Equip.	Tooling	Spec. Test Equipment	Facilities (New)	Facilities (Mod.)	Material	Manufacturing	Prod. Assurance	Acceptance Test	GSE (Tooling Maint.)	Sustaining Engr.	Checkout	Launch Ops.
Prim. Proc. & Data Store. S/C Syst. Cont. Unit EDHE SDHE ACMU GFE Interface C. Rcvr. Beacon NB Transmitter WB Transmitter Tape Recorder Assy	Replace with: Computer Data Dist. Unit Interface & Timing Unit Software Reduce Qty. from 4 to 2 Delete; Track NBT Reduce Qty from 2 to 1 Reduce Qty from 2 to 1 Delete GFE Interface	H H H H M (-X-) L H L - H	H H H H M (-X-) L H L - H	H H H - - (-) - H - - H	- - - - - - - - - - H	H H H - - - L - - H	H H H - - - L - - H	M M M - - - - - - H	- - - - - - - - - -	- - - - - - - - - -	H H H - - - M M M H	H H H - - - M M M H	M M M - - - - - - H	M M M - - - - - - H	- - - - - - - - - H	H H H - - - M M M H	H H H - - - M M M H	M M M - - - - - - H

H = High Cost Impact "(-)" = Cost increase
M = Moderate Cost Impact
L = Low Cost Impact

Fig. 3-3 CDPI Subsystem Changes vs Cost Categories Affected

3.3.2 Basic Payload Effects

The basic Payload Effects cost drivers are listed on Fig. 3-4. Their impact on program costs will be discussed in detail in later sections. They are listed here only to provide background on the general relationship of payload design approaches to costs.

3.3.3 Payload Designs vs Engineering Costs

Engineering costs are directly related to the complexity of the designs to be created. To reduce engineering costs, payload designers must strive for design simplification, beginning in the concept design phase and continuing through the detail design phase. Much of the complexity of spacecraft has been due to overspecification of functional requirements and of limitations on the size, weight, and power consumption of equipment. When the Space Shuttle replaces expendable launch vehicles for the transportation of space payloads, the factors which have led to overspecification will lose much of their significance, specifications can be less restrictive, and payload designers will have the freedom to create simpler, less costly designs.

Simple, low-cost design pays off in reduction of engineering costs; examples are:

<u>Low-Cost Design Approach</u>	<u>Engineering Required</u>
Fewer parts	Lesser quantity of drawings and specifications
Ruggedized, high-safety factor structure	Reduced level and degree of analysis
Decreased-density equipment packaging and assembly	Simpler detail drawings and design analyses
Relaxed dimensional tolerances	Fewer interface correlations
Use of pre-qualified hardware	Fewer and simpler specifications

The design engineering cost represents between 25 percent and 36 percent of total RDT&E; it is therefore very important that a simple cost-effective payload

- PAYLOAD DEVELOPMENT AND PRODUCTION COSTS
 - Trade weight/volume for design simplicity (reduce mechanisms, use high safety margins)
 - Use low-cost materials and off-the-shelf components
 - Establish hardware commonality - standard components, spacecraft, interfaces, GSE
 - Simplify ground/flight tests
- GROUND AND FLIGHT OPERATIONS
 - Provide for orbit checkout, repair, refurbishment
 - Reduce payload life (MTBF and redundancy)- use retrieval, refurbishment, reuse
 - Extend system usefulness - use hardware update and modernization

Fig. 3-4 Payload Effects Cost Drivers

design be coupled with equally simple and direct engineering drawing procedures, engineering analytical methods, and engineering management. This will assure that otherwise attainable savings will not be dissipated in unnecessary documentation or in over-specification.

3.3.4 Payload Design vs Development, Qualification, and Acceptance Testing Costs

In the design of payloads for launch by the Space Shuttle there are a number of departures from past practice that will lead to the reduction of testing costs.

3.3.4.1 Use of Shuttle Permits Less Testing. The firm knowledge that no payload will be committed to its mission orbit without a successful system check-out in the cargo bay of the orbiting Shuttle, and the knowledge that a malfunctioning payload can be recovered for repair in orbit or for return to earth will minimize the fear of the consequences of failure that has historically led to the incorporation of redundant equipment and backup systems not clearly required for mission success.

3.3.4.2 Proper Application of Redundancy. The elimination of functional redundancy obtained from different sets of components in favor of redundant identical components will eliminate the design engineering and the development and qualification costs associated with one set of components. Similarly the elimination of backup systems will further reduce costs.

3.3.4.3 Increased Safety Factors. The payload weight and volume capability of the Shuttle will permit the design of very conservative payload structure with factors of safety high enough to eliminate requirements for structural load testing.

3.3.4.4 Modularization of Equipment. An additional benefit to payload design deriving from the Shuttle payload capability will be true modularity of equipment packaging, which has been inhibited by the weight and volume constraints of expendable launch vehicles. Well-designed equipment modules of a payload system may be removed from the system and replaced by an equivalent module

without perturbation of the system function or calibration. Such interchangeability of modules will simplify, and reduce the costs of, system level testing.

3.3.4.5 Use of Previously-Qualified Hardware. Payload program testing costs, as well as design and development costs, will be reduced by incorporation of previously qualified components.

3.3.4.6 Costs of Testing. Development and qualification test hardware represents between 30 percent and 45 percent of RDT&E costs. Once the types of tests required have been determined and the quantities for each test prescribed, the hardware becomes essentially a fixed cost (based upon duplicates of the flight hardware design). The development and qualification testing represents between 10 percent and 15 percent of RDT&E costs.

Considering these statistics, it is very important to minimize test hardware by:

- (1) Specify the absolute minimum test hardware to verify the performance
- (2) Utilize the same hardware for more than one test if at all possible

Of course, the optimum low-cost approach involves elimination of certain testing which is either redundant with some other test or does not add to the absolute verification of hardware capability.

3.3.5 Payload Design vs Tooling and Manufacturing Costs

3.3.5.1 Tooling Relationship to Design. The costs of tooling for space payloads are directly related to the tolerances specified for the assembly and alignment of structures and for the installation of equipment. The larger, heavier structures of Shuttle-launched payloads should not require as much tooling as the lighter, more flexible structures of historical payloads.

Also, tooling costs can be reduced if, in the design of a payload structure, a single structurally-stable plane can be established on which all equipment

modules requiring alignment can be mounted. This concept was found to be feasible in designs for both a low-cost OAO-type payload and for a low-cost SEO-type payload.

3.3.5.2 Manufacturing Approaches Affected by Design. Manufacturing costs can be reduced if payload designers avoid sandwich materials, chemical milling, machine contouring, and other techniques for weight control commonly used in the design of payload for expendable boosters but unnecessary in the design of payloads for the Shuttle.

Low-density packaging of equipment into modules and of modules into payload structures will help to reduce manufacturing costs by providing free access to all components.

The manufacturing costs of electronic equipment can be reduced by low-density packaging because of improved access for the placement of parts and for inspection, which in turn results in the reduction of scrap and rework.

3.3.5.3 General Design Guidelines. A design may be relatively simple from the engineer's viewpoint, but can also be difficult or expensive to produce. Care must be exercised by the designer to follow-through and provide the procurement, tooling, manufacturing, and product assurance organizations with the utmost flexibility in obtaining minimum costs. The following payload design approaches are typical of those directly related to lower costs and should be used wherever possible:

- Specify simple, easy-to-fabricate structure; use aluminum in lieu of titanium, beryllium
- Eliminate special machining or metal removal processes such as chemical milling
- Provide liberal tolerances on all parts

- Specify use of readily-procured materials in commercially available forms, gauges (extrusions, sheet metal).
- Provide maximum possible volume in part/component packaging to allow easy assembly, access for inspection.
- Specify readily-available processing methods (plating, coatings, thermal treatment).
- Locate alignment-critical equipment in single area of payload to allow simplified fixturing.

3.3.6 Payload Design vs Ground Handling and Checkout

There are a number of often overlooked things the payload designer can do to minimize the costs of GSE and the recurring operations associated with handling, transportation, checkout, storage and logistics support. If such considerations are incorporated at the outset of the program they do not add to the design burden, eliminate the necessity to modify designs later to incorporate support provisions, reduce the amount of new GSE that must be designed, and minimize the amount of STE and tooling needed to produce the payload. Some of the factors listed below are system criteria, some relate to structural and mechanical design, and others impact electrical and/or software design; they are all important in minimizing total payload cost.

3.3.6.1 Rugged Structure Effect. The design requirements for structures, sub-assemblies, components, and piece parts for low cost payloads should include the ground handling and transportation loads; this is contrary to some past practices where weight limitations dictated that ground conditions should not place limitations upon the design. However, the cumulative effect of (a) special handling, (b) special testing, (c) special ground support equipment, and (d) all the associated procedures, validations and documentation is higher development and operating costs. These costs can be minimized under the "low-cost" payload ground rules. Simple design trade studies that consider the cost of design, manufacture, test, maintenance, operation and documentation of the

GSE will usually provide a quick indication of whether to "beef up" or otherwise increase the capability of the payload or to require additional complexity of ground operations and GSE.

For example, maximum Shuttle-imposed loads are 3.3g, whereas typical in-plant dollies probably do not impose loads of more than $\pm 2g$ vertically and $\pm 1g$ in the horizontal plane. However, if shipment of the payload or components by truck or by air is necessary, loads of $\pm 6g$ may be encountered. Rail shipment is characterized by shock loads that may range from $\pm 6g$ to $\pm 30g$ lengthwise of the rail car, and hence rail transportation should be avoided. Therefore, a component such as a solar array panel should be designed to be self-supporting for simplicity in assembly, inspection, and testing, and to withstand $\pm 6g$ ground transportation loads. Shipping and storage containers will still be necessary to protect the components from contaminants and to support them at the designed load bearing points, but complex shock mitigation systems and recording equipment, and special handling are not required. Similar treatment should be given to other payload elements such as antennas, main structure, modules and large experiment structure or packages. It should be noted that choice of materials, finishes and assembly techniques are as important in the ground environment as the structural load factors, and the term "rugged" applies to the total environment.

3.3.6.2 Hard Points and Lift Provisions. Suitable mounting provisions and load paths are designed into payloads to mate with booster adapters or specified primary interfaces, but often the need to hoist, tilt, rotate and translate in order to fabricate, assemble, test, ship, store and mate is neglected. In the past this has resulted in the design of expensive handling fixtures, complex shipping containers and tricky mating maneuvers; the removal of severe payload weight and volume constraints allows the incorporation of lifting pads or hard points, lifting eyes or holes, handholds, and tiedown brackets, holes, recesses or cleats. Such provisions should be provided for the assembled spacecraft and independently for each interchangeable module and replaceable component as appropriate to its size and weight. Cables, slings, spreader bars,

blocks, etc., are standard items and their usage may be shared; special cradles, fixtures, grips, etc., that are payload-peculiar will be charged to the program. Some special equipment may be necessary, such as cantilevered booms to assist in module installation or removal after a payload is installed in the cargo bay; perhaps this can be the same equipment used for on-orbit maintenance.

3.3.6.3 Alignment References. It is often desirable to be able to determine the principal axes of payloads for checkout as well as installation into Shuttle. External optical reference marks or targets allow this to be done quickly with common optical instruments, and reverified as often as necessary. Where the reference is transferred from a plug-in module a target on the module may also be desirable. If the payload is specifically destined for use with the Shuttle, the targets should be located so they are visible through the cargo bay doors and/or access ports. Each payload module should be self-aligning to the payload main structure to a specified accuracy, and for any position or attitude of the payload.

3.3.6.4 Thermal Environment Effect. The thermal design is usually optimized for the space environment, but the ground checkout and pre-launch environment must not be neglected. GSE can supply dry, conditioned air or a nitrogen purge to a general area of the payload, but it requires equipment and personnel that must be maintained and trained, slows up operations and is usually avoidable if consideration has been given to thermal paths and radiative surfaces. While power-on or load cycles can be limited under ground conditions such limitations slow up testing, increase integration complexity and are, in general, undesirable and within the designer's prerogative to control.

3.3.6.5 Protection of the Payload. All appendages, alignment devices, connectors, fasteners, sensors, etc., should be protected against handling damage, preferably by placement on or within primary structure, or, alternatively, by simple fenders or guards. Protective caps or covers should be used for optics, connectors and surfaces sensitive to dust, moisture, and other contaminants; these should be designed for ready accessibility for removal during test or

launch operations. Wherever possible systems and materials that do not require such protection should be selected.

3.3.6.6 Checkout of Payload. This is one of the most important considerations in low-cost design because the function extends from the piece-part level of procurement through the entire life cycle of the payload(s). Specific examples of low-cost payload design requirements are: (1) eliminating or reducing testing by designing for large factors of safety, ample performance margins, broad tolerances in operating parameters; (2) self-checking or built-in test provisions for each "black box"; (3) separate functional verification of redundant elements; (4) interchangeability of replaceable modules/"black boxes" without recalibration; and (5) independent safety-monitoring instrumentation (when the payload is to be flown in a manned Shuttle). The payload system should be configured to take advantage of automatic, computer-controlled test equipment to provide consistency of test methods/equipment and data in the factory, at the launch base, on orbit, and for checkout of logistics spares. Compatibility with a standard payload test set (PTS), if the PTS concept has been adopted and implemented, will satisfy several of the above-listed requirements and provide a standard power, command and data interface with the Shuttle.

3.3.7 Payload Design vs Launch and Flight Operations

Recurring operations, particularly on programs having long operational life, account for an appreciable fraction of the total program cost; this is principally evident in the manning level required. Systems designed to require less support, simpler procedures and fewer activities or operations will cost less. Mandatory ground rules should include use of: (1) common support equipment; (2) standard RF links and data formats; (3) shared control facilities; (4) STADAN and NASCOM services, including TDRS when available; and (5) existing government data processing capabilities.

3.3.7.1 Effect of Safety Factors. All pressure vessels, tanks, pyrotechnics, radiation sources, or toxic materials used in low-cost payloads must be self-safing, inherently safe because of the safety factors used in the design, or

capable of being rendered safe by remote command. It is desirable that pre-flight servicing, including loading of propellants and pressurants, be done quickly and without the necessity of clearing the area of personnel or requiring that other activities in the vicinity be suspended. Safe-Arm plugs or switches should be provided to minimize hazards to launch vehicles and personnel during the pre-countdown joint flight acceptance compatibility tests.

3.3.7.2 Test Connectors. Means should be provided for conducting RF closed-loop tests without having to obtain range clearance to radiate. Safety instrumentation sensor outputs should be routed directly to test connector so checks can be made without having to activate payload power, signal conditioning and data subsystems. Hardline controls should be provided for safing and initialization. All test connectors should be designed fail-safe for disconnect.

3.3.7.3 Software. The payload system software should be compatible with payload test set interfaces and should be designed for ease of validation as well as for ease of programming changes during test and operations. The need for computer programmers in routine operations modes should be avoided. Extensive use of independent subroutines for specific operations is desired, with internal program controls to prevent inadvertent memory dumps or initiation of hazardous sequences. It is important that the software system design be established concurrently with hardware design requirement formulation. On-board memory capacity should be at least twice the estimated basic requirement so that straightforward programming techniques can be used and real-time changes can be made without having to employ sophisticated memory conservation devices. (These comments apply equally to the payload and to automatic payload test equipment.)

3.3.7.4 Autonomy of the Payload. Maximum autonomy of payload operation is the single most effective design feature for reducing launch and flight operations cost, both direct charges to the program and services supplied by other government-supported activities. An autonomous payload should be able to maintain its health (i.e., power, attitude, temperature, configuration and sensor protection) for long periods without ground intervention, including substitution

of backup equipment or functions to correct failure modes. This implies a self-check capability, a large stored command repertoire, and a passive 'storage' mode. As a result, the ground tracking and data acquisition system and the ground control center need not provide continuous coverage for the payload, need handle fewer health parameters, generate fewer and simpler commands, and have more freedom for "conflict scheduling" of network resources. The mission planners have more flexibility in planning payload operations and in modifying operations as a result of experiment data or changed requirements. Sustaining engineering support of payloads can also be shared because subsystem specialists and computer programmer support are generally not needed for updates and modifications to the payload on-board computer program.

3.3.7.5 Impact of TDRS. The long-range plans of the NASA Office of Tracking and Data Acquisition provide for the implementation of a network of Tracking and Data Relay Satellites, and the phasing out, as the TDRS become operational, of some of the remote tracking stations that are presently part of STADAN or MSFN. Considerable overlap of service is required during this period because the operational period of some of the satellites will span the transition. Changes in frequency, format and antenna pointing may be required, either by ground-commanded configuration change or by a Shuttle revisit and module replacement. Cost of including this versatility in a payload design must usually be funded by the payload, but the operational savings accrue to the T&DA. Hence, dual capability does not provide payload cost savings, though total cost to the government may be decreased.

Section 4

PAYLOAD REPAIR/REFURBISHMENT EFFECT ON DESIGN AND COSTS

To provide for the maximum reduction in program cost of Shuttle-launched low-cost payloads, it is necessary to employ repair and refurbishment and reuse techniques for not only the payload but for equipment modules and components which comprise the payload. Design of the payload to facilitate repair/refurbishment, in orbit and on the ground, has a direct bearing on the program costs. This section is devoted to describing the concepts of repair and refurbishment, the relationships with design approaches, and the magnitude of dollar savings which can be obtained on typical payload programs. Throughout the section, OAO and SEO payload program reference data are used.

4.1 BASIC APPROACH TO PAYLOAD REPAIR/REFURBISHMENT AND COST EFFECTS

The concept developed for repair and refurbishment includes these basic elements:

- Repair of initial-launch payload on-orbit following pre-deployment checkout
- Periodic refurbishment of the payload by replacement of equipment modules
- Orbit repair of random failures which occur between refurbishments
- Refurbishment (and reuse) of equipment modules by replacement of components therein
- Refurbishment (and reuse) of components by replacement of parts and subassemblies therein.

Figure 4-1 lists the basic steps involved in this overall concept. Throughout this discussion, the meaning of the two primary terms are:

"Repair" - the replacement of a failed module with a new or refurbished module thereby returning the payload to operational status (but not changing the overall life expectancy of the payload).

"Refurbishment" - the replacement of all hardware elements which have degraded over the operating time, thereby restoring the payload (or module or component) to its initial operating life expectancy.

4.1.1 Repair of Payloads on Orbit

Repair of payloads on orbit falls into two categories: (1) repair following the rigorous launch/ascent, and (2) repair of random failures which may occur between the periodic refurbishment points.

4.1.1.1 Checkout and Repair after Launch/Ascent. Investigation of payload failure modes has revealed that a significant percentage of total failures occurs in the launch/ascent phase of payload total life or shortly thereafter. Although further detailed analysis of this condition is required, it appears desirable to isolate and eliminate these failures where possible. It is, therefore, required that replacement (spare) modules be carried to orbit with the payload. Checkout of the payload on orbit (using the Shuttle-carried payload checkout set) will reveal any failure or degraded performance of a module; the module can then be replaced prior to payload deployment to orbit.

There are several different modules in each payload with multiple quantities of some of these. It is planned (at least for the early launches, until statistical failure data can be accumulated) that one of each of the different modules (except the solar array unit) be carried to orbit with each payload launch. A typical listing of modules designed for the low-cost OAO are shown on Fig. 4-2. Equivalent data for the SEO modules is given on Fig. 4-3.

4-4

Subsystem	Module		Quantity per Payload	Delivered Cost per Single Module (\$ Million)	Unit Weight (lb)
	No.	Type			
Electrical	1	Battery	1	.919	448
	2	Power	1	.434	196
	-	Solar Array	2 ea.	.368	175
CDPI	1	Computer	1	.716	322
	2	Communication	1	.904	104
Stabilization & Control	1,2	Primary Attitude	1 ea.	1.011	161
	3	Secondary Att.	1	1.055	150
	4	Wheel	1	.815	186
Attitude Control	1,2 3,4	Attitude Control	1 ea.	.241	207
Experiment	1	Electronic	1	.765	156
	2	Electro-Mech.	1	.917	173
TOTALS				8.145	2268

1 lb = 0.4536 kg

Fig. 4-2 Low-Cost OAO Modules

LMSC-A990558

Subsystem	Module		Qty. per P/L	Delivered Cost Per Single Module (\$1000)	Unit Weight (lb)
	No.	Type			
Electrical	1	Battery	1	187	170
	2	Power Control	1	402	106
	3,4	Paddle Drive	1 ea	120	39
	5,6	Solar Paddle	1 ea	264	69
Commun., Data Proc., Instrumen.	1,2	Data Handling	1 ea	665	81
	3	Communications	1	889	87
Stabilization & Control	1	Sensing & Flight Control	1	1117	98
	2	Momentum	1	504	125
Attitude Control	1,2	Attitude Control	1 ea	139	143
	3,4				
Experiment	1	Photographic	1	1293	419
	2	Vidicon Camera	1	381	86
Totals				5961	1423

1 lb = 0.4536 kg

Fig. 4-3 Low-Cost SEO Modules and Costs

4.1.1.2 Repair of Random Failures. Because of the comparatively high cost of transportation to perform a repair revisit, it is not desirable to perform "repair only" on a low earth orbit payload when a random catastrophic failure occurs within the last three months of a 1-year operating period. Rather, a refurbishment should be performed. If a failure occurs prior to the nominally-scheduled 1-year point, the refurbishment will be rescheduled. Similarly, if satisfactory operation of the payload continues beyond the 1-year point, the refurbishment can be delayed. The statistical probability of failure occurrence (or wearout) has been assumed to average a 1 year time period for an OAO-type payload.

On a Synec orbit payload (SEO), it is not proposed to perform "repair only," when a random catastrophic failure occurs. Rather, a payload replacement will be performed and the failed SEO will be returned to earth for refurbishment. The statistical probability of occurrence of random failures on SEO-type payload has been assumed to average a 2-year time period.

4.1.1.3 Spares for On-Orbit Repair. The cost of spares replacements for repair of typical payloads has been estimated. It has been assumed that a group of 4 SEO launches will require replacement of two modules or an average of one module for each 2 SEOs. The average cost is assumed to be maintained in a randomly distributed fashion over the program periods (detail analysis will be able to ascertain module failures and their occurrence probability).

Similarly, the average cost of a repair module for the OAO-type payload has been determined. This cost is also assumed to be applied randomly over a 6-year program with one module requiring replacement each year. Costs to cover these repairs should be included in the estimate for all program spares.

4.1.2 Refurbishment of Payloads

4.1.2.1 OAO-Type Payload Refurbishment in Orbit. It is planned to periodically (at the end of each 1-year operating period, or as varied by actual failure experience) launch a Shuttle with a set of replacement modules. The payload would

be retrieved by the Shuttle in low earth orbit and all modules replaced while the payload was tethered to the Shuttle on the extended deployment/retrieval gear or within the Shuttle cargo bay. The new modules would be installed in the payload: (1) by Shuttle crew members working in EVA or non-pressurized IVA; (2) by telefactor robots remotely controlled from the Shuttle crew compartment; (3) by automated devices; or (4) by combinations of these.

The Shuttle will return the failed or spent modules to earth for refurbishment.

4.1.2.2 SEO-Type Payload Refurbishment. It is planned to periodically (at the end of each 2-year operating period, or as varied by actual failure experience) launch a replacement payload with a Shuttle/Tug. The failed or spent payload would be returned to earth for refurbishment. The payload will accommodate refurbishment on orbit but because the Tug must be returned to earth for propellant refill in the mode assumed, the payload is also returned to earth for the refurbishment. For this operation, a ready-to-launch replacement payload must be available on the ground at all times.

The first, second, and third of a set of 4 payloads must be refurbished and ready for relaunch within 60 days (to match the average launch cycle time of 60 days between launches). Refurbishment of the payload will comprise replacement of the equipment modules and re-checkout of each payload. Because of the longer time span required for module-level refurbishment, two sets of modules should be held in standby; this allows approximately 4 months for turnaround time for refurbishment of a set of modules.

4.1.3 Refurbishment of Components

4.1.3.1 Component Refurbishment Concept. The lowest level of refurbishment planned is the component (two or more components are installed in each equipment module. It is planned that component refurbishment will be accomplished in field repair/refurbishment depots which are manned by technicians skilled in the various separate subsystems of several payloads; for example, electrical components, stabilization and control components, etc. Parts for this component refurbishment presumably would have been ordered at the time of payload

initial procurement (following a thorough logistic analysis and spares provisioning) and would be stocked in the depot in bonded stores. Also, required quantities of test sets would be supplied to the depot to allow complete test/checkout of the component after refurbishment.

The actual refurbishment of the component should be limited to replacement of those parts and sub-assemblies which can be readily removed and installed; conversely, reuse without replacement is planned for as many pieces of hardware as possible. Contractor-supplied repair and overhaul instruction manuals would be procured and used by depot personnel as guides for the refurbishment work. To facilitate the refurbishment activity, the design of the component hardware should be carefully done to be compatible with the basic depot disassembly-assembly procedures.

4.1.3.2 Refurbishment of Low-Cost OAO Components. An analysis has been made of the refurbishability of the various components in low-cost modules. Figure 4-4 lists for two subsystems of an OAO-type payload: (1) the components; (2) the assumed replaceable hardware elements; and (3) the estimated cost of replacement parts, disassembly/assembly labor, QA and test labor. In some instances, the component is not refurbishable; an example is the battery in the electrical power subsystem. In other cases, a new component is relatively inexpensive when compared to the replacement parts and labor needed to refurbish it; in these cases refurbishment is not planned -- examples are the computer, solar aspect sensor, transducers.

4.1.4 Refurbishment of Modules

4.1.4.1 Basic Approach. The basic approach to module refurbishment is listed in Fig. 4-5. The modules should be designed for this type of easy disassembly and reassembly.

4.1.4.2 Module Refurbishment. The costs of refurbishment for a typical low-cost SEO-type payload module are tabulated in Fig. 4-6. The "replacement parts" cost includes both the cost of parts required to refurbish the components and

Subsystem/ Module	Component	Replaceable Hardware Elements	Comp. Initial Recur. Cost	Cost of Component Refurbish.				Comp. Saving
				Repl. Parts	Disassy Assy	QA & Test	Total	
ELECTRICAL:								
Battery (No. 1)	Battery (3)	All	.030	-	-	-	-	-
	Pow. Cont. Unit	PCBs	.240	.170	.004	.006	.180	.060
	Pow. Reg. Unit	Xfmr, PCBs	.100	.070	.002	.003	.075	.025
	St. of Chg. Un.	PCBs	.080	.050	.001	.001	.052	.028
	Diode Box	PCBs	.040	.030	.001	.001	.032	.008
	Grd. Pow. Rel.	Cont. Points	.080	.001	.001	.001	.003	.005
	Batt. Current Shunt	Instrument.	.013	.007	.001	.001	.009	.004
Power (No. 2)	Volt. Reg/Conv.	PCBs	.060	.040	.002	.002	.044	.016
	Volt. Inverter	Xfmr, PCBs	.090	.060	.002	.003	.065	.025
	Pow. Dist. Unit	Fuses, CBs	.050	.030	.005	.002	.037	.013
CDPI:								
Computer (No. 1)	Computer (2)	All	.110	-	-	-	-	-
	I/F Tim. Unit(2)	PCBs	.125	.080	.004	.005	.089	.036
	Cold Plate	None	.010	-	-	-	-	.010
Comm. (No. 2)	Tape Recorder	PCBs, Motors	.125	.080	.006	.004	.090	.035
	NB Xmtr	PCBs	.030	.020	.001	.001	.022	.003
	WB Xmtr	PCBs	.150	.100	.003	.004	.107	.043
	Cmd. Rcvr(2)	PCBs	.080	.050	.002	.002	.054	.026

Fig. 4-4 Component Level Refurbishment Costs - OAO (\$ Million)

- DISASSEMBLY - Remove module cover, remove components
- HARDWARE REPLACEMENT - Replace components with new or refurbished items. Replace parts in selected components. Replacement hardware procured as spares and delivered concurrently with payload.
- COMPONENT TEST - Test all new or refurbished components prior to installation into module using checkout set.
- INSPECTION - Inspect all hardware to be re-used in accordance with payload maintenance manual (includes mechanical parts, structural elements, chassis, electrical cabling, etc.)
- ADJUSTMENT/ALIGNMENT - Check and adjust or calibrate as necessary in accordance with payload maintenance manual.
- ASSEMBLY AND TEST - Reassemble module and perform complete functional checkout using checkout set and payload simulator.

Fig. 4-5 Ground Refurbishment Approach for Modules (Field Repair Depot)

	Repl. Parts	Eng. Sup. 5%	Comp Disassy Assy	Module Disassy Assy	Comp QA & Test	Module QA & Test (80%)	Total Refurb Cost	% of New Cost
Electrical - Battery No. 1	.047	.006	.001	.002	.001	.060	.117	63
Power Cont.No.2	.124	.012	.011	.006	.011	.100	.264	66
(ea) Paddle Dr. No.3,4	.012	.006	.001	.003	.001	.032	.055	46
(ea) Solar Paddle No.5,6	-	-	-	-	-	-	-	100
CDPI - Data Handling,	.280	.014	.012	.003	.008	.056	.373	56
(ea) No. 1, 2								
Commun. No.3	.336	.012	.006	.002	.007	.080	.443	50
S&C - Flight Cont.No.1	.453	.025	.009	.006	.015	.108	.616	55
Momentum No. 2	.129	.020	.010	.006	.005	.072	.242	48
Attitude Control (ea) No. 1,2,3,4	.007	.003	.010	.003	.005	.032	.060	43
Exper. - Photo No. 1	.124	.080	.016	.016	.013	.252	.501	39
TV Camera No.2	-	-	-	-	-	-	-	100

Fig. 4-6 Cost of Module/Component Refurbishment - SEO

the cost of new replacement components (non-refurbishable). A nominal amount of support engineering has been included for the field depot work (5 percent of the engineering used by the contractor in the initial fabrication, assembly, and test of the delivered OAO hardware). Disassembly/assembly labor and QA/test labor costs are shown separately for the component and module-level work. The cost of a refurbished module ranges between 22 percent and 66 percent of the cost of a new OAO or SEO-type module.

4.1.5 Cost Savings with Refurbishment

Total savings resulting from refurbishment on typical unmanned payload programs are quite significant. In the following paragraphs, the module replacement cycle is explained and the program dollar savings are provided for typical OAO and SEO payloads. Similar savings can accrue on other programs wherein the refurbishment and reuse principle is employed.

4.1.6 Module Replacement Schedule and Costs

The module replacement schedule and replacement costs are tabulated on Fig. 4-7 for a 10-year program on a typical SEO-type payload. The SEO modules are replaced at 2-year intervals. The refurbished SEO costs \$3.54 million (or \$4.07 million at the end of the 4th and 8th years) as compared with a new SEO cost of \$9.63 million.

4.1.7 Overall Program Costs with Refurbishment

Estimates for complete programs, including launch and operations costs, have been made, utilizing the basic cost data derived in the bottom-up cost estimates on the low-cost payloads. To illustrate the dramatic impact of payload refurbishment (and design to accommodate) on payload program costs, data for a 10-year SEO program, with cumulative figures for 4, 6, 8, and 10 years; are tabulated in Fig. 4-8. The savings of \$360.87 million for the 10-year program represent a saving of about 50 percent for the Shuttle-supported refurbished SEO approach.

Subsystem	MODULES REPLACED AND COST (\$ Million)							
	End of 2nd Yr.		End of 4th Yr.		End of 6th Yr.		End of 8th Yr.	
	Mod. No.	\$	Mod. No.	\$	Mod. No.	\$	Mod. No.	\$
Electrical	1	.117	1	.117	1	.117	1	.117
	2	.264	2	.264	2	.264	2	.264
	3	.055	3	.055	3	.055	3	.055
	4	.055	4	.055	4	.055	4	.055
			5	.264			5	.264
			6	.264			6	.264
Communications, Data Processing, Instrumentation	1	.373	1	.373	1	.373	1	.373
	2	.373	2	.373	2	.373	2	.373
	3	.443	3	.443	3	.443	3	.443
Stabilization & Control	1	.616	1	.616	1	.616	1	.616
	2	.242	2	.242	2	.242	2	.242
Attitude Control	1	.060	3	.060	1	.060	3	.060
	2	.060	4	.060	2	.060	4	.060
Experiment	1	.501	1	.501	1	.501	1	.501
	2	.381	2	.381	2	.381	2	.381
Refurbished SEO		3.54		4.07		3.54		4.07
New SEO (Refurb.)		9.83		9.83		9.83		9.83

Fig. 4-7 Typical Module Replacement Schedule and Cost of Refurbishment for Low-Cost SEO (10-Yr. Mission)

Cost Elements	Expendable-Launched (\$ Million)				Shuttle-Launched (\$ Million)			
	Length of Program				Length of Program			
	4 Yrs.	6 Yrs.	8 Yrs.	10 Yrs.	4 Yrs.	6 Yrs.	8 Yrs.	10 Yr.
Non-Recurring Cost	97.99	97.99	97.99	97.99	85.70	85.70	85.70	85.70
Unit Cost of Payloads (incl. 1 backup unit)	100.52	145.20	189.88	234.56	49.15	49.15	49.15	49.15
Launch Cost	127.20	190.80	254.40	318.00	40.80	61.20	81.60	102.00
Operations Cost	27.12	40.68	54.24	67.80	23.90	35.85	47.80	59.75
Module Replacement (Cumulative) (inc. savings by refurb of modules & components)	-	-	-	-	14.16	30.44	44.60	60.88
Totals	352.83	474.67	596.51	718.35	213.71	262.34	308.85	357.48
Savings					139.12	212.33	287.66	360.87

Fig. 4-8 Savings with Payload and Module/Component Refurbishment on 2-Year SEO

4.2 DESIGN ANALYSIS REQUIRED FOR IMPLEMENTATION OF REPAIR/REFURBISHMENT CONCEPT

Initial studies have been performed to assure the technical feasibility of implementing payload refurbishment and on-orbit repair. A listing of the basic areas investigated is shown on Fig. 4-9. These same areas will require detail design study and execution in actual payload hardware designs. The following paragraphs briefly describe the background data for the items listed.

4.2.1 Deactivation and Attitude Stabilization of Payload

In retrieval of payloads by the manned Space Shuttle or by an unmanned Space Tug, the orbiting payload must first be made stable so that docking with Tug, manipulator, or Shuttle can be accomplished. Secondly, any potentially hazardous functions must be deactivated so that the safety of the space crew handling the payload is assured.

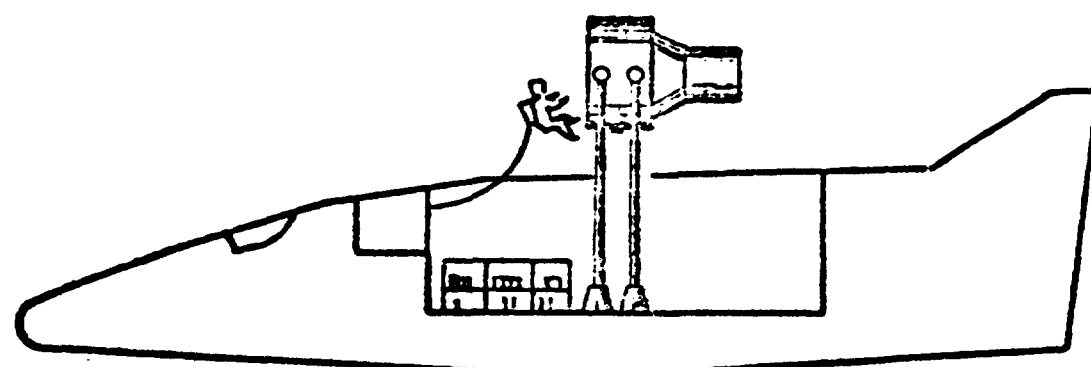
4.2.2 EVA vs Remote Manipulators

In the actual replacement of equipment modules on payloads in orbit, the modules can be manipulated by automated devices or moved by space crew in EVA. The activities involved are: (1) move module from stowage in the Shuttle to the payload; (2) remove failed module from payload; (3) insert replacement module; and (4) return failed module to stowage position.

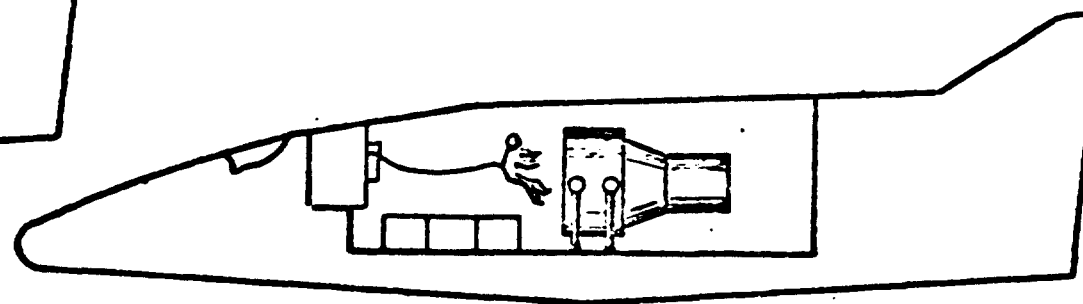
Figure 4-10 illustrates four basic modes of module replacement. The automated handling device is not illustrated; however, its feasibility has been studied in detail by NASA/GSFC. Also, a rather detailed study has been done on evaluation of telefactor robots under the cognizance of NASA/MSFC. The "pressurized IVA" mode shown requires a rather comparatively large investment in a pressurizable repair "hangar" in which crew members can work in shirtsleeve conditions. The early Space Shuttle activity will probably employ EVA and robot manipulators (free flying or fixed-mounted to Shuttle). In either case, the payload design must provide maximum capability for rapid equipment module

- DEACTIVATION AND ATTITUDE STABILIZATION OF PAYLOAD
- EVA VS REMOTE MANIPULATORS
- SPACE TUG OPERATIONS WITH THE SHUTTLE
- RECALIBRATION OF PAYLOAD AFTER MODULE REPLACEMENT
- GROUND-BASE RESPONSE TIME
- TURN AROUND TIME FOR GROUND REFURBISHMENT
- ON-ORBIT RE-CHECKOUT AFTER MAINTENANCE/REFURBISHMENT
- PAYLOAD WEAROUT/RELIABILITY VS REFURBISHMENT CYCLE

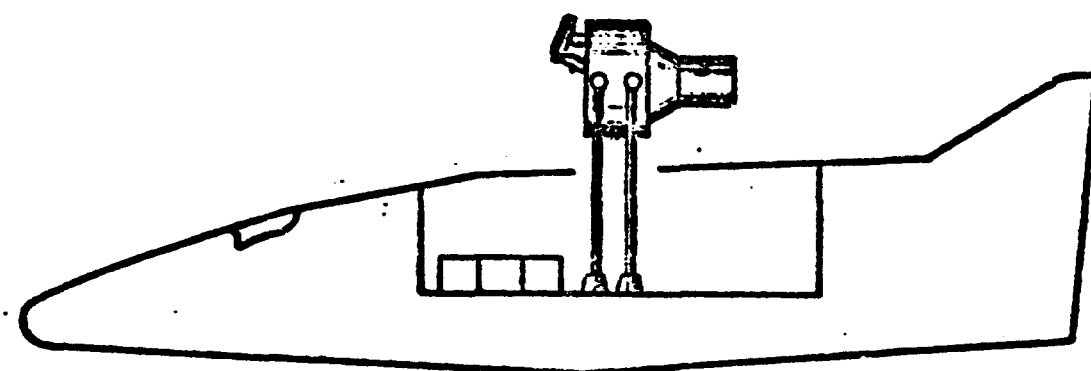
Fig. 4-9 In-Orbit Maintenance/Refurbishment Analyses



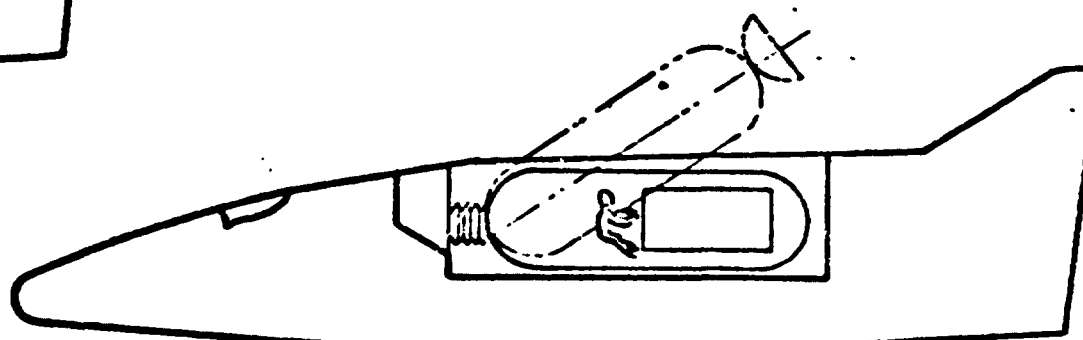
RESTRAINED EVA



NON-PRESSURIZED IVA



REMOTE CONTROLLED ROBOT



PRESSURIZED IVA

Fig. 4-10 In-Orbit Repair/Refurbishment Concepts

replacement. Modular design to accommodate this requirement is discussed in detail in Section 8.

4.2.3 Space Tug Operations with the Shuttle

The current NASA plans include use of a Space Tug to transport special payloads to and from syneq orbit and to boost planetary payloads to injection velocities. The Tug will be carried in the Shuttle cargo bay. In most instances, the payload will be pre-mounted to the Tug. Upon attaining low earth orbit position, the Tug and payload will be deployed from the Shuttle and the Tug remotely ignited.

The payloads will require adapter attachments to the Tug, and in the case of a recoverable payload, will also require a docking ring and special docking-stabilization features in the control subsystem.

4.2.4 Recalibration of Payload After Module Replacement

The payload module design must be carefully executed to allow replacement of equipment modules with minimum influence on the overall payload alignment and calibration. No readjustment nor recalibration should be necessary following an in-orbit repair activity.

4.2.5 Ground Base Response Time

The elapsed time (down time) between failure of an orbiting payload and the point at which a replacement can be put into orbit or an in-orbit repair or refurbishment executed is sometimes classified as critical. There will be an average response time and a critical response time in the Shuttle transportation system. Use of any mode other than average will probably dictate a dedicated and high-cost revisit of the Shuttle. Very careful consideration should be given in the payload system planning, therefore, to allow for the maximum down time and thereby reduce the costs of response to a random payload failure.

4.2.6 On-Orbit Checkout After Repair/Refurbishment

After any repair or refurbishment of the payload on orbit, a re-checkout must be performed to assure flight readiness. The payload design should be established to require a minimum number of mandatory verifications to be accomplished so that checkout can be accomplished in a reasonable elapsed time using a Shuttle-carried payload checkout set (discussed elsewhere in this document).

4.2.7 Payload Wearout/Reliability vs Refurbishment Cycle

Refurbishment cycles of one year (for an OAO-type payload) and two years (for an SEO-type) have been established for preliminary program costing purposes. As payload designs are initiated for future payloads, the hardware reliability characteristics should be traded-off with repair and refurbishment cycles to obtain a cost-effective balance for the specific design.

4.3 PAYLOAD MODULARIZATION FOR REPAIR/REFURBISHMENT

To obtain the large cost benefits of payload in-orbit repair and refurbishment, the payload design must incorporate provisions for easy replacement of equipment by a space crewman and the capability for verifying functional adequacy of the repaired or refurbished payload by a reasonably simple on-orbit checkout.

4.3.1 Groundrules for Payload Modular Design

Basic criteria have been established for the design of modules. They are listed on Fig. 4-11. The module weight and size must be compatible with the combined capability of the Shuttle-carried payload manipulators and the space crew handling the modules in zero-g. The maximum cost of a single module should not exceed about 5 to 10 percent of the recurring cost of the payload (as module cost increases, the cost of a single-module repair increases).

- **DIVIDE PAYLOAD SUBSYSTEMS INTO MINIMUM QUANTITY OF MODULES CONSISTENT WITH:**
 - **MAXIMUM WEIGHT/SIZE WHICH CAN BE READILY INSTALLED OR REMOVED**
 - **MAXIMUM COST OF A SINGLE MODULE**
- **SEGREGATE COMPONENTS WHICH HAVE HIGH PROBABILITY OF REPLACEMENT FROM THOSE WHICH HAVE HIGHER PREDICTED LIFE.**
- **ESTABLISH OPERATING TOLERANCES ON INDIVIDUAL MODULES SO THAT MODULE REPLACEMENT WILL NOT REQUIRE PAYLOAD RECALIBRATION.**
- **PROVIDE SIMPLE FUNCTIONAL AND MECHANICAL INTERFACES BETWEEN MODULES.**
- **PROVIDE FOR EASY ACCESS TO AND REMOVAL/INSTALLATION OF MODULES WITHOUT NEED FOR SPECIAL TOOLS.**

Fig. 4-11 Modular Design for Payload In-Orbit Maintenance/Refurbishment

4.3.2 Typical Compartmentization for Modules

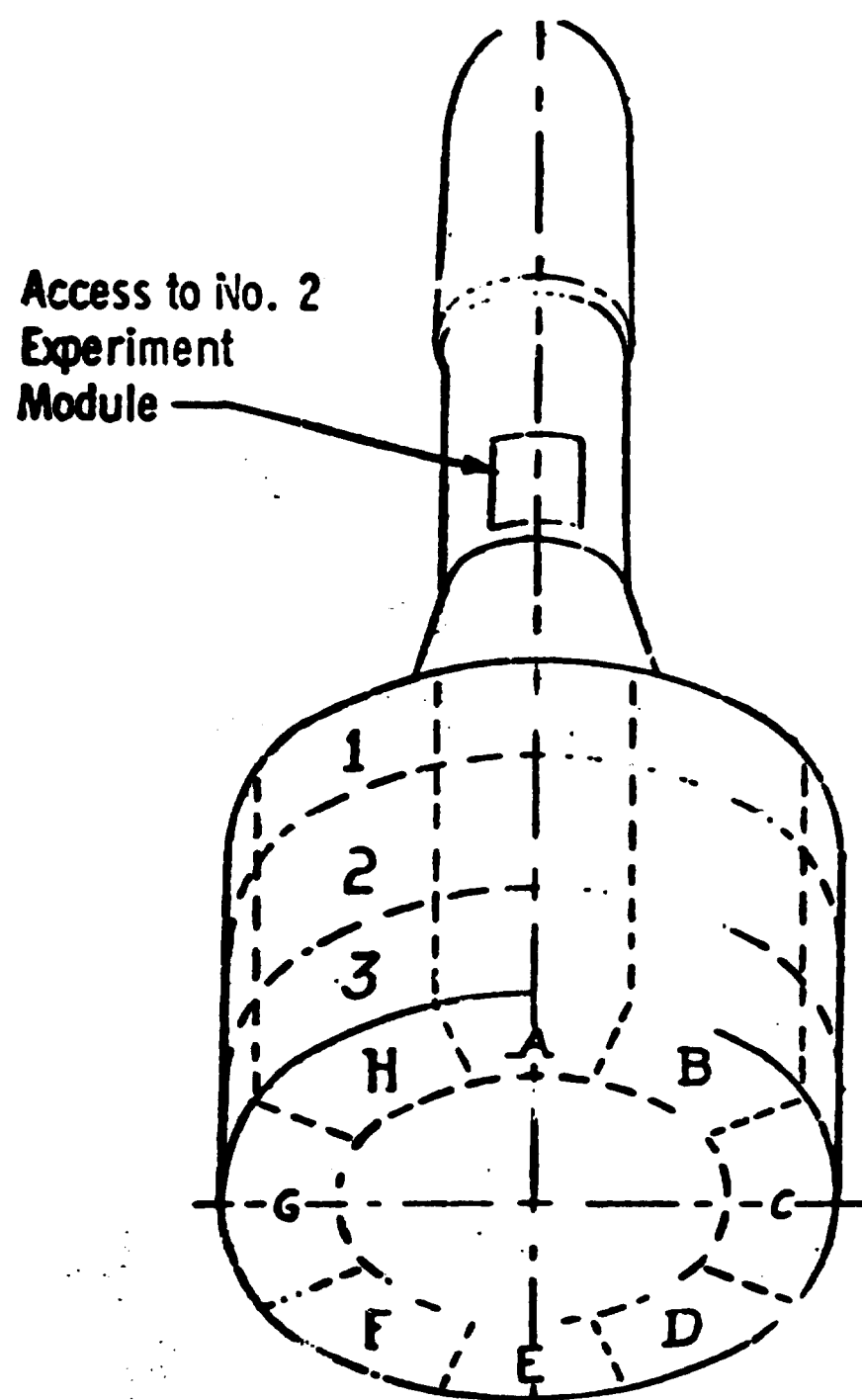
Two basic approaches can be utilized for module location and access for removal/installation:

- (1) Large-Diameter Payloads - On these payloads, which are 10 ft (3.05 m) to 15 ft (4.55 m) in diameter, the clearance between the cargo bay bulkheads of the Shuttle and the payload is too small for easy access. In these cases, consideration should be given to a central access tunnel with modules installed peripherally and module removal accomplished internally.
- (2) Smaller-Diameter Payloads - On these payloads, where external access within the Shuttle cargo bay is available, modules can be arranged to provide external removal.

Access into the central tunnel of the larger payload can be accomplished by a space crewman entering via a hatch in the forward bulkhead of the Shuttle cargo bay (the payload must be mounted with the central tunnel opening facing forward in the Shuttle). A typical module location arrangement for a large OAO-type payload is shown in Fig. 4-12. An arrangement of modules in a smaller SEO-type payload is shown on Fig. 4-13. The spacecraft structure is provided with guides and supports for the modules. Complete networks of module interconnecting electrical harnesses are installed into the structure and terminate with connectors at each module.

4.3.3 Typical Equipment Modules

A set of module-types comprising a low-cost SEO-type payload are illustrated on Fig. 4-14. A sample listing of the equipment comprising the modules of each subsystem of an SEO-type payload is shown on Figs. 4-15a, -15b, 15c. The configuration of a typical module is shown in Fig. 4-16. The general configuration of a typical compartment into which the module is installed is shown in Fig. 4-17.

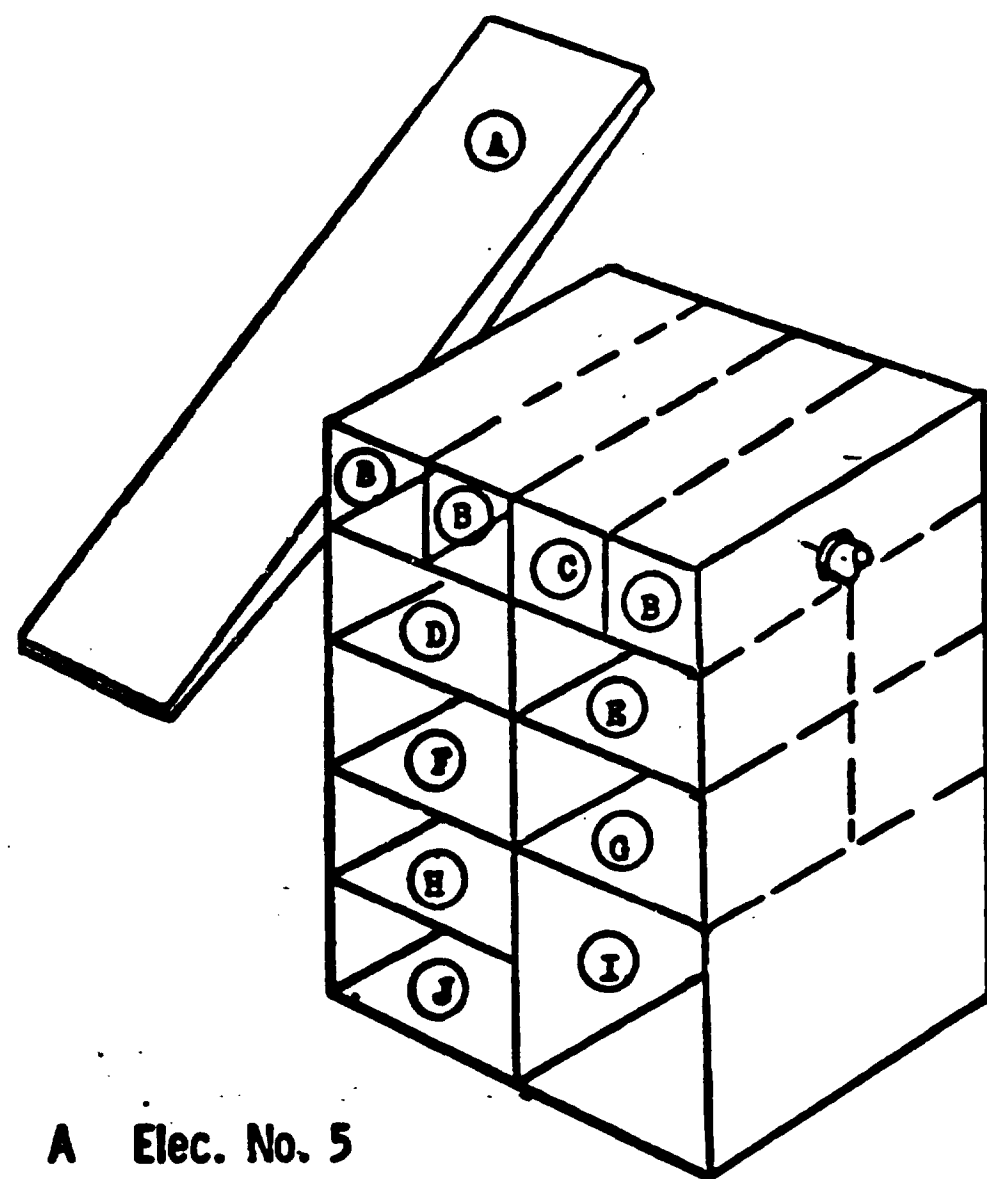


<u>Compartment</u>	<u>Module</u>
A-1	Empty
A-2	Electrical Power No. 2
A-3	Attitude Control No. 1
B-1	Empty
B-2	Electrical Power No. 1
B-3	Empty
C-1	Empty
C-2	Empty
C-3	Attitude Control No. 2
D-1	Stabilization & Control No. 1
D-2	Experiment No. 1
D-3	Empty
E-1	Stabilization & Control No. 3
E-2	CDPI*Data Distribution Unit**
E-3	Attitude Control No. 3
F-1	Stabilization & Control No. 2
F-2	CDPI* No. 1
F-3	Empty
G-1	Empty
G-2	CDPI No. 2
G-3	Attitude Control No. 4
H-1	Empty
H-2	Stabilization & Control No. 4
H-3	Empty

* Communications, Data Processing, & Instrumentation
 ** Fixed unit, not replaceable in orbit

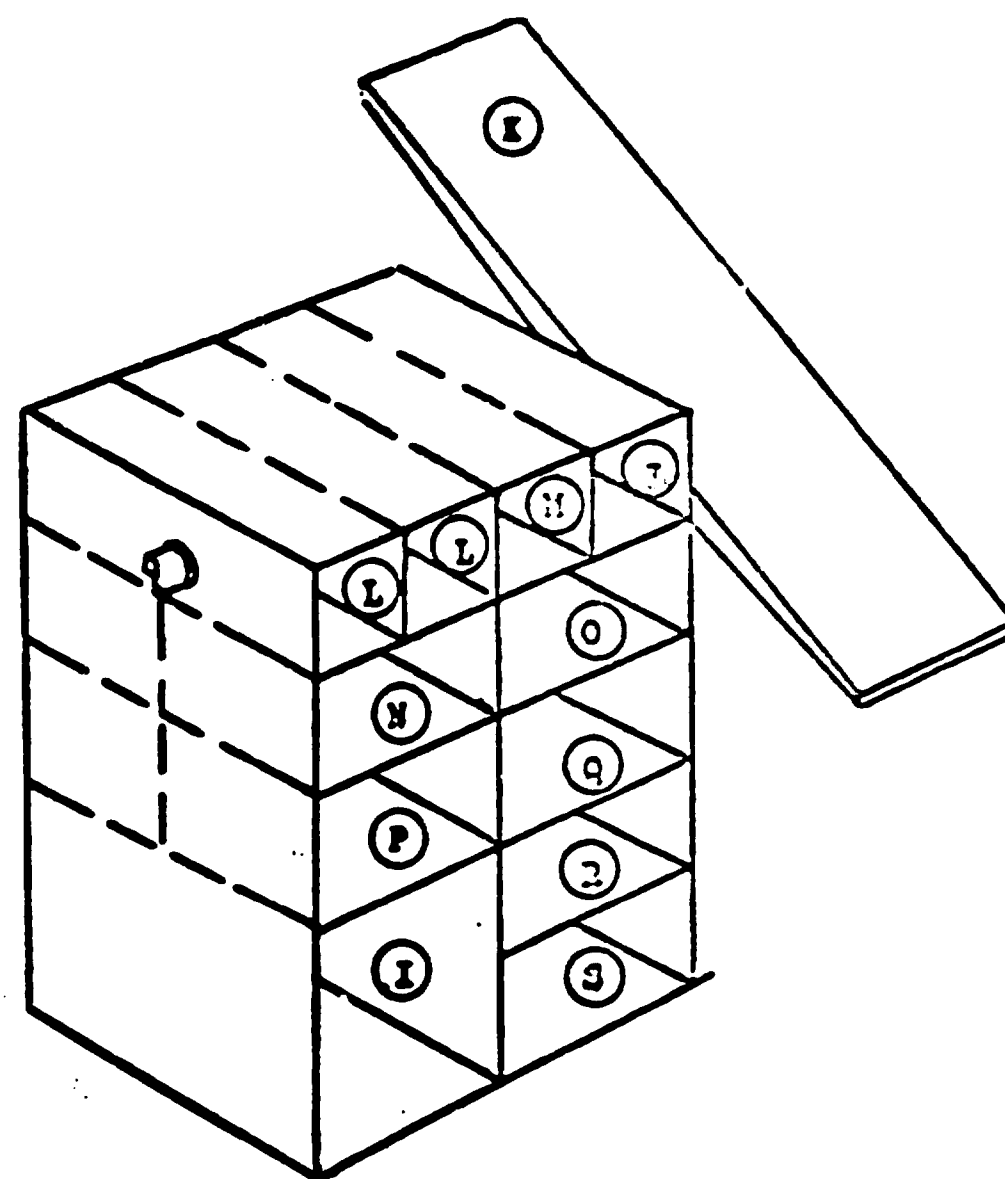
Fig. 4-12 Low-Cost OAO Module Locations

4-23



East Side

- A Elec. No. 5
- B Empty
- C Elec. No. 3
- D Elec. No. 1
- E CDPI No. 1
- F ACS No. 1
- G ACS No. 2
- H CDPI No. 3
- I Exper. No.1
- J Exper. No.2

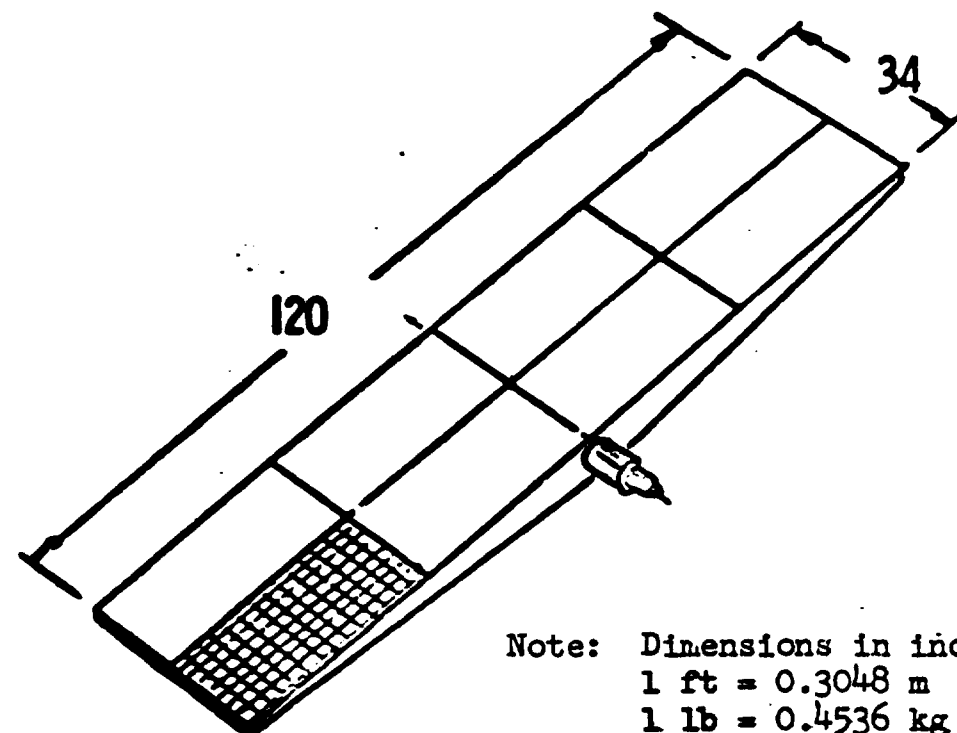


West Side

- K Elec. No. 6
- L Empty
- M Elec. No. 4
- N CDPI No. 2
- O Elec. No. 2
- P ACS No. 3
- Q ACS No. 4
- R S&C No. 1
- S S&C No. 2

Fig. 4-13 SEO Module Location & Arrangement

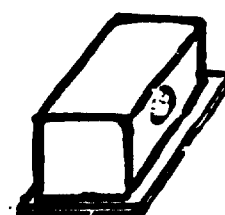
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SOLAR PADDLE MODULE (2) - 69 lb* each

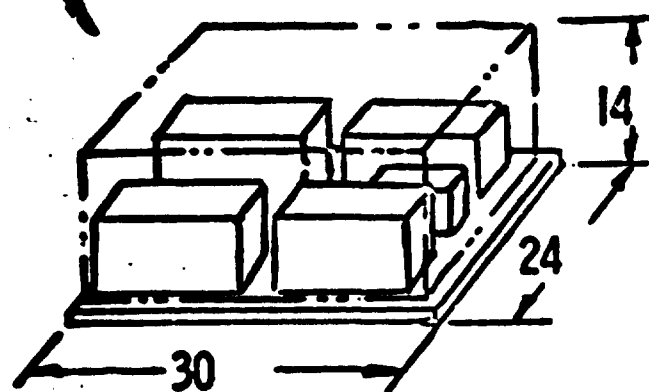
- 8 Panels, each 17 x 30; 672 cells per panel
- 2 x 2 cm solar cells, qty per paddle - 5376
- Solar cells are phosphorous diffused N/P silicon; .012 thick with .020 coverglass. (in.)
- Slipring on paddle shaft transfers array power and instrumentation to brush assy. on spacecraft

Note: Dimensions in inches
1 ft = 0.3048 m
1 lb = 0.4536 kg



Paddle Drive Module (2) 39 lb* each

Battery Module - 170 lb*



- Including 15% contingency

Power Control Module - 105 lb*

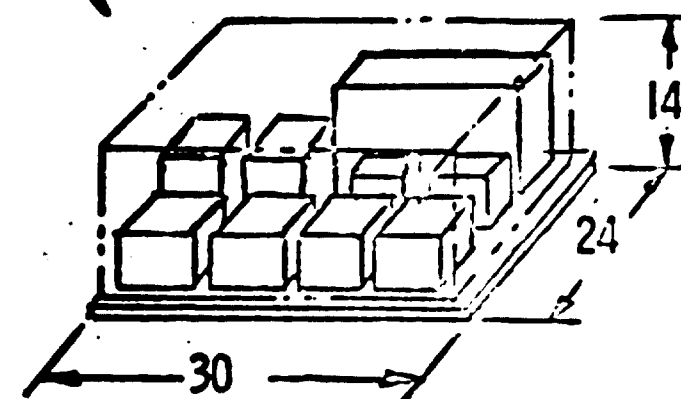


Fig. 4-14 Low-Cost SEO Electrical Subsystem

SUBSYSTEM	MODULE	EQUIPMENT IN MODULE	MODULE WEIGHT
Electrical	Battery No. 1	• NiCd Battery (4) - 12 Amp. Hr. • Current Shunt Assy. • Module Base/Cover • Internal Cables	Basic 154 lb 10% Cont. 16 Total 170 lb
Electrical	Power Control No. 2	• Battery Charge Controller (4) • State-of-Charge Unit • Power Distribution Unit • Ground Power Relay • Motor Controller - S/A Drive • Pulse Generator - S/A Drive • Module Base/Cover • Internal Cables	Basic 92 lb 15% Cont. 14 Total 106 lb
Electrical	Paddle Drive (2 Req'd) No. 3 No. 4	• Drive Motor (2) • Gear Box • Module Base/Cover • Internal Cables	Basic 34 lb 15% Cont. 5 Total 39 lb
Electrical	Solar Paddle (2 Req'd) No. 5 No. 6	• Solar Cell Panel (8) • Paddle Structure • Voltage Regulator (Diodes) • Internal Wiring	Basic 60 lb 15% Cont. 9 Total 69 lb

1 lb = 0.4536 kg

Fig. 4-15a Low-Cost SEO Modules (1 of 3)

SUBSYSTEM	MODULE	EQUIPMENT IN MODULE	MODULE WEIGHT
Communication, Data Processing, Instrumentation (CDPI)	Data Handling (2 Req'd) No. 1 No. 2	• Tape Recorder/Encoder • PCM Multiplexer/Encoder • Command Decoder • Module Base/Cover • Internal Cables	Basic 70 lb 15% Cont. 11 Total 81 lb
Communications, Data Processing, Instrumentation	Communications No. 3	• Transponder (2) • TWTA • Modulation Selector (2) • Module Base/Cover • Internal Cables	Basic 75 lb 15% Cont. 12 Total 87 lb
Stabilization & Control	Sensing & Flight Control Electronics No. 1	• Flight Control Electronics • Horizon Sensor (2) • Solar Aspect Sensor (2) • Module Base/Cover • Internal Cables	Basic 85 lb 15% Cont. 13 Total 98 lb
Stabilization & Control	Momentum No. 2	• Gimballed Momentum Wheel • Wheel Drive Electronics • Rate Gyro Package • Wheel Safety Shield • Module Base/Cover • Internal Cables	Basic 109 lb 15% Cont. 16 Total 125 lb

1 lb = 0.4536 kg

Fig. 4-15b Low-Cost SEO Modules (2 of 3)

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SUBSYSTEM	MODULE	EQUIPMENT IN MODULE	MODULE WEIGHT
Attitude Control	Attitude Control (4 req'd) No. 1 No. 2 No. 3 No. 4	• Tank • Valves: Fill, Relief, Solenoid Latching • Press. Regulator • Thrust Valve Cluster with 4 Nozzles • Plumbing • Transducers • Internal Support • Module Base/Cover • Internal Cabling	Basic 125 lb 15% Cont. 18 Total 143 lb
Experiment	Photographic No. 1	• Camera • Processor/Dryer • Optical-Mech. Scanner (2) • Film Handling Assy. • Electronic Assy. • Nitrogen System • Film Supply • Bimat Supply • Internal Structure • Case • Internal Cabling	Basic 365 lb 15% Cont. 54 Total 419 lb
Experiment	Vidicom Camera No. 2	• Vidicon Camera Assy (2) • Module Base/Cover • Internal Cables	Basic 75 lb 15% Cont. 11 Total 86 lb

1 lb = 0.4536 kg

Fig. 4-15c Low-Cost SEO Modules
(3 of 3)

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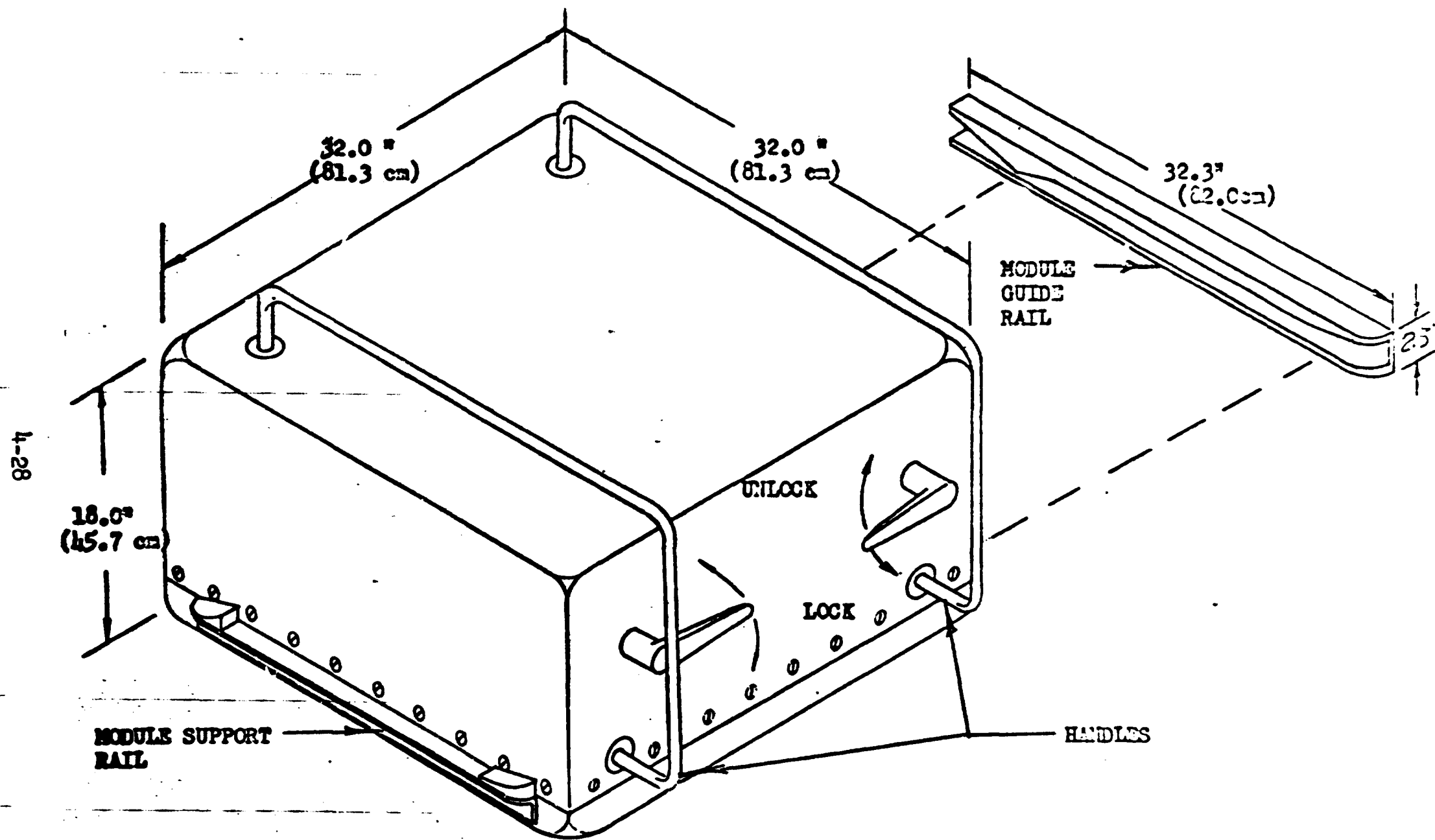
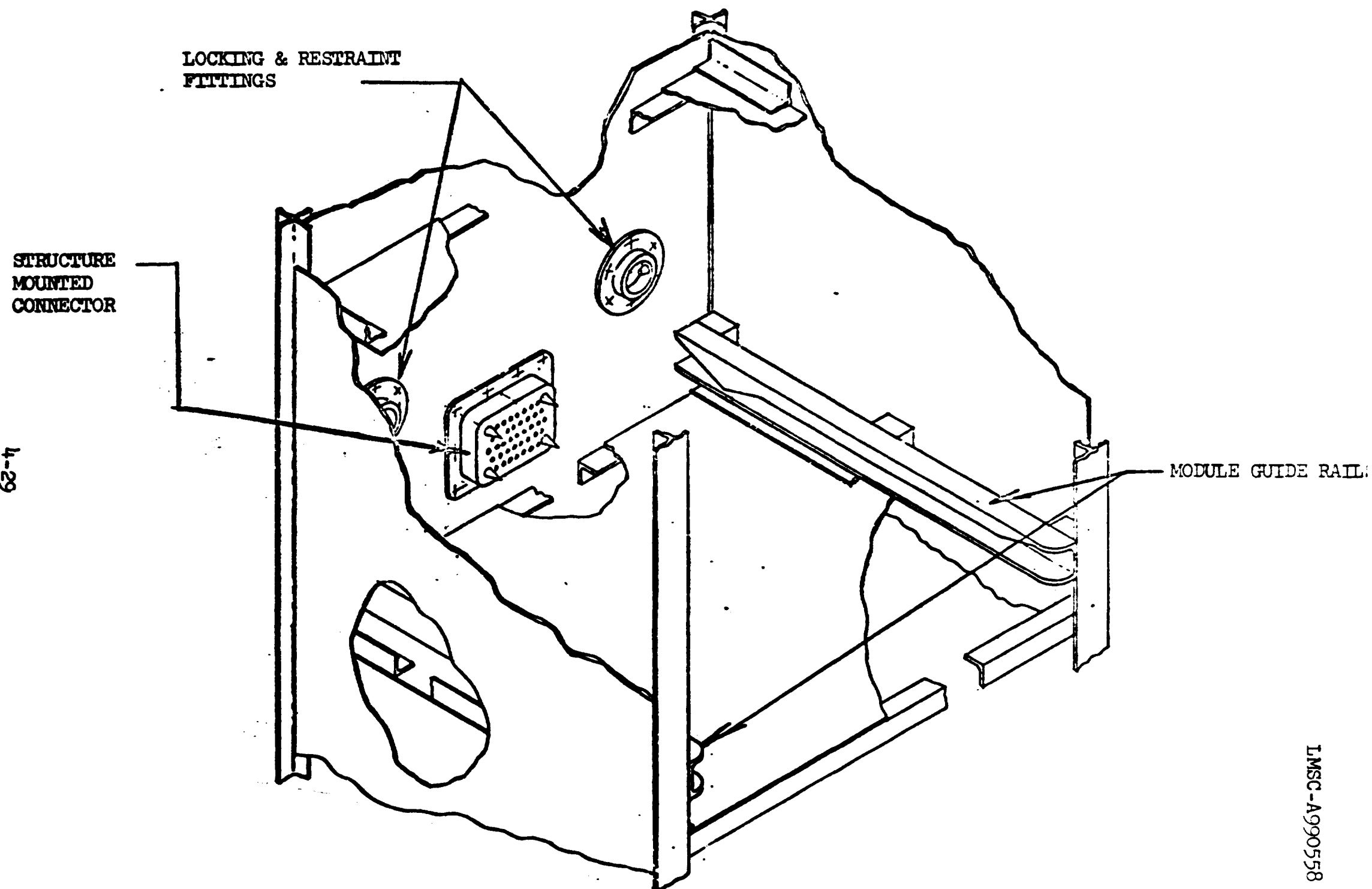


Fig. 4-16 Typical Spacecraft Module

4-29



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Fig. 4-17 Typical Spacecraft Equipment Module Compartment

Section 5

SPACE SHUTTLE SYSTEM INTERFACES AND EFFECTS ON PAYLOAD DESIGN

This section has been deleted and will be superseded by a separate document being issued by NASA: "SPACE SHUTTLE/PAYLOAD INTERFACE DOCUMENT."

Section 6

COST REDUCTION APPROACHES TO PAYLOAD SYSTEM ENGINEERING

6.1 MANDATORY SYSTEM PERFORMANCE AND DESIGN REQUIREMENTS

Over-specification has been a significant factor in escalating the costs of space programs, and it begins with the System Performance and Design Requirements Specification prepared by the government program office and/or the contractor. Cost consciousness must be fostered among the scientists, engineers and managers responsible for program planning, with the objective of obtaining requirements specifications based on cost/value analysis.

6.2 SIMPLIFIED ENGINEERING DOCUMENTATION

Much of the escalation of the costs of space programs is attributable to the documentation requirements. Program planners should carefully evaluate documentation needs, and impose only those requirements essential to the orderly execution of the program. In particular, they should avoid the imposition of requirements simply because it is safer to require too much and avoid the risk of requiring too little.

When contractual documentation requirements permit, simplification of contractor engineering documentation can result in significant savings.

6.3 SIMPLIFIED EQUIPMENT SPECIFICATIONS

Engineering organizations also prepare design specifications that are more restrictive than they need to be, because it is safer (but more costly) to err in that direction. Program offices must encourage engineers to specify lower equipment performance and other design requirements, and effective review procedures must be established to control over-specification of requirements for both in-house fabricated and procured equipment.

6.4 REDUCED TESTING REQUIREMENTS

Typically, spacecraft equipment and assembled spacecraft have been subjected to very comprehensive, rigorous, and often repetitive testing programs to establish the level of confidence felt to be necessary for the making of launch decisions. Such programs have been costly to execute and often costly in wornout or damaged equipment. The fear of spacecraft failure during launch or early in orbital flight and of the consequences of such a catastrophe has been the principal motivator for traditional testing programs. Much of this fear will be alleviated by the Space Shuttle which makes possible in-orbit checkout of spacecraft before they are committed to orbit, monitoring of their early orbital performance, and their recovery for repair in orbit or their return to earth in the event of early malfunction. Once fears of the consequences of failure have been alleviated it should be possible to reduce significantly the scope and cost of testing programs.

In addition, load testing of the structures and mechanisms of Shuttle-launched spacecraft may be significantly reduced since they may be designed without weight constraints and with high factors of safety.

6.5 SPECIAL RELIABILITY CONSIDERATIONS

6.5.1 Cost/Reliability Relationship

It is well known that the cost of raising the predicted reliability of space vehicle systems and subsystems above nominal values increases exponentially as the ultimate reliability of 100 percent is approached. Part selection becomes more rigorous, reliability testing becomes more comprehensive, and more and more redundant components and backup functions are required. For example, a typical cost estimating relationship (CER) showing the relationships between cost and reliability for a typical space payload subsystem is presented in Fig. 6-1.

The exponential increase of cost as reliability is increased beyond 0.75 (75 percent) is most significant. If the reliability required of complex subsystems

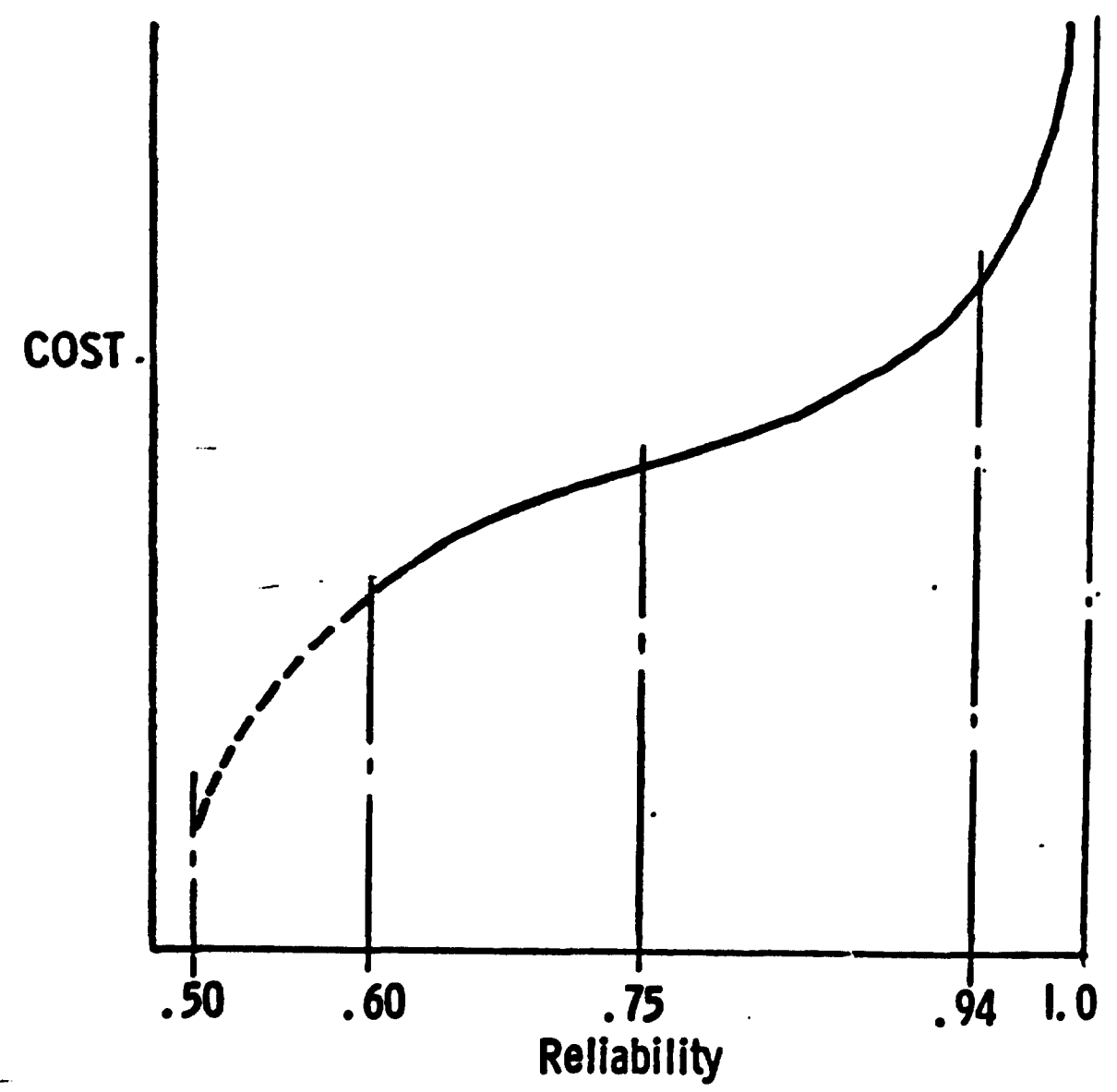


Fig. 6-1 Cost/Reliability Relationship for Typical Space Payload Subsystem

can be limited to about 0.7, appreciable savings can be realized in the design, development, and production of space payloads. Such limitation of reliability requirements will be possible for Shuttle-launched payloads because of the capability of the Shuttle to checkout payloads in orbit prior to deployment, to repair them in orbit, and to recover them for reuse.

6.5.2 Use of MIL-Spec or Aircraft Parts

One cost-saving change from historical approaches to payload design made possible by the lowered reliability requirements of Shuttle-launched payloads may be the use of MIL-Spec and possibly aircraft parts instead of Hi-Rel parts. During the LMSC Payload Effects study, investigations were made of the relative costs of various grades of parts and of the practicability of using lower grade, and hence lower cost, parts in future space payloads. These investigations were limited in scope because of task priority agreements with NASA, and further study is required to establish the validity of using lower-grade parts in payload design. However, cost-saving potential exists and application of the lower-grade parts should be considered if short operating-life or refurbishment cycle is desirable. The four basic grades of parts and their relative costs are defined in Fig. 6-2. A sample list of comparative costs for specific parts is shown in Fig. 6-3.

The failure rate of MIL-Spec parts is about two or three times that of Hi-Rel parts. Also, the failure rate of aircraft parts is about ten times that of Hi-Rel parts; but more importantly, the parametric drift rate of aircraft parts is about four times the Hi-Rel part drift rate.

In general, MIL-Spec parts are comparable in weight to Hi-Rel parts; equivalent aircraft parts are significantly heavier as indicated by the sample listing of Fig. 6-4. The failure rates of aircraft and MIL-Spec parts are compared in Figs. 6-4 and 6-5.

6.5.2.1 Application of MIL-Spec Parts. A preliminary analysis was made of the application of MIL-Spec parts to an SEO-type payload. It was determined

HI - RELIABILITY - COST = \$x

PRODUCED ON SPECIAL PRODUCTION LINES IN MOST CASES; MANY PARTS 100% TESTED FOR PERFORMANCE; ALL PARTS SCREENED AND BURNED IN; LIFE OF 2-3 YEARS WITH GOOD PARAMETRIC STABILITY IS POSSIBLE; FAILURE RATES VERY WELL DOCUMENTED

MIL-SPEC - COST = \$0.3x TO \$0.5x

SAMPLE SELECTED FROM PRODUCTION RUN FOR TEST TO SPEC. REQTS. -
SAMPLE PASSING TEST WILL ACCEPT TOTAL LOT; FAILURE RATE 2x TO 3x
HI-REL; APPROX. 70% OF HI-REL PARAMETRIC STABILITY

AIRCRAFT - COST = \$0.15x TO \$0.2x

PARTS TESTED TO PARAMETRIC NORMS; SAMPLE FROM EACH LOT SUBJECTED TO
SIMULATED STRESSES AND SHOCKS EXPECTED IN AIRCRAFT FLIGHT -
SAMPLE FAILURE REJECTS TOTAL LOT; FAILURE RATE ABOUT 10x HI-REL PARTS;
DRIFT RATE ABOUT 4x HI-REL

COMMERCIAL - COST = \$0.05x TO \$0.1x

MANUFACTURED IN VERY LARGE QUANTITIES; SUBJECTED TO SIMPLE GO-NO-GO
TESTS, SAMPLING TECHNIQUES TO ACCEPT/REJECT LOTS; NO FAILURE RATE DATA
EXCEPT BY LOTS REJECTED

Fig. 6-2 Grades of Parts

PART OR COMPONENT	COST (APPROXIMATE \$ PER UNIT)			
	Commercial	Aircraft	Mil-Spec	Hi-Rel
Pressure Transducer	1.65	4.00	10.00	65.00
Transistor	0.25	0.75	2.50	7.50
Resistor	0.17	0.60	2.20	4.90
Coax Cable RG 58U	0.14/ft.	0.35/ft	1.15/ft	2.40/ft
Capacitor (Ceramic cap)	0.24	0.47	1.60	4.85
Connector (16 pin)	0.60	1.35	3.15	8.00
Integrated Circuit (and Gate and Paramp)	1.15	2.75	7.00	16.60

Fig. 6-3 Comparison of Cost for Different Grades of Parts

	Manufacturer	AIRCRAFT TYPE		HIGH-RELIABILITY	
		Failure Rate (per 10 ⁶ hr.)	Weight (lb)	Failure Rate (per 10 ⁶ hr.)	Weight (lb)
DC-DC Converter	Westinghouse	460	38	25	12
Press. Reg. System (N ₂)	AiResearch	240	48	16	7
Transceiver	Collins	28	18	5	8
IRU	M-H	60 (747)	28	23	18
Tape Recorder	HP	66	72	11	18

1 lb = 0.4536 kg

Fig. 6-4 Weight and Failure Rate Comparison - Aircraft and Hi-Rel Parts

	MANUFACTURER	FAILURE RATE (per 10^6 hrs.)	
		Aircraft	High-Reliability
Rate Gyro	Kearfott	75	5
Silicon Transistor	TI 2N22A	14	0.3
Oper. Amplifier	Fairchild MA-709	54	0.2
Latching Relay	James Pond Clark	240	24
Solenoid Valve (N_2)	Sterer	16	1.7
Voltage Regulator	Collins	21	0.4
Battery NiCd	Yardney	30	0.6
Pressure Transducer	GE	175	7

Fig. 6-5 Aircraft vs High Reliability Part Failure Rate Comparison

to be feasible to use MIL-Spec parts in lieu of Hi-Rel parts if the design operating life of the SEO (the time between refurbishments) could be limited to nine months.

6.5.5.2 Application of Aircraft Parts. Also, a preliminary analysis has been made of the application of aircraft parts to an OAO-type payload, and it was determined that the operating life of the payload would be reduced to four months (from a one-year design life with Hi-Rel parts) because of the relatively high parametric drift rates of aircraft parts. Additional analysis is necessary before firm conclusions can be reached concerning the application of MIL-Spec or aircraft parts to space payloads; however, the potential for cost saving is significant and warrants further study.

6.5.3 Predicted vs Achieved Reliability

Justification for lower system reliability requirements for Shuttle-launched payloads is afforded by analysis of the historical disparity between the predicted reliability of expendable-launched payloads and the reliability they have actually achieved in orbit. (Examples are given in Fig. 6-6). This disparity is evidence that the reliability requirements for expendable-launched payloads have been conservative. The in-orbit payload checkout capability and the payload recovery capability of the Shuttle will permit the establishment of less conservative reliability requirements for Shuttle-launched payloads. Lower reliability requirements will reduce design, development, testing, and production costs of payloads.

6.5.4 Adversity Factors

Another factor that has contributed to the conservative reliability requirements imposed on expendable-launched payloads is the Adversity factor, a multiplier applied to failure rates to account for the effects of environment. Adversity factors defined by MIL-STD-217A are shown in Fig. 6-7.

Program	Predicted Reliability	Achieved Reliability
LMSC Small Research Satellite (SRS)	< 0.80	0.90
HEOS-A, German	0.85	0.98
Lunar Orbiter	0.75	0.90
Mariner	0.75	0.90

Fig. 6-6 Predicted Reliability vs Achieved Reliability

ENVIRONMENTAL EXPOSUREADVERSITY
FACTOR

LAUNCH OPERATIONS -----	90	▲	2
AIRCRAFT OPERATIONS -----	18		
ROAD TRANSPORT -----	15		
RAIL TRANSPORT -----	5		
SHIPBOARD OPERATIONS -----	3		
SUBMARINE OPERATIONS -----	2		
SPACE OPERATIONS -----	1.5		

▲ Some Contractors/Agencies believe launch stress should be 900 - 1500; others use as low as 4. LMSC uses 10 max. for solid-state hardware, 35 max. for other.

2 This factor is not time dependent.

6-7 Adversity Factors for Payload Reliability Prediction
(Data from MIL-HDBK-217A)

The launch environments to which payloads are exposed by the Space Shuttle are less severe than those experienced by expendable-launched payloads. Therefore, the adversity factor for Shuttle launch should be lower and, as such, contribute to lower payload costs. The Shuttle adversity factor can approach 18 for launch/ascent and, with corrective action of in-orbit checkout following launch/ascent, can further approach the 1.5 factor used for "space operations".

Section 7
GUIDELINES FOR LOW-COST PAYLOAD DESIGN

This section contains a summarized set of design approaches and recommendations which are directed toward achievement of cost-effective payloads. Sub-section 7.1 deals with the general aspects of material and equipment selection, low-density packaging, and modularization of equipment. Sub-sections 7.2 and subsequent contain data pertinent to the specific subsystems of a payload:

• Structures and Mechanisms	. 7.2
• Experiment Installations	7.3
• Stabilization & Control	7.4
• Communications, Data Processing, & Instrumentation	7.5
• Electrical Power	7.6
• Attitude Control	7.7
• Environmental Control	7.8

7.1 GENERAL GUIDELINES

7.1.1 Low-Cost Materials

In general, for comparable applications, materials that are inexpensive to buy and fabricate, such as steel and aluminum, are heavy; and light materials, for example titanium and beryllium, are expensive. In the past, payload weight limitations have prevented the free use of heavy materials and simple methods of fabrication; and the costs of spacecraft have been higher than they would have been without weight limitations. In designing spacecraft that will be orbited and supported by the Space Shuttle, full advantage should be taken of the great payload capability of the Shuttle, and low-cost materials and fab-

7.1.2 Pre-Qualified Equipment

Most space programs have made some attempt to use pre-qualified equipment to obtain the obvious cost savings. However, existing qualified equipment has often been considered and rejected because it was not optimized functionally, or in size and weight, for the contemplated application.

The relaxation of weight, volume, and power constraints on spacecraft design will permit designers to consider a much wider range of qualified equipment for a given application, and to select equipment that historically would have been rejected as over-designed, oversize, or overweight. Much more emphasis must be placed on avoiding RDT&E costs in future space programs.

7.1.3 Standardized Equipment

The development and qualification of standardized equipment for spacecraft promises major overall savings for the Nation's space program. There has been very little standardization of equipment in the past because the emphasis has been on the optimization of equipment function, size, and weight for the requirements of each space program. When such optimization is no longer required because of the elimination of constraints on payload weight and volume, the development of standardized equipment can proceed with assurance that it will be widely used. There should be established: (1) design requirements for commonly-used types of spacecraft equipment, (2) common industry sources for such equipment, and (3) specifications which require that where possible the selection of equipment for new spacecraft shall be from among the available standard equipment.

7.1.4 Low-Density Packaging of Electronic Equipment

Electronic equipment for space programs has evolved in the direction of reduced size and weight, and increased packaging density. In general, this evolution has been accompanied by cost escalation. The relaxation of weight and volume

constraints in the Shuttle era should permit the development of simpler, lower density packaging techniques for electronic equipment; resulting in the reduction of RDT&E and recurring costs of such equipment.

Low density electronic packaging saves labor costs in design, modification, fabrication and assembly, repair, and inspection. Additional study is required to determine quantitatively the magnitude of cost savings possible through low density packaging.

7.1.5 Modularization of Equipment Installations

In the design of spacecraft for launch by the Space Shuttle, equipment installations should be modularized to permit in-orbit removal and replacement. The capability to repair a spacecraft in the Shuttle cargo bay before releasing it is a major factor in the reduction of space program costs that will be realized through the use of the Shuttle. The following guidelines are applicable to the design of equipment modules for Shuttle payloads:

- Divide payload subsystems into minimum quantity of modules consistent with:
 - Maximum weight/size which can be readily installed or removed by a shuttle crewman
 - Maximum cost of a single module
- Segregate components which have high probability of replacement from those which have higher predicted life.
- Establish operating tolerances on individual modules so that module replacement will not require payload recalibration.
- Provide simple functional and mechanical interfaces between modules.
- Provide for easy access to and removal/installation of modules without need for special tools.

Modularization of equipment installations also contributes to cost savings in spacecraft ground operations prior to launch.

7.2 STRUCTURES AND MECHANISMS SUBSYSTEM DESIGN GUIDELINES

7.2.1 Basic Recommended Design Approaches

Historically, the design of spacecraft has been severely constrained by weight and volume limitations resulting from the modest payload capability of launch vehicles.

When weight and volume constraints are relaxed or eliminated, significant reductions in the cost of spacecraft structures and mechanisms can be made by using the following guidelines in design:

- (1) Select a simple spacecraft configuration that requires only a simple structure.
- (2) Provide volume for low-density equipment installations to simplify installation design and to insure complete accessibility of equipment.
- (3) Use high factors of safety for sizing structural elements to reduce design and analysis efforts and to reduce or eliminate static loading testing.
- (4) Increase dimensional tolerances to simplify tooling, and to reduce fabrication, assembly and quality control efforts.
- (5) Simplify structural elements and minimize the use of contour-machined parts to reduce design, analysis, fabrication and assembly efforts.

- (6) Use commercially available grades and sizes of aluminum sheet and extrusions for most structural elements.
- (7) Minimize the use of sandwich materials.
- (8) Do not use beryllium, composite materials or other high-cost materials.
- (9) Avoid use of higher-cost parts such as forgings and castings.
- (10) Eliminate deployment mechanisms whenever the launch vehicle payload envelope permits fixed installation of solar panels, antennas, sensors and other equipment.
- (11) Avoid sophistication and miniaturization of mechanisms.

7.2.2 Examples of Low-Cost Designs of Structures and Mechanisms

7.2.2.1 Orbiting Astronomical Observatory (OAO).

a. Initial-Design OAO-B Structures & Mechanisms. The actual OAO-B structures and mechanisms are shown in Fig. 7-1. The basic structure is an octagonal cylinder approximately 7 ft (2.1 m) wide and 10 ft (3.0 m) high. It includes a central tube 48 in. (1.2 m) in diameter and 118 in. (3.0 m) long in which the experiment package is installed. The eight primary trusses are machined and chem-milled from aluminum plate. The forward and aft bulkheads and the equipment shelves form 48 equipment bays. Secondary structure is provided in each bay for the support of one or more items of equipment. Hinged doors provide access to the installed equipment.

b. Low-Cost Design - OAO Structures & Mechanisms. The general configuration of the equivalent low-cost design of the OAO is shown in Fig. 7-2 and the primary structure of the equipment section in Fig. 7-3. The equipment section is cylindrical, 13 ft (4.0 m) in diameter and 8 ft (2.4 m) long. Twenty-four

7-6

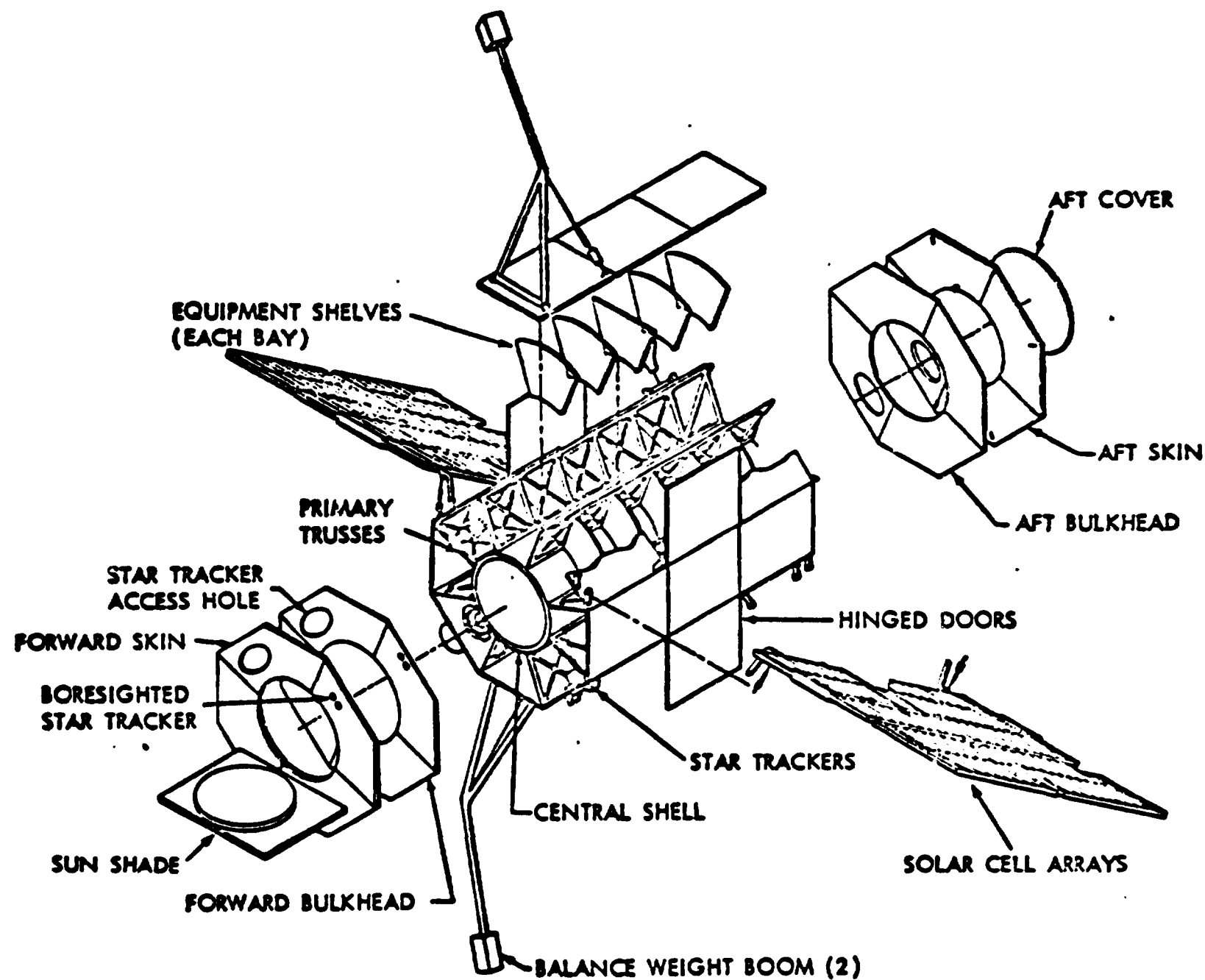


Fig. 7-1 OAO-B Structures and Mechanisms

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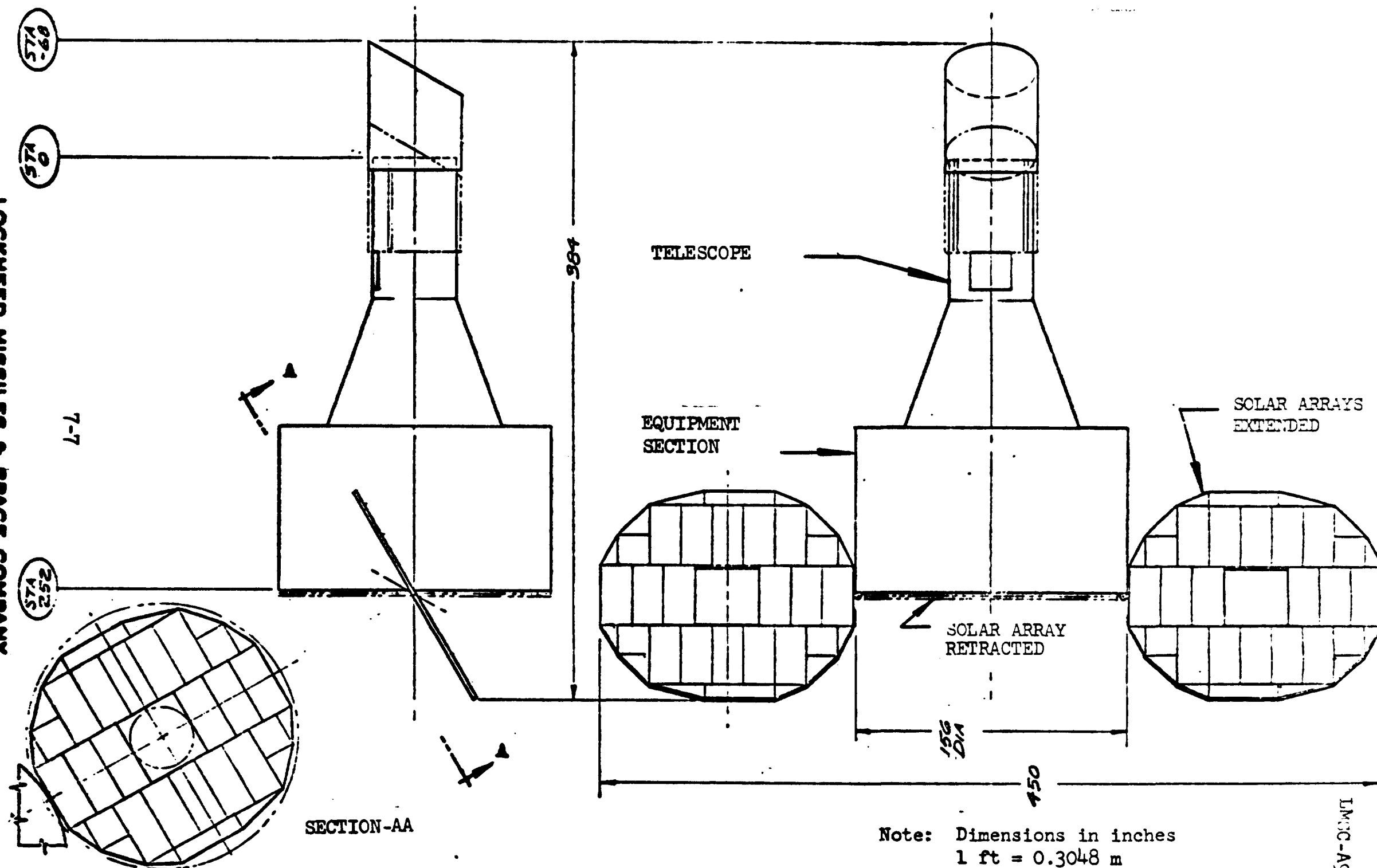


Fig. 7-2 Low-Cost OAO Configuration

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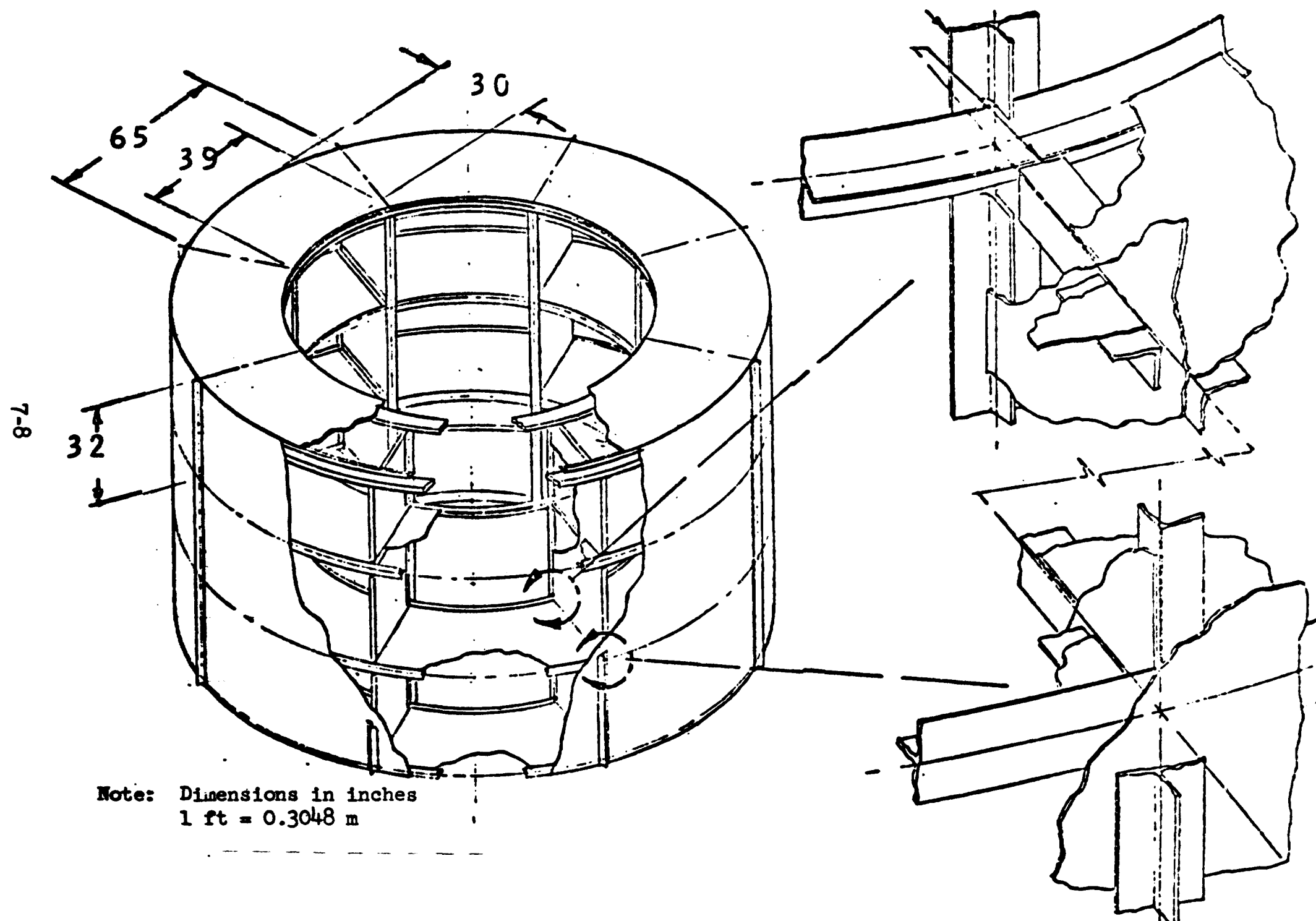


Fig. 7-3 Low-Cost OAO Equipment Section Structure

equipment compartments surround an 8 ft (2.4 m) diameter by 8 ft (2.4 m) long empty volume in which ground technicians or Space Shuttle crewmen may work, installing or removing equipment modules from the equipment compartments. Modules are retained by mechanisms that are designed to be operated by a space-suited Shuttle crewman.

The equipment section structure is a riveted assembly consisting primarily of commercially available aluminum sheet and extrusions. Factors of safety of three or greater are used to reduce design and analysis efforts and to reduce or eliminate static load testing. The forward bulkhead of the equipment section is the reference plane for alignments in the low-cost OAO, and machined surfaces are provided for the attachment of the experiment package and for indexing equipment modules containing attitude reference equipment such as star trackers and inertial reference units.

The aft bulkhead is of sheet metal construction. Access to the work space inside the equipment section is through a 4-ft (1.2 m) square door.

The low-cost OAO equipment section structure is simpler and includes many less piece parts than the OAO-B equipment section structure, low-cost materials and high factors of safety are used, and analysis and static load testing are minimized. For these reasons the cost of developing and manufacturing the low-cost OAO structure was found to be significantly lower than the corresponding costs of the initial-design OAO-B. Figure 7-4 shows a tabulation of comparative costs.

7.2.2.2 Synchronous Equatorial Orbiter.

a. Initial Design - SEO Structures & Mechanisms. An SEO-type payload is shown in Fig. 7-5. The primary structure of this spacecraft consists of three decks interconnected by truss assemblies. Like the structure of the Lunar Orbiter, from which it was derived, it was optimized to minimize weight and was therefore costly to design, develop, qualify and produce.

Spacecraft	Initial Design-Subsystem Costs (millions of dollars)		Low-Cost Subsystem Costs (millions of dollars)	
	RDT&E	Average Unit	RDT&E	Average Unit
OA0	9.04	5.10	6.69	0.95
SEO	6.9	1.2	6.0	0.6
SRS	1.28	0.12	1.09	0.11

Fig. 7-4 Cost Comparisons - Structure & Mechanisms Subsystem

7-11

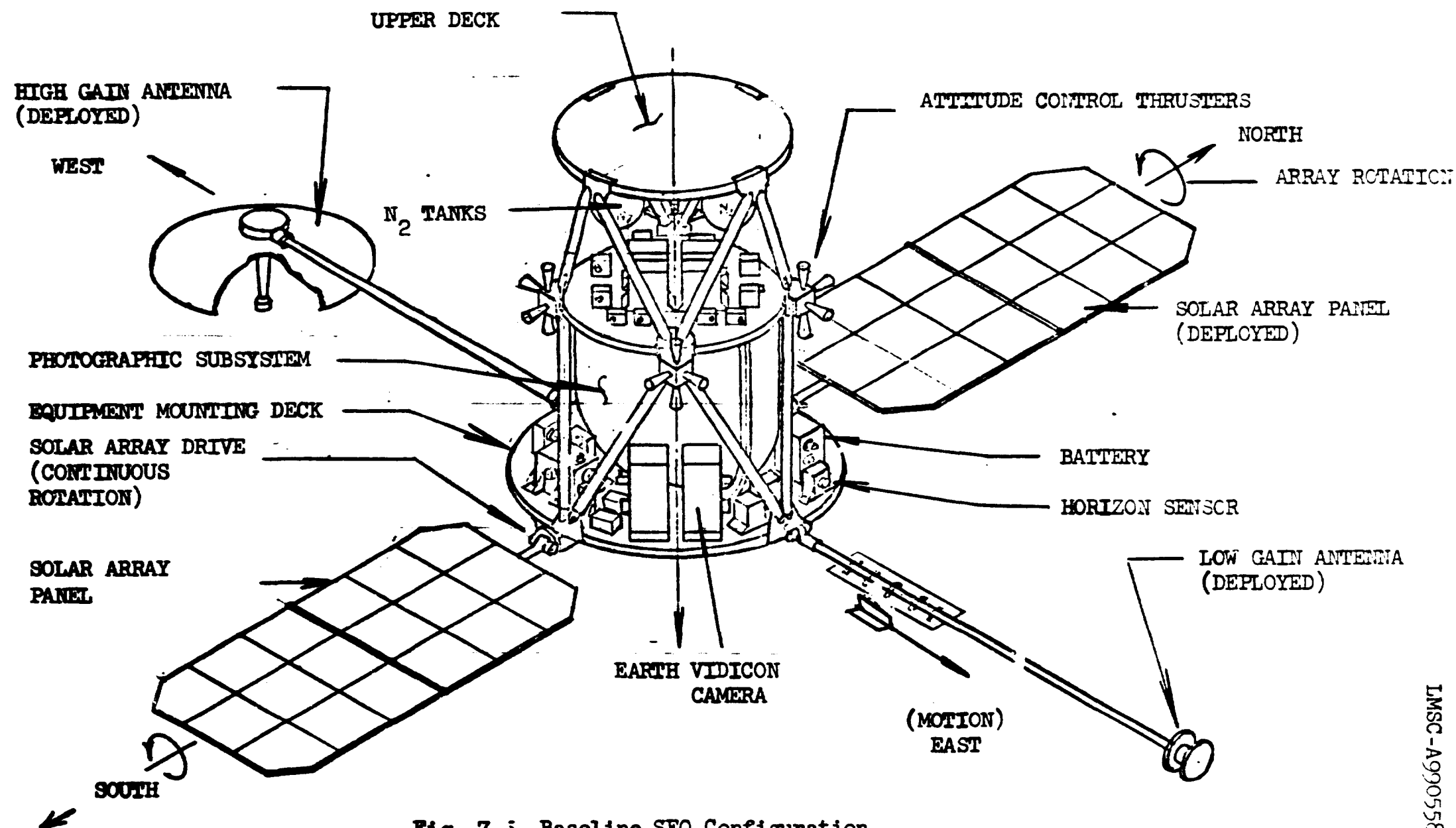


Fig. 7-5 Baseline SEO Configuration

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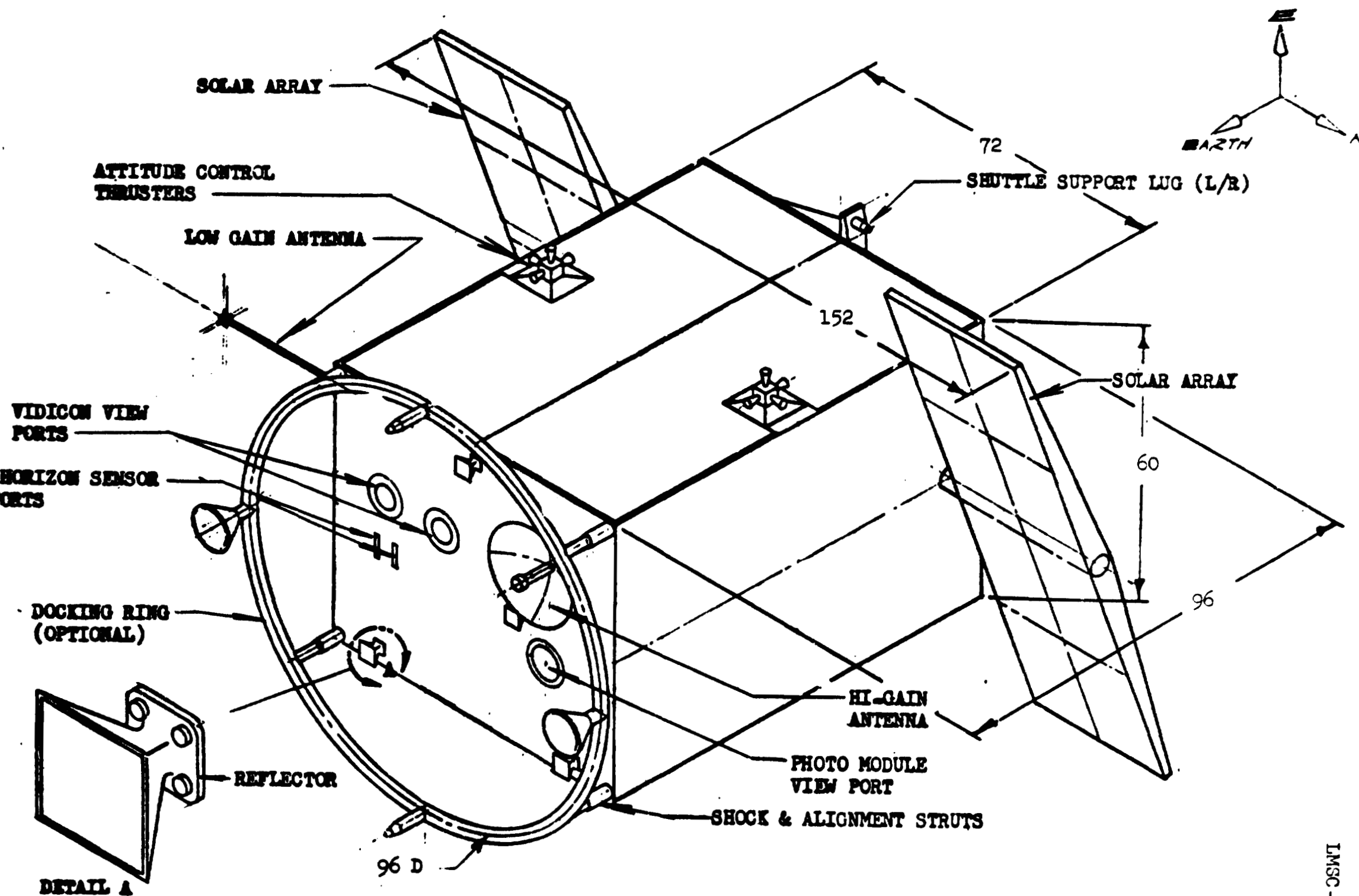
b. Low-Cost SEO Structures & Mechanisms. The general configuration of the equivalent low-cost design of the SEO is shown in Fig. 7-6 and the low-cost SEO structure in Fig. 7-7. The structure is a rectangular box divided into compartments in which readily-removable equipment modules are installed. The modules are designed to be removed and replaced by a space-suited Shuttle crewman. The structure is a riveted assembly made up primarily of aluminum sheet and extrusions. Stainless steel angles provide thermal isolation of the top and bottom panels, and stainless steel hinges thermally isolate the doors. Factors of safety of three or greater are used to reduce design and analysis efforts and to reduce or eliminate static load testing.

The design guidelines listed in paragraph 7.2.1 were followed in the design of the low-cost SEO structure and mechanisms subsystem and its estimated costs as shown in previous Fig. 7-4, particularly the average unit costs, are significantly lower than the costs of the initial-design SEO.

7.2.2.3 Small Research Satellite. The initial design SRS spacecraft was a modification of the early P-11 spacecraft shown in Fig. 7-8. The major difference in configuration between the baseline SRS and the P-11 is the addition of an identical solar module frame to each of the six frames shown.

a. Initial-Design SRS Structures & Mechanisms. The Structures and Mechanisms subsystem of the SRS included the basic structures shown in Fig. 7-9 and the solar module frames, all of which were optimized for minimum weight and, therefore, relatively costly to design, develop, qualify and manufacture.

b. Low-Cost SRS Structures & Mechanisms. The general configuration of the low-cost SRS is shown in Fig. 7-10 and the low-cost SRS structure in Fig. 7-11. Four deep beams made up of aluminum sheet and extrusions form an "eggcrate" assembly in which subsystem equipment may be installed. This assembly is strengthened by heavy tee extrusions connecting the outboard mid-points of the deep beams, and by angles connecting the outboard corners of the beams. Shear strength is provided by a heavy shear panel at the mid-plane of the structure, and by bulkheads that close the ends of the four triangular compartments of



Dimensions in inches
1 ft = 0.3048 m

Fig. 7-6 Low-Cost SEO General Configuration

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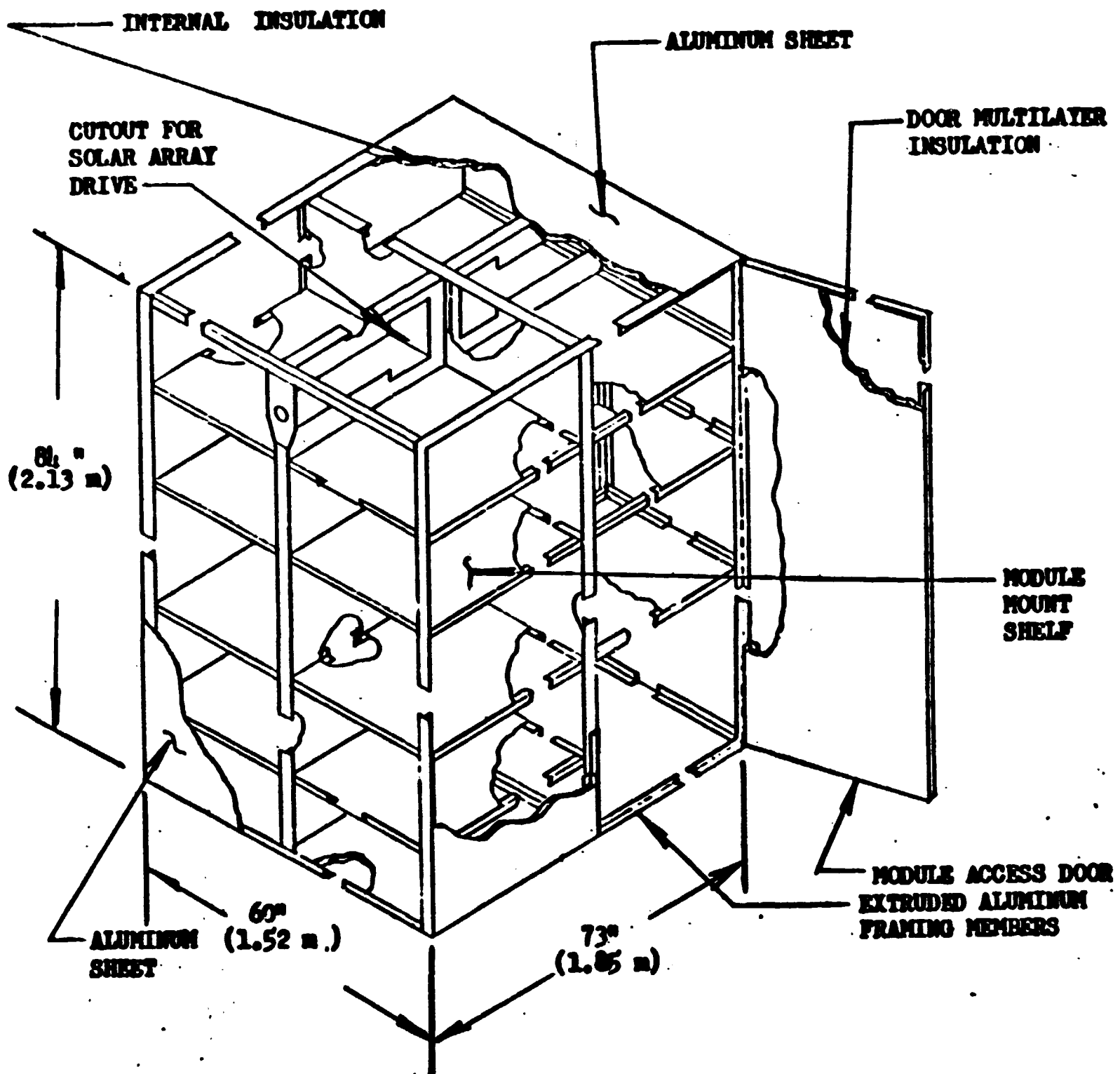


Fig. 7-7 Low-Cost SEO Structure

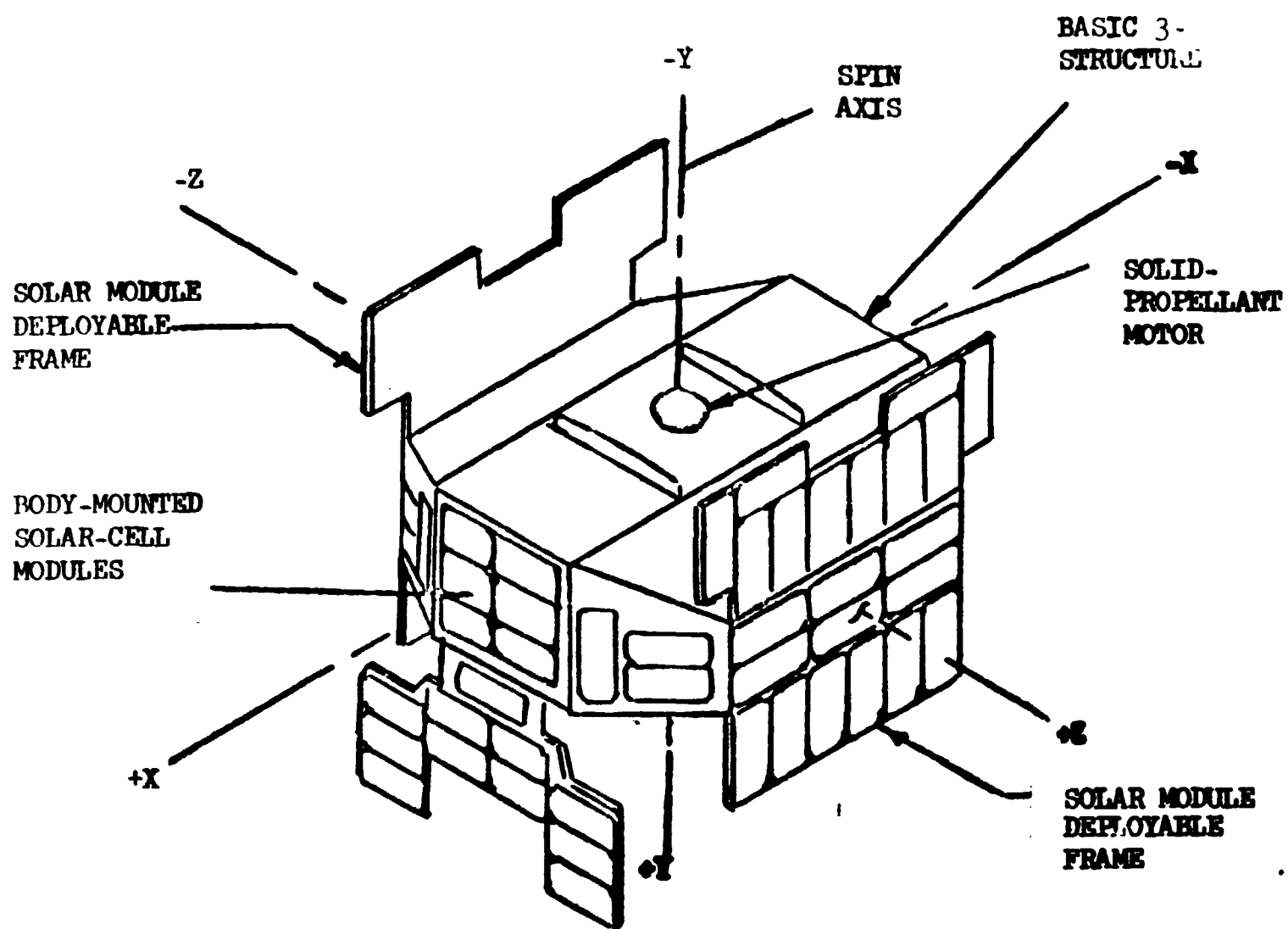
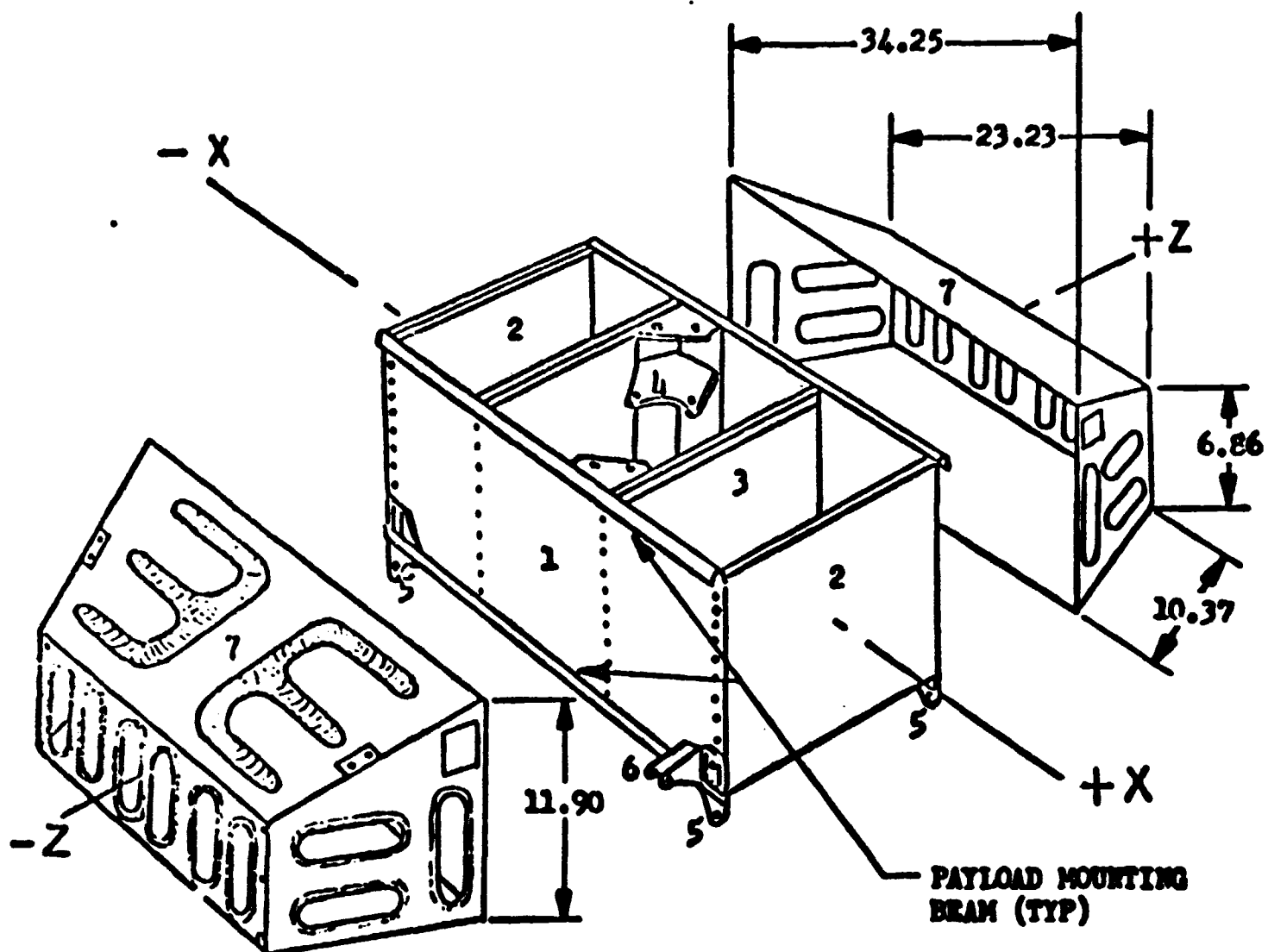


Fig. 7-8 Initial-Design SRS

Note: Dimensions in inches
1 ft = 0.3048 m



- | | |
|----------------------------|----------------------------------|
| 1. Main Beam Assembly | 5. Mounting Lugs |
| 2. End Panel Assembly | 6. Spin Nozzle Brackets |
| 3. Interior Panel Assembly | 7. Wing structure (Payload Area) |
| 4. Motor Mount | |

Fig. 7-9 Basic SRS Structure

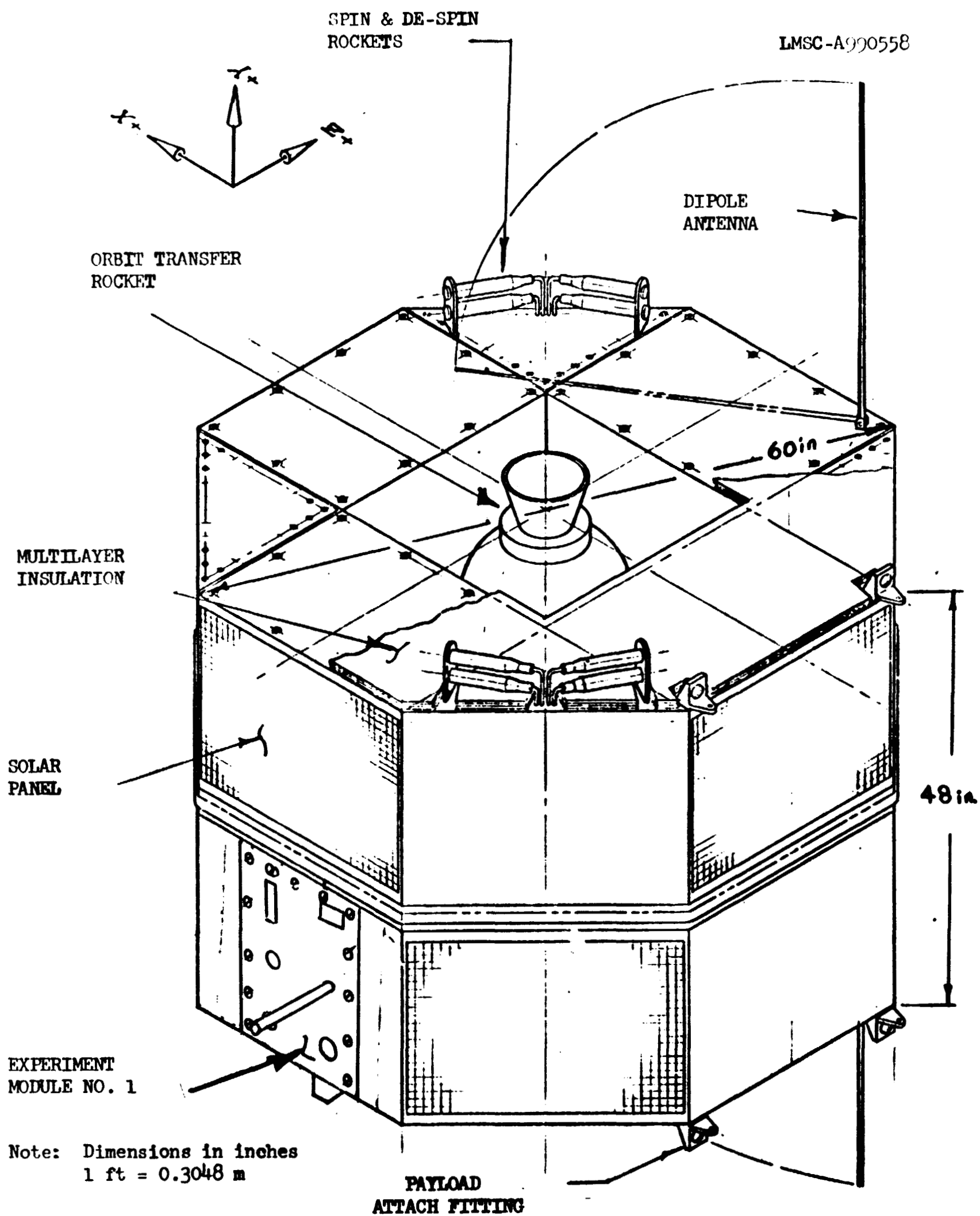


Fig. 7-10 General Configuration - Low-Cost SRS

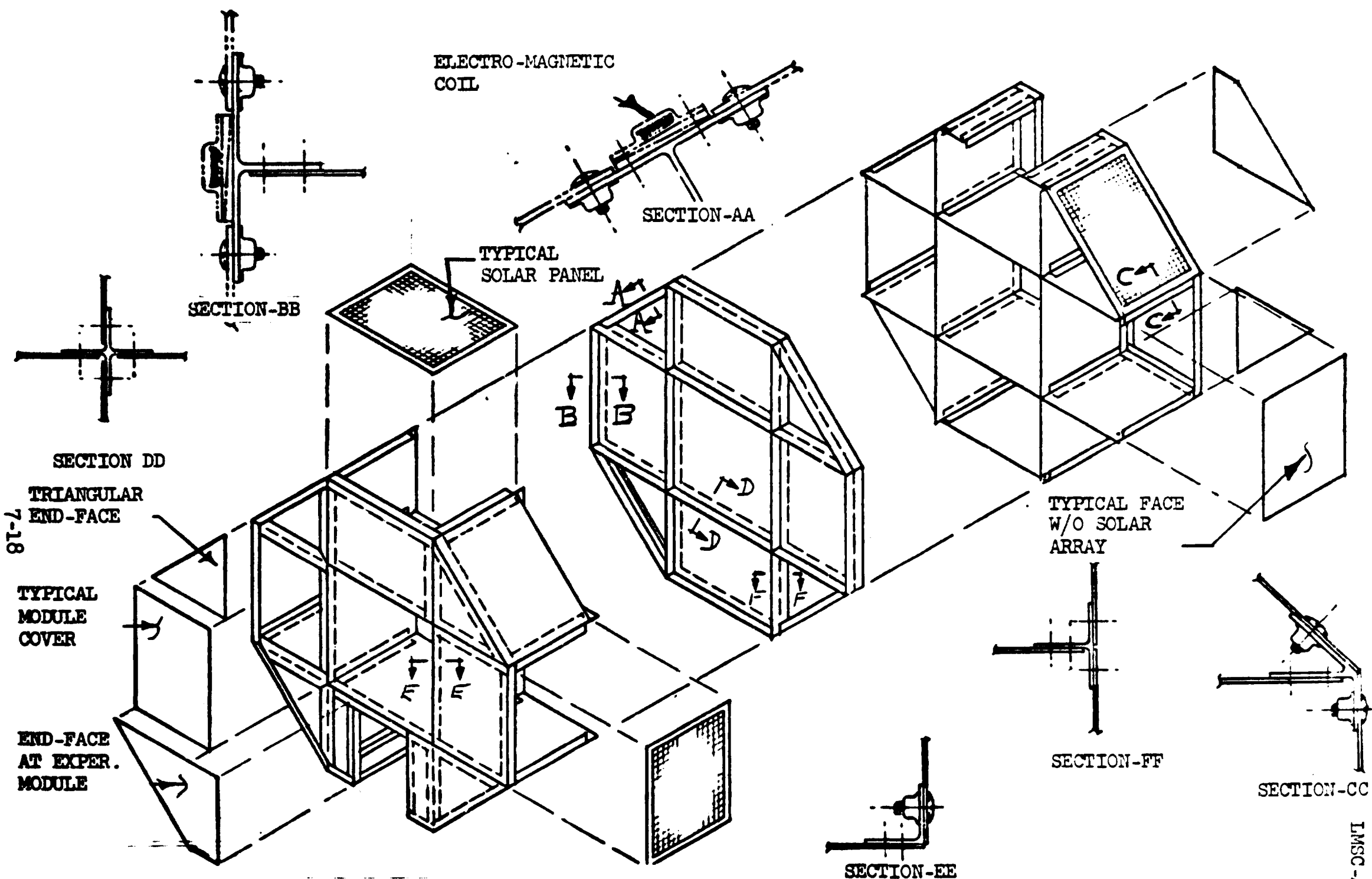


Fig. 7-11 Low-Cost SRS Structure

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"operate". Each of the eight plane surfaces of the octagonal primary structure assembly provides for the attachment of two solar panels or skin panels. As configured for the HiGlo mission the SRS has eight solar panels, three skin panels incorporating view ports for sensors, and five plain skin panels. The +Y and -Y faces of the SRS structure are closed by aluminum skins and multi-layer insulation blankets. The structure assembly includes secondary structure for the attachment of equipment modules, rocket motors, antennas, and fittings for handling and deployment of the SRS. The SRS structure has been designed with generous provisions for augmentation of the baseline HiGlo mission. Volume is available for the addition of experiment and subsystem equipment including a second propulsion rocket. Also, the average power available for subsystem and experiment equipment can be increased from 50 watts to approximately 100 watts by the installation of additional solar cell panels. The strength of the structure, based on factors of safety of three or greater, permits the addition of equipment without extensive analysis and redesign.

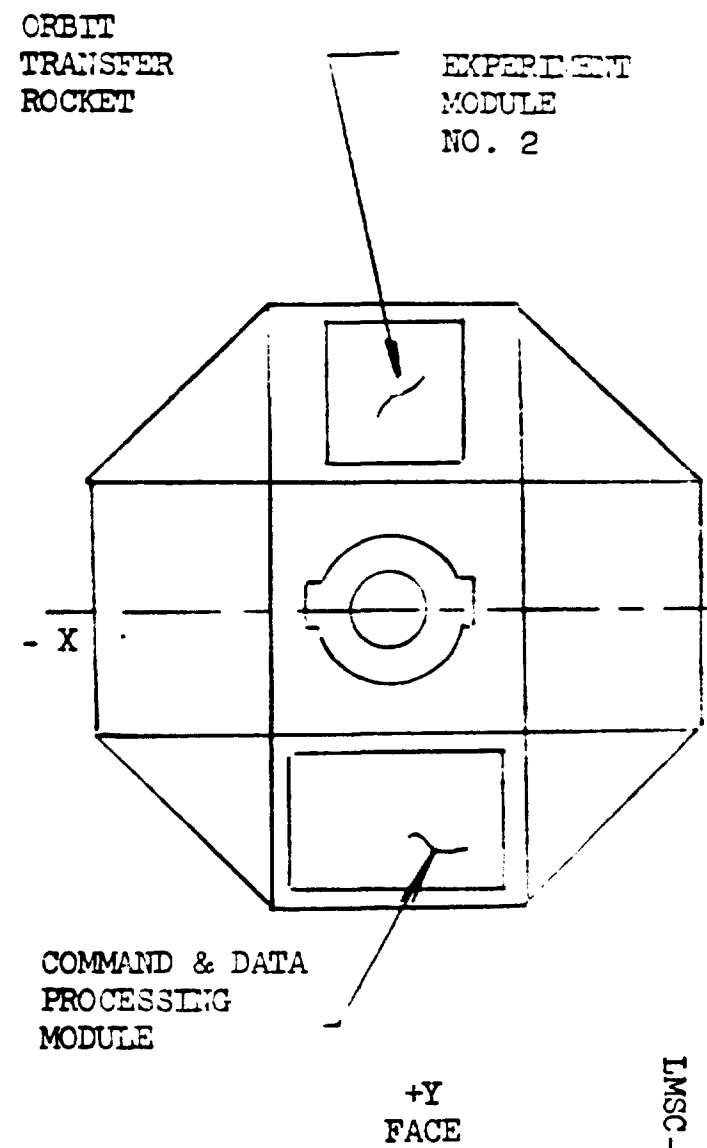
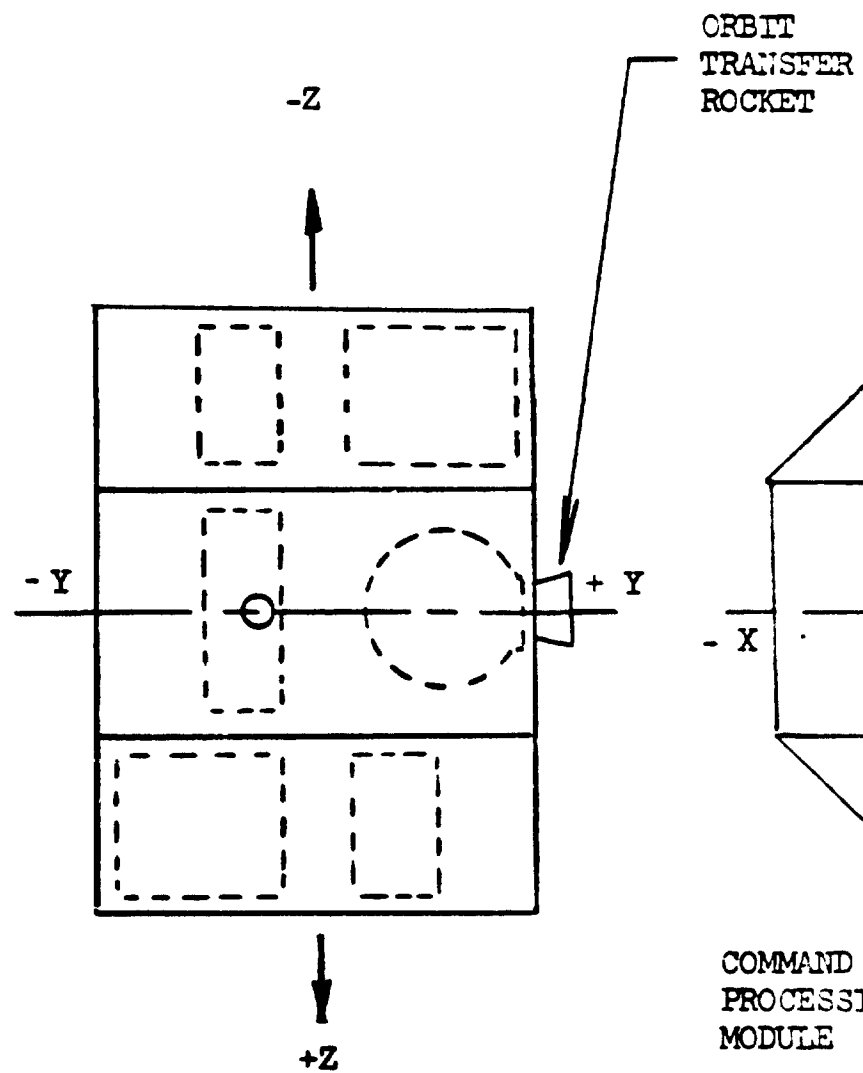
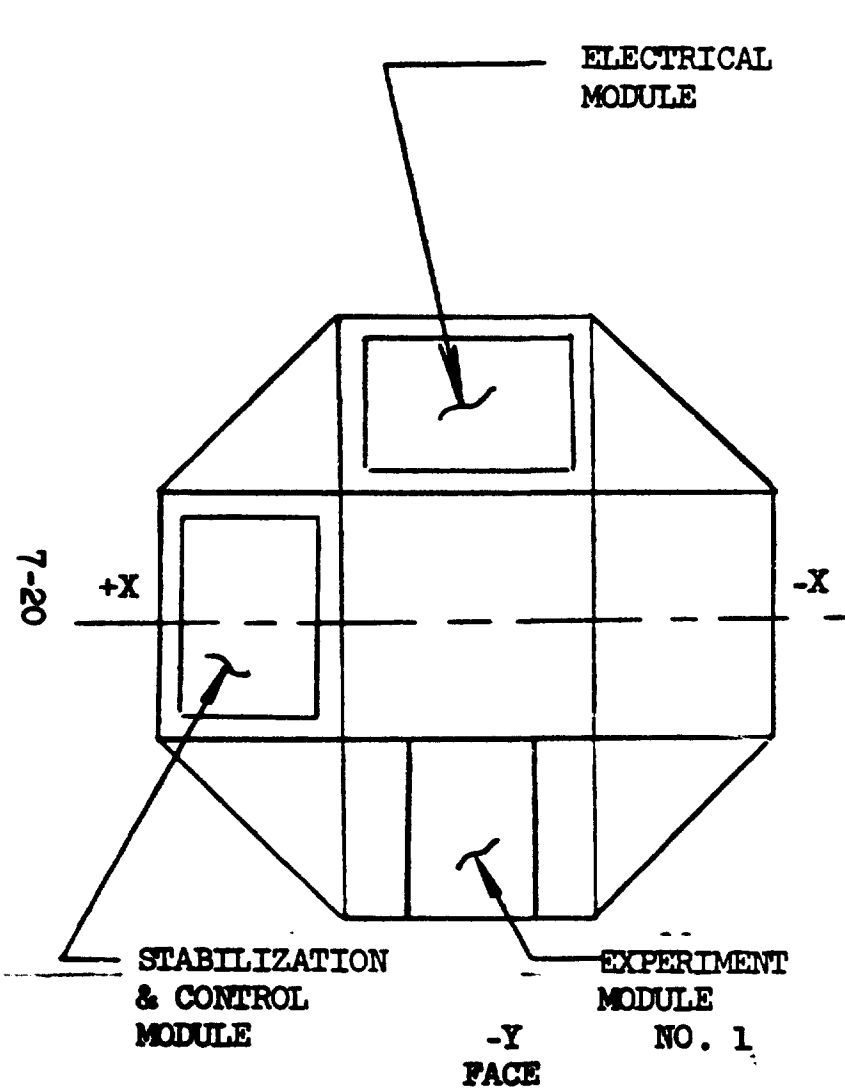
Most of the equipment of the HiGlo SRS is packaged in readily removable modules to reduce development and test costs. The location of modules in the SRS is shown in Fig. 7-12.

The design guidelines of paragraph 7.2.1 were used in the design of the low-cost SRS structure and mechanisms subsystem, but in addition, significant provisions were made for the growth of other subsystems. As a result, the cost comparison of previous Fig. 7-4 does not show cost reductions comparable to those made in the design of the low-cost OAO and SEO structures and mechanisms subsystems.

7.3 EXPERIMENT INSTALLATION DESIGN GUIDELINES

7.3.1 Basic Recommended Design Approaches

Costs of Experiment Installations can be reduced by using the following approaches in design:



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- (1) Select simple experiment package configurations, taking full advantage of the greater payload capability of new launch vehicles.
- (2) Apply Structures and Mechanisms Subsystem design guidelines (paragraph 7.2.1) to experiment-package structures and mechanisms.
- (3) When experiment thermal control requirements differ significantly from other spacecraft subsystem requirements, isolate the experiment to simplify thermal control of both the experiment and the spacecraft.
- (4) Design for in-orbit maintenance of experiment installations by modularization of equipment.
- (5) Design low-density experiment installations with provisions for additions of or changes to experiment equipment.
- (6) Avoid mechanisms that are not self-supporting in one-g.
- (7) Design low-density electronic packages to reduce design, development and manufacturing costs.
- (8) Eliminate need for in-flight adjustments or alignments
- (9) Avoid use of high cost material such as beryllium for weight reduction
- (10) Eliminate close-tolerance machining for weight reduction
- (11) Avoid miniaturization for weight and volume reduction

7.3.2 Examples of Low-Cost Designs of Experiment Installations

7.3.2.1 Orbiting Astronomical Observatory (OAO). The design of a low-cost Experiment Package for an OAO-type payload has been developed. The design is derived from the Goddard Experiment Package (GEP) of the OAO-B shown in Fig. 7-13.

a. Initial-Design OAO-B Experiment Installation. The GEP was designed to be installed within the central tube of the OAO-B spacecraft, and its thermal control requirements constrained and added to the complexity and cost of the thermal control design of the spacecraft. Conversely, the GEP design was complicated by the thermal environment provided by the spacecraft. Weight constraints imposed on the design of the GEP led to the use of beryllium optical mirrors that were more costly than mirrors of conventional optical materials. The optical properties of the beryllium mirrors were unstable in one-g, and optical calibration procedures were consequently more complex and costly than they would have been if rigid but heavier mirrors had been used. In-orbit focus control of the GEP was required to compensate for calibration uncertainty due to thermal effects and the instability of the beryllium mirrors.

b. Low-Cost OAO Experiment Installation. The general configuration of a low-cost Experiment Package for the OAO is shown in Fig. 7-14; additional details are shown in Figs. 7-15, 7-16, 7-17, and 7-18. The basic optical design of the GEP is unchanged. However, all the optical elements are fabricated from a conventional optical material, CerVit (a fused silica material) which has a very low coefficient of thermal expansion. All the mirrors and the grating are made from solid CerVit blanks having a diameter-to-thickness ratio of approximately six to insure adequate stiffness in a gravity environment and to facilitate fabrication and calibration.

The low-cost Experiment Package structure is an aluminum shell stiffened by aluminum rings and longerons. All material is commercially available aluminum. Factors of safety of three or greater are used in the design of the structure (a) to insure the stiffness required to preserve the alignment of the optical elements, (b) to reduce the design and analysis effort, and (c) to eliminate requirements for static load testing.

The primary mirror support is shown in Fig. 7-15. The support frame is a weldment of extruded aluminum channel designed for direct attachment to six of the internal longeron members of the shell structure. Three tangential mirror supports provide for mirror alignment.

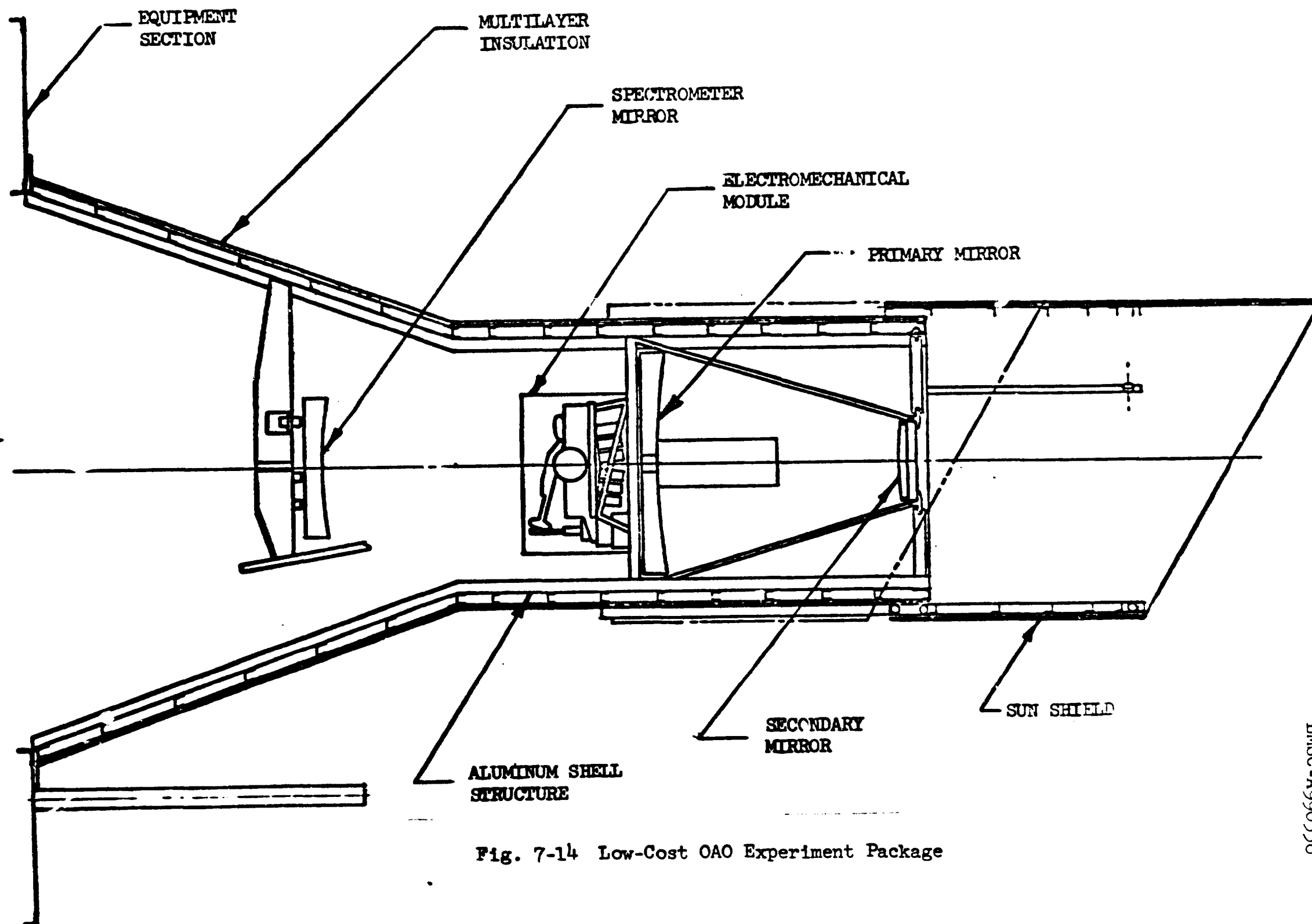


Fig. 7-14 Low-Cost OAO Experiment Package

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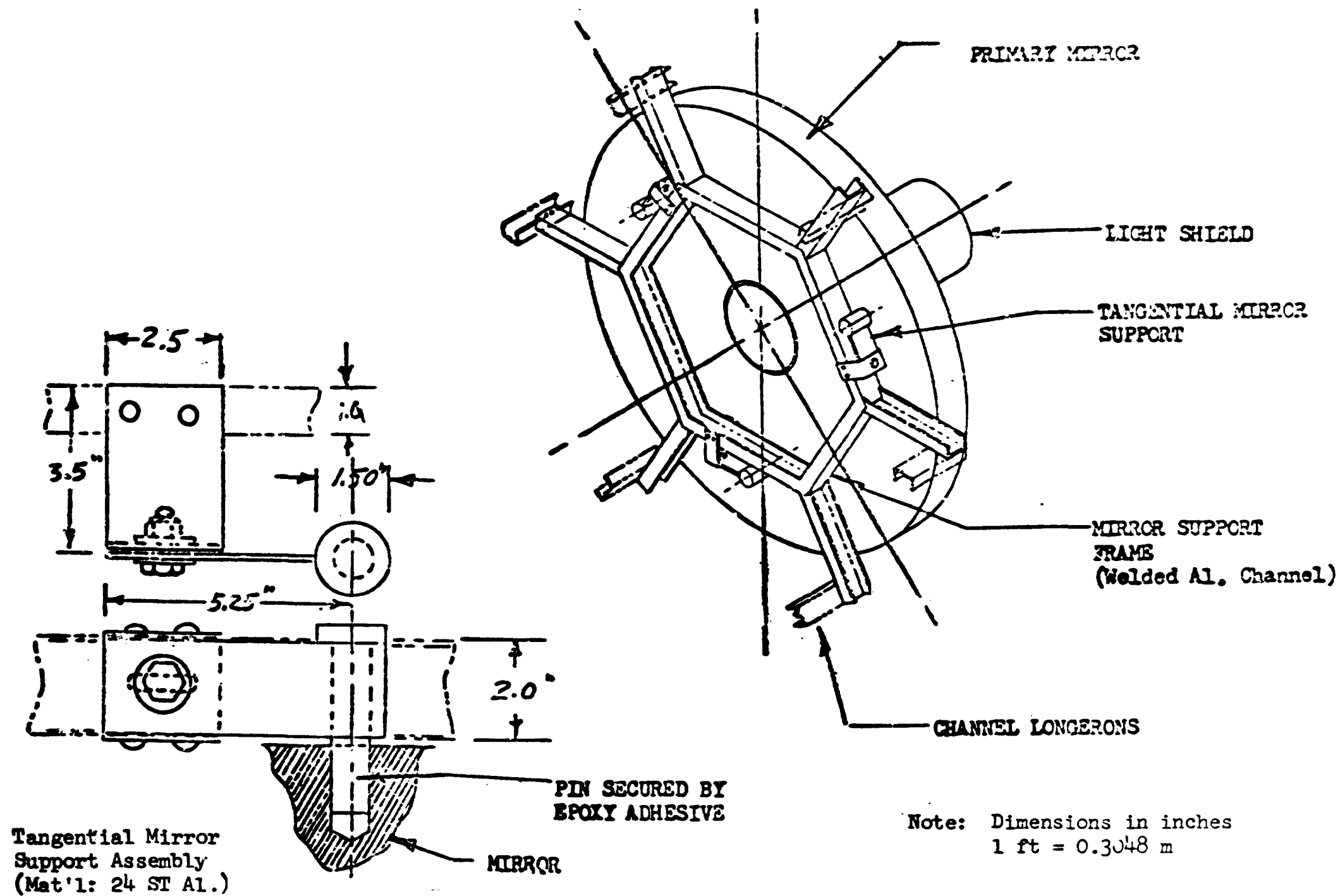
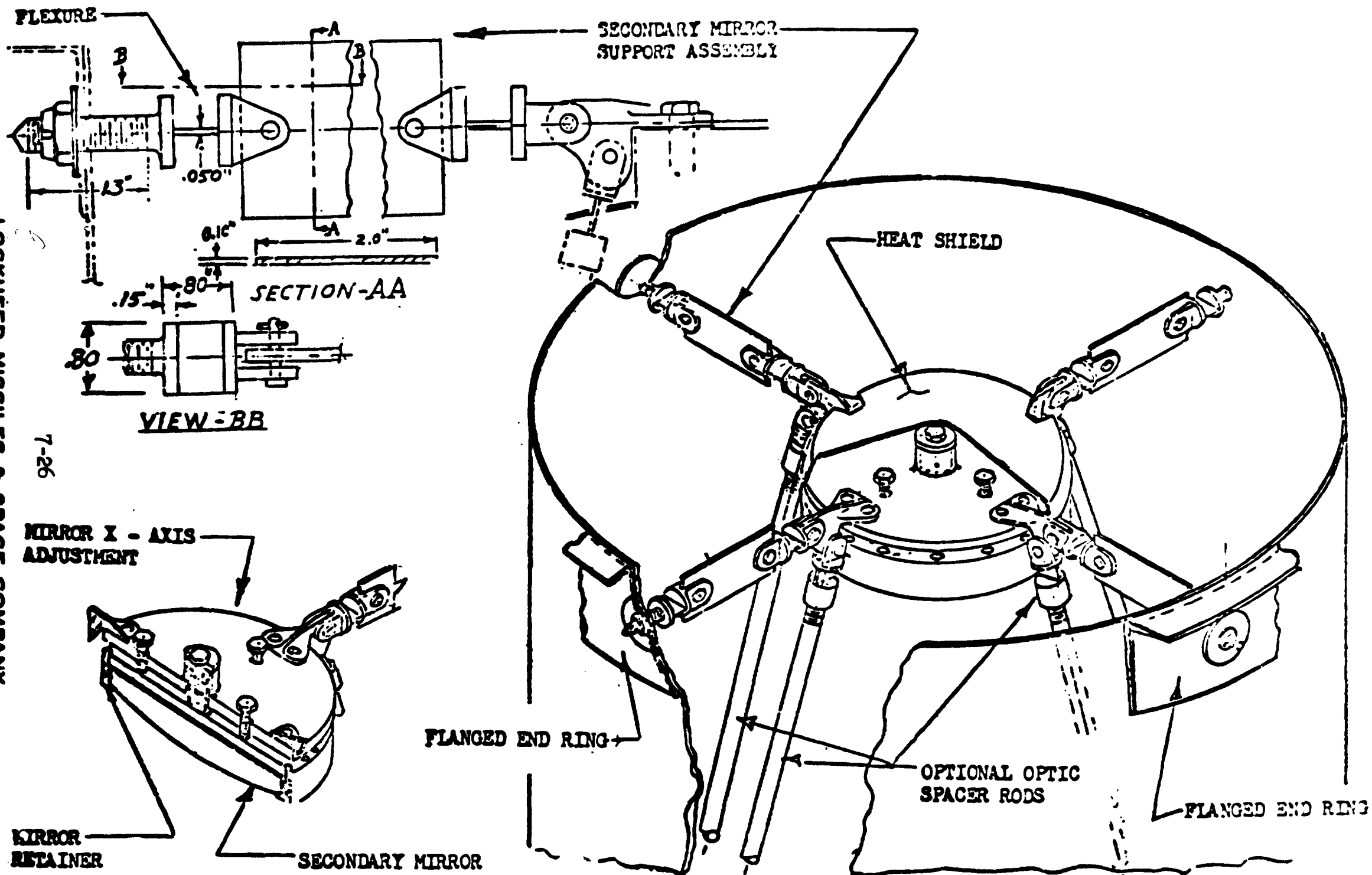


Fig. 7-15 Primary Mirror Support

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Note: Dimensions in inches
1 ft = 0.3048 m

Fig. 7-16 Secondary Mirror Support

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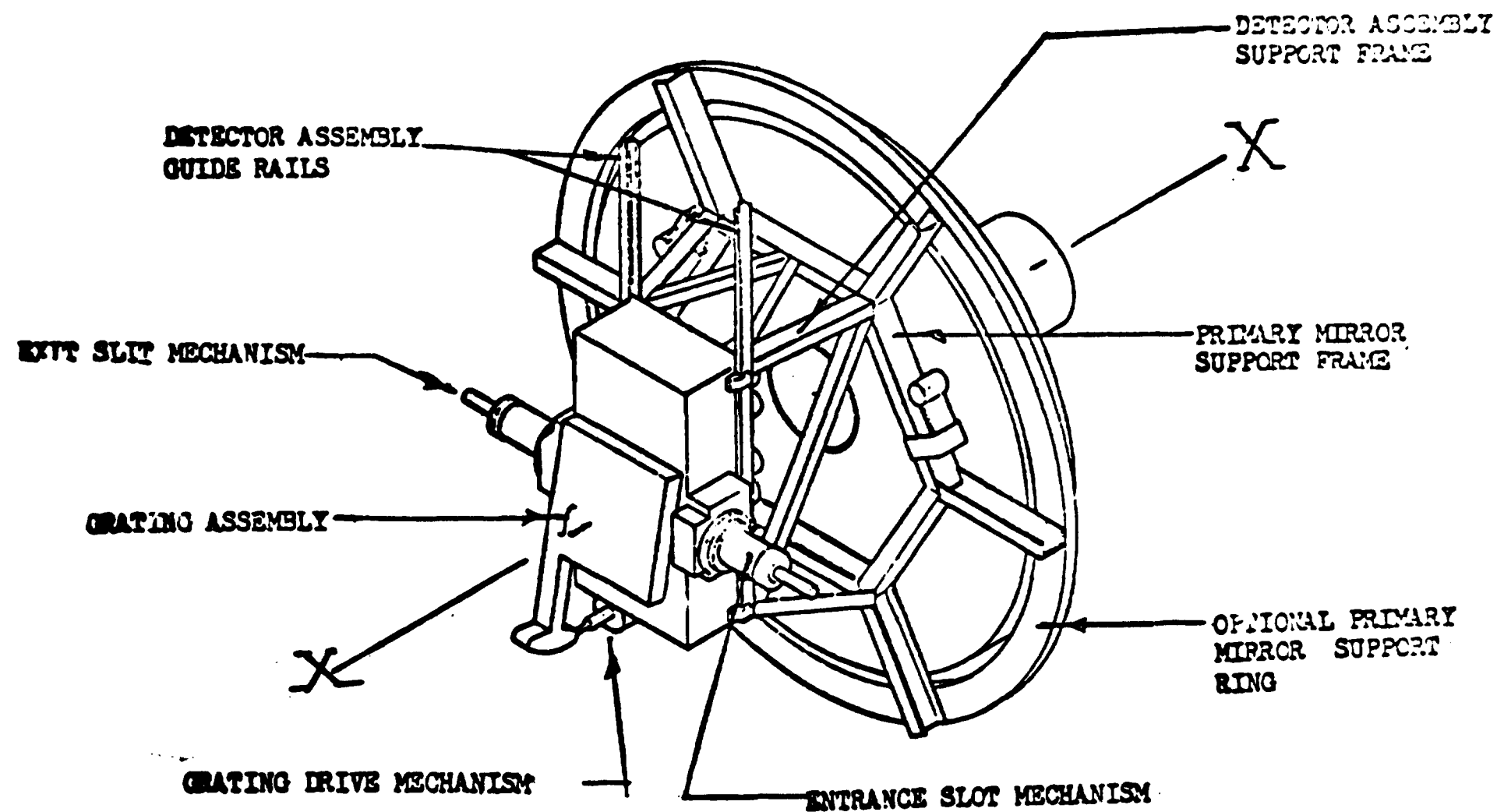


Fig. 7-17 Spectrometer Detector Assembly Mounting

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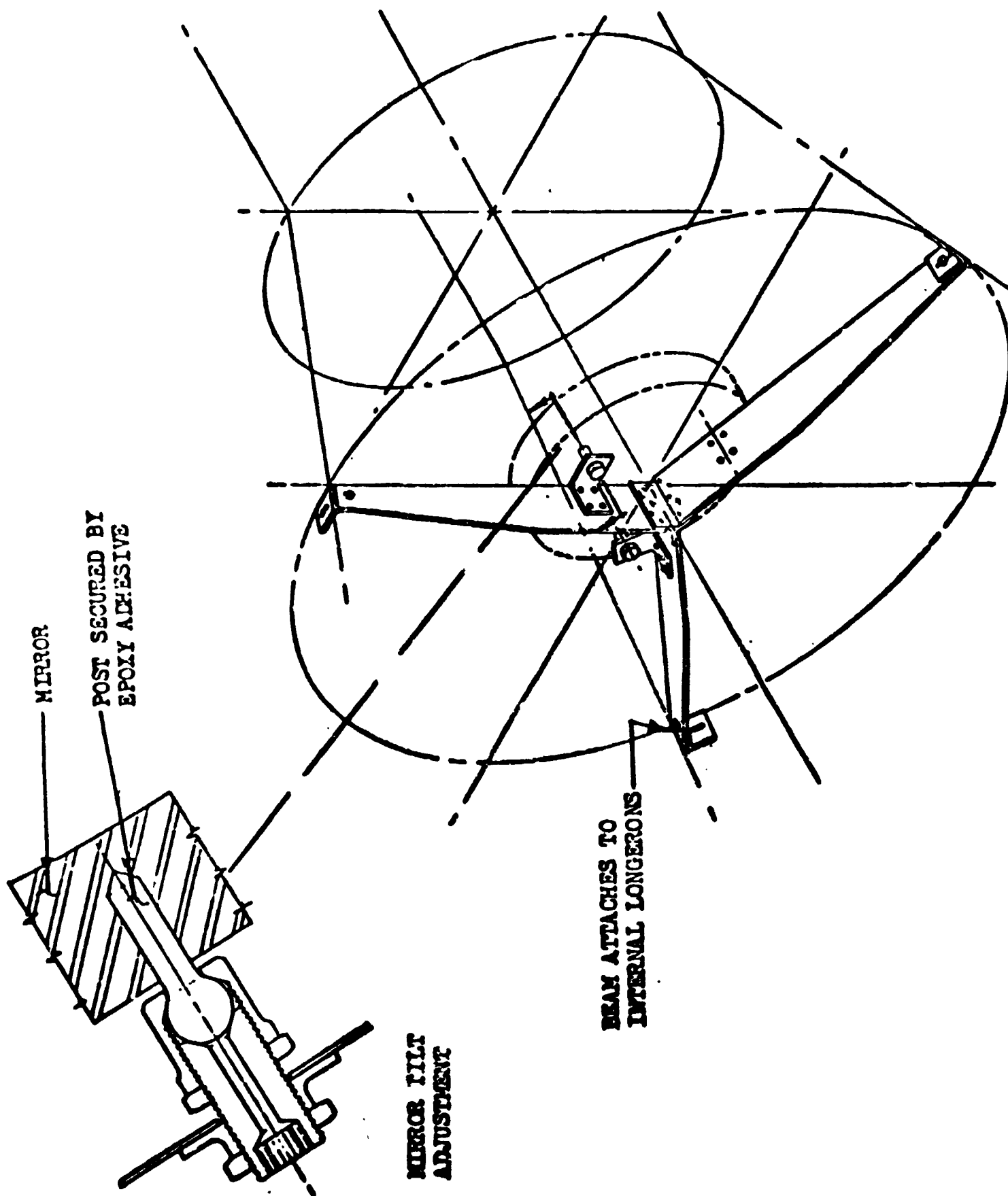


Fig. 7-18 Spectrometer Mirror Installation.

The secondary mirror is installed at the forward end of the shell structure as shown in Fig. 7-16. Support and lateral positioning of the mirror are provided by four adjustable flexure links. Alignment of the mirror is provided by three adjustment screws.

The detector and grating assembly is supported as shown in Fig. 7-17. The assembly is mounted on rails and may be removed and replaced in orbit.

The spectrometer mirror is supported by the "tripod" beam assembly shown in Fig. 7-18. The three beams are attached to three internal longerons of the shell structure. No provision for lateral or longitudinal adjustment of the mirror is necessary. However, tilt adjustment is required and is provided by three adjustment screws.

Additional cost savings in the design of the low-cost Experiment Package were achieved by the elimination of the in-orbit focus provisions, which are no longer required because of the rigidity of the structure and the virtual elimination of temperature gradients attained by isolating and insulating the experiment package. The design guidelines listed in paragraph 7.3.1 were followed in the design of the low-cost Experiment Package. The cost, as shown in Fig. 7-19, particularly the average unit costs, are significantly lower than the corresponding costs of the Goddard Experiment Package of OAO-B.

7.3.2.2 Synchronous Equatorial Orbiter (SEO).

a. Initial Design - SEO Experiment Installation. The Lunar Orbiter weight and volume were constrained by the limited payload capability of the launch vehicle, and a comprehensive weight saving program was necessary in the design of the Photographic Subsystem (from which the SEO Experiment was derived). Lightweight materials including beryllium were used, parts were optimized to reduce weight and compact packaging was necessary to minimize volume. These requirements resulted in relatively high RDT&E costs and affected recurring costs. The camera installation in the Lunar Orbiter is shown in Fig. 7-20; its schematic function is pictured in Fig. 7-21.

	Baseline Subsystem Costs (millions of dollars)		Low-Cost Subsystem Costs (millions of dollars)	
	RDT&E	Average Unit	RDT&E	Average Unit
QAO	11.52	5.00	9.91	2.91
SEO	35.1	2.01	25.5	1.41

Fig. 7-19 Cost Comparison - Experiment Package

7-31

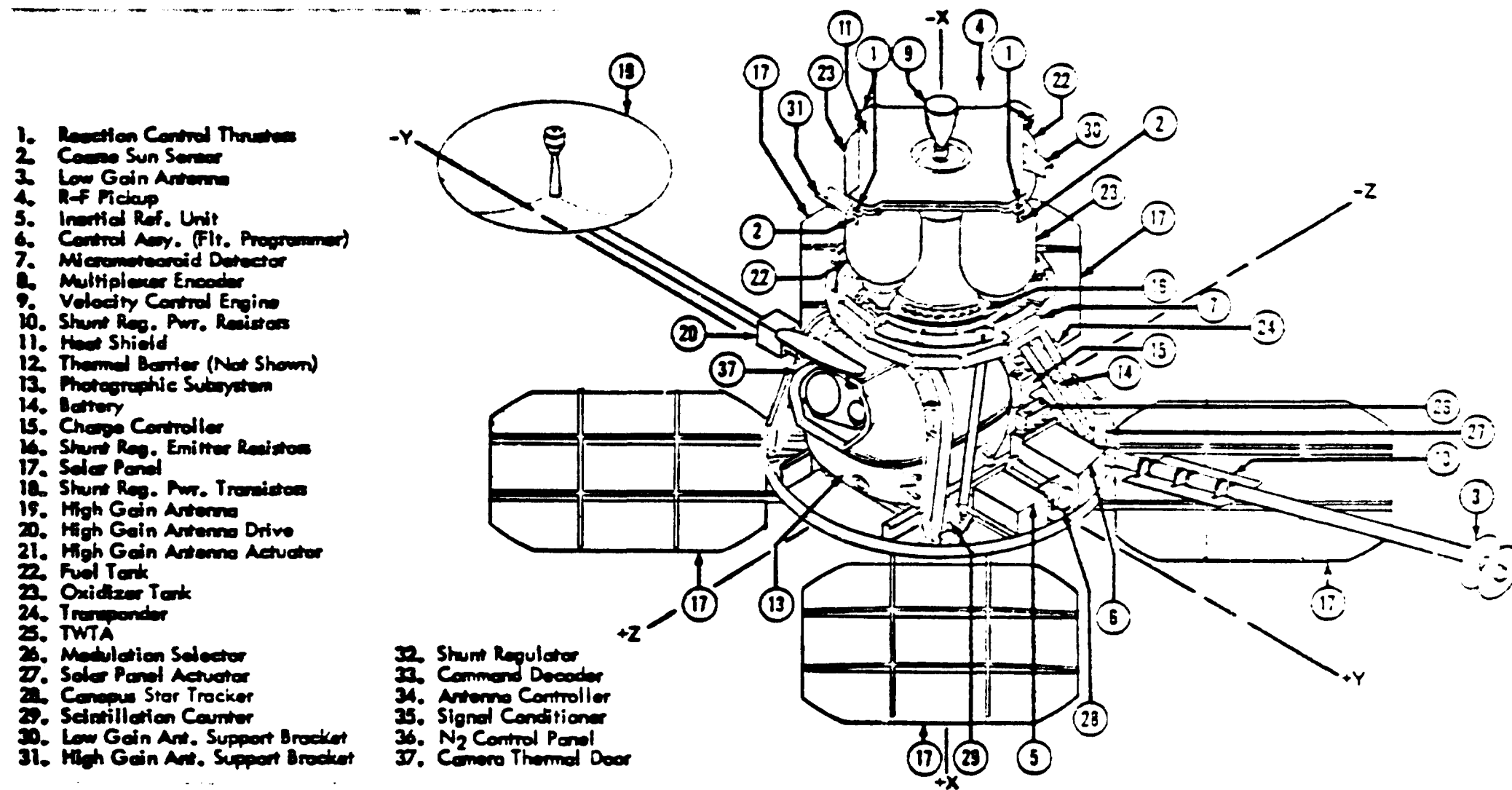


Fig. 7-20 Lunar Orbiter Spacecraft Configuration (Camera Front View)

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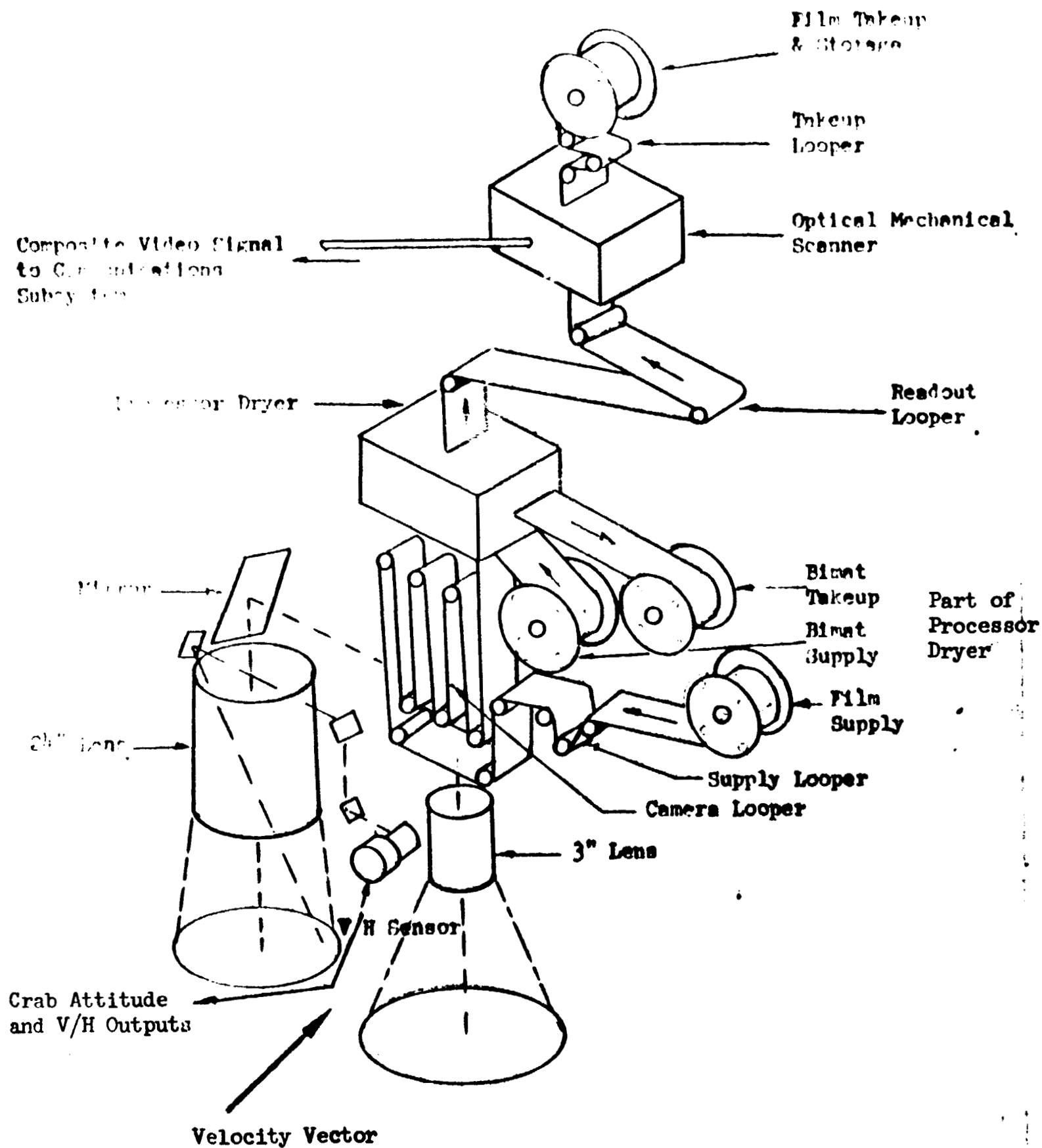


Fig. 7-21 Lunar Orbiter Photo Subsystem Schematic

b. Low-Cost SEO Experiment Installation. The general configuration of the low-cost Photographic Subsystem is shown in Fig. 7-22. The design was not constrained by weight and volume limitations; and simplification, including simplification of components, was possible. The resultant cost savings are mainly evident in the reduction of RDT&E costs by approximately 25 percent.

The Photographic Subsystem equipment is all installed on an "L-shaped" aluminum frame that is supported in the cylindrical, aluminum case by rails. The equipment assembly may be readily withdrawn from the case for access to components and for loading film and bimat.

The very limited effort applied to the design of the low-cost Photographic Subsystem did not permit the application of low-cost design guidelines to the redesign of the most costly component of the subsystem, namely the optical-mechanical scanner. Reduction of the cost of that component should be possible. The costs of the low-cost Photographic Subsystem and those of the baseline subsystem are compared in previous Fig. 7-19.

7.4 STABILIZATION & CONTROL SUBSYSTEM DESIGN GUIDELINES

7.4.1 Basic Recommended Design Approaches

Stabilization and Control subsystems are usually the most costly of spacecraft because of the complexity of the sensors, actuators and electronic equipment required for their mechanization. The designs of such subsystems have been constrained by weight, volume, and power limitations, and have been complicated by redundancy and backup modes incorporated to maximize probability of mission success. Efforts to reduce the costs of Stabilization and Control subsystems should seek to reduce their complexity and to eliminate the cost penalties that derive from weight control requirements.

Costs of Stabilization and Control subsystems can be reduced by using the following guidelines in design:

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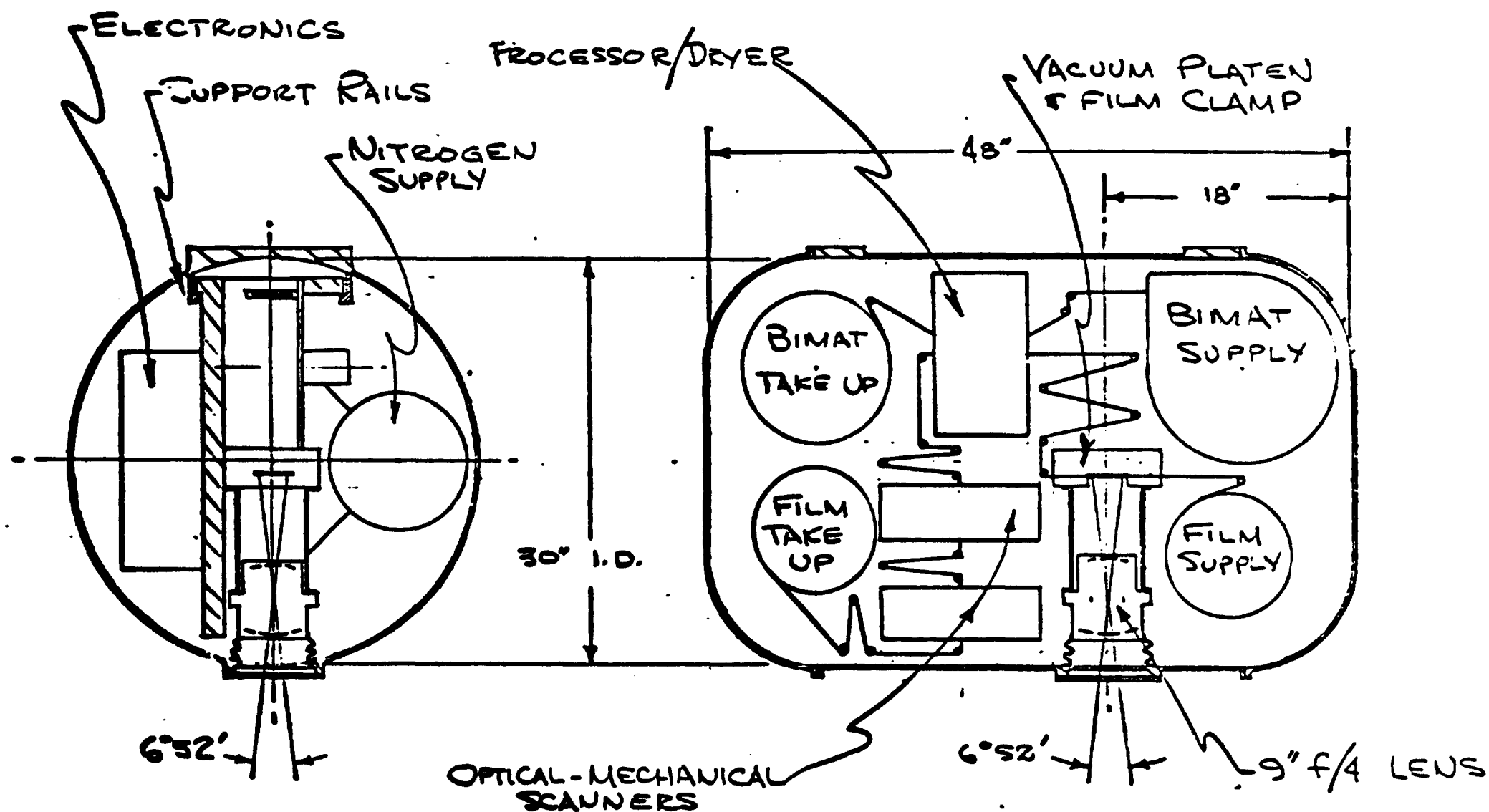


Fig. 7-22 Low-Cost SEO Photographic Subsystem

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- (1) Select the simplest system that will meet specification requirements, to reduce design, analysis, fabrication, and testing efforts.
- (2) Do not over-specify performance requirements.
- (3) Establish reliability goals based on in-orbit checkout capability of the Space Shuttle.
- (4) Limit equipment redundancies and backup operating modes to those specifically required by reliability goals.
- (5) Simplify equipment design by taking full advantage of the greater weight and volume capability afforded by spacecraft designed for new launch vehicles.
- (6) Avoid new technology development but exploit new technology that has been reduced to hardware.
- (7) Tradeoff the use of a general purpose computer for Stabilization and Control and Data Processing functions against alternate mechanizations.
- (8) Design low-density electronic equipment.
- (9) Minimize command and data requirements.
- (10) Design for in-orbit maintenance by modularization.

7.4.2 Examples of Low-Cost Designs of Stabilization and Control Subsystems

7.4.2.1 Orbiting Astronomical Observatory (OAO).

a. Initial Design - OAO-B Stabilization & Control Subsystem. The functions of the Stabilization and Control Subsystem of the OAO-B are as follows:

- To stabilize the spacecraft following launch vehicle separation, and establish a preselected attitude with precision.

- To orient the spacecraft, as required, to the reference attitudes dictated by the mission objectives.
- To hold the spacecraft in a reference attitude with the required accuracy over long periods of time.

To perform these functions, the OAO-B spacecraft employed gyros, solar sensors, star trackers, and magnetometers as sensors with inertia wheels, nitrogen gas-jets, and magnetic torquers as actuators for control and momentum unloading.

With each of the sensors there was, in general, an associated electronics unit called "signal processor," and with each of the actuators there was, in general, an associated electronics unit designated a "controller." In addition, there was a logic unit, or "programmer," which sequenced the Stabilization and Control subsystem through its various modes of operation. This proliferation of electronic units added complexity and cost to the subsystem.

After injection into orbit, separation of the spacecraft from the Centaur resulted in tumbling of the spacecraft. Gyros and solar sensors detected spacecraft rate and position and produced error signals needed to actuate the gas jets and inertia wheels for stabilization. Initial stabilization was programmed to result in one of two orientations - solar paddles normal to the sun or spacecraft aft end normal to the sun. From either of these orientations the spacecraft was programmed such that control was transferred to the gimbaled star trackers. This was accomplished in one of two methods:

- Star search - The spacecraft orientation (three axes) was determined from solar aspect sensor and magnetometer data, while holding on the inertial reference unit. Guide Star positions were computed with respect to this spacecraft orientation. The star trackers then were programmed to search and lock on the stars by executing a star search routine.

- Roll search - The initial stabilization placed the aft end of the spacecraft normal to the sun. A roll search around this sunline was initiated at a given spacecraft rate. A number of guide stars were selected to form a pattern and the star trackers were programmed to the calculated guide star positions. The spacecraft automatically switched to tracker control when the trackers acquired the guide stars simultaneously.

The spacecraft then had a stellar frame of reference to which subsequent satellite movements were related. In addition, the inertial reference unit (IRU) could now be reset to establish its inertial three axes reference. Spacecraft maneuvers required to point the experiment were performed under tracker or IRU control. Two-axes control of the satellite was transferred to the experiment for extremely accurate pointing of the spacecraft (fine control mode).

Each of the three major spacecraft axes had a large and a small inertia wheel. The large (coarse-control) wheel slewed the satellite, while the small (fine-control) wheel stabilized the satellite against small disturbing torques after the desired pointing had been attained. In the fine-control mode, the satellite maintained its pointing direction with an accuracy of $0.1 \text{ } \widehat{\text{sec}}$ by using an error signal derived from experiment optics.

To prevent the inertia wheels from becoming saturated, the gas jet system and a magnetic torquing system (a set of orthogonal electromagnetic coils) permitted momentum dumping.

In the coarse-control mode, using star trackers for error sensors, the three axes pointing accuracy was within $1 \text{ } \widehat{\text{min}}$ with a drift rate of less than $15 \text{ } \widehat{\text{sec}}$ in 50 minutes.

b. Low-Cost OAO Stabilization & Control (S&C) Subsystem. The low-cost OAO S&C subsystem retains many of the features of the equivalent initial-design OAO but differs from the baseline subsystem in: (1) Relying more upon on-board data processing and less on ground processing and communication; (2)

eliminating equipment dedicated to alternate/backup modes, (3) reducing the amount of analog hardware by integrating all data-processing and sequencing functions into a programmable general-purpose digital computer, and (4) eliminating primary actuator equipment whose functions can be performed efficiently by cold gas jets.

Figure 7-23 shows the functional and equipment relationships. The inertial reference unit (IRU) and two gimballed star trackers (GST) provide attitude data to the computer which combines them in a Kalman filter algorithm to compute spacecraft attitude precisely. Errors from the reference attitude produce signals to drive reaction wheels and cold gas jets to supply 3-axis stabilization and control. Primary attitude control torques are obtained by varying the speed of the reaction wheels, equivalent to the OAO "fine" wheels. Inertially-secular disturbance torques can cause the wheels to saturate (reach maximum speed); wheel desaturation is accomplished by torquing the spacecraft with the low-level cold gas jets.

Reorientation from one attitude reference to another is accomplished by a series of single axis slews derived by the on-board computer and using the high-level cold gas jets. IRU-detected attitude errors control attitude about the non-slew axes during slewing.

The Boresighted Star Tracker and GEP error sensor perform the same roles as for the baseline OAO Subsystem, namely, provide a more precise pitch and yaw attitude reference than that obtained from the gimballed star trackers (GSTs).

The principal of operation of the primary attitude determination function is to obtain accurate long-term attitude information from the GSTs and a computer-stored star catalog, while the IRU gyros provide a precise short-term reference. The computer processes and mixes the information from both sources.

By continuously maintaining a record of the instantaneous attitude in the computer, the ability to acquire stars and to maneuver is considerably enhanced over the baseline OAO, and the chances of loss of star reference are

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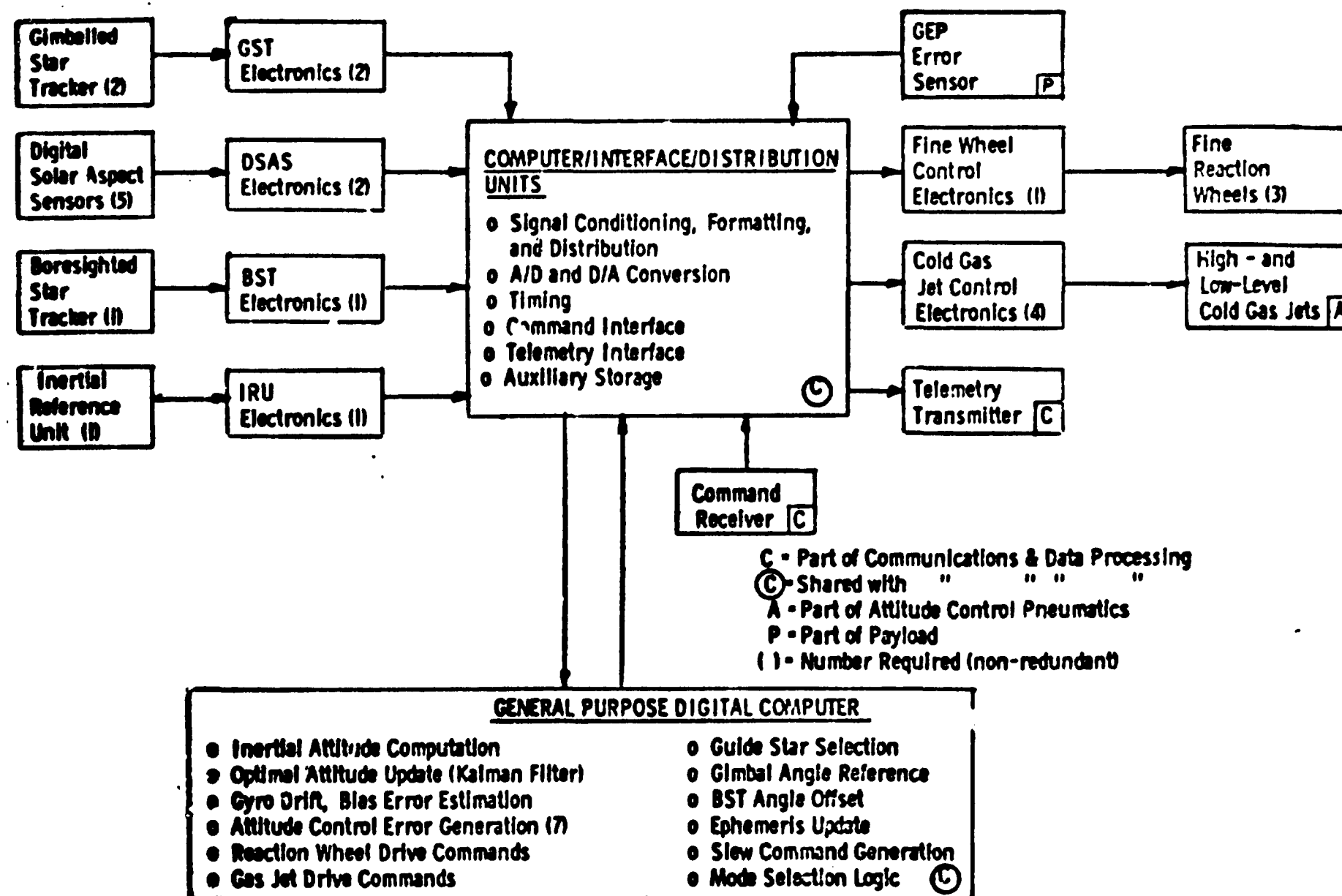


Fig. 7-23 Block Diagram - Low-Cost OAO S&C Subsystem

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greatly diminished. This higher assurance of success in finding, holding, and reacquiring stars should reduce the level of design analysis, verification and testing efforts and lessen the need for backup modes and equipment on-board. By reducing the ground support of flight operations to periodically transmitting blocks of spacecraft reference attitudes for payload orientation, not only are ground operations minimal, but the data link support is correspondingly reduced in requisite quantity and quality, not to mention criticality of timing. By replacing hardware with software not only are non-recurring and recurring costs reduced but the cost impact of changes which inevitably occur during development (due to inadequate performance or changed requirements) is considerably softened.

The major functional features of the low-cost OAO S&C subsystem are listed in Fig. 7-24, together with the cost reduction features of the design. Cost reduction was accomplished by the elimination of all OAO-B equipment and operating modes not essential to compliance with the OAO-B system requirements specification; and by incorporating a general purpose computer, thus reducing the number of electronic packages needed for S&C functions and substituting software for hardware in the processing of data. The costs of the low-cost OAO Stabilization & Control Subsystem are compared with the costs of the baseline OAO-B S&C Subsystem in Fig. 7-25.

7.4.2.2 Synchronous Equatorial Orbiter (SEO). The design is derived from the Stabilization and Control Subsystem of the Lunar Orbiter.

a. Initial-Design SEO Stabilization & Control Subsystem. The SEO is carried into synchronous orbit by a Burner II stage. During the transfer-orbit coast to synchronous orbit the Burner II/SEO combination is slowly rolling in the apogee burn attitude for gyro drift-averaging, and thermal control.

After synchronous orbit injection and stage separation, the SEO must orient itself to the earth vertical-velocity vector-orbit normal coordinate frame and maintain attitude stability on station, within specified tolerances, for two years.

FUNCTIONAL

- RETAINS ONLY ESSENTIAL OAO OPERATING MODES/EQUIPMENT
- CONTINUOUSLY COMPUTES SPACECRAFT ATTITUDE ON-BOARD
- MAKES ATTITUDE CONTROL INDEPENDENT OF LONG-TERM ATTITUDE REFERENCE
- HAS HIGH DEGREE OF AUTONOMOUS CAPABILITY
- USES SOFTWARE INSTEAD OF HARDWARE FOR SIGNAL PROCESSING/LOGIC/SEQUENCING
- RETAINS SOME DEGRADED BACKUP MODE CAPABILITIES

COST REDUCTION

- ELIMINATES TWELVE PRIMARY OAO ELECTRONICS PACKAGES
- ELIMINATES OAO COARSE MOMENTUM WHEELS AND MAGNETIC TORQUERS
- ELIMINATES OAO COARSE AND FINE SUN SENSORS
- ELIMINATES TWO GIMBALLED STAR TRACKERS
- ELIMINATES EQUIPMENT DEDICATED TO ALTERNATE MODES (RAPS)
- REDUCES RDT&E COSTS ASSOCIATED WITH DESIGN/REQUIREMENT CHANGES
- REDUCES RDT&E COSTS ASSOCIATED WITH DESIGN VERIFICATION AND TESTING
- REDUCES FLIGHT SUPPORT OPERATIONS AND EQUIPMENT INCLUDING DATA PROCESSING AND DATA LINK

Fig. 7-24 Features of Low Cost OAO S&C Subsystem

Spacecraft	Baseline Subsystem Costs (millions of dollars)		Low-Cost Subsystem Costs (millions of dollars)	
	RDT&E	Average Unit	RDT&E	Average Unit
QAO	76.65	11.50	29.10	5.90
SEO	15.9	3.3	9.6	1.9

Fig. 7-25 Cost-Comparison Stabilization & Control Subsystems

The initial-design SEO Stabilization and Control (S&C) subsystem has the following functions:

- To control attitude transients from Burner II separation
- To acquire Earth and Yaw references
- To meet SEO pointing requirements (two degrees, all axes) for two years
- To stabilize and control SEO attitude during east-west station-keeping maneuvers.
- To reorient SEO to the orbit reference attitude from any attitude following loss of reference for tumbling rates up to 3 deg/sec.
- To point the SEO spacecraft to the sun with near-zero rates following primary system failure.

The operating modes of the subsystem are as follows:

- Initial Earth Reference Acquisition
- Initial Yaw Reference Acquisition
- Yaw Reference Hold
- Earth Reference Hold
- East-West Stationkeeping
- Attitude Reference Reacquisition
- Momentum Unload
- Hold on Gas (Backup)

The equipment required to mechanize the subsystem is as follows:

- Flight Electronics Control Assembly
- Earth Horizon Sensor
- Sun Sensors (2)
- Rate Gyro Package

- Flight Electronics Switching Unit
- Polaris Star Tracker
- Reaction Wheels (3)

This equipment is derived from the equivalent Lunar Orbiter equipment retaining maximum commonality of operating and design concepts, and within the constraints of the weight and volume limitations of the Lunar Orbiter program.

b. Low-Cost SEO Stabilization & Control Subsystem. In contrast to the initial-design SEO, the low-cost SEO is activated and checked out aboard the Shuttle, in low earth orbit, carried to its final position, and separated in its operational attitude from the Space Tug.

If the Low-Cost SEO should fail to pass all critical tests before Tug departure, it can be retrieved and returned to low orbit or to earth for repair. Should the SEO fail during the two-year life it may be retrieved for reuse or salvage.

The Low-Cost SEO S&C Subsystem is simpler than that of the Baseline SEO. Figure 7-26 compares the implementation of the two SEO spacecraft S&C subsystems and Fig. 7-27 compares the operational modes for the two designs.

The low-cost SEO stabilization and control subsystem (Fig. 7-28) is based on active control of a two-degree-of-freedom CMG. Attitude control torques in pitch are obtained by varying the wheel speed while roll and yaw control torques result from tilting the wheel spin axis. The momentum of the gimbaled wheel supplies gyroscopic restraint of yaw attitude. A horizon sensor is the source of earth-pointing error signals and scanning laser radar on the Space Tug provides pitch, yaw and roll attitude errors for docking.

Two wide-angle digital solar aspect sensors and a 3-axis rate gyro package are included to aid in reorientation to earth reference in the event of a reversible system failure (e.g., intermittent power supply outage). The same units, in conjunction with cold gas thrusters, comprise an anti-tumbling system to permit safe revisit after subsystem failure.

ITEM OF EQUIPMENT	APPLICATION	
	BASELINE SEO	LOW-COST SEO
Flight Electronics Control Assembly	Yes	Yes
Earth Horizon Sensor	Yes	Yes
Sun Sensors (2)	Yes	Yes
Rate Gyro Package	Yes	Yes
Flight Electronics Switching Unit	Yes	No
Polaris Star Tracker	Yes	No
Reaction Wheels (3)	Yes	No
Gimballed Momentum Wheel	No	Yes
Docking Reflectors	No	Yes

Fig. 7-26 Comparison of Baseline and Low-Cost SEO S&C Subsystem Implementation

OPERATING MODE	APPLICATION	
	BASELINE SEO	LOW-COST SEO
Initial Earth Reference Acquisition	Yes	No
Initial Yaw Reference Acquisition	Yes	No
Yaw Reference Hold	Yes	No
Earth Reference Hold	Yes	Yes
East-West Stationkeeping	Yes	Yes
Attitude Reference Reacquisition	Yes	Yes
Momentum Unload	Yes	Yes
Hold on Gas (Backup)	Yes	Yes
Docking	No	Yes

Fig. 7-27 Comparison of Baseline and Low-Cost SEO S&C Subsystem Modes

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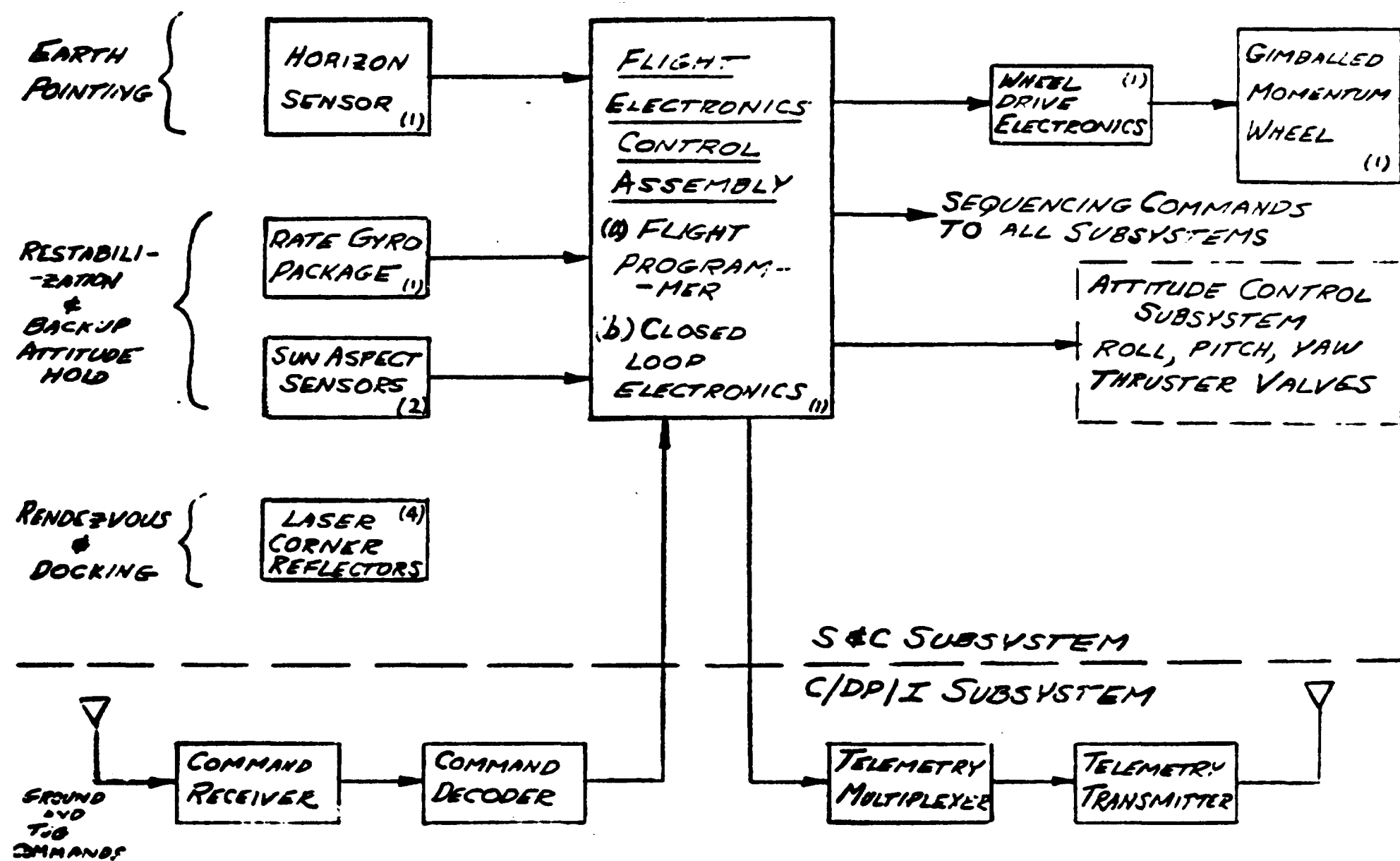


Fig. 7-28 Low-Cost SEO Stabilization & Control Subsystem

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The Flight Programmer section of the Flight Electronics Control Assembly provides timing, sequencing, and logical computations for all SEO subsystems. It accepts commands for real-time execution and for storage from the communications (CDPI) Subsystem. The Closed Loop Electronics Section converts sensor signals and command messages into control logic and actuation signals.

The double-gimballed reaction wheel control system (Fig. 7-29) is a new concept now being applied to three-axis, long-life satellite attitude control. The system uses a single biased angular momentum reaction wheel, mounted on a restricted angle, two degree-of-freedom pivot arrangement for angular momentum control in roll, pitch, and yaw. The low-cost stabilization system requires no yaw sensor.

Cost reductions were accomplished in the design of the low-cost SEO S&C subsystem by the following means:

- Combining the redundant Flight Control Electronics Units of the baseline SEO S&C Subsystem into a single package
- Using 1970 state-of-art off-shelf components in lieu of more expensive Lunar Orbiter components
- Substituting a gimballed momentum wheel for three reaction wheels
- Eliminating 2 Polaris Star Trackers (gimballed momentum wheel eliminates requirement for yaw sensing)
- Eliminating the Flight Electronic Switching Unit (used on Lunar Orbiter for pyrotechnic initiation).
- Repackaging all electronic assemblies using low-cost techniques
- Modularization of subsystem equipment

The estimated costs of the low-cost SEO S&C Subsystem are compared with the costs of the initial-design SEO S&C Subsystem in previous Fig. 7-25.

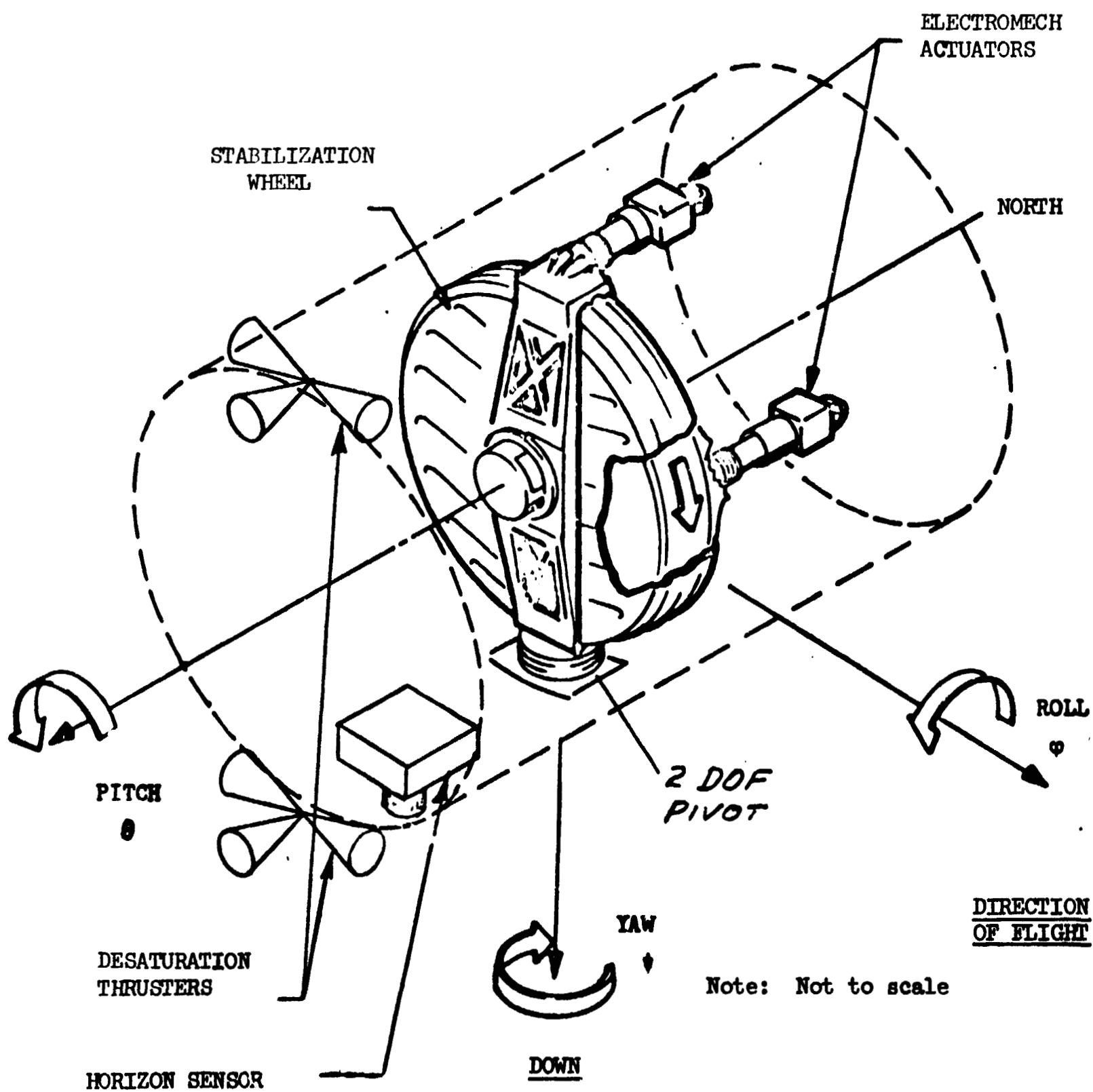


Fig. 7-29 Dual-Gimballed Wheel System

7.5 COMMUNICATIONS, DATA PROCESSING, & INSTRUMENTATION (CDPI) SUBSYSTEM DESIGN GUIDELINES

7.5.1 Basic Recommended Design Approaches

The CDPI Subsystem includes much complex electronics equipment; and to accomplish cost savings in design, development, and procurement means must be found to reduce the quantity of equipment required and also to reduce its complexity. Digital computers can perform programming, control and data processing functions, and the cost-saving potential of computers should be thoroughly evaluated in the system design of a new payload. Relaxation of weight and volume constraints permits the design of low-density electronic equipment that is inherently less costly to design, develop, and produce than the high-density electronics characteristic of many historical spacecraft. Standardization of CDPI equipment will relieve space programs of the costs of design and development of many program-peculiar elements and will result in significant savings also.

Costs of CDPI subsystems can be reduced by using the following approaches in design:

- (1) Select the simplest system, that will meet specification requirements, to reduce design, analysis, fabrication, and testing efforts.
- (2) Do not over-specify performance requirements.
- (3) Establish reliability goals based on in-orbit checkout capability of the Space Shuttle.
- (4) Limit equipment redundancies and backup operating modes to those specifically required by reliability goals.
- (5) Simplify equipment design by taking full advantage of the greater weight and volume capability afforded by spacecraft designed for new launch vehicles.
- (6) Avoid new technology development but exploit new technology that has been reduced to hardware.

- (7) Tradeoff the use of a general purpose computer for CDPI, Stabilization and Control, and other spacecraft functions against alternate mechanizations.
- (8) Increase the volume of electronic equipment (x2 or more) to reduce packaging density, thus reducing design and manufacturing costs.
- (9) Use larger, higher-rated parts in circuit design to reduce stress on parts, increase confidence in performance, and reduce testing costs.
- (10) Design for in-orbit maintenance by modularization to make possible the program cost savings that derive from checkout and repair in the Space Shuttle cargo bay.

7.5.2 Examples of Low-Cost Designs of Communication, Data Processing and Instrumentation Subsystems

7.5.2.1 Orbiting Astronomical Observatory (OAO).

a. Initial Design - OAO-B CDPI Subsystem. The functions of the OAO CDPI Subsystem are as follows:

- collect, store, and transmit experiment data to ground receiving stations
- collect, store, and transmit experiment and spacecraft equipment status information to ground receiving stations
- provide a beacon signal for ground stations to track the spacecraft
- receive, decode, and distribute real-time and stored command information from ground stations to control operation of observatory equipment
- provide observatory system timing and programming functions.

Functional relationships of the OAO CDPI Subsystem are shown in Fig. 7-30.

Communications equipment for receiving ground command signals and transmitting tracking signals, command verification signals, and spacecraft and experiment data consisted of four radio links:

- Radio command (149.52 MHz)
- Radio tracking beacon (136 MHz)
- Wideband telemetry (400 MHz)
- Narrowband telemetry (136 MHz)

The radio command system included redundant command receivers used with two antennas in a diversity receiving system. Four receivers and two antennas for redundancy provided the radio link for ground control of the spacecraft subsystems and experiment apparatus.

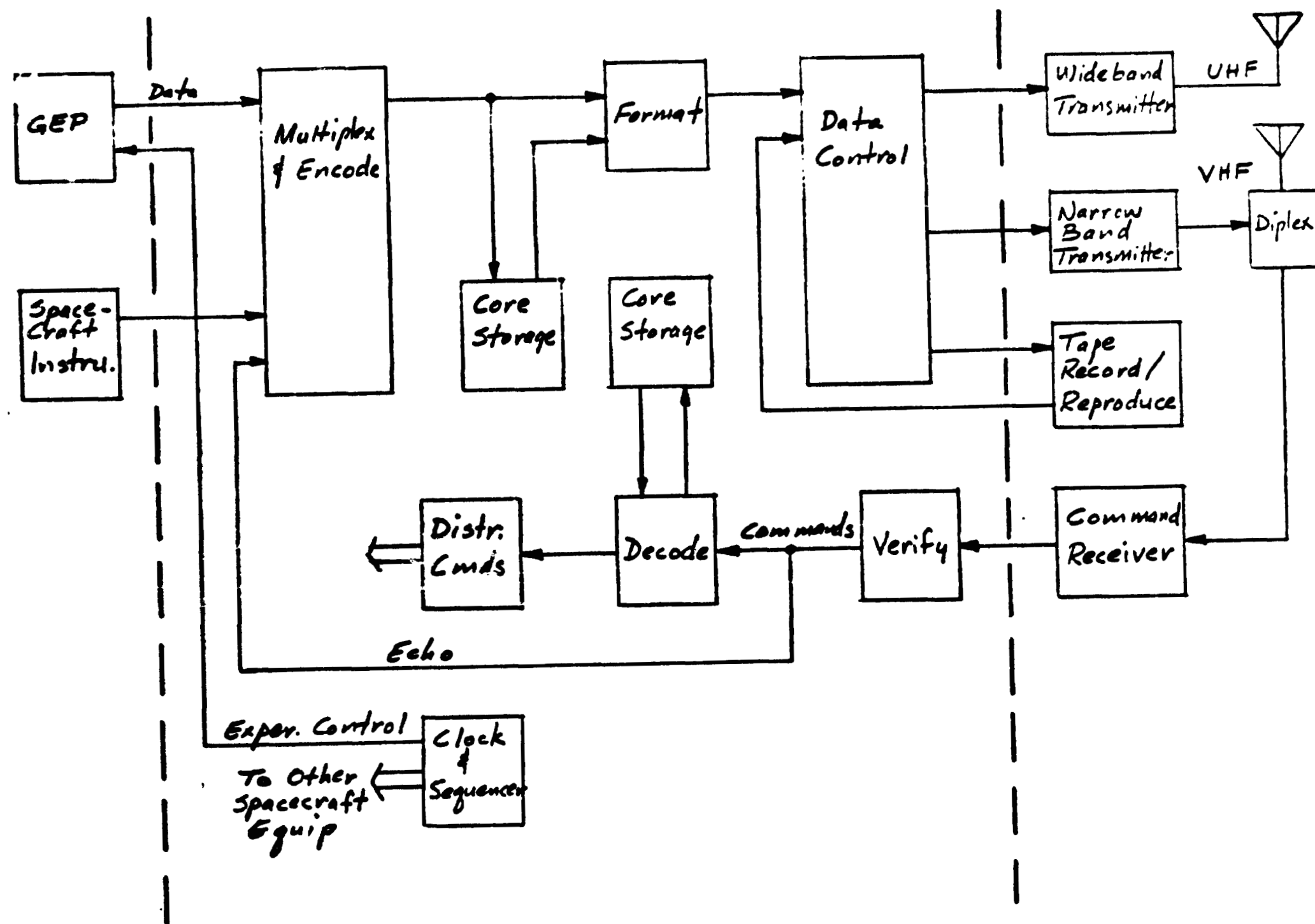
The radio tracking beacon supplied a continuous signal to facilitate ground tracking of the spacecraft. Ground command selected either of two identical transmitters.

The wideband telemetry link consisted of two transmitters. Ground command transmitted analog or digital data from the experimenters' units or digital data from the spacecraft. Ground command also selected the data to be transmitted: analog data were transmitted only in real time; digital data were transmitted from storage or in real time.

The narrowband telemetry link consisted of two identical transmitters. Ground command selected to transmit spacecraft status data, command verification, command memory dump, or stored status or experiment data.

From a functional standpoint, excluding the redundancy of transmitters and receivers, and antenna radiators, an antenna set operated at approximately 136 MHz with the tracking beacon and narrowband telemetry transmitter and at approximately 148 MHz with the command receivers. An ultrahigh frequency

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Fig. 7-30 OAO CDP&I Subsystem Functional Diagram

antenna set worked with the wideband telemetry at approximately 400 MHz. This antenna arrangement furnished nearly omnidirectional coverage about the observatory.

The onboard data-processing subsystem included circuits and storage capabilities to verify, decode, store, and distribute digital spacecraft-control and experiment commands. It also stored digital data and transmitted them from storage on ground command. The system provided timing signals for internal synchronization and for use by experiment and spacecraft equipments. The system also included spacecraft and experiment data handling equipment to accept digital data, accept and convert analog data to digital data, and to assemble those data into a suitable format for storage or real-time transmission to the ground. The digital and analog gate circuitry was available for the experimenter to program data storage. Random programming of experiment data into storage was permissible. Real-time transmission of experiment data backed up the data-storage system. During real-time transmission, data sources were sampled in a cyclic time sequence controlled by a real-time programmer included as part of the experiment data-handling equipment.

The command system contained circuits for decoding, distributing, and storing commands received from the ground-command station over the radio command receiver system. The command decoder handled the real-time commands to be executed immediately after verification in the spacecraft, and the stored commands to be placed in storage (after onboard verification) for execution at a later time. In addition, all commands could be retransmitted to the ground. Both real-time and stored commands could be transmitted over the narrowband telemetry link, and command memory could be transmitted over the wideband telemetry link to permit ground verification. Stored commands were programmable and could be executed in small time-address increments. If two or more commands had the same execution time code, they were executed within the same increment in the same order received.

A system clock provided the timing signals or pulses required by the observatory and synchronized all data words and command words.

Encoded analog data and data originally generated in binary form were assembled into a suitable format for storage or real-time transmission to the ground over the narrowband-transmitter system.

The equipment required to mechanize the initial-design OAO-B CDPI Subsystem is shown in the following list.

<u>Item</u>	<u>Quantity</u>
Command Receiver	4
Radio Tracking Beacon	2
Narrow-Band Transmitter	2
Wide-Band Transmitter	2
Diplexer	2
Hybrid Junctions	2
Antenna	4
Primary Processor & Data Storage	1
Programmer & St. Signal Controller	1
Experiment Data Handling Equip.	1
Spacecraft Data Handling Equip.	1
Spacecraft Systems Controller	1
Signal Conditioning Unit	1
Auxiliary Command Memory	1
Instrumentation (Set)	1

The Experiment and Spacecraft Data handling equipment read, encoded, and assembled digital, bi-level, and analog data signals into a format suitable for transmission or storage.

The Primary Processor and Data Storage unit and the Programmer and Star Tracker Signal Controller performed the following functions:

- Verify the accuracy of ground commands
- Provide command storage

- Execute stored or real-time commands
- Supply all basic signals for the spacecraft
- Provide data storage
- Control the sequencing of operation
- Control the mode of operation of the gimbaled star trackers

The functions of the other equipment are self-evident.

b. Low-Cost OAO Communication, Data Processing, and Instrumentation Subsystem.

Functional elements between the dashed lines of previous Fig. 7-29 comprise the Data Processing System (DPS). Two possible approaches to the DPS design are (1) separate command decoders, programmers, multiplexers, A-D converters and core storage as in the Baseline OAO-B CDPI Subsystem design, and (2) use of a central computer in an integrated data processing system. The low-cost Stabilization and Control subsystem requires the use of a digital computer to perform control system computations. Since the computer contains storage elements and provides flexibility in sequencing, formatting, decoding and multiplexing functions, it can be utilized to great advantage as the central unit in the DPS. With this approach, inflexible hardware is eliminated by the use of computer software programs which can be changed readily to suit evolutionary experiment and spacecraft operational changes. Therefore this approach has been adopted in the low-cost CDP&I Subsystem design.

The low-cost OAO CDPI Subsystem block diagram is presented in Fig. 7-31.

The Data Processing System (DPS) consists of a Data Distribution Unit (DDU), Interface and Timing Unit (ITU), and redundant computers. The integrated DPS performs all the data handling, timing, sequencing, command decoding, S&C Subsystem computation, and data storage functions. The DDU and ITU provide the interfaces with all other spacecraft and experiment equipment. The DDU is essentially a junction box plus submultiplexers for instrumentation. The ITU provides timing, multiplexing, analog-to-digital conversion, and is the primary computer/spacecraft interface. The computer provides the computation and storage functions.

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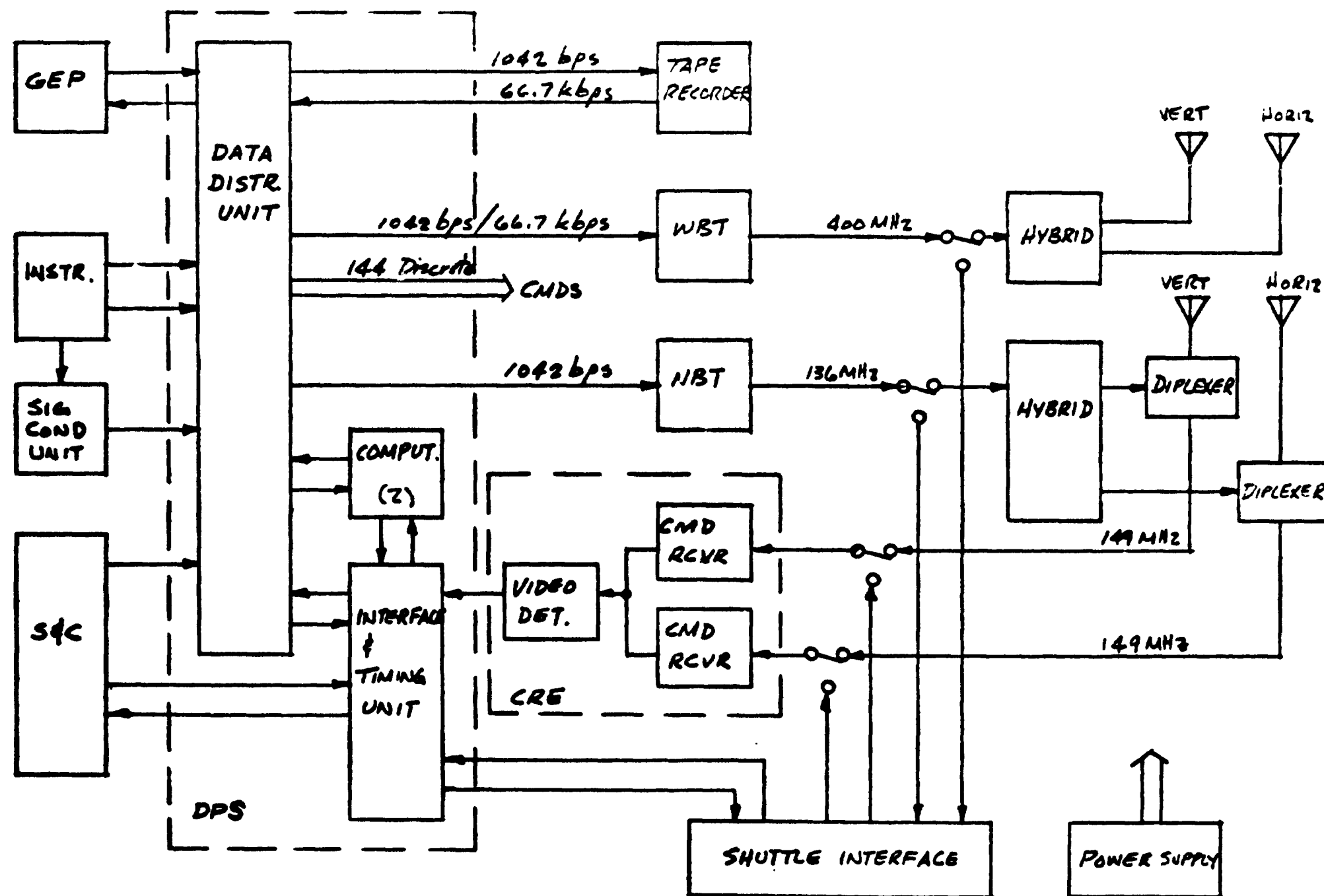


Fig. 7-31 Low-Cost OAO CDP&I Subsystem

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The tape recorder is operated under control of the computer via the DDU and records data at 1042 bps and plays back at 66.7 kbps. Output of the tape recorder is routed via the DDU to the WBT for transmission to the ground stations. Recording capability is 43.2×10^6 bits for a maximum recording time of 12 hours.

The Wideband Transmitter (WBT) accepts modulation data at either 1042 bps or 66.7 kbps from the DDU and transmits it on a carrier frequency at 400 MHz through two orthogonal, linearly polarized antennas. The transmitter power output is equally divided by the hybrid to feed the two antennas.

The Narrowband Transmitter (NBT) accepts modulation data at 1042 bps from the DDU and transmits it as PCM/PM on a carrier frequency of 136-137 MHz via the two orthogonal, linearly polarized VHF antennas. The hybrid divides the transmitter power equally between the two antennas; the diplexers provide isolation between the telemetry and command frequencies which share the same antenna.

Two AM command receivers are used to provide polarization diversity reception which assures command reception regardless of spacecraft attitude. The receiver outputs are combined for a common video detector which demodulates the FSK subcarriers to recover the PCM command words. The digital command words are then presented to the computer via the ITU for verification and execution or storage.

Interface with the Space Shuttle launch vehicle is by means of coaxial RF lines and RF switches which enable exercise of the payload including telemetry transmitters and command receivers. The Shuttle interface equipment may either provide on-board checkout and diagnostics or serve as a relay station to the data acquisition network.

The equipment of the baseline OAO-B CDPI Subsystem and the low-cost OAO CDPI Subsystem are compared in Fig. 7-32.

Item	OA0-B CDPI	Low-Cost OA0 CDPI
Command Receiver	4	2
Radio Tracking Beacon	2	-
Narrow-Band Transmitter	2	1
Wide-Band Transmitter	2	1
Diplexer	2	2
Hybrid Junctions	2	2
Antenna	4	4
Primary Processor & Data Storage	1	-
Programmer & St. Signal Controller	1	-
Experiment Data Handling Equip.	1	-
Spacecraft Data Handling Equip.	1	-
Spacecraft Systems Controller	1	-
Signal Conditioning Unit	1	-
Auxiliary Command Memory	1	-
Instrumentation (Set)	1	1
Digital Computer	-	2
Interface & Timing Unit (ITU)	-	2
Data Distribution Unit (DDU)	-	1

Fig. 7-32 OA0-B and Low-Cost OA0 CDPI Subsystem Equipment Comparison

The digital computer, ITU, and DDU of the low-cost OA0 CDPI subsystem replace seven major electronic units of the OA0-B CDPI subsystem, and replace numerous electronic units of the OA0-B Stabilization & Control subsystem. The incorporation of the computer is the most significant cost-saving change made in redesigning the CDPI subsystem. Two computers and two ITUs in time-share redundancy are required to meet the reliability goal for the subsystem.

The estimated costs of the low-cost OAO CDPI Subsystem are compared with the costs of the initial-design subsystem in Fig. 7-33.

7.5.2.2 Synchronous Equatorial Orbiter (SEO).

a. Initial Design - SEO/CDPI Subsystem. The functions of the SEO CDPI subsystem are as follows:

- (a) Transmit photographic, vidicon, spacecraft performance, and command verification data from the Spacecraft to ground stations.
- (b) Receive and decode commands transmitted from ground stations to the SEO and to temporarily store the received commands for verification purposes.
- (c) Condition all spacecraft data other than video data prior to multiplexing/encoding and to multiplex/encode all non-video data prior to telemetering to ground stations.
- (d) Sense certain conditions and parameters indicative of the Space subsystems' performance.

The baseline SEO CDPI subsystem includes the following major equipment: An S-band transponder, a tape recorder, a high-gain antenna, a traveling wave tube amplifier, a modulation selector, a command decoder, a low-gain antenna, a PCM Multiplex-Encoder, and pressure and temperature sensors.

The block diagram of the subsystem is presented in Fig. 7-34.

Digital and analog instrumentation signals representing spacecraft equipment status are sent to the Multiplexer/Encoder to be combined with signals from the environmental sensors and command verification data.

Spacecraft	Initial Design Subsystem Costs (millions of dollars)		Low-Cost Subsystem Costs (millions of dollars)	
	RDT&E	Average Unit	RDT&E	Average Unit
OA0	40.82	4.60	20.90	2.75
SEO	25.9	3.9	17.8	2.8

Fig. 7-33 Cost Comparison - Communication, Data Processing, & Instrumentation Subsystems

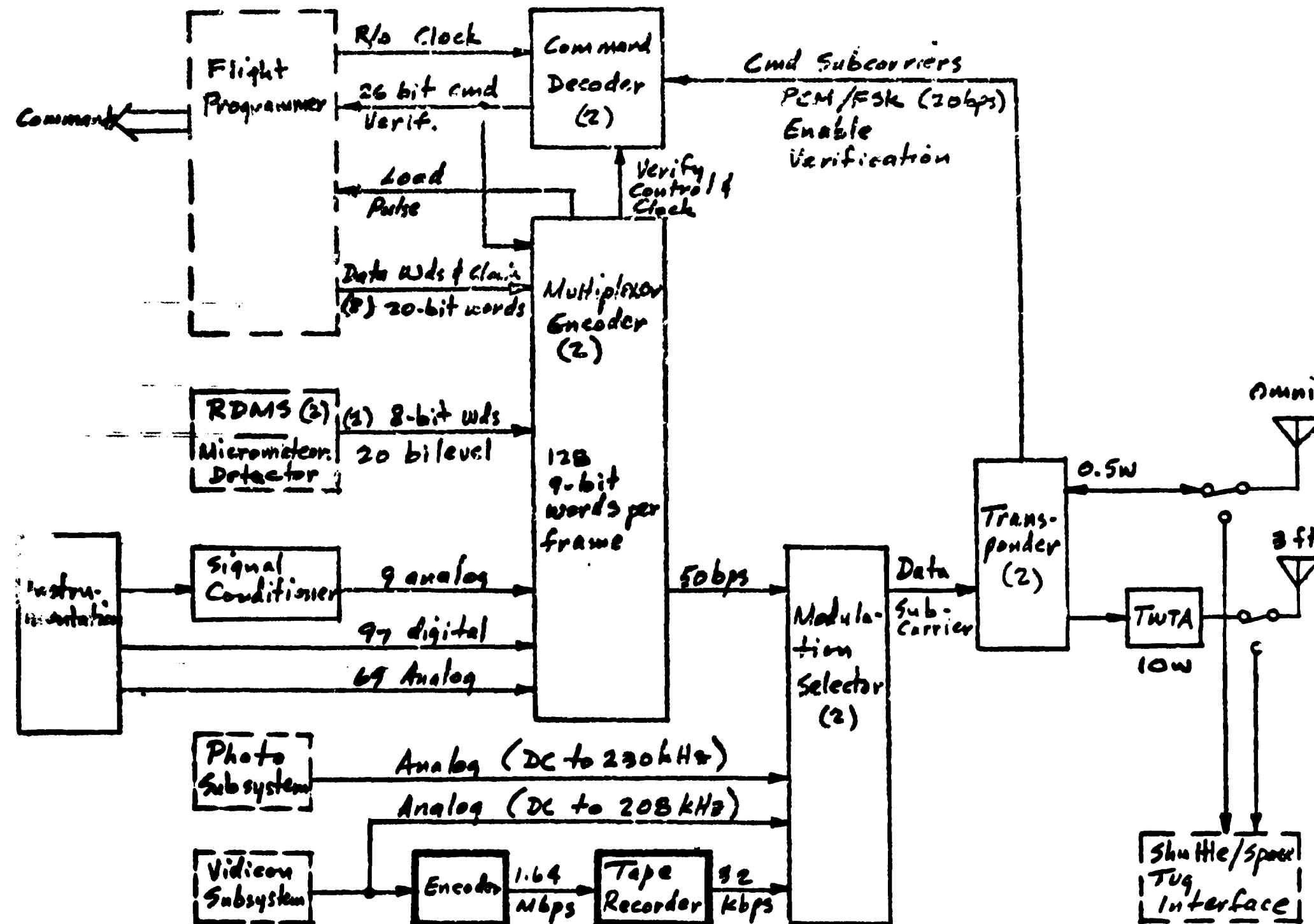


Fig. 7-34 SEO CDR&I Subsystem Block Diagram

The Multiplexer/Encoder consists of an analog multiplexer and A-D converter to digitize 78 analog channels to 8-bit accuracy and interleave the resulting digital data with the data of the digital input channels. The output data frame consists of 128 9-bit words including a 43-bit Legendre code for frame synchronization. The data frame is read out serially at a 50 bps rate. The data has a PCM/NRZ-M format and is presented to the Modulation Selector where it is FSK-modulated onto a subcarrier frequency at 30 KHz. The 30 KHz status telemetry subcarrier is combined with either one of the experiment subcarriers to form a baseband signal to phase modulate the carrier frequency.

The Photo Subsystem produces a DC to 230 KHz analog signal. This analog signal is presented to the Modulation Selector where it is vestigial-sideband modulated onto a 310 KHz subcarrier. The video subcarrier and a subcarrier reference, obtained by dividing the 310 KHz subcarrier by 8, is linearly summed with the status telemetry subcarrier during photo readout. The baseband then phase modulates the carrier frequency for transmission.

The Vidicon Subsystem video signal is encoded to 9 bits per picture element to provide a 1.64 Mbps serial bitstream. This data is stored in a digital tape recorder which is required to store up to 36 frames (3.9×10^8 bits total). Readout rate is slowed to 32 Kbps serial PCM. This data is frequency modulated (FM) onto a subcarrier frequency at 150 KHz for transmission sequentially with the photo data. As a backup mode, the analog signal may be modulated onto the photo data subcarrier for realtime readout.

The Modulation Selector selects and modulates data subcarriers under control of the Flight Programmer. Four operational modes are provided. Mode 1 provides telemetry subcarrier only for combination with range code data to modulate the carrier. Mode 2 provides telemetry plus photo data subcarriers to modulate the carrier. Mode 3 provides telemetry only to modulate the carrier for high power transmission. Mode 4 provides vidicon data plus telemetry subcarriers to modulate the carrier.

The transponder receives a carrier frequency in the range of 2110 to 2120 MHz which can be modulated with command subcarriers and range code signals. In the coherent mode, the received carrier is translated up in frequency by the ratio of 240/221 to provide a coherent transmitter frequency. The PRN range code and command subcarriers are demodulated in the receiver portion of the transponder. The range code is combined with the telemetry subcarrier to form a baseband signal which is modulated onto the downlink carrier frequency to provide two-way range tracking. The command subcarriers are presented to the command decoder for command extraction. Phase modulation of the carrier is used to control the spectral distribution of the modulated signal. The modulation index is selected to optimize the power distribution between the carrier and data subcarriers. Mode 1 uses the 0.5W output of the transponder to drive the omni antenna for transmission of range and telemetry data. The other modes use a portion of the transponder power output to drive the 10W TWT amplifier for transmission via the high gain antenna. The high gain antenna has a 10-deg beamwidth and depends upon the attitude stability of the vehicle to keep it pointed at the ground station.

The command decoder demodulates command subcarriers received from the transponder and stores each command temporarily. After verification for validity, the command is routed to the Flight Programmer and is readout to the ground via telemetry.

b. Low-Cost SEO CDPI Subsystem. The block diagram of the baseline SEO CDPI Subsystem (previous Fig. 7-34) is also applicable to the low-cost SEO CDPI Subsystem; and the description of the operation of the baseline SEO CDPI applies equally to the low-cost SEO CDPI.

Cost savings have been accomplished in the design and development of the low-cost SEO CDPI by the application of the design guidelines in paragraph 7.5.1. The costs of the low-cost SEO CDPI and the baseline SEO CDPI are compared in previous Fig. 7-33.

7.6 ELECTRICAL POWER SUBSYSTEM DESIGN GUIDELINES

7.6.1 Basic Recommended Design Approaches

Most Electrical Power subsystems of spacecraft employ solar arrays for the conversion of solar energy to electrical energy and batteries for the storage of electrical energy. They include equipment to control the charging of the batteries, the conversion of voltages, and the distribution of electrical power.

Changes in the design of spacecraft batteries do not promise significant cost savings. However, if battery sizes can be limited in number and standardized, cost savings will accrue to the national space program.

The cost of equipment for regulation and control of electrical power can be reduced by simpler, low-density packaging and by standardization.

Relaxation of weight and volume constraints will result in lower costs for solar arrays, which can be of simpler design and can be made of low-cost structural materials and lower-rated, less costly solar cells.

Costs of Electrical Power subsystems can be reduced by using the following guidelines in design:

- (1) Select the simplest system, that will meet specification requirements, to reduce design, analysis, fabrication, and testing efforts.
- (2) Do not over-specify performance requirements.
- (3) Establish reliability goals based on in-orbit checkout capability of the Space Shuttle.
- (4) Limit equipment redundancies and backup operating modes to those specifically required by reliability goals.

- (5) Simplify equipment design by taking full advantage of the greater weight and volume capability afforded by spacecraft designed for new launch vehicles.
- (6) Avoid new technology development but exploit new technology that has been reduced to hardware.
- (7) Design low density electronic equipment.
- (8) Design for in-orbit maintenance by modularization.

7.6.2 Examples of Low-Cost Designs of Electrical Power Subsystems

7.6.2.1 Orbiting Astronomical Observatory (OAO).

a. Initial Design - OAO Electrical Power Subsystem. The major equipments for the OAO-B Electrical Power Subsystem (EPS) are shown in Fig. 7-35. A silicon solar cell array and three secondary nickel cadmium batteries supplied continuous power to the system loads during orbital light and dark periods, respectively. A dc-dc converter and a dc-ac inverter furnished power to each of the spacecraft subsystem equipments requiring regulated dc or ac input voltages. The nominal unregulated bus voltage was 30 volts and varied from 24 volts to 35 volts, depending on the particular OAO operating mode.

The Electrical Power Subsystem comprised the following major equipments:

- (1) Solar Cell Array - The solar cell array converted solar energy to electrical energy. The array consisted of four main and four auxiliary paddles with solar cells on one side only. The cells were diffused junction N on P silicon (1 cm x 2 cm) with 6 mil (0.15 mm) coverglasses. The array contained 53,992 cells, and the operating voltage was 0-50 VDC approximately.

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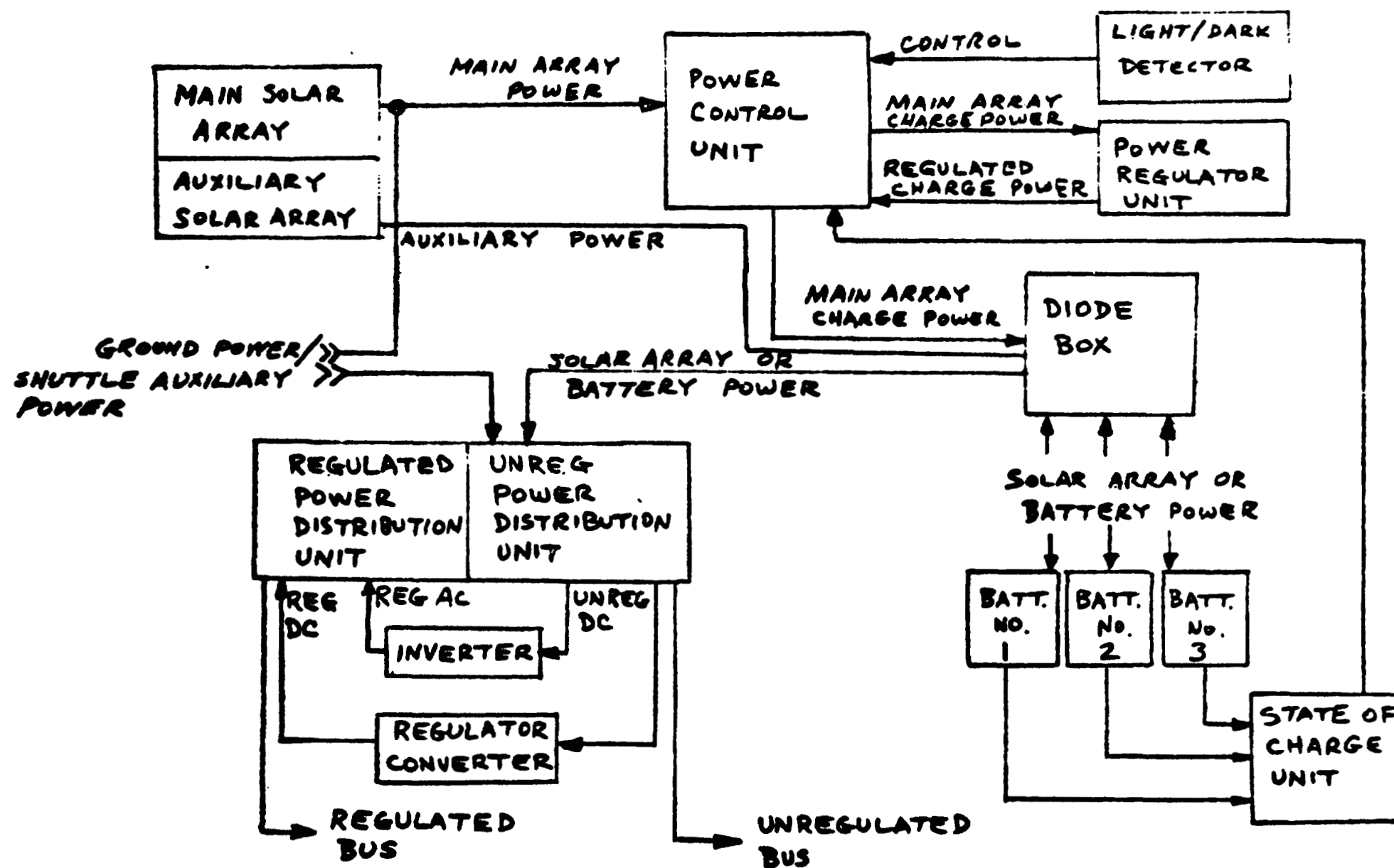


Fig. 7-35 OAO-B Electrical Power Subsystem

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- (2) Voltage Regulator and Converter - The voltage regulator and converter converted unregulated (23v to 35v) D.C. voltage to five regulated D.C. voltage levels (+28v, -28v, +18v, +10v, -10v).
- (3) Voltage Inverter - The voltage inverter converted unregulated (23v to 34v) D.C. voltage to regulated AC voltage (115v 1 ϕ , 115v 2 ϕ , and 115v 3 ϕ).
- (4) NiCd Batteries - The NiCd batteries supplied power to the spacecraft during dark periods and aided in supplying peak loads during light periods. The batteries supplied power at 23 volts to 29 volts.
- (5) Power Controller Unit (PCU) - The power controller unit controlled individual battery charging, provided solar array power to the unregulated bus, provided ground command capability, and provided bi-level telemetry information on the Electrical Power Subsystem status.
- (6) Power Regulator Unit (PRU) - The power regulator unit regulated battery charging voltage to the level indicated by the PCU, provided a signal to the PCU statusing the main regulator, and provided an alternate source of regulated voltage following main regulator failure.
- (7) Diode Box - The diode box supplied power to the unregulated bus during both light and dark periods, provided diode isolation between batteries and transferred battery charge current from the charge bus to the batteries.
- (8) Power Distribution Panel (PDU), Regulated and Unregulated - The power distribution panel provided the central distribution point for all spacecraft power, and housed all power line fuses, current sensors, relays, etc.
- (9) State of Charge Unit (SOCU) - The state of charge unit monitored battery charge and discharge operation using ampere-hour integrators and adhydrides, and telemetered data. The unit was not normally in

the primary battery charging loop and required two enabling operations to provide an input to the PCU.

- (10) Battery Current Shunt Assy - The battery current shunt assembly measured battery charge/discharge currents and provided proportional outputs to the SOCU Amp-Hr integrators.

The energy requirements imposed on the EPS were directly dependent on the operational mode of the satellite. The energy supply capabilities of the EPS depended on the satellite orientation with respect to the sun, because of the fixed array configuration. The satellite energy status was monitored and the operating modes were controlled in order to prevent stored energy falling below predetermined minimum requirements. An operating mode, sunbathing, could be initiated if the system energy state were running down. In sunbathing the satellite was oriented to position the fixed solar array normal to the sun's rays. The satellite was powered down and all excess electrical power went into returning the batteries to full charge.

In general, the EPS operated as follows:

- (1) The satellite entered the sunlight, with the batteries having supplied all energy requirements during the previous dark period.
- (2) Battery charging commenced in the shunt mode (direct connection of the solar array to all batteries in parallel).
- (3) When a battery limiting condition was reached, the charge mode was changed from shunt to regulated mode, and a taper charge was then established on that battery for the remainder of the light period.
- (4) The satellite entered the dark period, battery charging ceased, and the batteries supplied the spacecraft load.

b. Low-Cost OAO Electrical Power Subsystem. The block diagram of the baseline OAO-B Electrical Power Subsystem (previous Fig. 7-35) is also applicable

the low-cost OAO Electrical Power Subsystem; and the descriptions of the equipment and operation of the baseline OAO-B EPS are, in general, applicable to the low-cost OAO EPS. However, significant changes have been made in the design of the solar array to reduce costs. The low-cost OAO solar array, shown in Figs. 7-36 and 7-37, consists of two rigid paddles in contrast to the two articulated paddles, each consisting of four panels, of the initial design OAO-B.

Each of the two paddles of the low-cost OAO solar array consists of 16 8" x 40" (0.2 x 1.02 m) panels, four 18" x 20" (0.46 x 0.51 m) panels, and two 9" by 40" (0.23 x 1.02 m) panels all bolted to an aluminum structural frame. The panels are made of aluminum sheet stiffened by aluminum angles flush-riveted to one side of the sheet. The solar cells are cemented to the other side in appropriate series-parallel grouping. Solar paddle structures constructed in this way are heavier but considerably less costly than paddle structures of honeycomb and other lightweight materials.

2 cm x 2 cm solar cells (rather than 1 cm x 2 cm cells), are used in the solar array of the low-cost OAO with consequent savings in assembly labor. 20 mil (0.5 mm) glass covers are used and losses due to breakage in fabrication and installation of the covers are reduced. 2 cm x 4 cm solar cells should be used when available to further reduce solar array assembly costs.

The most significant factor contributing to the lower cost of the low-cost OAO solar array is the use of many more solar cells from a production lot of cells thus reducing the cost per cell. A lot of solar cells is first inspected for mechanical defects and approximately 25 percent are rejected. The remaining cells are then tested to determine a current rating for each cell. The current rating distribution of a typical lot of cells is presented in Fig. 7-38. It has been the practice in space programs to reject approximately one-third of the lowest rated cells (all cells below 132 ma of the example lot) to minimize the number of cells (and hence the weight) of a solar array with a given power output.

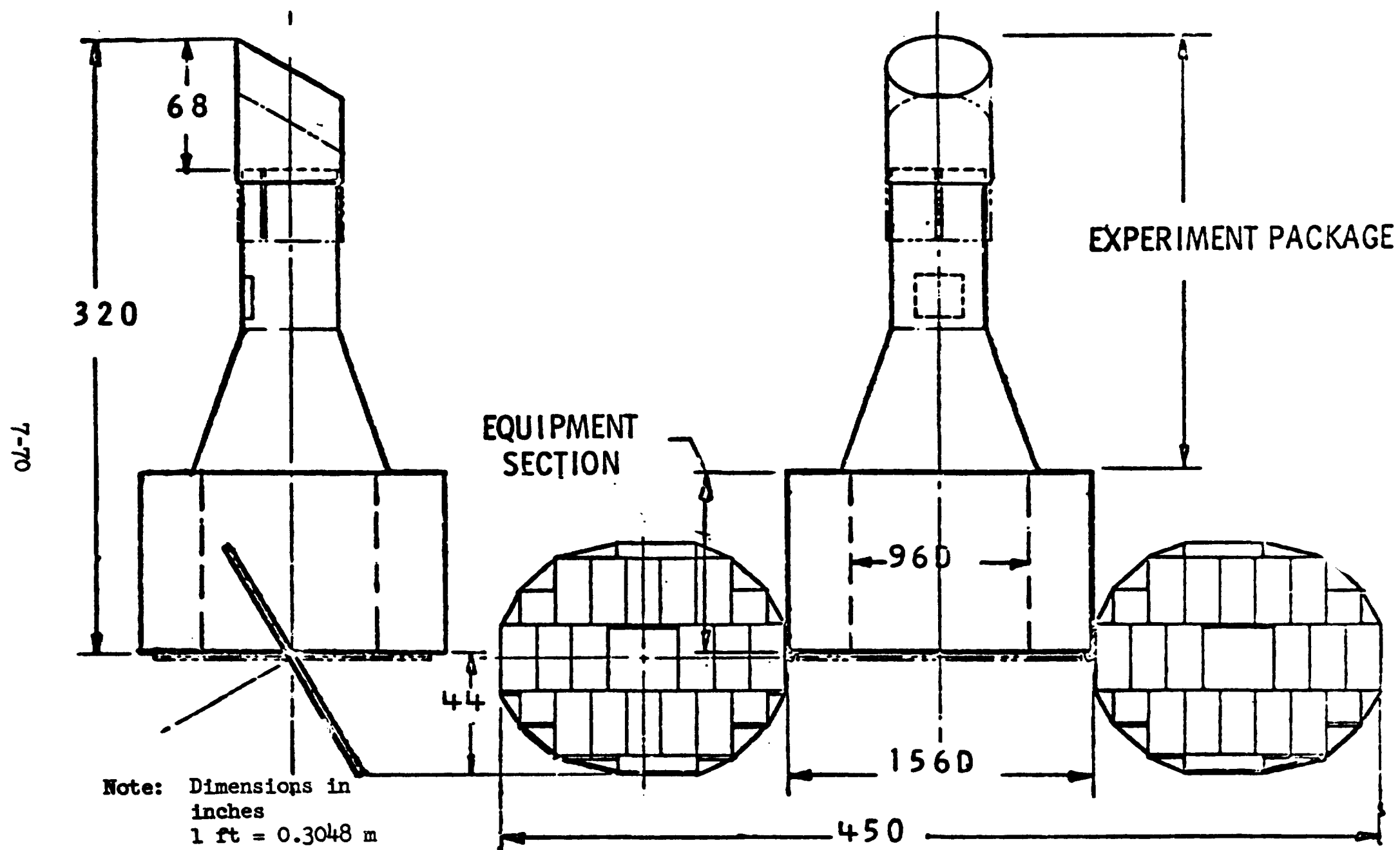


Fig. 7-36 Low-Cost OAO Configuration

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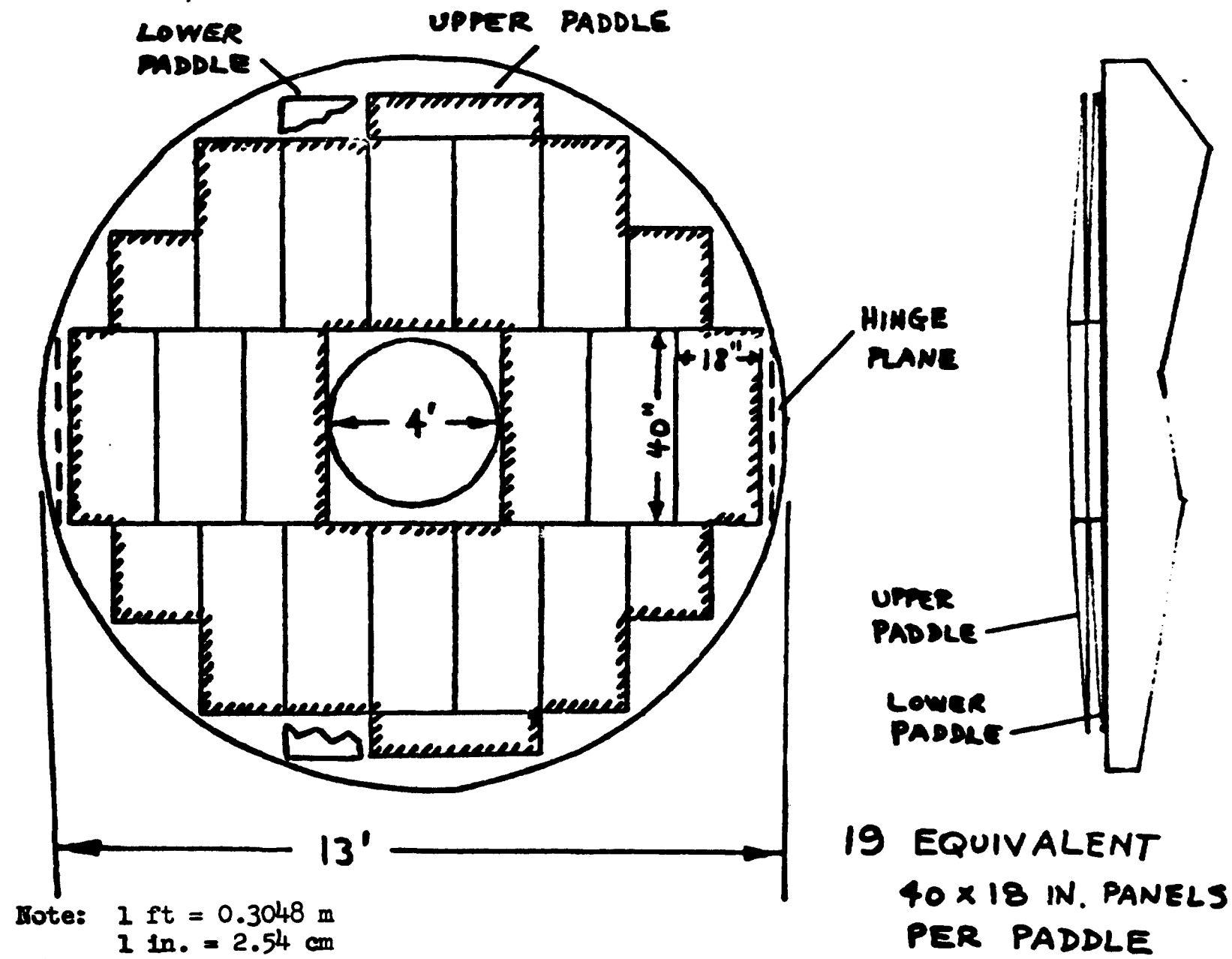
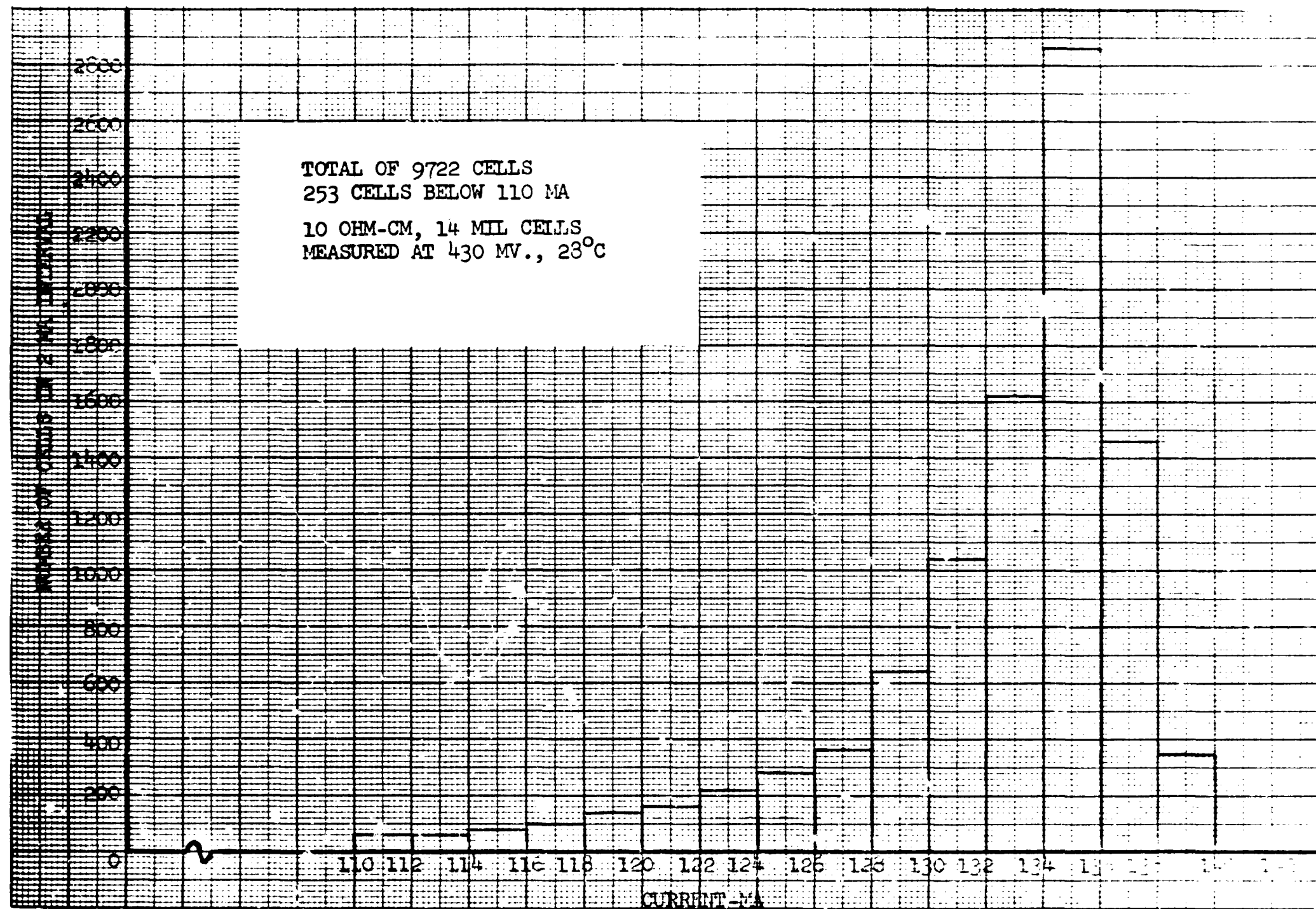


Fig. 7-37 Low-Cost OAO Solar Paddle

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Fig. 7-38 Solar Cell Performance Distribution

For the low-cost OAO solar array weight is not a constraint and approximately 97.5 percent of the functional cells of a manufacturing lot (all cells above 110 ma of the example lot) may be used.

When 97.5 percent of the functional cells of a lot, rather than 66.7 percent, are used, the average cost per watt of output power is reduced 30 percent (cost reduction) and the total number of cells in an array increases 2 percent (weight increase).

Other cost saving techniques used in the design of the low-cost OAO Electrical Power subsystem include simplified, low density packaging of charge controllers, regulators, inverters and other components.

The costs of the low-cost OAO EPS are compared with the costs of the baseline OAO-B EPS in Fig. 7-39.

7.6.2.2 Synchronous Equatorial Orbiter (SEO).

a. Initial Design - SEO Electrical Power Subsystem. The major equipments for the baseline SEO Electrical Power Subsystem (EPS) are shown in Fig. 7-40. A silicon solar cell array and four secondary nickel cadmium batteries supply continuous-power to the system loads during orbital light and dark periods, respectively. The power system provides unregulated dc power to the bus supplying the spacecraft subsystems. The subsystems condition this dc power as required with the individual power supplies. The nominal unregulated bus voltage is 28 volts and may vary from 23 volts to 35 volts, depending on the SEO operating mode.

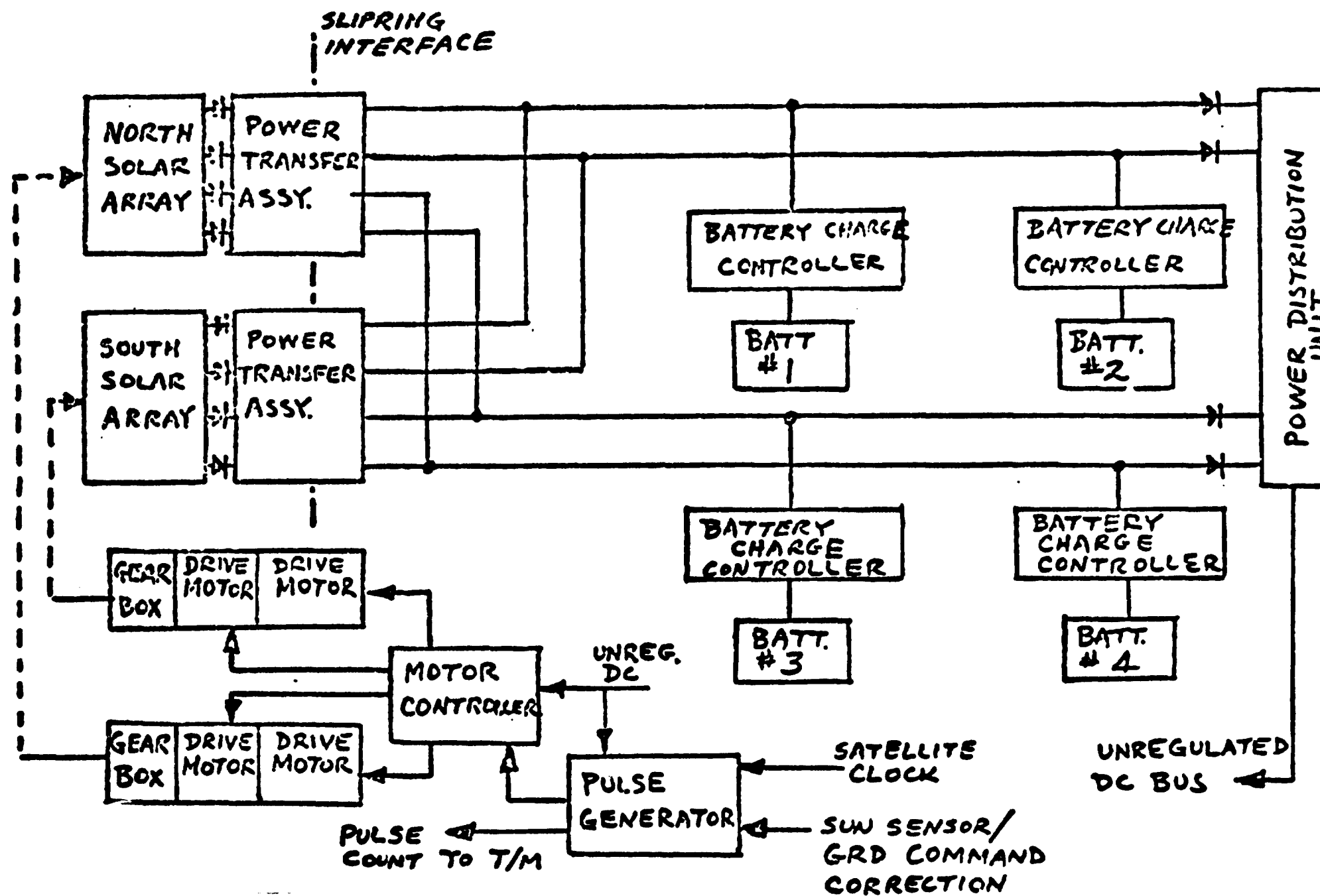
The Electrical Power Subsystem comprises the following major equipments:

- (1) Solar Cell Array - The solar cell array converts solar energy to electrical energy. The array consists of two paddles with cells on one side. The cells are phosphorous diffused N/P silicon (2 x 2 cm), 12 mils (0.3 mm) thick with 20 mil (0.5 mm) coverglasses.

Spacecraft	Baseline Subsystem Costs (millions of dollars)		Low-Cost Subsystem Costs (millions of dollars)	
	RDT&E	Average Unit	RDT&E	Average Unit
QAO	17.08	3.90	11.24	2.80
SEO	12.4	1.9	10.2	1.7
SRS	3.95	0.7	2.15	0.41

Fig. 7-39 Cost Comparison Electrical Power Subsystems

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Fig. 7-40. Baseline SEO Electrical Power Subsystem

The array contains 10,752 cells. The array operating voltage is 0-50 VDC approximately. The two array paddles are individually driven to track the sun in one axis and a slipping power transfer assembly is provided for each paddle.

- (2) Voltage Regulator - On-array Zener diode shunt voltage limiters operate to provide an unregulated 24 to 35 vdc to the Power Distribution Unit and the Battery Charge Controllers.
- (3) NiCd Batteries - NiCd batteries supply power to the spacecraft subsystems during dark periods and aid in supplying peak loads during light periods. The batteries supply power at 23 to 29 volts.
- (4) State-of-Charge Unit - The state-of-charge unit monitors battery charge and discharge operation using ampere-hour integrators and provides data on battery charge status to the T/M system. The voltage is 23-24 VDC unregulated.
- (5) Battery Current Shunt Assembly - The battery current shunt assembly measures battery charge and discharge currents and provides proportional outputs to the State of Charge Unit for use in Amp. Hr. integrators.
- (6) Battery Charge Controllers - Battery charge controllers control battery charging to levels appropriate for the temperature and state of battery charge.
- (7) Tracker Pulse Generator - The tracker pulse generator provides input pulses to the solar array drive motor controller at the fixed rate required for sun tracking. The timing reference used is the SEO clock timer output.
- (8) Motor Controller - The motor controller provides switching of unregulated 28 vdc to the phases of the stepper motors in proper sequence and for timed duration to rotate the solar array. The time

of energizing phases is established by the pulses received from the Tracker Pulse Generator.

- (9) Tracker Drive Motors - The tracker drive motors step 90 deg when activated by the Motor Controller and provide the force to rotate the solar array, in steps, through a gear box.
- (10) Power Transfer Assembly - A slipring assembly attached to each rotating solar paddle shaft allows array power and instrumentation signals to be picked up by a brush assembly attached to the vehicle structure. A power transfer assembly is used on each of the two array paddles and continuous 360 deg tracking of the arrays is provided.
- (11) Power Distribution Unit - The power distribution unit provides the central distribution point for all spacecraft unregulated 28 vdc power, and houses line fuses, current sensors, relays, etc.

In synchronous equatorial orbit the SEO experiences eclipse shading for from zero to 72 minutes for two periods of 44 days each during one year. The EPS operates as follows:

- (1) The satellite enters sunlight, with the batteries having supplied all energy requirements during the preceding dark period.
- (2) Battery charging commences at the maximum rate and continues until the next dark period.
- (3) The satellite enters the dark period, battery charging ceases, and the batteries supply the spacecraft load.
- (4) When the eclipse season is over, the batteries are continuously trickle charged.

b. Low-Cost SEO Electrical Power Subsystem. The block diagram of the SEO Electrical Power Subsystem (Fig. 7-40) is also applicable to the low-cost

SEO Electrical Power subsystem; and the descriptions of the equipment and operation of the baseline SEO EPS are, in general, applicable to the low-cost SEO EPS.

The solar array of the low-cost SEO shown in Fig. 7-41 incorporates the design concepts employed in the design of the solar array for the low-cost OAO described in para. 7.6.2.1(b). Aluminum sheet metal is used for the paddle structure and the solar cell substrate; and 97.5 percent of the functional solar cells from each manufacturing lot of cells are used. Simplified, low density packaging is used in the design of charge controllers and other EPS equipment.

The estimated costs of the low-cost SEO Electrical Power Subsystem are compared with the costs of the baseline SEO EPS in previous Fig. 7-39.

7.6.2.3 Small Research Satellite (SRS). The SRS EPS is relatively simple and is configured as shown in Fig. 7-42. The principal difference between the low-cost SRS EPS and the baseline SRS EPS is in the contrasting solar array configurations.

The solar array of the baseline SRS consists of six erectable, articulated solar paddles and some body-mounted solar cells. The solar array of the low-cost SRS consists entirely of body mounted solar cells; the cells are cemented into eight stiffened aluminum panels 20" x 26.9" (0.51 x 0.69 m) that are bolted to the spacecraft structure as described in previous para. 7.2.2.3. This low-cost design is possible because of the larger surface area of the low-cost SRS, which, in turn, is made possible by the relaxation of weight and volume constraints on the design of the spacecraft. The estimated costs of the low-cost SRS EPS are compared with the costs of the initial-design EPS in previous Fig. 7-39.

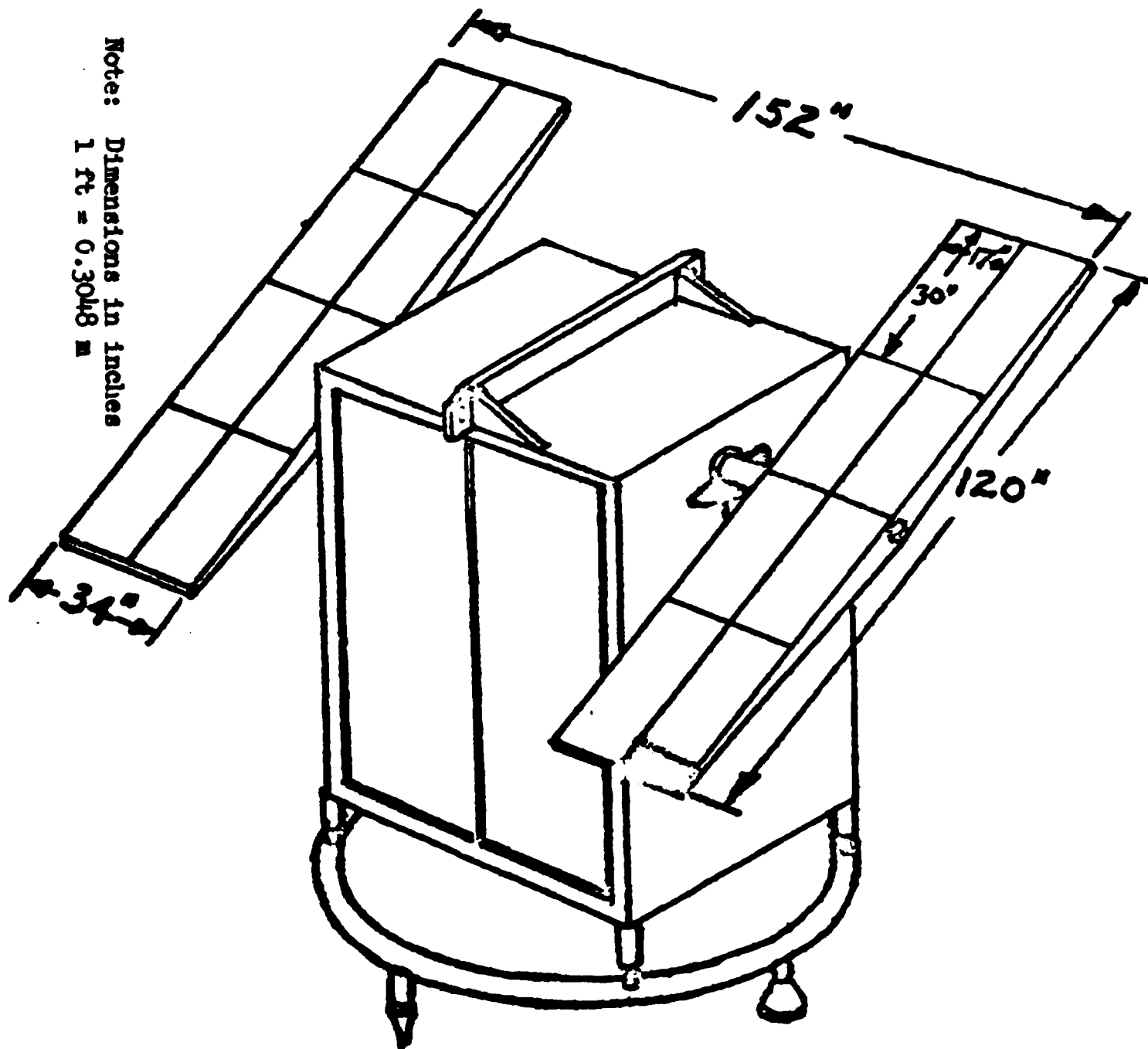


Fig. 7-41 Low-Cost SEO Solar Array



Fig. 7-42 Small Research Satellite Electrical Power Subsystem

7.7 ATTITUDE CONTROL SUBSYSTEM DESIGN GUIDELINES

7.7.1 Basic Recommended Design Approaches

In this document, by definition, the Attitude Control subsystem is limited to systems that produce torques by the expulsion of mass for stabilization of spacecraft. Consideration of such systems at this time is further limited to those employing cold gas. Essentially simple, such systems are relatively low in cost compared to other spacecraft subsystems; however, modest cost reductions may be accomplished in their design by simplification and by the elimination of high cost materials. Furthermore, when weight and volume restrictions on spacecraft design are relaxed, cold gas systems may be used to the exclusion of such costly control systems as momentum wheels and magnetic torquing systems.

7.7.2 Example of Low-Cost Designs of Attitude Control Subsystems

A low-cost Attitude Control Subsystem for an OAO-type payload has been designed to replace the equivalent subsystem of the initial-design OAO-B.

a. Initial-Design OAO Attitude Control Subsystem - This subsystem comprised:

- Coarse Momentum Wheels (3 wheels)
- Primary Pneumatic (cold gas) System
- Secondary Pneumatic (cold gas) System
- Magnetic Wheel Unloading System

These systems are described in succeeding paragraphs.

- (1) Coarse Momentum Wheels - The coarse momentum wheels were used to slew the spacecraft from one pointing attitude to another. Each wheel and its drive motor were an integral unit. The motor was built "inside-out" with the stator inside rather than outside of the rotor. Thus, the rotor had a larger moment of inertia than it would have if it were a conventional design. The motor was a 40 pole, two-phase device that could not start or continue to run unless voltage was present in both phases. The coarse wheels operated intermittently.

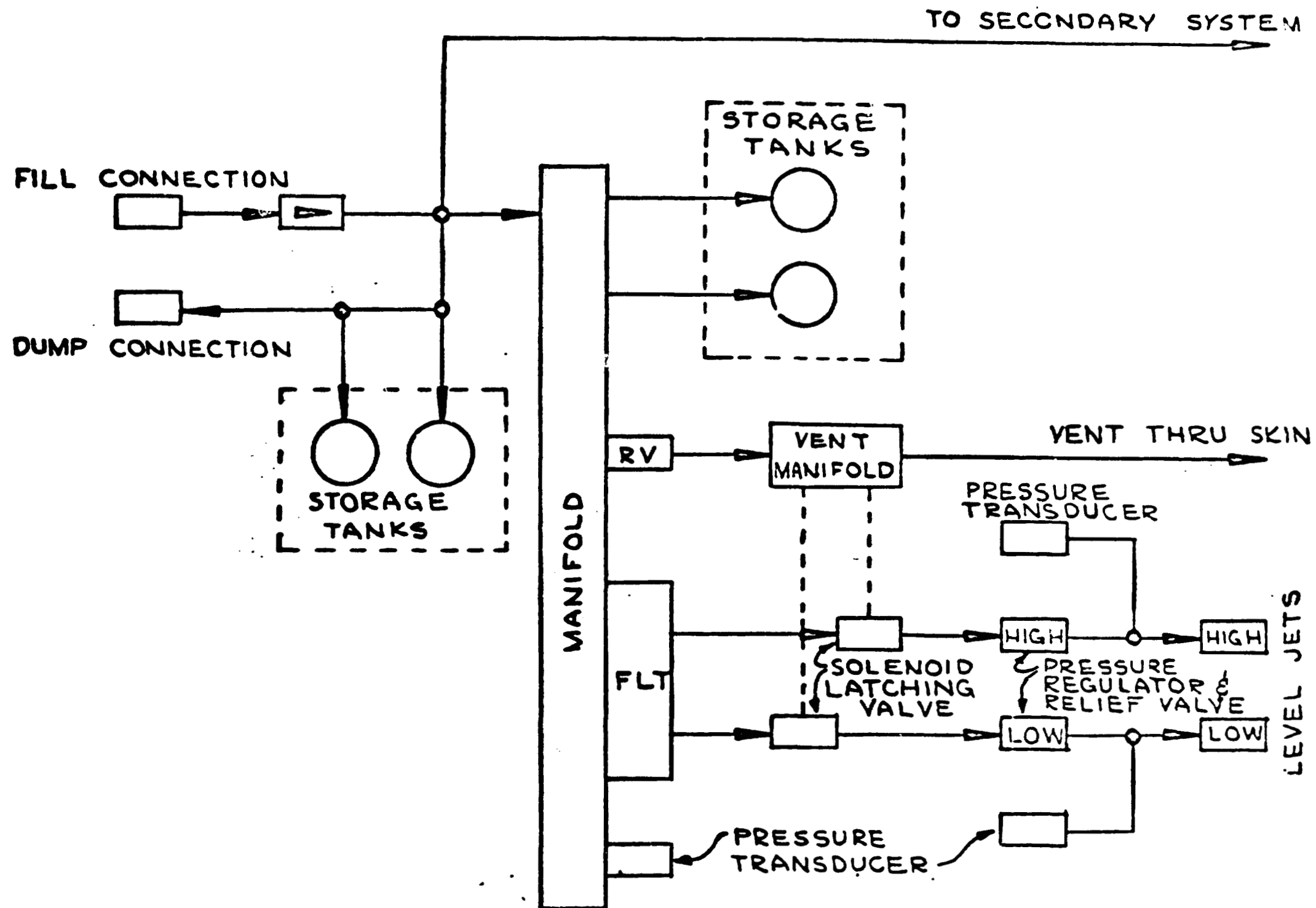
- (2) Pneumatic System - The pneumatic system performed the functions of providing reaction mass and thrust for initial stabilization of the OAO, for torque unloading of the OAO and for RAPS (Rate and Position Sensor) control of the OAO. It was composed of a primary system and a secondary system which were interconnected such that both systems were "filled" through a common fill fitting and could be "dumped" through a common dump fitting. Both the primary and secondary high thrust jets were normally used for initial stabilization or restabilization. The primary low thrust jets were used for RAPS sunbathing or attitude-hold modes of operation.

The primary pneumatic system (Fig. 7-43) used pressurized dry nitrogen propellant stored in four spherical tanks, each capable of holding 8 lbs (3.6 kg) of dry nitrogen at a pressure of 3250 psi ($22 \times 10^6 \text{ N/m}^2$).

The Secondary Pneumatic System, Fig. 7-44, was almost identical to the Primary System. There were four tanks holding 8 lbs (3.6 kg) of dry nitrogen each at 3250 psi ($22 \times 10^6 \text{ N/m}^2$) connected to the manifold. Normally during initial stabilization and roll search, both Primary and Secondary systems would be operating. Either the primary or secondary system could be shut off by ground command. This might be desired if small corrections in vehicle rates or orientation were desired or if the jets on one system were stuck open. The secondary high thrust jets, only, were fired by signals from the RAPS controller.

The 18 nozzle assemblies used in both the Primary and Secondary Pneumatic systems were similar. All assemblies utilized the same type of solenoid shutoff valve mechanism. The jet nozzle ends differed with some set at an angle other than 90 deg to the structural surface to which they were secured.

In space, thrust per nozzle at nominal pressures were 0.1 lbs (0.44 N) for the High Thrust Jets and .0021 lbs (.009 N) for the Low Thrust Jets.



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Fig. 7-43 OAO-B Primary Pneumatic System

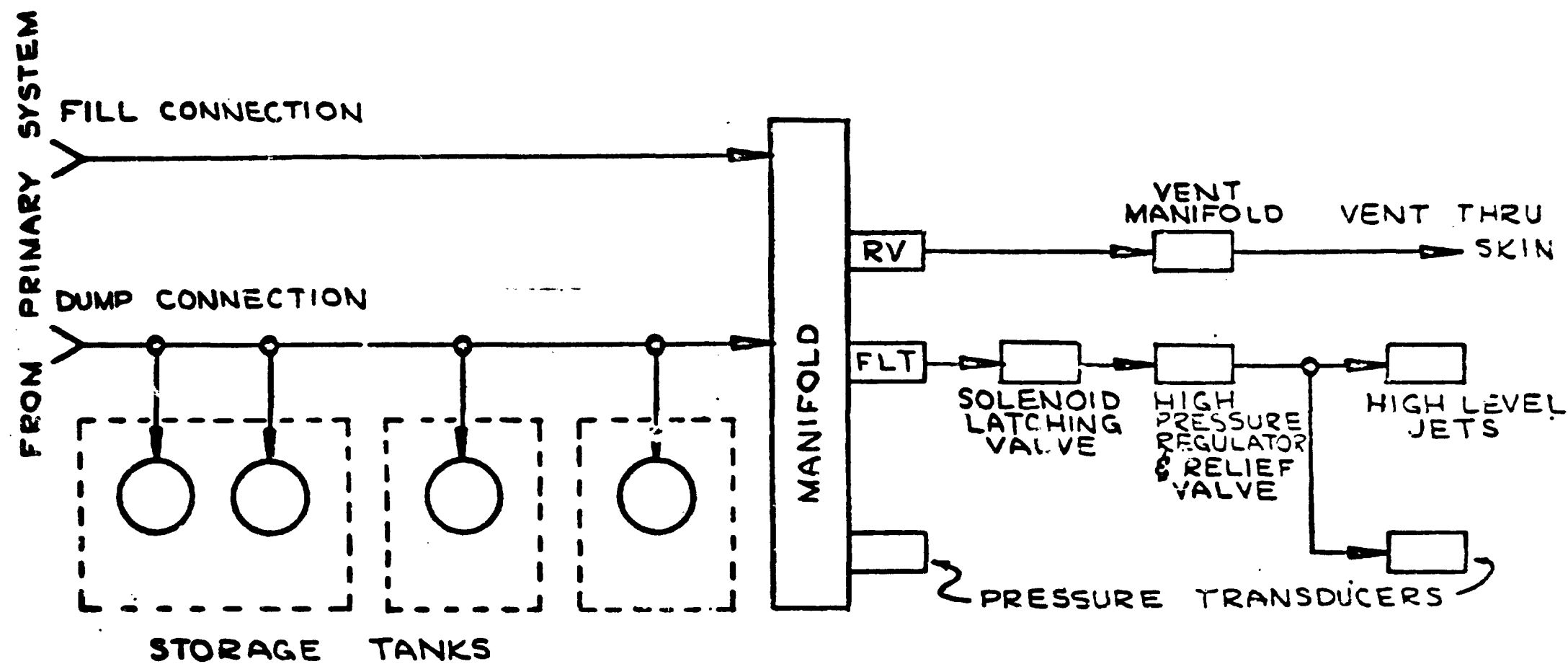


Fig. 7-44 OAO-B Secondary Pneumatic System

- (3) Magnetic Unloading System - The Magnetic Unloading System (MUS) was a device on the OAO that provided continuous unloading of extraneous torque on the spacecraft by the generation of a magnetic field which interacted with Earth's magnetic field in such a manner as to produce a torque in the opposite sense of the disturbing torque. Although the MUS was considered as a backup for the Low Thrust Jets, where the MUS could develop sufficient torque to counteract the extraneous torques, it was considered as the preferable system since its reaction was smoother and its use obviated the need to fire the Low Thrust Jets which resulted in a net saving in reaction mass.

It was also used to unload the continuously-running fine momentum wheels, part of the Stabilization and Control Subsystem.

The MUS consisted of a Magnetometer, and its electronics, which sensed the components of the Earth's magnetic field along each of the vehicle axes, a Signal Processor which performed an algebraic summation of the Magnetometer's outputs and the signals from the FWJC's, and three Torquer Bars with their associated electronics. One bar was parallel to each vehicle axis. The Torquer Bars generated the magnetic fields that interacted with the Earth's field and produced a torque on the spacecraft.

- b. Low-Cost OAO Attitude Control Subsystem. The low-cost OAO Attitude Control Subsystem is a conventional cold gas control system that replaces the Coarse Momentum Wheels, the Pneumatic System, and the Magnetic Unloading System of the initial-design OAO-B. It provides thrust for stabilization and slewing of the spacecraft, for unloading the fine momentum wheels, and for fine-pointing of the spacecraft if the fine momentum wheels fail. It consists of four identical modules that are installed 90 deg apart in the aft section of the OAO. Each module contains three high level thrusters and three low level

thrusters. The high level thrusters are used for stabilization and slewing of the vehicle. The remaining functions are performed by the low level thrusters. The schematic diagram of the module is presented in Fig. 7-45.

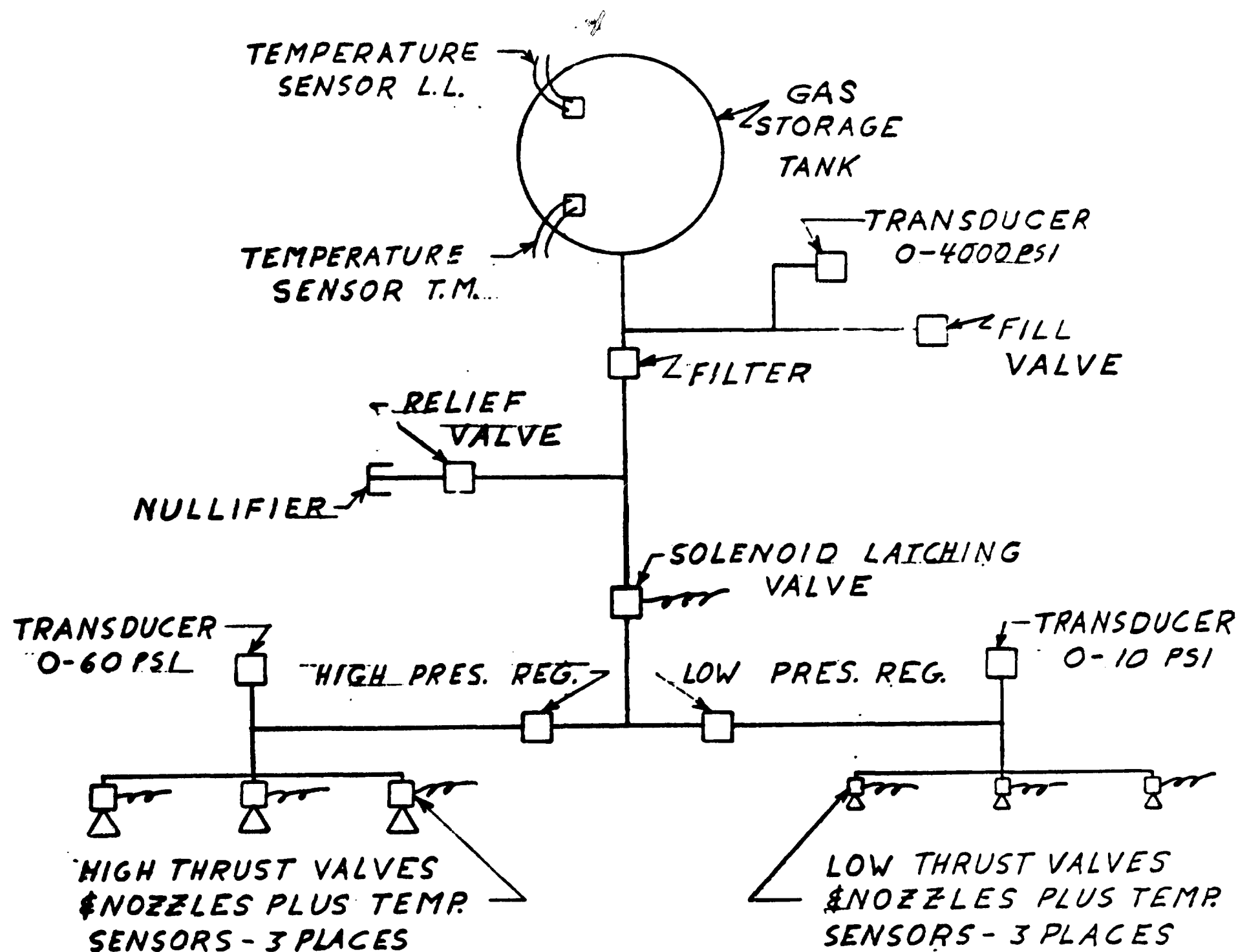
The gas storage tank is capable of holding 100 lbs (45 kg) of dry Freon 14 at a pressure of 3000 psi ($21 \times 10^6 \text{ N/m}^2$), providing a total impulse of 4500 lb-sec (20,000 Ns). The tank is pressurized through a fill valve connected to the tank. The fill valve contains a needle valve which is opened to fill or bleed the tank and closed to maintain pressure in the tank. A 5 micron filter is connected into the outlet side of the Tee fitting leading to the solenoid latching valve. Also connected into line between the tank and the solenoid latching valve is the relief valve. This valve opens and vents gas when pressure rises above 3400 psi ($28 \times 10^6 \text{ N/m}^2$) for any reason. The solenoid latching valve is normally open and is closed when there is a leak in the downstream system, or when there is an indication that either regulator is malfunctioning and causing an abnormal pressure to be supplied to the thrusters. The next component in the system is the high pressure regulator which reduces the tank pressure to 35 psi ($0.24 \times 10^6 \text{ N/m}^2$) for the three high level thrusters. Each thruster produces 0.1 lb (0.44 N) thrust.

Also downstream of the solenoid latching valve the line branches to the low pressure regulator which reduces the tank pressure to 5 psi ($3.4 \times 10^4 \text{ N/m}^2$) for the three low level thrusters. Each low-level thruster produces 0.0021 lb (0.009 N) thrust.

The low-cost OAO Attitude Control subsystem module is designed to be replaced in orbit by a Space Shuttle crewman and hence emphasis is on safety. All components such as valves and regulators are operated at less than one-third of proof pressure; and the gas storage tank is designed to boiler-code standards. The tank is made of aluminum rather than titanium to reduce its cost. The resulting weight penalty is acceptable to the low-cost OAO.

The costs of the low-cost OAO Attitude Control Subsystem are compared with the costs of the OAO-B pneumatics system in Fig. 7-46.

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Note: 1 psi = 6895 N/m²

Fig. 7-45 Low-Cost OAO Attitude Control Subsystem Module

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Spacecraft	Baseline Subsystem Costs (millions of dollars)		Low-Cost Subsystem Costs (millions of dollars)	
	RDT&E	Average Unit	RDT&E	Average Unit
QAO*	5.10	1.50	4.30	1.0.

* QAO-B Pneumatic System Only.

Fig. 7-46 Cost Comparison Attitude Control Subsystems

Though appreciable savings are indicated they are not the total savings resulting from the low-cost design. The cost savings resulting from the elimination of the coarse momentum wheels and the magnetic unloading system are included in the cost savings accomplished by the redesign of the OAO Stabilization and Control Subsystem (discussed previously in this section).

7.8 ENVIRONMENTAL CONTROL SUBSYSTEM DESIGN GUIDELINES

7.8.1 Basic Recommended Design Approaches

The Environmental Control subsystem comprises primarily the on-orbit thermal control system. Passive thermal control techniques are used almost exclusively in unmanned spacecraft and are relatively low in cost compared to other spacecraft subsystems. However, the design of the thermal control subsystem for complex spacecraft such as the OAO-B-type require extensive analysis, testing, and detail design of the installation of insulating materials. Appreciable cost savings can be made if such a thermal control system can be extensively simplified. Such simplification has been attained in the design of the thermal control system of the low-cost OAO discussed following.

7.8.2 Example of Low-Cost Design of a Thermal Control System

7.8.2.1 Initial-Design OAO-B Thermal Control System. The following description of the initial-design OAO-B Thermal Control system is quoted from NASA SP-133, "Scientific Satellites, page 366. "The OAO structure (Fig. 7-47) is basically a tube, 122 centimeters in dia. and 300 centimeters long. It is surrounded by 48 truncated equipment bays, arranged in an octagonal pattern. The sequestering of OAO equipment into small bays insulated from the main satellite structure is the key to successful passive thermal control. Superinsulation made of aluminized Mylar covers all but one side of each bay (Fig. 7-48). The bulk of the heat flowing out of (or into) each bay follows the path between the honeycomb mount and the aluminum satellite skin. Thus, the bay skins can be painted or finished in a manner appropriate to the enclosed equipment. The small equipment packages in the bays are each handled separately, as if they were small, passively controlled satellites (LMSC emphasis).

The major heat input to the main OAO structure and the contained telescope is leakage through the superinsulation of the equipment bays. By careful insulation and design, heat inputs to the structure through fittings and supports

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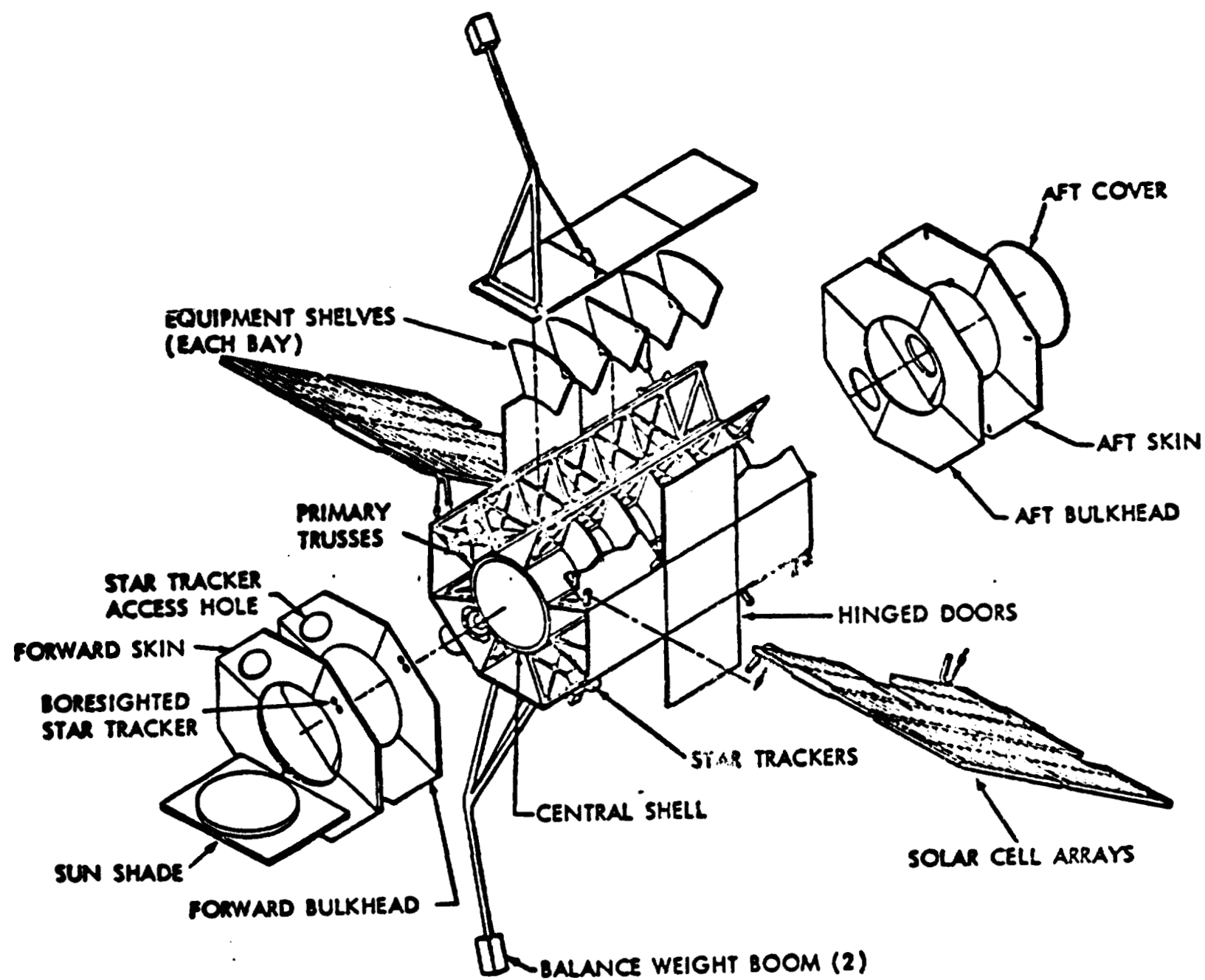
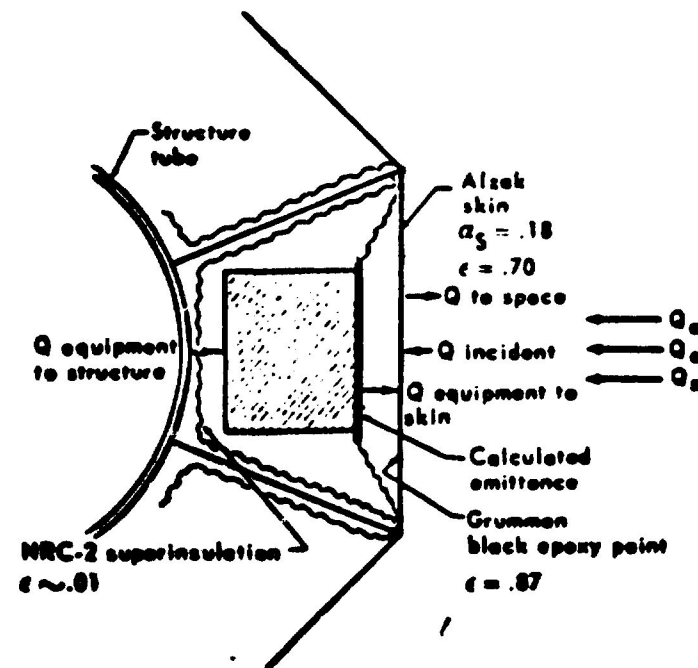


Fig. 7-47 OAO Spacecraft, Exploded View

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Plan view of an OAO equipment bay, showing an instrument package surrounded by super-insulation on all sides except the one facing the outer skin.



Side view of the OAO equipment bay.

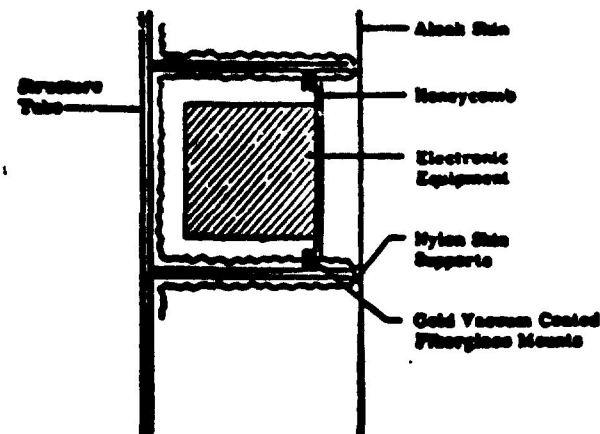


Fig. 7-48 Thermal Control of OAO-B Equipment Bays

are minimized. Heat leaves the structure through radiation escaping via the open tube ends and heat transfer to nonequipment bays and end skin sections. Since the heat sinks are difficult to control, the heat flow into the structure is varied by changing the amount of superinsulation around the bay walls. The design temperature of the cylinder and contained telescope in the OAO is about $-30^{\circ} \pm 15^{\circ}\text{C}.$ "

The most significant statement in the above description of the OAO-B Thermal Control system is the following: "The small equipment packages in the bays are each handled separately, as if they were small, passively controlled satellites." Such a design requires a very extensive analytical effort and proliferation of detail design data. In addition, the testing required for verification of the design is extensive and complex.

7.8.2.2 Low-Cost OAO Thermal Control System. The low-cost OAO Thermal Control System consists only of passive thermal control materials as shown in Fig. 7-49. The thermal isolation of the Experiment Package from the Equipment Section, made possible by the elimination of weight and volume constraints, results in significant simplification of the thermal control of the Low-Cost OAO relative to that of the baseline OAO-B.

When the Experiment Package is isolated from the Equipment Section, the temperature of the latter may be permitted to vary over a relatively wide range, thermal isolation and control of individual equipment units is eliminated or greatly reduced, and significant cost savings are realized.

The estimated costs of the low-cost OAO Thermal Control System are compared with the costs of the OAO-B Thermal Control System in Fig. 7-50.

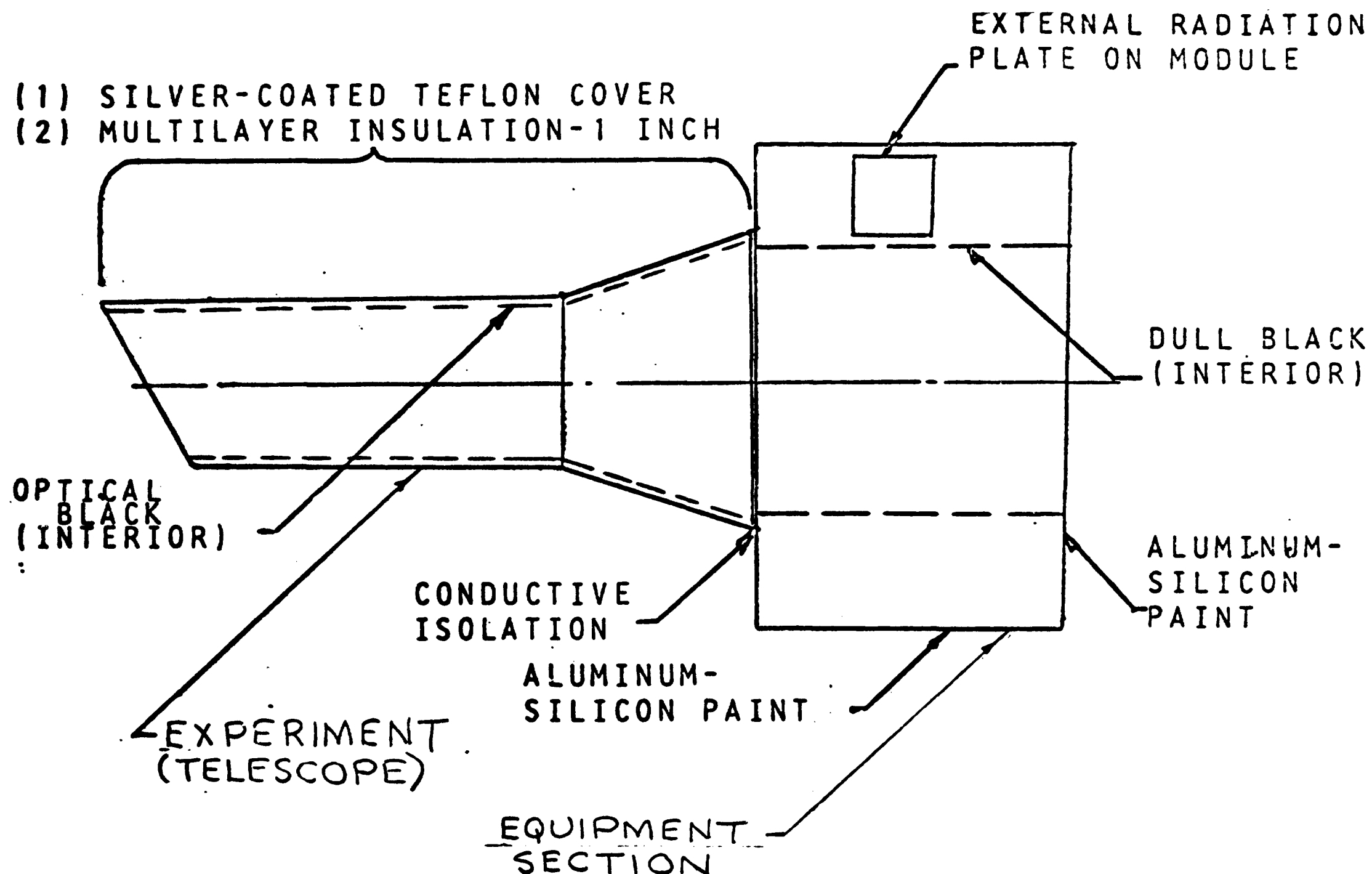


Fig. 7-49 Low-Cost OAO Thermal Control (Passive)

Cost Category	Baseline OAO-B Thermal Control Sys. (millions of dollars)	Low-Cost OAO Thermal Control Sys. (millions of dollars)
RDT&E	6.05	2.9
Average Unit	1.0	0.39

Fig. 7-50 Cost Comparison - Thermal Control Systems