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LMSC-D154696

DESIGN GUIDE
FOR
LOW COST STANDARDIZED
PAYLOADS

VOLUME II

30 APRIL 1972

CONTRACT NAS W-2312

LOCKHEED MISSILES & SPACE COMPANY, INC.
A SUBSIDIARY OF LOCKHEED AIRCRAFT CORPORATION

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DESIGN GUIDE
for
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Space Systems Division

LOCKHEED MISSILES & SPACE COMPANY

FOREWORD

This document, "Design Guide for Low-Cost Standardized Payloads" has been prepared by Lockheed Missiles & Space Company, Inc., Space Systems Division, as part of the total effort under NASA Contract NAS W-2312, covering the Payload Effects Follow-On Study. The specific effort for the Design Guide was identified as Task 1.9.

For ease of reference, the general information covering the design and effects of low-cost and standard-element payloads is contained in Volume I. This Volume II is a supplement volume, containing copies of the 16 Engineering Memos prepared by LMSC describing the point designs of payloads and their subsystems.

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DEFINITIONS OF TERMS

Many of the terms used herein have various connotations within the Aerospace community. Therefore, as a guide, some of the basic terms are defined below.

PAYLOAD SYSTEM — describes collectively: (1) the payload; (2) the payload/Shuttle adapters, and any deployment or separation devices required to effect a separation of the payload from the launch vehicle; (3) payload ground support equipment; (4) payload flight support equipment including spare module support racks, payload checkout equipment, and special payload umbilicals.

PAYLOAD — the total operating entity, such as a satellite, that is launched into orbit by the Shuttle; it comprises spacecraft and experiments but excludes Shuttle related elements - such as platforms or adapters - that are non-functional relevant to the orbiting satellite.

BASELINE PAYLOAD — a current unmanned payload used to provide a basis for the development of low-cost or standard payloads and for cost comparisons.

LOW-COST PAYLOAD — A payload designed for launch by the Space Shuttle or by a future large-expendable launch vehicle. Such a payload is designed (1) without the traditional costly constraints on weight and volume, and (2) for in-orbit repair or refurbishment.

SUBSYSTEM — A major functional group of equipment which is essential to the operation of a spacecraft. Spacecraft subsystems include:

- Structures & Mechanisms
- Electrical Power
- Stabilization & Control
- Attitude Control
- Communications, Data Processing & Instrumentation
- Environmental Control
- Propulsion

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.
STANDARD SUBSYSTEM	a major spacecraft subsystem (stabilization and control; communication, data processing, and instrumentation; electrical power; attitude control) designed for application to a significant number of mission-peculiar or standard spacecraft.
STANDARD SUBSYSTEM MODULE	a plug-in assembly of components forming a major segment of a standard subsystem, and having standard mechanical, electrical, and thermal interfaces.
STANDARD SUBSYSTEM MODULE VARIANT	a standard subsystem module modified by the addition, deletion, or substitution of a single component.
STANDARD SPACECRAFT	a small quantity of different types of spacecraft incorporating standard subsystem modules, each type capable of replacing a significant number of the mission-peculiar spacecraft defined in the NASA mission model. The spaceframe, integral wiring harnesses, and thermal control elements of each standard spacecraft type are standardized.
CLUSTER SPACECRAFT	a spacecraft incorporating standard subsystem modules and capable of supporting concurrently the experiment/sensor packages of several of the missions defined by the NASA Mission Model.
RELIABILITY	the probability that a system, subsystem, component, or part will satisfactorily perform its intended function without catastrophic failure for a prescribed period of time, within a prescribed environment.
CONFIDENCE LEVEL	the probability that the reliability figure-of-merit predicted for a system, subsystem, component, or part is correct.
REPAIR	an action taken to restore a failed system to an operating state. The action may be scheduled or unscheduled, and consists of: • Diagnosis of the failure condition • Removal of the failed system element • Replacement of the failed element with a similar element in operating condition • Checkout of the system post-maintenance to assure proper operation within prescribed limits.

PARTIAL REFURBISHMENT	A maintenance action expected to prevent future failure. In this study it is assumed that when a repair visit to the system becomes necessary to repair a failed system element, other system elements which have not yet failed will be approaching their theoretical point of first failure. These latter elements will be removed also and replaced as assurance that the system will be protected against failures occurring soon after a repair visit.
FULL REFURBISHMENT	A maintenance action (analogical to a complete overhaul) occurring shortly prior to, or at the theoretical MMD point of the system, where MMD denotes the useful operating life terminal point as dictated by the limits of the design. The action consists of removal and replacement of all dynamic system elements, whether or not they have exhibited failure. Following full refurbishment, the spacecraft is assumed to be in the "as new" state and capable of operating another period equal to the spacecraft MMD.
MAINTENANCE LEVEL (SPACECRAFT)	The hardware level at which maintenance action takes place. Since the systems in question are modularized at the subsystem level, all maintenance actions are confined to removal and replacement of the module, or modules exhibiting failure, or approaching a theoretical failure point.
MAINTENANCE INTERVAL	That period of elapsed time between any one maintenance action and the next, as scheduled in the overall maintenance program. The interval is predicated upon expectable wearout rates, and expected failure incidence.

N O T I C E

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INTRODUCTION

1.1 INITIAL LOW-COST PAYLOAD DESIGNS

In the initial Payload Effects Analysis Study (NAS W-2156), three low-cost refurbishable payloads were designed, based upon the design/performance requirements of the following baseline payloads:

Payload	Cognizant Agency
• Orbiting Astronomical Observatory (OAO)	NASA/Goddard (Grumman)
• Synchronous Equatorial Orbiter (SEO)	NASA/Langley (Boeing)
• Small Research Satellite	USAF (LMSC)

The low-cost designs were described in detail in LMSC Engineering Memos, copies of which were supplied to NASA agencies for review/comment. These memos, now used only for historical reference, are listed below. Excerpts of these memos were included in the initial "Design Guide for Space Shuttle Low-Cost Payloads", LMSC-A990558, dtd 30 June 1971.

Payload	LMSC Engineering Memorandum (Not Published)				
	Payload General Description	Payload Subsystem			
		Stabilization and Control	Communication Data Processing Instrumentation	Electrical Power	Attitude Control
Orbiting Astronomical Observatory	PE-7	PE-2	PE-3	PE-4	PE-5
Synchronous Equatorial Orbiter	PE-27	PE-22	PE-23	PE-24	PE-25
Small Research Satellite	PE-47	PE-42	PE-43	PE-44	PE-45

1.2 FOLLOW-ON PAYLOAD EFFECTS STUDY

The results of the initial study effort indicated a dramatic impact of payloads upon the 1979 and beyond space program. In fact, it appeared that unless payload savings could be implemented (by use of low-cost payload design/manufacturing/testing techniques and by refurbishment/reuse of payloads), the Shuttle program would not be economically feasible. For this reason, a follow-on study was sponsored by NASA/HQ, with co-direction from two directorates, OSSA and OMSF.

The new study had the following principal objectives:

- Create additional point designs of future spacecraft, incorporating not only features of low-cost and refurbishability, but also establishing spacecraft hardware standardization at three levels*:

    (1) Standard Subsystems and Modules
    (2) Standard Spacecraft
    (3) Cluster Spacecraft

- Prepare program plans and cost estimates for the low-cost standardized spacecraft hardware and establish dollar savings relative to baseline (traditional design) payload programs.

- Determine effect of the new payload designs upon the Space Shuttle system and the constraints which the Shuttle (in its current configuration) might place upon full realization of future payload cost reduction potential.

- Prepare a Designers Guide (as a sequel to the initial document mentioned in par. 1.1), updating the principles of low-cost payload design for Space Shuttle application and providing additional methodology for design and application of standard hardware to future spacecraft.

1.3 THE UPDATED DESIGN GUIDE

This document, "Design Guide for Low-Cost Standardized Payloads", in two volumes, is the result of effort on the last objective in the above list.

Volume I summarizes the economic impact of low-cost, standardized spacecraft hardware and provides special information pertinent to implementation of the critically-needed payload cost-reduction principles.

This volume, Volume II, contains 16 preliminary LMSC Engineering Memorandums documenting the point designs of spacecraft hardware that were carried out during the Payload Effects Follow-On Study. The Engineering Memorandums are identified in Fig. 1-1.

By agreement with NASA/HQ, these Engineering Memorandums are informal documents prepared without the constraints of format normally imposed on published material. The intention is to convey maximum technical data for minimum cost, which is a major objective of the Payload Effects Studies.

Copies of these documents are contained herein for general reference purposes.

1.4 CONTENT ARRANGEMENT

A copy of each Engineering Memo is contained in this volume. Each is preceded by a separator sheet upon which the number and title appears. Additionally, the separator sheet bears the "PE" number of the memo on the right-hand edge as an index aid.

All memos are in numerical order by PE number, PE-102, PE-103, etc., and are listed in the Table of Contents.

Payload	IMSC Engineering Memorandum (Preliminary)					
	Payload General Description	Payload Subsystem				
		Stabilization and Control	Communication Data Processing Instrumentation	Electrical Power	Attitude Control	Propulsion
Earth Observatory Satellite	PE-106	PE-102	PE-103	PE-104	PE-105	-
U.S. Domestic Communication Satellite	PE-126	PE-122	PE-123	PE-124	PE-125	-
Mars Viking	-	-	PE-133	-	-	PE-137
Standard, Large Astronomical Observatory Spacecraft	PE-146	PE-146	PE-146	PE-146	PE-146	-
Standard Earth Observatory Spacecraft	PE-156	PE-156	PE-156	PE-156	PE-156	-
Cluster Earth Observatory Spacecraft	PE-166	PE-166	PE-166	PE-166	PE-166	-
Cluster Astronomical Observatory Spacecraft	PE-186	PE-186	PE-186	PE-186	PE-186	

Fig. 1-1 IMSC Engineering Memos Contained in Vol. II

PE-102

STANDARD EOS

STABILIZATION AND CONTROL

SUBSYSTEM

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: STANDARD EOS STABILIZATION AND CONTROL SUBSYSTEM	EM NO: PE-102 REF: DATE: 30 November 1971
AUTHORS: R. J. Pollak	APPROVAL: ENGINEERING F.C. Bolton Wayne Muller

60 pages

PRELIMINARY

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LIST OF ABBREVIATIONS

ACP	Attitude Control Propulsion
C/DP/I	Communications/Data Processing/ Instrumentation Subsystem
EOS	Earth Observatory Satellite
FHT	Fixed Head Star Tracker
FOV	Field-of-View
GP	General Purpose
IRA	Inertial Reference Assembly
PMB	Pitch Momentum Bias
RGP	Rate Gyro Package
SAS	Solar Aspect Sensor
S&C	Stabilization & Control
TARS	Three-Axis Rate Sensor (Same as IRA)

SUMMARY

The standard EOS Stabilization and Control Subsystem is required to maintain the spacecraft in an earth-oriented attitude for up to one-year and provide readout for ground-based precision attitude determination. The principle of operation of the primary attitude determination function is to obtain accurate long-term attitude information from a star sensor and a computer-stored star catalog, while three-axis rate sensor gyros provide a precise short-term reference. A computer processes and mixes the information from both sources. A check on the vehicle orientation in space is available to ground stations by readout of solar aspect sensors. These sensors would also be used for attitude reacquisition should a catastrophic event cause the spacecraft to tumble.

Attitude control torques are obtained by varying the speed of reaction wheels or pulsing attitude control thrusters. Momentum absorbed by the wheels is continuously reduced as the magnetic torquers interact with the ambient earth field under control of the CDPI computer. Should an unexpectedly large disturbance torque cause the wheels to saturate (reach maximum speed); desaturation is accomplished by torquing the spacecraft with the attitude control jets.

Reorientation from one attitude reference to another, if required, is accomplished by a series of slews using the wheels and/or gas jets which also control TARS-detected attitude errors about the non-slew axes.

The standard EOS S&C Subsystem is implemented entirely with equipment planned for the Standard Spacecraft S&C inventory. The options available to the user and the rationale for their selection will be presented in a forthcoming EM.

The factors leading to the decision to utilize star sensing/tracking for the primary attitude reference are listed in Appendix A. In summary, it can be shown that the need for high precision spacecraft attitude determination makes a less accurate (e.g., horizon) sensor redundant. The ready availability of an up-to-date ephemeris to the spacecraft via the TDRS, the low-cost of on-board attitude calculation offered by fourth-generation, general-purpose aerospace processors, the existence of a number of fixed head (gimballess) star tracker suppliers, and the benefits of development cost-sharing common to all high-utility standard/cluster/special-purpose spacecraft components far outweigh the (possible) higher initial cost and natural tendency to specify direct attitude measurements for earth-oriented spacecraft.

The standard EOS S&C subsystem utilizes two fixed-head star trackers (FHT) with narrow-angle (~ 5 deg) optics for both on-board (coarse) attitude determination and for after-the-fact ground (precision) attitude determination. Since, after the first orbit or two, all accuracy requirements can be met with but one FHT, the second unit provides essentially full mission redundancy.

Subsystem design trade considerations, described in Appendix A, are:

(1) Earth horizon vs Star Sensing

(2) Fixed head vs Gimballed head star trackers

(3) Reaction Wheels vs mass expulsion for attitude control torques

(4) Magnetic vs Mass Expulsion for wheel unloading

(5) Pitch Momentum Bias (PMB) vs Zero momentum system

(6) Three-axis magnetometer vs software model of the earth's magnetic field

(7) Two gyro triads vs skewed six-pack

(8) Combined reaction wheel-horizon sensor vs separate sensor and wheels

I. PROBLEM STATEMENT

A. Ground Rules and Constraints

The Stabilization & Control (S&C) Subsystem will be designed under the following ground rules and constraints:

1. Space Shuttle/Space Tug-launched
2. 1971-72 technology (concepts reduced to practice)
3. One-year life without refurbishment; up to three reuses
4. Subsystem reliability goal: 0.85
5. Low-cost: Trade weight increase for cost reduction
6. Minimize number of variants: Favor over-design in preference to a multiplicity of mission-peculiar parts
7. Subsystem is checked out and initialized at shuttle separation

Figure 1 shows the standard EOS spacecraft arrangement.

B. Functional Requirements

The Standard EOS Spacecraft Stabilization & Control Subsystem has the following functions:

(1) To stabilize the EOS spacecraft following shuttle separation, and establish the preselected attitude with a precision of ± 0.5 deg.

(2) To hold the spacecraft in the reference attitude with the required accuracy over periods of time up to 1 year with growth capability to 2 years.

(3) To provide a measure of instantaneous spacecraft attitude of sufficient accuracy to permit ground-based after-the-fact attitude determination within ± 0.002 deg (1σ).

(4) To hold the spacecraft attitude rates to less than ± 0.005 deg/sec, all axes.

(5) To reorient the spacecraft to the reference attitude from any attitude, following loss of reference due to reversible system failures, for tumbling rates up to 10 deg/sec.

(6) To point the spacecraft at the sun with near zero attitude rates following primary subsystem failure.

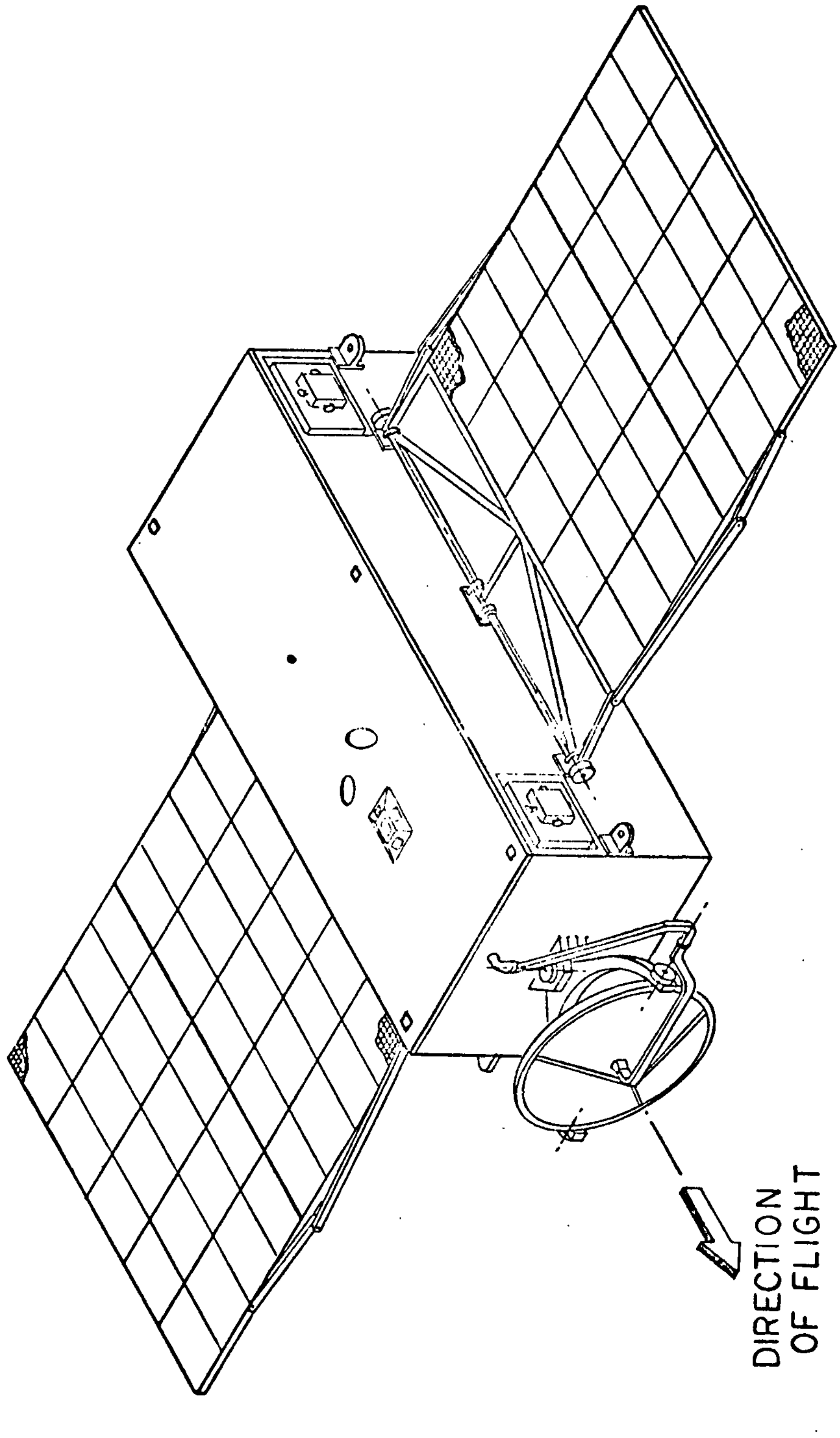

Fig. 1 Configuration Standard EOS Spacecraft

SC: 1" = 6.4'

II. DESCRIPTION OF STANDARD EOS S&C SUBSYSTEM

A. Subsystem Operation

(1) Primary Mode

Figure 2 shows the functional and equipment relationships. The three-axis rate sensors (TARS) and one or two fixed-head star trackers (FHT) 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 or gas jets to supply three-axis stabilization and control.

The S&C subsystem functional flow (Fig. 3) uses the on-board CDPI computer for attitude determination. The initial attitude at shuttle separation has been stored in the computer. The TARS continuously measures attitude changes which are processed in the computer to update the stored value. Thus, a numerical record exists of the spacecraft current attitude in an inertial coordinate frame. This attitude is compared to a stored reference attitude and the errors used to drive the wheels or jets. Note that this entire operation is proceeding independently of the star tracker references, which are, in effect, "off-line". The attitude reference sensor outputs are periodically sampled and optimally blended with the gyro-determined attitude to yield a best-estimate update to the attitude in storage at that instant but the sensor operation is decoupled from spacecraft attitude control. (Fig. 4)

The subsystem design approach takes advantage of the repeatability and stability of inertial-grade gyros for measurements of high data-rate attitude changes. Because random variations in gyro parameters are very small (better than 0.01 deg/hr) only discrete updating is necessary. The periodic star fixes, via the filter in the computer, bound long-term attitude errors and, at the same time, update the estimated random gyro drifts, scale factor errors, and alignment biases.

In effect, the gyros provide high bandwidth attitude data with unbounded errors (drift) whereas the star sensor provides low bandwidth data with bounded errors. After combination, these data yield high bandwidth attitude knowledge with bounded errors (Ref. 4).

(2) Secondary (Anti-Tumbling) Mode

A significant portion of the anticipated cost savings associated with the Shuttle derives from the planned return to the payload for maintenance, resupply, or recovery for refurbishment/repair.

It is axiomatic that it will be necessary to approach and grasp the satellite prior to performing any of these activities. While various special techniques and mechanisms can no doubt be conceived which will permit this if the satellite is tumbling randomly about any or all of its axes, it seems clear that it would be simpler, safer, and cheaper to assure that the satellite will be stabilized at Shuttle arrival. This is an easy requirement to satisfy: the main decision required is to what degree will an independent capability be provided. The equipment required is:

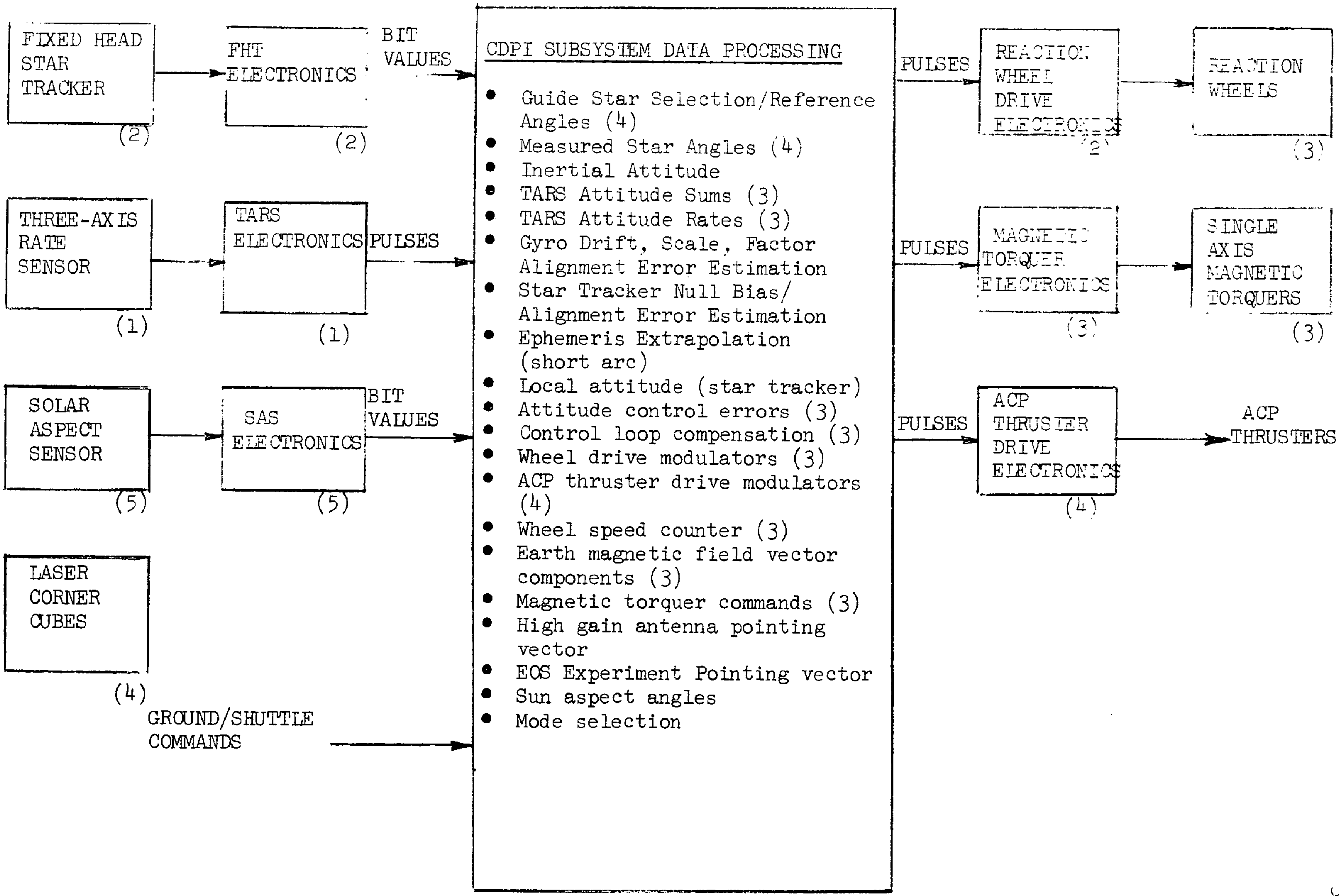

Fig. 2 Standard EOS Stabilization & Control Subsystem - Primary Mode (including Redundant Equipment)

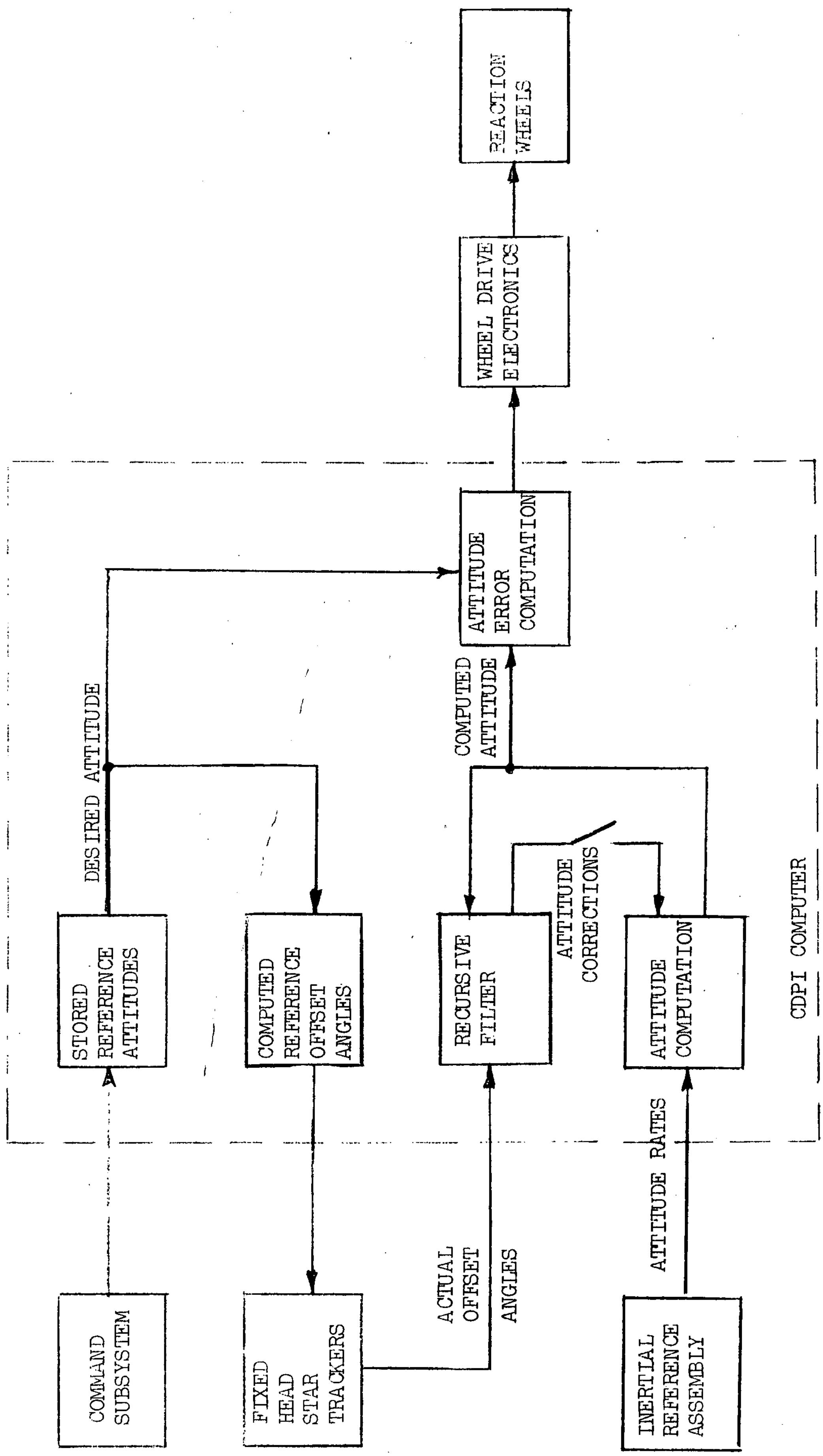

Fig. 3 Stabilization and Control Functional Flow

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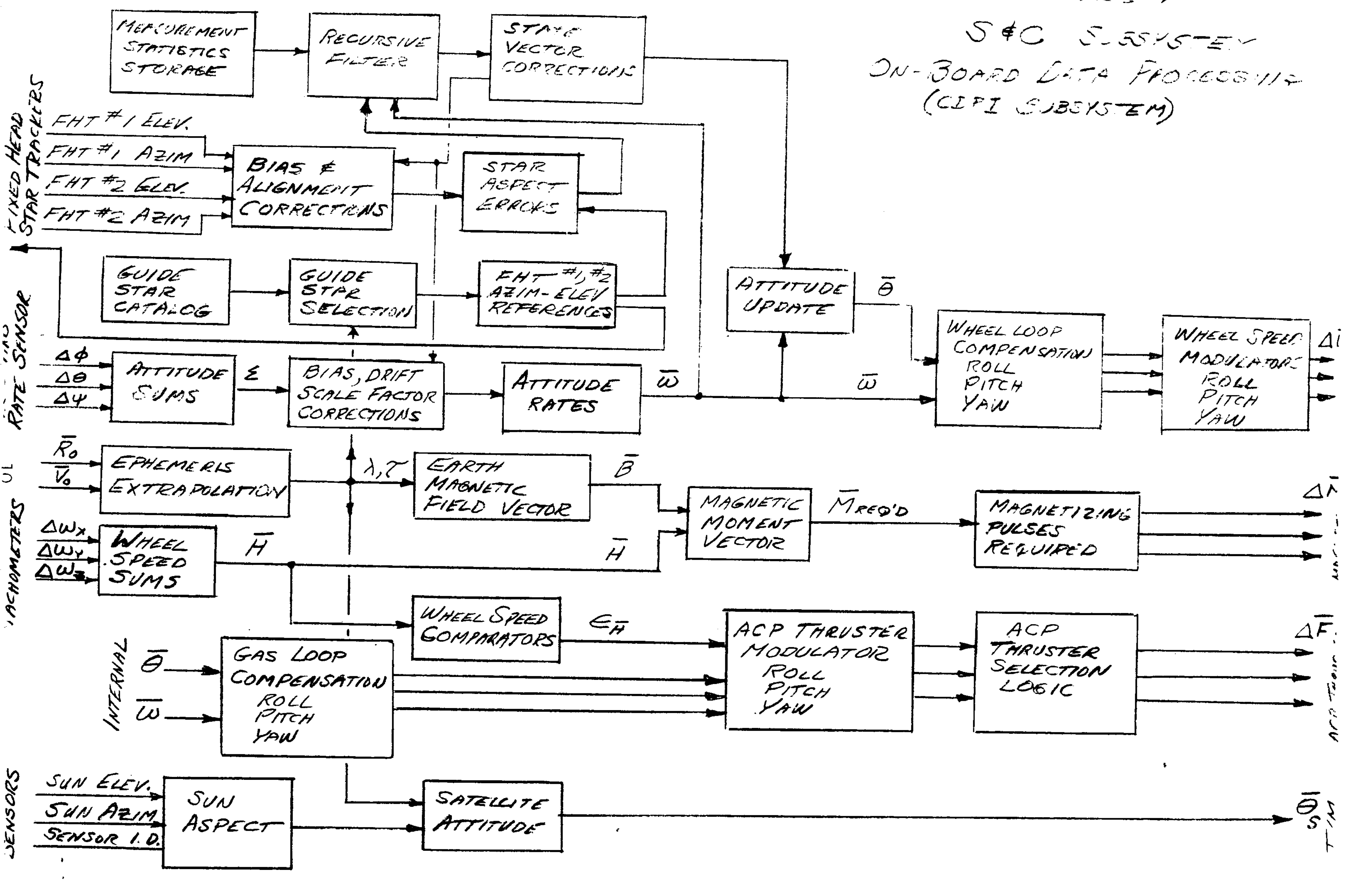
FIGURE 4

S&C SUBSYSTEM

ON-BOARD DATA PROCESSING

(CIPI SUBSYSTEM)



(1) 3-axis rate gyro package (S&C)
(2) Coarse sun sensor (S&C)
(3) Electronics package (S&C)
(4) Cold-gas jets (3-axis)
(5) Cold Gas Tank
(6) Command Receiver/Decoder
(7) Battery
(8) Payload Shutters

Ideally, these parts would be packaged as an entity, independent of the primary S&C subsystem. At the other extreme, no special parts would be added, the functions all provided by the primary equipment. In this case there would obviously be much less assurance of having a quiescent satellite since the electronics and gas jets are tied to the normal system operation.

Figure 5 shows the functional and equipment relationships of the secondary operating mode, used only after an irreversible failure of a piece of gear of the primary mode, including the CDPI computer.

The backup or anti-tumbling system would either be commanded "on" or automatic under some set of on-board logic. The sequence of events would be:

(1) Rate Stabilization - Rate gyros null all rates

(2) Roll Search - Bias signal starts roll rate using cold gas

(3) Sun Acquisition - Sun in sensor FOV removes rate bias signal; sensor and rate gyros drive attitude to lock on sun at desired solar aspect.

(4) Attitude Hold - Sun sensor and rate gyros hold satellite to sunline. (Very slow rotation about sunline due to gyro threshold possible.)

B. Subsystem Implementation

Equipment Complement

The S&C subsystem implementation represents one variant of the Standard Spacecraft S&C Subsystem.

Table 1 shows the S&C subsystem equipment list with estimated weights and power.

Associated with each star tracker, the reaction wheels set, each solar aspect sensor, the three-axis rate sensor, and each magnetic torquer is an electronics package. Its functions will be tailored to produce a low bit rate digital electronic interface with the computer.

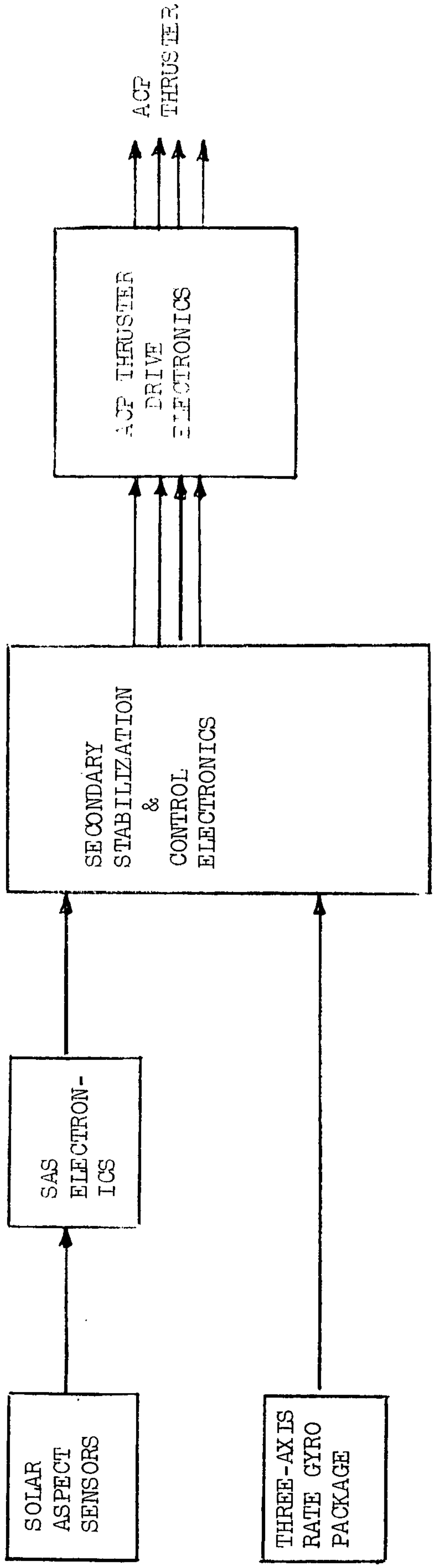


Fig. 5 Standard EOS Stabilization & Control Subsystem - Secondary

Table 1

LOW-COST EOS STABILIZATION & CONTROL SUBSYSTEM**

Item	Qty.	Unit Weight (lb)	Unit Power (watts)	Est. Failure Rates (hr x 10^6) & Duty Cycles (in %)	Total Weight (lb)	Total Avg. Power (watts)
Three Axis Rate Sensor/ Electronics	1/1	15	30	10 (100)	15	30
Fixed Head Star Tracker/ Electronics	2*/2*	13	11	5 (100)	26	22*
Sun Aspect Sensor/ Electronics	5/5	1/2	0/1	3 (1)	15	5
Reaction Wheels/ Electronics	3 /2*	18/9	4/9	2/6 (100)	72	21
Magnetic Torquers/ Electronics	3/3	2/1	1/30 (Pulse)	1(100/1)	9	4
Laser Corner Cubes	4	2	Passive	Passive	8	Passive
ACP Drive Electronics	4	7	12	3(25)	28	12
Rate Gyro Package (Backup)	1	2	15	10(1)	2	15
Secondary S&C Electronics (Backup)	1	5	5	10(1)	5	5
					180 lb	114 W

* Including redundancy, if required.
** Not including module weights or component mounting bases, brackets, pads, electrical connectors, cables, etc.

A brief description of the functions of each part and specifications of currently-available units follows.

(1) Three-Axis Rate Sensor (TARS)

The functions of the rate sensor are to:

(a) Provide damping signals for attitude stabilization,

(b) Detect and measure all body angular motions for input to the computation of vehicle attitude,

(c) Measure body rates during attitude maneuvers.

The three-axis rate sensor uses three single-degree-of-freedom rate integrating gyros mounted with their input axes forming a nominally-orthogonal triad. The TARS outputs are three asynchronous trains, each pulse representing a fixed increment of rate experienced by the vehicle about the respective gyro input axis. Each pulse in a train is summed in the CDPI computer in a separate up/down counter. The contents of the counters are periodically strobed into the computer for solution of attitude change over the sample period. The benign angular motion of the spacecraft enables this solution to be accomplished at a relatively low frequency without the usual concern for "coning" errors of strapdown attitude computations.

Suitable units are produced by Bendix, Honeywell, Kearfott, and Nortronics. Figures 6 and 7 and Table 2 describe the Honeywell GG 2200.

(2) Fixed Head Star Tracker (FHT)

The Fixed Head Star Tracker is a two-axis sensor designed to provide high accuracy error signals from celestial targets (stars and planets). Fixed head trackers have been designed for a variety of applications, including fine tracking for three-axis rocket control systems and satellite stabilization control loops.

The FHT has the capability to automatically acquire and precisely track a star electronically, without the use of mechanical gimbals or rotating components (Fig. 8). The star tracker has the ability to scan a large field-of-view, select a star in the field, switch to a much smaller field of view, and provide two axis position information with excellent resolution on the

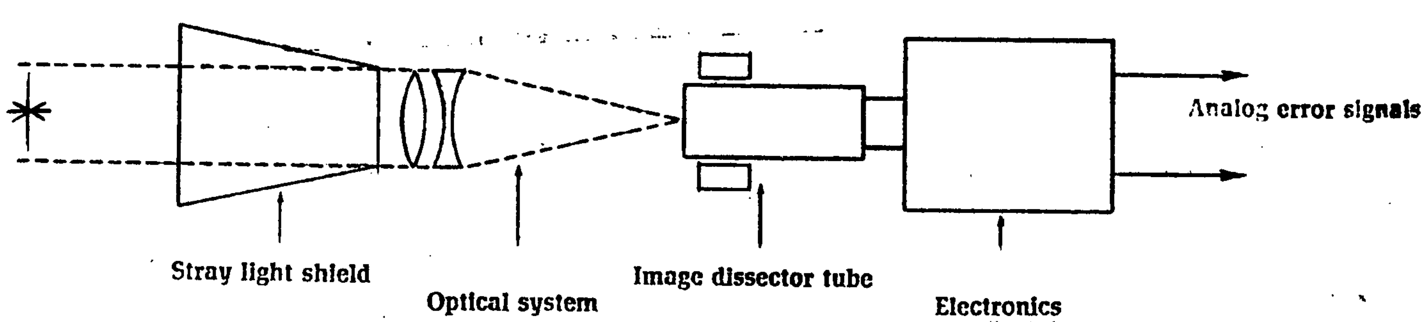


Fig. 8 Functional Parts of the Fixed Head Star Tracker

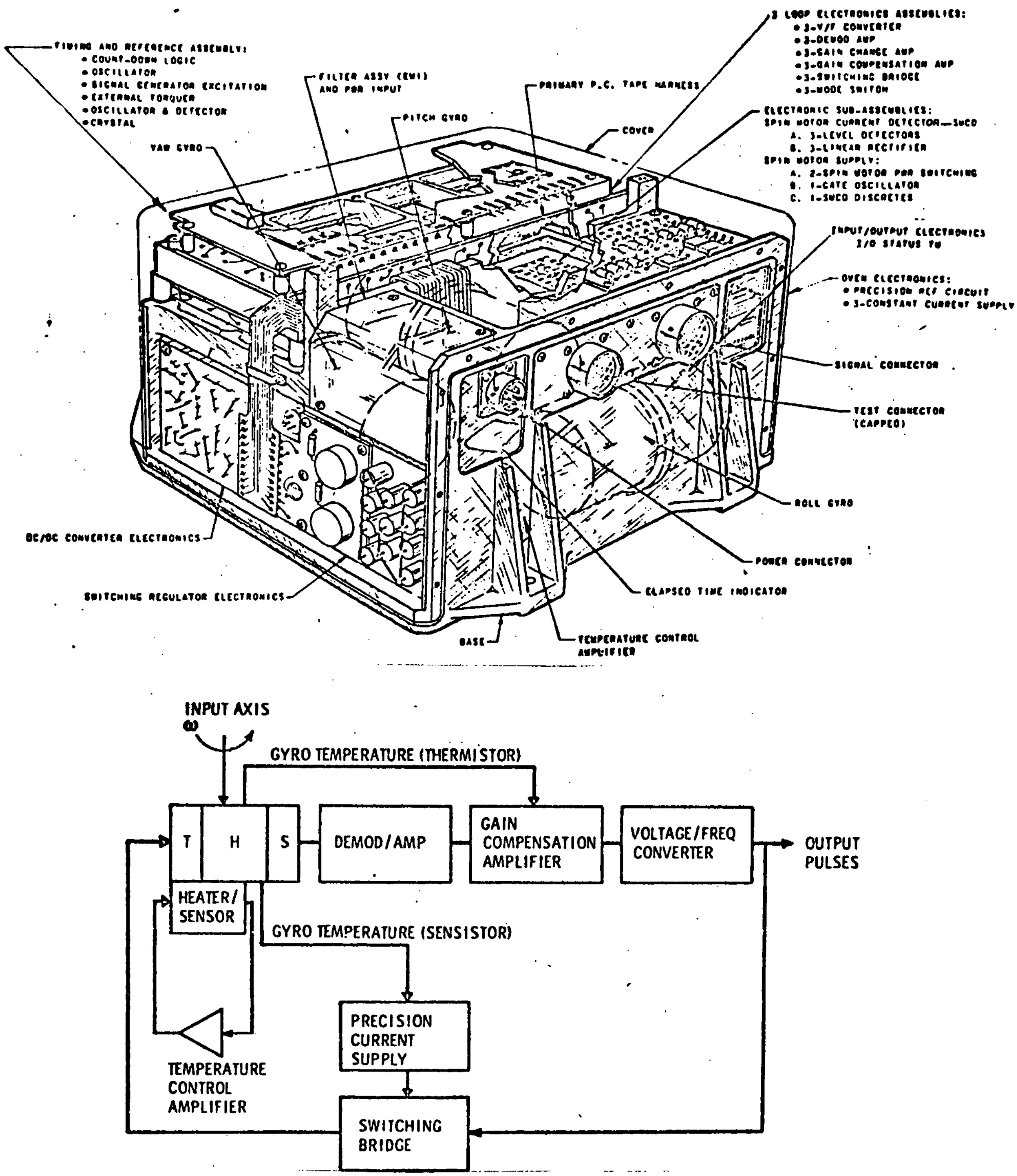
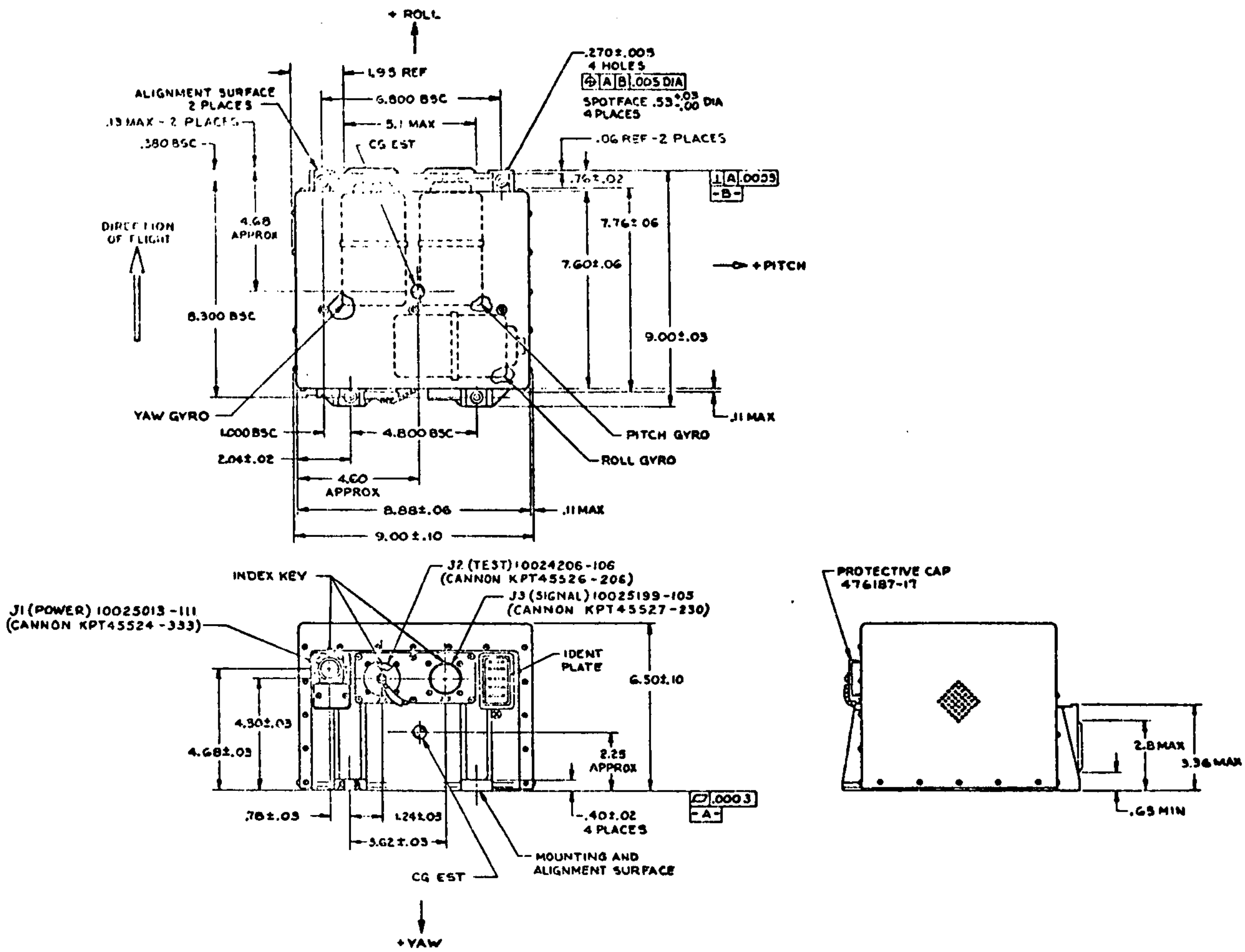


Fig. 6 Three-Axis Rate Sensor (1969)



4 - TORQUE RECOMMENDATION FOR MOUNTING BOLTS 12±1 LB/FT

3 - MOUNTING HARDWARE RECOMMENDATION: HIGH TENSILE STRENGTH BOLTS (130,000 PSI MIN) ST984508 OR EQUIVALENT WITH NAS620C416 WASHERS

2 - THE IRA GYRO INPUT AXIS ARE ORTHOGONAL WITHIN ±4 ARC MINUTES.

1 - THE IRA PITCH AND YAW GYRO INPUT AXES ARE ALIGNED TO WITHIN ±4 ARC MINUTES AND THE ROLL GYRO INPUT AXIS TO WITHIN ±2 ARC MINUTES OF THE IRA MOUNTING AND REFERENCE SURFACES (DATUMS A AND B)

Fig. 7 GG 2200 IRA (Honeywell)

Table 2

GG 2200 IRA SPECIFICATIONS

Performance

The performance with synchronized digital outputs and self-initiated mode changes is as follows:

Performance requirement

- 250 watts maximum starting (260 watts maximum during 30 second spinmotor runup)
- 50 watts maximum running

Warm-up time --- 30 minutes (maximum)

Internal timing reference

G-insensitive draft	1 deg/hr max.
Random drift	0.1 deg/hr max.
G-sensitive drift	1.0 deg/hr/g max.
Rate threshold	0.9 deg/hr max.
Pulse weight, low range	0.0656 sec
Pulse weight, high range	2.100 sec
Scale factor stability	±0.1%
Bandwidth	10.0 ± 2.0 Hz
Rate uncertainty	2.7 deg/hr max.
Maximum pulse rate	9600 pps
Gyro run-up time	30 sec max.
Telemetry	
• IRA internal temperature	(1)
• Gyro temperature	(3)
• Mode status	(3)

Control Inputs

Control inputs are:

Timing reference (1)	76,800 Hz
Rate synchronization (1)	9,600 Hz
High rate command (3)	+ 5.0 vdc
Torquing command (1)	+ 28 vdc

Control Outputs

Control outputs are:

Mode status (3)	+4.75 vdc
Rate signals (3)	+4.75 vdc
SMRD (1)	+4.75 vdc

Table 2 (Cont.)

Test Interface

(1) Test inputs, analog
Plus and minus gyro torquing

(2) Test outputs, analog
Rate outputs for above inputs

(3) Test monitor outputs
Internal timing reference
Reference voltage
Spinmotor excitation
Signal generator excitation
Oven temperature sensor

(4) Test control input

Spinmotor off	28 vdc

Design Features

Design features include the following:

Weight	17.75 lbs max.
Volume	450 cubic inches (9 x 9 x 7)
Input axis alignment	
Pitch and yaw	4 min
Roll	2 min
Emissivity	0.8 minimum

star being tracked (Fig. 9). The two output signals are proportional to the angular displacement of the star from the boresight of the tracker. The light sensor used in the fixed head star tracker is an image dissector tube. The image dissector tube is usually chosen for star sensor applications because of its high sensitivity, low noise, linear response to input energy, and its relative mechanical ruggedness. In operation, the electrons leaving the photocathode of the tube pass through a limiting aperture, the size of which determines the instantaneous field of view. Magnetic deflection permits scanning of the entire photocathode surface. The result is high signal-to-noise ratio achieved by the elimination of most of the background light.

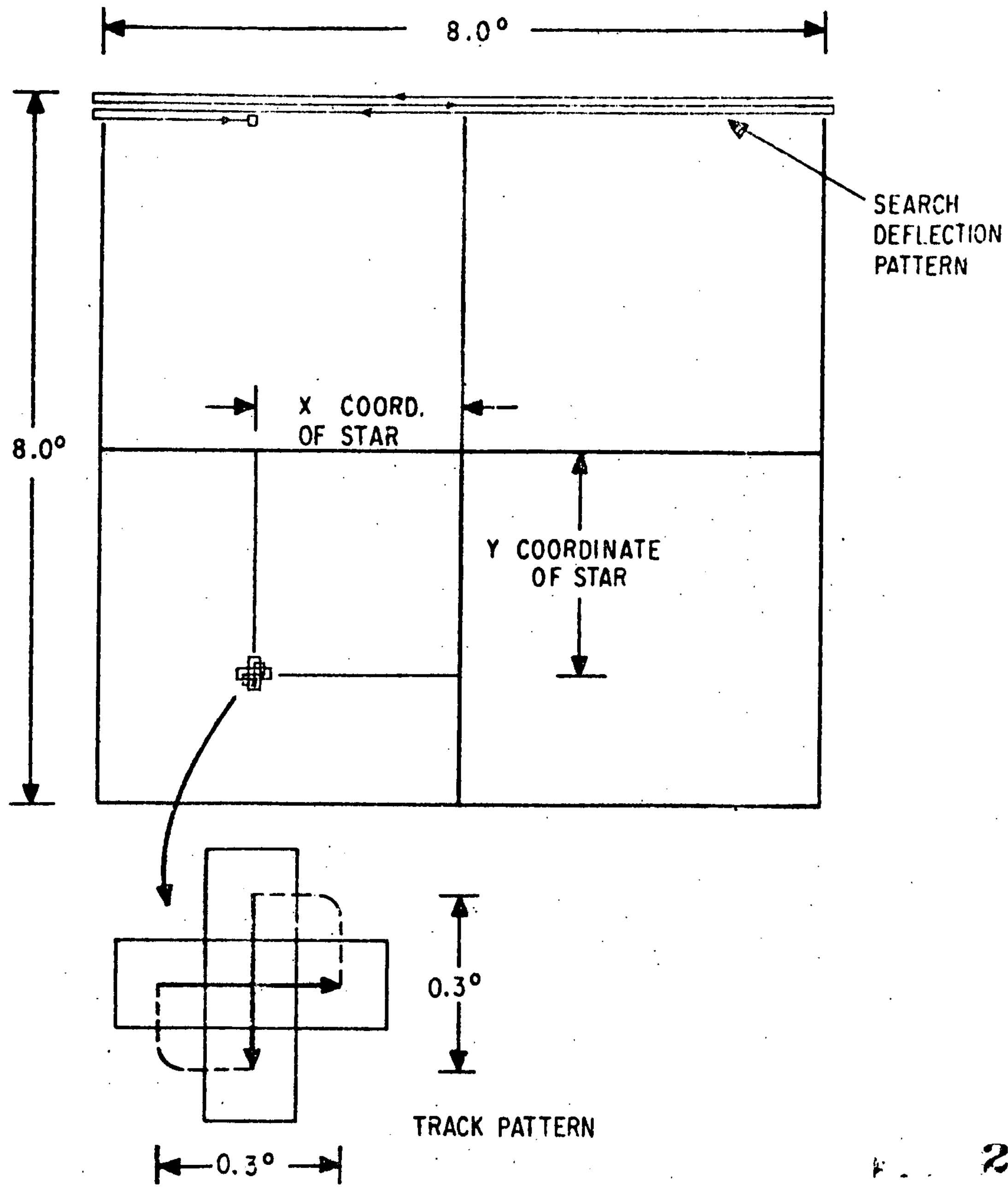


Fig. 9 Scan Geometry

The Fixed Head Tracker is packaged in a single, self-contained assembly, requiring only +28 vdc and appropriate control signals for operation.

Specifications are given in Table 3 for the newest Ball Bros. sensor, shown in Fig. 10, derived from their flight-proven "STRAP" Unit. Functionally-similar units have been manufactured by Ball Bros., Bendix, Hycon, and ITT Aerospace.

Table 4 lists some of the considerations which led to the decision to use a minimum of two and a maximum of four star tracker heads.

To meet the EOS design goal of attitude determination after ground data processing within 22 $\widehat{sec}$ (3σ), a number of changes from the tracker specifications shown in Table 3 are suggested by Ball Brothers. These are:

Parameter	Value
Search Scan (raster pattern)	$4^{\circ} \times 4^{\circ}$
Search Time (full frame)	5.5 sec
Track Pattern (cross scan)	$0.2^{\circ} \times 0.2^{\circ}$
Star Brightness (minimum for GO V Star)	+4th magnitude
Track Mode Noise Equivalent Angle	< 1.0 $\widehat{sec}$
Output Time Constant	0.5 sec
Estimated Total Error with Calibration & Compensation (over full 4° diameter field of view)	6 $\widehat{sec}$ (1σ)
Weight of Single Tracker (including electronics)	10 lbs
Power Consumption	< 8 watts
Volume - Sensor Head	4 in. dia. x 14 in. long
Electronics	2 x 4 x 6

In addition to these changes, electrostatic, rather than magnetic, focusing is recommended and instrument calibration should be repeated whenever the magnetic or temperature environment undergoes a change.

The FHT employs a sensor to preclude permanent damage from direct sunshine or moonshine. The sensor has a field of view of $\pm 3^{\circ}$, that activates a shutter internal to the FHT optical system, when the sun or moon enters its view.

Table 3

TYPICAL FIXED HEAD STAR TRACKER PARAMETER SUMMARY
(Ball Bros.)

Photo Sensor	
Manufacturer and type	ITT F4012RP Image Dissector
Focusing method	Magnetic
Deflecting method	Magnetic
Cathode Spectral type	S-20
Aperture dimensions	0.008 inch square
Lens	
Type	Super-Farron
Mfg.	Farrand Optical
Focal length	76 mm
T-stop	1.0
Resolution (diameter of 90% energy point)	0.004 inch
Field of View	$8^{\circ} \times 8^{\circ}$
Instantaneous field of view	0.153°
Search Mode	
Scan type	Raster
Search scan time	50 ms
Number of search scans	80
Total frame time	4.0 seconds
Star signal dynamic range visual magnitude (GO V)	+6.0 to +3.0
Probability of successful acquisition of 6.0 magnitude star ... greater than	0.95
Probability of noise occurrence during a 4-second search frame ... less than	0.1

Table 3 (Cont.)

Track Mode	
Scan type	Cross
Scan time each direction	50 ms
Scan period	100 ms
Output bandwidth	0.016 Hz
RMS output noise (NEA) less than	3.0 sec
Calibrated accuracy	10 sec
General	
Tracker power consumption -- less than	5 watts
Tracker weight	10.6 lbs

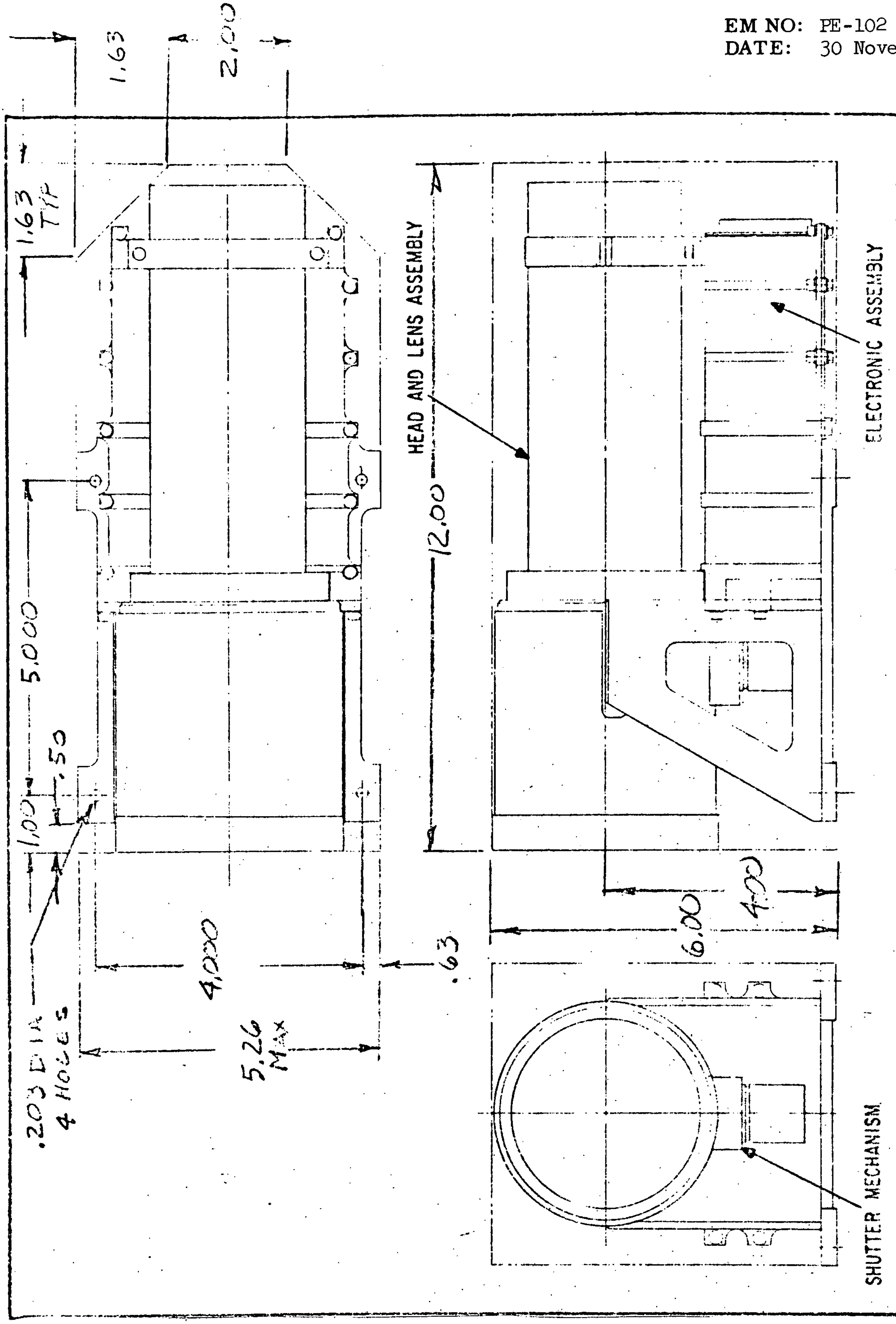


Fig. 10 Three-View Layout of Typical Star Tracker (Ball Bros.)

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Table 4

CONSIDERATIONS ON THE REQUIRED NUMBER OF FIXED HEAD STAR TRACKERS

Physical (Avg. time between fixes, max. time between fixes)

Field-of-view
Sensitivity (Star magnitude → No. of stars available)
Orbit attitude (earth occultation)
Inertial reference sensor errors (error build-up)
Sun/Moon interference schedule

Operational (Error build-up allowed)

Need for three-axis axes
Need for continuous fixes

Functional

Single type of module vs dissimilar modules
Redundancy requirements

TARS
FHT

Conclusions

1-Year: One FHT - Adequate for accuracy but inadequate reliability
1-Year: Two FHT - More than adequate accuracy and adequate reliability
2-Years: Three FHT - Adequate reliability but dissimilar modules
2-Years: Four FHT - Best for overall program cost considering two identical modules and standard interfaces

The shutter is spring-loaded in the open position and a signal serves to drive it shut. An override command is provided in the event the sensing device should fail in the energized position.

The FHT is further protected from oblique light impingement by a tubular light baffle which is centered about the optical axis.

Fixed Head Tracker Electronics

The output of the star tracker consists of a rectangular waveform which is asymmetric, except when the star is centered (Fig. 11). Additionally, left/right and up/down gating signals are provided in synchronism with the scan drive signals. A typical gating signal, is shown at (a) below. The photomultiplier tube output signal during the "up" scan only for a centered star is shown in (b) and for a vertically displaced star, (c) and (d). (Sensor outputs during "left", "down", and "right" scans have been neglected in the figure for clarity.)

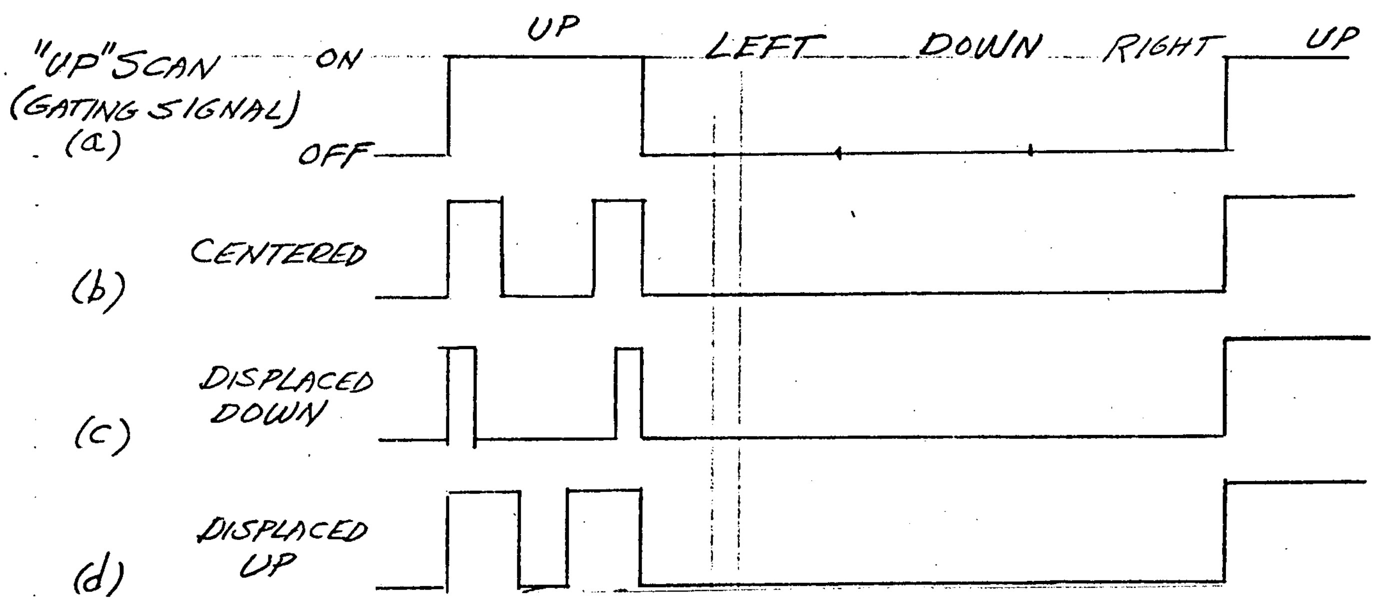

Fig. 11 FHT Output Signals

During the period of the sensor output, and while the scan gate is on, a clock pulse is counted by a gated counter. This count is transferred to the computer as a measure of off-axis orientation of the star being scanned. To reduce the susceptibility of the system to scan timing changes, a second counter is provided for the entire gating time. A servo signal can be derived from this count and used to control the oscillator.

Timing of the electronics is such that at least 1000 counts of the oscillator will occur during a single FHT scan. A period significantly shorter than one oscillator cycle will be used to set data in the output register and an "initiate transfer" signal will be presented to interface unit. The output register must be read and cleared during the following scan so that it will again be ready to receive data.

Additional functions of the FHT electronics include transmittal of the computer scan offset commands to the FHT, power control, and transfer of telemetry signals to the IU. The scan offset generator must include a holding register to retain the most recent command and a pair of analog outputs to be applied to the vertical and horizontal deflection amplifiers in the FHT (Fig. 12).

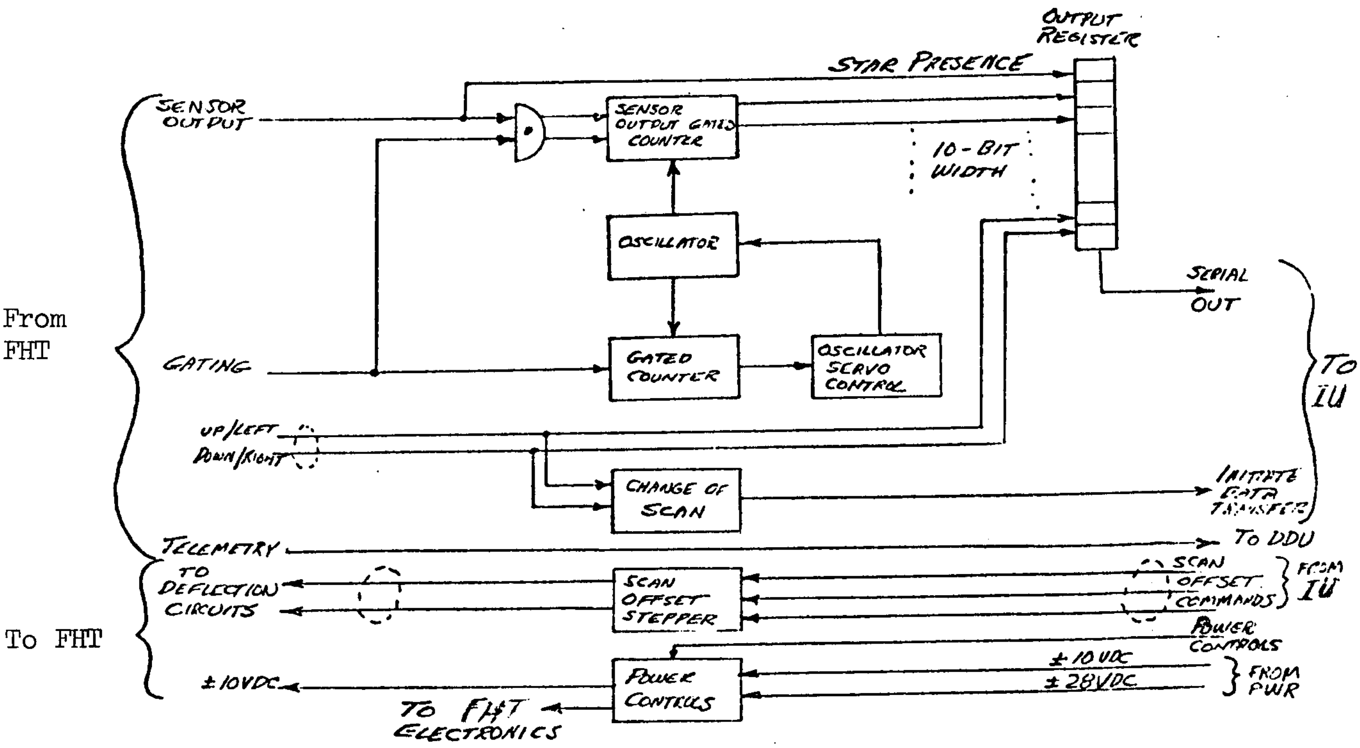

Fig. 12 FHT Electronics

(3) Reaction Wheels

Three single-axis, orthogonally-mounted reaction wheels will be provided in a single module to supply primary control torques. Reaction wheels do not eliminate the need for attitude control propulsion for momentum dumping, coarse acquisition, and backup control modes but do greatly reduce thruster duty cycles and total impulse required, as well as provide important benefits to the system (Appendix A).

The principal design characteristics of the Bendix OGO yaw reaction wheel are shown in Table 5 and Figs. 13 and 14.

Reaction wheel design has not changed much since the mid-1960's even though momentum wheel technology has advanced considerably.

Table 5

OGO YAW AXIS REACTION WHEEL

DESIGN DATA

General:

Momentum at 1250 rpm	8.84 ft-lb-sec.
Inertia of rotating parts	0.059 slug ft^2
Total unit weight	17 lb
Overall dimensions (inches)	12 O. D. x 4.75 high
Mounting	4-lug, with flat side heat sink
Life requirement	1 year
Reliability requirement	0.999 for 10,000 hours
Ambient temperature specified	22°F to 113°F

Motor Data (inside out):

Stall torque	10 oz-in. minimum
Stall power (total both phases)	43 watts maximum
Power requirement	115 volts, 400 cps, 2-phase square wave, with 90-degree phase relationship
Linearity of speed torque curve	Peak torque at a slip of 0.5
Starting voltage with 115 volts on one phase	3 volts maximum
Synchronous speed	1500 rpm

Tachometer Data:

Permanent magnet pulse generator pulses per revolution	8
Sensor output	Speed and direction (saw tooth cam)

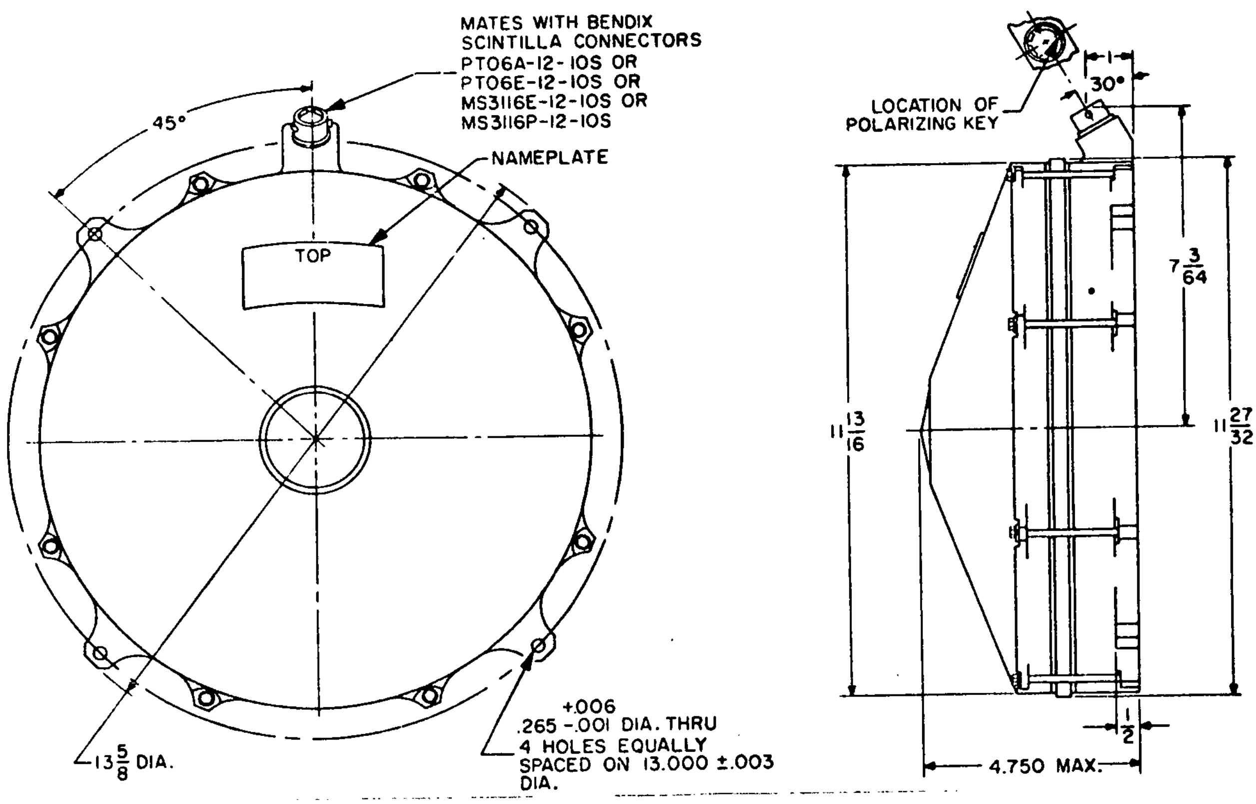

Fig. 13 OGO Yaw Axis Reaction Wheel - Outline

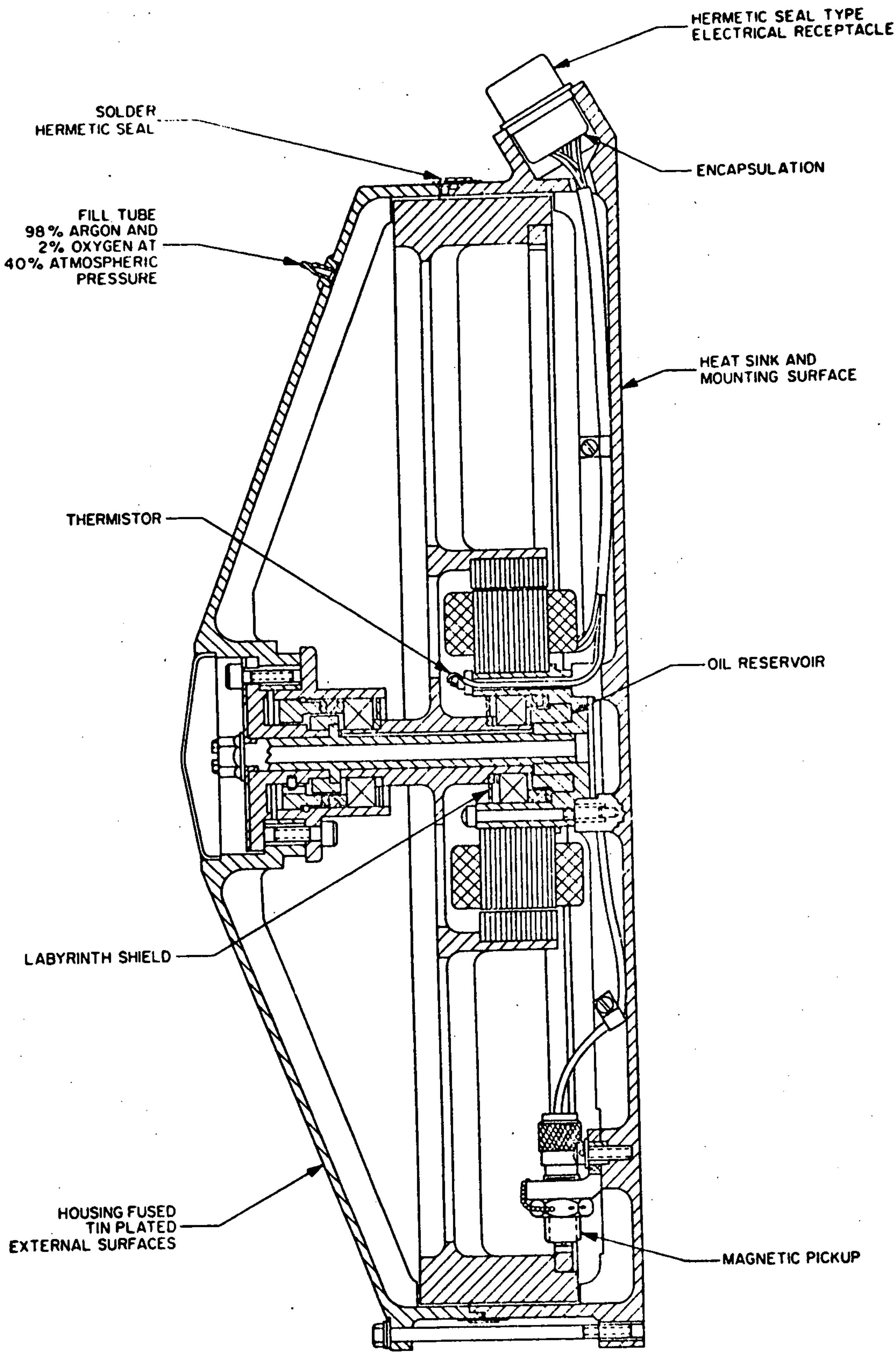

Fig. 14 OGO Yaw Axis Reaction Wheel -- Cross-Section

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Wheel Drive Electronics

The Wheel Drive Electronics package contains the wheel's power supply, pulse modulators (Fig. 15), and motor drivers (Fig. 16).

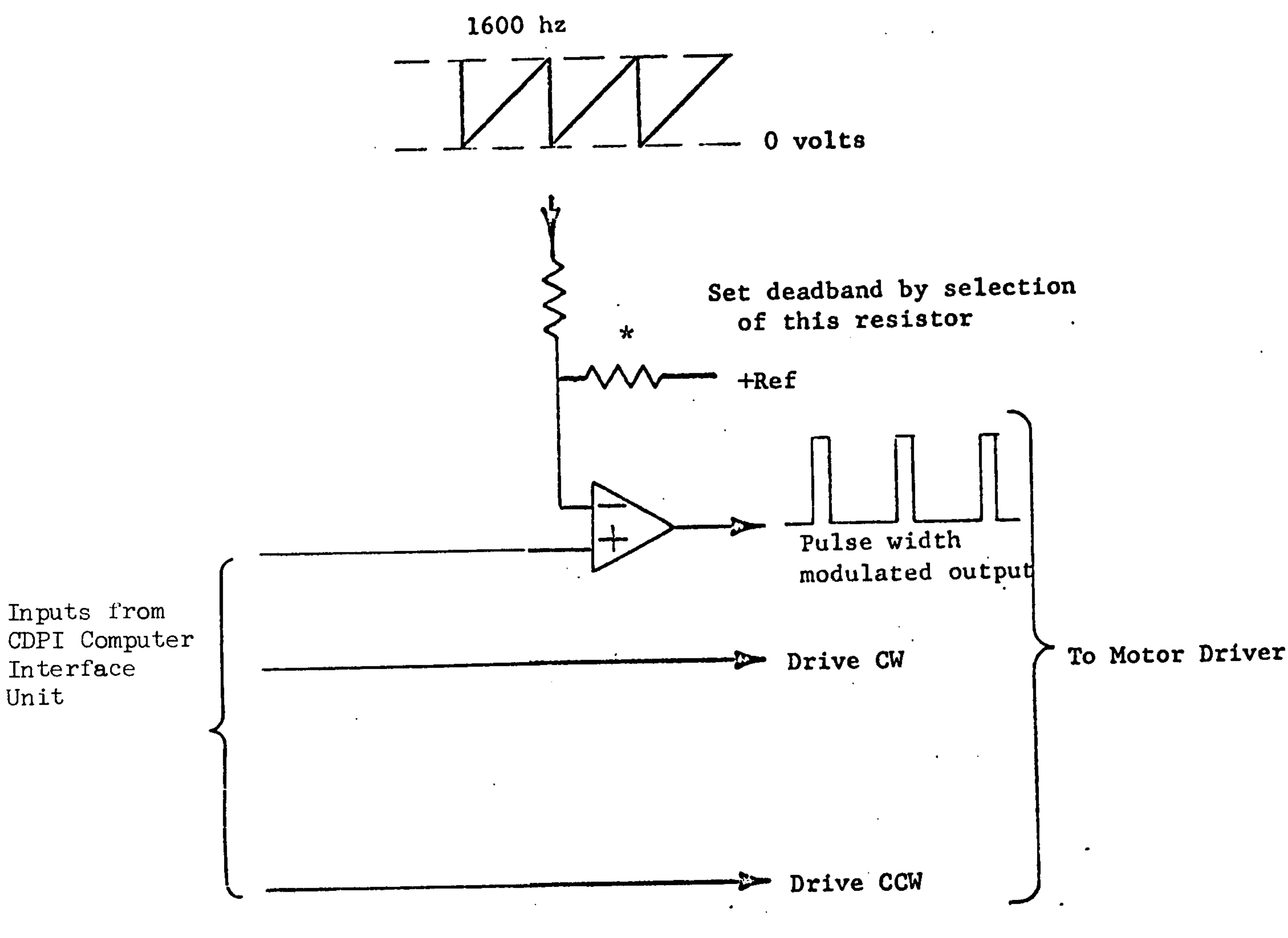


Fig. 15 Pulse Modulator

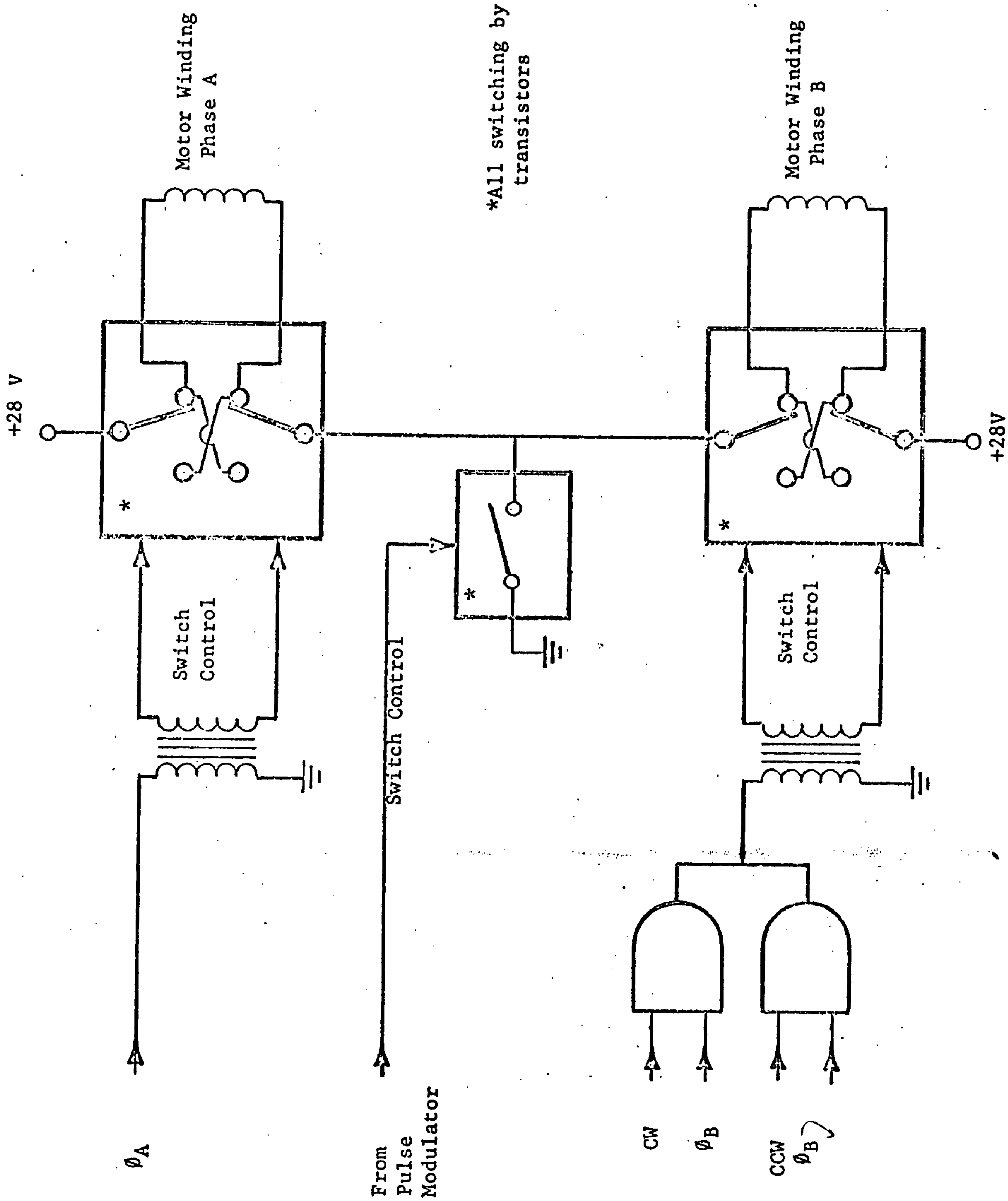


Fig. 16 Motor Driver

The pulse modulator converts the input signal, a train of width-modulated dc pulses, to a train of width-modulated 1600 hz pulses. The width is directly proportional to the control signal.

The modulator circuit also provides polarity information on a separate pair of lines, marked cw and ccw. These three signals control the motor driver.

The motor driver produces a motor torque that is nearly directly proportional to the duty cycle of the 1600 hz pulse train. The torque is, in fact, equal to the duty cycle times the available torque.

The motor driver consists of a pair of transistorized reversing switches, one associated with each motor winding. These switches are driven by 400 hz square waves that are transformer-coupled into the driver control circuits. The direction of the resulting torque, whether clockwise (cw) or counterclockwise (ccw) is determined by the cw and ccw inputs from the pulse modulator. These signals control the polarity of square wave applied to phase B. A third switch, common to both motor windings, is controlled by the pulse width modulated signal from the pulse modulator. The result can be looked at as a linear modulation of the current in the motor windings, thus producing torque proportional to duty cycle.

The power supply converts the +28 volt primary power to regulated voltages required by the circuits in the system. The power supply consists of a square loop transformer dc to ac converter, operating at 1600 hz. This is followed by rectifiers off appropriate secondary windings to produce the dc voltages. The 1600 hz square wave from the transformer is counted down as shown to produce the 400 hz two-phase reference signals for the motor driver. A sawtooth generator converts the 1600 hz square wave to a sawtooth signal for use by the pulse modulator.

Digital Solar Aspect Sensor

The Solar Aspect Sensor (SAS) is used to determine the two angles which describe the position of the sun vector with respect to the spacecraft. Information that describes the sun vector, plus a binary code to identify the sensor being read, is presented by the electronics unit to the vehicle telemetry in digital form. Figure 17 shows the basic principle. Light passing through a slit on the top of a quartz block is screened by a Gray-coded pattern on the bottom of the block to either illuminate or not illuminate each photocell. The angle of incidence determines which photocells are illuminated. The outputs from each cell are amplified, and the presence ("one") or absence ("zero") of a signal is stored and processed in the electronics to provide the desired output to telemetry.

Table 6 and Figs. 18 and 19 give specifications for the Adcole 15672 sensor and 15671 electronics.

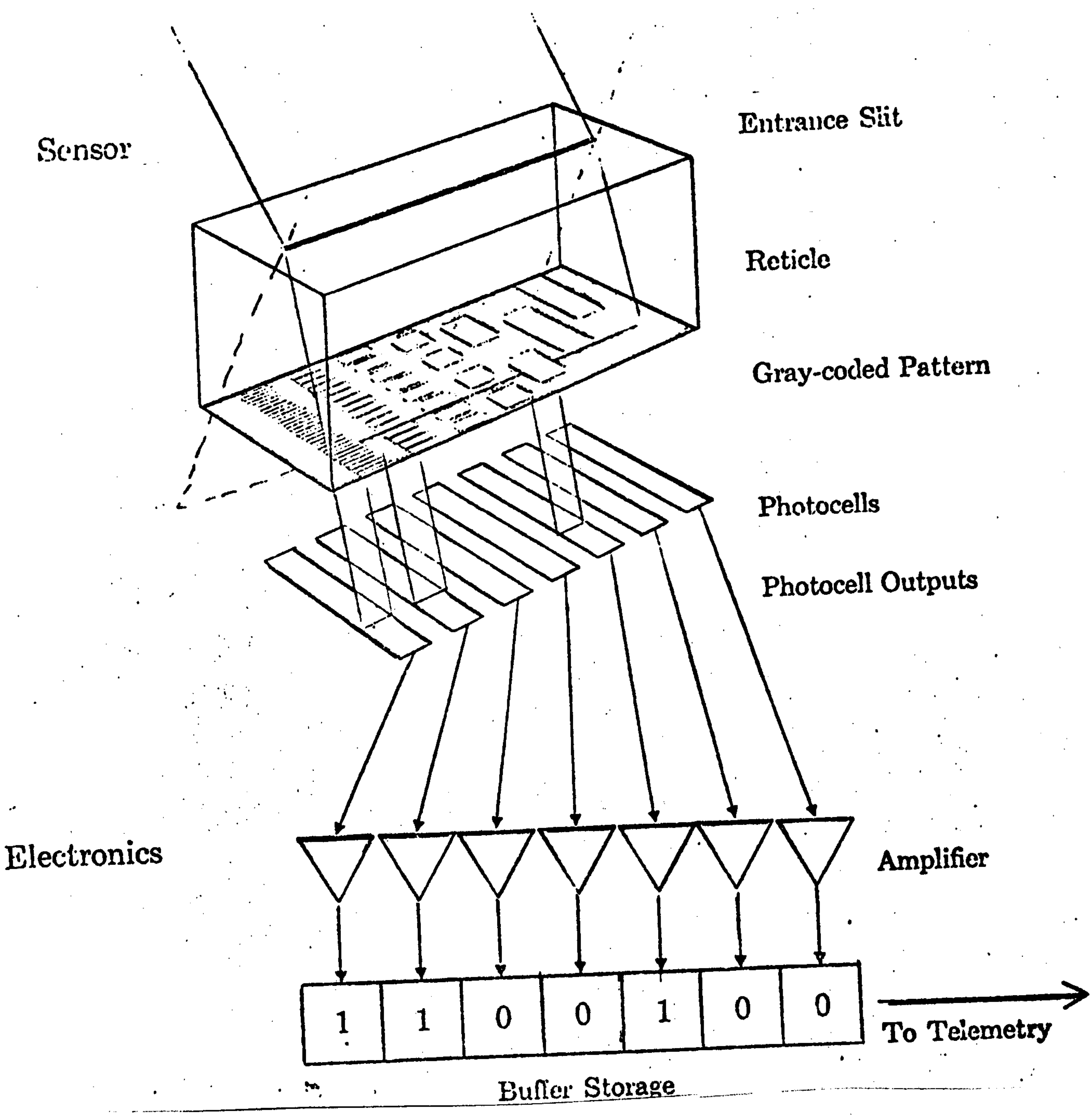


Fig. 17 - SAS Schematic

Table 6

SUMMARY SPECIFICATIONS

PERFORMANCE CHARACTERISTICS - SYSTEM

Sensor Field of View	$64^{\circ} \times 64^{\circ}$
Resolution	$1/8^{\circ}$
Transition Accuracy	Better than ± 6 arc min.
Output	Parallel Gray Code, 9 bit/axis
Output Signal Levels	Logic "O" 0.0 $\pm$0.5 VDC
	Logic "1" 3.0 VDC min.
Maximum Power Requirement	28 $\pm$4 VDC .35 watts max.

MECHANICAL CHARACTERISTICS - SENSOR MODEL 15672

Size:	Per drawing 15672
Weight:	0.7 lbs.
Cells:	N on P Silicon with redundant leads
Connector:	Bendix JT02A-14-37P (005)

MECHANICAL CHARACTERISTICS - ELECTRONICS MODEL 15671

Size:	Per Drawing 15671
Weight:	1.5 lbs.
Connectors:	Bendix 21-245-216-26P
	Bendix JT02A-14-37P (005)

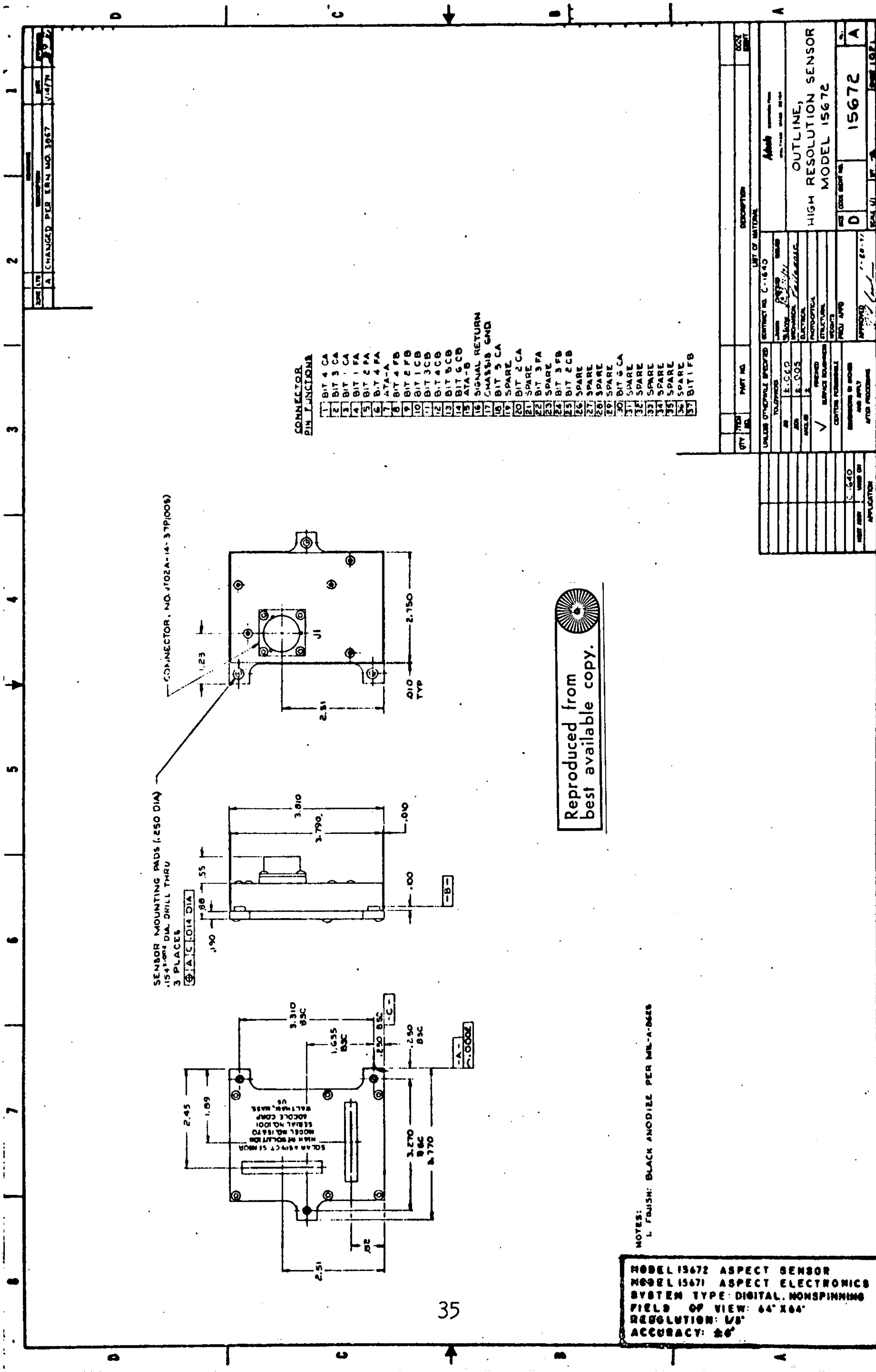

OUTLINE, HIGH RESOLUTION SENSOR MODEL 15672
15672
CONNECTOR PIN FUNCTIONS
SENSOR MOUNTING PADS (.250 DIA)
CONNECTOR, NO. 1102A-14-37P(005)
Reproduced from best available copy.
NOTES:
1. FINISH: BLACK ANODIZE PER MIL-A-8625
MODEL 15672 ASPECT SENSOR
MODEL 15671 ASPECT ELECTRONICS
SYSTEM TYPE: DIGITAL, NONSPINNING
FIELD OF VIEW: 64° X 64°
RESOLUTION: 1/8°
ACCURACY: ±6'

Fig. 18

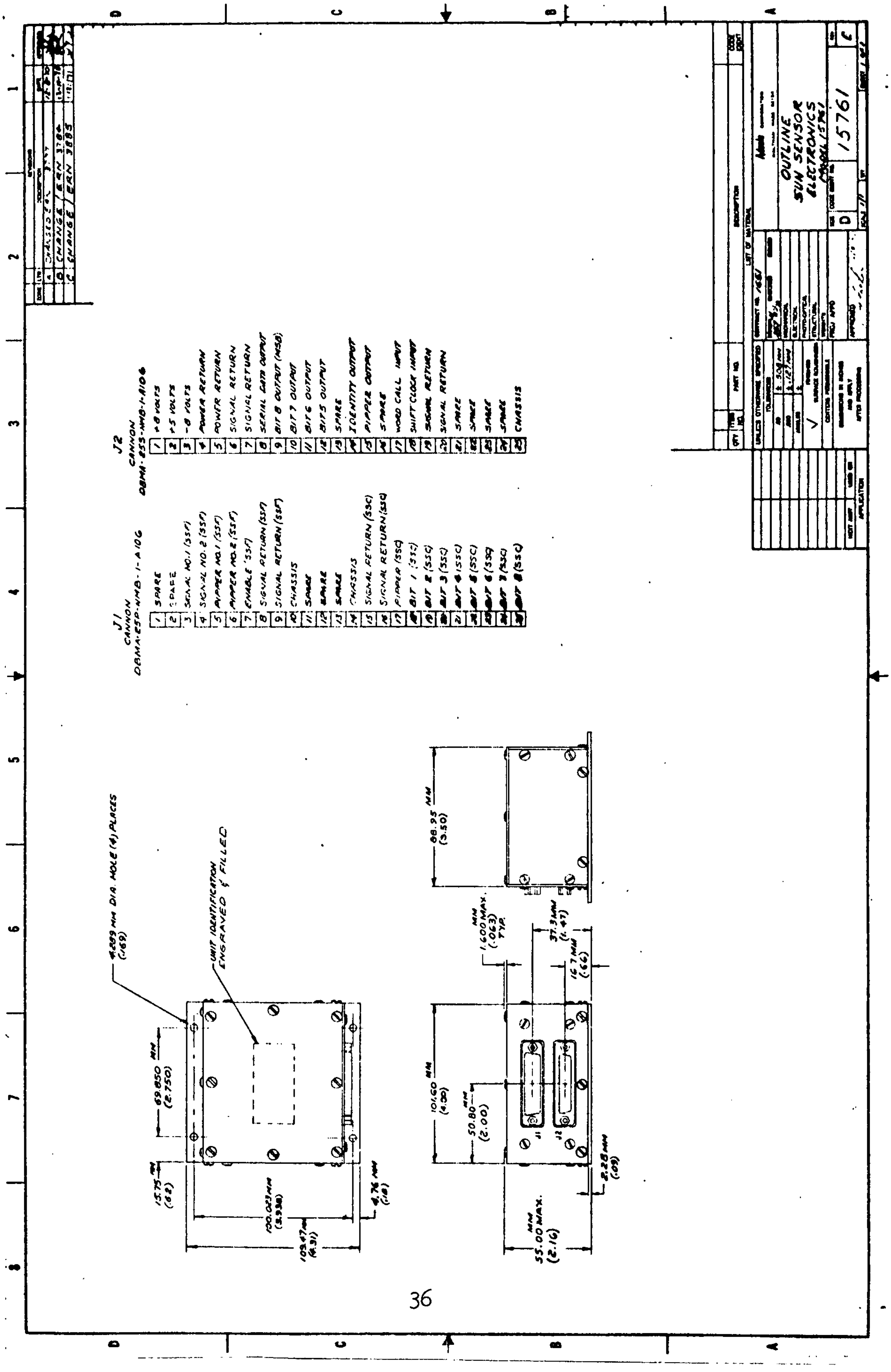


Fig. 19

(5) Rate Gyro Package

The rate gyro package measures angular rates about the roll, pitch, and yaw axis. The backup system requirements are lax enough to permit their fulfillment by a wide variety of gyros. A typical package uses three Nortronics GR-G5 subminiature fluid-filled rate gyros. Table 7 and Fig. 20 summarize the rate gyro package characteristics.

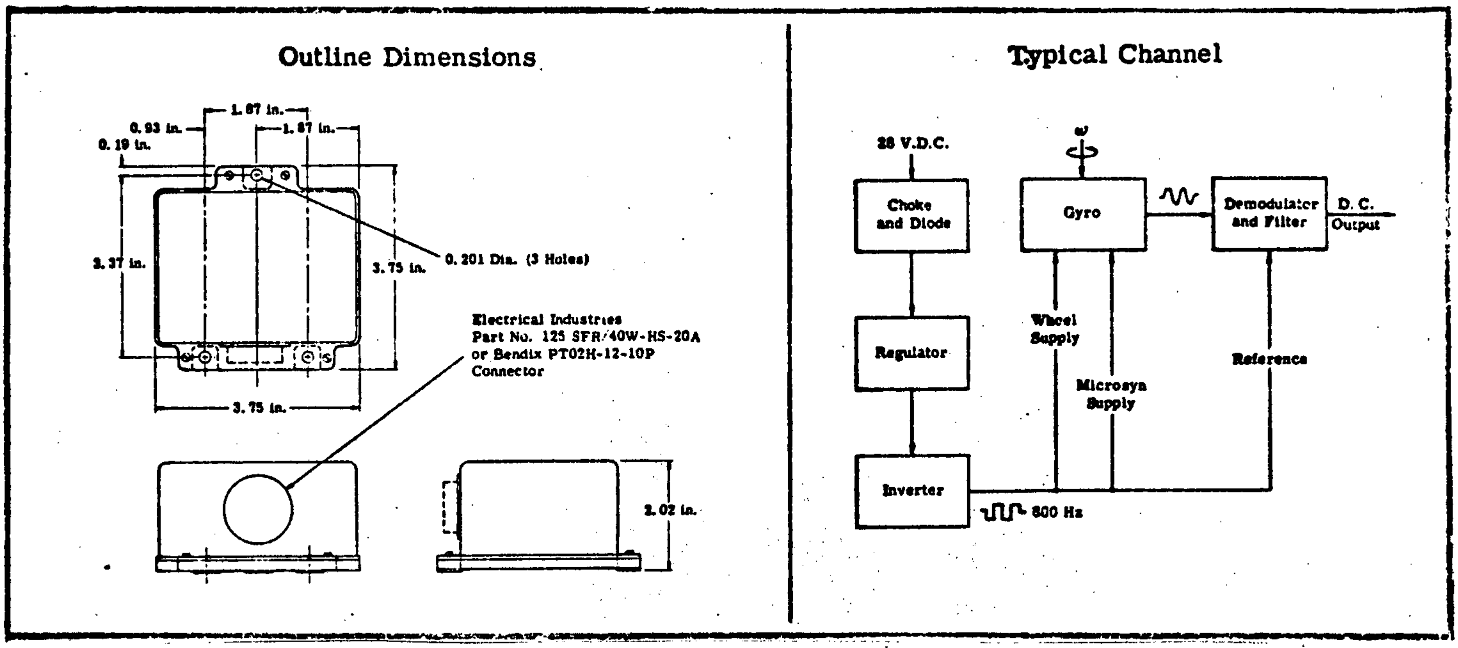


Fig. 20 Rate Gyro Package (Nortronics)

(6) Docking Reflectors

The Docking Reflectors are the target part of the ITT Scanning Laser Radar System developed for MSFC.

The target equipment consists of a quartet of 4" optical corner cube reflectors. The corner cube reflectors allow the Shuttle to acquire and track the EOS out to a range of 75 nm for rendezvous. The scanning laser radar equipment is Shuttle-borne; the function of the EOS portion is solely to reflect back the laser beam.

It is desirable for the EOS to correct its attitude with respect to the Shuttle docking LOS; four cubes provide sufficient information to the Tug to enable it to compute the EOS LOS angles and roll angle for RF transmission to EOS.

Table 7

RATE GYRO PACKAGE SPECIFICATIONS

MEASURED CHARACTERISTICS

Weight	2.0 lb. (max.)
Outline Dimensions	3.75 x 3.75 x 2.0 in. (max.)
Power Input	15 w. (max.) at 31 vdc
Input Voltage Limits	28 ± 3 vdc
• Full-Scale Output	2.5 ± 2.5 vdc
• Output Impedance	5000 ohms (max.)
Output Load Resistance	500K ohms (nominal)
• Ripple	25 mv. peak-to-peak (max.)
• Zero Rate Setting	±1/2 % FS
Input Range (Pitch, Roll, Yaw)*	100°/sec.
Maximum Input Rate	600°/sec.
Output Voltage Overrange Limits	-0.75 vdc, 7.0 vdc
• Output Stability, Input Voltage Variations	1/2% FS
Repeatability	1% FS
• Threshold*	0.01°/sec.
• Resolution*	0.01°/sec.
• Hysteresis*	0.1°/sec.
Operating/Storage Temperature*	-35°F to +160°F/-65°F to +200°F
Temperature Sensitivity	Zero Rate Output 1% FS/100°F (max.) Scale Factor 3%/100°F (max.)
Warm-up Time	10 minutes
• Gyro Spin Motor Acceleration Time	30 sec. (max.)
Gyro Gimbal Deflection Angle	±2° typical
Acceleration Sensitivity	
Linear*	0.05°/sec./g
Angular	0.08°/sec./rad./sec.^{2}
Linearity	1/2% FS, from 0 to half scale 2% FS, half scale to full scale
Service Life	1000 hr. (min.) or one year
• Insulation Resistance	10 megohms (min.) at 50 vdc
• Damping Ratio*	0.5 to 0.9
• Natural Frequency (min.)*	35
Environments	
Shock	250 g peak sawtooth, 5 msec.
Vibration	0.1 g^2/Hz, 20-2000 Hz
Storage Temperature	-65°F to +200°F
Radio Frequency Interference	MIL-I-8161D

*Parameters that are a function of the rate sensor used
• Acceptance tested parameters
NOTES:

1. The three output signals are isolated from input common and from each other.
2. The output signals are protected from any damage occurring as a result of inadvertent shorting.
3. The standard three-axis DC/DC configuration is also available with control output of ±5 vdc.
4. There are several models of the Standard GR-G5 Rate Gyro to choose from, accommodating full-scale rate inputs from 20 to 10,000°/sec. Variable limits for natural frequency, acceleration sensitivity, threshold and resolution as a function of input rate are shown in the G5 parameter table.

The magnetic torquer is a single-axis variable permanent magnet system for a) in-orbit magnetic balancing of spacecraft, and/or b) wheel momentum control through interaction with the earth's magnetic field. It is a permanent magnet whose dipole moment can be controlled. Once the desired magnetic state is reached, power may be turned off and this state will be retained indefinitely until further changes are desired. There is usually a substantial power-saving advantage over current carrying coils or electromagnets, since power is required only when the dipole moment is changed. Other advantages include small size and weight. Table 8 and Fig. 21 give the specifications of a magnet produced by Ithaco, Inc.

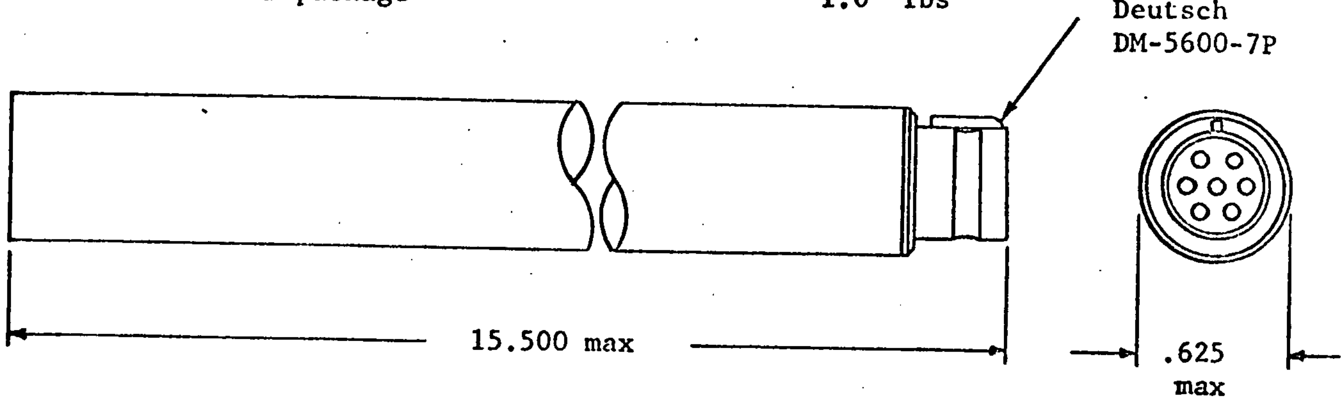
Table 8

MAGNETIC TORQUER

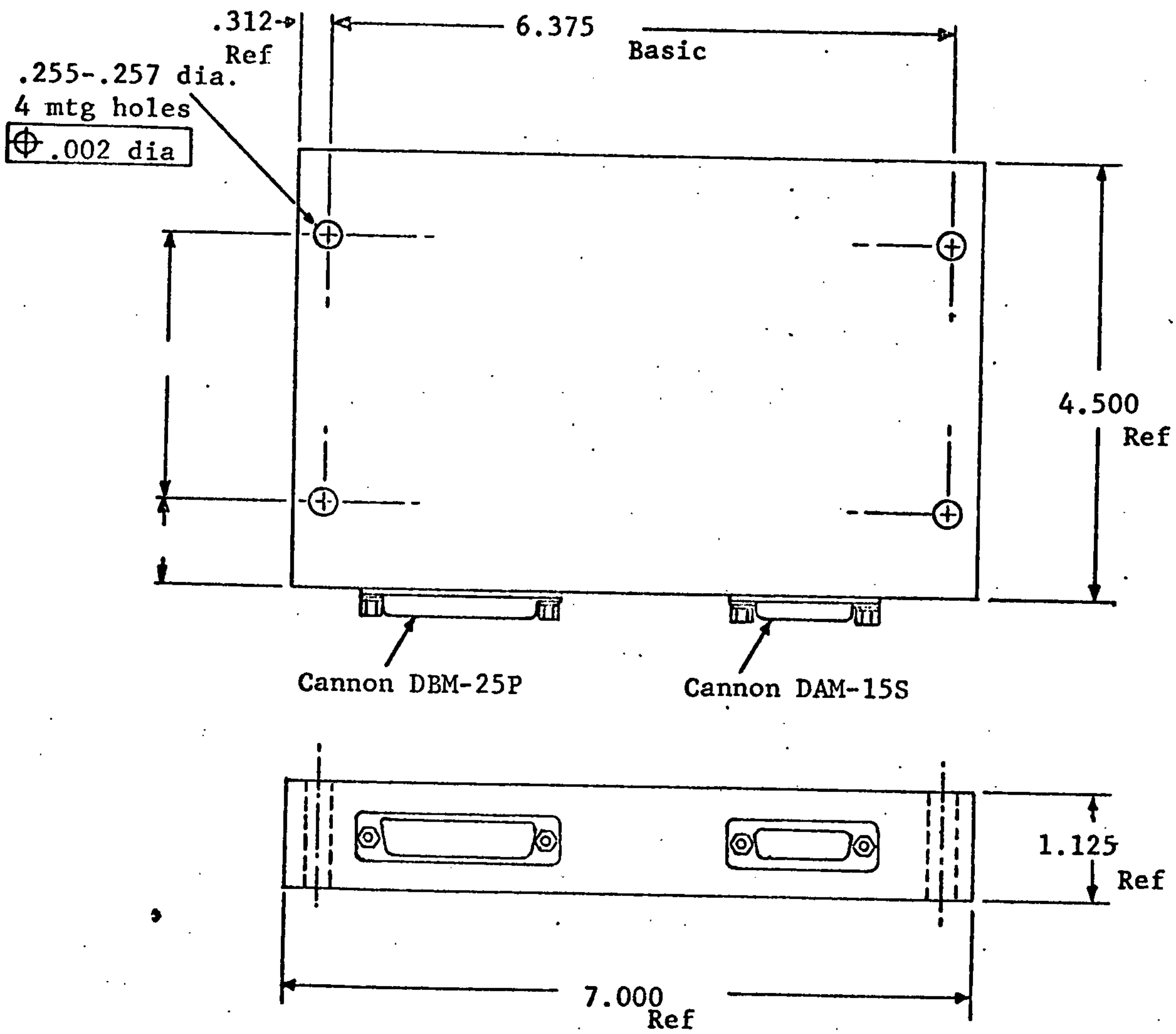
SPECIFICATIONS:	
Maximum dipole moment*	±10,000 pole-cm min
Accuracy	±3% of maximum moment, relative to the calibration curve provided
Repeatability	±.1% of maximum moment
Resolution	±.1% of maximum moment
Switching time	$3X10^{-6}$ sec/pole-cm
Switching energy	10^{-4} watt-sec/pole-cm
Gain (ANALOG MODE)	2 pole-cm/mv
Rate of moment change(INTEGRATE MODE)	1 pole-cm/sec/mv
Operating temperature	-10°C to +60°C
Storage temperature	-50° to +85°C
Power requirements	+28 volts ±2V @ 1 amp peak +10 volts ±1V @ 15 ma -10 volts ±1V @ 15 ma
COMMANDS:	
Interface	12 volt latching relay coils at 500 ohm. Teledyne 422 series
Functions	ENABLE DISABLE ANALOG MODE INTEGRATE MODE
TELEMETRY:	
Interface	Analog -5 to +5 volts @ 2 Kohm max Digital > 0V = "1" @ 2 Kohm max < 0V = "0" @ 2 Kohm max
Functions	COMMAND STATUS (2 digital) DIPOLE MOMENT (.4 mv/pole-cm) MAGNET TEMP (100 mv/°C)

MECHANICAL:

Weight - Magnet .75 lbs
Control package 1.0 lbs



MAGNET



CONTROL PACKAGE
Fig. 21

C. S&C Subsystem Modules/Interfaces

(1) Guidelines

An important factor in determining the cost of revisit/repair/refurbishment operations is the manner in which the S&C equipment is physically grouped. These groupings could range from a single part to the complete S&C subsystem; a group could even mix parts from two or more subsystems. The selected arrangement was arrived at by application of the modularity guidelines (Table 9).

(2) Module Description (Fig. 22)

The Primary Sensing module has provisions for mounting two star trackers in a skewed but mutually perpendicular orientation.

Figures 23 and 24 show schematically the Secondary Sensing and Reaction Wheel Modules, respectively.

(3) S&C Subsystem Interfaces

The module interconnections are shown in Fig. 25. The subsystem command and telemetry functions are listed in Tables 10 and 11, respectively.

The use of reaction wheels and magnetic torquers for wheel desaturation effectively decouples the functions of the S&C and ACP subsystems. (See Appendix A, Parts (3) and (4)). An electrical/mechanical interface exists in the ACP thruster drive electronics packages - one is located in each ACP module but is considered part of the S&C Subsystem for convenience.

Figure 26 shows the EOS equipment locations and Fig. 27 the recommended location and orientation of the S&C subsystem modules.

Table 9

MODULARITY GUIDELINES

(1) Minimize number of modules and types of modules.

(2) Pick well-defined electrical interfaces which are parameter tolerance-sensitive.

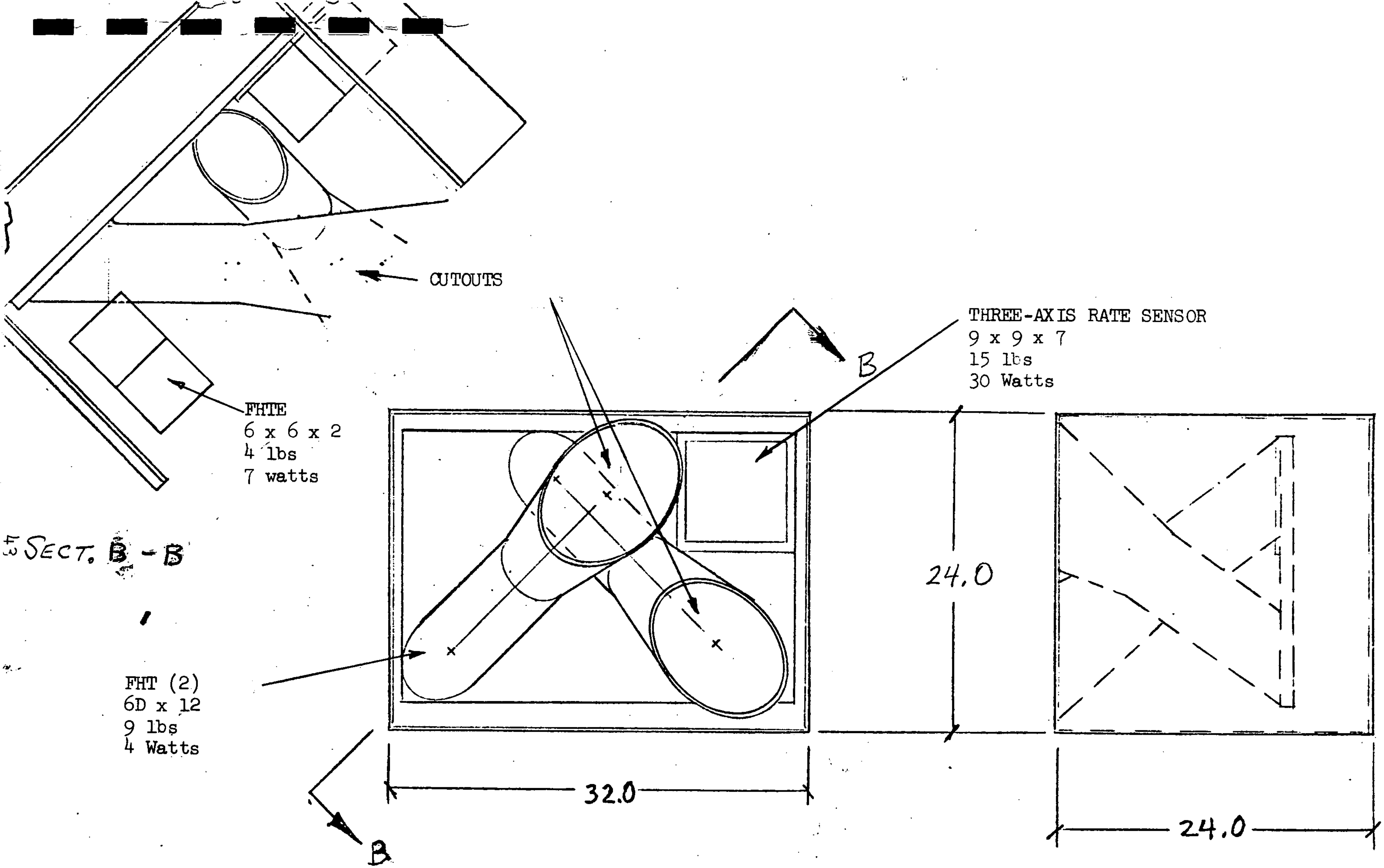
(3) Keep number of electrical connections at each module as small as possible.

(4) Keep module size down to approximately 18" x 24" x 30".

(5) Keep module cost to less than 5% of spacecraft cost.

(6) Place redundant assemblies into separate modules wherever a failure could cause a chain reaction with catastrophic consequences.

(7) Minimize EMI generation and susceptibility.



EQUIPMENT WEIGHT: 41 lbs
EQUIPMENT POWER: 52 Watts

Fig. 22 Primary Sensing Module

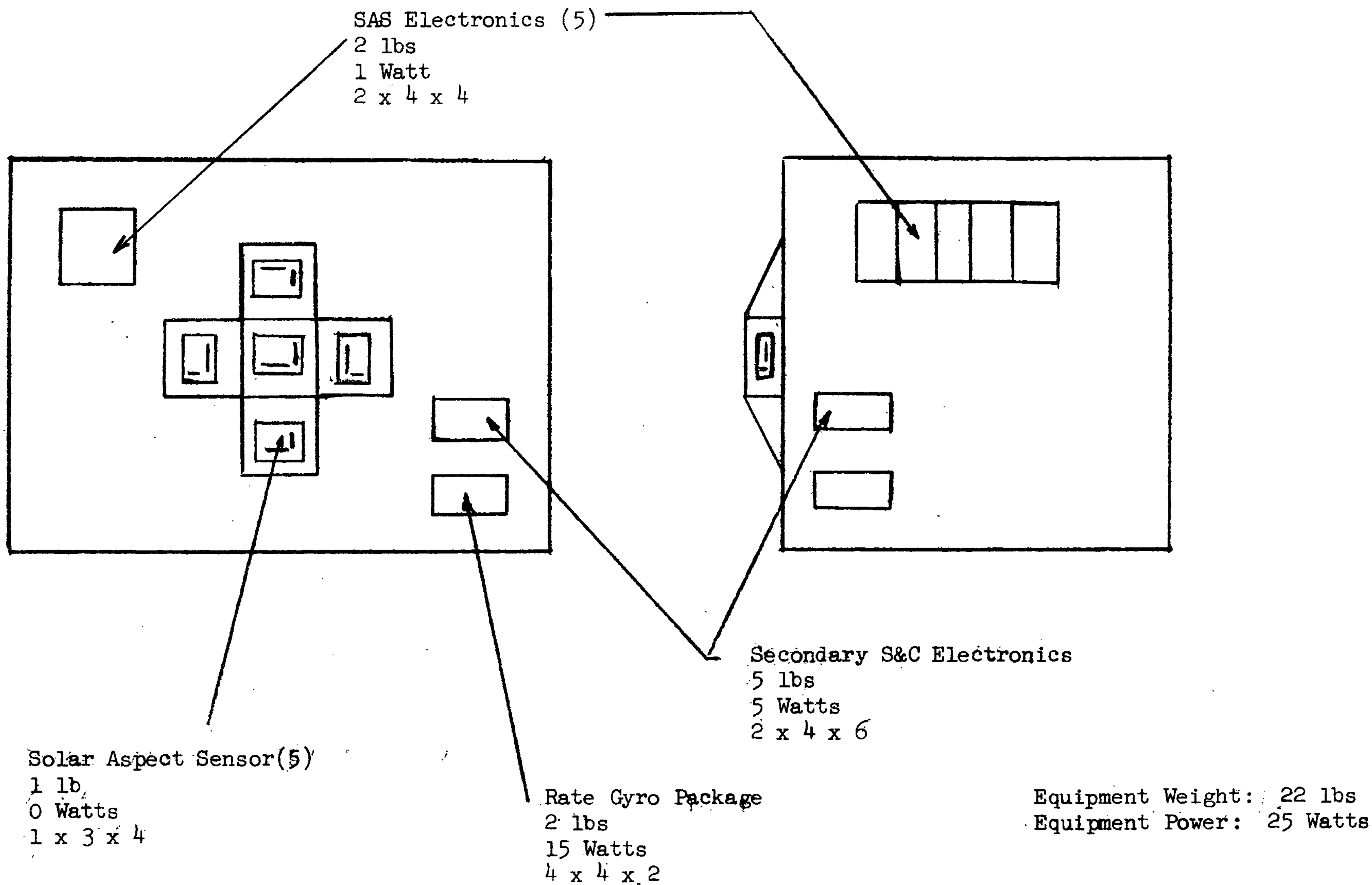


Fig. 23 Secondary Sensing Module

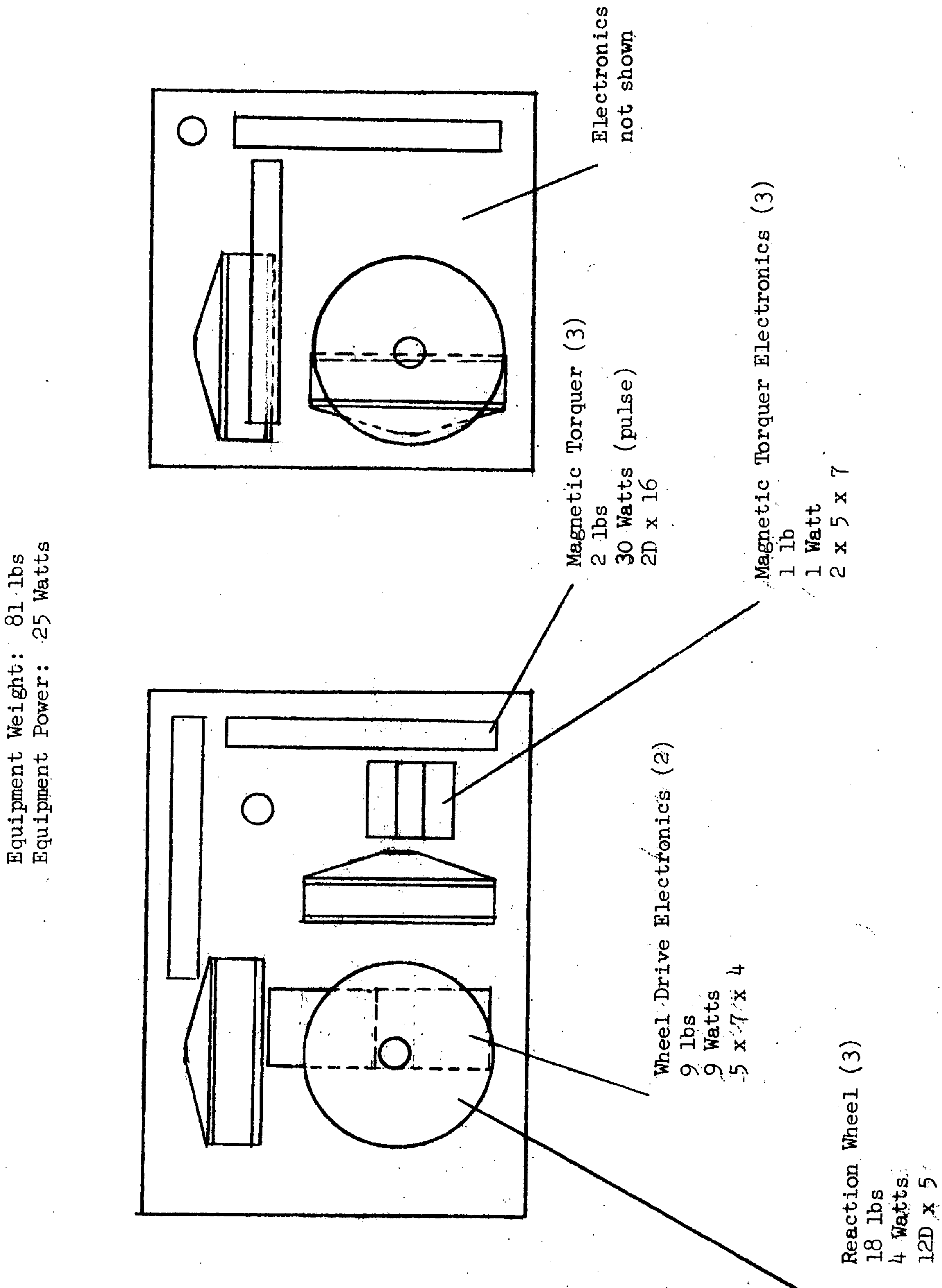


Fig. 24 Reaction Torque Module

* Note: Each ACP Module contains one thruster drive electronics package

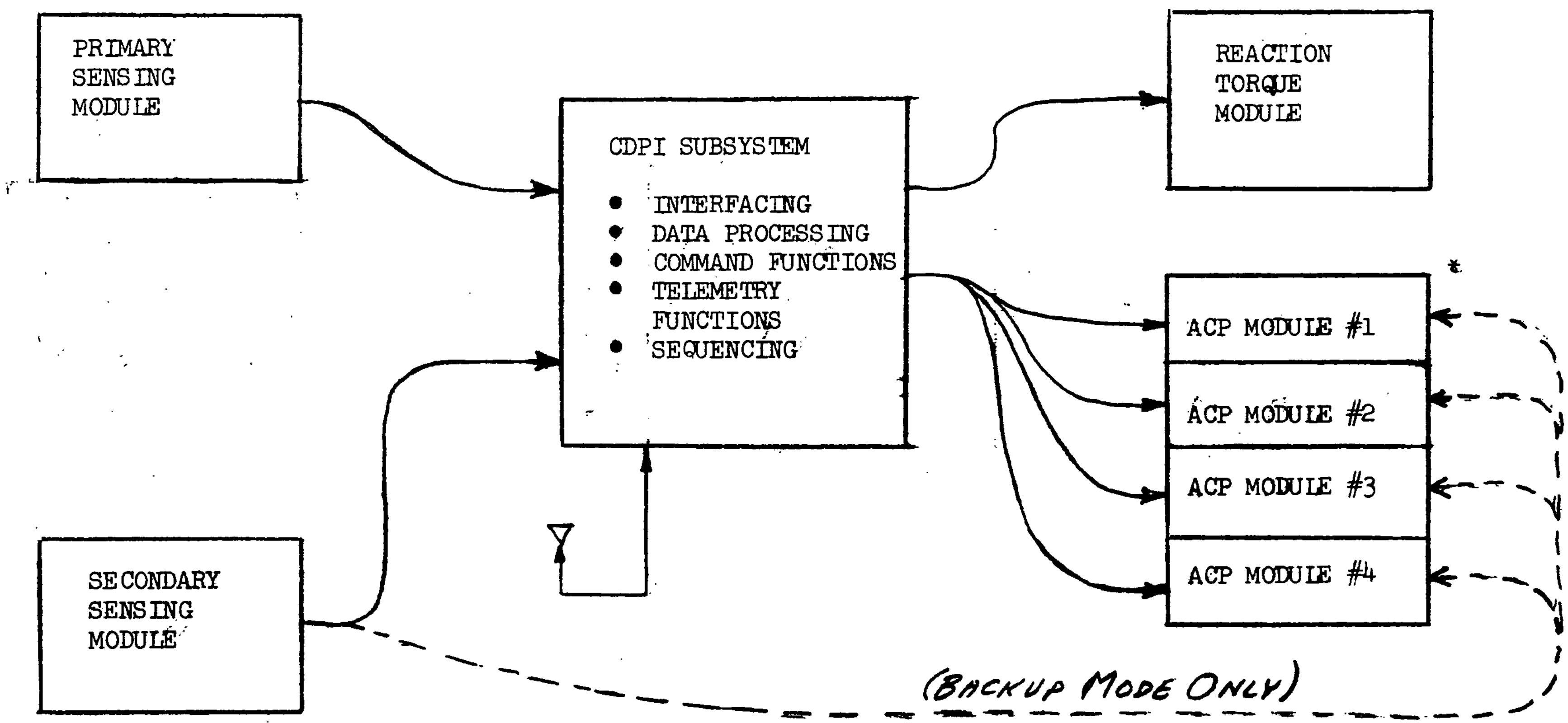
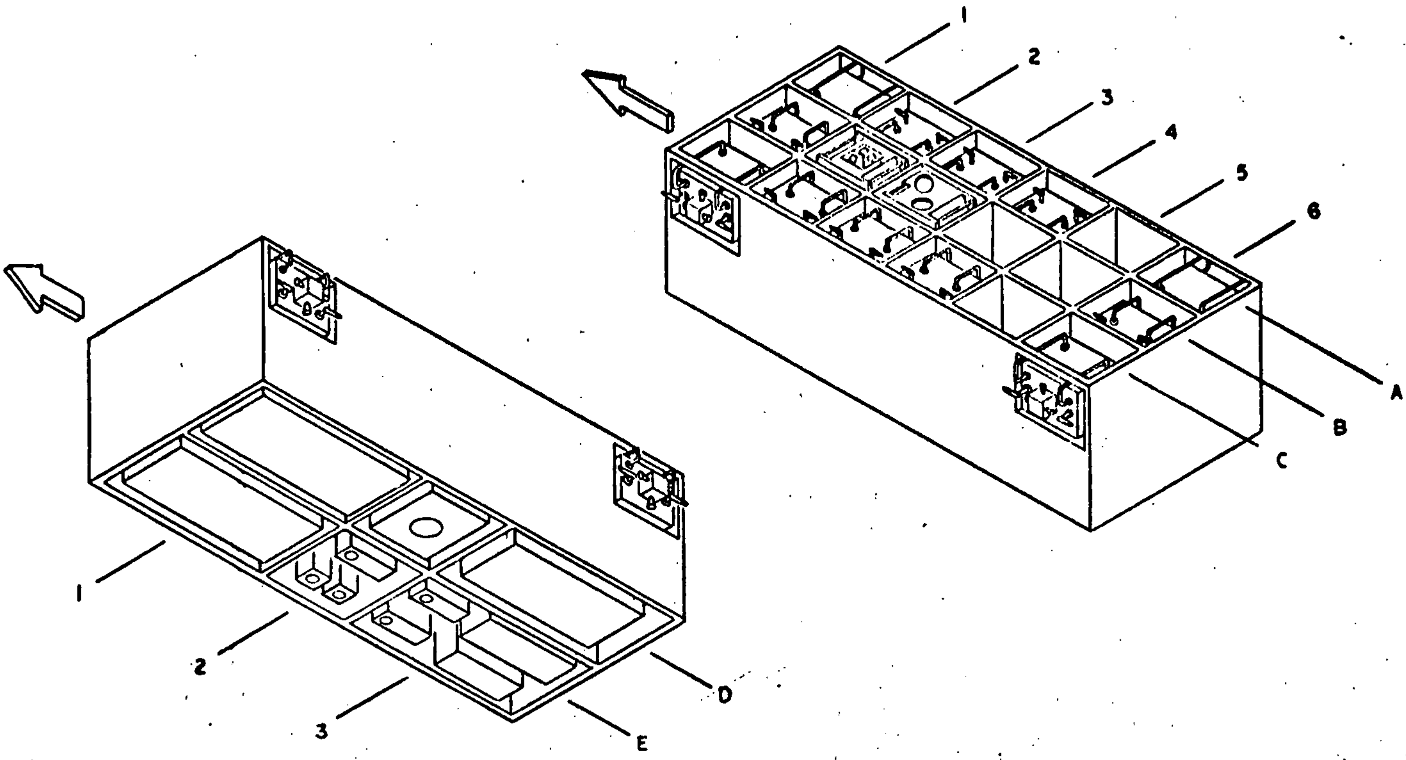


Fig. 25 Low-Cost EOS S&C Module Interconnections



Mission Equipment

D-1 Passive Microwave Radiometer
(λ = 0.81 cm)

D-2 Thematic Mapper

D-3 Passive Microwave Radiometer
(λ = 2.81 cm)

E-1 Passive Microwave Radiometer
(λ = 6.01 cm)

E-2 Ocean Scanning Spectrophotometer
Atmospheric Pollution Sensor
Upper Atmosphere Sounder

E-3 Cloud Physics Radiometer Sea Surface Temp. Radiometer
Passive MW Radiometer (λ = 1.67 cm)
Passive MW Radiometer (λ = 1.40 cm)

Spacecraft Subsystem Modules

A-1 Attitude Control Module No. 1
A-2 S & VHF Band Communication Module
A-3 Battery Module No. 1
A-4 Power Control Module
A-5 Empty
A-6 Attitude Control Module No. 2
B-1 K-Band Communication Module
B-2 S&C Secondary Reference Module
B-3 S&C Primary Reference Module
B-4 Empty
B-5 Empty
B-6 Reaction Torque Module
C-1 Attitude Control Module No. 3
C-2 Data Processing Module
C-3 Battery Module No. 2
C-4 Battery Module No. 3
C-5 Empty
C-6 Attitude Control Module No. 4

Fig. 26 Standard EOS Equipment Locations

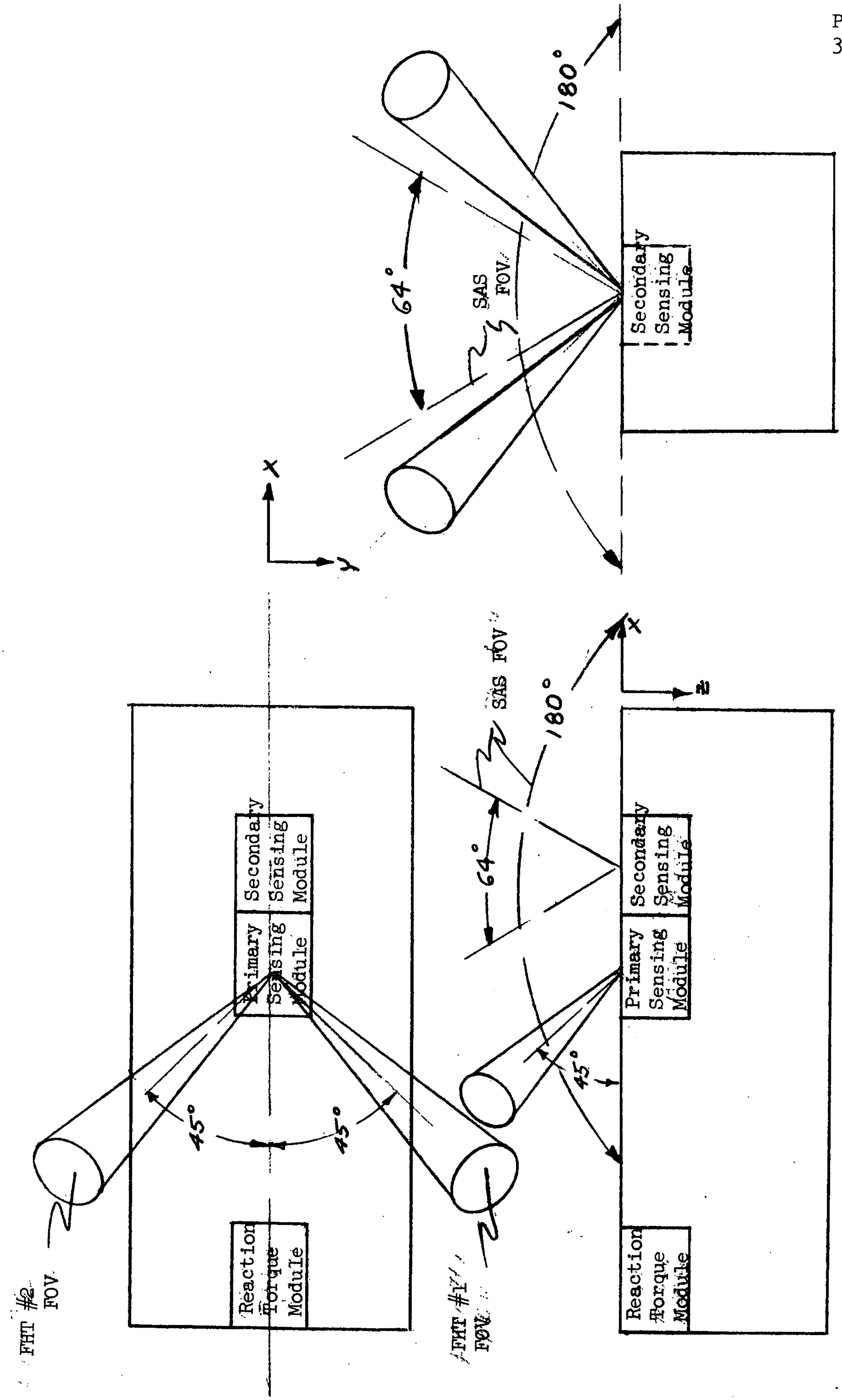


Fig. 27 S&C Subsystem Equipment Location & Orientation

Table 10

STANDARD EOS

S&C COMMAND LIST

Pitch + gas
Pitch - gas
Roll + gas
Roll - gas
Yaw + gas
Yaw - gas

Wheel speed nominal - roll
Wheel speed nominal - pitch
Wheel speed nominal - yaw

Step magnet + pitch
Step magnet - pitch
Step magnet + roll
Step magnet - roll
Step magnet + yaw
Step magnet - yaw

Bias pitch +
Bias pitch -

Turn off TARS
Turn off wheel drive electronics #1
Switch to sun hold mode
Turn off pneumatics
Turn off wheel drive electronics #2

FHT #1, #2

Power on/off
Command elevation offset positive/negative
Command azimuth offset positive/negative

Target Star coordinates (data)
100 vectors (typical)

2 20-bit words per vector
1 12-bit word per vector

Ephemeris Update (data)
6 elements
1 12-bit word per element

Table 11

STANDARD EOS

S&C TELEMETRY LIST

Fixed Head Tracker #1, #2 Elevation Output
Fixed Head Tracker #1, #2 Azimuth Output

TARS Output - roll
TARS Output - pitch
TARS Output - yaw

Wheel speed and direction - roll
Wheel speed and direction - pitch
Wheel speed and direction - yaw

Wheel drives output - roll
Wheel drives output - pitch
Wheel drives output - yaw

Temperature - Star Trackers #1, #2
Temperature - Gyros - roll/pitch/yaw
Temperature - Wheels - roll/pitch/yaw
Temperature - Sun sensor
Temperature - Electronics (20)

Gyro Spin Motor detector - Roll
Gyro Spin Motor detector - Pitch
Gyro Spin Motor detector - yaw

Rate Gyro Package outputs (3)

Star Sensor sun presence (2)
ACP Valve currents (16)
ACP pressures, temperatures (4)
Operating Mode (4)
Sun Sensor outputs (5 - 11 bit words)

E. Reliability

The reliability goal for the standard EOS Stabilization and Control subsystem is 0.850 for one year of orbit life. This goal is attained in the designs described as shown in Fig. 28.

- Subsystem allocated reliability 0.850
- Subsystem operating life .... 1 year, 8760 hrs.
- To meet the allocation, effective overall failure rate must not exceed $\lambda = 18.3 \times 10^{-6}$
- With all modes of redundancy considered allocated λ per S/S element is shown in Sheet 2.
- With all redundancy modes considered, computed effective λ is shown in Sheet 2.
- Subsystem is approx. 23% over redundified in the design. R = 0.869 vs 0.85 required.
- Subsystem will meet requirement with only one module containing "FHST" assemblies 2 each and "TARS" assemblies 1 each.
- Forms of redundancy used: Standby, complex, binomial, time share, functional back-up, multi-modal.

Definition:

Effective failure rate connotes that failure rate pertaining, which recognizes all of the redundancy techniques applied. Example: Actual failure rate summation is the sum of all serially applied parts failure rates. Effective failure rate is computed when the redundancy of two assemblies is calculated. 2 units in standby redundancy each having a failure rate summation of 18.5×10^{-6} .... R = 0.985. Then effective failure rate is: 1.73×10^{-6} by calculation.

Fig. 28 Standard EOS Stabilization and Control Subsystem (Sheet 1 of 2) Reliability Analysis Summary

SUBSYSTEM ELEMENT	QTY	1	2	3	4	5	6	REQ'D $\epsilon\lambda$	COMPUTED $\epsilon\lambda$	$\Delta\epsilon$ / RED %	REMARKS
FIXED HEAD STAR TRACKER	4			X			X	1.6	1.2	.4 / 25%	S/S CAN OPERATE WITH ONLY 2 FHST + 1 TARS. REDUNDANCY IS COMPLEX & MULTIMODAL
FHST ELECTRONICS	4			X							
3 AXIS RATE SENSOR	2		X				X	2.0	1.5	.5 / 25%	BACKS UP FHST & TARS.
TARS ELECTRONICS	2		X								SAME REDUNDANCY SCHEME.
SOLAR ASPECT SENSORS	5					X		2.5	2.0	.5 / 20%	BINOMIAL. 5 AVAILABLE 4 NEEDED. BACKS UP FHST + TARS
SAS ELECTRONICS	5					X					
LASER CORNER CUBES	4	←			N/A		→	0.3	0.3	—	PASSIVE DEVICES. NO TYPE OF REDUNDANCY APPLIED.
DATA PROCESSOR UNIT (COMPUTER)	1				X		X	4.0	4.0	—	UNIT HAS 20 FUNCTIONS FOR S/S $\epsilon\lambda = 4.0$ $\epsilon\lambda\bar{x} = 0.2$ per f.
REACTION WHEELS	3							3.5	3.5	—	NEED ALL 3 WHEELS.
WHEEL DRIVE ELECTRONICS	2		X								FUNCTIONAL BACK UP - COLD GAS
SINGLE AXIS MAG TORQUER	3		X	X				2.4	2.0	.4 / 16.6%	BACK UP TO WHEELS IN COMPLEX-STD BY MODE.
TORQUER ELECTRONICS	3		X	X							
ACP THRUSTER		←			N/A		→			.5 / 25%	THRUSTERS (16) ARE SCORED AGAINST ACS/S. MULTIMODE AND TIME SHARE REDUNDANCY ARE USED.
ELECTRONICS	4						X	2.0	1.5		
SUMMATIONS								$\Sigma\epsilon\lambda$ 18.3 / $R_{S/S}$ .85	$\Sigma\epsilon\lambda$ 16 / R .87	EXTRA RED'C'Y ≈ 23%	SYSTEM MEETS R required WITH ONE MODULE OF 2 FHST + 1 TARS

- 1 PARA: ACTIVE REDUNDANCY
- 2 STAND-BY; ONE ON LINE ONE BACK UP
- 3 COMX: COMPLEX REDUNDANCY, MULTI CASE STD-BY OR ACTIVE
- 4 TIME SHARE: T.S. SHARED ON LINE TIME
- 5. BIN. BINOMIAL R. X UNITS AVAILABLE Y NEEDED.
- 6 MuMo: MULTIMODAL REDUNDANCY. ONE OR MORE MODES USED + TIME SHARING

Fig. 28 Standard EOS Stabilization and Control Subsystem Reliability Analysis Summary (Sheet 2 of 2)

III. PROBLEM AREAS

A. Development

(1) To avoid redundant equipment to meet the reliability goal, particular attention should be paid to increasing the MTBF of the three-axis rate sensor.

(2) Some difficulty has been experienced in obtaining unit-to-unit performance repeatability of electronic star trackers.

Further tracker development will be necessary to consistently achieve the accuracy goal (one part in 800), and, at the same time, realize a unit cost substantially below $100K.

B. Operations

(1) Electronic star tracker operation is degraded by stray magnetic fields. It is therefore important to avoid locating sources of EMI near the star sensors and to calibrate them in their actual environment. In the latter situation, it may be necessary to provide shielding for the sensors. These considerations imply the possible need for a magnetic simulation facility for spacecraft development and testing.

(2) For fine pointing mission spacecraft, the need to assure alignment stability between the star trackers and the mission equipment within about 10 sec after spacecraft shipping and handling, and exposure to the launch and orbital thermal environments will impose important spacecraft design constraints.

(3) To assure initial star acquisition without requiring wide-angle optics, the star tracker-shuttle GNC reference alignment must be controlled to about 0.25 deg. This alignment could be held through spacecraft-shuttle adapter design, calibrated by an optical link, or obviated by having the spacecraft compute its attitude from Shuttle liftoff.

IV. REFERENCES

1. Pollak, R. J., "Low-Cost OAO Stabilization and Control System"; PE-2; Feb, 1971.

2. Pollak, R. J., "Low-Cost SEO Stabilization and Control"; PE-22; Mar, 1971.

3. Pollak, R. J. - "Low-Cost SRS Stabilization and Control"; PE-42; Apr, 1971.

4. Davis, W. R. et al - "Space Precision Attitude Reference System (SPARS) Phase-1A Final Report: Vol. I Technical Report; LMSC-B099041; SAMSO TR-69-72; Feb 1969

5. Advanced Spacecraft Capabilities Study; Technical Proposal, Vol. I; LMSC-B200910; Oct 1970, SECRET

6. "Reaction Wheels for Space Vehicles"; Engineering Report; Publication No. 6311-5, Nov, 1963.

7. Pollak, R. J. - "Standard Spacecraft S&C Subsystem"; PE-62; July, 1971

8. Earth Observatory Satellite (EOS) - Definition Phase Report - Vol. I; GSFC; Aug, 1971.

APPENDIX A

SUMMARY OF IMPLEMENTATION TRADEOFFS

(1) Earth horizon vs Star sensing for earth pointing missions
Decision: Utilize horizon sensors for some earth pointing missions

Rationale for utilizing star sensing:

Although:

- Most missions require medium or high precision attitude determination,
- The same star sensing equipment can be used both for attitude determination and attitude control reference thereby reducing cost,
- Ephemeris information will be routinely available via the TDRS,
- The added on-board/ground data processing burden will not be prohibitively costly,
- The sharing of star sensing/data processing development costs will prevail.

It is also true that:

- Many missions require earth-orientation of spacecraft,
- Many mission require only low or medium attitude precision,
- Earth horizon sensor & software are lower cost,
- No ground support required (ephemeris update), associated with horizon sensing.

(2) Fixed Head vs Gimballed Head Star Tracker
Decision: Utilize two or more fixed head trackers.

Rationale for using FHT's:

Fixed Head Trackers:

- FOV is tied to vehicle attitude
- Acquisition time is function of initial conditions
- Sensitive to ambient/stray magnetic fields
- Small FOV (compared to GHT FOV)

Gimballed Head Trackers

Less reliable (moving parts)
Less accurate (alignment)
More difficult to protect thermally
Shorter life (mechanical wearout modes)
More costly
Greater size, weight, power
Larger vehicle skin cutout

(3) Reaction Wheels vs Mass Expulsion only
Decision: Utilize 3 single-axis reaction wheels

Rationale for utilizing wheels:

- Momentum Storage/Variable Torque reduces ACP thruster duty cycle and attitude control impulse required.
- Momentum storage/variable torque decouples S&C and ACP designs, simplifying functional interfaces.
- Momentum storage/variable torque permits decoupling of spacecraft configuration design (disturbance torques) and ACP design (impulse required).
- Momentum storage/variable torque facilities fine pointing control.
- Simplified functional interface between S&C and ACP Subsystems improves electrical interface, facilitates module interchangeability.
- Reduces propellant exhaust product deposition.

(4) Magnetic Unloading vs Mass Expulsion only
Decision: Utilize three-axis magnetic torquers for wheel momentum unloading with mass expulsion backup.

Rationale for inclusion of magnetic torquing:

- Reduced disturbance torque environment (roll yaw gyroscopic torque) associated with lower mean angular momentum (wheels kept unloaded)
- Reduced thruster duty cycles/relaxed life testing specs
- Reduced reaction wheel wear/increased life
- Virtual elimination of unloading disturbance torques and wheel vibration disturbance
- Reduced control gas impulse load
- Ease of implementing torque commands in software
- Added ability to trim residual environmental torques (such as gravity gradient, solar, magnetic).

(5) Pitch Momentum Bias (PMB) vs Zero Momentum System
Decision: Implement a zero momentum S&C concept.

Rationale for zero momentum bias:

- PMB System sizing is spacecraft/orbit dependent.
- Accuracy requirements difficult to meet by passive means (unless spacecraft design constrained to limit environmental torques).
- Other requirements (i.e., offset pointing, slewing, payload scanning, orbit adjust, and restabilization) favor the use of gyros and active control.
- System design does not preclude using wheel speed bias as backup or adding momentum wheel to fill special need.

(6) Three-axis magnetometer vs software model of earth's field for unloading torque computation
Decision: Put model of earth's magnetic field in software.

Rationale for software approach:

- High precision associated with magnetometers not required for unloading function
- Low precision ephemeris needed on board for other purposes
- Magnetic cleanliness not required
- Software more reliable and lower cost
- Simple software model is adequate

(7) Two Gyro Triads vs Skewed Six Pack
Decision: Provide orthogonal triad, possibly mounted in a skewed relationship. Use only one triad where adequate for reliability.

Rationale for choosing two separate triads:

- The reliability of a six-pack of gas-bearing gyros is not necessary to meet the reliability goal.
- Six-pack refurbishment requires processing more "good" gyros than does a triad.
- Stocking six-packs costs more than triads.
- Two skewed orthogonal triads can provide a higher reliability than parallel redundant units, although possibly less accurate than the six-pack.

<u>Note</u>: Similar reasoning applies to reaction wheels - but here the motivation to go to the six-pack is even lower, due to wheels having higher reliability than gyros. 63

(8) Combined Reaction wheel horizon sensor vs separate sensor and wheels.
Decision: Separate sensor and wheels

Rationale for using two single-purpose devices:

- Requirements can be met without momentum bias.
- Combined device mounting requirements more constraining.
- Separate devices easier to place in proper modules (i.e., sensor colocated on TARS mounting block).
- Only one type of wheel needs to be stocked.
- Less likelihood of requiring replacement of functioning equipment.
- Separate sensor potentially more accurate.
- Non-scanning wheels have longer life.
- Separate devices easier to adapt to expanded mission requirements, e.g., synchronous equatorial.
- Separate devices easier to operate.

3

APPENDIX B

PRECISION ATTITUDE DETERMINATION ERROR BUDGET (3σ)

	(Sec)
Fixed Head Tracker Error*	17
Ephemeris Error (60 m)	12
IRA Errors (drift, scale factor, bias)*	8
	22 sec

It is assumed that alignment uncertainties between the tracker boresight and experiments can be calibrated to a satisfactorily low level.

*After calibration.

PE-103

PE-103

STANDARD EOS

COMMUNICATIONS, DATA PROCESSING,

& COMMUNICATIONS SUBSYSTEM

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: STANDARD EARTH OBSERVATORY SATELLITE CDPI SUBSYSTEM	EM NO: PE-103 REF: DATE: 11/30/71
AUTHORS: M. Loeb, D. F. Wald	APPROVAL: J.E. Bolton ENGINEERING Wayne Miller

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EM NO: PE-103
DATE: 11/30/71

STANDARD EARTH OBSERVATORY SATELLITE - CDPI SUBSYSTEM

1.0 Introduction

This report is prepared to discuss the Communication, Data Processing, Interface and Instrumentation Subsystem (CDPI) as it relates to the formulation of the projected standardized spacecraft configurations considered in the Payload Effects Follow-On Study. The CDPI includes the spacecraft transmitter, receivers, antennas, all data handling and routing hardware, digital processing equipment and system instrumentation; it is the major interfacing subsystem of the spacecraft. A block diagram of the CDPI is shown in Fig. 1

A basic problem in devising a standardized CDPI, which will provide the significant reduction in non-recurring costs standardization affords, is to definitize the optimal functional hardware/software entities making up the CDPI and its related spacecraft subsystems. If these system elements are conceived of as physical modules, then the problem converts to a definition of function and description of each module in turn. Some modules may be essentially unifunctional, as would be obtained with a transmitter or receiver in a single package. Other modules may be multifunctional, as with a unit containing command, telemetry and selected data processing equipment. Although the possible variations in module function and content are very large, practical considerations, e.g., size, maintenance requirements, power demands, packaging constraints, etc., limit the number of different module configurations which may be actually employed.

An underlying functional requirement for all modules is the need for module interfacing. This need may be satisfied by individual interface modules, or selected interface functions may be allocated to multifunctional modules. It should be noted that this need as well as demands imposed by primary functional requirements, may be served by software; hardware is not the only alternative.

In order to realize optimal functional groupings in modules two approaches may be used, as illustrated in Fig. 2. Two types of modules, sensor and interface, are conjectured in each configuration on the figure. In the first, a separate design approach, the sensor modules are structured to be highly self-contained; they include support electronics, data accumulation, encoding and formatting. The only functions of the interface module are to furnish terminal connections for the sensor modules and to supply a single line output from the interface module via a multiplexer. This approach maximizes sensor module independence since the interface module is not required for proper sensor module functioning. There is a concurrent reduction in interface module requirements.

With the second approach, an integrated design, the capability of the interface module is expanded. This allows for a considerable simplification of both the sensor modules and the overall system design; however, sensor modules are completely dependent upon the interface module for operation. This is largely due to inclusion of the support electronics in the interface module.

Both approaches have advantages and disadvantages. Neglecting the impact of standardization for the moment, the following effects are obtained with each approach:

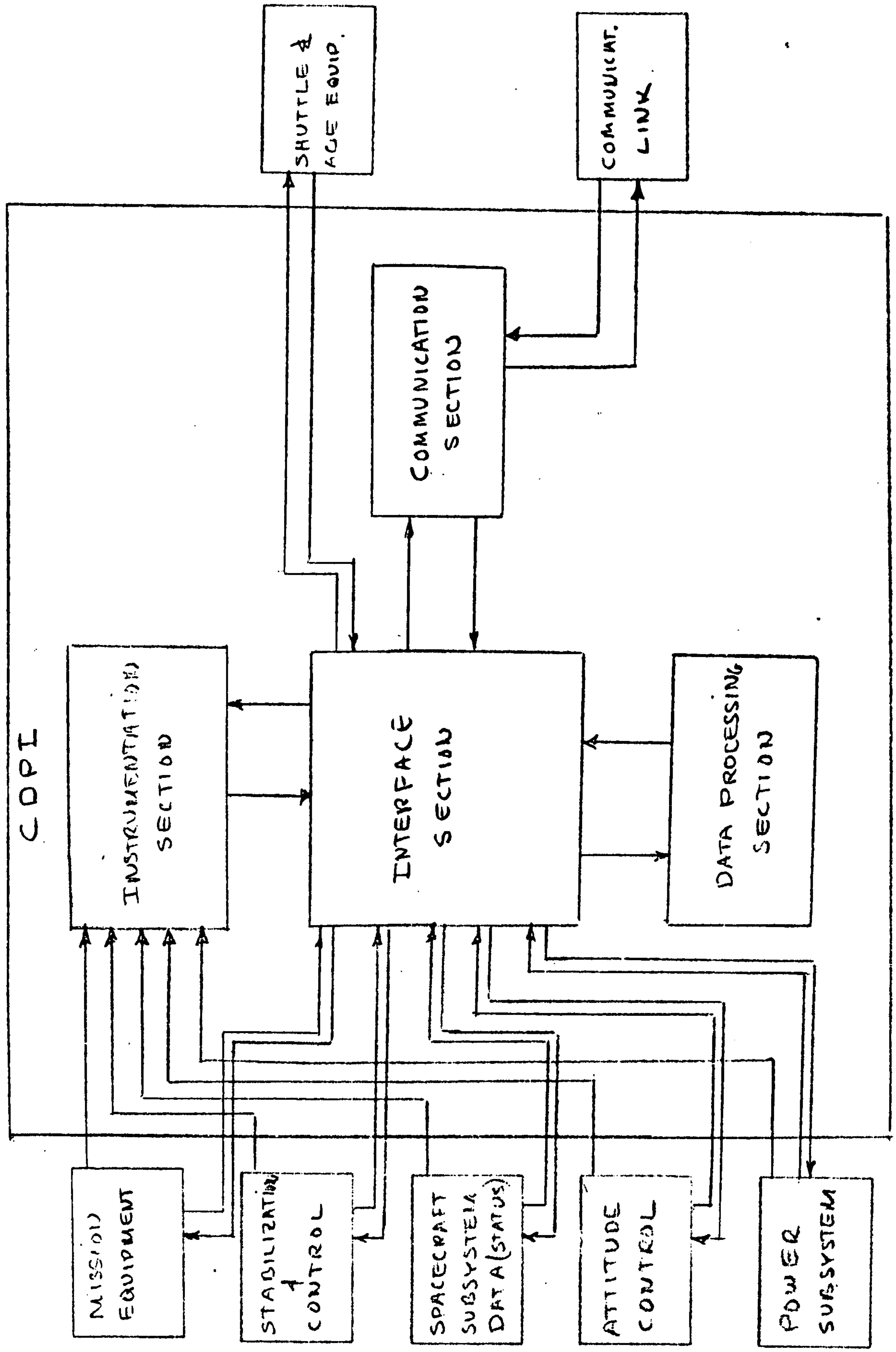
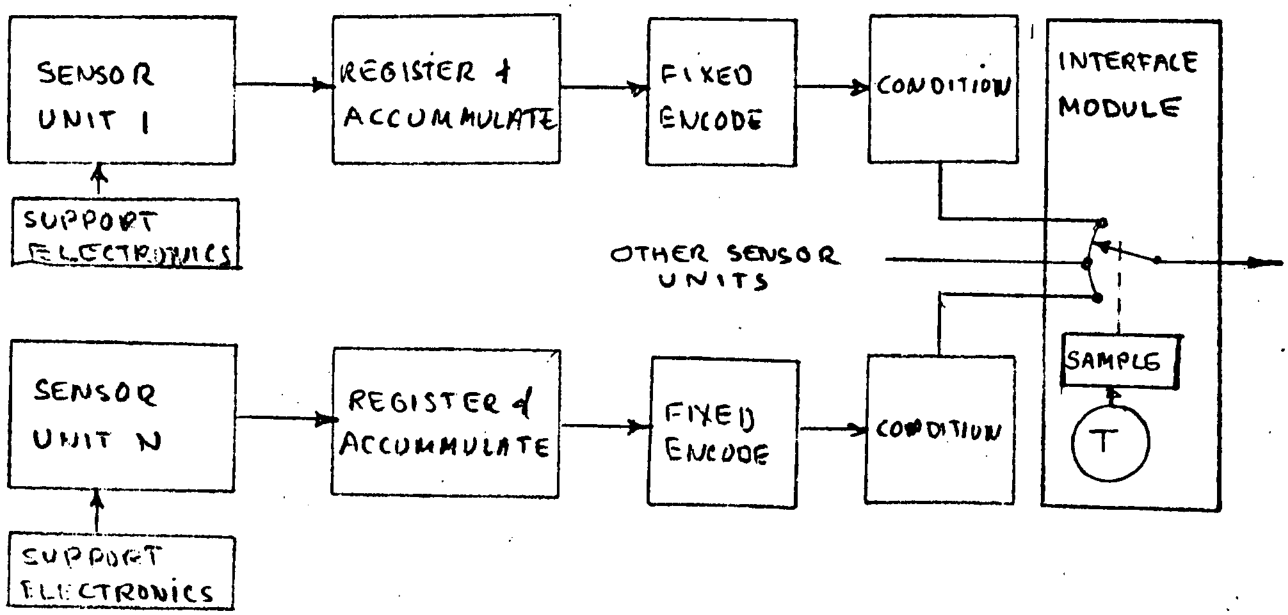


Fig. 1 Communication, Data Processing, Interface & Instrumentation Subsystem

APPROACH #1 - SEPARATE DESIGN



APPROACH #2 - INTEGRATED DESIGN

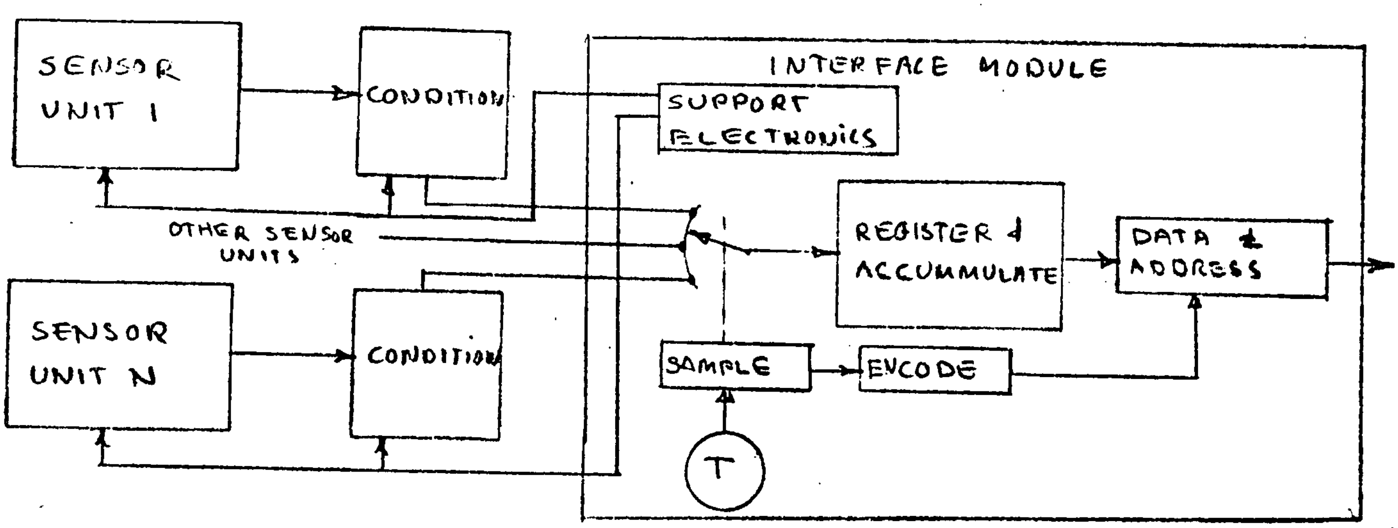


Fig. 2 Alternate Approaches to CDPI Subsystem Design

Separate Approach

Effect	Reason
Easier "add-on" capability	Module independence reduces possibility of changes to other modules when modules are added or subtracted from the module set making up the system.
More latitude in design	This is also a result of module independence. The designer of a specific module has control of a larger number of parameters.
Less reliable	This is a result of the larger number of component parts.
More costly	This is also due to increases in component or component part count.

Integrated Approach

Greater system simplification	Equipment commonalities are used to greatest advantage.
Lower cost	Physical realization results in a reduced part count.
Greater reliability	Another result of reduced part count.
More care exercised in design	Greater module interdependency.

The advantages and disadvantages listed above are less clear-cut when standardization is considered. This pertains in particular to costs. For example, if a system is designed which employs both standardization and a separate approach, the cost benefits of standardization might outweigh the cost benefits an integrated approach would yield. It is evident that the optimum method is to combine standardization and an integrated design to attain maximum benefits; however, the question arises as to the feasibility and realizability of this combination in standard-spacecraft CDPI subsystems. In order to answer this question the relative characteristics of standardization and integrated designs must be considered.

The use of standardization, as it pertains to standardized spacecraft, may be interjected in two ways:

- Establishment of standard design criteria — This includes standards for dimensioning electrical connection, thermal isolation and other specifications
- Utilization of inventoried items — Units or modules which do not require changing, regardless of mission

There are many spacecraft components which may be standardized and inventoried, e.g., amplifiers, antennas, transmitters, couplers, etc. In the ideal, configuring a specific spacecraft for a particular mission would be reduced to assembling and interconnecting selected items from the inventory. This ideal may not be fully realizable, considering the broad range of potential missions to which standard spacecraft may be applied.

The integrated approach, a design and analysis process which is oriented towards total system requirements in working on a particular subsystem or system element, results in employing multifunctional system entities to satisfy intersystem commonalities. In the spacecraft the two primary foci for such integration are the computer and interface sections of the CDPI. The computer software may be used to meet selected needs of the CDPI, Stabilization and Control (S&C), Attitude Control and Mission Equipment while the interface section provides interface hardware functions for these same subsystems. However, the computer software and interface units are the most likely system elements which would require some customization to reflect mission or equipment changes if they are introduced.

In order to minimize the impact of custom design on the computer software and interface units of the CDPI, so that the cost benefits of combining an integrated approach with standardization may be obtained to the maximum extent, a core design concept will be employed for the CDPI and related subsystems. This approach is illustrated in Fig. 3. The core unit consists of those structural, electrical, thermal, and other subsystem entities which will be utilized in every mission and are invariant, despite mission. The entire core unit may be a composite standardized, integrated structure; it may be partially or completely modularized for convenience in fabrication, maintenance and repair. The core unit includes standardized interface "faces" to which standardized spacecraft subsystem components are attached. If special purpose mission or other equipment must be included in a particular flight, a customized interface component will be interjected between the non-standard equipment and the standard interface. One face of the customized interface component will be standardized to enable interconnection with the core unit; however, other faces may be special purpose in character.

It is important to note that the core unit concept applies to computer software as well as the CDPI hardware. Various computational, input/output, checkout and fault isolation, executive, data manipulation and processing routines will remain resident in the computer, regardless of mission. This software forms the core set. Other software modules which may be added to the core set, in support of changing or adding new standardized equipment may also be preprogrammed to form a standard software module library. If special purpose software modules are required, they will include interfacing routines to the core set. Standardizing rules must also be established on data storage and resource allocation.

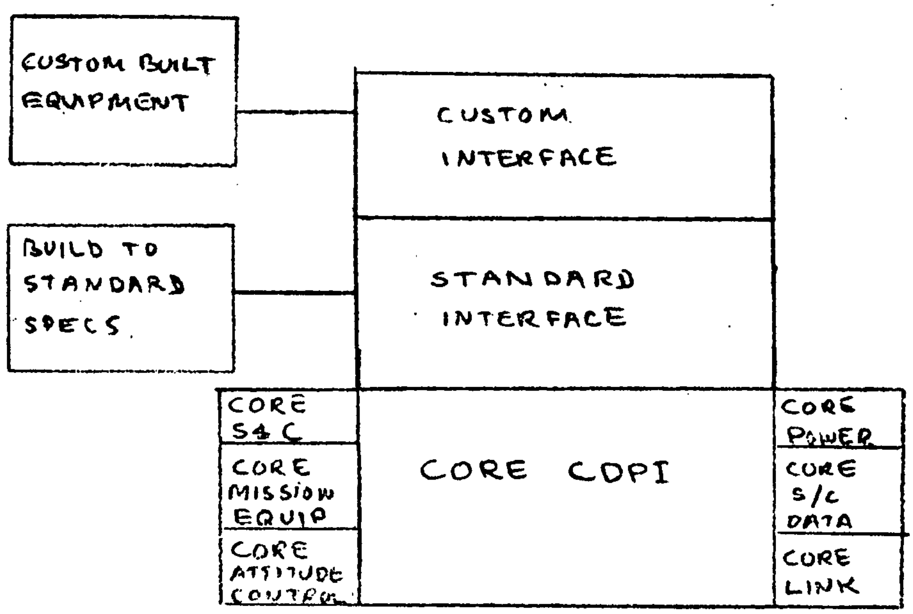


Fig. 3 Core Configuration for CDPI Subsystem

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The core unit concept applied to software provides significant cost savings, particularly in the area of program validation. After an initial validation of the core set and standardized modules, software validation for a particular flight is devoted primarily to special purpose sequences, program dynamics, module interfaces and timing. This represents a significant reduction in validation compared to a software system which would require detailed validation procedures on all software components prior to each flight.

In order to attain a definition of the standard spacecraft CDPI subsystem, including the core unit, and an identification of those components which may be standardized and those which may have to be customized, a point design will be prepared, based upon Earth Observatory Satellite (EOS) requirements. The point design will include representative equipment, equipment module configurations, inter and intra-module interfacing and input/output. Rationale and supportive data will be supplied to justify the approach used. 1971-75 technology will be assumed in the design derivation.

2.0 CDPI Functional Requirements

2.1 Basis for Requirements

The CDPI functional requirements presented here are established to meet the needs of a point design standard spacecraft capable of fulfilling EOS A-C missions, as described in Reference 1. These missions are constrained to 300-600 mile, polar, sun synchronous orbits, where communication to and from the ground is via the Mark I-C Tracking and Data Relay Satellite (TDRS) discussed in References 2-4, or a similar system. Provision is also made for communication between the EOS & Shuttle.

The functional requirements of the CDPI are determined by and subservient to the requirements of other spacecraft subsystems and the uplink/downlink communications and data handling system. Thus a significant change in one or more of these other system entities will have a profound effect on CDPI requirements and design. This dependency is increased through utilization of the integrated-system design approach, the methodology of the work performed here. In order to reduce the sensitivity of the CDPI to changes in the other systems, the CDPI capacity and capability will include margins for growth and change. This will necessitate exceeding the minimal demands of a point design to some degree. The point design minimal functional demands impacting the CDPI are identified in the following paragraphs; growth allowances will be introduced in the subsequent discussion of CDPI design.

2.2 Mission Equipment

The mission equipment may be subdivided into four classes, based upon output data rate and characteristics:

1. Ultra High Data Rates - Thematic Mapper
2. Medium & High Data Rates - Imaging Radiometers
3. Low Data Rates - Microwave Radiometer and Low rate sensors
4. Analog - TV, mission equipment monitoring

The required quantities and rates of data which must be handled by the CDPI for each of the above classes is listed in Fig. 4. It should be noted that the actual rate handled by the CDPI for each sensor will depend upon whether data buffering is included in the sensor support equipment. This choice is a subject of trade study.

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Sensor	No. of Channels	Data Rate (1)	Data Rate (2)	Data Quantity	Rationale for Rates & Quantities	Control
Oceanic Scanning Spectrometer	Not Deter.	1.76 Mb/sec	0.54 Mb/sec	2.7×10^9 pixels/ day	20 spectral images of 926 x 926 Km scene every 143 sec. 277 scenes/day - 2.7×10^9 pixels/day - 8 bits /pixel (no real time req.)	Not det. Assume 16
Sea Surface Temp Imaging Radio-meter	5 + 1 cali-brat.	1 Mb/sec(3)	0.33 Mb/ sec(3)	159,700 bit(3) buffer to in-clude El. Phy. Rad.	2 samples/IFOV-digitize to 10 levels (3) 28% scan ef-ficiency 10 bits/sample 10% synch.	16 discretes
Cloud Physics Radiometer	5 + 1 cali-brat.	.66 Mb/sec(3) (1.83 Mb/sec for both	.22 Mb/ sec(3) (.66 Mb/sec for both.)	Included in (3) above.	Included in above. (3)	16 discretes
Thematic Mapper	7	30-80 Mb/sec	30 Mb/sec	5×10^{10} pixels/ day. 6.4×10^8 bits/ scene	462 scenes, 3234 images 4200x4299x6x7+) 1300x1300x7)pixels/scene	Analog drive to gimbal or stepping mo-tor controller & gimbal 16 discretes
Upper Atmos. Sounder (LRIR)	4	<004 Mb/sec	No buffer	74 Mb/orbit 4	13×10^2 bits/sec - 95 min./orbit 4	10
Atmos. Pollution Sensor	3	< .01 Mb/sec	No buffer	included in above 4	included in above (4)	6
Passive Multi-Channel M Rad.	5	.01 Mb/sec	No buffer	included in above 14	Included in above (4)	12
RBV camera	1	10 MHz	analog	N/A	N/A	2
						94 discretes total + counter drive

(1) Generated from sensors A/D conversion
(2) Assuming forming bytes and buffering
(3) Treated as a pair in Reference 1
(4) Data quantity sum of three sensors

Fig. 4 Mission Equipment Data Handling Requirements

Input controls to the mission equipment, primarily discretes, will also be required to activate or deactivate various sensors, drive motors or other devices. Estimates of the control requirements are included in Fig. 4. It is seen that approximately 94 control discretes are estimated plus a positional analog or digital control for the Thematic Mapper.

2.3 Stabilization & Control Subsystem

The functional requirements of the CDPI which are established by Stabilization and Control Subsystem (S&C) demands fall into the three areas of computational services, uplink commands and downlink telemetry. The requirements in each of these areas are identified in Reference 5 and reiterated here for ease of referral in Figs. 5-7.

2.4 Attitude Control Subsystem

The functional requirements of the CDPI which are derived from the Attitude Control Subsystem (ACS) are in three main areas - signal steering, computational services and downlink telemetry. Signal steering is needed to route the driver pulses to the appropriate gas thrusters. Four propulsion modules, with 4 thrusters per module are specified. The selection of the thrusters is obtained from two computational algorithms. The first provides redundant couples, in the instance of propulsion module malfunction as shown in Figs. 8-9; the second processes commands issued by the S&C. The telemetry requirements are included in the Fig. 7 estimates for the S&C.

2.5 Uplink/Downlink Communication & Data Handling System

It is assumed that the uplink/downlink communication and data handling system incorporates tracking and data relay satellites at synchronous orbit altitudes, and all communication to and from the spacecraft will be via the relay satellites. As stated in Reference 2, with two relay satellites 132° apart, line of sight coverage is obtained for greater than 85 percent of the global area at low earth (300 nm) spacecraft orbits; 100 percent coverage is obtained at the operational orbit altitudes of 600 nm. Judicious placement of two relay satellites or the placement of 3 equally spaced relay satellites would enable utilizing real time communication. This eliminates the need for on-board spacecraft bulk storage, e.g., tape recorders.

The TDRS Mark I-C will be able to handle a broad range of frequencies in the RF spectrum, including VHF, S and K. High and ultra-high data rates would necessitate S or K band; there is no restriction on low data rates, e.g., 1 to 20 Kilobits per second. The actual selections of frequency band, relative to function are a subject of trade study.

Presuming no bulk storage and performance of a trade study on frequency band selection, functional requirements impacting the CDPI are as listed in Fig. 10.

• Attitude Sum	$(\Sigma\Delta\theta_i)$ i = 1, 2, 3 5/sec
• Attitude Rate	50/sec
• Gyrocompass Decoupling	(summed differences) 1 /sec
• Control Loop Compensation	3 axis
• Wheel Drive Modulation	1600 pps - 3 axis
• Attitude Control Thruster Modulation	3 axis
• Wheel Speed	1200 pps - 3 axis
• Magnetic Field Vector (computed)	magnitude, direction 1/10 sec
• Magnetic Torque Command	vector cross product - 1/sec
• High Frequency Antenna Pointing Vector	5/sec
• EOS Experiment Pointing Vector	> 100/sec
• Low Precision Ephemeris Update	1/sec
• Sun Sensor Signal Processor	1/sec
• 3-Axis Attitude Update	1/10 sec
• Star Sensors (3)	20 pps

Fig. 5 S&C Computational Services Required from CDPI

Pitch + gas
Pitch - gas
Roll + gas
Roll - gas
Yaw + gas
Yaw - gas

Wheel speed nominal - roll
Wheel speed nominal - pitch
Wheel speed nominal - yaw

Step magnet + pitch
Step magnet - pitch
Step magnet + roll
Step magnet - roll
Step magnet + yaw
Step magnet - yaw
Bias pitch +
Bias pitch -

Turn off TARS
Turn off wheel drive electronics
Switch to sun hold mode
Turn off pneumatics

Ephemeris Update
Update ephemeris (six elements)
    6 elements
    1 12-bit word per element

FHT #1, #2

    Power on/off
    Command elevation offset positive/negative
    Command azimuth offset positive/negative

Target Star Coordinates

    100 Vectors (0 to 378)
    2 20-bit words per vector (+4)
    1 12-bit word per vector

Fig. 6 S&C Command List

Fixed Head Tracker #1, #2, Elevation Output
Fixed Head Tracker #1, #2, Azimuth Output

TARS Output - roll
TARS Output - pitch
TARS Output - yaw

Wheel speed and direction - roll
Wheel speed and direction - pitch
Wheel speed and direction - yaw

Wheel driver Output - roll
Wheel driver Output - pitch
Wheel driver Output - yaw

Temperature - Star Trackers #1, #2
Temperature - Gyros - Roll/Pitch/Yaw
Temperature - Wheels - Roll/Pitch/Yaw
Temperature - Sun Sensor
Temperature - Electronics (20)

Gyro Spin Motor Detector - Roll
Gyro Spin Motor Detector - Pitch
Gyro Spin Motor Detector - Yaw
Rate Gyro Package Outputs (3)

Star Sensor Sun Presence (4)
ACP Valve currents (16)
ACP pressures, temperatures (4)
Operating Mode (4)
Sun Sensor outputs (-5 - 11 bit words)

Fig. 7 S&C Telemetry List

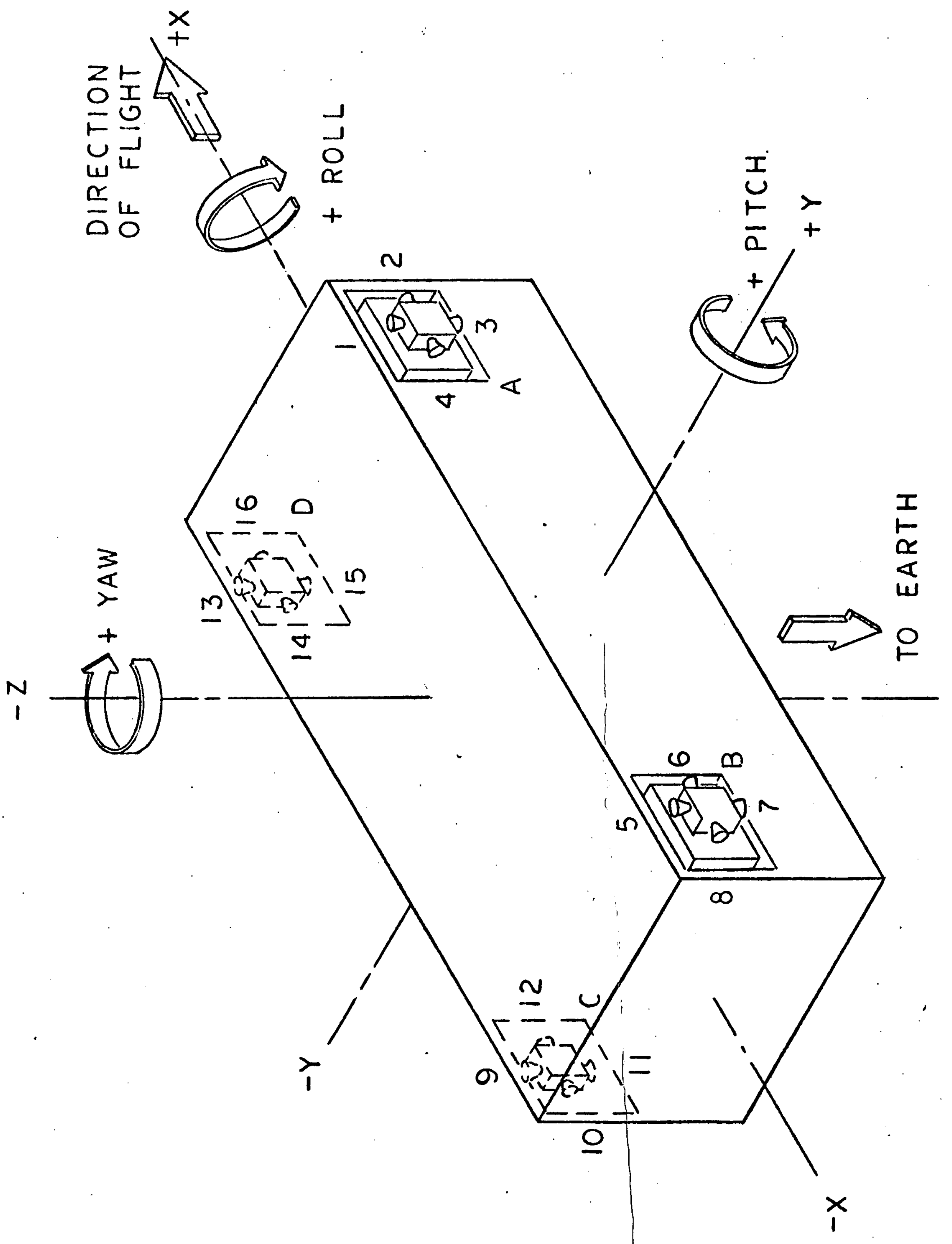

Fig. 8 Standard EOS Attitude Control Subsystem Installation

Motion	Active Thruster
+ Pitch	3 & 5, 9 & 15
- Pitch	1 & 7, 11 & 13
+ Roll	1 & 15, 5 & 11
- Roll	3 & 13, 7 & 9
+ Yaw	2 & 10, 6 & 14
- Yaw	4 & 12, 8 & 16

Fig. 9 Redundant Couples to Compensate for Disabling One Propulsion Module

	Item	Number
Uplink Communication -	Commands	512
	Ephemeris data	7 terms
	Orbit adjust data	4 terms
	Verification information	1 word
	Initialization information	undetermined
	Test & diagnostic data	undetermined
Downlink Communication-	Housekeeping & status monitoring*	500
	Mission equipment data	as per Fig. 4
	Verification information	1-3 words
	On-board data processing results	undetermined
	S&C data	as per Fig. 7
	Propulsion Subsystem data	as per Fig. 7
Ranging	Delay lock loop correlation	N/A
	Receive and transmit range codes	10-32 bit words at undetermined rate
Acquisition & Tracking -	Cooperative targeting	Trade Study
	Initial communication for acquisition	Trade Study
	Maintenance of lock-on	Trade Study

* Mixture of analog (low and high level), bilevel and digital.

Fig. 10 Uplink/Downlink Communication & Data Handling System Impact on CDPI Reqmts.

2.6 Power Subsystem

The power subsystem will require support from the CDPI to perform control switching functions. These controls include the preprogrammed or commanded switching discretes governing activation or deactivation of mission equipment, S&C components, etc. Approximately 150 discrete controls will be necessary. In addition voltage setting controls may be required to establish gain or bias settings in S&C, communications or mission equipment electronics. Six voltages over the range of 0-28 vdc will be assumed.

2.7 Instrumentation Requirements

In addition to mission equipment and S&C sensors, additional instrumentation is required to monitor the operating conditions and status of spacecraft equipment, including the mission equipment itself. This monitoring instrumentation consists of transducers and various pickoff devices located at selected points in the spacecraft. Some instrumentation sensors are active, requiring excitation voltages as inputs; however, a number are passive devices and handling of the output signals is the primary concern.

Because of the large number of instrumentation sensors required in the spacecraft, sampling and multiplexing will be necessary. In addition signal conditioning will be needed to transform output signals to proper voltage and impedance levels. Assuming a full complement of EOS mission equipment components, the signal conditioned instrumentation is estimated as follows:

Pressure transducers	17	
Temperature sensors	56	
Current sensors	30	
Voltage sensors	25	
Total	128	instruments needing signal conditioning

Other instrumentation will be necessary which will also require analog to digital conversion. These will be in two ranges - 0 to 5 vdc and 0 to 200 mv dc, where 110 such devices will be assumed. In addition, there are a number of bilevel channels which indicate either on/off or operation is within a required value. To enable flexibility in the system 128 analog and 128 bilevel channels will be assumed.

It should be noted that sensors for the mission equipment will be provided and identified by the mission equipment manufacturers. Thus, the requirements identified here serve to establish multiplexing and conditioning requirements rather than the instrumentation itself.

Summarizing, total instrumentation requirements are assumed as follows:

Analog low level	31 channels
Analog high level	225 channels
Bi-level	128 channels

Digital information regarding antenna pointing and Thematic Mapper offset will be required. Thus three channels for handling digital information will be necessary in addition to the above. 12 bit data words will be assumed sufficient.

3.0 CDPI TRADE STUDIES

The CDPI functional requirements, determined by both the spacecraft subsystem servicing demands and intra-CDPI communications, delay, timing, storage and processing needs, are given in the Functional Diagram of Fig. 11. There are eight primary functional information and control paths in the figure -

- Mission equipment to rf link (3)
- Up link command and data to storage and controlling functions
- S&C sensors to attitude controllers (3)
- Control of power to the various subsystems

Although the functions which must be accomplished in each of these paths are identified in the figure, actual mechanization of specific functions are dependent upon tradeoff results. The main tradeoffs are as follows:

- rf frequencies and equipment configuration
- software vs hardware
- utilization of data buffering in mission equipment data paths

Each of these will be discussed prior to development of the CDPI design.

3.1 RF Frequencies and Equipment Configuration

A number of possibilities exist; however, within the framework of projected TDRS capabilities, these reduce to four main options. These alternates are given in Figs. 12-15. The equipment complement to realize each of these options is identified in Fig. 16. The two primary options, 1 and 4, will be discussed in turn.

3.1.1 Option #1 Discussion

Option #1 divides the required user communication services among three bands. Back-up communications for both telemetry and command utilize VHF (135-150 MHz). The spacecraft has an omni-antenna which allows communication with the ground either directly or via the TDRS. Assume that telemetry, command and ranging are placed on S-Band (1.8 GHz-uplink, 2.25 GHz-downlink). Under this condition communications with the ground is possible only through the TDRS at S-Band. Wide-band data would be transmitted to the TDRS at K-band (15.3 GHz).

Because of the high ERP required to develop the requisite channel capacity, a large aperture (6 foot diameter) is required on the spacecraft. The narrow beamwidths associated with this large aperture (0.8° at K-Band and 5.5° at S-Band) will require a cooperative closed-loop antenna tracking system on both the TDRS and the spacecraft. Initial acquisition may be provided by either S&C computer processed commands or through data supplied via the VHF links. Coarse action tracking is then established at S-Band, and precision tracking ($\pm 0.1^{\circ}$ or better) is continued at K-band. The TDRS to spacecraft K-band link is used to develop antenna tracking error signals and tracking updates from the TDRS.

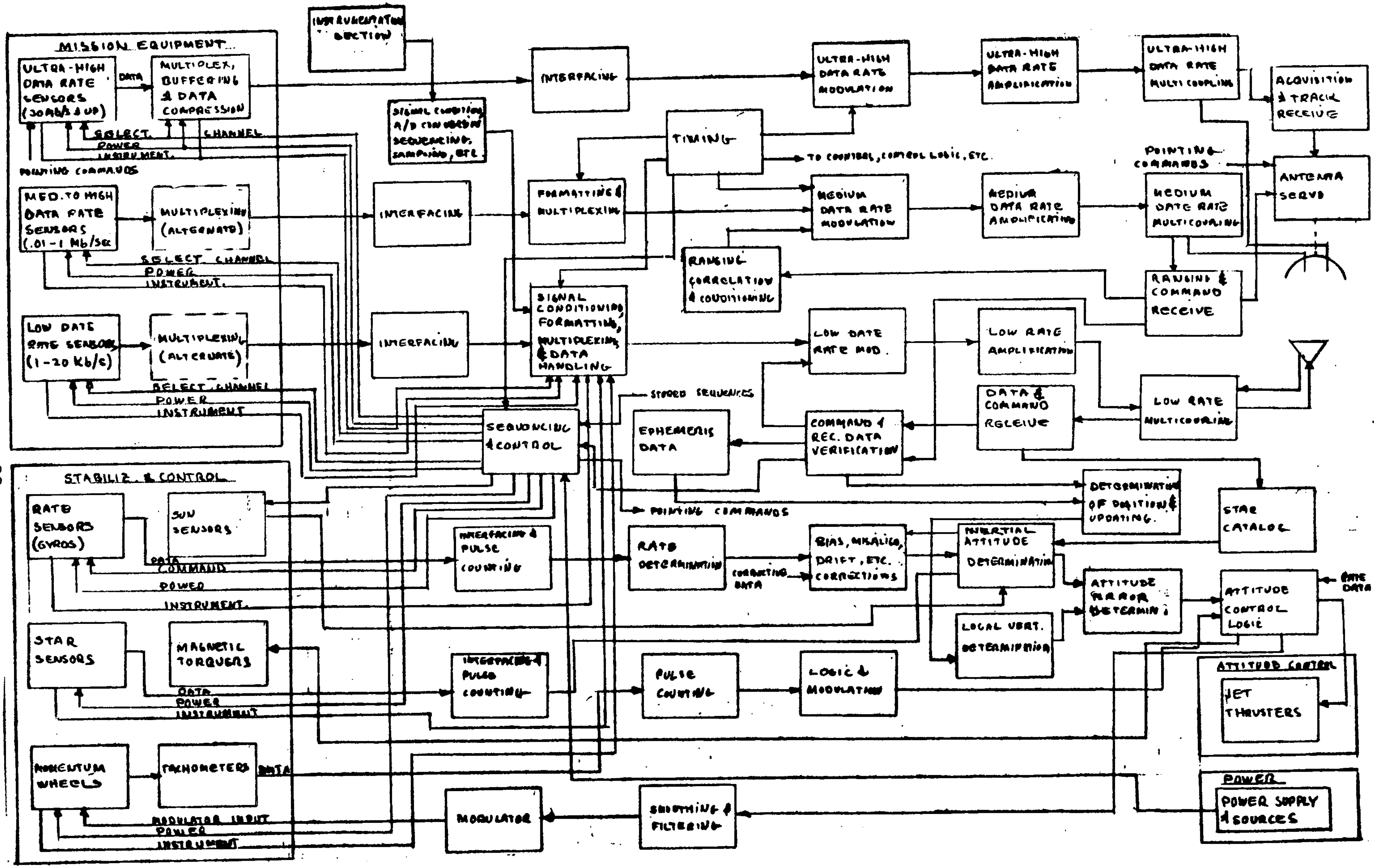

Fig. 11 Standard EOS CDPI Functional Diagram

Function	Implementation
Command & Computer Update	1. Data Rate - 10 Kb/s 2. Frequency - S-Band (Primary), VHF 3. Modulation - SIK on Range Code at S-Band 4. Output - Serial PCM to Computer for Command Decoding or Computer Update
Telemetry	1. Data Rate - 1.2 Mb/s on S-Band 10 Kb/s on VHF 2. Frequency - S-Band (primary), VHF 3. Modulation - QPSK on S-Band 4. Input - Serial PCM from VIP or MIRP
Wide Band Data	1. Primary - Data Rate 30-100 Mb/s Optional - Analog Bandwidth 10 MHz 2. Frequency - K-Band (15.3 GHz) 3. Modulation - Digital - PCM/QPSK on Carrier Analog - FM on Carrier 4. Input - Serial PCM from MOMS
Ranging	1. Digital - PN Range Code (1 Mb/s clock) 2. Frequency - S-Band (both directions) 3. Modulation - Uplink - Range Code SIK'ed by Command Downlink - Range Code and TM in quadrature on carrier
Acquisition & Tracking	1. Rough pointing from computer 2. Coarse pointing at S-band ($\pm 1^{o}$) 3. Fine Pointing and Tracking at K-Band ($\pm .1^{o}$)

Fig. 12 Frequencies & Equipment Configuration vs Function
Option #1

Function	Implementation
Command	1. Digital - Data Rate - 10 Kb/s 2. Frequency - VHF (149 MHz) 3. Modulation - PCM/Bi-Phase or FSK 4. Output - Serial PCM to computer for Command Verification & Decoding
Telemetry	1. Digital - Data Rate 1.2 Mb/s 2. Frequency - K-band (15.3 - GHz) 3. Modulation - PCM/Bi-Phase on Subcarrier 4. Input - Serial PCM from multiplexed spacecraft status and low rate experiments
Wide-Band Data	1. Primary - Digital - Data Rate = 30 Mb/s Optional - Analog - Bandwidth = 10 MHz 2. Frequency - K-Band (15.3 + GHz) 3. Modulation - Digital - PCM/QPSK on carrier Analog - FM on carrier 4. Input-serial PCM directly from experiment
Ranging	1. Primary - PN Range Code 10 Mb/s crack Backup - VHF minitrack 2. Frequency - K band primary (both directions) VHF Backup 3. Modulation - PCM/Bi-phase on carrier at K-band
Acquisition & Tracking	1. Rough pointing by computer 2. Coarse pointing by defocused K-band beam cooperative with TDRS 3. Fine pointing by focused K-band beam

Fig. 13 Frequencies & Equipment Configuration vs Function
Option #2

Function	Implementation
Command	1. Digital - Data Rate - 10 Kb/s 2. Frequency - VHF (149 MHz) 3. Modulation - PCM/Bi-Phase or FSK 4. Output - Serial PCM to computer for Command Verification & Decoding
Telemetry	1. Primary - Data Rate 1.2 Mb/s Backup - Data Rate 10 Kb/s 2. Frequency - K-Band primary VHF Backup 3. Modulation - PCM/Bi-phase 4. Input - Serial PCM from Multiplexer Hi/Lo Rate switching required
Wide Band Data	Same as Option #4
Ranging	Same as Option #4
Acquisition & Track	Same as Option #4

Fig. 14 Frequencies & Equipment Configuration vs Function Option #3

Function	Implementation
Command	1. Digital - Data Rate - 10 Kb/s 2. Freq. - Primary K-Band, VHF Backup 3. Modulation - S/K on Range Code on K-Band, PCM/Bi-Phase or FSK on VHF 4. Output - Serial PCM to Computer for Command Verification & Decoding
Telemetry	1. Digital - Data Rate - 1.2 Mb/s 2. Frequency - K-Band, VHF Backup (Low Rate) 3. Modulation - S/K on Returned Range Code 4. Input - Serial PCM from Multiplexed Spacecraft status and Low Rate Experiments
Wide-Band Data	1. Primary - Digital - Data Rate - 30 Mb/s Optional - Analog - Bandwidth - 10 Mb/s 2. Frequency - K-Band 3. Modulation - Digital PCM/QPSK on carrier Analog - FM on carrier 4. Input - Serial PCM directly from experiment
Ranging	1. Primary - PN Range Code 30 Mb/s clock Backup - VHF Minitrack 2. Frequency - Primary K-Band, Backup VHF 3. Modulation - PCM/Bi-Phase on Carrier (primary)
Acquisition & Tracking	1. Rough pointing by computer 2. Coarse pointing cooperatively by defocused K-Band beam 3. Fine pointing by focused K-band beam

Fig. 15 Frequencies & Equipment Configuration vs Function
Option #4

Option #	VHF		S-Band		K-Band		
	RCVR	XMTR	RCVR	XMTR	RCVR	Driver(s)	TWTA
1	X	X	X	X	(X) Used for Acq + Track	X	X
2	X	X				XX	X
3	X	X			(X) Used for Acq + Track	XX	X
4	X	X			X	XX	X

(X) Equipment poorly utilized.

Fig. 16 Equipment Requirements to Realize Options

The VHF (Backup) communication channels employ straightforward PCM/Bi-phase modulation techniques. Because of the omni antenna on the spacecraft, RFI from earth-based stations may become a serious problem. Some system of automatic channel switching in the presence of RFI may be needed if RFI cannot be controlled by frequency assignments. Should the problem of RFI appear too serious to handle by the above methods, spread-spectrum techniques involving modulo-2 addition of data to a user-specific PN code will be employed.

The S-Band and K-Band channels with moderate to ultra-high data rate requirements must use highly efficient modulation techniques in order to utilize the available relay channel capacity most effectively. The S-Band uplink (tracking and command) is composed of a PN range code (approx. 1.2 Mb/s clock) which is sequence-Inversion-keyed by the 10 Kb/s command data stream. The PN code is coherently regenerated at the spacecraft when it is used for detection of the command data stream and for return to the ground terminal. The coherently regenerated range code and the 1.2 Mb/s data stream both modulate the downlink transmitter with a quadrature relationship between them. Because of this orthogonal independence between the ranging and telemetry signals, they can both be transmitted asynchronously if required, in the same channel. The signals can be separated on the ground using standard synchronous detection techniques.

The K-band downlink (30 Mb/s wideband data) employs straight forward quadri-phase shift keying (QPSK) for improved communications efficiency. Alternate bits of the 30 Mb/s data stream are assembled into two 15 Mb/s data streams by means of conventional clocked steering logic. These two 15 Mb/s data streams each bi-phase modulate low-level carriers with a quadrature phase relationship. The two level carriers are combined and then amplified by a 50W TWTA. At the ground terminal, the two 15 Mb/s data streams are recovered by asynchronous detection and recombined into the original 30 Mb/s data stream.

Using the channel capacities of the spacecraft TDRS as a basis, as shown in Fig. 17, link analysis data for Option 1 are shown in Fig. 18. These data are summarized in Fig. 19 for convenience. It is seen that adequate link margins are obtained with the specified antenna gains and transmitted output power levels; however, data rates must remain close to specifications, particularly at S-band.

Options 2 and 3 do not utilize S-band for any functions and command communication is via VHF. Though these options represent some simplification through the elimination of S-band, a more advantageous configuration is provided in Option 4.

Link Designation	Nominal Frequency (GHz)	Direction	Normalized User ERP (dBW)	Mk 1 G/T (dB/K^{o})	Mk 1 ERP (dBW)	Normalized User G/T (dB/K^{o})	Norm. Overall* Link P/KT (dB-Hz)
VHF	0.137	User to TDRS	0.0	- 14.0	-	-	+ 44.6
VHF	0.149	TDRS to User	-	-	+26.0	-30.0	+ 54.6
S-Band	2.25	User to TDRS	0.0	0.0	-	-	+ 33.8
S-Band	1.80	TDRS to User	-	-	+39.0	-30.0	+ 44.9
K-Band	15.3	User to TDRS	0.0	+ 12.0	-	-	+ 28.4
K-Band	14.3	TDRS to User	-	-	+52.0	-30.0	+ 38.7

*Includes -1.0 dB degradation in TDRS to Ground Link.

Fig. 17 User/TDRS Channel Capacities

VHF (User to TDRS):

*Available Normalized Channel Capacity (ERP = 0 dBW)	+ 44.6 dB-Hz
Actual User ERP (5W, + 10 dB Gain)	+ 17.0 dBW
Available Actual Channel Capacity	+ 61.6 dB-Hz
Channel Signalling Rate (10 Kb/s)	+ 40.0 dB-Hz
Req'd S/N (PCM/Bi-Phase Mod)	+ 10.0 dB
Req'd Channel Capacity	+ 50.0 dB-Hz
Link Margin (No RFI)	+ 11.6 dB

VHF (TDRS to User Spacecraft):

*Available Normalized Channel Capacity (G/T = -30 dB/°K)	+ 54.6 dB-Hz
Actual User G/T (-4 dB Gain, 400°K)	- 30.0 dB/°K
Available Actual Channel Capacity	+ 54.6 dB-Hz
Channel Signalling Rate(10 Kb/s)	+ 40.0 dB-Hz
Req'd S/N (PCM/Bi-Phase Mod)	+ 10.0 dB
Req'd Channel Capacity	+ 50.0 dB-Hz
Link Margin	+ 4.6 dB

S-Band (User Spacecraft to TDRS):

*Available Normalized Channel Capacity (ERP = 0 dBW)	+ 33.8 dB-Hz
Actual User ERP (10W, +30 dB Gain)	+ 40.0 dBW
Available Actual Channel Capacity	+ 73.8 dB-Hz
Channel Signalling Rate (1.2 Mb/s)	+ 60.8 dB-Hz
Req'd S/N (PCM/QPSK Mod.)	+ 7.0 dB
Req'd Channel Capacity	+ 67.8 dB-Hz
Link Margin	+ 6.0 dB

S-Band (TDRS to User Spacecraft):

*Available Normalized Channel Capacity (G/T = -30 dB/°K)	+ 44.9 dB-Hz
Actual User G/T (+26.5 dB Gain, 1000°K)	- 3.5 dB/°K
Available Actual Channel Capacity	+ 71.4 dB-Hz
Channel Signalling Rate (1.2 Mb/s - Range Code Clock)	+ 60.8 dB-Hz
Req'd S/N (SIK of 10 Kb/s data on range code)	- 10.0 dB
Req'd Channel Capacity	- 50.8 dB-Hz
Link Margin	+ 20.6 dB

*Available Channel capacities used in analysis are obtained from: "GSFC Mark 1 Tracking and Data Relay Satellite (TDRS) System Concept", Phase A Study Final Report, Vol. 2, December 1969.

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Fig. 18 Option #1 Link Analyses

K-Band (User Spacecraft to TDRS):

*Available Normalized Channel Capacity (ERP = 0 dBW)	+ 28.4 dB-Hz
Actual User ERP (50W, +45 dB Gain)	+ 62.0 dBW
Available Actual Channel Capacity	+ 90.4 dB-Hz
Channel Signalling Rate (30 Mb/s)	+ 74.8 dB-Hz
Req'd S/N (PCM/QSPK Mod)	+ 7.0 dB
Required Channel Capacity	+ 81.8 dB-Hz
Link Margin	+ 8.6 dB

K-Band (TDRS to User Spacedraft):

*Available Normalized Channel Capacity (G/T = -30 dB/°K)	+ 38.7 dB-Hz
Actual User G/T (+44 dB Gain, 10,000°K)	+ 4.0 dB/°K
Available Actual Channel Capacity	+ 72.7 dB-Hz
Channel Signalling Rate (10 Kb/s)(1)	+ 40.0 dB-Hz
Req'd S/N (PCM/Bi-Phase)	+ 10.0 dB
Req'd Channel Capacity	+ 50.0 dB-Hz
Link Margin	+ 22.7 dB

* Available Channel capacities used in analysis are obtained from: "GSFC Mark 1 Tracking and Data Relay Satellite (TDRS) System Concept", Phase A Study Final Report, Vol. 2, December 1969.

(1) Assumes a 100 Kb/s PN Code for cooperative tracking control purposes.

Fig. 18 Option #1 Link Analyses (Cont.)

Communication Channel		User Xmtr. Power or G/T	User Antenna Type & Gain	Channel Capacity (dB-Hz)	Signalling Rate (Mb/s)	Channel Function	Modulation Format	Multi-Access Capability	Channel Margin (dB)
VHF	User to TDRS	5W (+7dBW)	Omni -4 dB	+ 61.6	0.01	Backup Telemetry	PCM/Bi-Phase on Carrier	Assigned Channel Slots	+ 11.6
VHF	TDRS to User	- 30.0 dB/°K	Omni -4 dB	+ 54.6	0.01	Backup Command	PCM/Bi-Phase on Carrier	Same as above	+ 4.6
S-Band	User to TDRS	10 W (+10dBW)	6' dish +30 dB	+ 73.8	1.2	Primary TM + Returned Range Code	Dual PCM/Bi-Phase in Q Quadrature TM on one channel ranging on other	Assigned Channels	+ 6.0
S-Band	TDRS to User	- 3.5 dB/°K	6' dish +26.5dB	+ 71.4	1.2 (Range code clock) 0.01 Command	Primary Command + Range Code	Sequence inversion keying (SIK) of command on the range code	Same as above	+ 20.6
K-Band	User to TDRS	50W (+17dBW)	6' dish +45 dB	+ 81.8	30	Wide Band Data	PCM/QSPK on Carrier	None	+ 8.6
K-Band	TDRS to User	+ 4.0 dB/°K	6' dish + 44 dB	+ 72.7	0.1	Cooperative Antenna Tracking Data	PCM/Bi-Phase on Carrier	None	+ 22.7

Fig. 19 Option #1 Link Analysis Summary Table

3.1.2 Option #4 Discussion

Option #4 divides the communication services between only two bands. VHF is used for backup telemetry and command, while all primary services (ranging, telemetry, command and wideband data transmission) employ K-Band. VHF operation is identical with Option #1. At K-Band, the uplink service (TDRS to USER spacecraft) employs a PN ranging signal which is sequence-inversion keyed (SIK) with command data. The downlink service (USER spacecraft to TDRS) has a single 50 watt TWTA which amplifies both the telemetry/ranging and wideband data carriers with the power divided in the ratio of 1 to 24 (2 watts and 48 watts, respectively). The wideband data carrier is identical with Option #1, but the telemetry data is SIK 'd on the 30 Mb/s clock ranging signal on the other carrier.

Acquisition and tracking are similar to Option #1, with rough antenna pointing done by VHF commands or computer derived signals and tracking performed by use of the K-Band signals. The acquisition procedure must be modified, however, because of the very narrow beamwidth of the K-Band antenna. Either electrical or mechanical defocusing of the beam is necessary, or else a mechanical or electronic scanning of the narrow beam is required to achieve lock-on.

Data obtained from a link analysis of Option #4 is given on Fig. 20 using the channel capacity information supplied in Fig. 17 as a basis. As with Option #1 adequate link margins are obtained at both VHF and K bands. It should be noted that these data are conservative, since they include sufficient margins for system losses. A block diagram of a configuration meeting Option #4 requirements is shown in Fig. 21 for illustrative purposes.

Comparing Option #1 with Option #4, the following relative advantages are obtained:

Option #1	Option #4
Greater utilization of projected TDRS capability	Pre-emphasis of K-band
Medium-high data rate communication feasible without development of K-band TDRS & spacecraft equipment	Entire program dependent on K-band developments
More complex equipment configuration	Simplification of communication section
Antenna feed, rotary joints, etc., more of a problem	Simpler front end configuration

Although a number of mission equipment sensors are projected for future use which require ultra-high bandwidths, e.g., Thematic Mapper and Aperture Radar, the majority utilize much lower bandwidths. S-band is adequate for these lower rate sensors, presuming the aggregate of their rates is within bounds. Because of this Option #1 is more desirable in that the communication section capability is better matched to needs. For this reason, Option #1 will be used as the basis for the CDPI communication section design. It should be noted, however, that Option #1 would be less satisfactory when medium-high data rates exceed 1.2 Mb/sec. This would occur if buffering is not incorporated in the high data rate channel. This matter will be examined in the discussion of buffering, Para. 3.3.

VHF Telemetry (User Spacecraft to TDRS):

Available normalized channel capacity (ERP = 0 dBW)	+ 44.6 dB-Hz
Actual User ERP (5W, +10dB gain)	+ 17.0 dBW
Available Actual Channel Capacity	+ 61.6 dB-Hz
Channel Signalling Rate (10 Kb/s)	+ 40.0 dB-Hz
Req'd S/N (PCM/Bi-Phase Mod)	+ 10.0 dB
Req'd Channel Capacity	+ 50.0 dB-Hz
Link Margin (No. RFI)	+ 11.6 dB

VHF Command (TDRS to User Spacecraft):

Available Normalized Channel Capacity (G/T: -30 dB/^{O}K)	+ 54.6 dB-Hz
Actual User G/T (-4 dB Gain, T = 400^{O}K)	- 30.0 dB/^{O}K
Available Actual Channel Capacity	+ 54.6 dB-Hz
Channel Signalling Rate (10 Kb/s)	+ 40.0 dB-Hz
Req'd S/N (PCM/Biphase Mod)	+ 10.0 dB
Req'd Channel Capacity	+ 50.0 dB-Hz
Link Margin	+ 4.6 dB

Fig. 20 Option #4 Link Analysis

K-Band - Wide Band Data (User Spacecraft to TDRS):

Available Normalized Channel Capacity (ERP = 0 dBW)	+ 28.4 dB-Hz
Actual User ERP (48W, +45 dB Gain)	+ 61.7 dBW
Available Actual Channel Capacity	+ 90.1 dB-Hz
Channel Signalling Rate (30 Mb/s)	+ 74.8 dB-Hz
Req'd S/N (PCM/QSPK Mod)	+ 7.0 dB
Required Channel Capacity	+ 81.8 dB-Hz
Link Margin (Wide Band Data)	+ 8.3 dB

K-Band - Telemetry & Ranging (User Spacecraft to TDRS):

Available Normalized Channel Capacity (ERP = 0 dBW)	+ 28.4 dB-Hz
Actual User ERP (2 watts, +45 dB Gain)	+ 48.0 dBW
Available Actual Channel Capacity	+ 76.4 dB-Hz
Channel Signalling Rate (Telemetry = 1.2 Mb/s)	+ 60.8 dB-Hz
Req'd S/N (PCM/S/K Mod)	+ 10.0 dB
Req'd Channel Capacity	+ 70.8 dB-Hz
Link Margin (Telemetry)	+ 5.6 dB
Channel Signalling Rate (Ranging = 30 Mb/s Clock)	+ 74.8 dB-Hz
Req'd S/N (PCM/Bi-Phase)	- 10.0 dB
Req'd Channel Capacity	+ 64.8 dB-Hz
Link Margin (Ranging)	+ 11.6 dB

Fig. 20 Option #4 Link Analysis (Cont.)

K-Band - Command & Ranging (TDRS to User Spacecraft):

Available Normalized Channel Capacity (G/T = -30 dB/°K)	+ 38.7 dB-Hz
Actual User G/T (+44 dB Gain, 1000°K)	+ 4.0 dB/°K
Available Actual Channel Capacity	+ 72.7 dB-Hz
Channel Signalling Rate (Command = 10 Kb/s)	+ 40.0 dB-Hz
Req'd S/N (PCM/SIK Mod)	+ 10.0 dB
Req'd Channel Capacity	+ 50.0 dB-Hz
Link Margin (Command)	+ 22.7 dB
Channel Signalling Rate (Ranging = 30 Mb/s Clock)	+ 74.8 dB-Hz
Req'd S/N (PCM/Bi-Phase)	- 10.0 dB
Req'd Channel Capacity	+ 64.8 dB-Hz
Link Margin (Ranging)	+ 7.9 dB

Fig. 20 Option #4 Link Analysis (Cont.)

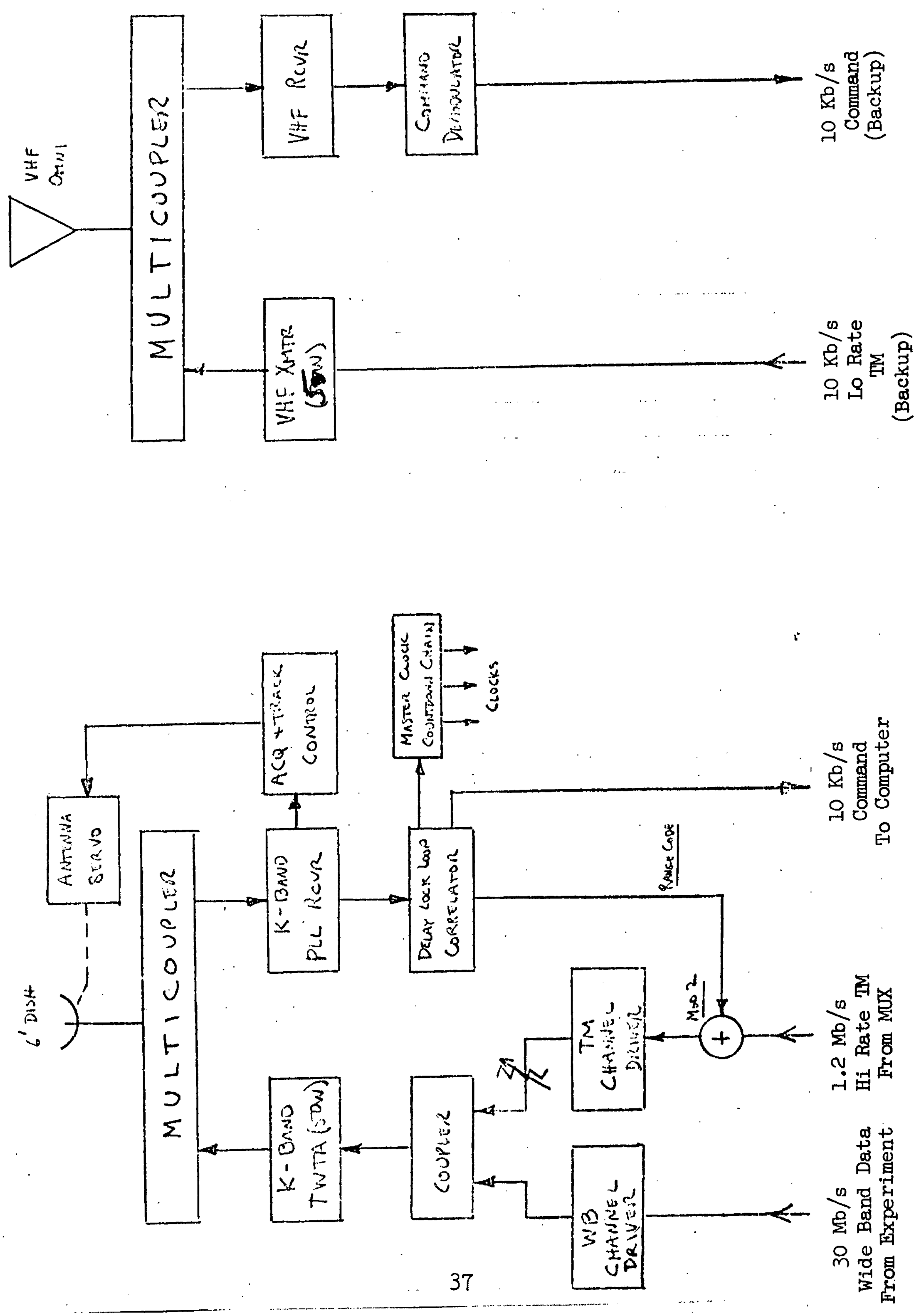

Fig. 21 Option #4 Block Diagram

3.1.3 Addition of Video Camera

Although the functional requirements of the baseline design do not include provisioning for a video camera, the impact of such a sensor on the design is of concern. Since the signal bandwidth is so wide, i.e., greater than 10 MHz, K-band is necessary. Option #4 might prove to be the better choice, considering all mission equipment requirements.

Presume simultaneous operation of the 30 Mb/sec digital link, TM, ranging and video camera. In this case the spacecraft effective radiated power (ERP) must be increased 3 to 4 db. This may be accomplished by increasing antenna diameter or transmitted output power. A compromise solution consists of raising the antenna diameter to 8 ft (47.5 db) and the power output from the TWT amplifier to 60 watts. A link analysis based on these values is shown in Fig. 22, assuming a power split ratio of 15:40:5 for the ultra-high rate digital data, video camera, and TM links, respectively.

There are problems in mechanizing the composite system described above, adding to RDT&E costs. These include requirements for precise control of the relative power levels of the different carriers, maintenance of a high degree of frequency stability, and attaining high TWTA linearity to minimize the presence in the signal of inter-modulation products. These problems, coupled with increased power and antenna dimensioning will necessitate more detailed examination, if a video camera must be included in the mission equipment sensors at a later date.

3.2 Software vs Hardware

If the functions of the CDPI, delineated on the Functional Diagram of Fig. 11, are examined it is seen that many may be realized through computer processing software, instead of by hardware implementation. This pertains in particular to S&C computational processes, logic control of the equipment serviced by the CDPI, command verification and processing and low data rate telemetry sequencing and formatting. Other functions, e.g., control sampling of sensor data, data processing support of high data rate sensors through gridding or other techniques (as identified in Reference 1) etc., may also be implemented.

In the integrated approach employed in this design, utilization of software is maximized to obtain the advantages of equipment commonalities. Other benefits include the significantly increased flexibility and relative ease of making changes in system function or operation. Also, depending upon what is actually done, ground computation and data handling may be considerably reduced.

There are however some disadvantages to the use of integrated design in software. The most predominant are as follows:

- A failure in the computer or software will result in failure of a number of spacecraft subsystems

- Software preparation, validation and verification has proven to be expensive in the past.

- Relative newness of computer technology has resulted in lack of familiarity on the part of some potential users. In addition there have been negative experiences with software which have been broadly disseminated, and in some cases wrongly interpreted.

K-Band - Wide-Band Data (User to TDRS):

Available Normalized Channel Capacity (ERP = 0 dBW)	+ 28.4 dB-Hz
Actual User ERP (15W, +47.5 dB Gain)	+ 59.2 dBW
Available Actual Channel Capacity	+ 87.6 dB-Hz
Channel Signalling Rate (30 Mb/s)	+ 74.8 dB-Hz
Req'd S/N (PCM/QPSK Mod)	+ 7.0 dB
Req'd Channel Capacity	+ 81.8 dB-Hz
Link Margin (Wide Band Data)	+ 5.8 dB

K-Band - Video Camera (User to TDRS):

Actual User ERP (40W, +47.5 dB Gain)	+ 63.5 dBW
Available Actual Channel Capacity	+ 91.9 dB-Hz
Channel Signalling Rate (10 MHz)	+ 70.0 dB-Hz
Req'd S/N* (FM, M - 2 Mod)	+ 16.0 dB
Req'd Channel Capacity	+ 86.0 dB-Hz
Link Margin (Video Signal)	+ 5.9 dB

K-Band - T/M & Ranging (User-to-TDRS):

Actual User ERP (5W, +47.5 dB Gain)	+ 54.5 dBW
Available Actual Channel Capacity	+ 82.9 dB-Hz
Channel Signalling Rate (1.2 Mb/s)	+ 60.8 dB-Hz
Req'd S/N (PCM/Bi-Phase)	+ 10.0 dB
Req'd Channel Capacity	+ 70.8 dB-Hz
Link Margin (TM)	+ 12.1 dB

* For 40dB Post-detection S/N

Fig. 22 Option #4 - Link Analysis Including Video Camera

In order to mitigate against the broad influence of computer failure, alternate equipment and paths must be available which bypass the computer. In the designs of the standard spacecraft, the only kinds of failure of major consequence are electrical shorts which may irrepairably damage the components and spacecraft tumbling. The former type of failure is compensated by fail safe design; the latter is accounted for by a separate S&C/Attitude Control assembly. The unit obtains a pointing reference from the available sun sensor; a separate rate sensing unit is included to mitigate against failure of the primary rate sensing source.

Software preparation, verification and validation costs have proven to be expensive in the past because of the criticality of the software to system or personnel safety, mission/equipment limitations and the attendant demands for error free programs. If reasonable care is exercised and a prudent program arranged, software failure need not result in catastrophic failure of the spacecraft, i.e., non-repairable or non-correctable modes. Errors that may be present and not recognized until after injection into orbit may be corrected by link communication or Shuttle revisit. There are of course certain software sections that are more critical than others, e.g., discrete controls, command processing, power controls, etc. These will require somewhat greater care in preparation and validation. Other sections are not so demanding, e.g., mission equipment output data handling, telemetry, etc. The primary concern, from the viewpoint of safety, is that these software routines do not adversely affect the more critical software sections. Thus a less exhaustive preparation and validation effort is needed in these software sections.

Despite negative attitudes, computers are being used at an increasing rate and sophistication in space applications. Advances in technology, e.g., MOS-LSI circuitry, which will provide significant improvements in size, power, weight, cost and reliability, will accelerate computer applications even further. This growth cannot be ignored and provisioning the standard spacecraft with a computer, with the flexibility and system simplifications it affords, will aid in attaining a system with broad applicability and prevent early antiquation of the spacecraft.

3.3 Data Buffering

In Fig. 4 of Para. 2.2, two data rates are specified for each mission equipment sensor package. The first, called the raw data rate, is the rate from the package after digitizing, combining and serializing the data from all sensor channels on a common bus; the second is the rate after buffering the common bus data in some storage medium, e.g., magnetic cores. Buffering provides a reduction in output rate because useful data output from the sensors is not continuous. For example, consider the Sea Surface Imaging Radiometer, an instrument which typifies the medium rate sensors. This sensor contains a rotating mirror, with its rotation axis parallel to the spacecraft roll axis. Since the spacecraft is oriented normal to the local vertical, the mirror will pick up useful information during only a fraction of a full revolution. This fraction is termed scanning efficiency. In the case of the imaging radiometer, the efficiency is only 27.8 percent. Thus, for approximately two-thirds of the time no useful data are obtained from the instrument. Thus, if the useful data obtained from one scan are stored in a buffer, the output rate from the bufffer may be one-third that of the input rate. Note that input to the buffer is intermittent and output is continuous; however, the reduction in rate enables reducing communication channel and ERP requirements.

The penalty for buffering is of course the buffer itself. This penalty may be high. For example, a 140 kilobit buffer is needed with the Ocean Scanning Spectrometer, while a 1.5 - 2.5 megabit capacity is necessary with the Thematic Mapper.

In order to make a decision regarding utilization of buffering, the penalty of buffering compared to improvements in the communication section must be considered. In addition other factors must be examined, e.g., state of the art. The two main problem areas are the ultra-high rate sensors, particularly the Thematic Mapper and the medium-high rate sensors.

Consider first the Thematic Mapper. In accordance with 1971-75 technology, reasonable design parameters are a 50 watt TWT amplifier and a six ft antenna at K band. With quadriphase modulation, transmission to a TDRS rated satellite, etc., a reasonable bound on data rate is approximately 50 Mb/sec. This corresponds to a Thematic Mapper with approximately 50 percent scan efficiency. According to Reference 1, a scan efficiency between 30 to 80 percent is estimated. The 50 percent efficiency represents 10 percent less than the mean of 55 percent. It will therefore be assumed that buffering will not be required in the standard spacecraft ultra-high rate channel. If further experience in Thematic Mapper design results in lower efficiencies, it will be assumed that buffering will be included as part of the mission equipment.

A somewhat different result is obtained with the high data rate sensors, as seen in Fig. 23. With buffering the rate from all sensors is approximately 1.2 Mb/sec; without buffering this rate would increase to approximately 3.2 Mb/sec. The higher rate would necessitate utilization of K-band to maintain the link margins at conservative levels; however, S-band may be used at the lower rate, as seen in previous discussions.

In general, utilization of K-band for the high rate data and elimination of buffering might prove less costly; however, actual savings are indefinite at this time. Of course utilization of an all K-band configuration would aid in resolving the problem (Option #4), but this is not a good strategy, as previous discussion showed.

Since an absolute resolution of this tradeoff cannot be made at this time, the CDPI designs developed in this memo will assume medium-high data rate sensor information are buffered; however, the option of K-band and no data buffering will be kept open for further study.

4.0 CDPI Design

4.1 Communication Section

A block diagram of the Communication Section, as it is related to the other CDPI sections making up the CDPI subsystem, is shown in Fig. 24. This section is in accordance with Option #1 of the trade study, Par. 3.1.1. As seen in the figure, the communication section is in three parts, where the first part is a K-band configuration devoted exclusively to the ultra-high 30 Mb/sec data rate sensor information communication. The second part, an S-band configuration combines medium-high data rates, ranging and command functions. The third part provides a VHF channel for telemetry of low data rate information and serves as a backup link should there be a failure in the other channels. Backup consists of safing the system or performing those mission functions for which communication remains open, i.e., ultra-high and low data rate

Find Impact of a 3.2 Mb/s TM Rate vs a 1.2 Mb/s Rate:

S-Band (Option #1):

Req'd channel capacity for +6.0 dB Margin is 67.8 dB-Hz at 1.2 Mb/s

For 3.2 Mb/s, Channel capacity must increase +4.3 dB

This would require either:

1) Increasing TWTA (solid state amplifier is out) power to 27 watts

2) Increasing Antenna diameter to 10 feet

3) A combination of the above, e.g.,
An 8 foot dish and a 20 watt TWTA (solid state is again out)

K-Band (Option #4):

No significant impact. Power could be split between wide band + TM carriers in the ratio of 9:1 (45 watts + 5 watts respectively to accommodate this requirement)

Fig. 23 Buffering Tradeoff

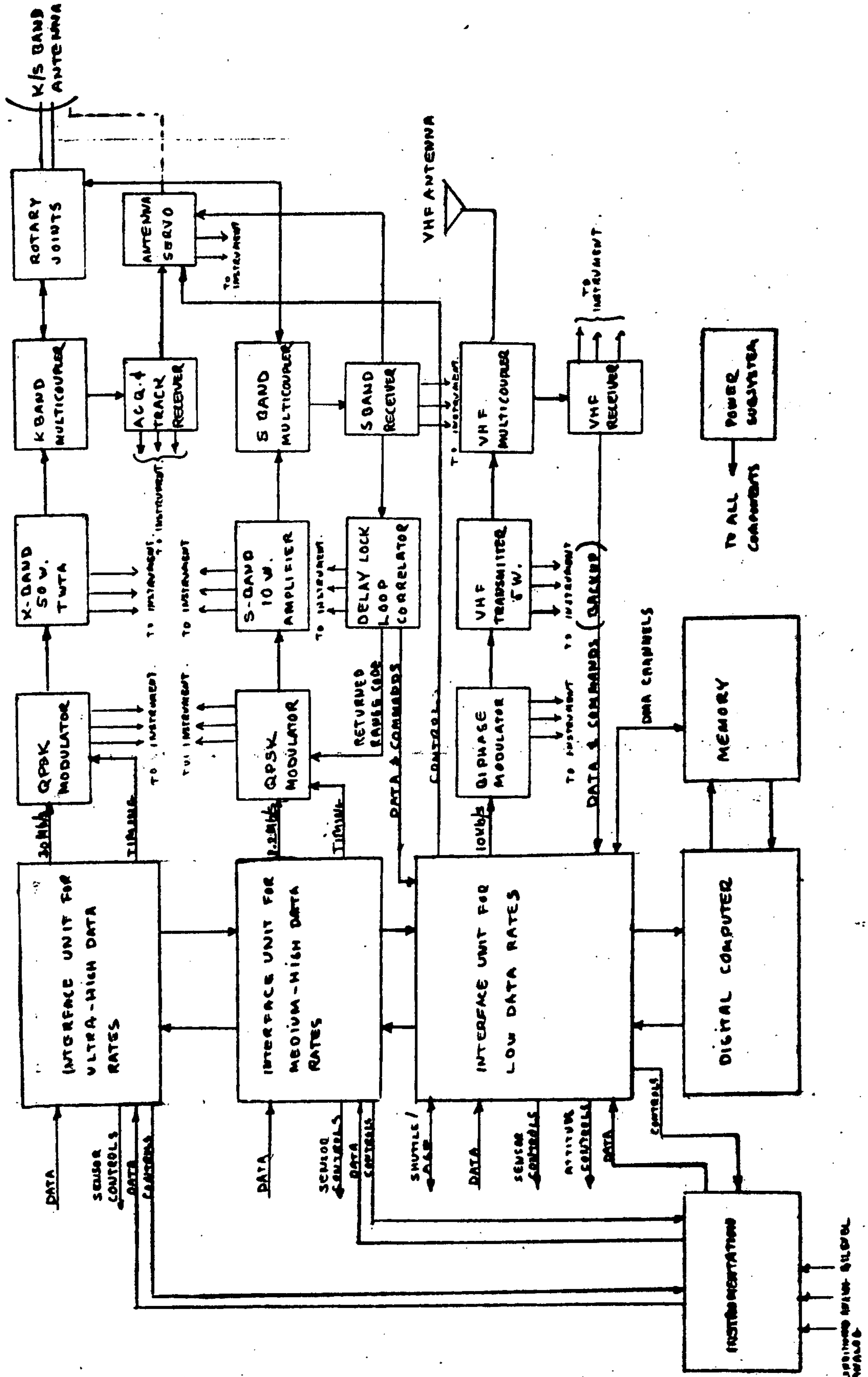

Fig. 24 Standard EOS CDPI Communication Section

operation with loss of S-band, or medium high and low data rates with loss of K band. It should be noted that ranging data would be lost with loss of S-band. A backup ranging system may be included in K-band as an alternate; however, this backup is not included in the design since safing the system is the primary protective mode.

The main specification for each major Communication Section component is given in Fig. 25. Estimates of weight, size, power and cost of these components are listed in Fig. 26.

Housekeeping instrumentation outputs and power control inputs are required to and from the Communication Section. Interconnection, multiplexing, conditioning, etc., for these components are provided via the Ultra-high Data Rate Interface Unit.

4.2 Ultra-High Data Rate Interface Unit

As seen in Fig. 24 of Para. 4.1, there are three units in the Interface Section, corresponding to the three channels of the Communication Section. The first unit, which interconnects the Thematic Mapper with the K-band modulator, as well as other interface units, is shown in Fig. 27.

A number of information and control channels are provided in the Interface Unit. Output from the Thematic Mapper 30 Mb/sec output channel is fed via coaxial cable to a buffer amplifier. The amplifier provides the gain necessary to drive the K-band modulator as well as supplies a proper impedance match to the Thematic Mapper/modulator circuitry. The amplifier may be switched on and off via power control logic.

Instrumentation channels are also provided to obtain status and housekeeping data. These include signal conditioned and non-signal conditioned analog data and bilevel data. A subcommutating sampler and multiplexer is supplied in the analog outputs and a sample and hold register is furnished for the bilevel data. These devices reduce interface interwiring requirements.

Three types of control are incorporated - power on-off, as with the amplifier previously discussed, sensor channel control through sequence drive logic, and a slewing control to orient the Thematic Mapper in accordance with either ground command or a routine stored in the digital computer. It should be noted that timing registers and drive logic are included to serve the particular requirements of the components served by the Interface Unit. The originating source for the timing is in the Interface Unit for the low data rate sensors.

Additional channels have been incorporated in the figure to illustrate the hardware impact on the Interface Unit of a video camera. Although this design does not include a video camera, the extra capability is added on to the Interface Unit to allow for future growth.

The basic description of each component in the unit is listed in Fig. 28. Assuming packaging with plug-in logic boards into a cabling/connector assembly for all timing, gating and logic, and mounting the amplifiers on the logic boards in addition to the remaining circuitry; the following estimates are obtained -

1.	K-Band TWTA	50 watts out, Gain + 20 dB
2.	K-Band QPSK Mod.	30 Mb/s in, Output ~ 0 dBW
3.	K-Band PLL Rcvr.	$BW_{(RF)}$ = 200 MHz, NF = + 12 to + 14 dB
4.	K-Band Multicoupler	1 XMTR, 1 RCVR, Power 100 Watts, Loss < 1 dB
5.	S-Band XMTR	10 Watts out, input - up to 3 Mb/s, QPSK Mod
6.	S-Band PLL RCVR.	$BW_{(RF)}$ = 20 MHz, NF = +5 to + 7 dB
7.	S-Band Multicoupler	1 XMTR, 1 RCVR, Power 30 Watts, Loss < 1 dB
8.	Delay Lock Loop Correlator	Cannot define without more study
9.	VHF Transmitter	5 Watts out, 10-20 Kb/s Input, Bi-Phase Mod
10.	VHF RCVR & Demod.	BW = 50 KHz, NF = +2 dB
11.	Antenna Servo Electronics	Cannot define without more study
12.	Parabolic Antenna	Diameter 6 ft, gain at S-Band + 30 dB, Gain at K-Band + 45 dB, Dual S&K-Band Feeds, Rotary Joints (waveguide & coax) Total line loss < 2 dB at both bands
13.	VHF Antenna	Turnstile or equiv, gain + 10 dB Total line loss < 1 dB

Fig. 25 Component Specifications

Component	Weight (lbs)	Power (watts)	ROM Cost x 10^3 Non-Re-curring	Recur-ring	Size
1. K-Band TWTA (50W out)	12	250	500	50	12"x4"x4"
2. K-Band QPSK Mod/Driver	3	10	100	20	3"x3"x2"
3. K-Band PLL Receiver (Acq. & Track)	3	5	100	15	3"x3"x2"
4. K-Band Multicoupler	3	-	20	5	4"x4"x2"
5. S-Band Transmitter (10W out)	6	80	100	20	6"x6"x6"
6. S-Band Receiver	3	5	50	10	3"x3"x2"
7. S-Band Multicoupler	3	-	-	5	2"x2"x2"
8. Delay Lock Loop Correlator	2	4	20	5	3"x3"x3"
9. VHF Transmitter (5W)	6	40	10	5	4"x4"x4"
10. VHF Receiver & Demod.	3	5	10	5	3"x3"x2"
11. Antenna Servo Electronics	8	20	30	10	8"x6"x6"
12. Antenna Servo Motors & Gears	15	20	-	-	(Part of Antenna Assy)
13. Antenna Feed (S&K Band)	4	-	-	-	(Part of Antenna)
14. Antenna (Dish - 6 ft)	10	-	1000	100	75" dia. x 30"
15. Antenna Mount (6' dish)	10	-	-	-	(Part of Antenna)
16. Antenna (VHF)	5	-	20	5	30"x10"x10"
17. Rotary Joint (K-Band)	3	-	50	5	Part of Antenna
18. Rotary Joint (S-Band)	2	-	10	5	Part of Antenna
19. VHF Multicoupler	1	-	-	1	2"x2"x2"
20. S-Band QPSK Mod/Driver	3	10	100	20	3"x3"x2"

Fig. 26 Communication Section Component Description

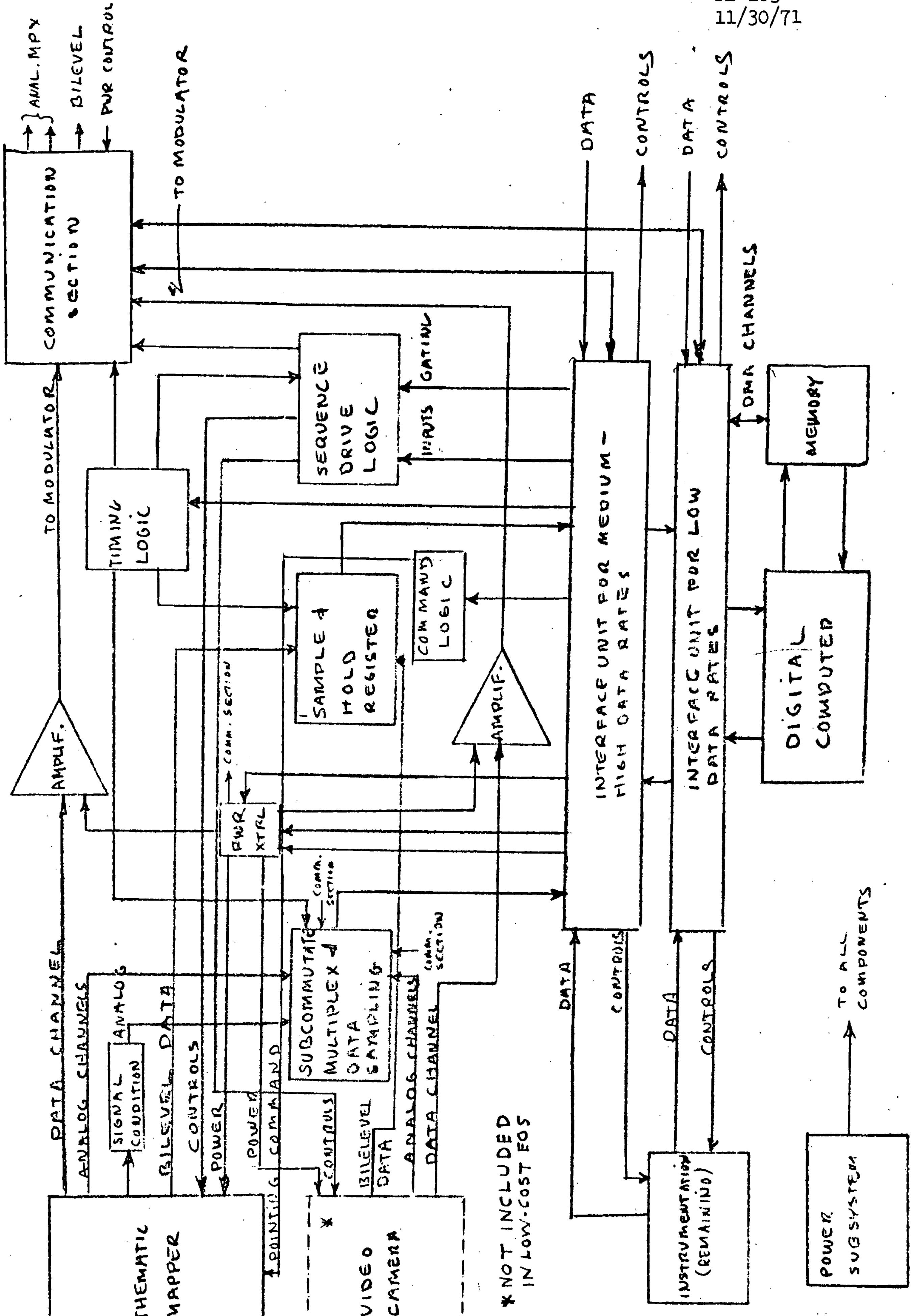

Fig. 27 Standard EOS CDPI Ultra-High Data Rate Interface Unit

Amplifiers -	(1) Gain $<$ 10 dB, bandwidth 100 Mc (2) Gain $<$ 10 dB, bandwidth 20 Mc
Analog Subcommutator - Sample & Multiplex	64 gates and one amplifier
Bilateral Sampler - Hold Register	2 16 bit registers
Sequence & Drive - logic	5 bit register - 32 "and" gates
Power Control -	5 gates
Timer -	Frequency divider. Assume 12 bit register and 4 gates
Logic -	TTL MSI
Packaging -	38 MSI circuits/board + mother board

Fig. 28 Ultra-High Data Rate Interface Unit Components

Logic Boards	Ckts/Board	Total Wt.	Total Power	Total Size
2	38 MSI + Amplifier	3 lbs	6W	< 0.1 ft^3

The cost for the unit would be approximately as follows:

Item	Base Cost	End Cost
Artwork fabrication	$1400/board	$ 2800
Components (TTL + FET)	300	600
Wiring and Packaging	2000	2000
Total Estimated cost		$ 5400

4.3 Medium-High Data Rates Interface Unit

A block diagram of the second unit in the Interface Section is shown in Fig. 29. The purpose of this unit is four-fold:

- provide an interface with medium-high data rate mission equipment and the S-band portion of the Communication Section
- supply multiplexing for medium-high data rates
- perform subcommutative multiplexing, control sequencing, etc., on mission equipment instrumentation and operation
- act as an interface to both the ultra-high and low data rates Interface Units

Except for the number of instrumentation and control signals and the employment of broadband amplifiers, the primary difference between the ultra-high and medium-high Interface Units is the inclusion of high data rate multiplexing.

The multiplexer combines the data channel outputs from the three mission equipment sensor units. In order to remain within S-band data rate handling capabilities premultiplex buffering is incorporated in the sensor packages. Signal conditioning is contained in each sensor data channel within the Interface Unit to enable matching sensor output to multiplexer input requirements and to provide a means for adapting the circuitry to a change in the mission equipment.

The high data rate multiplexer employs two shift registers. The first is loaded by the sequential samples from the three data channels. This results in an alternating bit pattern in the register. When the register is filled, the assembled data are transferred to a second register in parallel. Steering and gating logic is included in the transfer paths so that the data are properly organized for subsequent output. A block diagram of the high data rate multiplexer circuitry is shown in Fig. 30.

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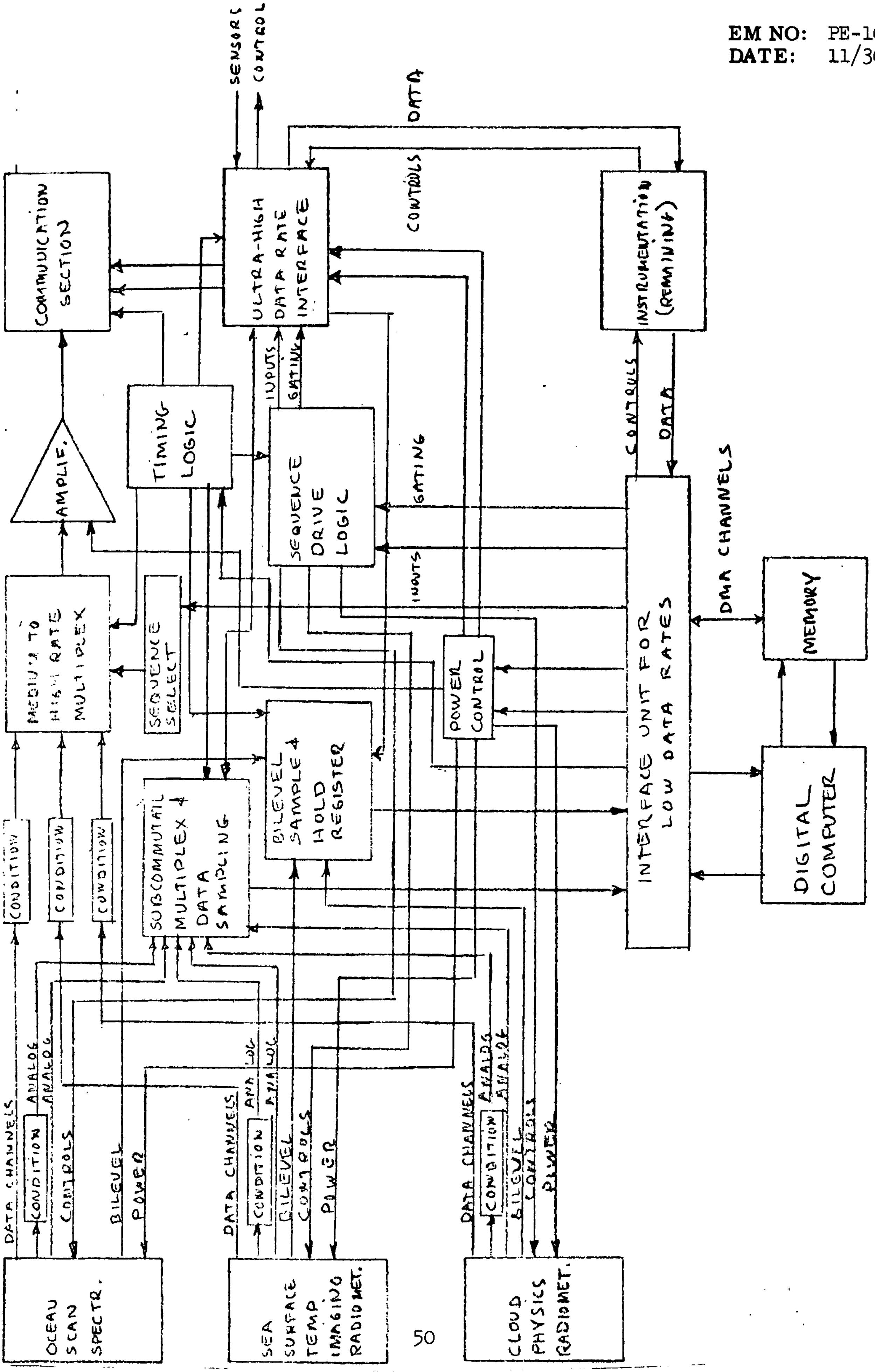

Fig. 29 Standard EOS CDPI Medium-High Data Rate Interface Unit

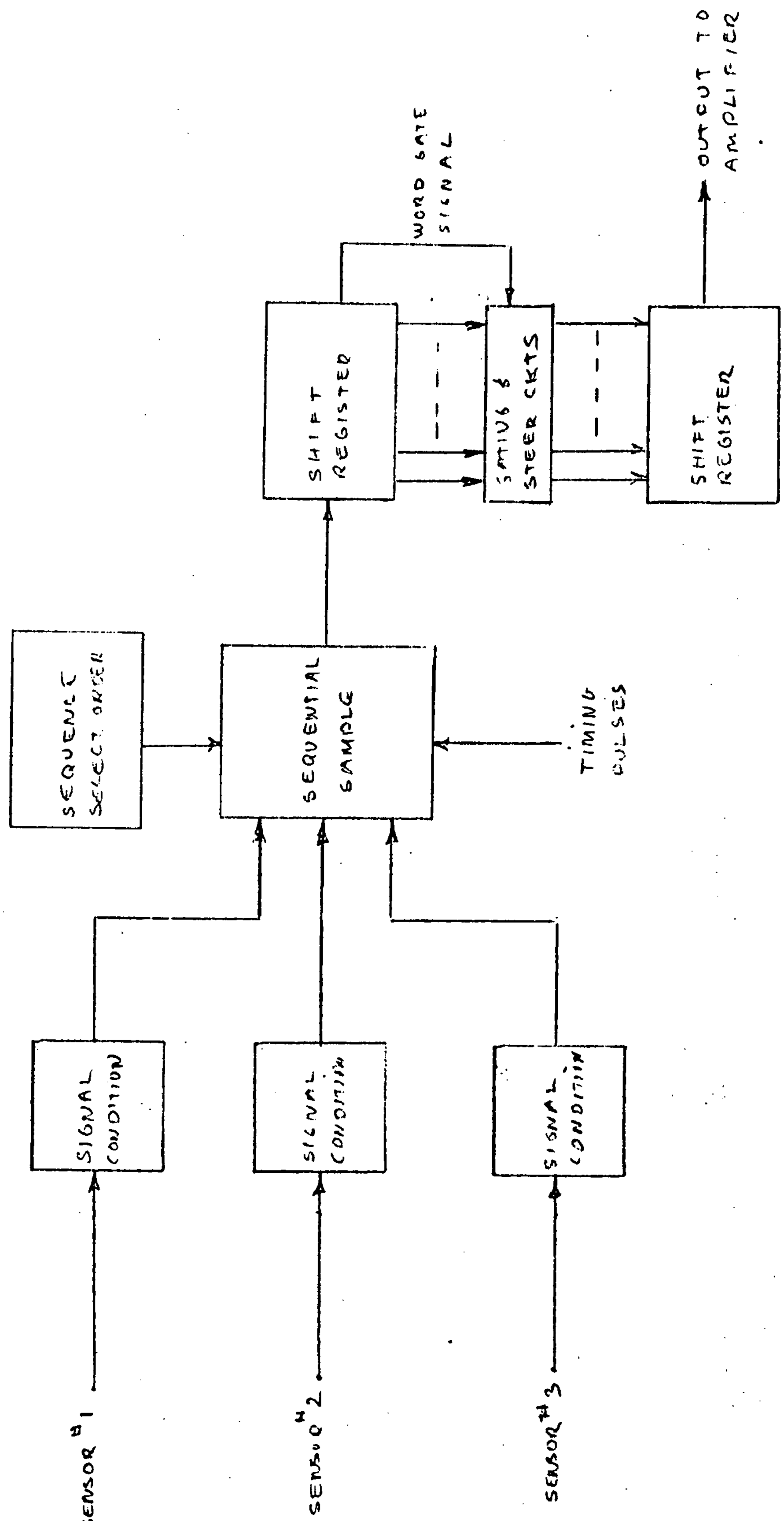

Fig. 30 Standard EOS CDPI High Data Rate Multiplexer

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The component complement of the Medium-High Data Rate Interface Unit is listed in Fig. 31. Assuming the same type of electronics and packaging as the Ultra-High Data Rate Interface Unit, the following estimates are obtained.

Logic Boards	Ckts/Board	Total Wt.	Total Power	Total Size
3	38 MSI + Amplifier	4 lbs	7 W	0.15 ft^3

The cost for the unit would be approximately as follows:

Item	Base Cost	Final Cost
Artwork Fabrication	\$ 1,300/board	\$ 3,900
Components (TTL & FET)	250	750
Wiring & Packaging	3,000	3,000
Total Estimated Cost		\$ 7,650

4.4 Low Data Rate Interface Unit

A block diagram of the Low Data Rate Interface Unit, the final component of the Interface Section, is shown in Fig. 32. This unit is divided into three parts. The first is devoted to S&C interfacing functions, e.g., counters, amplifiers, gates, etc.; the second serves as the focus for instrumentation and mission equipment functions, e.g., main analog multiplexing and A/D conversion, timing reference, etc. The third part is the digital computer external input/output interface containing input/output registers, data channel controls and logic. The functions of the S&C components are described in Reference 5 and the instrumentation and mission equipment functions are similar to those supported by other Interface Units, except data rates are much lower. The primary point of departure is the part containing computer input/output interfacing (asterisked Modes in Fig. 32).

A mechanization scheme for realizing computer interfacing is shown on Fig. 33. In the illustrated process, a common data channel is used for all devices attached to the channel line. These devices are the counters, mission equipment output data registers, A/D converter output holding register, etc. The output of each device is assembled with an identifier code and a parity indicator to form a computer input word. Delivery of the word to the computer may be accomplished in three ways - under direct computer addressing control, via cyclic sampling under strobe control, via cyclic sampling under strobe control and through the external clock.

Computer exercise control via the stored program. The data from a particular device may be obtained by sending the device identifier code down the select line. The code is interpreted by the various device decoders. The "true" code output gates the selected word. At the same time a blocking signal is sent to the other gates in the chain to prevent gating of other devices, if operated by an alternate mode controller.

High Data Rate Multiplexer -	Serial bit stream. Maximum input per channel 500 K bits/sec. Maximum output 1.2 M bits/sec. Order of sequence under control of sequence select logic registers 3 SS conditioning amplifiers - 18 gating + steering circuits - 2 18 bit registers 1 solid state power amplifier
Analog Subcommutator - Samples & Multiplex	28 gates and 1 amplifier
Bilevel Sample + - Hold register	1 8 bit register
Sequence Driver Logic -	6 bit register 48 "and" gates
Power Control -	6 gates
Timer -	Frequency divider. Assume 12 bit register and 5 gates
Logic -	TTL
Packaging	38 bit MSI circuits/board + amplifier + mother board

Fig. 31 Medium-High Data Rate Interface Unit Components

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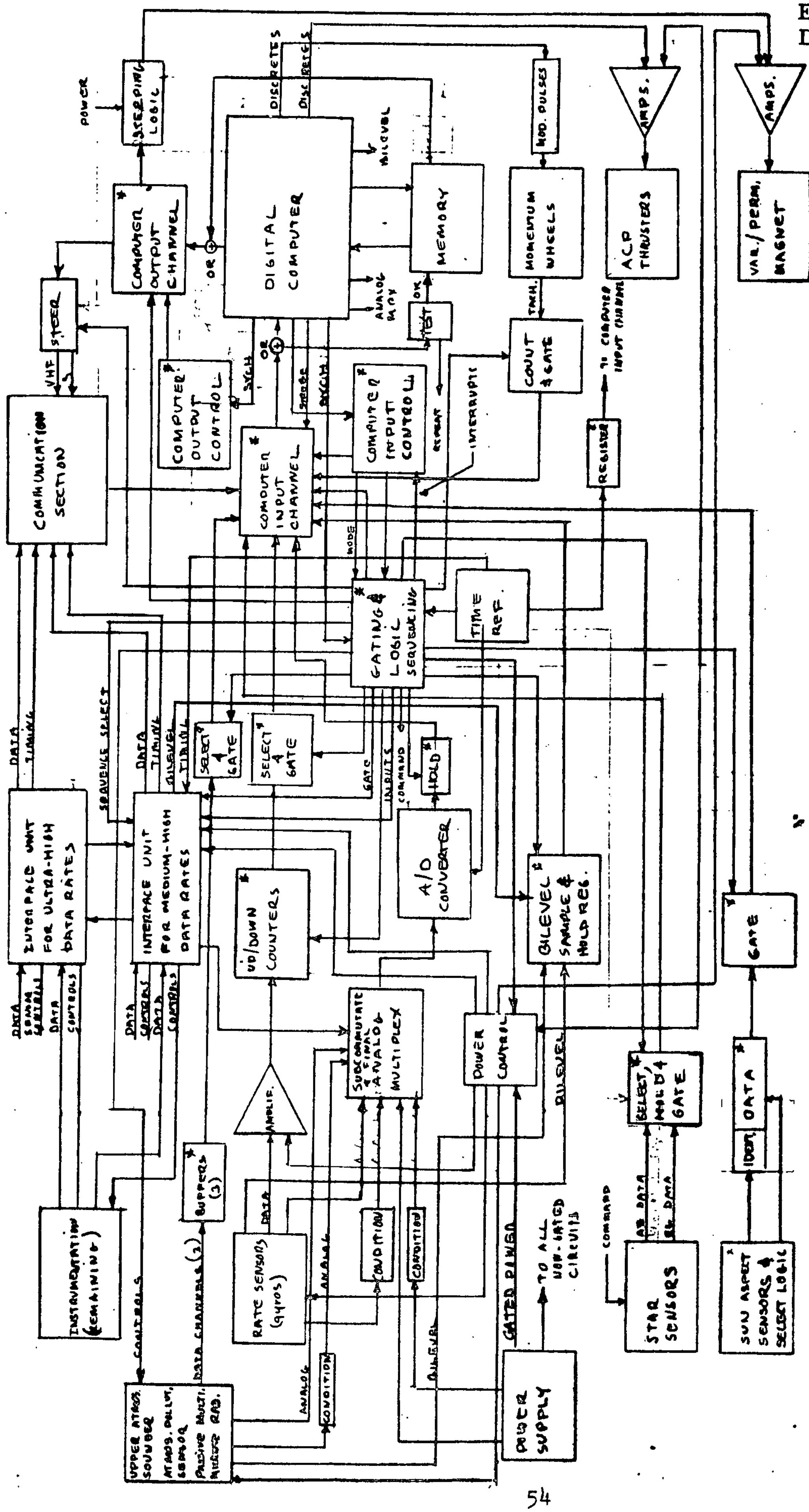
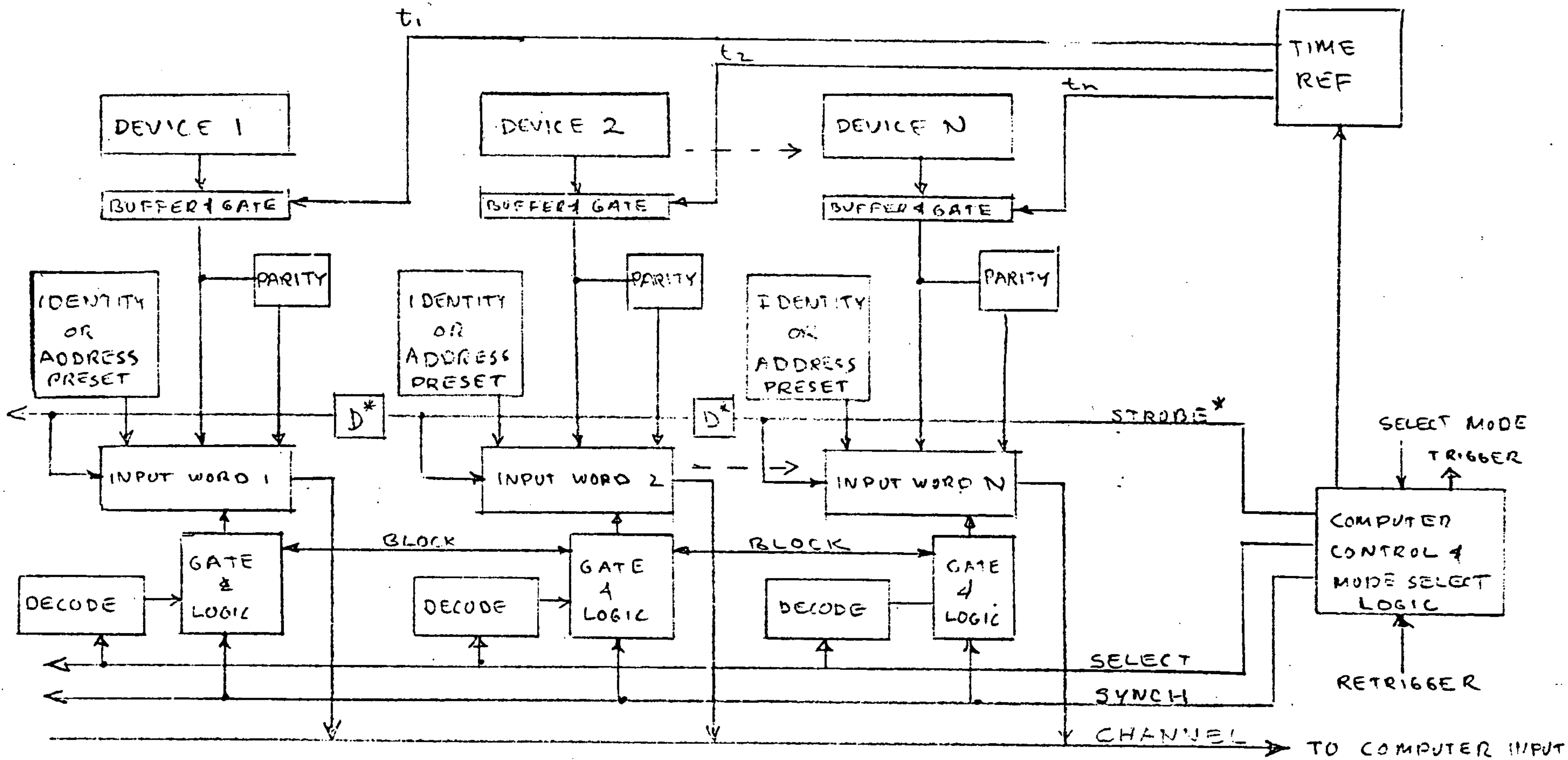


Fig. 32 Standard EOS CDPI Low Data Rate Interface Unit

* CHANNEL ADDRESSING PER FIGURE 3F



MODES

1. OUTPUT TO CHANNEL UNDER COMPUTER CONTROL
2. OUTPUT TO CHANNEL UNDER EXTERNAL CLOCK CONTROL
3. CYCLIC OUTPUT UNDER STROBE (OPTIONAL*)

LEGEND

t_n – TIMER INPUTS
D – DELAY

Fig. 33 Standard EOS CDPI Channel Addressing Implementation

Cyclic sampling of device contents may be realized by pulses transmitted down the strobing line. Delay lines are interjected in the line, with varying lengths, to account for the differing cycle times of the various devices.

The third mode, output under the external clock, utilizes the timing source dividing registers for alternate sequencing output from the various devices in the channel. Activation of the timing control, as with the other modes, is under computer mode control select logic. This provides considerable flexibility in device operation and data processing.

The main components of the Low Data Rate Interface Unit are listed in Fig. 34. It is to be noted that the list excludes buffer, gating and other functions associated with the mission equipment and S&C units. It is assumed that these components will contain the proper interface circuitry for operation with the data channel.

In order to provide a better interface structure with improved flexibility, each part of the Low Data Rate Interface Unit will be mounted on separate logic boards. This means that three types of circuitry boards will be included as identified on Fig. 35. This approach increases the number of logic boards, with low packing density per board. Assuming the same type of structure as the other Interface Units, the following estimates are obtained:

Logic Boards	Ckts/Board	Total Wt.	Total Power	Total Size
5	38 MSI + Time Ref. + A/D Convert.	6 lbs	10 W	0.25 ft^3

The cost for the unit would be approximately as follows:

Item	Base Cost	Final Cost
Artwork Fabrication	$1,500/board	$ 7,500
Components (TTL, Xtal, etc.)	350	1,650
Wiring & Packaging	4,000	4,000
	Total Estimated Cost	$ 13,150

It should be noted these estimates include the circuitry for input to direct memory access (DMA). DMA and data channel interfacing will be discussed in the following paragraphs.

Board Type		
1	Analog Sampler + Multiplexer Subcommutator Section	60 gates + 1 amplifier
1	Final Analog Multiplexer	6 gates + 1 amplifier
2	Up/down counters (3) (assure < 10K pps at sample rate of 50 samples/sec	3 8 bit registers + logic
1	Bilevel sample + hold register	2 16 bit registers
1	Power control	6 gates
1	Timer Xtal reference 15 ppm accuracy + divider register	Crystal oscillator and frequency divided. Assume 16 bit register and 20 gates
1	Sequence Driven Logic	6 bit register 38 "and" gates
3	Buffer registers for data channel - 12 required & 4 for growth	16 16 bit registers 128 "and" gates for divide and logic control
2	Buffer amplifiers	3 solid state
	Logic	TTL

Fig. 34 Low Data Rate Interface Unit Components

4.5 Data Processing Section

A block diagram of the Data Processing Section and its interconnections with other CDPI sections and Interface Units is shown in Fig. 35. This section serves the computational, on-board decision making, sequencing control, command verification, storage and formatting processes identified on the Functional Diagram of Fig. 11.

4.5.1 Computer Specifications

The heart of the Data Processing Section is a digital computer containing appropriate memory, arithmetic, storage input/output and control electronics. The capabilities required in this computer are approximately as follows:

4.5.1.1 Word Length. Accuracy requirements on most computations are usually much less than one part in 10^4. Some computations, e.g., attitude - one part in 10^6 and position-one, part in 10^5, require high accuracy; however, double precision may be used in these instances. This suggests a word length of 16 bits would suffice.

4.5.1.2 Memory Capacity. The memory capacity required is based upon the following word count estimates. Double precision computations are included:

Function	16-bit Word Totals
Stability & Control	2500
Sequencing & Command Processing	2500
Telemetry Formatting	350
Ephemeris (short arc)	384
Commands	512
Star Data	320
Received Data & Command Tests	350
I/O function routines, etc.	700
AGE routines	250
Attitude Control	150
	8016 words

This estimate indicates an 8K 16 bit memory would be adequate; however, a 12K memory would be specified to allow for star tracker calibration process storage (2000 words) and other growth items.

4.5.1.3 Memory Type. A random access memory is required, with non-destructive readout (NDRO) desired. The use of read-only memory sections may be used for the more critical subroutines; however selective write circuit lockout under program or command control would also be effective.

In order to meet flexibility requirements and effectively mechanize the computer input method described in Para. 4.4, Fig. 33, a direct access memory (DMA) is required. DMA allows for entering or extracting data from memory without interrupting the computer processes. The DMA may be obtained via cycle stealing, or separate addressing control may be used.

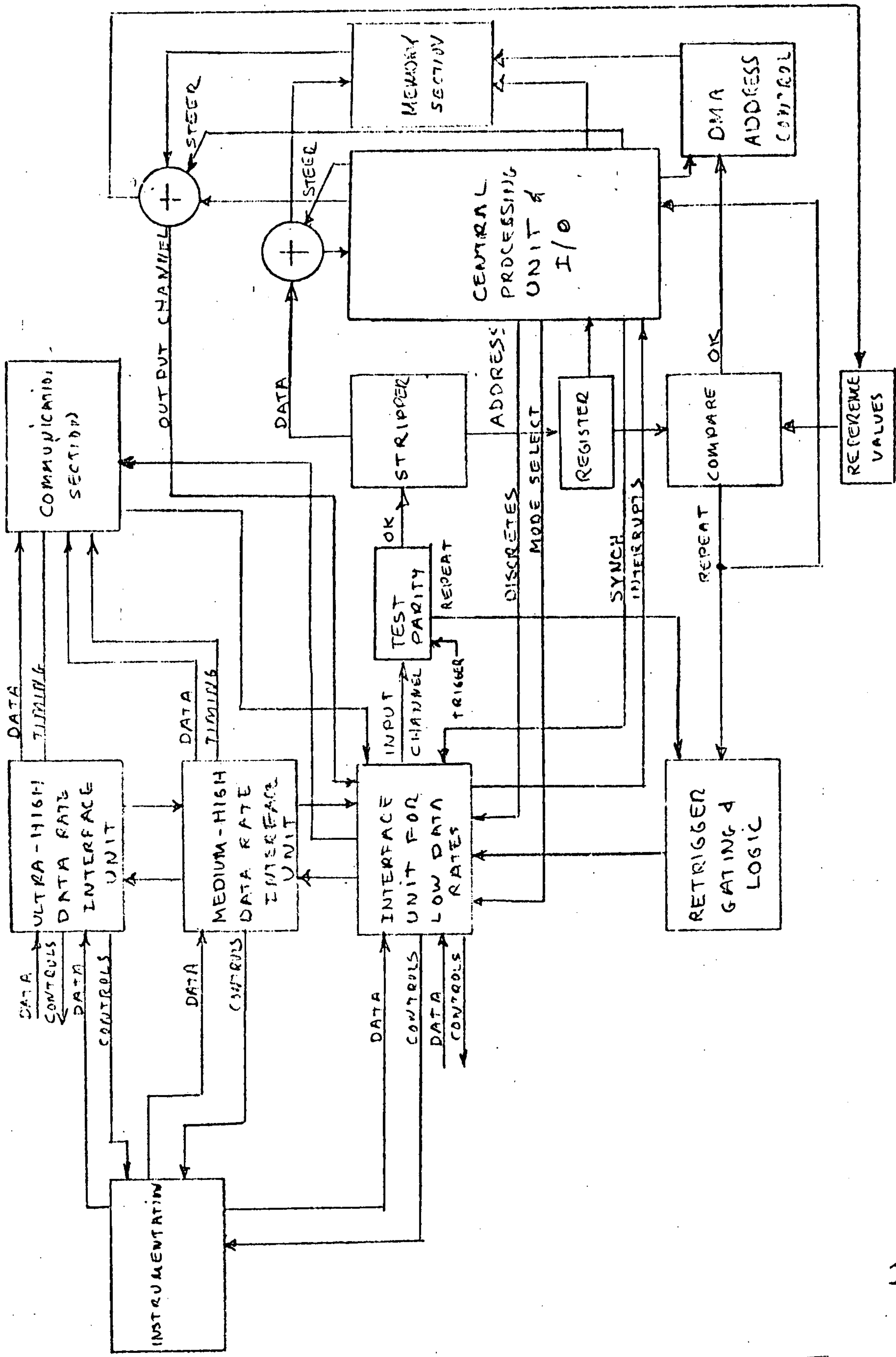


Fig. 35 Standard EOS CDPI Data Processing Section

4.5.1.4 Operating Speed. Speed requirements are not definitive at this time; however, a computer with an add time of 10 microseconds and a multiply time of 25 microsecond should suffice. Of course higher speeds would be desirable.

4.5.1.5 Instruction Repertoire. A standard mix of input/output, arithmetic, memory addressing, logical data manipulation and indexing would be sufficient. An instruction count in excess of 30 instructions is desired. Double precision instructions would also be of assistance.

4.5.1.6 Index Registers. A minimum of one index register is required; however, multiple index registers would provide desired flexibility.

4.5.1.7 Arithmetic Type. There are no specific performance requirements on arithmetic. The standard 2's complement, fixed point, parallel computer would be adequate. A double length accumulator is essential, however.

4.5.1.8 Input/Output. Input and output should be through two mechanizations - computer control and DMA. Buffer registers are an option since they may be supplied via the external Interface Unit. Both serial and parallel channels are desired. Operation of the input/output data channel processes are detailed in Para. 4.5.2.

The computer must handle at least three interrupts-input or register, timing cycle, and power fail. Other interrupt handling capability is desired.

Issuance of fixed and variable discretes under address control are desired. A two address for on and off respectively (variable discrete) or a single address (fixed discrete) control is acceptable.

4.5.1.9 Weight, Size, Power. Although specific weight, size and power requirements are to be determined, a fourth generation technology is assumed. This would result in a computer with the following characteristics:

Weight	< 18 lbs
Size	< 0.35 ft^3
Power	< 70 W

4.5.2 Data Channel Interface

DMA has a disadvantage in that noise or erroneous data could alter memory contents and may destroy crucial routines. In order to prevent this possibility special circuitry are added to the data channel, as shown in Fig. 35. This electronics has been accounted for and is physically included in the Low Data Rate Interface Unit; however, it is shown in Fig. 35 as a separate section for the sake of clarity.

As seen in Fig. 33, data words arising from data channel devices contain three parts - identifier or address, parity and the data itself; they are transmitted in serial mode. A trigger signal derived from mode control activates parity test hardware on the word being transmitted. If the received word is found correct, it is advanced to the stripper and steering circuit; if it fails, retriggering logic is activated and the word output repeated.

Two outputs are obtained from the stripping circuit - the data portion of the word and the word identifier or address. The identifier is compared with reference values, obtained from the computer by stored program control, in comparison logic. A number of tests may be performed, e.g., testing whether the address is legal or within a specified area of memory. If the address is acceptable, DMA address control is acquired and the data portion of the word entered into the specified location. An error in comparison would result in operating the retrigger logic.

It should be noted that data may also enter or leave the computer under direct program control. This alternate is useful in telemetry or command input/output processing.

4.5.3 Computer Example

The computer specifications delineated in the previous paragraphs may be satisfied by a number of computers undergoing development at the present time. One machine in particular is completely designed and is being demonstrated at the present time; it is extremely well suited to the standard spacecraft CDPI. This is the CDC 469. The technical characteristics of this computer are listed in Fig. 36. It is seen that this computer equals or exceeds specifications in most respects. The one exception is availability of DMA in the demonstration model. Fortunately, however, DMA may be added, with up to 16 devices directly on the data channel, with a nominal wiring change in this computer. Extra circuitry will be required but this is accomplished external to the computer itself.

Prices on the computer, as published by CDC, are given in Fig. 37. These prices do not include DMA modification; however, the added increment would be nominal.

4.5.4 Software

Software mechanization for the CDPI has not been determined as yet; however, some estimates may be obtained based upon past experiences. Assuming an 8K memory load then

basic software cost	\$ 170,000
validation expense (40% basic cost)	68,000
	\$ 238,000

This cost does not include documentation, instruction manuals, training and other costs arising from system integration, management and business operations.

TECHNICAL CHARACTERISTICS

General Data

Weight: 2.5 lbs (8K)
4.0 lbs (16K)

Dimensions: 4.2"h x 4.13"w x 3.0"d (8K)
4.2"h x 4.13"w x 5.0"d (16K)

Power Consumption: $\approx$12 watts (8K)
$\approx$13 watts (16K)

Input Voltages: $\pm$ 15 VDC, + 5 VDC, -3VDC

Circuits: High level PMOS/CMOS/IC devices

Environment: Designed to MIL-E-5400

Cooling: None required (0^oC to $+50^oC$)

Estimated Reliability:

Central Processor (LSI, 14 devices total)

Type: Binary, parallel, general purpose single address, plus file address

Repertoire: 42 instructions (some double precision)

Word Length: 16 bits

Register Files: 16 Addressable 16-bit word files

Interrupts: 3 external levels, plus 1 direct execute

Arithmetic: Fractional, fixed point, two's complement. Hardware multiply and divide.
Typical execution times:
Add: 2.4 μsec
Double Precision Add: 3.6 μsec
Multiply: 10.4 μsec
Divide: 30.4 μsec

Memory

Type: Random access, word-organized, NDRO (electrically alterable) Plated Wire Memory (PWM)

Word Length: 16 bits

Capacity: 8K words NDRO expandable to 65K in 8K word increments

Read Cycle Time: 1.6 microseconds

Write Cycle Time: 2.4 microseconds

Access Time: 500 nanoseconds

Input/Output

1-16 bit parallel, party line, buss input
1-16 bit parallel, party line, buss output
4-bit address control lines
1-serial input channel
1-serial output channel
External clock input
69-90 KHz - parallel continuous word rate I/O
400 KHz - parallel burst word rate I/O
130 KHz - serial bit rate I/O

Optional I/O: Solid state keyboard and display suitable for navigational, checkout, and general purpose use can be integral to the computer. Also available are multiplexed A/D input channel(s), buffered and non-buffered peripheral device channel(s) on a Quote Special Equipment (QSE) basis.

Standard Interface: TTL

Software

Assembler, Simulator, Plus Library and Diagnostic Routines

Optional Support Equipment

Programmer's Console with integral CRT display and power supply.

P.W. Memory Loader with integral paper tape reader and power supply.

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* Information furnished by Control Data Corp., Minneapolis, Minn.

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Fig. 36 CDC-469 Example Computer*

Standard 469 Computer & Support Equipment

Prices - Schedule A

Computer Model No. CDC-16-4/8/16-T-1-B

Qty.	469 & 4K	469 & 8K	469 & 16K	Qty.	Programmers Console	PWM Loader
1-2	32,560	37,950	65,000	1	9,800	11,700
3-5	29,800	34,900	58,500	2	8,800	10,500
6-10	28,250	32,800	48,100	3	8,350 ea	9,850 ea
11-25	26,800 ea	31,100 ea	42,200 ea	4	8,000 ea	9,450 ea
26-50	25,000 ea	29,000 ea	38,000 ea	5	7,750 ea	9,100 ea

These prices are to be used for quotation as standard hardware (no 633's).

*Information furnished by Control Data Corp., Minneapolis, Minn.

Fig. 37 Price List*

5.0 Packaging Approach

In order to adapt the CDPI designs to meet standard spacecraft requirements, a modular packaging structure will be used. Allocation of the various CDPI sections to the modules are subidivided in accordance with core vs non-core functions, i.e., whether the module is part of the core unit structure or external to the core unit.

Considering the mission equipment as it relates to the CDPI, a natural division exists between the Ultra-high Data Rate components and the remainder of the spacecraft. This includes the Thematic Mapper, Ultra-High Data Rate Interface Unit, and the K-band communication equipment. The only points of commonality between these components and the remainder of the spacecraft are the antenna and interfacing functions. This division suggests a separate CDPI module for the Thematic Mapper/K-band functions.

The remainder of the CDPI is more interrelated and probably required for all missions. This suggests incorporation into one or two modules which serve as the CDPI core unit. An appropriate division is the medium-high data rate and low rate functions; however, some interchange may be desirable to present a more balanced heat and mass load. For example, the communication units may be included in one module and the remaining units in the other module. In a similar fashion, the computer may be included in the module containing the communication equipment to reduce lead lengths from the computer to the telemetry section; however, closer association with the Interface Unit seems more desirable. The subdivision selected here will be based upon the data rate differences.

6.0 Conclusions & Recommendations

A point design of a spacecraft CDPI which employs the low cost concepts of standardization and integrated design has been developed; based upon EOS low earth orbit missions. It was found in proceeding through the design that a number of commonalities exist between the various sections making up the CDPI. The most pronounced are in the areas of interfacing, communication support and data processing. These commonalities may be exploited by providing standardized equipment or software to meet requirements. Typically, data rate demarcates the possibility of direct input to a computer vs multiplexing vs no multiplexing at all. Instrumentation for status and housekeeping functions also have a great deal of commonality in data handling and routing processes and standard components may be provided to meet such needs.

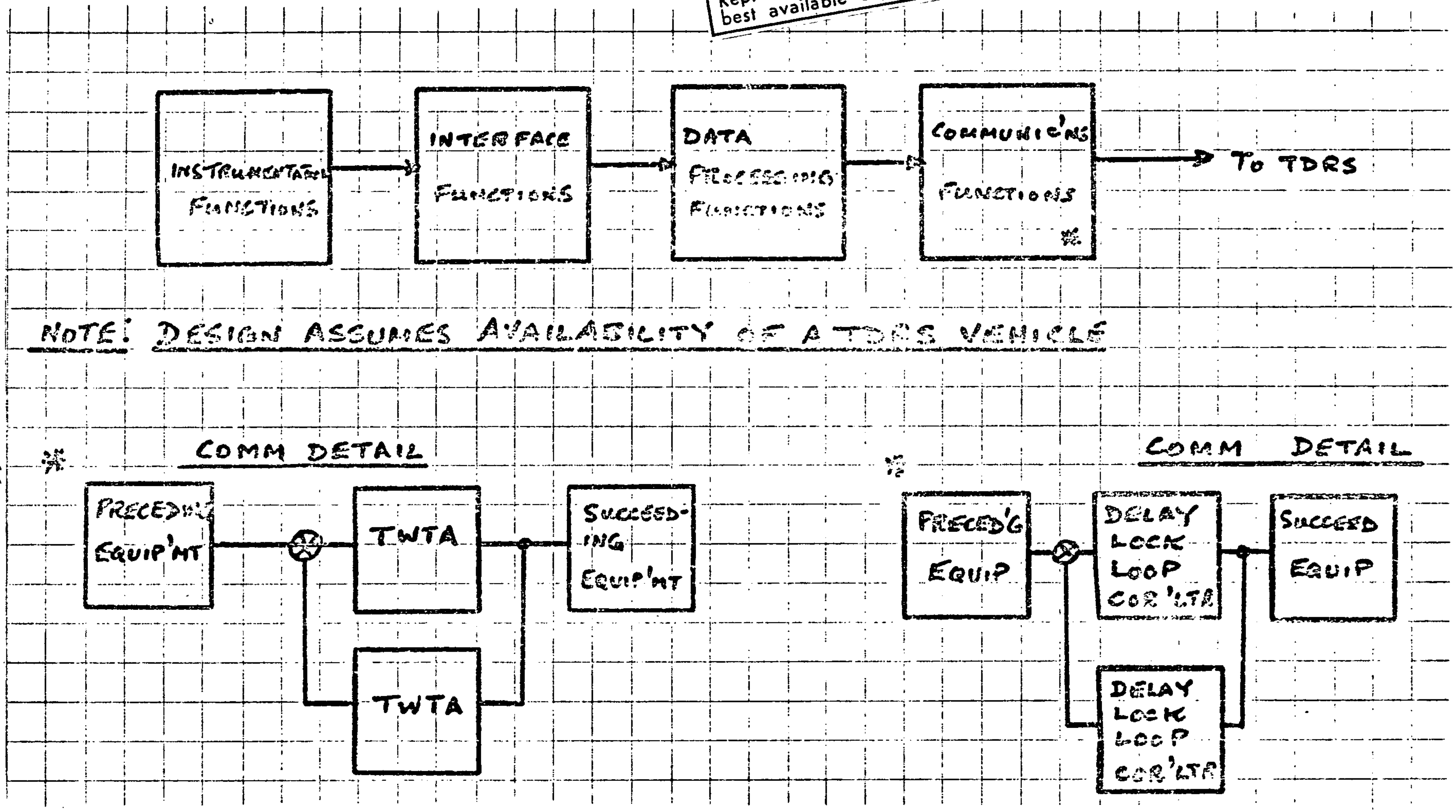
It was found that customizing the design is most likely to be required as state-of-the-art limitations imposed by data rate handling requirements are approached; however, lower data rate equipment is more readily available and also more amenable to standardization. The high degree of similarity between the low and medium-high data rate Interface Units demonstrates this possibility.

The strong possibility of standardizing CDPI interfacing should be investigated in more depth, since it is primarily in interfacing that standardization most often yields maximum benefits. Other commonalities than data rates may exist, e.g., signal waveform, voltage levels and other characteristics. Software interfaces should be considered as well as hardware electronics.

Other point designs need investigation and development to obtain more "point data". These include synchronous orbit and deep space missions, particularly those missions which will utilize a number of spacecraft.

Much of the design developed in this report is based upon Space Shuttle launch and revisit. Thus, the only alternate to operational or partial operational modes is a safe mode. This presumes lifetime requirements are such that expensive, high reliability components and component parts are not required. There is a tradeoff between Shuttle visit frequency and spacecraft costs which needs to be examined. This pertains in particular to K-band communication components, e.g., TWT amplifiers.

The reliability goal of the CDPI subsystem for one year of orbit life is 0.864. The reliability goal has been met by the proposed design as shown in Fig. 37.



* Both K and S Bands

Fig. 38a Standard EOS CDPI Subsystem Reliability Block Diagram
(Sheet 1 of 2)

- Allocated reliability 0.864 for 1 year.

- Approach: minimal equipment redundancy, functional backup, i.e., K-Band Hi Data Rate backed by S-Band in Communications. VHF Partial Back-up to both. Subsystem is <u>FAIL SAFE</u>, i.e., loss of any one function does not involve any other in the loss.

- Subsystem breakdown into 3 parts:

Interface Functions	$R = 0.974$
Computer Functions	$R = 0.960$
Communication Functions	$R = 0.930$

- Interface elements mostly passive, interconnect boards $\epsilon\lambda$ 2.95×10^{-6} approximately 2950 interconnects at 1 bit per cross connection.

- Computer functions 10 major logic boards + memory. Memory effective λ 2570 bits, logic $\epsilon\lambda$ 2000 bits $\epsilon\lambda$ 4.57×10^{-6} (1 bit = 1×10^{-9} λ)

- Communications critical areas (a) TWTA's in K&S Bands. Solution standby redundancy with Time Share. (b) Delay lock loop correlator. Solution standby redundancy with solid state switching. $\epsilon\lambda$ 7.97×10^{-6}

- Allocated $\Sigma\lambda = 16.55 \times 10^{-6}$ $\Sigma\epsilon\lambda$ with redundancy a, b -- $\Sigma\epsilon\lambda = 15.49$

- Subsystem will meet 1 year requirement.

Fig. 38 Standard EOS CDPI Subsystem Reliability Block Diagram (Sheet 2 of 2)

7.0 References

1. Earth Observatory Satellite (EOS) Definition Phase Report - Vol. 1 Goddard Space Flight Center; dated August 1971.

2. Tracking and Data Relay Satellite System; An Overview, P. F. Barritt, E. J. Habib. AIAA Paper No. 70-1305.

3. GSFC Mark 1 Tracking and Data-Relay Satellite (TDRS) System Concept - Phase A Study Final Report Vol. 1, Goddard Space Flight Center, dtd Nov. 1969.

4. Appendixes to GSFC Mark 1 Tracking and Data-Relay Satellite (TDRS) System Concept - Phase A Study Final Report Vol. 2. Goddard Space Flight Center, dated December 1969.

5. Standard EOS Stabilization and Control Subsystem; R. J. Pollak, EM PE-102, LMSC Report; dtd 30 Nov 1971.

PE-104

EOS

ELECTRICAL POWER SUBSYSTEM

PE-104

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: STANDARD EARTH OBSERVATORY SATELLITE: ELECTRICAL POWER SUBSYSTEM	EM NO: PE-104 REF: DATE: 30 November 1971
AUTHORS: G. A. Deppe	APPROVAL: ENGINEERING J. C. Bolton Wayne Muller

18 Pages

CONTENTS

PRELIMINARY

Lockheed Missiles & Space Company

EM NO: PE-104
DATE: 30 November 1971

1.0 Introduction

The electrical power subsystem described herein is a recommended preliminary design for incorporation into a standard shuttle-launched spacecraft to perform the Earth Observatory Satellite (EOS) mission. The electrical load is an average of 1000 watts.

2.0 Requirements and Guidelines

The EOS mission requirements are:

1. Altitude of 485 miles.

2. Orbit inclination of 93° (sun synchronous)

3. Launch time of noon ± 3 hours, ($\beta \pm 45^{\circ}$, and remaining in the launch β throughout the mission).

4. Life of 1 year; design goal of 2 years

5. 1000 watt average load

The main guideline is low overall mission cost. Since the satellite is to be launched by the Space Shuttle, lower costs can be realized due to the reduced satellite weight and volume constraints relative to those imposed by current booster launch systems. Other advantages offered by the Shuttle-launch concept are these: in-orbit checkout, in-orbit maintenance and, return of the spacecraft to ground for refurbishment, if warranted. 1971 state-of-the-art is to be used. Minimizing non-recurring costs (early expenses) to a greater extent than recurring costs (later expenses) is a goal.

Considering these guidelines, the following design approaches are appropriate:

1. Modularize the components into modules which

    a. Are small enough in size for in-orbit handling and replacement

    b. Have efficient interfaces with other modules (minimum interconnects)

    c. Are low enough in cost to be used as on-orbit replacement modules.

2. Subsystem modules to be:

    a. Solar Array Module (2 required)

    b. Battery and Battery Charge Control Module (3 required)

    c. Power Distribution Module

    d. Harness Module

3. Little checkout at launch pad; deployments and final in-orbit checkout via an umbilical to the checkout set in the Shuttle before final release.

4. Utilize non-graded solar cells with larger solar array area (cost saving on purchased cells).

5. Simplify folding elements of solar-array paddles (shuttle allows fixed arrays or simpler erection).

6. Utilize larger-area fixed solar array in lieu of self-orienting array with sun-seeking mechanism. (Shuttle provides added volume which allows use of fixed array.)

7. Minimize redundancy (less material and fabrication costs).

8. Use off-shelf or MIL-STD components in place of Hi-Rel components (cost savings on parts).

9. "Overdesign" to minimize acceptance and qualification testing (less testing labor).

10. Avoid miniaturization (less fabrication costs).

3.0 Description of the Electrical Power Subsystem (EPS)

The proposed EPS consists of a fixed (non-tracking) solar array, 6 nickel-cadmium batteries which are charge-controlled by array switching, and a DC-DC regulator. The system provides a nominal 28 volt bus which varies from 25.0 to 28.0 volts. The regulator will provide 28.0 VDC ± 2% for the equipment needing close regulation. A functional block diagram is shown in Fig. 1.

This system is very consistent with the guidelines because it can easily be adapted to the design approaches listed in para. 2.0. It has been used on three large Air Force programs at Lockheed with outstanding success. Its simplicity is largely responsible for its success.

Large battery capacity (more weight) with its attendant low depth-of-discharge, makes elaborate charge-rate controllers unnecessary (at a given temperature, battery life is inversely proportional to depth-of-discharge). Array open circuit switching conveniently dissipates excess energy as heat out on the large solar array (by not converting solar energy to electrical energy). This eliminates the large bank of heat dissipating resistors associated with shunt regulators. These resistors are traditionally difficult to locate because of the large quantity of heat to be dissipated. The batteries are on the bus full time, establish the array operating point, and serve the same function as a shunt regulator, thus making the system unregulated bus a direct energy transfer system. They also completely absorb the array switching transients and absorb the array cold-to-hot high voltage transients that are characteristic of array-on-bus and battery switching systems.

An array angle adjustment, before launch, compensates for β-angles other than 0 degrees.

Lockheed Missiles & Space Company

EM NO: PE-104
DATE: 30 November 1971

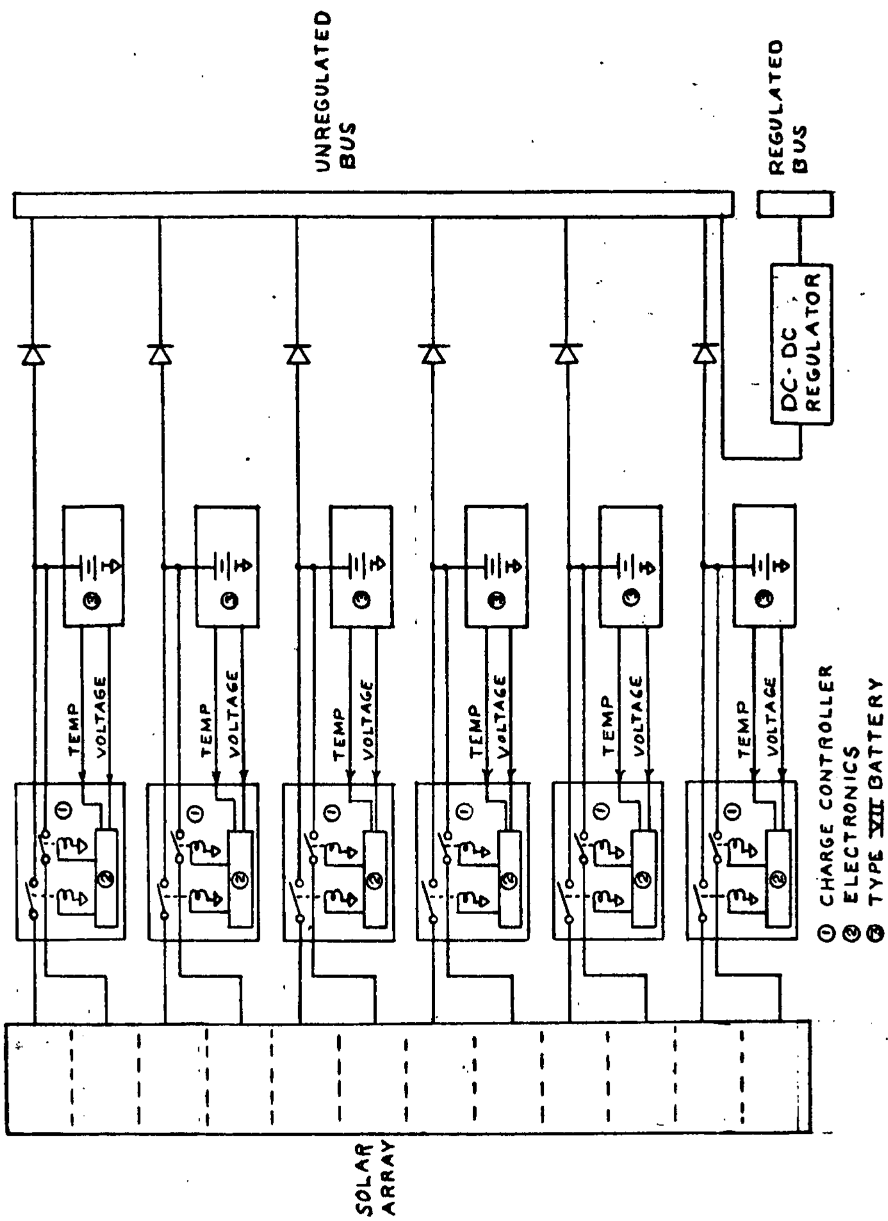


Fig. 1 Standard EOS Electrical Power Subsystem

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An equipment list, listing size and weight for each item is shown in Table 3-1.

Table 3-1

EQUIPMENT LIST

Component	Size	Wt. (ea.) (lbs)	Per Vehicle Qty.	Per Vehicle Wt. (lbs)
Solar Array Panel	31.8 x 21.6 in. ($4.76\ ft^2$)	7.8*	80	624
NiCd Battery	7 x 7 x 21 in.	70	6	420
Charge Controller	8 x 9 x 10 in.	6	6	36
Power Dist. Unit	19 x 16 x 5 in.	32	1	32
DC to DC Regulator	9 x 8 x 20 in.	37	1	37
Harness Assy		320	1	320

Refer to Section 6.0 for total EPS weight.

* The array panel weight is itemized as follows:

2.14 lbs (.45 lbs ft^2) for .012 in. cells, .020 in. coverglass, and printed circuit on .001 Kapton.
5.66 lbs aluminum substrate

7.8 lbs per panel

4.0 Equipment Description

Solar Array

The beginning-of-life solar array output necessary to support equipment loads of 1000 watts average over a 2 year period is calculated as follows:

1000 watts, using equipment
+ 100 watts, battery charging losses (1000/2 x 20%)
+ 55 watts, 5% radiation degradation in 2 years

1155 watts average needed from solar array

The solar array consists of 80 panels mounted on two fold-out wings as shown in Fig. 2. Each array wing is mounted on a shaft so that the wing can rotate $\pm 45^{\circ}$ with respect to the spacecraft body. This is to compensate for the $\pm 45^{\circ}$ β-angle. The angle will be set before launch. Since the β-angle does not change throughout the mission because the orbit is sun-synchronous, this one adjustment is adequate throughout the mission. The solar array is retracted when the spacecraft is returned to earth by the Shuttle.

A fixed array was selected instead of a tracking array because of lower overall costs, and much higher reliability was gained at the expense of increased weight and array area. Tradeoff calculations are presented in the Appendix.

A flexible array does not seem to offer a cost advantage at this time and in addition is not operational to date. Many of the flexible array manufacturing techniques that have been developed, such as etched circuit cell interconnects and induction soldering, will be utilized, however.

Each of the 80 panels is 31.8 inches by 21.6 inches and consists of an assembly of cells, coverglasses, and etched circuits on a Kapton base, which is bonded to a .080 inch thick aluminum sheet.

The cells are 2 by 4 CM, N on P, .012 inches thick, 2 ohm-CM base resistivity, with wrap-around contacts. 97.5% of the production cells (after mechanical screening) will be used instead of the usual 66.7% yield. This reduces cell costs by 30% with a 2% increase in the number of cells needed (2% weight increase). The coverglass is .020 inch fused silica for maximum radiation protection (less cells) and less handling breakage (higher yield). Solar arrays for conventionally boosted spacecraft systems typically use .012-.014 inch thick coverglass for the weight advantage.

The electrical arrangement of the cells on a panel will be 69 in series by 7 in parallel, making a total of 483 cells per panel. Table 2 shows the cell output, and calculation for the required total number of cells, array area, and the number of panels required.

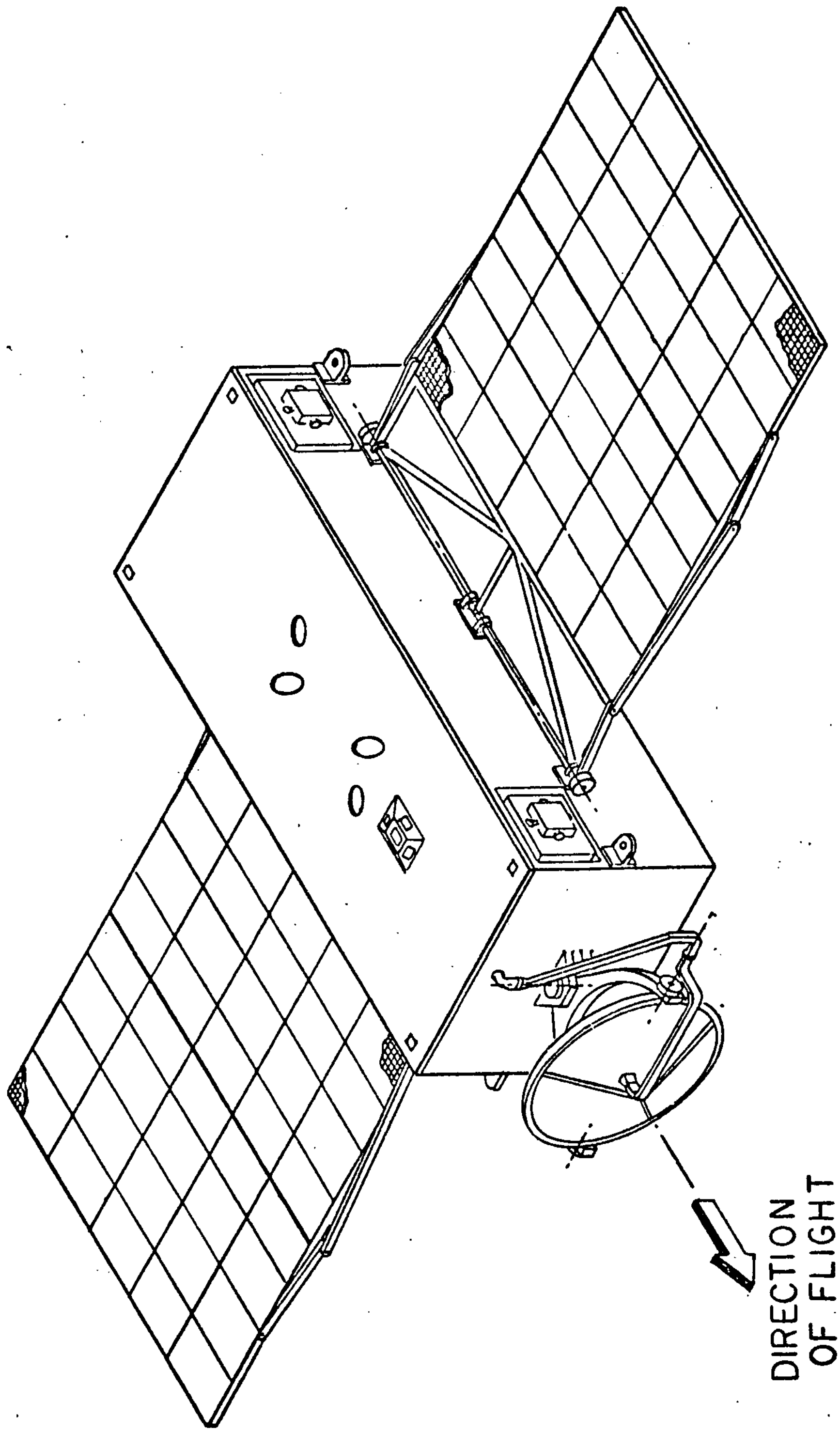

Fig. 2 Standard EOS Solar Array

Table 2

SOLAR CELL CHARACTERISTICS AND ARRAY SIZING

2 x 4 CM Cell Characteristics

Cell output, Normal to sun, Beginning of life:

Avg cell power, $28^{o}C$, AMO, 66.7 percent yield = 117.5 mW

Avg cell power, 97.5 percent yield = 117.5 x 0.98 = 115.1 mW

Avg cell power with 20 mil fused silica coverglass = 115.1 mW x 0.97 = 111.6 mW

Array Sizing

.1116 watts per cell at $28^{o}C$

.318 orbit effectivity for array at 0^{o} β (worst case)

.87 power decrease from $28^{o}C$ to $52^{o}C$. $52^{o}C$ is average array temperature (peak of 56^{o}) during the power producing period. (.55% decrease/^{o}C)

$$\text{number of cells} = \frac{\text{power needed from array}}{\text{cell output x array effectivity x temp degradation}}$$

$$\text{cells} = \frac{1155}{.1116 \times .318 \times .87} = 37{,}400 \text{ cells}$$

With 100 cells per ft^2 and 1 panel = 4.76 ft^2:

$$\text{array} = \frac{37{,}400}{100} = 374 \text{ ft}^2$$

$$\text{panels} = \frac{374}{4.76} = 78.6 \approx 80 \text{ panels (40 per wing)}$$

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Charge Controllers

Each battery has a charge controller which connects or disconnects a section of the array to the battery. The voltage "tail-up" as the battery nears full charge is used as the signal to cut off charge. This voltage level is modified by the battery temperature which is sensed by transducers in the battery and fed into the charge controller. The control is done in two stages. As the battery proceeds towards full charge, first one-half of the array section is turned off, and then if the battery voltage continues to rise another one-half volt, the other half of the array section is disconnected. In this way the array output can often balance the load with little or no battery cycling, except for the normal night-time cycling. Conversely as the battery voltage decreases, the array section is connected in two stages.

The control levels will be approximately as shown in Fig. 3, with K1 and K2 depicting the two levels of control. The approximately 3 volt dead band between the connect (charge-on) and disconnect (charge off) levels reflects the three volt difference between the battery charge voltage and its discharge voltage.

A high temperature charge cut-off (both sections) would probably be incorporated into the charge controller, to cut off the charge at about 90°F, regardless of the state of charge. This would interrupt the possible thermal run-away" mechanism of the battery in some abnormal situation. Additionally, a battery off-line ground command could be incorporated, which would cut a defective battery out of the circuit and connect the associated section of the array directly to the unregulated bus. This one section of the array would "load regulate" itself if its capacity did not exceed the load requirements.

Batteries

Nickel-cadmium batteries were chosen because of the long life requirements. The 40 ampere-hour capacity is necessary to keep a low depth-of-discharge (DOD) to maximize the battery life. An average 14% DOD has been calculated for a 40 amp-hr battery and the load. A NASA EOS report states 25% DOD to be a maximum, therefore 14% represents a good safety margin. The calculations for the DOD as well as the average charge rate per battery is shown in Table 3.

Power Distribution Unit

This unit distributes the power to the various using equipments and also contains fuses, current sensors, and power system telemetry conditioning networks as required.

DC-DC Regulator

A conventional DC-DC regulator will be selected when the regulated power needs are finalized.

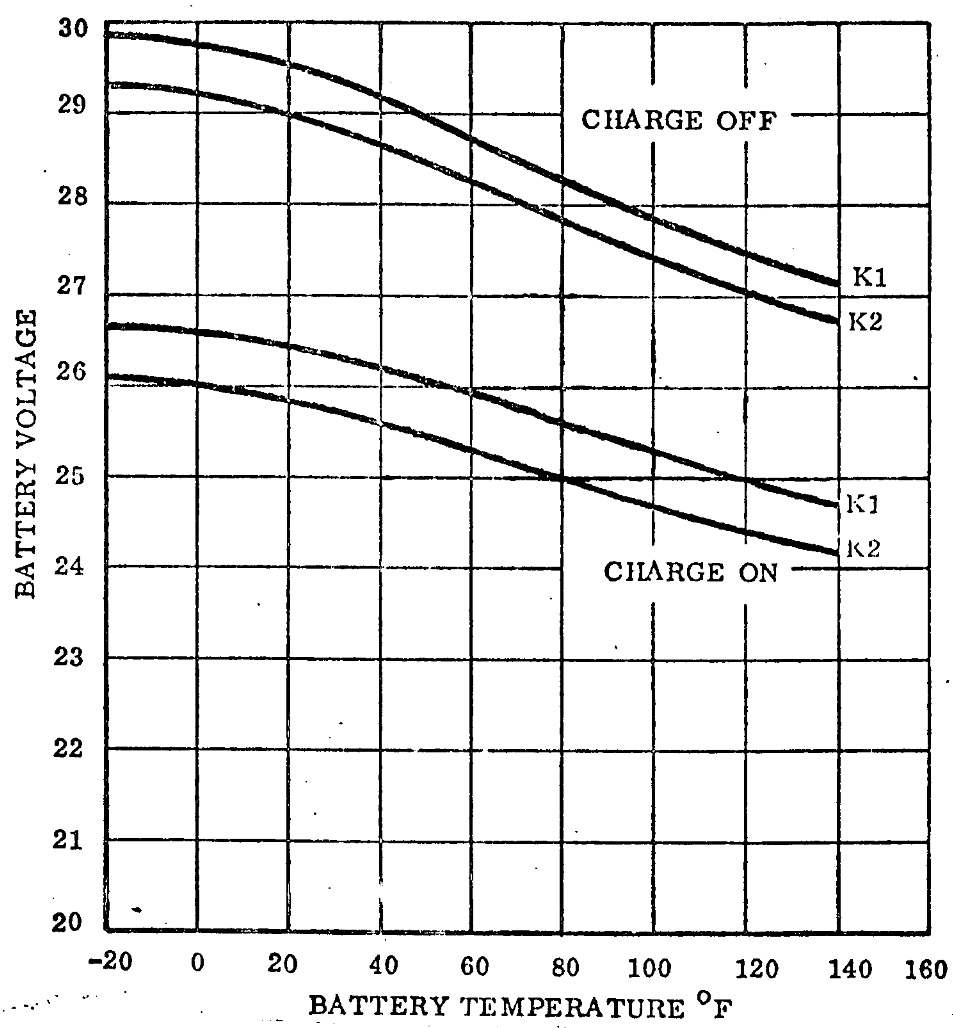


Fig. 3 Charge Controller Operating Characteristics

Table 3

BATTERY DEPTH OF DISCHARGE

$$\frac{\text{1000 watt load}}{\text{25 volts (discharge voltage)}} = 40 \text{ amps}$$

energy out during eclipse:

$$40 \text{ amps} \times \frac{\text{50 minute eclipse}}{60} = 33.3 \text{ amp-hrs}$$

total battery capacity = 40 A-hrs ea battery x 6 batteries = 240 amp-hrs.

Depth of discharge (average)

$$\frac{33.3}{240} = 14\% \text{ average}$$

Battery Charge Rate:

Charge put back in batteries at a 40 amp rate average

$$\frac{\text{240 amp hrs capacity}}{40} = 6$$

charge rate is C/6

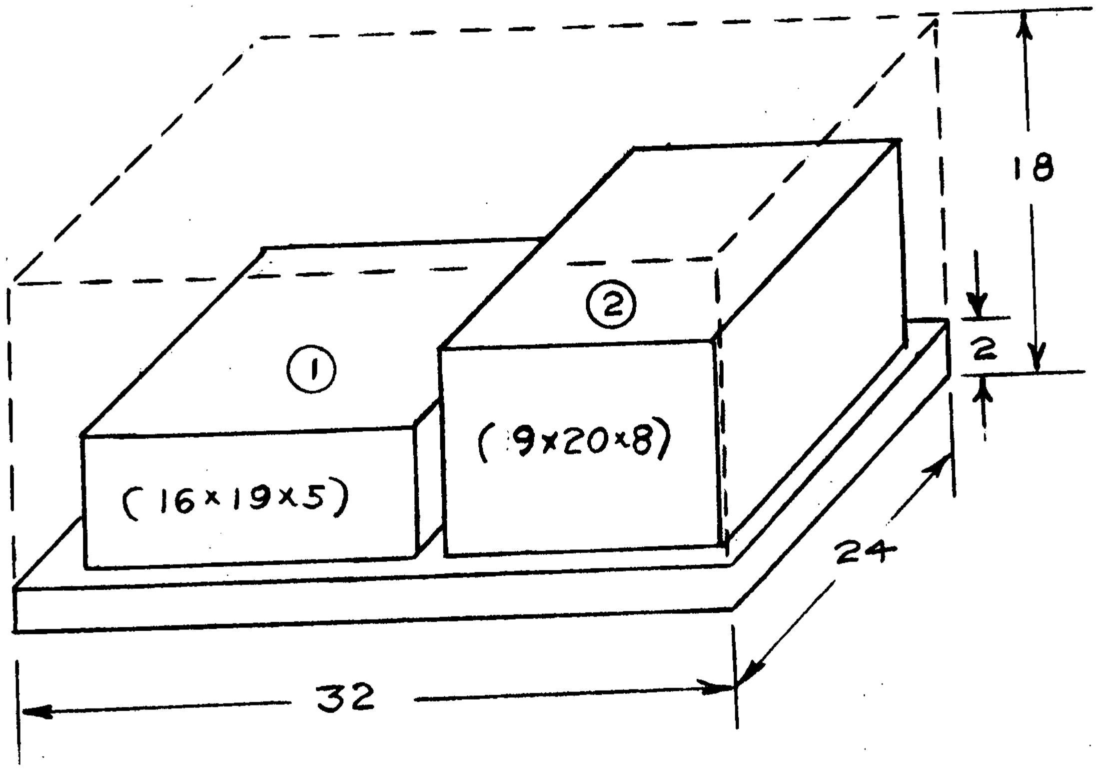
5.0 Modules for the EPS

The modularized concept for the power distribution is shown in Fig. 4, and the concept for the charge controllers/batteries is shown in Fig. 5.

6.0 Total EPS Weight

The weights for the EPS are itemized and totaled below.

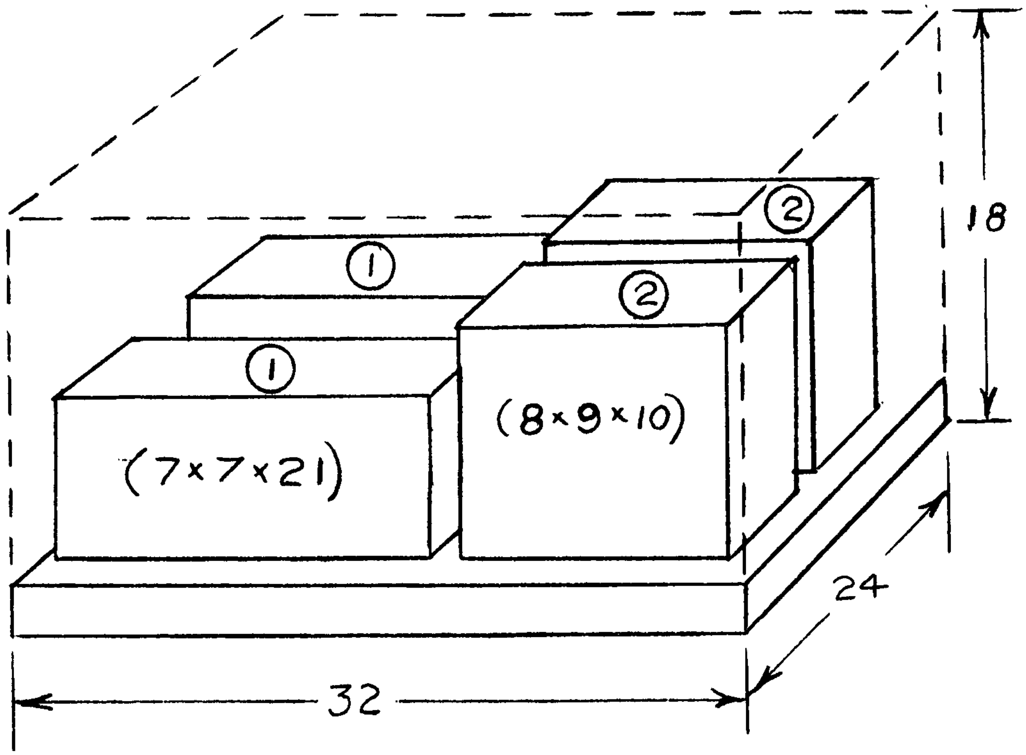
1 ea. Solar Array		
80 panels (7.8 lb ea)		624 lbs
4 Panel Support Structure (30 lb ea)		120
2 Deployment Mechanism (25 lb ea)		50
		794
15% contingency		120
		914 lbs
3 Battery Modules (206 lb ea incl. contingency)		618
1 Distribution Module (incl. contingency)		120
1 Spacecraft Cables (set)	320	
50% contingency	160	480
Total Weight, EPS		2132 lbs



EQUIPMENT

1	POWER DISTRIBUTION UNIT	①	32 lbs
1	REGULATOR	②	37
1	CABLES AND CONNECTORS (SET)		4
1	BASE		15
1	COVER		17
			105
	15% CONTINGENCY		15
			120

Fig. 4 Distribution Module
(1 Required)



EQUIPMENT

2	TYPE VII BATTERY	40 AH, NiCd	1 140 lbs
2	CHARGE CONTROLLER		2 12
1	CABLE AND CONNECTORS		3
1	BASE		15
1	COVER		17
			187
		10% CONTINGENCY	19
			206

Fig. 5 Battery Module
(3 Required)

7.0 Reliability

The reliability goal for the standard EOS Electrical Power subsystem is 0.889 for one year of orbit life. This goal is attained in the designs described as shown in Fig. 6.

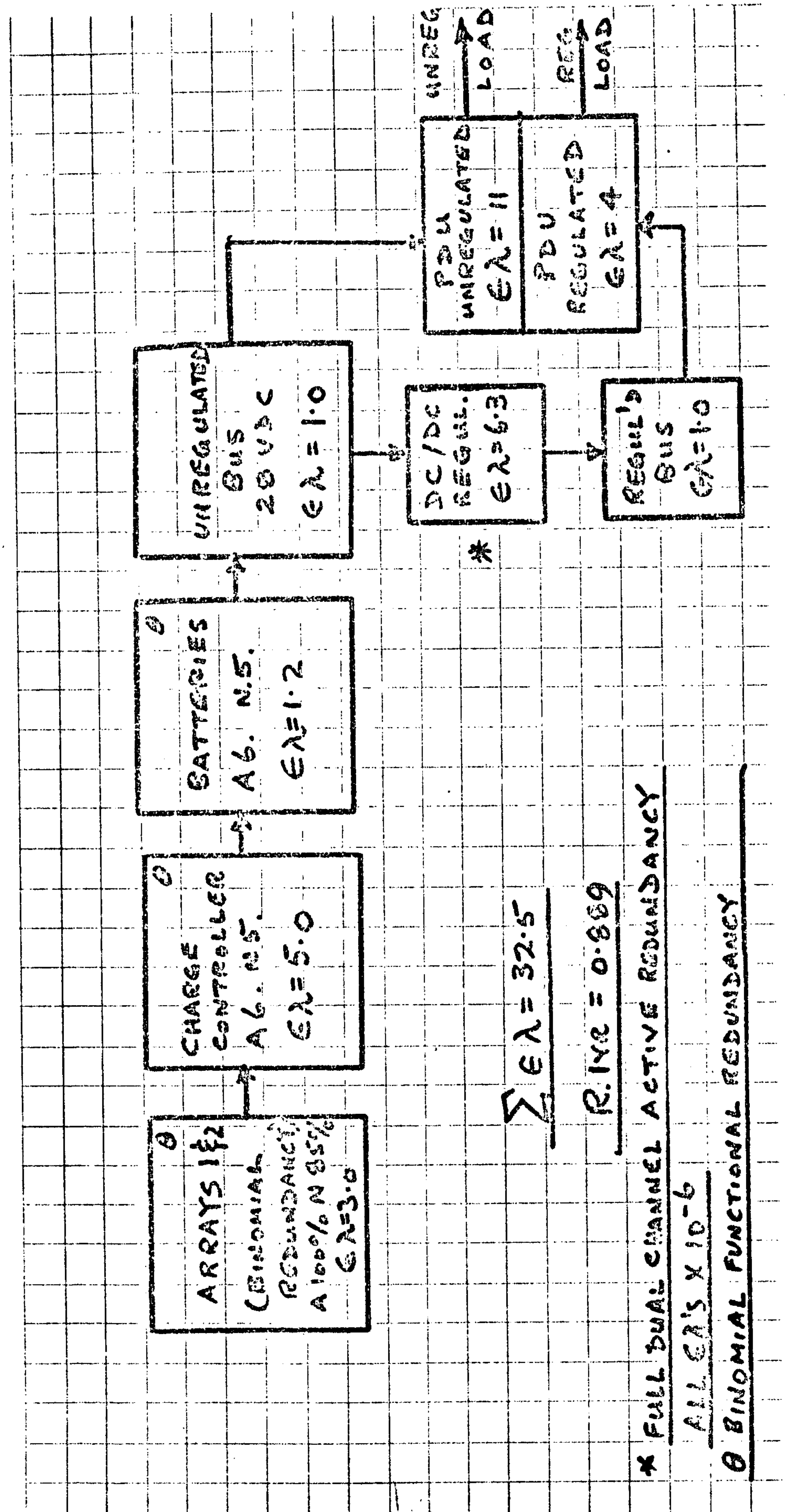


Fig. 6a Standard EOS Electric Power Subsystem Reliability Block Diagram (Sheet 1 of 2)

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- With exception of DC-DC regulator, subsystem is non redundant in the equipment duplication sense.

- Inherent reliability of design is achieved by safety factors, over-capacity vs all-up load, C > demand load

- System can tolerate 15% cell attrition and loss of one battery + charge controller, and still meet max. demand load. This places arrays and batteries + controllers in a condition of functional binomial case redundancy, i.e., 6 batteries + 6 controllers available, need only 5. 100% arrays available, need only 85%.

- Safety factor of approx. 2 on depth-of-discharge is with respect to all-up load during dark periods, which are approx. 50% of each orbit. Condition is highly conservative as demand load during darkness is <u>most</u> unlikely to exceed <u>40%</u> of dynamic load during solar illumination period. True safety factor ≈ 5.

- Reliability calculations recognize the safety factors of the design, in terms of binomial redundancy as indicated, load duty cycle as indicated, and active redundancy for the DC-DC regulator.

- Failure rates have been adjusted to reflect functional redundancies, and safety factors, and appear as effective rates, $\epsilon\lambda$'s
$\Sigma\epsilon\lambda = 32.5$ giving an R of 0.889 for 1 year on orbit.

Fig. 6b Standard EOS Electric Power Subsystem Reliability Block Diagram (Sheet 2 of 2)

APPENDIX

Tradeoff calculations for a fixed array vs a single-axis tracking array, assuming a β-adjust for both systems:

Number of cells for a tracking array:

.1116 watts per 2 x 4 cell
.5 array effectivity
.687 power decrease from 28°C to 85°C

$$\text{number of cells} = \frac{\text{power needed from array}}{\text{cell output x array effectivity x temperature degradation}}$$

$$\text{cells} = \frac{1155}{.1116 \times .5 \times .687} = 30{,}200 \text{ cells}$$

With 100 cells per sq ft and 1 panel = 4.76 sq ft:

$$\text{array} = \frac{30{,}200}{100} = 302 \text{ sq ft}$$

$$\text{panels} = \frac{302}{4.76} = 64 \text{ panels, or 32 per wing.}$$

Cost of increased number of cells for fixed array:

37,400 cells for fixed array (from section 4.0)
- 30,200 cells for a tracking array

7,200 cells increase for fixed array

Cost per cell for completed array (cell cost plus all of Lockheed's fabrication cost using "flex array" techniques) is estimated to be $18.25 per cell.
$18.25 x 7,200 cells = $131,000 per vehicle.

Tracking Array array-drive and slip-ring cost:

$ 75,000 - Ball Brothers Research Corp. verbal estimate
5,000 - Electronics for drive assembly

$ 80,000 x 2 per vehicle = $160,000 per vehicle

Net Savings for fixed array:

$160,000 - $131,000 = $29,000 per vehicle, plus approximately $200,000 for development and qualification tests for the drive/slip-ring assy.

PE-105

STANDARD EOS

ATTITUDE CONTROL SUBSYSTEM

PE-105

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: STANDARD EOS ATTITUDE CONTROL SUBSYSTEM	EM NO: PE-105 REF: DATE: 30 November 1971
AUTHORS: J. Cizauskas	APPROVAL: ENGINEERING F. C. Bolton, Wayne Miller

PRELIMINARY

9 pages

1. General

The Attitude Control Subsystem (ACS) provides thrust for reaction wheel unloading and backup, and emergency attitude hold for at least one month. The ACS consists of four identical modules installed on the outboard edges of the vehicle as depicted in Fig. 1. Each module contains four 1.75 lb rated thrusters, oriented as in Fig. 2 such that any three modules could provide 3 axis vehicle control. All four modules combined provide a total impulse of 6000 lb-sec thereby providing for at least 2 yrs. of orbital life. Loss of one module early in life has no appreciable effect, if any, on the 2-year orbit life.

2. Description

Each modular propulsion system consists of four major components. A fill valve is used to load Freon gas into a high pressure storage tank. The storage tank delivers Freon gas to the inlet of a pressure regulator over a decreasing pressure range from 1750 to 180 psia as the propellant is consumed by the thrusters. The regulator outlet provides 120 ± 10 psia gaseous Freon propellant to the common manifold of the 4 thrusters. As the vehicle guidance & control system requires external forces, the individual thruster valves are commanded open and closed to allow the 120 ± 10 psia gas to flow into the thruster chambers and expand out the nozzle producing thrust. A flow schematic of the ACS is presented in Fig. 3.

Freon 14 (CF_4), a dry, non-toxic, nonflammable, odorless gas that can be handled like gaseous nitrogen was selected for the ACS propellant. Each tank is initially charged with 38.5 lbs of Freon at an initial pressure of 1750 psig, which when blowndown to 180 psia minimum regulator inlet pressure, delivers 35 lbs of propellant for a total impulse of 1520 lb-sec. A 1750 psig maximum working pressure is selected to take advantage of the low Freon compressibility factor (Z) at that pressure and utilize an existing qualified low-cost stainless steel pressure vessel. Since Z factor is an inverse function of gas density, $Z = \frac{PV}{RT}$, a low Z factor allows efficient storage tank design when volume is not a primary constraint. Each tank is loaded through a fill valve located at the tank inlet/outlet fitting. This fill valve is opened or closed with a wrench and is capped after fill for redundant leakage protection. The pressurized Freon gas is supplied to the thruster via a 3600 psig rated inlet Regulator Valve Assembly which contains an inlet filter rated at 40 micron absolute, a solenoid latching valve, a regulator, and a downstream pressure relief valve with thrust nullifier. The regulator is supplied with Freon over a 1750 psig to 180 psig pressure range while supplying regulated 120 ± 10 psia gas to the thruster cluster assembly manifold. A solenoid latching valve controls tank pressure to the regulator dome which in turn positions the regulator main flow poppet. This solenoid latching valve is moved to either open or closed by a 80 millisecond electrical voltage pulse and has a position indicator which is monitored on TM. When the solenoid valve is open, gas pressure to the dome regulates outlet pressure to 120 ± 10 psia. When the solenoid valve is closed, the regulator main valve poppet closes and ceases to supply gas flow to the thrusters. Continued operation of the thruster after the regulator main poppet is closed, results in the downstream line pressure decreasing from 120 psia to 0 psia. The pressure relief valve is located downstream of the main flow poppet and

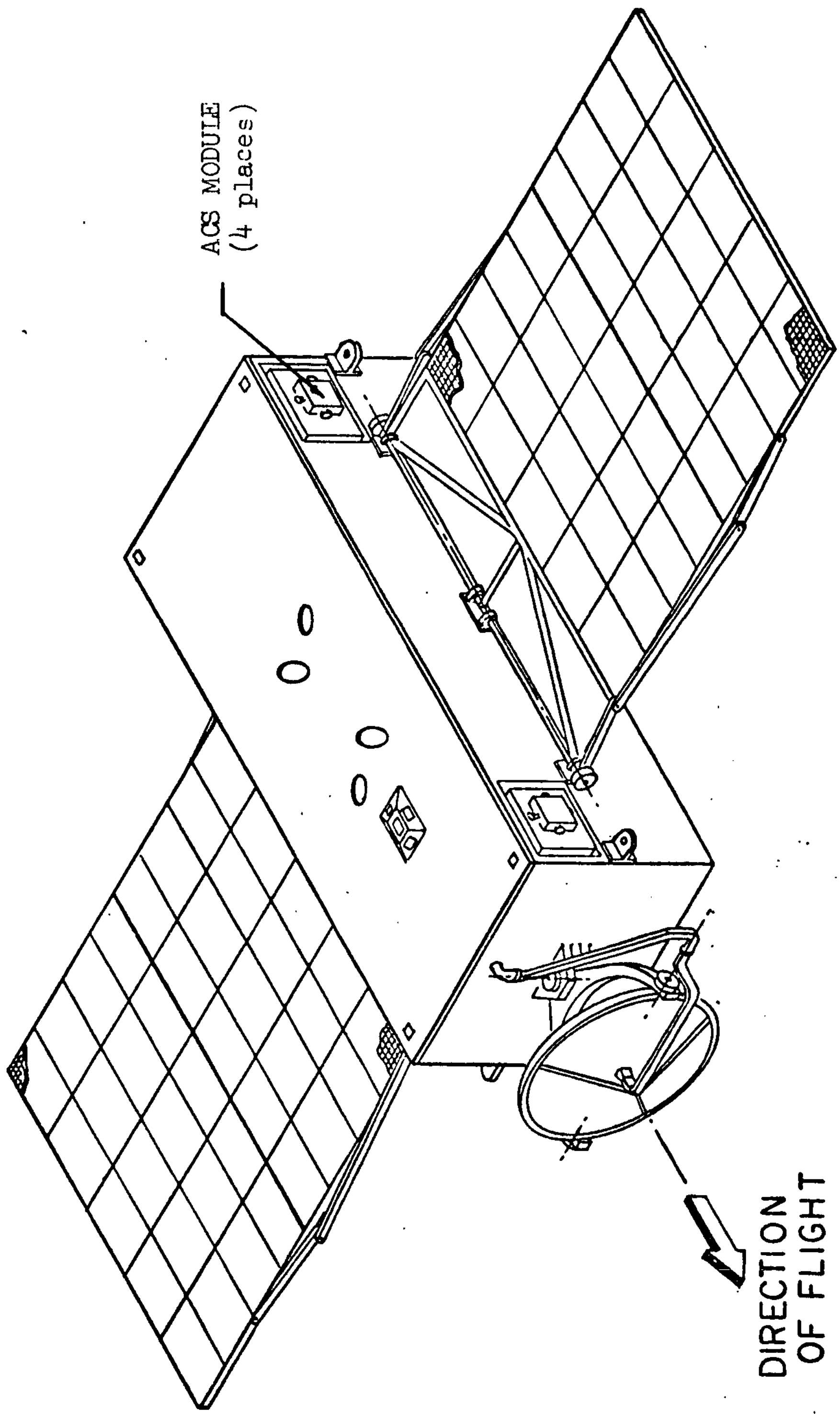

Fig. 1 Standard EOS Attitude Control Subsystem Installation

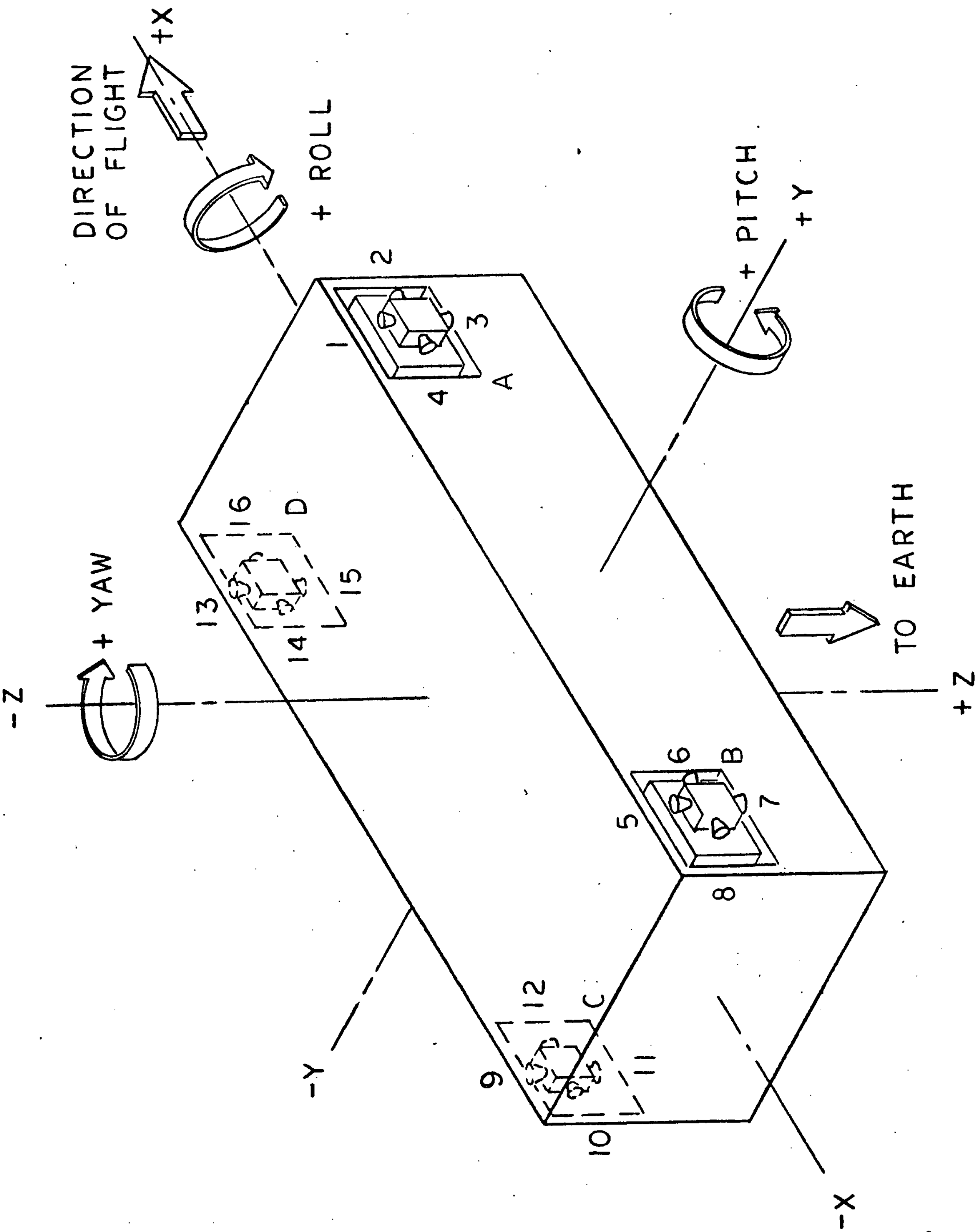

Fig. 2 Standard EOS ACS Thruster Orientation

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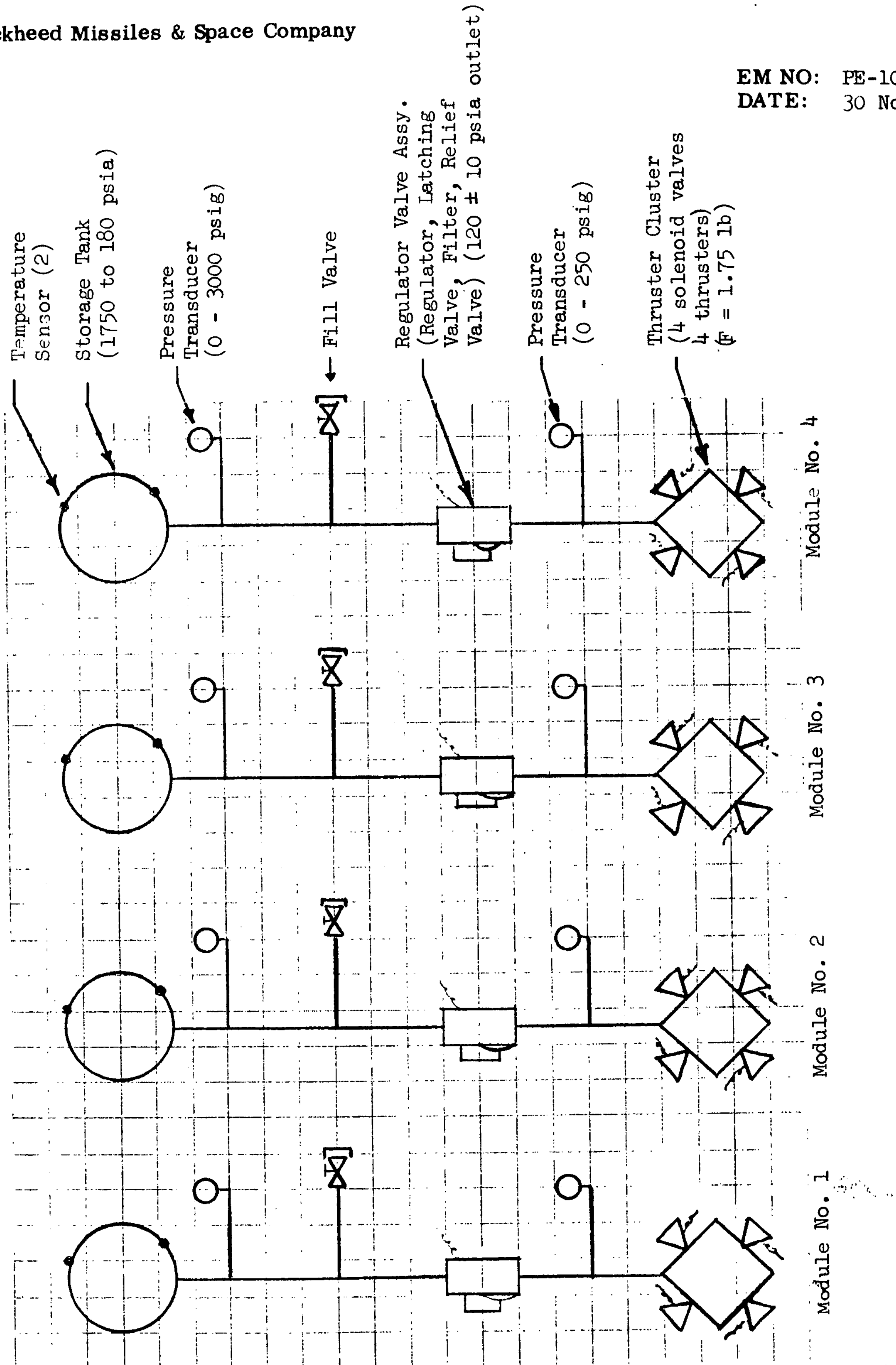

Fig. 3 Standard EOS Attitude Control Subsystem

vents should pressure exceed 160 psia due to a regulator malfunction or excessive main flow poppet leakage. This feature assures that the thrust valves will not be subjected to over-pressurization. The thrust valves are rated at 1000 psig burst. A 40 micron absolute rated inlet filter in the Regulator Valve Assy. protects the regulator and thruster valve poppet seals against excessive particle contamination to yield acceptable leakage rates.

Each thruster assembly contains an electrical valve which is driven by the Attitude Control System Electronics. The thruster nozzle is sized to deliver 1.75 lb of thrust at a 120 psia inlet pressure. Application of a 20 millisecond electrical pulse to a pair of thrusters limits the ACS minimum impulse bit to 0.07 lb-sec (2 x 1.75 x .020). Hence, the ACS requirements of limiting maximum thrust to 4 lbf per pair (2 x 1.75 = 3.5), and providing a 0.085 lbf-sec minimum impulse bit are satisfied.

3. Sensing Elements

Each module contains two pressure transducers, two temperature sensors, and a regulator valve indicating switch. A zero to 3000 psig range pressure transducer at the storage tank inlet/outlet and two temperature sensors attached to the external tank skin are provided for gas loading, orbital propellant mass statusing, and leakage detection. The zero to 250 psig range pressure transducer between the Regulator Valve Assy and the thrusters provides a regulator outlet pressure health check and can be used to check thruster valve leakage when the regulator main flow poppet is closed and the thrusters are inactive. Additionally, the pressure safety of the system is verified prior to manned access.

The Regulator Valve Assy latching solenoid valve has a position indicating monitor. Position indicators are used to determine if the valve is responsive to signal input, during operation and checkout. A valve indicating switch can be added to the thrusters if it is desirable to record thruster activity during orbit life.

4. Module

The ACS module shown in Fig. 4 is assembled using the major equipment listed in Table I plus miscellaneous tubing, bracketry, and electrical harnesses. Tubing is hard line stainless steel, brazed to component fittings, and designed to minimize potential leakage at joints. Each module assembly is locked in place in the vehicle by a mechanism designed to permit replacement of modules in orbit. The installation arrangement of the four modules as shown in Fig. 1 allows identical configuring of each module and hence a minimum stock inventory for module interchangeability.

5. Assembly and Test

Due to the relatively small size, low weight, and simplicity of the module it can readily be assembled and bench tested. All the previously acceptance tested major components are mounted, connected with the appropriately designed tubing, and brazed. The instrumentation is connected to a test panel, the storage tank pressurized, the system leak checked, and the system functionally checked out by providing command signals. After completion, the system is de-pressurized and stored prior to usage. Prior to vehicle installation, the storage tank is loaded with the appropriate quantity Freon 14 gas propellant.

Lockheed Missiles & Space Company

EM NO: PE-105
DATE: 30 November 1971

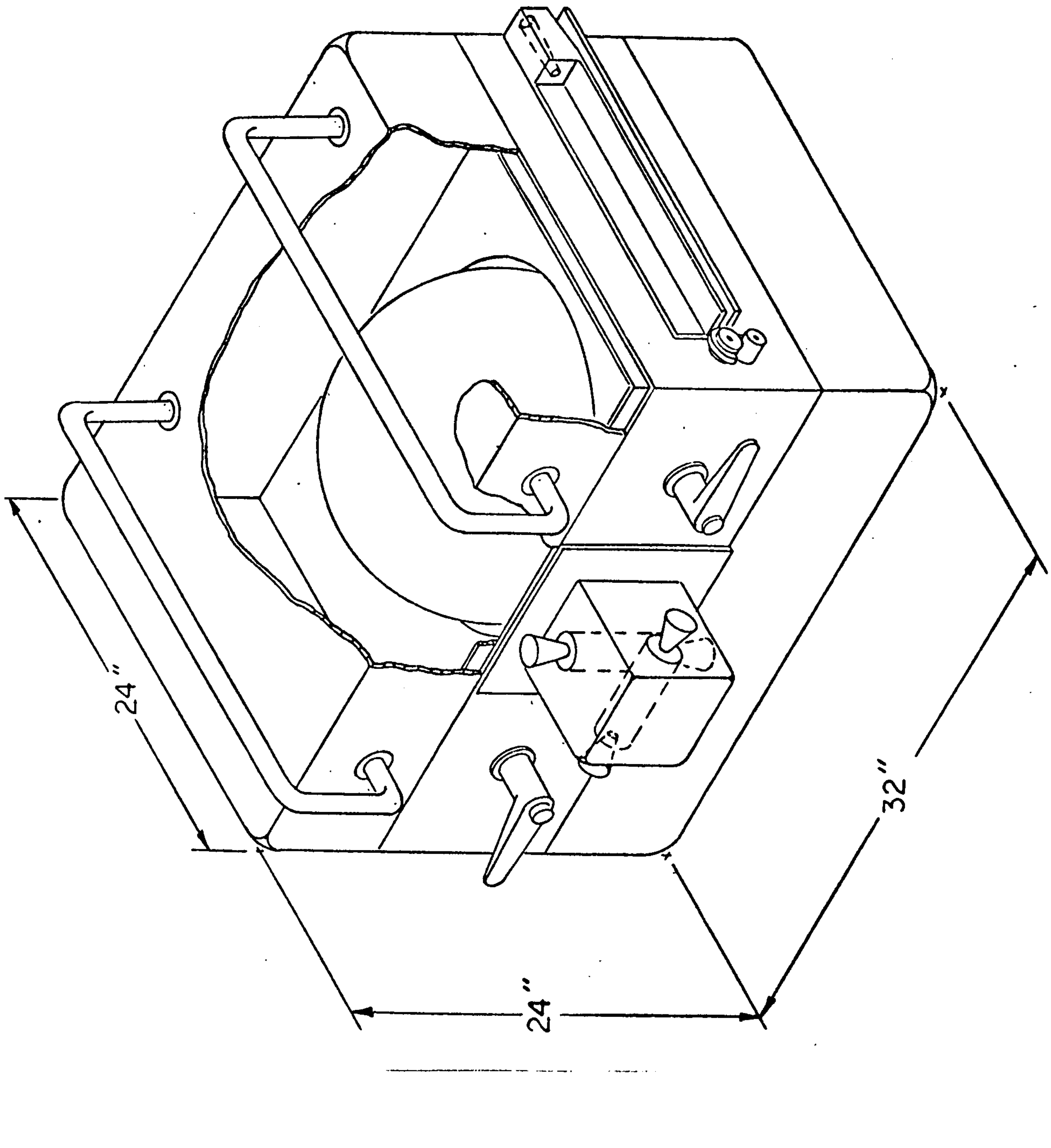

Fig. 4 Standard EOS Attitude Control Subsystem Module

Table I

ACS MODULE EQUIPMENT LIST

Equipment	Description	Weight	Cost ROM	Electrical Input	Electrical Output	Remarks
Storage Tank	ARDE, Inc. E3749 16" O.D. 301 Stainless 1960 in^3 7150 psig design burst	27.5 lbs	$ 3000	-	-	Qualified
Regulator Valve • Regulates • Latching solenoid • Filters • Relief & nullifier	IMSC 8100157 Rated at 3600 psig inlet 120 ± 10 psig 80 ms pulse command 40 micron absolute 175 psig crack	3.9 lbs	6000	22-29v	0-5v	Qualified
Fill Valve	IMSC 8106086-1 Valve shutoff plus cap 3000 psig rated	.25 lbs	400	-	-	Qualified
Thruster Cluster Ass'y.	Sterer 24050 1000 psig valve-burst .070 lb-sec MIB at 20 ms PW 4 thrusters per cluster 1.75 lb per thruster	3 lbs	5000	22-29v draws 1 amp at 28v	-	Qualified for 810K cycles vs 90K cycles req.
Pressure Transducers	IMSC 8100496 0-3000 psig - 13 0-250 psig - 11	.35 .35	1600 1600	22-29v 22-29v	0-5v 0-5v	Qualified Qualified
Temperature Sensor	IMSC 1618702-5 stick-on (2 req'd).	.07 lbs	100	22-29v	0-5v	Qualified

6. Preflight Sequence (Prior to installation of ACS module into EOS)

The regulated Freon 14 gas is loaded into the storage tank through a ground system filter and the module fill valve. At the completion of gas loading, the fill valve is turned closed with a wrench and the valve capped after the ground fill line is disconnected. The fill valve cap is checked for leakage, and final component checks are made audibly and by appropriate checkout equipment.

7. Design Objectives

Three design objectives were considered and met in designing the modular Attitude Control Subsystem. (1) The subsystem must provide attitude control after a single failure, (2) the cost of the module must be minimized and, (3) the subsystem must be simple and safe for ease of ground checkout, and man-rated for handling while charged with propellant.

The first design objective was achieved by providing four self-contained modules with sufficient propellant supply for a minimum of 24 months. Should a single failure make one module inoperative, the remaining three modules are capable of providing the required subsystem force functions without reduction in orbital mission life. Hence no redundant components were designed into the module resulting in design simplification.

The second design objective, low cost, was achieved by specifying existing qualified components to the maximum possible extent. All instrumentation, storage tank, fill valve, thruster cluster assembly and gas pressure regulator can be used "off the shelf" as qualified.

The Sterer PN 24050 Thruster Cluster Assembly specified was developed and qualified for the Apollo Program. For that application, a 350 psia inlet pressure was applied to yield a 5 lb thrust level unit. Reducing the inlet pressure to 120 psia for the present application results in a 1.75 lb thrust level. Additionally, the 1000 psig design burst pressure of the thruster valves allows an 8 to 1 safety factor of burst to working pressure for man-rated application.

Selection of a stainless steel storage tank results in a weight penalty of only 30% over expensive titanium, but a vessel that weighs approximately 50% less than an equal cost aluminum tank. The regulator chosen is efficiently designed to perform four individual functions: regulate pressure, shut-off flow, filter inlet gas, and provide downstream pressure relief. Combining all four functions within one valve housing results in a cost effective design.

The simplicity of the design is self-explanatory, and the man-rated safety aspect carefully considered by using high ratios (4 and 8 to 1) of equipment burst pressure to working pressure, adequate pressure instrumentation and use of Freon 14 which is a non-toxic, nonflammable gas that can be handled like gaseous nitrogen.

A reliability analysis for the ACS is presented in Fig. 5.

- Subsystem allocated reliability 0.9970
- Subsystem operating life .... 1 year, 8760 hours
- Operating mode .... Thrust on command
- Redundancy Mode: Binomial, 4 identical modules provided, Subsystem can operate on 3.
- Computation: $R = \sum_{r=m}^{n} \binom{N}{r} p^{r} (1 - p)^{n-4}$
- Subsystem basically the same as that of the Low-Cost OAO Vehicle. λ for module 42.2. Effective λ 3.4.
- R for 1 year 0.9978 assuming subsystem is dormant unless S/C subsystem is in failed mode. Duty cycle considerably less than 1.0% (anticipated)
- Module individual λ's:

Gas Storage Tank	0.20×10^{-6}
Pressure regulator	12.00
Fill Valve	4.50
Thruster Cluster Assy	7.00
Pressure XDCR (Hi)	5.50
Pressure XDCR (Lo)	5.50
Temp. Sensor	7.50×10^{-6}
	42.20

Fig. 5 Standard EOS ACS Module Reliability Data Summary

PE-106

GENERAL DESCRIPTION

STANDARD EARTH OBSERVATORY

SATELLITE

PE-106

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: GENERAL DESCRIPTION, STANDARD EARTH OBSERVATORY SATELLITE (EOS)	EM NO: PE-106 REF: DATE: 30 November 1971
AUTHORS: F. C. Bolton **PRELIMINARY**	APPROVAL: ENGINEERING F. C. Bolton Wayne Miller

36 pages

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1.0 General

The standard Earth Observatory Satellite (EOS) described in this Engineering Memo is to be launched by the Space Shuttle and is designed to be checked out and repaired, if necessary in the Shuttle prior to being placed into the mission orbit; to be repaired in orbit by th Shuttle during its design lifetime of one year; and to be recovered from orbit by the Shuttle after one year or longer for complete refurbishment and subsequent return to orbit. All communication with the standard EOS is assumed to be via a tracking and data relay satellite system of three equally spaced synchronous equatorial satellites. No on-board data storage is provided in the EOS design.

The nominal orbit of the EOS is near-polar circular and sun-synchronous, with altitude = 485 nm and inclination = 98 degrees. The number of orbits per day is 14.

The mission of the EOS is as follows:

- Provide a facility for the conduct of experimental research and development of advanced space systems for the earth observations disciplines in the latter half of the 1970-1980 decade.
- Obtain data in both the visible and infrared spectral bands to detect and distinguish the signatures of agricultural and forest resources and of natural thermal sources.
- Perform space observations of oceanographic phenomena and interactions of the ocean surface with the atmosphere to meet urgent needs for research data and the development of advanced operational sensors for the oceanographic and meteorological disciplines.
- Initiate a program to develop and test space sensors to monitor indicators of environmental quality, such as atmospheric pollution, on global and other appropriate scales.
- Provide a flexible data management system having the capability of providing data in appropriate formats and quantities on a timely basis, primarily for research purposes but also (as may be indicated during the initial research phase of one or more of the instruments) for quasi-operational use by appropriate agencies on a real-time basis.
- Develop a low-cost spacecraft for launch by the Space Shuttle, adaptable to supporting a wide variety of earth observations sensors and combinations of sensors.

To accomplish its designated mission, the EOS includes the seven sensors described briefly as follows:

1. Sea Surface Temperature Imaging Radiometer

   Five channels in the infrared and visible will provide precise daily global measurements of sea surface temperature at high (~ 2 km) spatial resolution, with correction for atmospheric effects in the line of sight. The data will contribute to our knowledge of ocean currents and the energy exchanges at the air-sea interface. This instrument will also provide measurements of components of the radiation budget of the earth-atmosphere system.

2. Cloud Physics Radiometer

Two visible and three infrared channels scanned from horizon to horizon will provide information on the physical characteristics and distribution of the cloud cover over the globe. The multispectral approach will permit the determination of such factors as cloud top pressure level, cloud thickness, droplet size, the density of condensed water and the distinction between ice clouds and those made up of liquid water droplets.

3. Oceanic Scanning Spectrophotometer

Ten to twenty narrow spectral bands in the visible spectrum will measure sea surface color, providing data needed for identification of nutrient rich and sterile areas of the oceans and their temporal changes. Data from this instrument together with the Sea Surface Temperature Imaging Radiometer will provide information on oceanic circulation and ecological processes, and possibly certain types of pollution, such as oil slicks.

4. Thematic Mapper

With seven spectral bands in the visible and selected atmospheric windows in the infrared, this instrument will provide data primarily for terrestrial resources survey. With an instantaneous field of view for 66 meters, and a 100 nm swath, the Thematic Mapper would continue and extend the earth resources surveys initiated with ERTS A and B. The addition of two infrared channels in particular will extend the ability to detect and distinguish the signature of agricultural and forest resources, and of natural thermal sources.

5. Passive Multichannel Microwave Radiometer

This instrument, operating in a number of bands to be selected in the wave length range between 1 and 11 cm will afford a global all-weather capability to measure such parameters as sea state, sea surface temperature, precipitation over the oceans, and the extent and nature of sea ice, as well as soil moisture and properties of snow and ice cover over the land.

6. Upper Atmospheric Sounder

Several candidate sensors are under study which can determine temperature profiles in the atmosphere above 30 km, determine the nature and distribution of atmospheric components such as ozone or water vapor, and contribute to our understanding of the dynamics and photochemistry of this region.

7. Atmospheric Pollution Sensor

Several types of instruments are under development for the detection and measurement of gases associated with atmospheric pollution. The sensor data, based on the Remote Gas Filter Correlation Analyzer, (AAFE 71-18) are representative of the payload allocation required for atmospheric pollution monitoring.

The principal reference used in the design of the standard EOS was the following:

Earth Observatory Satellite (EOS)
Definition Phase Report, Volume 1
August 1971
Goddard Space Flight Center
Greenbelt, Maryland

2.0 Standard EOS Configuration

The general configuration of the standard EOS is shown in Fig. 1 and the location of equipment in Fig. 2. With few exceptions, the spacecraft subsystem equipment has been packaged in modules that can be removed and replaced in orbit by a Space Shuttle crewman. This makes possible the checkout and rapid repair, if necessary, of the EOS in orbit if it should fail prior to recovery for refurbishment. The mission equipment (the seven sensors) have not been packaged in similar replaceable modules during this limited design study. However, it would be logical to modularize the sensors to gain the significant cost advantages and operational flexibility that modularization affords.

The standard EOS configuration provides spare volume for growth of either spacecraft subsystems or mission equipment.

The designations of the subsystem modules and the equipment contained in each module are listed in Fig. 3.

3.0 EOS/Shuttle Interfaces and Launch Concept

Figure 4 shows the standard EOS supported in the Space Shuttle cargo bay, and elevated above the bay prior to its release into the mission orbit. The EOS is supported at four points in the mid-plane by retractable frames extended from the side-walls of the cargo bay. These frames support the EOS, during the boost phase, without imposing any loads on the EOS due to Shuttle structural deformation.

The EOS is elevated out of the cargo bay by four tape booms, one at each of the four inboard corners of the spacecraft. The tape booms are synchronized, and their tips are constrained by a rectangular frame to insure their alignment with the mating cones on the spacecraft.

A single umbilical provides hardline connections between the Shuttle and the EOS while the EOS is attached to the Shuttle. It is disconnected by remote control prior to the release of the EOS. The EOS is checked-out in orbit through the umbilical.

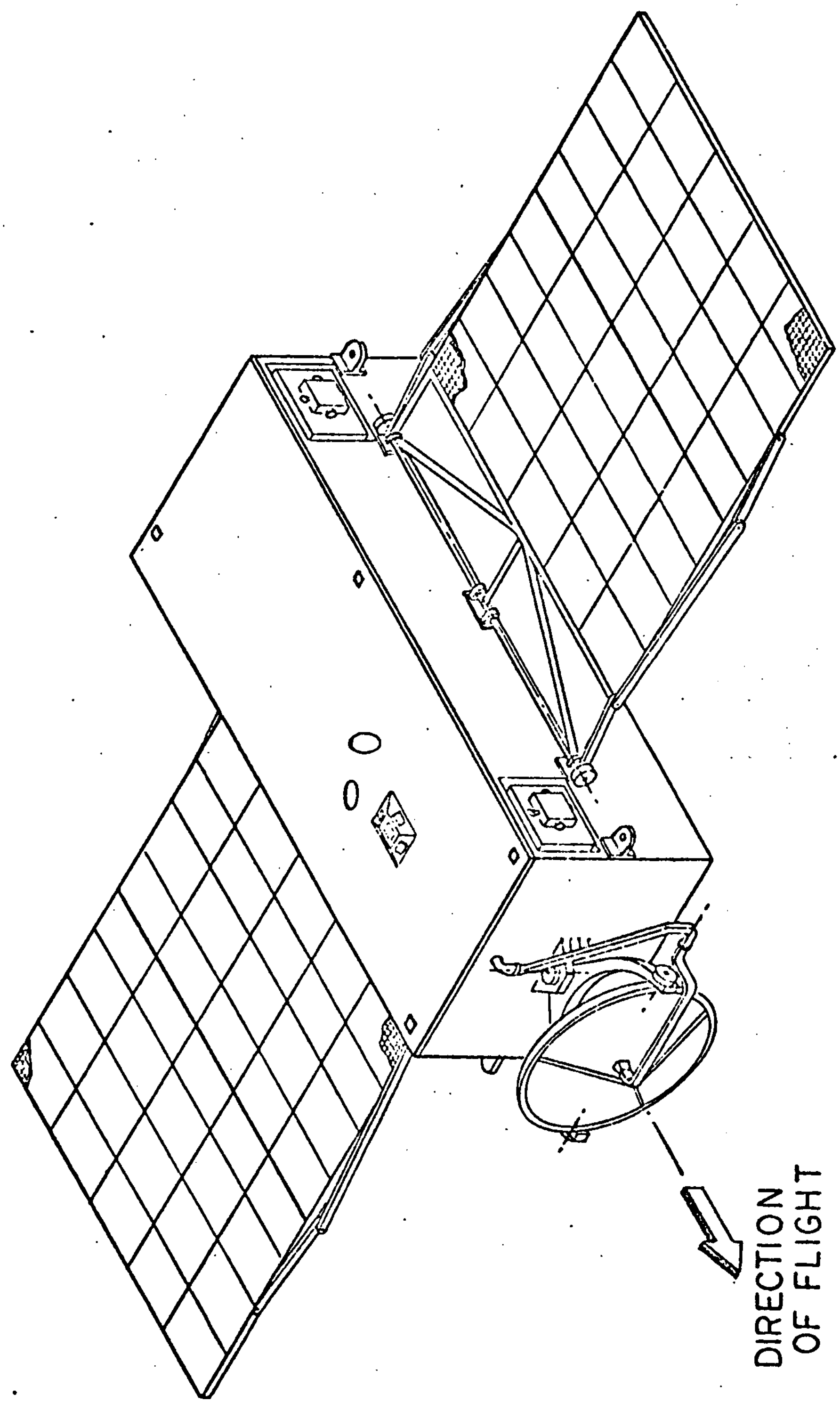

Fig. 1 General Configuration Standard Earth Observatory Satellite

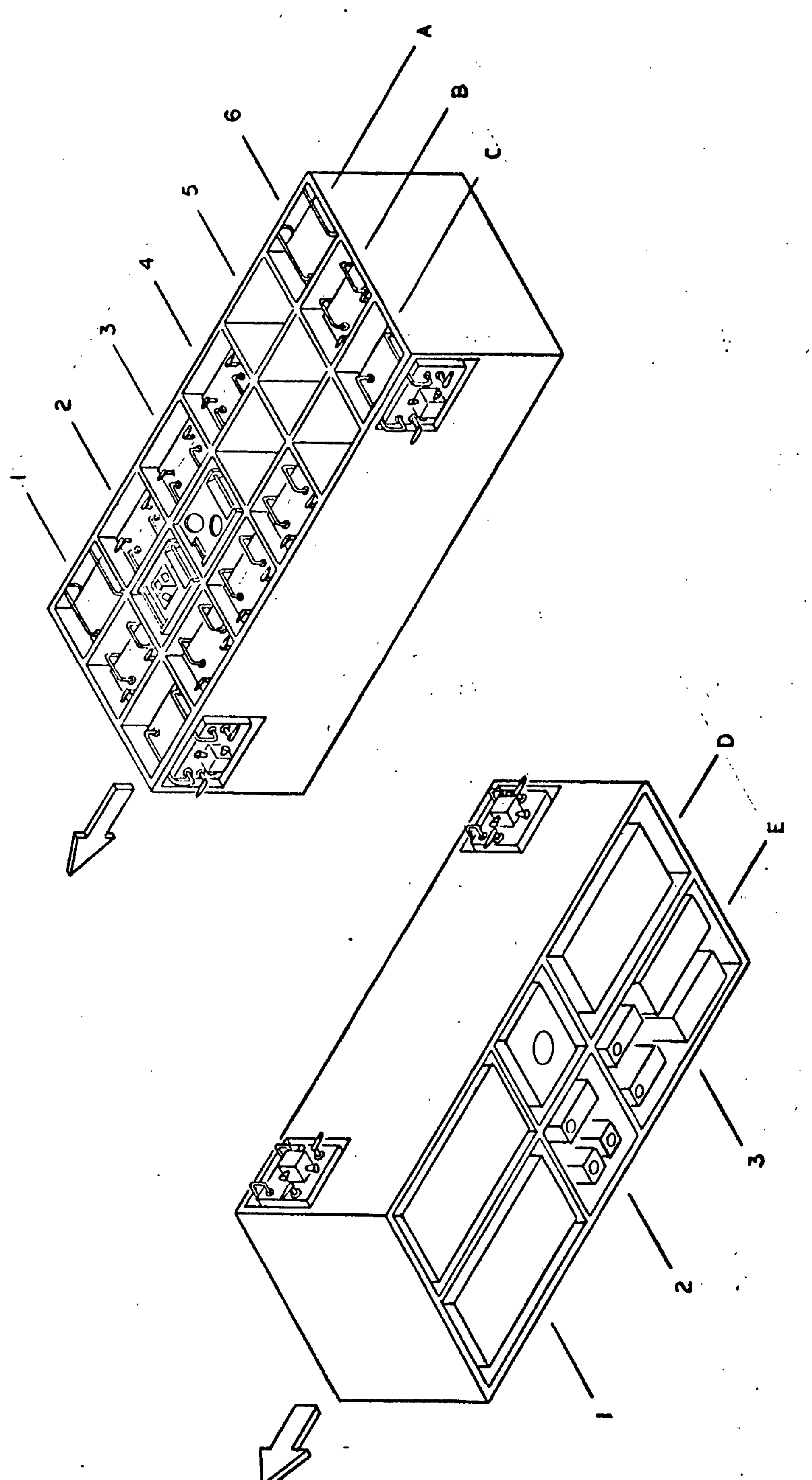

Fig. 2a Equipment Locations - Standard Earth Observatory Satellite
(Sheet 1 of 2)

MISSION EQUIPMENT

D-1	Passive Microwave Radiometer (λ = 0.81 cm)
D-2	Thematic Mapper
D-3	Passive Microwave Radiometer (λ = 2.81 cm)
E-1	Passive Microwave Radiometer (λ = 6.01 cm)
E-2	Ocean Scanning Spectrophotometer Atmospheric Pollution Sensor Upper Atmosphere Sounder
E-3	Cloud Physics Radiometer Sea Surface Temp. Radiometer Passive MW Radiometer (λ = 1.67 cm) Passive MW Radiometer (λ = 1.40 cm)

SPACECRAFT SUBSYSTEM MODULES

A-1	Attitude Control Module No. 1
A-2	S & VHF Band Communication Module
A-3	Battery Module No. 1
A-4	Power Control Module
A-5	Empty
A-6	Attitude Control Module No. 2
B-1	K-Band Communication Module
B-2	S&C Secondary Reference Module
B-3	S&C Primary Reference Module
B-4	Empty
B-5	Empty
B-6	Reaction Torque Module
C-1	Attitude Control Module No. 3
C-2	Data Processing Module
C-3	Battery Module No. 2
C-4	Battery Module No. 3
C-5	Empty
C-6	Attitude Control Module No. 4

Fig. 2b Equipment Locations - Standard Earth Observatory Satellite
(Sheet 2 of 2)

Subsystem	Module	Equipment in Module	Module Weight (lb)
Stabilization & Control	Primary Sensing Module No. 1	• Fixed Head Star Trackers (2) • FHST Electronics (2) • Three-Axis Rate Sensor • Precision Equipment Mount • Module Base • Module Cover • Cables and Connectors	Basic 91 lb 15% contingency 14 Total 105 lb
Stabilization & Control	Secondary Sensing Module No. 1	• Sun Aspect Sensor (5) • Sun Aspect Sensor Electronics • Rate Gyro Package • Secondary Stabilization & Control Electronics • Module Base • Module Cover • Cables & Connectors	Basic 56 lbs 15% contingency 8 Total 64 lbs
Stabilization & Control	Reaction Torque Module No. 1	• Reaction Wheel (3) • Wheel Support and Safety Shield • Wheel Drive Electronics • Magnetic Torquer (3) • Mag. Torquer Electronics (3) • Module Base • Module Cover • Cables & Connectors	Basic 133 lbs 15% contingency 20 Total 153 lb
Communication Data Processing & Instrumentation	K_u-Band Communication Module	• K-band TWTA (50 watts out) (2) • K-band PLL Receiver • K-band QPSK Modulator/Driver • K-band Multicoupler • Interface Unit (High Rate) • Module Base • Module Cover • Waveguide, Cables, Connectors	Basic 74 lbs 15% contingency 11 Total 85 lbs

Fig. 3a Standard EOS Subsystem Modules (1 of 3)

Subsystem	Module	Equipment in Module	Module Weight (lb)
Communication Data Processing & Instrumentation	S-Band/VHF Communication Module No. 1	S-Band Transmitter (10 watts out) S-Band Receiver S-Band QPSK Modulator/Driver S-Band Multicoupler Delay Lock Loop Correlator (2) Antenna Servo Electronics VHF Transmitter (5 watts out) VHF Receiver/Demodulator VHF Multicoupler Module Base Module Cover Cables and Connectors	Basic 68 lb 15% contingency 10 Total 78 lbs
Communication Data Processing & Instrumentation	Data Processing Module No. 1	Digital Computer Interface Unit (Med. & Low Rate) Timer Module Base Module Cover Cables & Connectors	Basic 79 lb 15% contingency 12 Total 91 lbs
Communication Data Processing & Instrumentation	Antenna Module No. 1	Antenna (6 ft. dish) Antenna Gimbal and Base Antenna Feed (S&K Bands) Rotary Joint (K-Band) Rotary Joint (S-Band) Antenna Servo Motors & Gears Antenna, VHF Waveguide, Cables, Connectors	Basic 65 lb 15% contingency 10 Total 75 lbs
Electrical Power	Battery Module (3 required) No. 1 No. 2 No. 3	NiCd Battery, Type 7, 40AH (2) Charge Controller (2) Module Base Module Cover Cables & Connectors	Basic 187 lb 10% contingency 19 Total 206 lbs

Fig. 3b Standard EOS Subsystem Modules (2 of 3)

Subsystem	Module	Equipment in Module	Module Weight
Electrical Power	Distribution Module No. 1	Power Distribution Unit Regulator Converter Module Base Module Cover Cables & Connectors	Basic 105 lbs 15% contingency 15 Total 120 lbs
Electrical Power	Solar Power Module (2 Required) No. 1 No. 2	Solar Power Panel (40) Panel Support Structure Deployment Mechanism	Basic 397 lbs 15% contingency 60 Total 457 lbs
Attitude Control	Attitude Control Module (4 Required) No. 1 No. 2 No. 3 No. 4	Gas Storage Tank Regulator Valve Assembly Fill Valve Thruster Cluster Transducers (set) ACS Electronics Plumbing Module Base Module Cover Cables & Connectors	Basic 87.5 lbs 15% contingency 13.5 Total 101 lbs

Fig. 3c Standard EOS Subsystem Modules (3 of 3)

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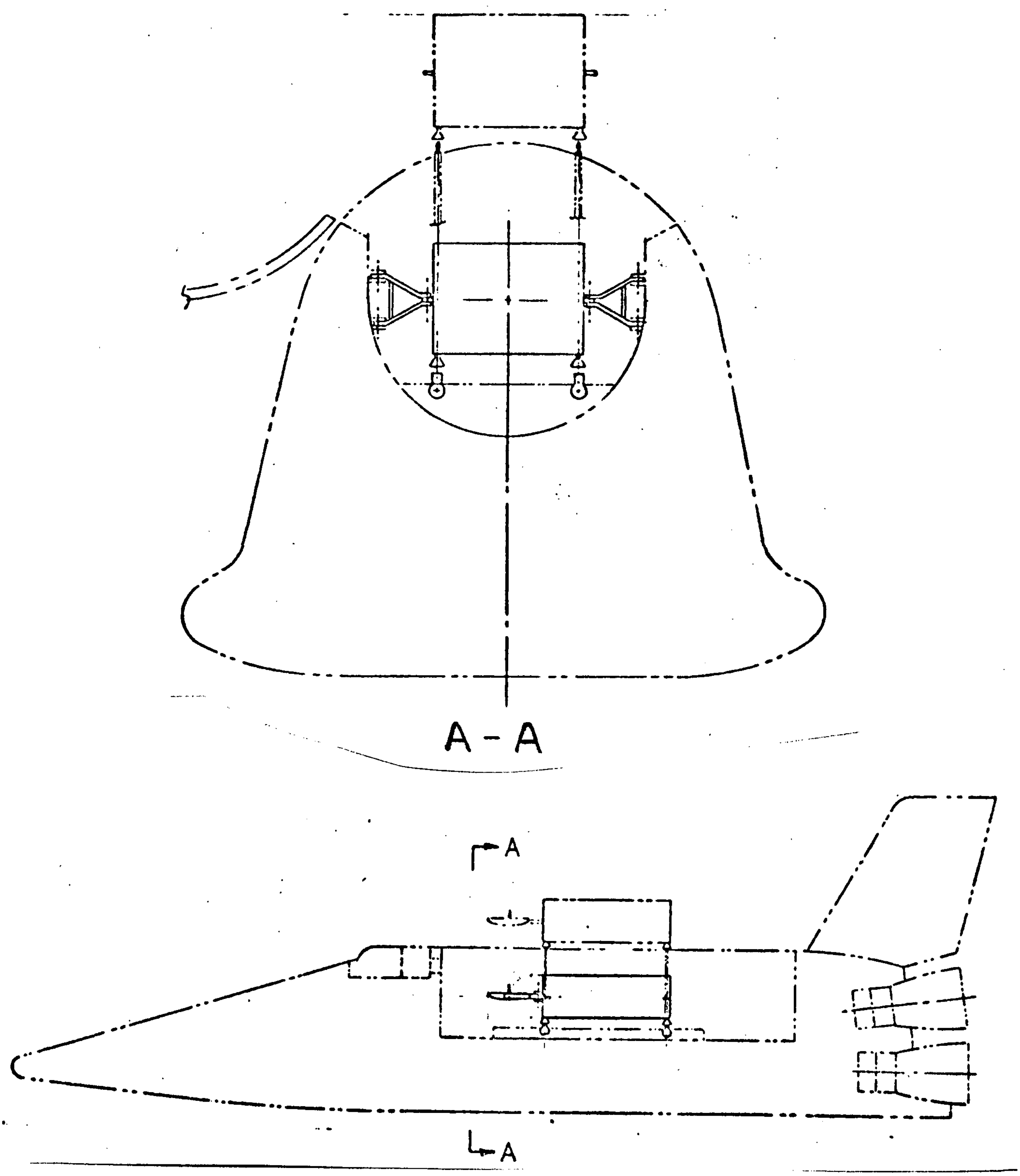

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Fig. 4 Standard EOS/Space Shuttle Mechanical Interface

4.0 Description of Standard EOS

Separate Engineering Memoranda have been written to describe the major subsystems of the EOS. They are as follows:

PE-102 Standard EOS Stabilization and Control Subsystem

PE-103 Standard EOS Communication, Data Processing, Interface, and Instrumentation Subsystem

PE-104 Standard EOS Electrical Power Subsystem

PE-105 Standard EOS Attitude Control Subsystem

These subsystem descriptions will be summarized briefly in subsequent paragraphs.

4.1 Structures Subsystem

The primary structure of the standard EOS is shown in Figure 5. The entire structure is made up of commercial grade aluminum sheet and extrusions. All structural elements are sized to have high factors of safety to minimize structural analysis and static load testing. The structure is symmetrical; therefore, the number of different parts is minimized and design and fabrication costs are reduced.

The structure is basically an "I-beam" with the mid-plane deck forming the web and the long sides forming the caps. The mid-plane deck is approximately six inches thick and hence very stiff. It is the baseline for alignment of equipment. Precision mounting surfaces are provided for establishing critical relationships such as that of the Thematic Mapper to the Primary Sensing Module. The electrical harness that interconnects the subsystem modules and the mission equipment is largely contained within the mid-plane deck. Four fittings, one at each corner of the deck, support the EOS during ground handling and in the Shuttle.

The volume bounded by the "caps of the I-beam" is subdivided into compartments to contain the spacecraft subsystem modules and the mission equipment. Eighteen uniformly sized compartments on the Zenith side accommodate fourteen subsystem modules and provide for additional modules if detail design determines that they are required. In-flight the subsystem modules are protected from solar radiation by three insulated doors. The doors may be opened for access to modules when the EOS is in the Shuttle cargo bay and the cargo bay doors are open.

Six compartments on the Earth side provide spaciously for the mission equipment. The available volume is sufficient to accommodate the mission equipment modularized for in-orbit replacement; however, the mission equipment has not been modularized in this limited design study. The large earth-oriented surface area of the standard EOS provides for flexibility in the selection and installation of earth sensors. The arrangement of sensors shown in Fig. 2 allocates approximately 70 percent of the earth oriented surface area to fixed Passive Microwave Radiometer (PMR) antennas. If additional surface area is required by other sensors the PMR antennas may be deployed.

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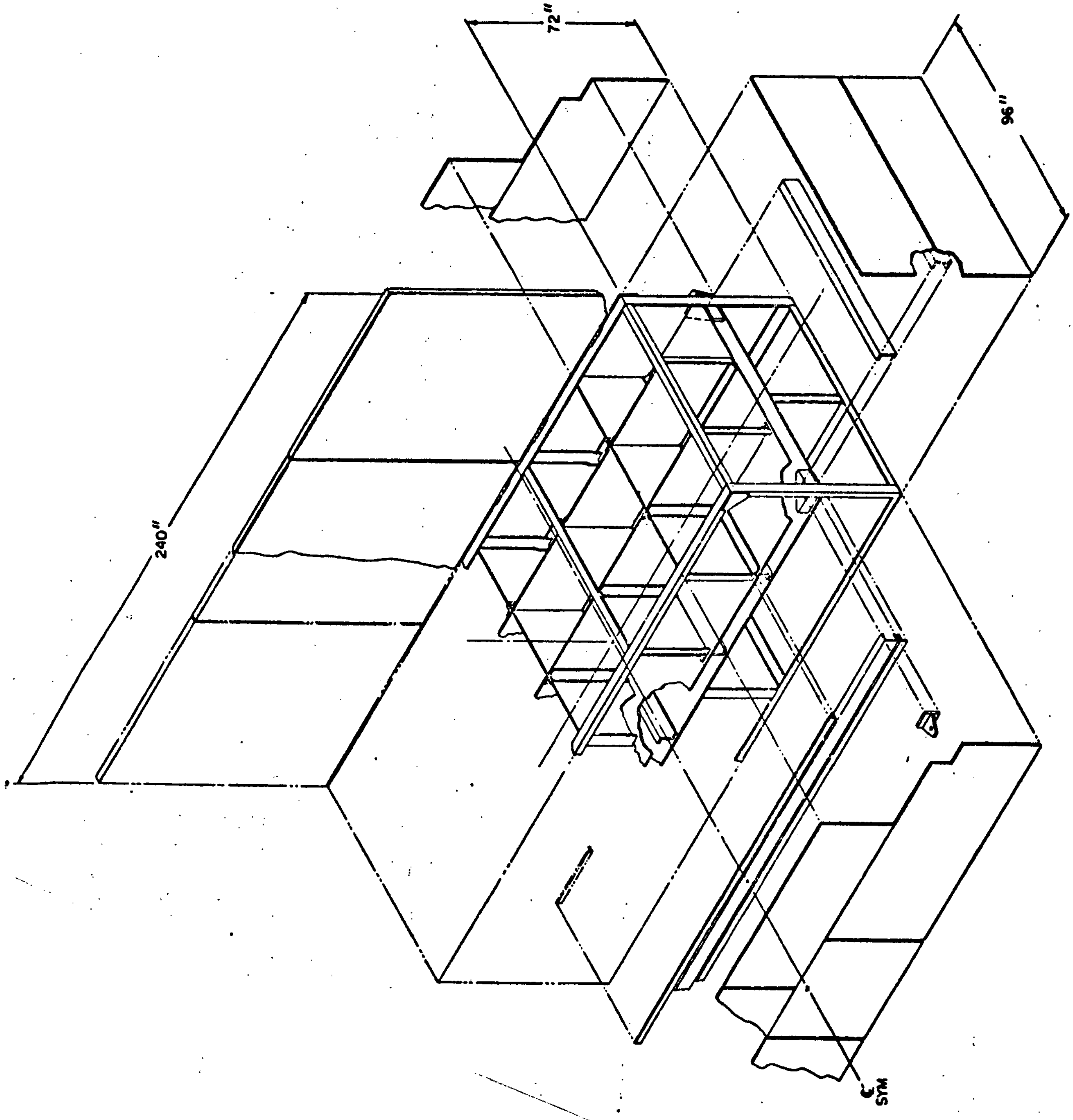

Fig. 5 Standard EOS Structure Assembly

4.2 Stabilization and Control Subsystem

The Standard EOS Stabilization and Control Subsystem is required to maintain the spacecraft in an earth-oriented attitude for up to one year and provide readout for ground-based precision attitude determination. The functional and equipment relationships of the primary stabilization and control mode are shown in Fig. 6, and those of the secondary mode in Fig. 7.

The principle of operation of the primary attitude determination function is to obtain accurate long-term attitude information from a star sensor and a computer-stored star catalog, while three-axis rate sensor gyros provide a precise short-term reference. A computer processes and mixes the information from both sources. A check on the vehicle orientation in space is available to ground stations by readout of solar aspect sensors. These sensors would also be used for attitude reacquisition should a catastrophic event cause the spacecraft to tumble.

Attitude control torques are obtained by varying the speed of reaction wheels or pulsing attitude control thrusters. Momentum absorbed by the wheels is continuously reduced as the magnetic torquers interact with the ambient earth field under control of the CDPI computer. Should an unexpectedly large disturbance torque cause the wheels to saturate (reach maximum speed) desaturation is accomplished by torquing the spacecraft with the attitude control jets.

Reorientation from one attitude reference to another, if required, is accomplished by a series of slews using the wheels and/or gas jets, which also control TARS detected attitude errors about the non-slew axes.

The Standard EOS S&C Subsystem is implemented entirely with equipment planned for the Standard Spacecraft S&C inventory.

The Standard EOS S&C Subsystem utilizes two fixed-head star trackers (FHT) with narrow-angle (~5 deg) optics for both on-board (coarse) attitude determination and for after-the-fact ground (precision) attitude determination. Since, after the first orbit or two, all accuracy requirements can be met with but one FHT, the second unit provides essentially full mission redundancy.

The Stabilization and Control subsystem equipment is packaged in three modules that are designed to be removed from the spacecraft in-orbit and replaced by equivalent modules without calibration or mechanical alignment. The locations of the S&C modules in the EOS are shown in Fig. 2 and the major equipment contained in each module is listed in Fig. 3.

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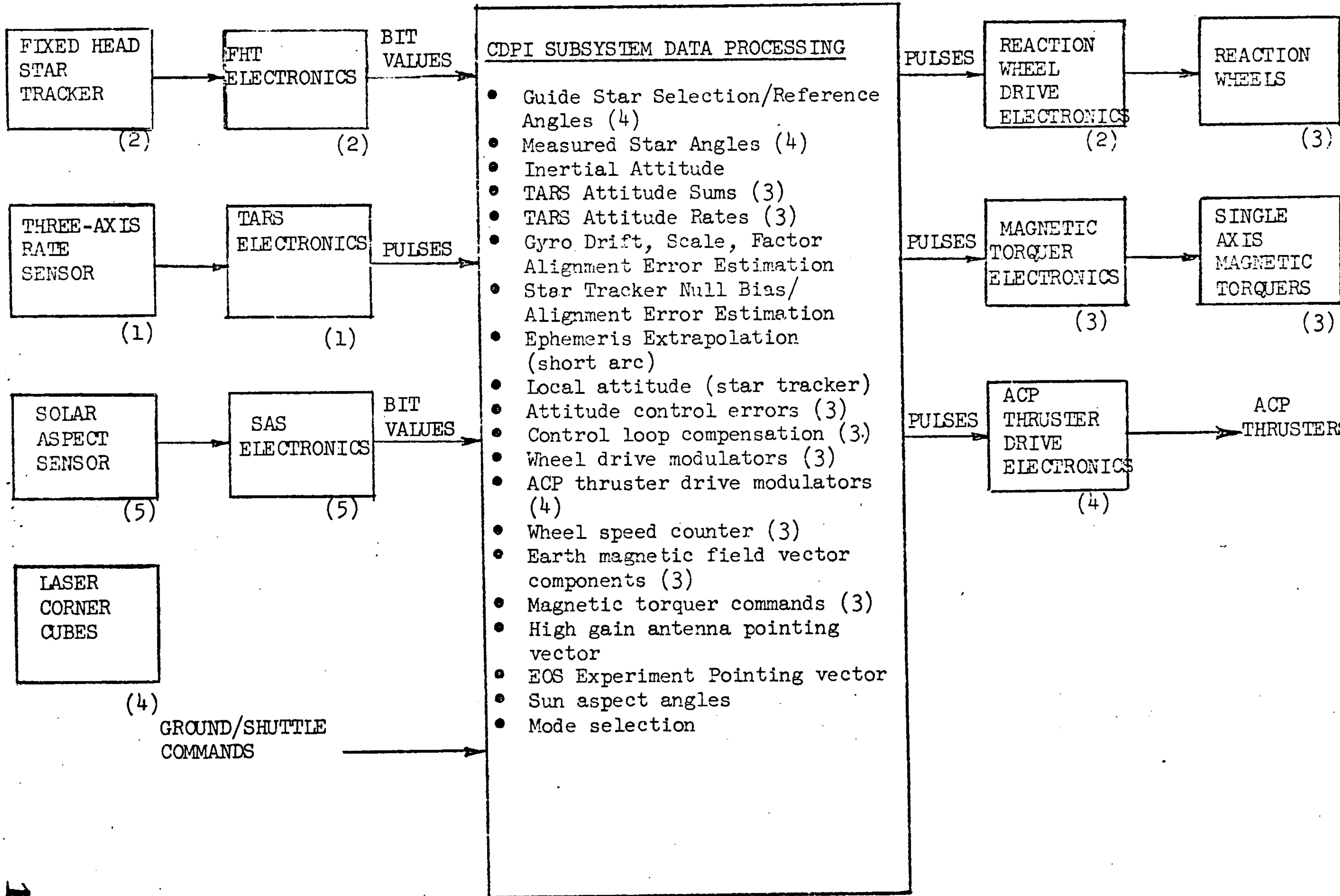

Fig. 6 Standard EOS Stabilization & Control Subsystem - Primary Mode (including Redundant Equipment)

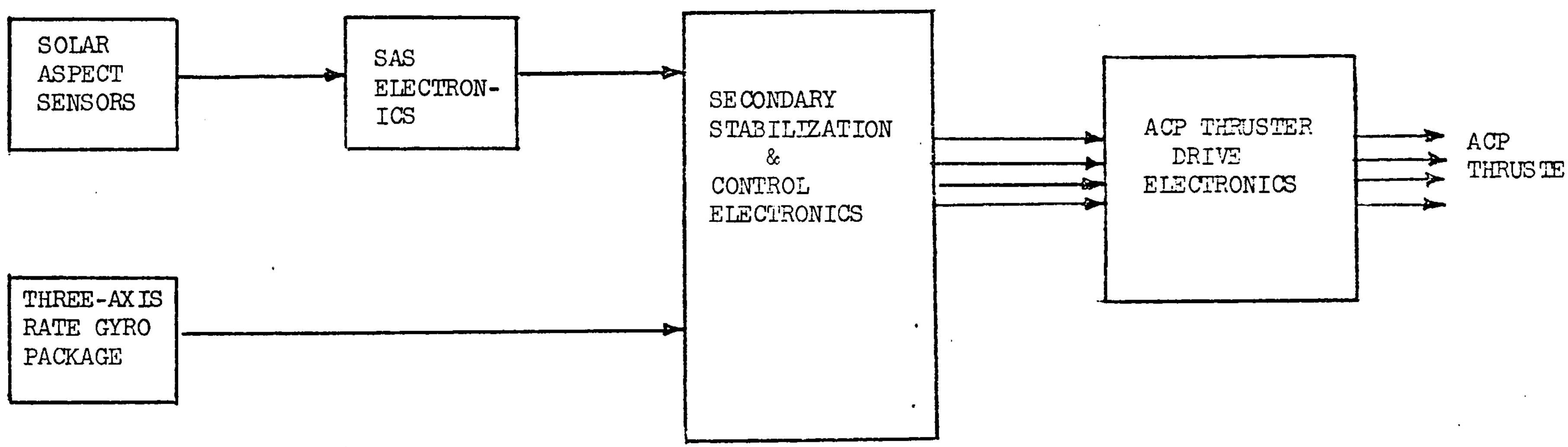

Fig. 7 Standard EOS Stabilization & Control Subsystem
- Secondary

4.3 Communication, Data Processing, and Instrumentation Subsystem

The Communication, Data Processing, and Instrumentation Subsystem (CDPI) must provide command, control, data processing including computation, and communication functions for the mission equipment and the spacecraft subsystems. The proposed CDPI subsystem is subdivided into four sections as shown in Fig. 8. Each of these sections is described in the following paragraphs.

Communication Section

A block diagram of the Communication Section, as it is related to the other CDPI sections, is shown in Fig. 9. As seen in the figure, the communication section is in three parts, where the first part is a K-band configuration devoted exclusively to the ultra-high 30 Mb/sec data-rate sensor information communication. The second part, an S-Band configuration combines medium-high data rates, ranging and command functions. The third part provides a VHF channel for telemetry of low data rate information and serves as a backup link should there be a failure in the other channels. Backup consists of safing the system or performing those mission functions for which communication remains open, i.e., ultra-high and low data rate operation with loss of S-band, or medium high and low data rates with loss of K-band.

Housekeeping instrumentation outputs and power control inputs are required to and from the Communication Section. Interconnection, multiplexing, conditioning, etc., for these components are provided via the Ultra-High Data Rate Interface Unit.

Ultra-High Data Rate Interface Unit

As seen in Fig. 9, there are three units in the Interface Section, corresponding to the three channels of the Communication Section. The first unit, which interconnects the Thematic Mapper with the K-band modulator, as well as other Interface units, is shown in Fig. 10.

A number of information and control channels are provided in the Interface Unit. Output from the Thematic Mapper 30 Mb/sec output channel is fed via coaxial cable to a buffer amplifier. The amplifier provides the gain necessary to drive the K-band modulator as well as supplies a proper impedance match to the Thematic Mapper/modulator circuitry. The amplifier may be switched on and off via power control logic.

Instrumentation channels are also provided to obtain status and housekeeping data. These include signal conditioned and non-signal conditioned analog data and bilevel data. A subcommutating sampler and multiplexer is supplied in the analog outputs and a sample and hold register is furnished for the bilevel data. Those devices reduce interface interwiring requirements.

Three types of control are incorporated - power on-off, as with the amplifier previously discussed, sensor channel control through sequence drive logic, and a slewing control to orient the Thematic Mapper in accordance with either ground command or a routine stored in the digital computer. It should be noted that timing registers and drive logic are included to serve the particular requirements of the components served by the Interface Unit. The originating

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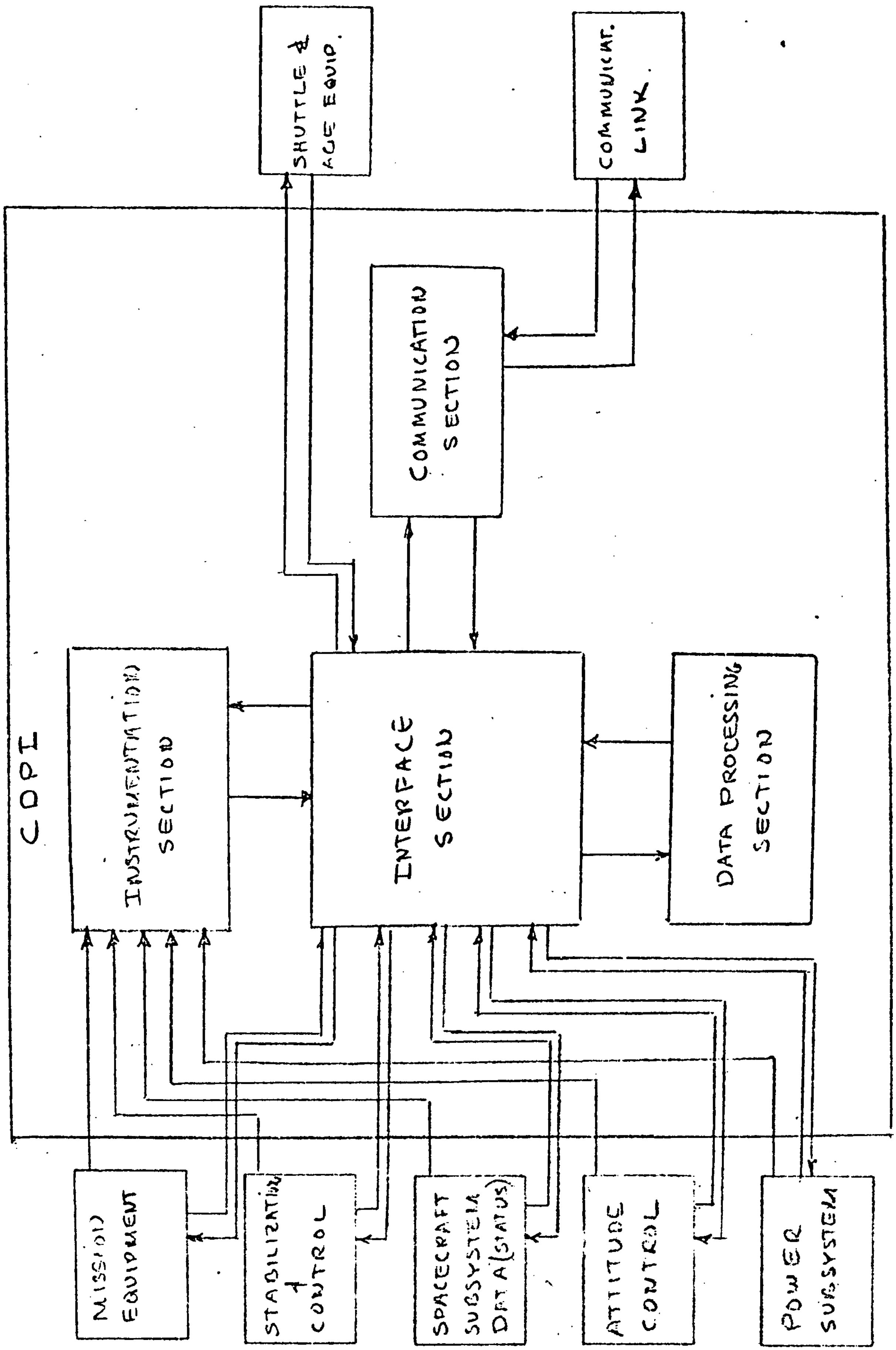

Fig. 8 Communication, Data Processing, Interface & Instrumentation Subsystem

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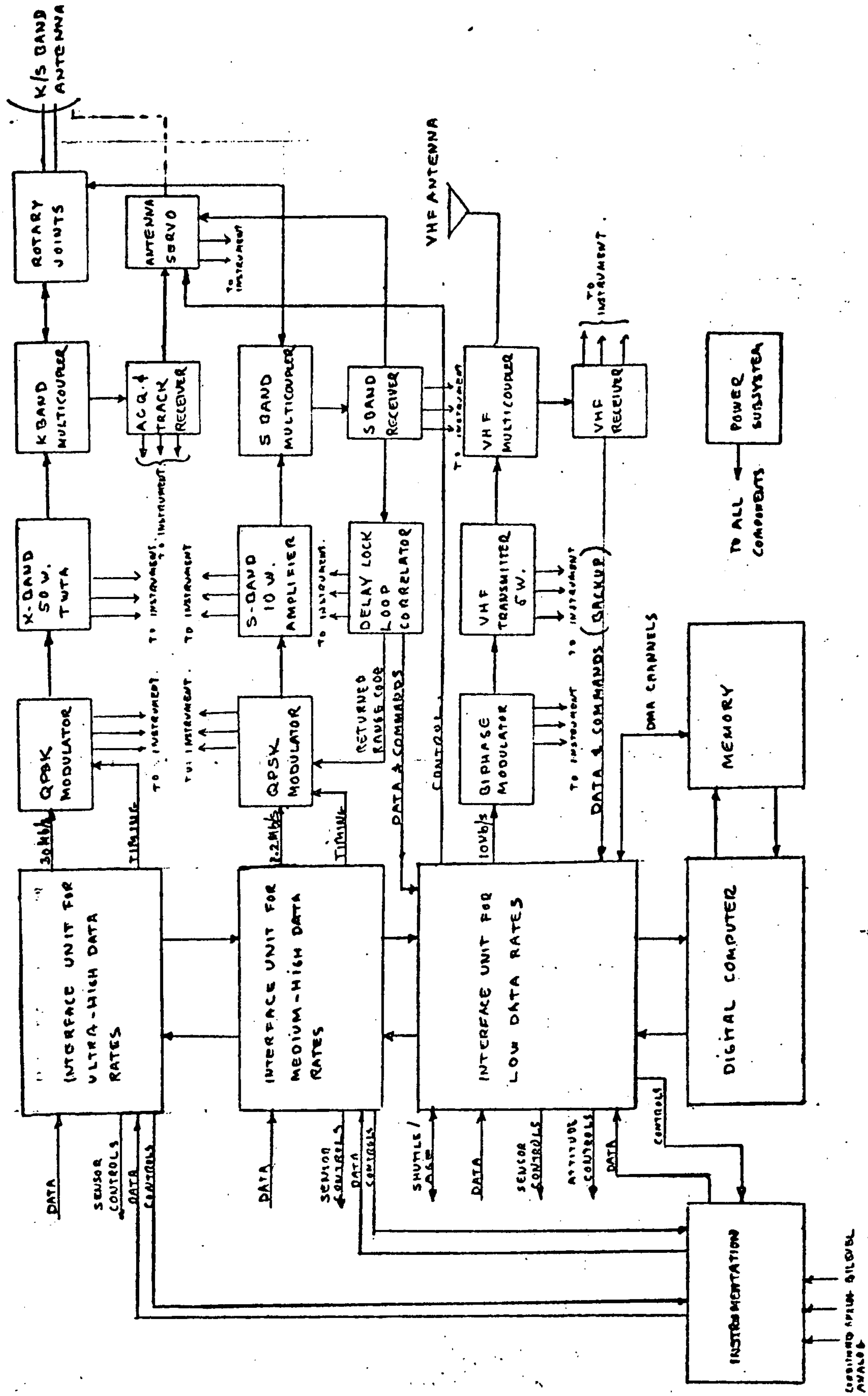

Fig. 9 Standard EOS CDPI Communication Section

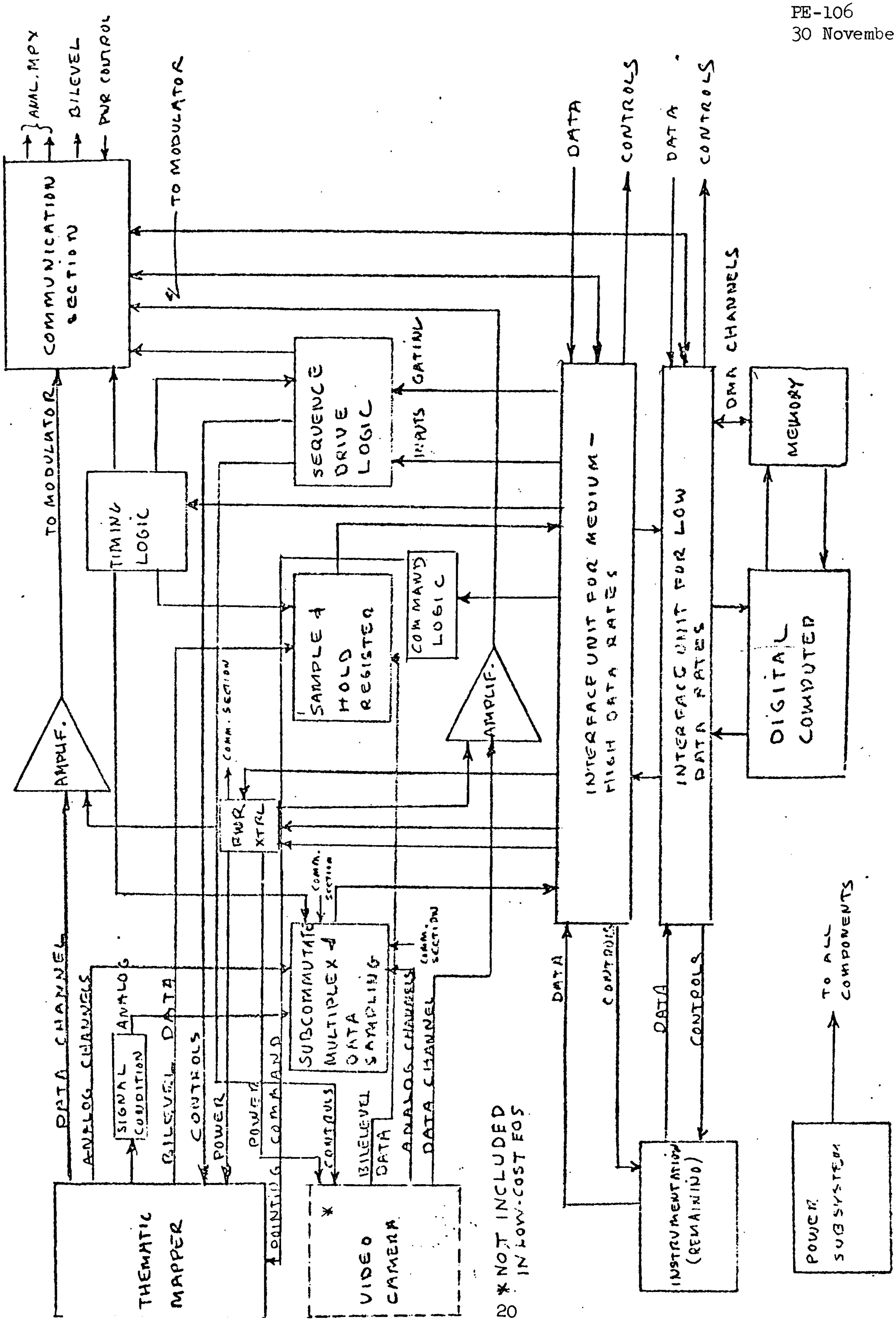

Fig. 10 Standard EOS CDPI Ultra-High Data Rate Interface Unit

source for the timing is in the Interface Unit for the low data rate sensors.

Additional channels have been incorporated in the figure to illustrate the hardware impact on the Interface Unit by a video camera. Although this design does not include a video camera, the extra capability is added on to the Interface Unit to allow for future growth.

Medium-High Data Rate Interface Unit

A block diagram of the second unit in the Interface Section is shown in Fig. 11. The purpose of this unit is four-fold:

- provide an interface with medium-high data rate mission equipment and the S-band portion of the Communication Section.
- supply multiplexing for medium-high data rates.
- perform subcommutative multiplexing, control sequencing, etc., on mission equipment instrumentation and operation.
- act as an interface to both the ultra-high and low data rates Interface Units.

Except for the number of instrumentation and control signals and the employment of broadband amplifiers, the primary difference between the ultra-high and medium-high Interface Units is the inclusion of high data rate multiplexing.

The multiplexer combines the data channel outputs from three mission equipment sensor units. In order to remain within S-band data rate handling capabilities, pre-multiplex buffering is incorporated in the sensor packages. Signal conditioning is contained in each sensor data channel within the Interface Unit to enable matching sensor output to multiplexer input requirements and provide a means for adapting the circuitry to a change in the mission equipment.

Low-Data Rate Interface Unit

A block diagram of the Low Data Rate Interface Unit, the final component of the Interface Section, is shown in Fig. 12. This unit is divided into three parts. The first is devoted to S&C interfacing functions, e.g., counters, amplifiers, gates, etc; the second serves as the focus for instrumentation and mission equipment functions, e.g., main analog multiplexing and A/D conversion, timing reference, etc. The third part is the digital computer external input/output interface containing input/output registers, data channel controls and logic.

The instrumentation and mission equipment functions are similar to those supported by other Interface Units, except data rates are much lower. The primary point of departure is the part containing computer input/output interfacing (asterisked Modes in Fig. 12).

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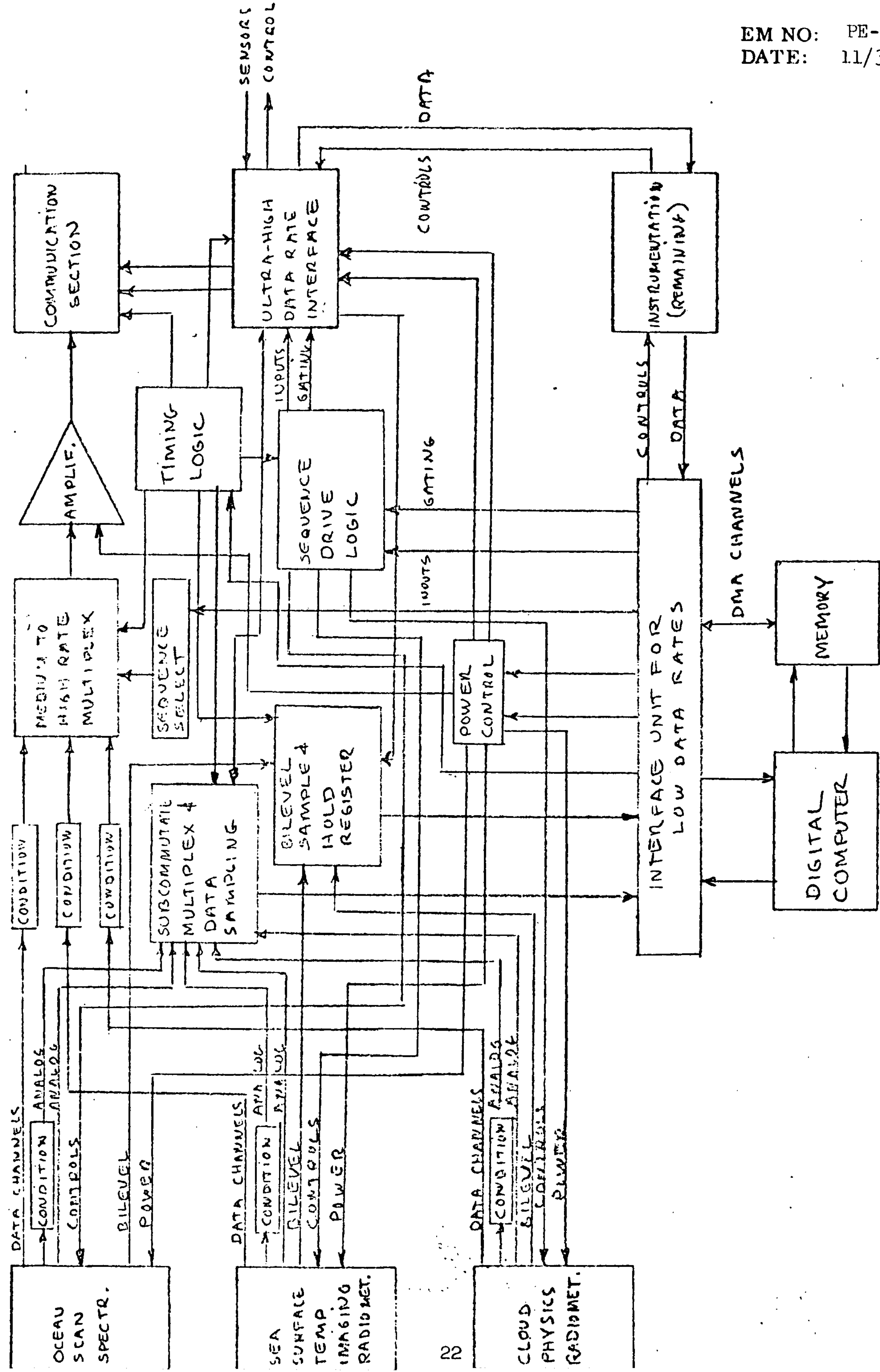

Fig. 11 Standard EOS CDPI Medium-High Data Rate Interface Unit

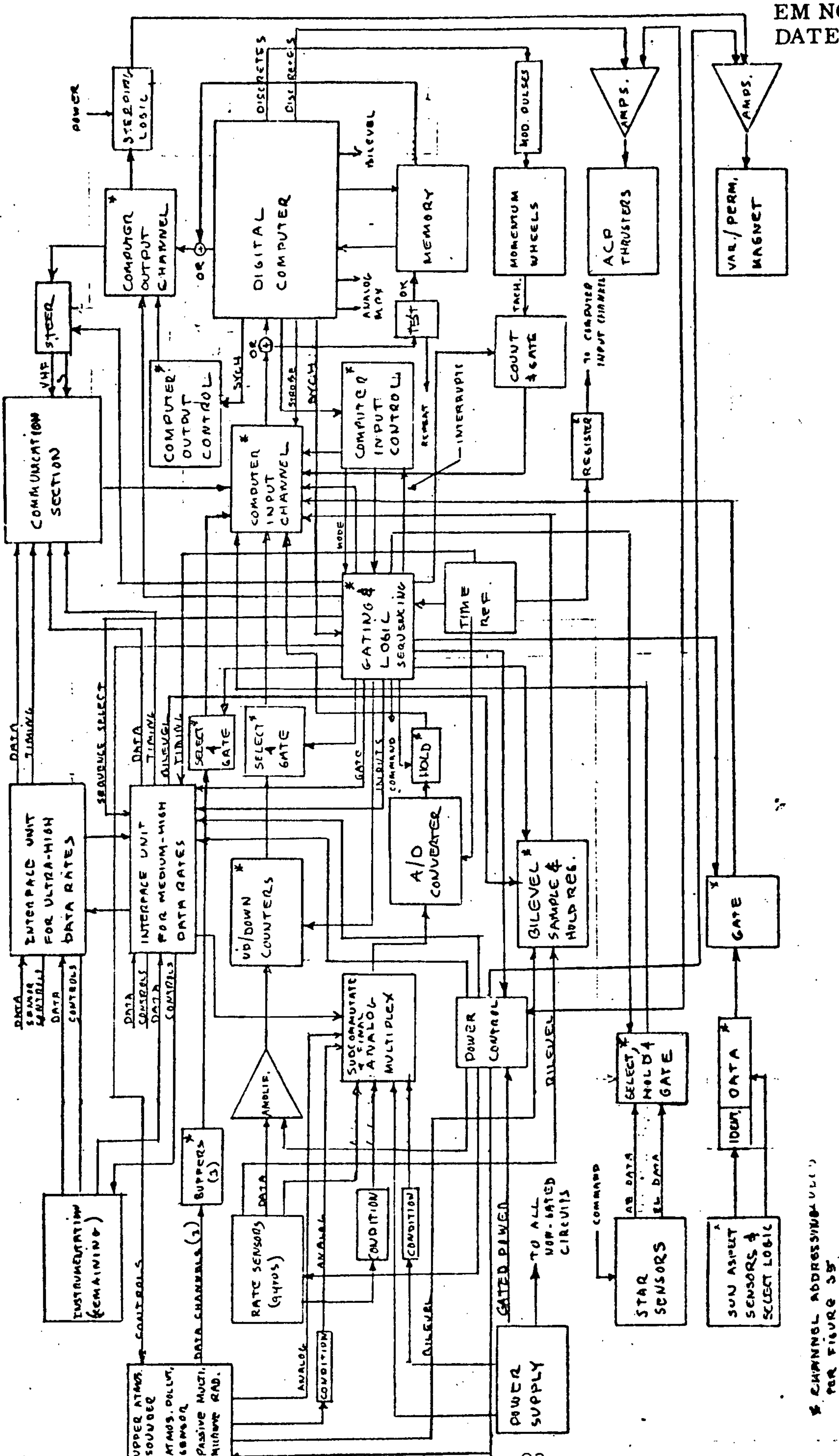

Fig. 12 Standard EOS CDPI Low Data Rate Interface Unit

Data Processing Section

A block diagram of the Data Processing Section and its interconnections with other CDPI sections and Interface Units is shown in Fig. 13. This section serves the computational, on-board decision making, sequencing, control, command verification, storage and formatting processes identified in the CDPI Functional Diagram, Fig. 14.

Software

If the functions of the CDPI, delineated on the Functional Diagram (Fig. 14) are examined it is seen that many may be realized through computer processing software; instead of by hardware implementation. This pertains in particular to S&C computational processes, logic control of the equipment serviced by the CDPI, command verification and processing and low data rate telemetry sequencing and formatting. Other functions, e.g., control sampling of sensor data, data processing support of high data rate sensors through gridding or other techniques, etc., may also be implemented.

In the design of the CDPI subsystem described in preceding pages, utilization of software is maximized to obtain the advantages of equipment commonalities. Other benefits include the significantly increased flexibility and relative ease of making changes in system function or operation. Also, depending upon what is actually done, ground computation and data handling may be considerably reduced.

The CDPI subsystem equipment is packaged in four modules that are designed to be removed from the spacecraft in-orbit and replaced by equivalent modules without calibration or mechanical alignment. The locations of the CDPI modules in the EOS are shown in Fig. 2 and the major equipment contained in each module is listed in Fig. 3.

4.4 Electrical Power Subsystem (EPS)

The proposed EPS consists of a fixed (non-tracking) solar array, 6 nickel-cadmium batteries which are charge-controlled by array switching, and a DC-DC regulator. The system provides a nominal 28 volt bus which varies from 25.0 to 28.0 volts. The regulator will provide 28.0 VDC ± 2% for the equipment needing close regulation. A functional block diagram is shown in Fig. 15.

Large battery capacity (more weight) with its attendant low depth-of-discharge, makes elaborate charge rate controllers unnecessary. (At a given temperature, battery life is inversely proportional to depth-of-discharge). Array open circuit switching conveniently dissipates excess power as heat out on the large solar array area (by not converting to electrical power). This eliminates the large bank of heat dissipating resistors associated with shunt regulators. These resistors are traditionally difficult to locate because of the large quantity of heat to be dissipated. The batteries are on the bus full time, establish the array operating point, and serve the same function as a shunt regulator, thus making the system a direct energy transfer system for the unregulated power. They also completely absorb the array cold-to-hot high voltage transients that are characteristic of array-on-bus and battery switching systems.

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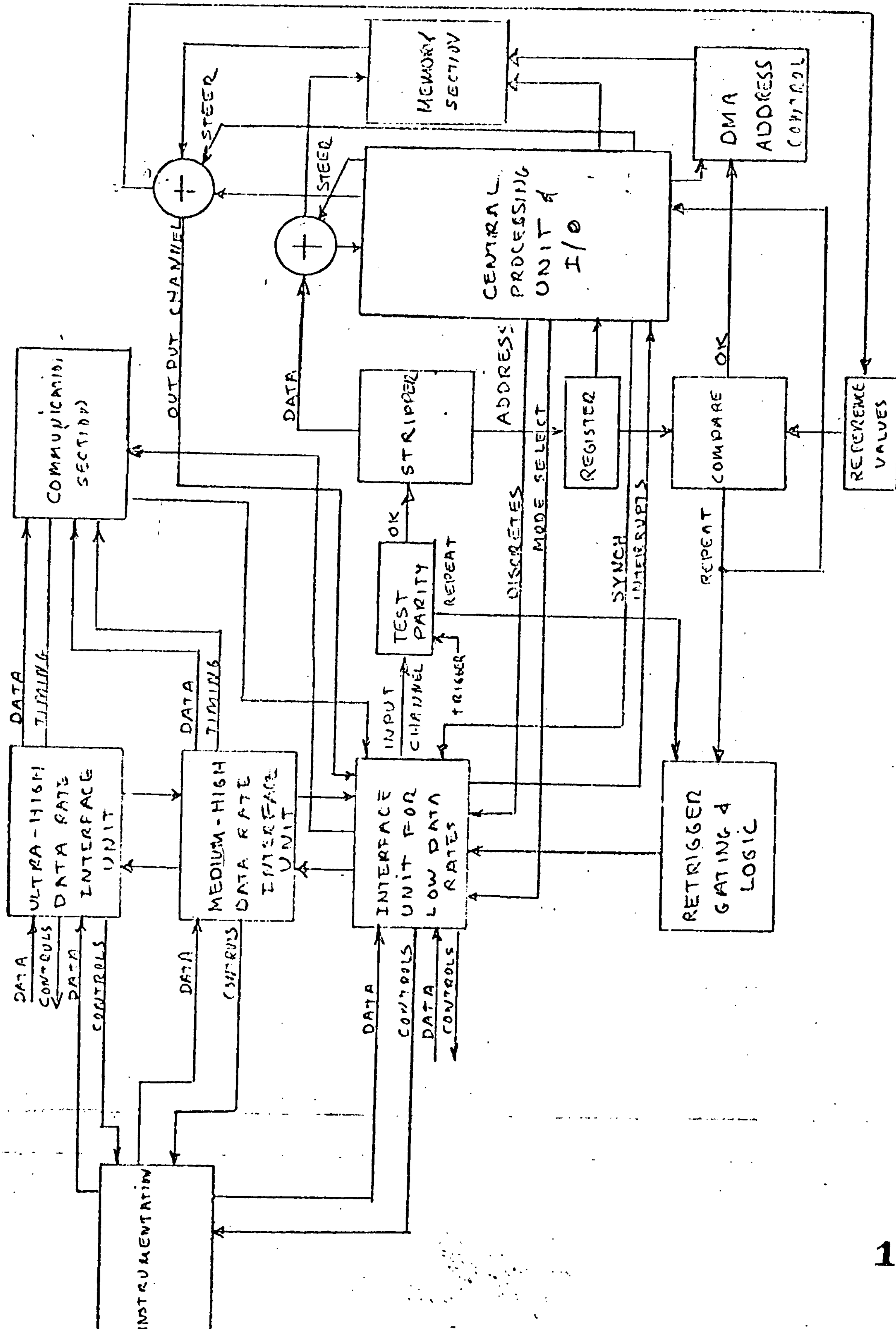

Fig. 13 Standard EOS CDPI Data Processing Section

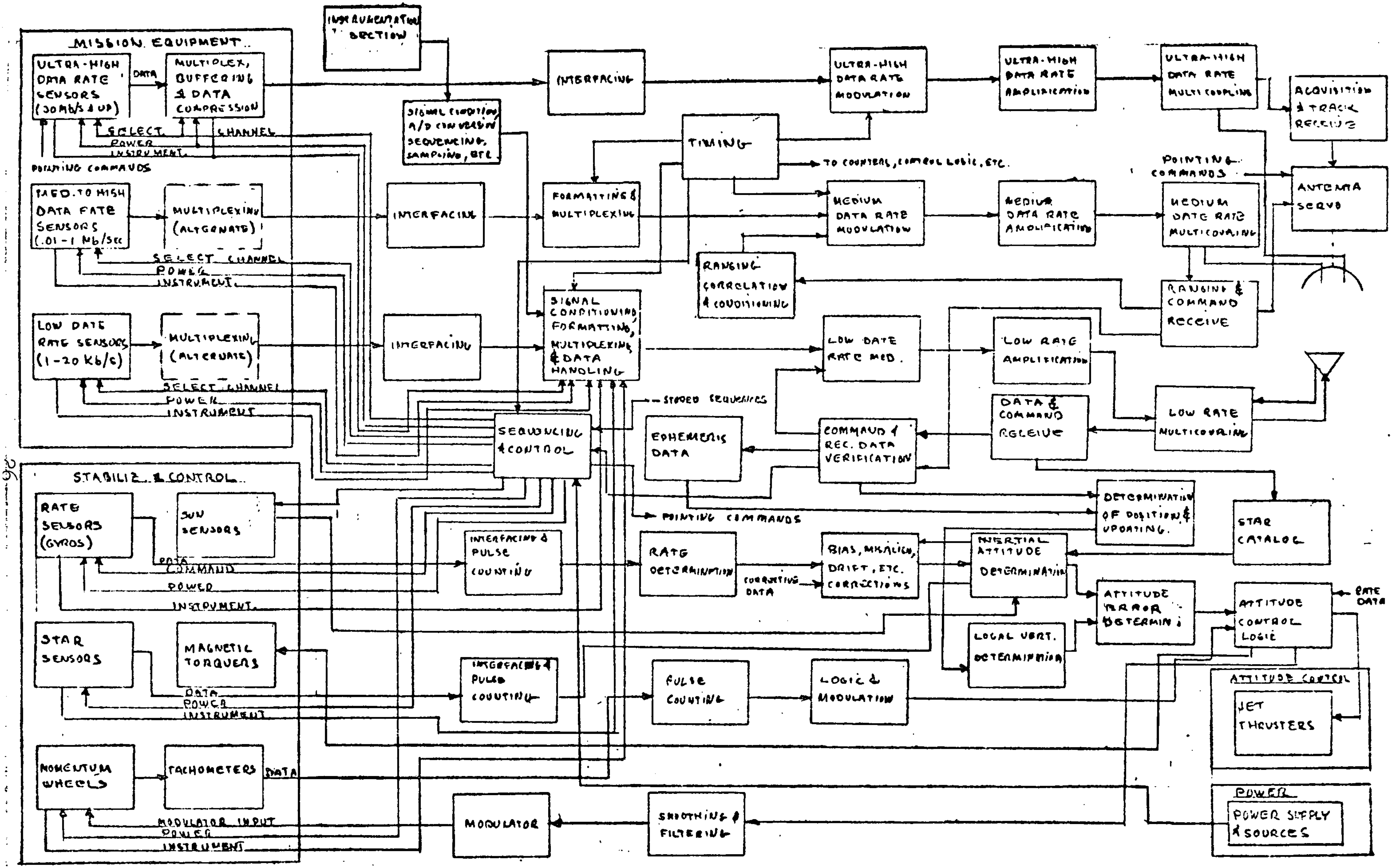

Fig. 14 Standard EOS CDPI Functional Diagram

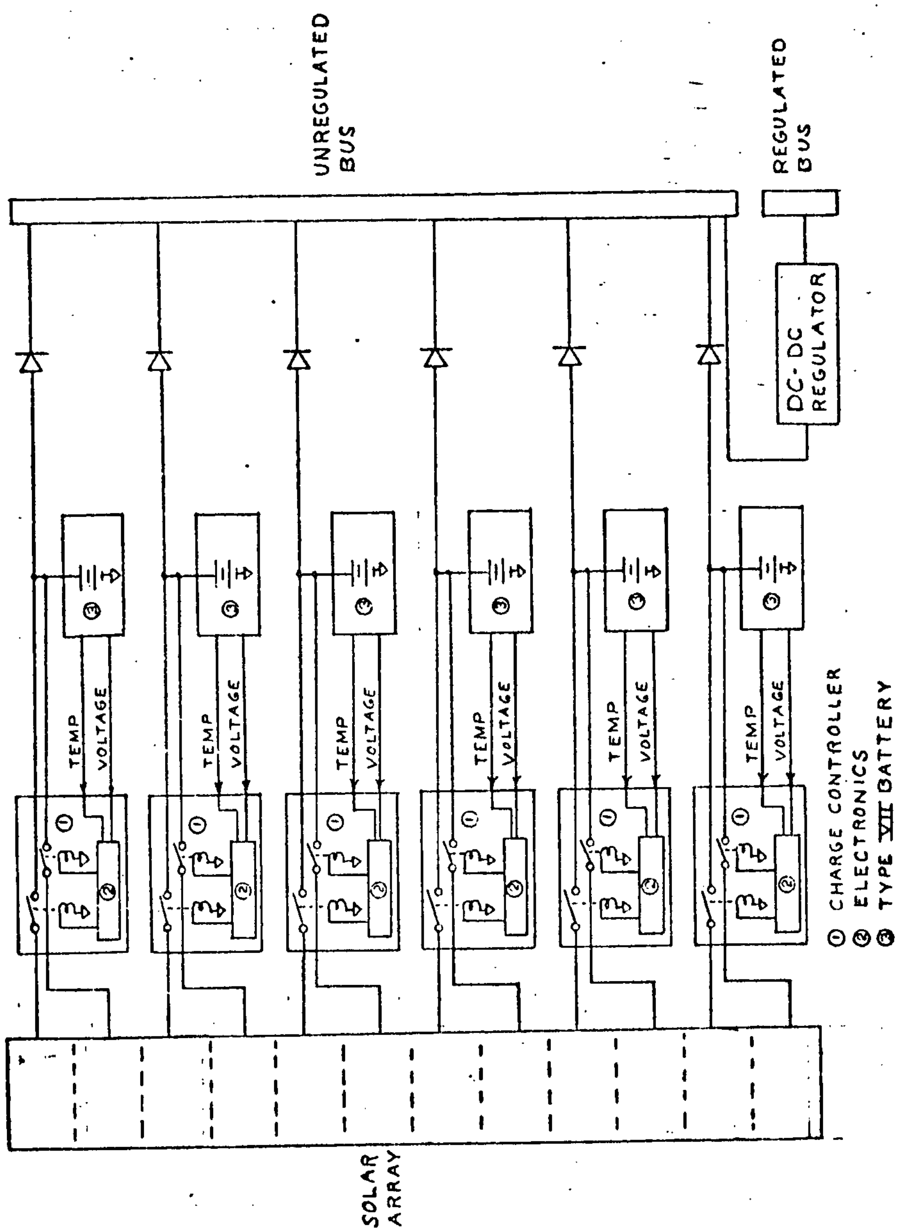

Fig. 15 Standard EOS Electrical Power Subsystem

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An array angle adjustment before launch compensates for β-angles other than 0 degrees.

Solar Array

The beginning-of-life solar array output necessary to support equipment loads of 1000 watts average over a 2 year period is 1155 watts.

The solar array consists of 80 panels mounted on 2 fold out wings as shown in Fig. 1. Each array wing is mounted on a shaft so that the wing can rotate $\pm 45^{\circ}$ with respect to the spacecraft body. This is to compensate for the $\pm 45^{\circ}$ β-angle. The angle will be set before launch. Since the β-angle does not change throughout the mission because the orbit is sun synchronous, this one adjustment is adequate throughout the mission.

The array is retractable in case the spacecraft is returned to earth.

A fixed array was selected instead of a tracking array because of lower overall costs and much higher reliability was gained at the expense of increase weight and array area.

The solar array is made up of 80 panels each 31.8 inches by 21.6 inches and consisting of an assembly of cells, coverglasses, and etched circuits on a Kapton base, which is bonded to a .080 inch thick aluminum sheet. The cells are 2 by 4 CM, N on P, .012 inches thick, 2 ohm-CM base resistivity, with wrap-around contacts. 97.5 percent of the production cells (after mechanical screening) will be used instead of the usual 66.7 percent yield. This reduces cell costs by 30 percent with a 2 percent increase in the number of cells needed (2 percent weight increase). The coverglass is .020 inch fused silica for maximum radiation protection (less cells) and less handling breakage (higher yield). The conventional-booster systems typically use .012-.014 inch thick coverglass for the weight advantage.

The electrical arrangement of the cells on a panel will be 69 in series by 7 in parallel, making a total of 483 cells per panel.

Batteries

Nickel-cadmium batteries were chosen because of the long life requirements. The 40 ampere-hour capacity is necessary to keep a low depth-of-discharge (DOD) to maximize the battery life. An average 14 percent DOD has been calculated for a 40 amp-hr battery and the load.

Charge Controllers

Each battery has a charge controller which connects or disconnects a section of the array to the battery. The voltage "tail-up" as the battery nears full charge is used as the signal to cut off charge. This voltage level is modified by the battery temperature which is sensed by transducers in the battery and fed into the charge controller. The control is done in two stages. As the battery proceeds towards full charge, first one-half of its array section is turned off, and then if the battery voltage continues to rise another one-half volt, the other half of its array section is disconnected. In this way the array output can often balance the load with little or no battery cycling, except for the normal night-time cycling. Conversely as the battery voltage decreases, the array section is connected in two stages.

Power Distribution Unit

This unit distributes the power to the various using equipments and also contains fuses, current sensors, and power system telemetry conditioning networks as required.

DC-DC Regulator

A conventional DC-DC regulator will be selected when the regulated power needs are finalized.

The Electrical Power Subsystem equipment is packaged in six modules that are designed to be removed from the spacecraft in-orbit and replaced by equivalent modules without calibration or mechanical alignment. The locations of the EPS modules in the EOS are shown in Figs. 1 and 2 and the major equipment contained in each module is listed in Fig. 3.

4.5 Attitude Control Subsystem (ACS)

The Attitude Control Subsystem (ACS) provides thrust for stabilization, reaction wheel unloading and backup, and emergency attitude hold for at least one month. The ACS consists of four identical modules installed on the outboard edges of the vehicle as depicted in Fig. 16. Each module contains four 1.75 lb rated thrusters oriented such that any three modules could provide 3 axis vehicle control. All four modules combined provide a total impulse of 6000 lb-sec, thereby providing for at least a 2-year orbital life.

Loss of one module early in life has no appreciable effect, if any, on reducing the 2-year orbit life.

Each ACS module consists of four major components: a gas storage tank, a fill valve, a pressure regulator and solenoid latching valve assembly, and a cluster of four thrusters. These major components are interconnected as shown in Fig. 17. The fill valve is used to load 38.5 lbs of Freon 14 gas into the stainless steel gas storage tank. The fill valve is opened or closed with a wrench and is capped for redundant leakage protection. The storage tank delivers Freon 14 gas to the inlet of the pressure regulator and valve assembly over a decreasing pressure range from 1750 to 180 psig as the gas is consumed. The pressure regulator and valve assembly contains an inlet filter rated at 40 micron absolute, a solenoid latching valve, a regulator, and downstream pressure relief valve with thrust nullifier. The regulator supplies regulated 120 ± 10 psia gas to the thruster cluster manifold. The solenoid latching valve controls tank pressure on or off to the regulator dome which in turn positions the regulator main flow poppet. This solenoid latching valve is moved to either open or closed by an 80 millisecond voltage pulse and has a position indicator which is monitored on TM. When the solenoid latching valve is open, gas pressure to the dome regulates outlet pressure to 120 ± 10 psia. If closed, the regulator main valve poppet closes and ceases to supply gas flow to the thrusters. Continued operation of the thrusters after the regulator main poppet is closed, results in the downstream line pressure decreasing

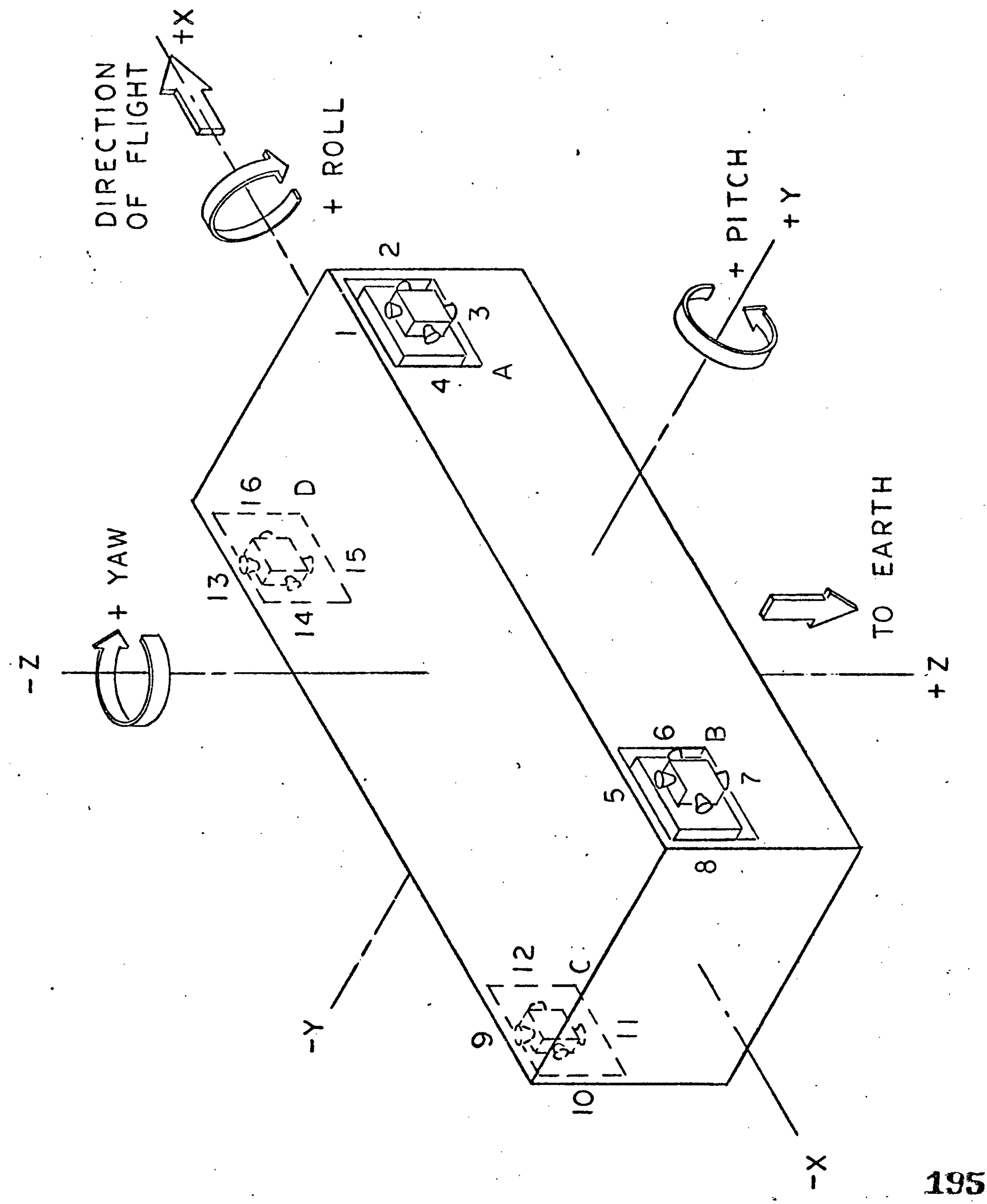

Fig. 16a StandardEOS ACS Thruster Orientation (1 of 2)

Vehicle Motion	Active Thrusters
+ Pitch	3 & 5 or 9 & 15
- Pitch	1 & 7 or 11 & 13
+ Roll	1 & 15 or 5 & 11
- Roll	3 & 13 or 7 & 9
+ Yaw	2 + 10 or 6 & 14
- Yaw	4 & 12 or 8 & 16

Couples are provided for all motions with one module disabled.

For Example Module A.

+ Pitch	9 & 15
- Pitch	11 & 13
+ Roll	5 & 11
- Roll	7 & 9
+ Yaw	6 & 14
- Yaw	8 & 16

Similarly couples are provided for each motion with Modules B, C, or D disabled

Fig. 16b Standard EOS ACS Thruster Orientation (2 of 2)

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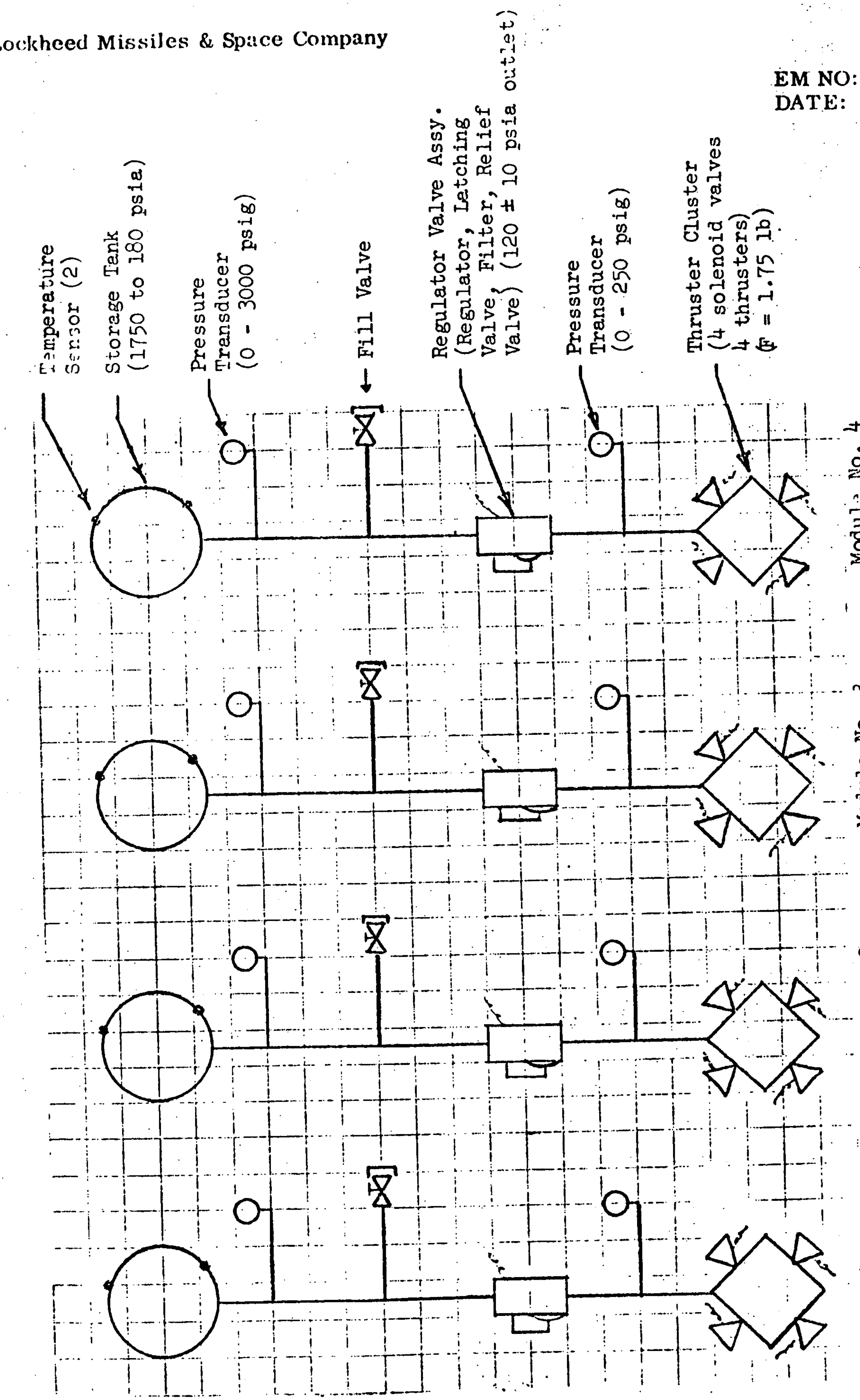

Fig. 17 Standard EOS Attitude Control Subsystem

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from 120 psia to 0 psia. The pressure relief valve is located downstream of the main flow poppet and vents should pressure exceed 160 psia due to a regulator malfunction or excessive main flow poppet leakage. This feature assures that the thrust valves will not be subjected to over-pressurization. The thrust valves are rated at 1000 psig burst.

Each of the four thruster assemblies in a cluster contains an electrical valve which is driven by the Attitude Control System Electronics. The thruster nozzle is sized to deliver 1.75 lb of thrust at a 120 psia inlet pressure. Application of a 20 millisecond electrical pulse to a pair of thrusters limits the ACS minimum impulse bit to 0.070 lb-sec.

Each module contains two pressure transducers, two temperature sensors, and a regulator valve indicating switch. A zero to 3000 psig range pressure transducer at the storage tank inlet/outlet and two temperature sensors attached to the external tank skin are provided for gas loading, orbital propellant mass statusing, and leakage detection. The zero to 250 psig range pressure transducer between the regulator and the thruster cluster provides a regulator outlet pressure check. It can also be used to check thruster valve leakage when the regulator main flow poppet is closed and the thrusters are inactive. Additionally, the safety of the system is verified by a pressure check prior to manned access.

The ACS module concept is shown in Fig. 18. The module is configured so that it may be used in any one of the four locations on the spacecraft, thus minimizing the number of spare modules required for in-orbit repair. The module is guided into its location on the spacecraft by rails and aligned and supported by two inboard pins and two outboard cams that engage machined grooves in the rails. The cams also transmit force from the cam actuators on the outboard face of the module to accomplish the controlled engagement and disengagement of the bulkhead-type electrical connectors on the in-board face of the module. The two wrap-around handles are designed to facilitate the handling of the module in orbit by a Space Shuttle crewman. This module is typical of the other subsystem modules.

4.6 Thermal Control

The thermal control of the EOS spacecraft will be accomplished primarily by passive thermal control techniques. To the extent possible thermal control of the spacecraft will be accomplished by the application of appropriate internal and external surface finishes and multilayer insulation. Equipment requiring temperature control within relatively narrow limits may require supplementary methods such as thermostatically controlled heaters. Some mission equipment such as the thematic mapper will require provisions for the cooling of sensors; however, such provisions are assumed to be part of the mission equipment.

4.7 Weight Summary

The weight summary for the standard EOS is shown in Fig. 19. The dry weight of the spacecraft including contingency is 6189 lbs. No additional contingency is included for the mission equipment. It is assumed that the total weight of the mission equipment, 1024 lbs, obtained from the Earth Observatory Satellite Definition Phase Report already includes appropriate contingency. It is estimated that packaging the mission equipment into modules for in-orbit removal and replacement would add 460 lbs to the dry weight of the spacecraft.

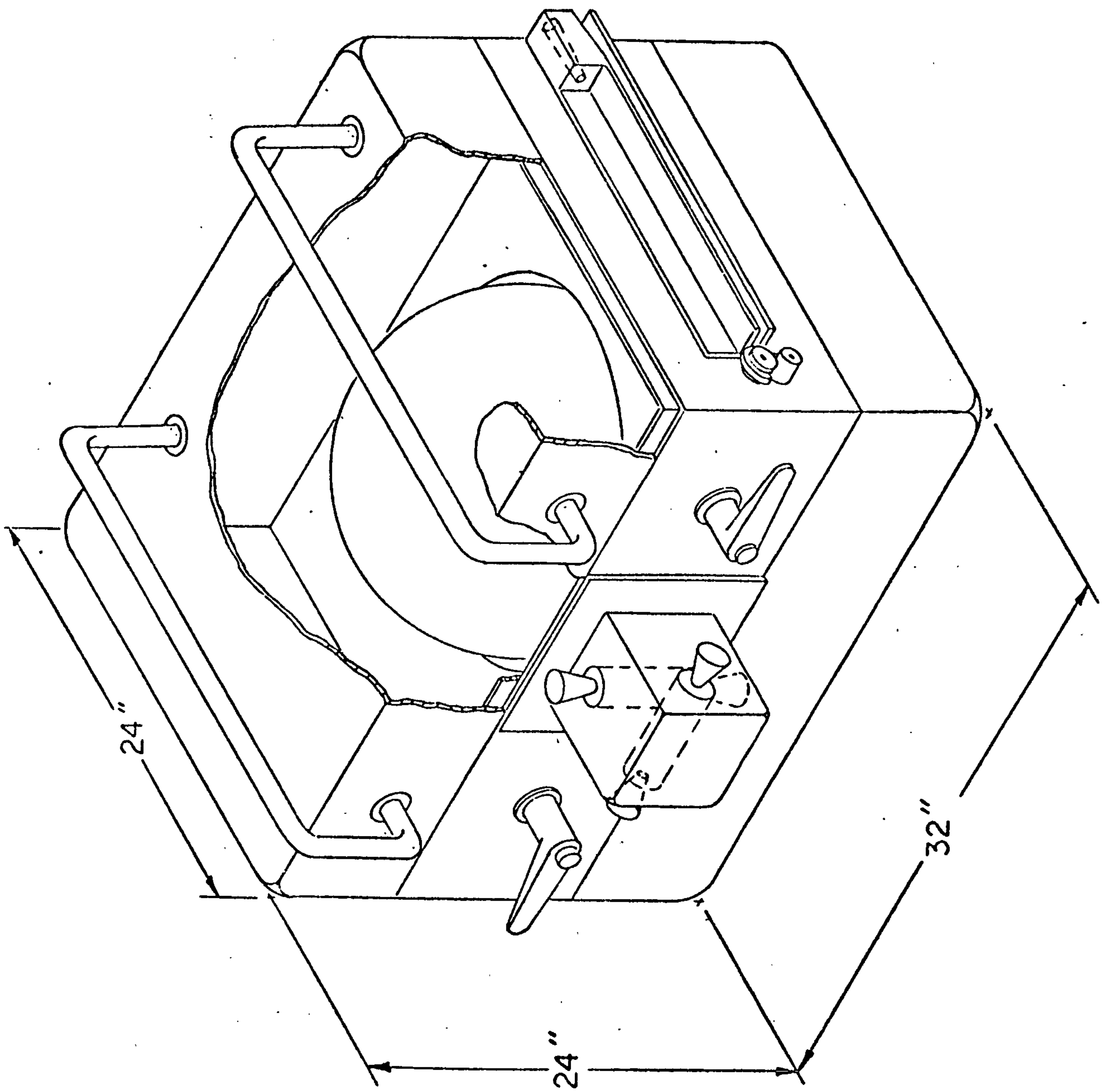

Fig. 18 Standard EOS Attitude Control Subsystem Module

Subsystem	Contingency	Weight
Structure & Mechanisms	15%	1660 lb
Environmental Control	15%	150
Stabilization & Control	15%	322
Communication, Data Processing, Inst.	15%	329
Electrical Power	20%	2132
Attitude Control	15%	404
Mission Equipment	Basic	1024
Mission Equipment supports, attachment hardware & electrical cables	15%	168
	Dry Weight	6189 lb
Propellant Freon 14		154
	Total	6343 lb

Fig. 19 Standard EOS Weight Summary

4.8 Reliability

The reliability goal for the standard EOS is 0.600 for one year of orbit life allocated as follows:

Payload	0.6000
Mission Equipment	0.9220
Spacecraft	0.6520
Structure	0.9996
Environmental Control	0.9996
CDPI	0.8640
Attitude Control	0.9970
Stabilization & Control	0.8500
Electrical Power	0.8890

The spacecraft reliability goal has been met by the designs described. More details of the designs and of the reliability analyses are contained in the referenced LMSC Engineering Memos.

PE-122

U. S. DOMESTIC COMMUNICATION

SATELLITE

STABILIZATION AND CONTROL

SUBSYSTEM

PE-122

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: U.S. DOMESTIC COMMUNICATION SATELLITE STANDARD SPACECRAFT - STABILIZATION & CONTROL SUBSYSTEM	EM NO: PE-122 REF: DATE: 24 Dec 1971
AUTHORS: R. J. Pollak — Prepared under cognizance of: Payload Integration, Orgn. 69-02 Space Space Systems Division	APPROVAL: J.C. Bolton ENGINEERING: Wayne Miller

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SUMMARY DESCRIPTION

The CSS stabilization and control subsystem has the following functions:

- To control attitude transients due to space tug separation
- To meet CSS pointing requirements (0.12 deg. roll and pitch, 0.5 deg. yaw) for five years
- To stabilize and control CSS attitude during east-west and north-south stationkeeping maneuvers and during space tug docking maneuvers
- To reorient CSS to the orbit reference attitude from any attitude following loss of reference for tumbling rates up to 50 deg/sec.
- To point the CSS spacecraft to the sun with near-zero rates following primary system failure.

The low cost CSS stabilization and control subsystem (Figure 1) 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 gimballed 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. During north-south stationkeeping, digital solar aspect sensors provide the yaw reference.

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

The general purpose digital computer in the CDPI subsystem provides timing, sequencing, and logical computations for the S&C subsystem. It accepts commands for real-time execution and for storage from the communications section. The processor section converts sensor signals and command messages into control logic and actuation signals.

The Comsat Standard Spacecraft S&C subsystem equipment complement represents one of the standard S&C subsysyem variants.

The double-gimballed reaction wheel control system (Figure 2) is a new concept now being applied to three-axis, long-life satellite attitude control. The system uses a 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 stabilization system requires no yaw sensor to perform its "normal" function, i.e., earth pointing.

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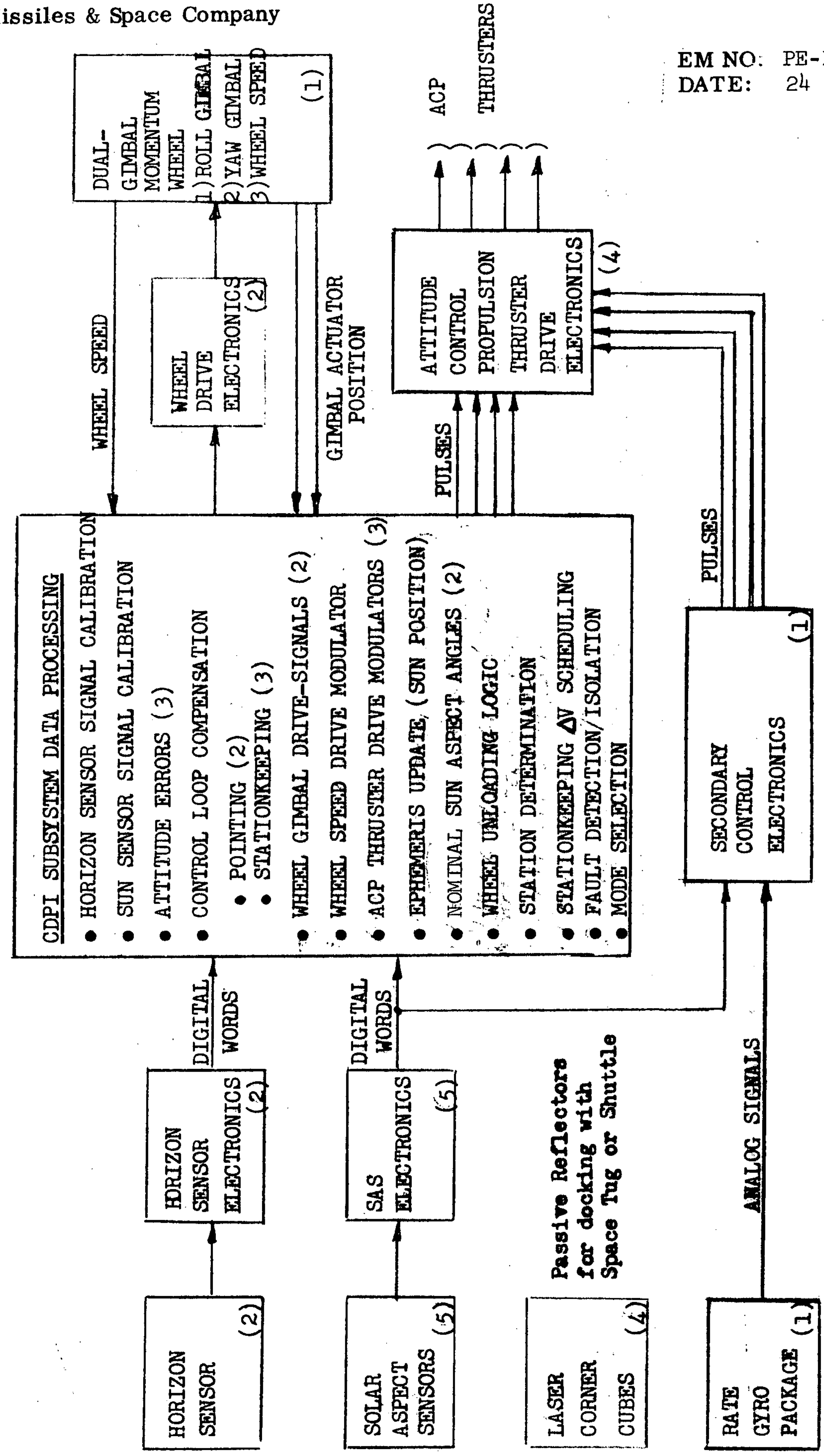

Fig. 1 COMSAT Standard Spacecraft Stabilization & Control Subsystem

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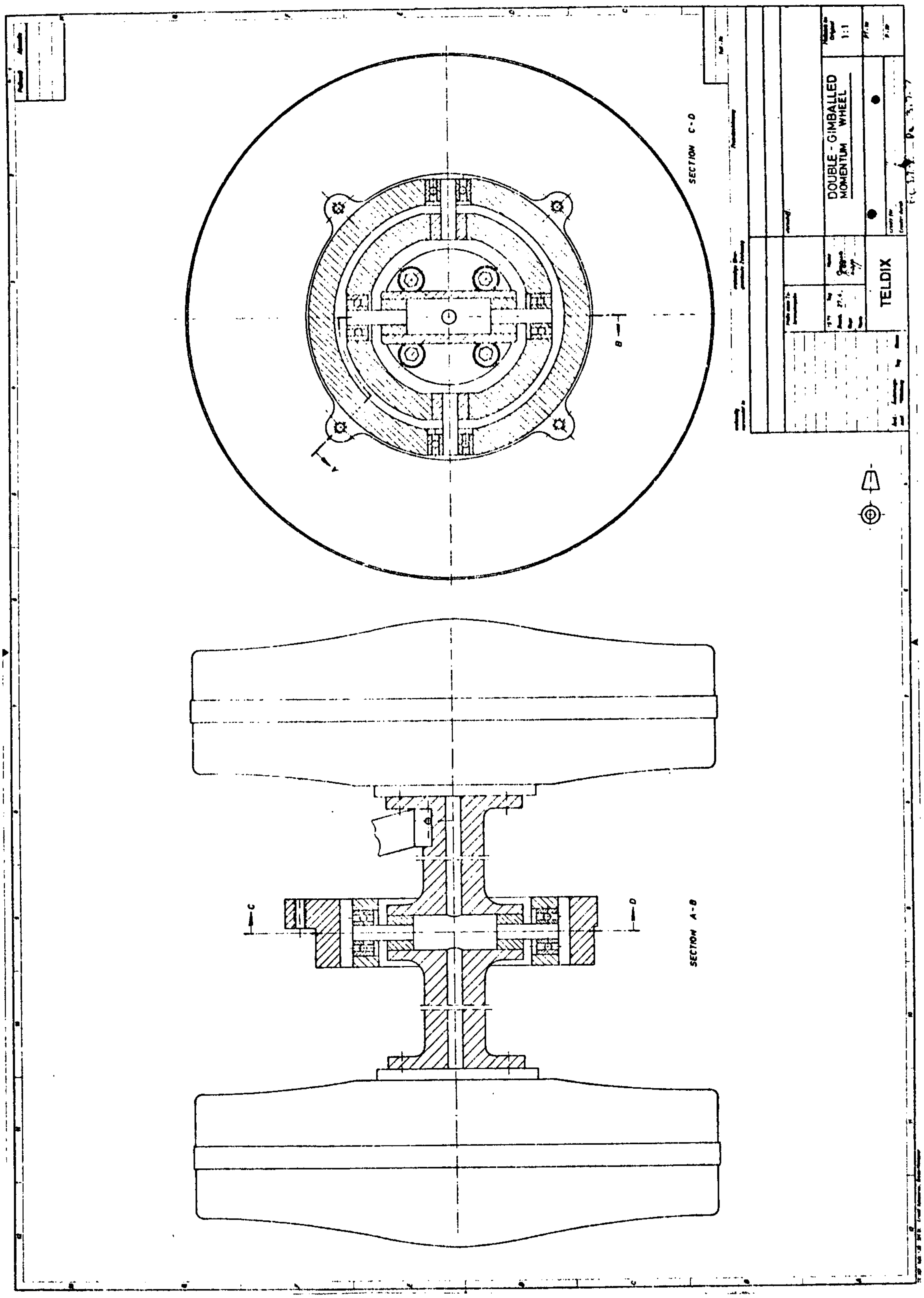

Fig. 2 Twin Momentum Wheel and Dual-Gimbal Arrangement

LIST OF ABBREVIATIONS

CSS	Comsat Standard Spacecraft
CMG	Control Moment Gyro
E, W	East, West
FOV	Field of View
GMW	Gimballed Momentum Wheel
GNC	Guidance, Navigation, and Control
HS	Horizon Sensor
IRU	Inertial Reference Unit
LOS	Line of Sight
R, P, Y	Roll, Pitch, Yaw
RGP	Rate Gyro Package
RSS	Root-Sum-Square
SAS	Solar Aspect Sensor
S&C	Stabilization and Control
TU	Test Umbilical
LCC	Laser Corner Cube

I. PROBLEM STATEMENT

The primary purpose of this report is to describe the Comsat Standard Spacecraft Stabilization and Control Subsystem in sufficient detail to permit the establishment of accurate cost estimates for CSS development, procurement and operations. The second use of this text is to supply design and interface information to other subsystem disciplines, specifically: attitude control, electrical power, spacecraft structure design and integration, communications and data processing, test operations, and reliability engineering.

A. Ground Rules and Constraints

The Stabilization and Control (S&C) subsystem must satisfy the following ground rules and constraints:

(1) Meet all CSS mandatory requirements

(2) Five-year orbit life

(3) Target subsystem reliability of 0.94

(4) 1971-1972 technology (concepts reduced to practice)

(5) Minimum cost: Trade weight increase for cost reduction

(6) Space Shuttle - launched, unmanned Space Tug syn-eq injection

B. Cost Factors

The following factors should be considered:

(1) The subystem will be checked out and operating at mission start (separation from the tug).

(2) The subsystem initial operating conditions can be provided by the Space Tug GNC Subsystem.

(3) Off-the-shelf hardware utilization, eliminates development and development test only - RDT&E costs for procurement, ground handling, qualification, checkout equipment, tooling, special test equipment and facilities must always be borne.

(4) Recurring cost elements include: Materiel, Manufacturing, Product Assurance, Acceptance Test, GSE/Tooling Maintenance and Sustaining Engineering.

(5) Operations cost elements include: ground transport, checkout, launch operations and orbit operations.

(6) Early non-recurring costs should be reduced at the expense of increased recurring costs (discounted dollars). Expected breakeven point is $5 recurring costs (increase allowed) for every $1 of non-recurring cost (saved).

C. CSS Operations Description

The Comsat Standard Spacecraft is an unmanned spsacecraft designed to provide telephone, telegraph, and television service to intra-U.S. users. The CSS will be placed into a near-equatorial, near-synchronous orbit by an umanned Space Tug. After drifting to the required station longitude (if not initially placed there) the spacecraft will supply the necessary east-west velocity corrections to attain synchronous speed and remain within ±0.1 deg. of station longitude and latitude for five years. Sufficient information for on-board determination of spacecraft location will be transmitted from ground tracking stations. This data will be processed into orbit adjust thrusting commands: direction, initiation time, and duration.

The spacecraft configuration (Figure 3) consists of an earth-oriented rectangular body and two single axis sun-tracking solar panels along the pitch axis. Principal spacecraft mass properties are:

Size:	8' x 6' x 10' excluding solar arrays
Weight:	4500 lb
MOI:	2800, 1500, 3500 slug-ft^2

D. Space Shuttle/Space Tug Interfaces (Reference 1)

The CSS will be designed to be compatible with the Space Tug and with the standard payload storage, deployment and retrieval structures and mechanisms of the Space Shuttle.

The CSS and Space Tug will be carried by the Space Shuttle to near-earth orbit. The Space Tug will then transport the CSS to synchronous equatorial orbit. The Space Tug will also be used to return the spent CSS from synchronous equatorial orbit to the Space Shuttle in near-earth orbit. Provision shall be made for docking the CSS to the Space Tug; and for deactivating the CSS Stabilization and Attitude Control Subsystems after docking latch-on is completed and prior to initiation of return to the Space Shuttle.

The CSS will be designed to be compatible with standard payload checkout equipment installed in the Space Shuttle cargo bay. Both pre-launch ground checkout and on-orbit checkout will be accomplished using this equipment. A checkout/monitoring station and a payload crew will be provided within the shuttle for man-in-the-loop operations. Interconnection between the CSS and the checkout equipment shall be by means of an umbilical cable (remotely released from the CSS at time of deployment by the Shuttle).

E. Specification Analyses

The moderate attitude accuracy, long lifetime, and large delta velocity requirements are factors which influence the selection of the stabilization and control concept and implementation. The need to provide the required functions at the least weight (consistent with low cost) is also a factor due to the limited payload capacity of the tug to synchronous altitude.

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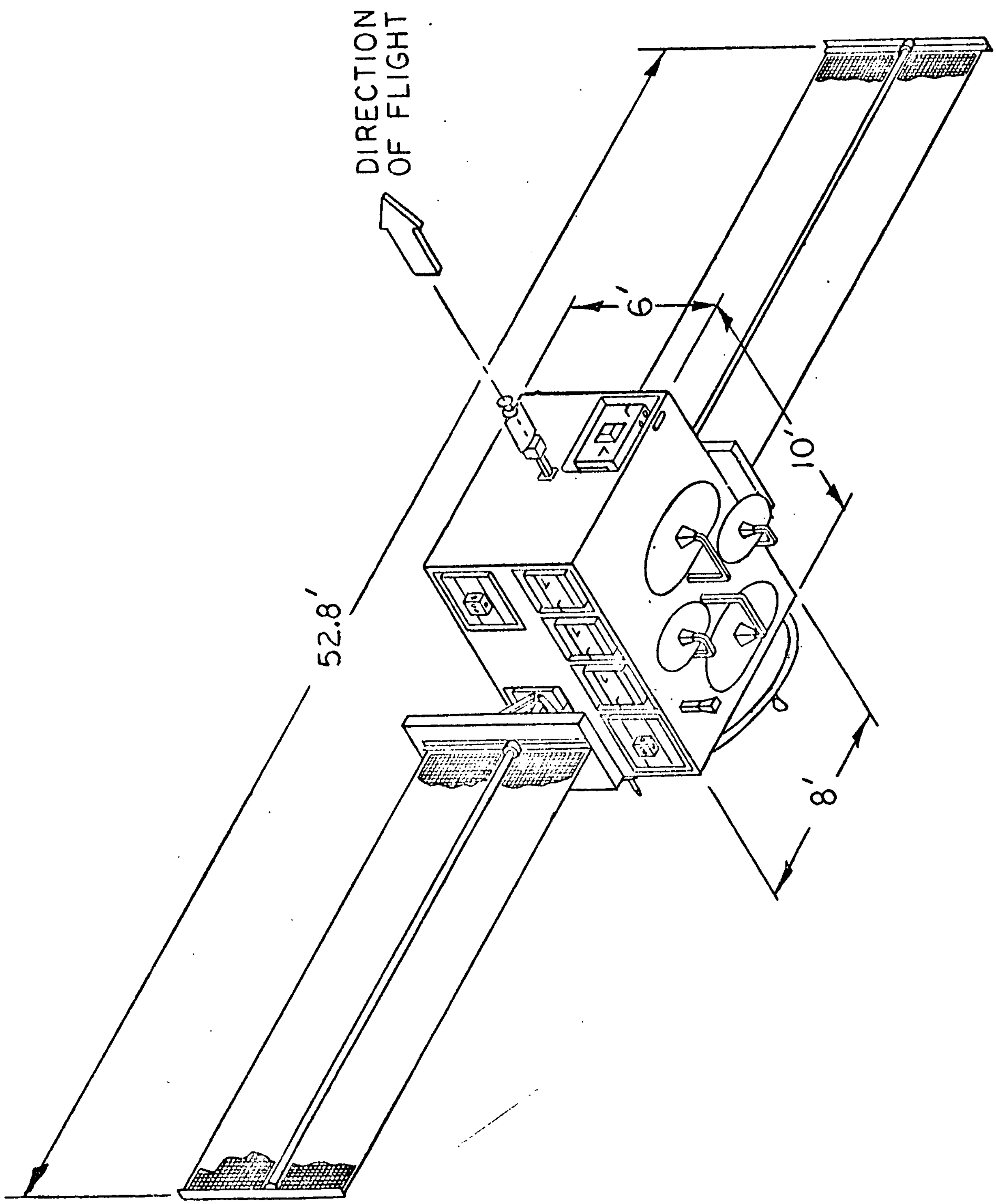

Fig. 3 COMSAT Standard Spacecraft

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These factors are summarized below:

Space Requirement	Impact on System Selection
• Moderate Accuracy	• Horizon Sensing • Pitch Momentum Bias
• Long-Life	• Twin Momentum Wheels • Redundant Horizon Sensors
• Large ΔV	• Sun Sensor for Yaw Reference • Thruster inverse modulation • Moderate Wheel Momentum
• Least weight (at same cost)	• Horizon Sensing • Pitch Momentum Bias • Sun Sensor for Yaw Control during north-south stationkeeping • Dual-Gimbal Momentum Wheel

II. SELECTION OF LOW-COST CSS S&C APPROACH

A. Functional Requirements

The subsystem must be able to perform in the following modes of operation.

Earth Pointing - the long term mode of operation during which precise earth-pointing is needed.

East-West and North-South Stationkeeping - short-term beam pointing during periods of linear velocity adjustment required to correct orbit parameters. The spacecraft returns to earth pointing mode at the conclusion of this mode.

Docking - same requirements as stationkeeping except no velocity changes occur.

Reacquisition - returns the spacecraft to earth pointing orientation after some unpredicted phenomena causes the loss of attitude reference.

Back-up Pointing - maintains beam pointing after malfunction of some S&C component. Back-up modes may result in degraded performance.

Anti-Tumbling - restricts angular velocity about all three axes to zero or near-zero rates. May include preferred sun orientation. Backs-up Docking Mode.

System-type selection was based upon beam pointing mode considerations; it will be shown that implementation of the other modes is the same for any of the candidate control concepts.

An 0.16-deg beam pointing accuracy specification was picked in order to limit the power losses of the narrow beam communications antennas to acceptable levels. The pointing error of the antenna will be divided into attitude stabilization accuracy (0.12°) and antenna-sensor misalignment (0.11°), since alignment tolerances can easily be held to this level.

The 0.12-degree attitude control requirement can be met by a variety of stabilization system concepts. Furthermore, the attitude reference sensor, often a major contributor to pointing error, can be selected on bases other than accuracy.

B. Candidate Stabilization and Control Concepts

Stabilization systems which meet the beam pointing requirement have been qualitatively evaluated (see Appendix A) considering: long-life and reliability, consequences of failure, compatibility with other modes, cost, development status, and physical characteristics.

The candidate systems were divided into sub-classes: three-axis systems, which require a yaw attitude error sensor and two-axis systems, which do not. In the latter case, yaw stability derives from the gyroscopic stiffness supplied by angular momentum directed along the orbital rate vector (nominal body negative pitch axis).

The prospective systems evaluated were:

Three-Axis Systems

a) 3 reaction wheels plus 3-axis gas unloading
b) 3-axis gas only (no wheels)

Two-Axis (momentum) Systems

c) Spinning body plus 2-axis gas unloading
d) Fixed pitch wheel plus 2-axis gas unloading
e) Single-gimbal pitch wheel plus 2-axis gas unloading
f) Double-gimbal pitch wheel plus 2-axis gas unloading

All of these systems will require mass expulsion for momentum desaturation, docking, pointing reacquisition, stationkeeping, and wheel backup.

C. Conclusions and Recommendations

1) Primary System

The attitude control system selected uses a 100 ft-lb-sec double-gimballed momentum wheel. It will have the lowest cost and highest reliability of any of the candidates. It is also anticipated that a momentum wheel system will be lighter than a 3-axis sensing and control system using reaction wheels. The weight advantage of the momentum wheel system is a consequence of the relatively relaxed yaw accuracy requirement. The power required for the two classes of system would be essentially the same.

The double-gimballed wheel system is also preferred over the other momentum wheel candidates on the basis of performance due to its superior dynamics and much lower thruster duty cycle.

All the two-axis systems use the gyroscopic torque of the wheel to provide yaw stability. The primary differences between competing 2-axis sensing systems is in the mechanism used to stabilize yaw. The fixed wheel with thruster system uses the yaw component of control torque from offset thrusters to damp yaw motions. The single-gimballed system also depends on gas for damping, for momentum desaturation, and for the restoring torque to directly counteract the inertial-fixed component of solar for the restoring torque to directly counteract the inertial-fixed component of solar torque. The double-gimballed wheel, similar to the fixed wheel system, uses the gyroscopic stiffness of the wheel to resist inertial-fixed solar torques, does not exhibit a nutation motion or a significant yaw overshoot due to roll transients.

Wheel status, life, and reliability are acceptable for CSS application. Momentum wheel stabilization has been used in a number of NASA and military space programs. Wheels as small as 5 and as large as 300 ft-lb-sec have been developed and flown. Sperry-Rand, General Electric and Bendix currently have wheel systems under development and all three companies are actively engaged in the development of large momentum "control-moment gyros" for space systems.

Either AC or DC brushless motors are capable of meeting performance and life requirements. DC brushless motor electronics were developed to commutate electrically and eliminate brushes but the complexity of the controls is such that reliability and cost have not been competitive with the AC motor until recently due to advances in integrated circuitry and rotor position sensors.

The most likely wheel failure mode would be eventual loss of lubrication. Because of this, all recently designed reaction and momentum wheels go to considerable effort to assure adequate lubricant in the bearings. Since the wheels operate in a near-vacuum to reduce windage losses, labrinth seals are required to maintain an adequate vapor pressure in the bearing. Standard oil and grease film lubrication techniques yield an operating life in excess of five years and additional lubrication techniques are being developed which should extend life to ten years.

All of the reaction and momentum wheels built and presently under development use conventional ball bearings. There has been a cursory examination of air bearings and fluid bearings but associated high power requirements have so far kept them from being serious contenders. The primary emphasis has been to use existing bearings by carefully controlling cleanliness, lubrication and preload. The bearings must be sized to withstand the worst expected boost environment and the preload must be heavy enough to prevent unloading during this period. If the environment were extremely bad it would be necessary to clamp the wheel with some caging mechanism during launch to avoid the very large bearings and preloads required to survive. However, this will not be required for the CSS.

The stationkeeping mode impacts two-axis systems. Torque unbalance due to mismatched thrusters and center of mass uncertainty is typically orders of magnitude larger than the solar torque. Roll and pitch thrust misalignment torques are corrected by normal thruster modulation. Without active yaw attitude control, the precession torque resulting from rotating the wheel angular momentum would be the sole counterbalance to the yaw stationkeeping disturbance torque. To meet the yaw angle accuracy an excessively large wheel momentum size would be required, even with a daily north-south stationkeeping interval. To keep the wheel to a practical size, active yaw control using the solar aspect sensors for yaw sensing during north-south stationkeeping is recommended.

Reacquistion from any attitude and rates up to 4 deg/sec requires a damping source and a wide-angle orientation reference. While the damping function can, in principle, be fulfilled by the sun aspect sensor using relatively complex electronics, a 3-axis rate gyro package, using inexpensive, spring-restrained gyros, accomplishes the same purpose at less cost.

2) <u>Secondary (Anti-Tumbling) Control System (Figure 4)</u>

A significant portion of the anticipated cost savings associated with the use of the Shuttle derives from the planned return to the payload for maintenance, resupply, or recovery for refurbishment/repair.

It is axiomatic that it will be necessary to approach and grasp the satellite prior to performing any of these activities. While various special techniques and mechanisms can no doubt be conceived which will permit this if the satellite is tumbling randomly about any or all of its axes, it seems clear that it would be simpler, safer, and cheaper to assure that the satellite will be stabilized when the Tug arrives. This is a simple requirement to satisfy; the main decision required is to what degree will an independent capability be provided. The equipment required is:

1) 3-axis rate gyro package*
2) Coarse sun sensor*
3) Electronics package*
4) Gas jets (3-axis)
5) Gas supply
6) Command Receiver/Decoder
7) Battery

Ideally, these parts would be packaged as an entity, independent of the S&C subsystem. At the other extreme, no new parts would be added, and the functions would all be provided by the primary S&C equipment. In this case there would obviously be much less assurance of having a quiescent satellite since the electronics and gas jets are part of the normal system operation. The equipment included in the S&C subsystem described herein is marked (*).

The backup or anti-tumbling system would either be commanded "on" or automatic under some set of on-board logic. The sequence of events would be:

1) Rate Stabilization - rate gyros null all rates

2) Roll Search - Bias signal starts roll rate using ACS thrusters

3) Sun Acquisition - Sun in sensor FOV removes rate bias signal; sensor and rate gyros drive attitude to lock on sun at solar panels' $\beta = 0$.

4) Attitude Hold - Sun Sensor and rate gyros hold satellite to sunline. (Very slow rotation about sunline due to gyro threshold will exist.)

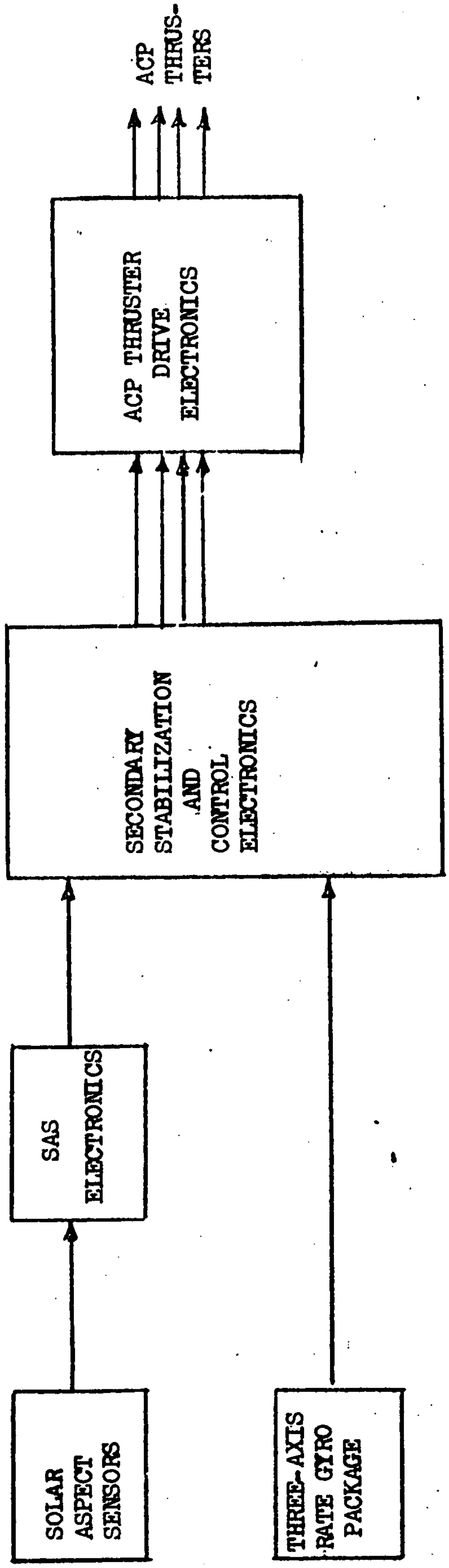

Fig. 4 CSS Secondary Stabilization and Control Subsystem

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III. DESCRIPTION OF S & C SUBSYSTEM

A. Design Approach

The first low-cost subsystem implementation iteration aims at satisfying mandatory functional requirements only; no backup modes are provided intentionally. Then the total capabilities inherent in the chosen mechanization are set down, including degraded backups. Finally, the design is augmented with the redundant equipment to meet the specified reliability/failure characteristic goals.

B. Operating Modes-Summary

The CSS S&C Subsystem operating modes are listed in Table 1. Five primary modes provide all required capabilities. The equipment used to implement the primary modes also provides a measure of degraded backup capability. These inherent backup modes are "degraded" in the sense of being less accurate, using gas at a higher-than-normal rate, and having higher thruster duty cycles. The low-cost S&C subsystem can continue and operation after the following failures:

Failure	Backup Mode
Wheel bearing, motor, etc.	Single wheel mode
Wheel Speed Control (Both)	Pitch Hot Gas
Wheel Gimbal Drive (Both)	Roll Hot Gas
Wheels (Both)	Pitch/Roll/Yaw Hot Gas

Two single-point failure modes: horizon sensing and wheel drive electronics are backed up by redundant equipment.

C. S & C Subsystem Implementation - Summary

The equipment needed to implement the low-cost CSS S&C subsystem is shown in Table 2, categorized according to the modes which each part supports. The S&C equipment is further described in Section VI. Table 3 shows the S&C subsystem equipment characteristics.

D. S & C Equipment Location

Figure 5 shows the installed locations and orientations of the S&C equipment in the CSS.

Most of the equipment is grouped into a pair of modules on the "earth" side of the spacecraft. The guidelines followed were:

(1) Grouping consistent with modularity guidelines (Section IV)

(2) Minimum number of modules

(3) Horizon and sun sensor fields-of-view unobstructed

(4) Electronics units close to sensors and actuators.

TABLE 1 - CSS S&C SUBSYSTEM OPERATING MODES

PRIMARY MODES

- Earth Pointing
- Momentum Wheel Unloading
- East-West Stationkeeping
- North-South Stationkeeping
- Docking
- Stabilization Reset

INHERENT BACKUP MODES

- Pitch Pointing (Wheel speed control backup)
- Roll/Yaw Pointing (Wheel gimbal control backup)
- Attitude hold (anti-tumbling/passive docking)

IMPLEMENTED BACKUP MODES

- Backup Horizon Sensing (Internal functional redundancy plus standby unit)
- Backup attitude control signal processing (Internal functional redundancy plus standby unit)
- Backup earth pointing (twin half-size wheels)

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Table 2

APPLICATION OF S&C EQUIPMENT TO OPERATING MODES

S&C PART	Primary Mode Use: Earth Pointing	Primary Mode Use: Momentum Unloading	Primary Mode Use: Stationkeeping	Primary Mode Use: Active Docking	Primary Mode Use: Stabilization Reset	Back-up Mode Use: Earth Pointing	Back-up Mode Use: Rate Null	Back-up Mode Use: Sun Pointing	Back-up Mode Use: Passive Docking
Horizon Sensor	X	X	X	—	—	X	—	—	—
General Purpose Digital Processor (CDPI)	X	X	X	X	—	X	—	—	
Momentum Wheel Assembly and Drive Electronics	X	X	X	X	—	X	—	X	X
Laser Reflector	—	—	—	X	—	—	—	—	X
3-Axis Rate Gyro Package	—	—	—	—	X	—	X	X	X
Sun Aspect Sensors	—	—	X	—	X	—	—	X	X
Reaction Jet Drive Electronics	—	X	X	X	X	X	X	X	X
Secondary Control Electronics	—	—	—	—	X	—	X	X	X

UNIT	QUANTITY (INCL. REDUNDANCY)	UNIT WEIGHT (LB)	TOTAL WEIGHT (LB)	PACKAGE SIZE	SUPPLIER	SUPPLIER PART NO.
Horizon Sensor (incl. HS elect.)	2	6	12	5x5x10	IMSC	5178178
3-Axis Rate Gyro Pkg. (incl. electronics)	1	3	3	4x4x2	Nortronics	79157-350
Sun Aspect Sensors (incl. electronics)	5	3	15	3x4x1 (2x5x4)	Adcole	15671, 2
Docking Reflectors	4	2	8	4x4x2	ITT	NAS 8-23973*
Gimballed Momentum Wheel (incl. electronics)	1	85	85	15Dx19	Sperry	2 - 45Q Wheels
ACP Drive Electronics	4	7	28	6x4x3	IMSC	New
Secondary control electronics	1	5	5	2x4x6	IMSC	New

Table 3 - CSS S & C EQUIPMENT LIST - I

*Contract Number

UNIT	QUANTITY (INCL. REDUNDANCY)	TOTAL POWER (WATTS)	DUTY CYCLE (%)	NO. OF ON-OFF CYCLES	FAILURE RATE (Per 10^6 Hr.)	UNIT COST ($K)
HS (incl. electronics)	2	10	100	1	4	40
RGP (incl. electr.)	1	15	1	2	10	6
SAS (Incl. electr.)	5	5	1	-	3	5
DR	4	0	-	-	-	0.25
GMW (incl. drive elect.)	1	16	100	1	0.7	200
ACPDE	4	12	$\ll 1$	1	3	10
SCE	1	5	1	1	10	25

Table 3 - CSS S & C EQUIPMENT LIST - II

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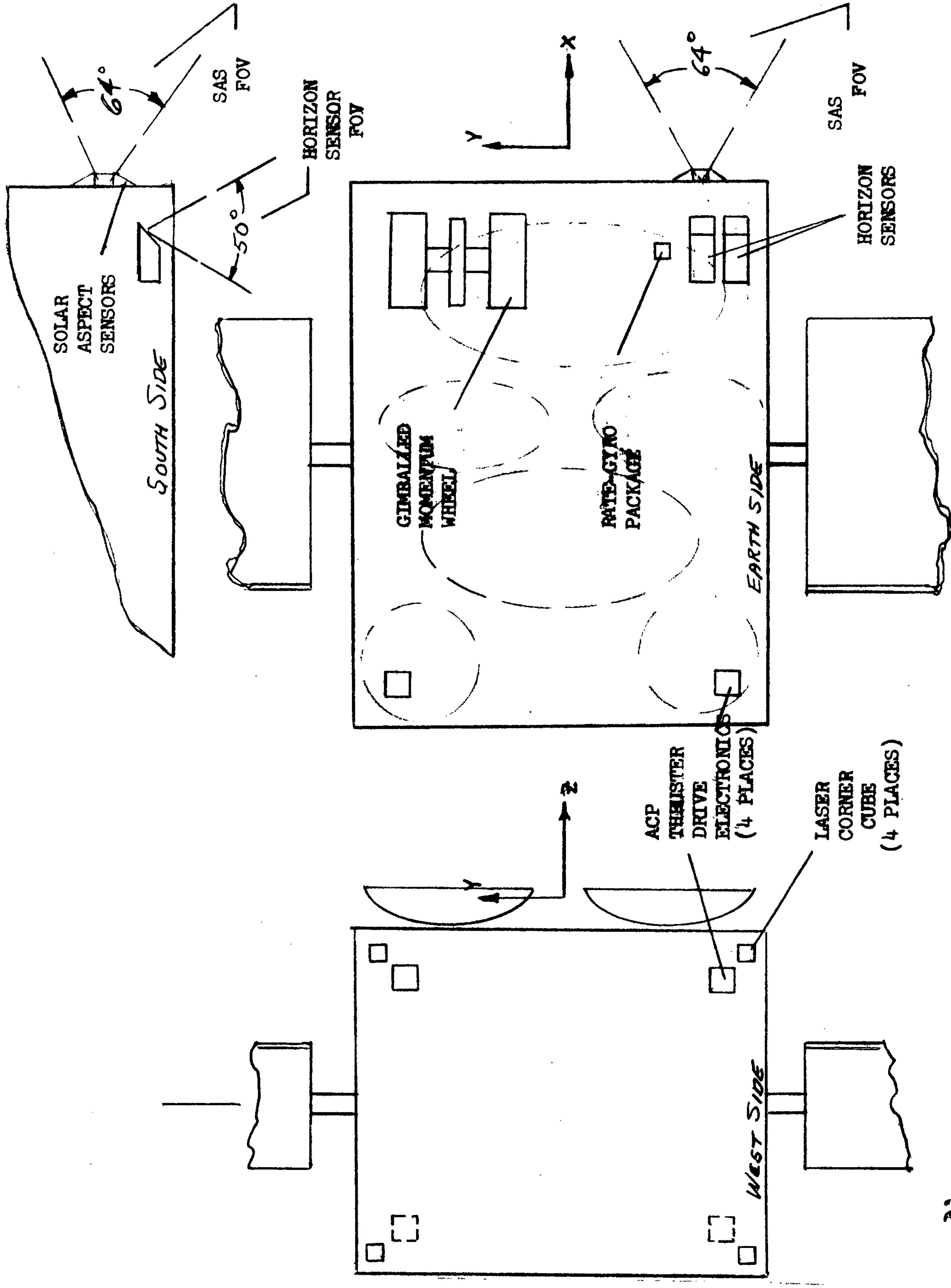

Fig. 5 Low-Cost CSS S&C Equipment

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E. Pointing Accuracy Budget (Table 4)

Although the required precision of pointing is not a critical factor, it is useful to show an error budget in order to learn where any effort should be placed to maintain or improve pointing accuracy and, conversely, where non-critical factors exist which permit easing of tolerances and attendant cost reduction.

The accuracies listed are conservative and require no special techniques or advancement in technology to attain. In fact, after a period of orbit data processing, it should be possible to reduce the bias errors.

F. S & C Subsystem - Special Characteristics which affect Development and Integration Costs

General

No special or unusual provisions are necessary to meet the dynamic or static accuracy requirements; alignment tolerances have been deliberately specified at levels which are easily attainable. No stringent manufacturing, handling, testing, or tooling problems are known to exist which would unduly impact subsystem development and integration costs.

Manufacturing

A rigid vertical mounting base is necessary for the momentum wheel to avoid amplification of the boost vibration environment and to maintain alignment within tolerances.

Testing

A dynamic test of the subsystem will be conducted using actual parts mounted on a three degree-of-freedom platform with a computer simulation of spacecraft dynamics. Such a test is normally conducted for all new spacecraft.

It may be necessary to provide protection against high-velocity fragments which could result from disintegration of a defective momentum wheel. This protection could be built into: (a) the wheel case (if specified),(b) the spacecraft structure, or (c) a special fixture.

G. Functional Operation

(1) Initialization -- After synchronous orbit injection, the Space Tug yaws and pitches to a horizontal attitude (tangent to the orbit) which orients the CSS in its operational attitude. The wheel is run-up, and a functional checkout of the Stabilization & Control Subsystem is performed with the Tug supplying attitude and rate stimuli. These tests, and those which follow CSS separation from the Tug, are further described in Section V.

Table 4 Pointing and Alignment Error Budgets

POINTING ERROR CONTRIBUTORS		EFFECTIVE POINTING ERROR (3σ)
Horizon Sensor Null		±0.03 deg
Horizon Sensor Linearity		±0.03
System Uncertainty (Quantization, etc.)		±0.03
$\pm0.7^\circ$ RSS Yaw Attitude Uncertainty		±0.07
Solar Pressure Torque	$\pm0.4^\circ$	
Stationkeeping Thrust	$\pm0.5^\circ$	
Wheel-to-Sensor Alignment	$\pm0.15^\circ$	
Attitude Control RSS Total		±0.09 deg
Seasonal Horizon Uncertainty		±0.03
Attitude Pointing Total		±0.12 deg
Antenna Beam-to-Horizon Sensor Alignment		±0.11
Antenna Pointing RSS Total		±0.16 deg

ALIGNMENT ERROR CONTRIBUTORS		EFFECTIVE POINTING ERROR (3σ)
• Horizon Sensor Alignment (RSS)		$\pm0.08^\circ$
Optical axis to Mounting Base	$\pm0.03^\circ$	
Mounting Base to Module	$\pm0.05^\circ$	
Module to Spacecraft	$\pm0.05^\circ$	
• Antenna Beam Alignment (RSS)		$\pm0.08^\circ$
Beam to Dish Boresight	$\pm0.03^\circ$	
Boresight to Mounting Base	$\pm0.05^\circ$	
Mounting Base to Spacecraft	$\pm0.05^\circ$	
• Horizon Sensor to Antenna Beam Alighnment		$\pm0.11^\circ$ (Pitch, roll)
• Wheel Alignment (RSS)		$\pm0.12^\circ$
H-Vector to Base	$\pm0.10^\circ$	
Mounting Base to Module	$\pm0.05^\circ$	
Module to Spacecraft	$\pm0.05^\circ$	
• Horizon Sensor to Wheel Alignment (RSS)		$\pm0.15^\circ$ (Yaw)

(2) Earth-Pointing Mode -- The double-gimballed momentum wheel attitude control system stabilizes the spacecraft to the orbital reference frame established by the radius vector from the center of the earth and the orbital angular velocity vector. The wheel operates as a biased reaction wheel with the nominal angular momentum pointing in the direction of the orbital angular velocity vector. Spacecraft pitch (rotation about the negative orbital angular velocity vector) is controlled by accelerating or decelerating the wheel in response to a pitch attitude error signal. Roll (rotation about the orbit velocity vector) is controlled by applying a reaction torque to the roll gimbal in response to a roll attitude error signal. Yaw (rotation about the negative radius vector) is controlled by gyro-compassing. Figure 6 shows the necessary data processing, carried out in the CDPI general purpose computer.

The pitch and roll rate and attitude gains are formed electronically through pseudorate modulators. The yaw rate and attitude gains are mechanically determined by the roll-to-yaw gimbal coupling angle and the angular momentum of the wheel times the average orbital angular rate (H_{W_O}) respectively.

This angular momentum control system has several unique features; it uses only one reaction wheel, requires only pitch and roll attitude information, and is steerable in roll, pitch, and yaw. The system does, however, require a two-degree-of-freedom movement of the wheel spin axis, but, since it utilizes a relatively large bias component of angular momentum, adequate control torques can be developed with small angle excursions permitting the use of flexure pivots for gimbals and flex leads for transfer of electrical power to the wheel.

A functional block diagram of the spacecraft attitude pointing control system is shown in Figure 7. The basic functions of the system are: (1) the roll, pitch and yaw attitude compensation, (2) the roll and yaw gimbal actuator error signals, (3) roll-to-yaw reaction control torque coupling which substitutes for the yaw sensor, (4) orbit motion decoupling (which computes the roll and yaw gimbal angle signals required for uncoupled reaction torques in a rotating reference control system), (5) the roll axis gyroscopic nutation decoupling signal, and (6) the pitch reaction wheel controller.

For the derived-rate functions, the pseudorate modulator is fairly simple and has good noise rejection properties. A pulsed type controller was selected rather than a continuous type because: (1) the electronics are simpler and take less average power, (2) a fast reaction wheel time constant can be obtained with less complexity, weight, and power, and (3) the orbit motion decoupler has better accuracy and drift characteristics with comparable complexity. The quantization errors of the integrators are minimized by being driven by pulsed signals approaching the maximum integration rate.

An inertially-fixed angular momentum vector not aligned to the pitch axis will cone about the orbital angular velocity vector as viewed from the rotating orbital reference coordinate frame. This means that the gimbal angles must oscillate at orbit frequency. In a frictionless, inertia-less

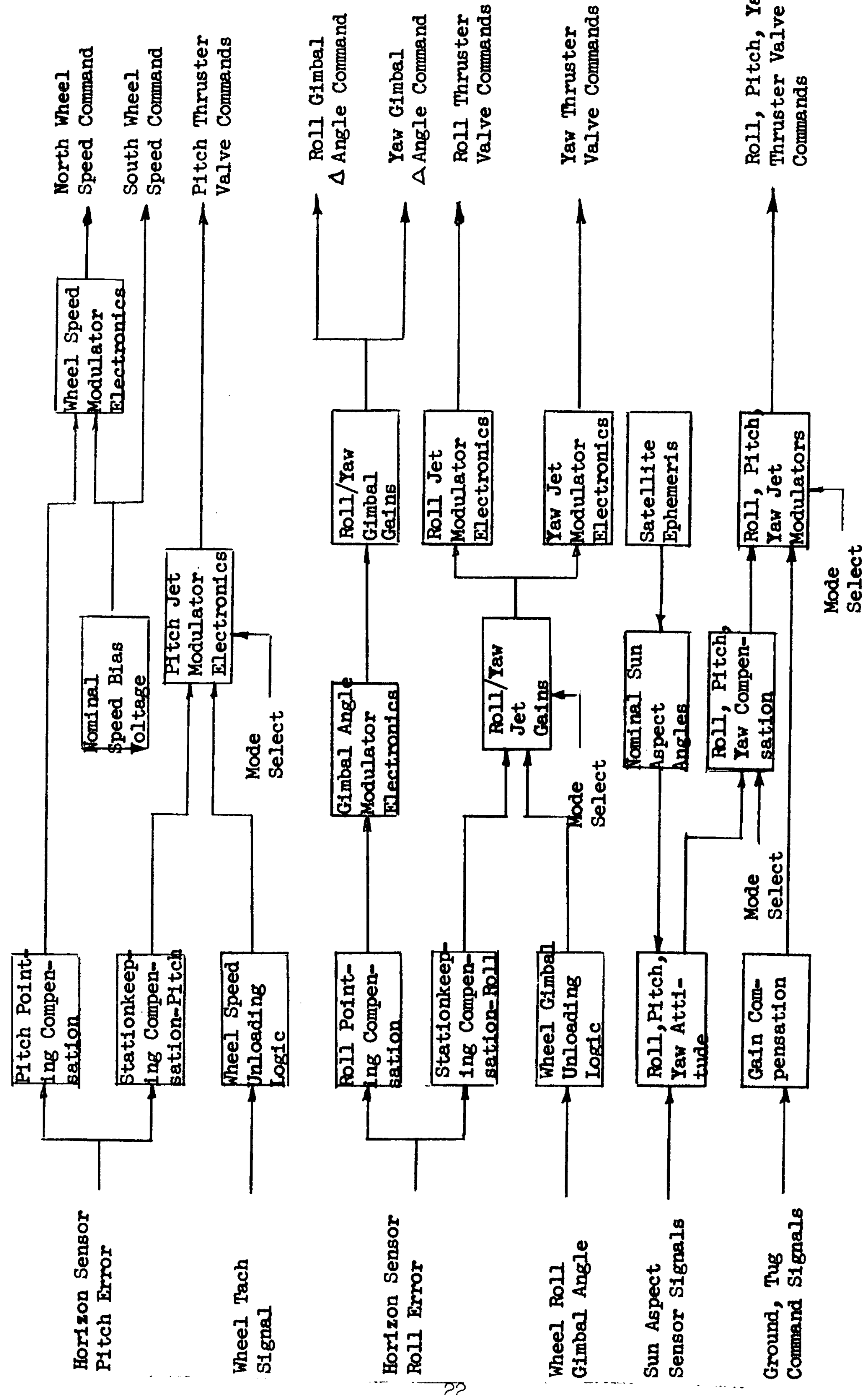

Fig. 6 Stabilization & Control Subsystem On-Board Data Processing (CDPI Subsystem)

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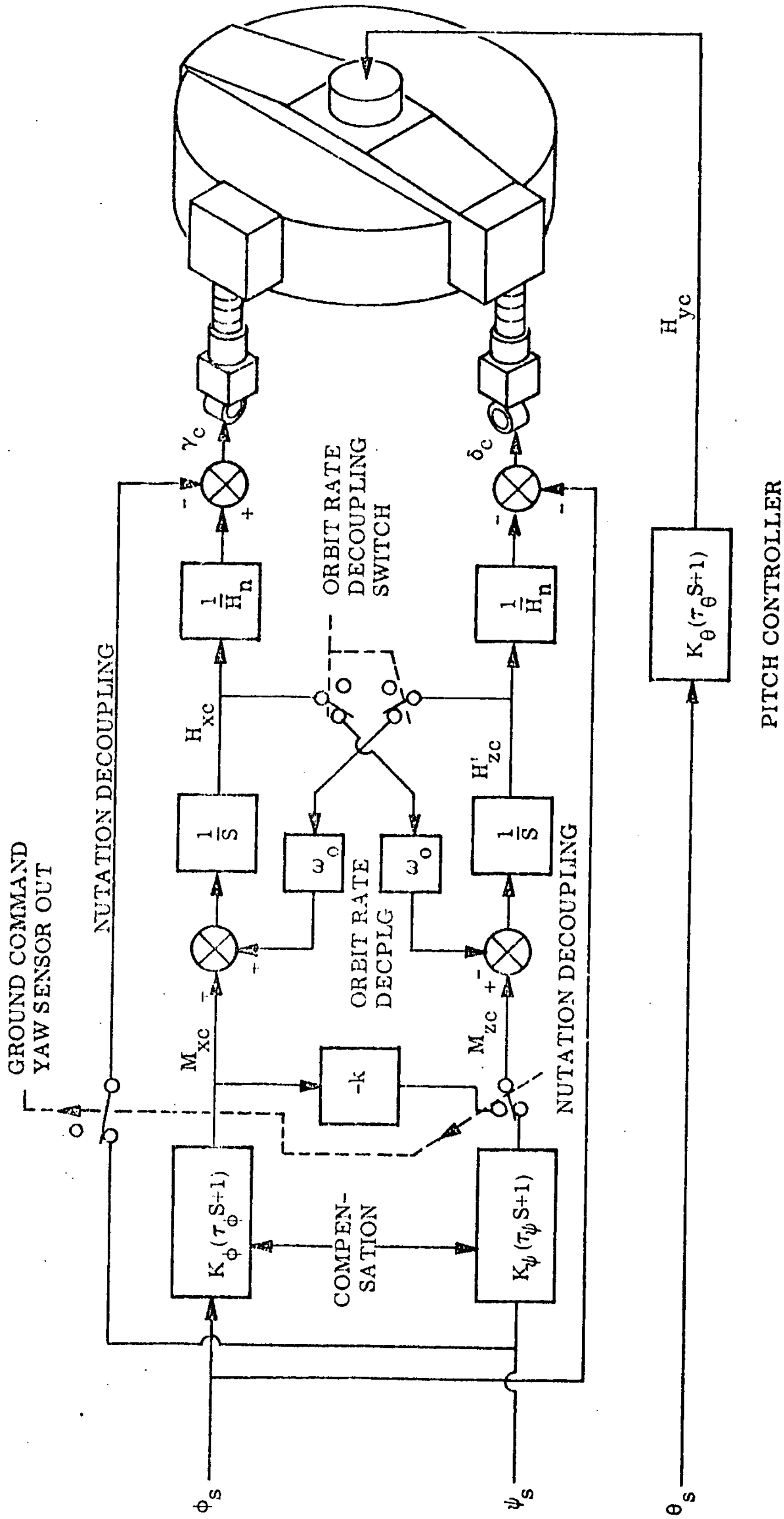

Fig. 7 DGMW Functional Block Diagram

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gimbal system, orbit motion decoupling would be accomplished automatically; the gyroscopic torque would be applied directly the the appropriate gimbal. The orbit motion decoupler supplies active gimbal drive commands to counteract friction and inertia without the need for an associated roll attitude error.

(3) Momentum Dumping (Desaturation)

Whenever the wheel speed exceeds an upper or lower bound, the momentum dumping system turns on a set of pitch reaction jets and at the same time activates a pitch error signal bias to avoid a steady state offset during the momentum dumping process. The sense of the applied torque is such as to cause the wheel to counter with speed changes towards nominal. When the angular momentum of the reaction wheel reaches the nominal value, H_N, the system is shut down. Roll axis angular momentum dumping is initiated whenever the yaw axis gimbal angle exceeds a set limit; the roll gas remains on until the yaw axis gimbal angle returns to zero.

(4) Stationkeeping

The stationkeeping control system maintains attitude pointing during periods of orbit velocity adjust. The attitude torques from moderate thrust misalignments makes it very desirable to have the thrusters used for stationkeeping simultaneously correct attitude errors. Such an implementation is shown in Figure 8 for one axis. Two thrusters bracket the center of mass and are used in an "inverse" modulation mode. When in the stationkeeping mode both thrusters are nominally on. Attitude errors that build up due to torque imbalance cause a jet to be switched off which results in a net correction torque. In steady state the larger torque-producing valve modulates while the other stays on.

The yaw sensor for use in stationkeeping operation is the sun sensor. The yaw angle can be determined during a significant portion of the orbit by measuring the angle from the spacecraft X-Z plane to the sunline. This angle is a function of the declination of the ecliptic with respect to the orbit plane and a function of the vehicle attitude and the position in its orbit. This function can be calculated in the computer from sun and computed orbital ephemeris data to reference the sun's azimuth position. The sun sensor output would then be compared to this reference to generate a yaw error signal to drive the yaw control system.

It has been shown that the sun aspect at the time of the north-south stationkeeping maneuver is never less than ± 23.5 deg. (Ref. 5).

To preclude imposing excessively long thrusting intervals on the north-south thrusters, it is planned to perform a daily ΔV correction using the north and south thrusters on alternate days. About 0.46 FPS per maneuver is necessary. This relatively small burn can be programmed by the on-board computer, i.e., be done "open loop." Then any accumulated errors in

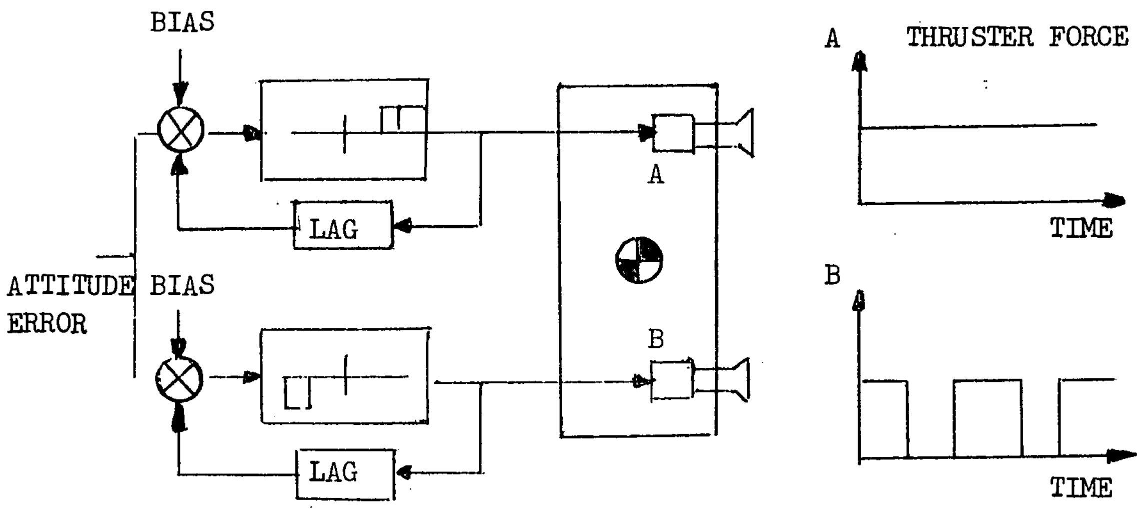

Figure 8 STATIONKEEPING WITH INVERSE MODULATION

satellite position, detected by tracking, would be removed by a vernier adjustment to a scheduled burn. It is planned to perform the calculation of drift from the nominal station and the associated orbit adjust burn in the CDPI on-board computer.

(5) Space Tug Rendezvous and Docking

There are two circumstances under which it will be necessary for the Tug to return to the CSS and dock. After initial separation, the Tug will stand by in the proximity of the CSS while initial operability is verified (Section VI). Should the CSS fail to prove acceptable, it will be returned to low orbit by the Tug. The other reason for providing a docking capability is so that the spacecraft can be retrieved for its salvage value at the completion of its mission. In either case it is assumed that all S&C equipment is operating normally.

The Tug will be vectored to a position inside a 30-deg cone about 5 NM behind the earth-oriented CSS. During the docking approach, the CSS horizon sensor reference is replaced by the tug-measured DCS roll, pitch, and yaw errors from the laser LOS. These attitude errors are transmitted to the CSS over the rf command link and the CSS attitude is kept aligned and indexed to the beam during the tug's approach.

(6) Reacquisition

The CSS is fully-operational at Tug departure and is not required to change from its earth-vertical orientation during the five-year mission life. Should a reversible failure occur, such as an intermittent power failure, and the spacecraft lose earth reference or tumble, recovery can be initiated automatically or by command. First, rates about all three axes are nulled by the hot gas jets driven by signals from the 3-axis rate gyro package. Then an earth search is executed by commanding a spacecraft rotation about its pitch axis. If a full rotation fails to detect the presence of the earth, the maneuver reverts to a sun search. The solar aspect sensor fields-of-view are oriented to provide a full 4π steradian coverage when the spacecraft is pitched through 360 degrees. When the sun enters the field-of-view of one of the solar aspect sensors, the search is stopped and the spacecraft is stabilized to the sunline. The earth search is repeated each hour thereafter, until the earth enters the field-of-view of the horizon sensor. Under the worst orientation-time of day situation this would require over five hours. This time lapse could be shortened to a matter of minutes by executing a 65° yaw maneuver and repeating the earth search should the first earth search fail to acquire after a rotation or two. Alternatively, if the time of day (sun position) is known, a prescribed maneuver from the sun-line to the earth-vertical is be performed for immediate reacquisition.

IV. SUBSYSTEM INTERFACES

A. Subsystem Equipment Modules

An important factor in determining the cost of revisit/repair/refurbishment operations is the manner in which the S&C equipment is physically grouped. These groupings could range from a single assembly to the complete S&C subsystem; a group could even mix parts from two or more subsystems. The selected arrangement was arrived at by application of the modularity guidelines (Table 5).

The modules and interconnections are shown schematically in Figure 9. Most of the S&C equipment is grouped into two modules: the sensing and the wheel modules. These modules are physically adjacent and the sensor module abuts the antenna alignment reference plane. Figures 10 and 11 depict the two S&C modules. The dashed lines on the sensing and electronics module represent cutouts for sensor viewing.

B. Attitude Control Propulsion Subsystem Functional Requirements

The Attitude Control subsystem provides reaction jet control for momentum wheel unloading, restabilization/reacquisition after a catastrophic upset and backup pointing in the event of wheel failure.

S&C requirements imposed upon the Attitude Control subsystem are summarized in Table 6.

The specifications assume a thruster couple at 8-ft separation, or equivalently, two thrusters at a 4-ft moment arm.

Table 5

MODULARITY GUIDELINES

(1) Minimize number of modules and types of modules

(2) Pick well-defined electrical interfaces which are parameter tolerance-insensitive

(3) Keep number of electrical connections at each module as small as possible

(4) Segregate into categories of higher and lower probability of operating longer than the specified mission time. "Higher" means about twice the survival time of "lower".

(5) Keep equipment volume within 26" x 26" x 40" module envelope

(6) Keep module cost to less than 5% of spacecraft cost

(7) Place redundant assemblies into separate modules wherever a failure could cause a chain reaction with catastrophic consequences

(8) Avoid EMI within and between modules

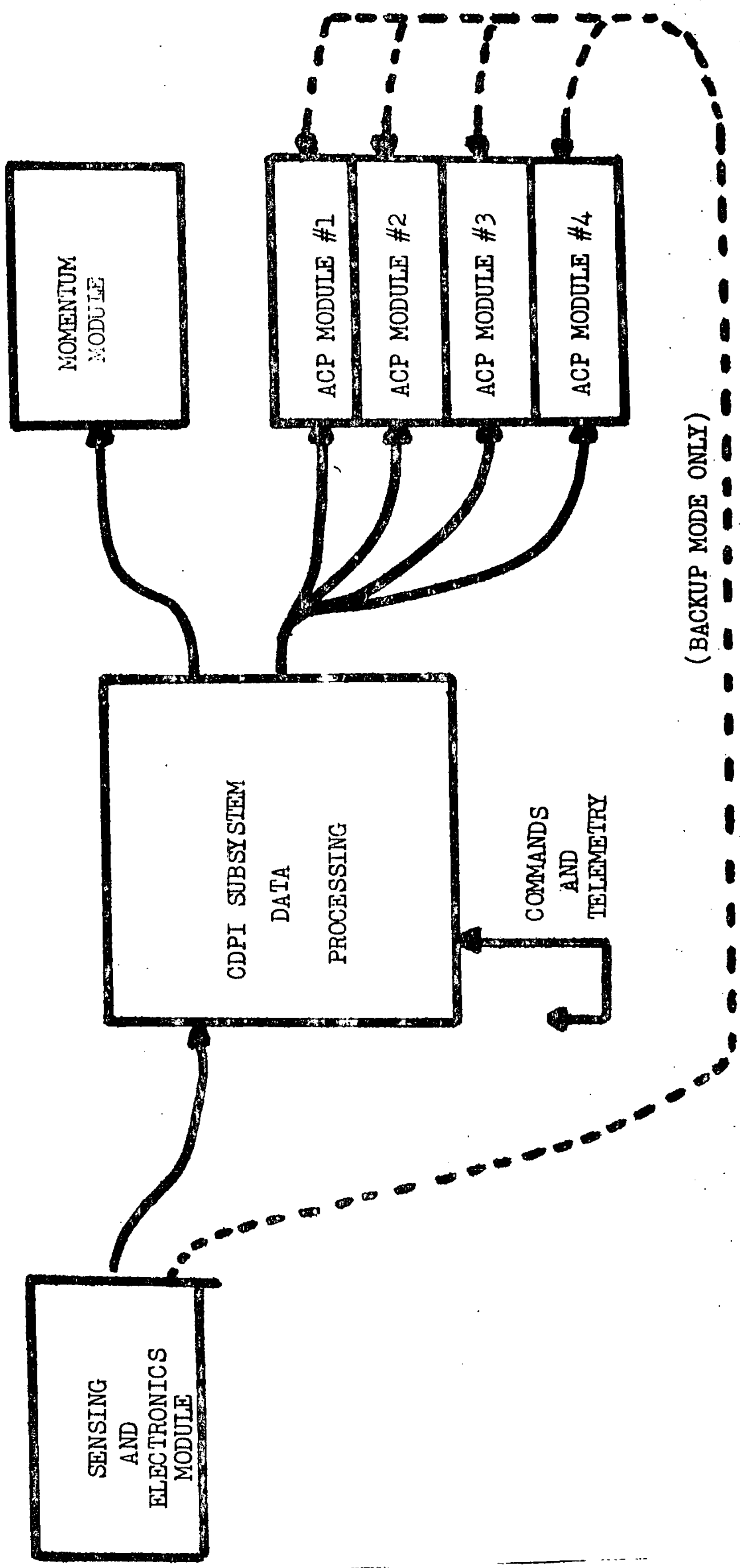

Fig. 9 CSS S&C Module Interconnection

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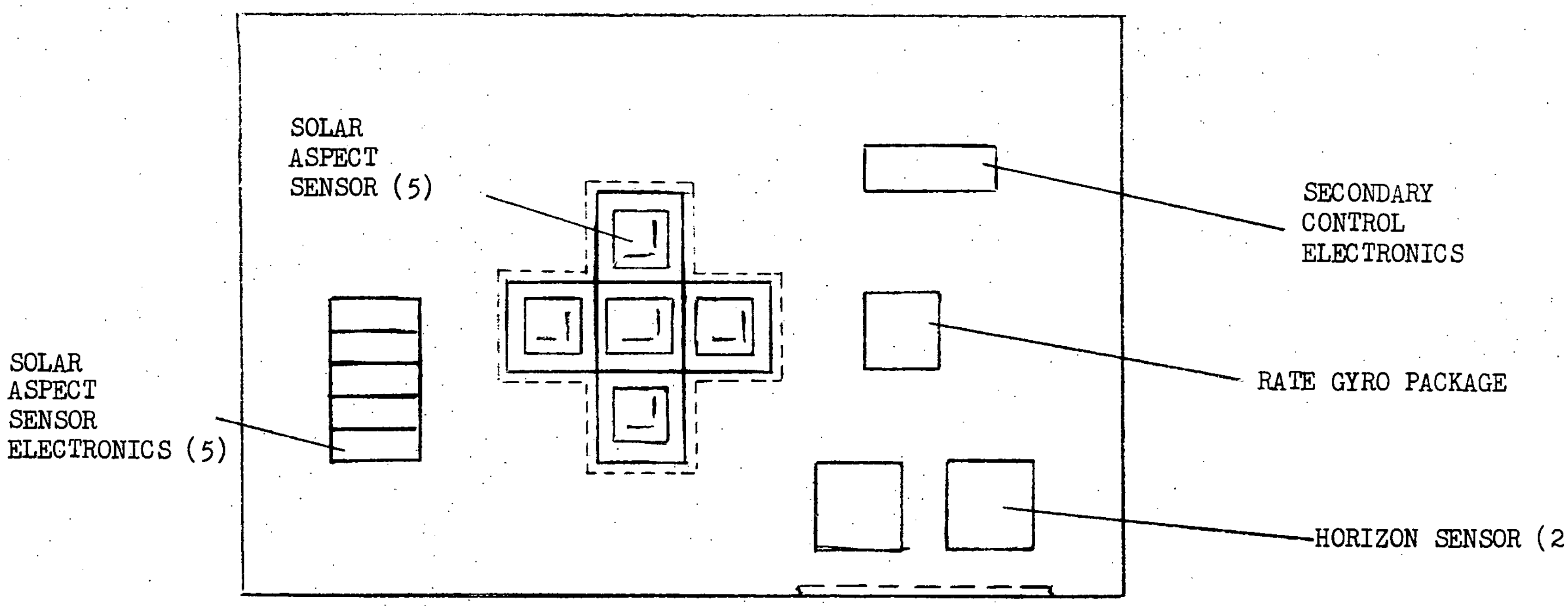
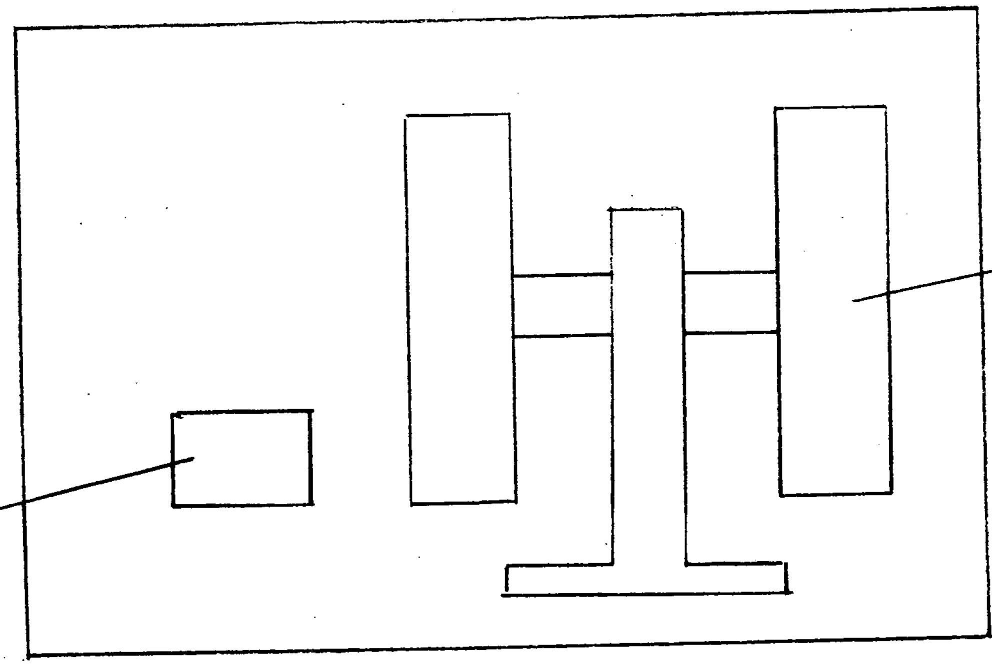


MODULE ALIGNMENT TOLERANCE: ± 3 arc min

ITEM	INTERNAL ALIGNMENT (Arc Min)	SIZE (in)	WEIGHT (lb)	POWER (w)	DUTY CYCLE (%)	FAILURE RATE (per 10^6 hrs)
Hroizon Sensor (2)	±3	5 x 5 x 19	12	10	100	4
Solar Aspect Sensor (5)	±6	3 x 4 x 1	15	5	1	3
SAS Electronics (5)		2 x 5 x 4				
Rate Gyro Package	±10	4 x 4 x 2	3	15	1	10
Secondary Electronics	-	2 x 4 x 6	5	5	1	10

Fig. 10 Sensing & Electronics Module



MODULE ALIGNMENT TOLERANCE: ±3 arc min

ITEM	INTERNAL ALIGNMENT (Arc Min)	SIZE (in)	WEIGHT (lb)	POWER (w)	DUTY CYCLE (%)	FAILURE RATE (per 10^6 hrs)
Momentum Wheel	±3	15D x 19	82	16	100	5
Wheel Drive Electronics	-	4 x 4 x 6	3			

Fig. 11 Wheel Module

Table 6 CSS Attitude Control Propulsion Requirements

1. CAPABILITIES

   Three-Axis Rotation (Uncoupled)
   Three-Axis Translation (Uncoupled)

2. FIVE-YEAR IMPULSE DEMAND

   a. North (or South) Thrusters
      1000 lb-sec initially
      65 lb-sec/day @ one time

   b. East (or West) Thrusters
      6000 lb-sec initially
      40 lb-sec/week

   c. All Thrusters
      3 lb-sec/week

3. Minimum Bit* (couples @ 8-ft)
   Less than 0.020 lb-sec required
   Less than 0.010 lb-sec desired

4. Thrust (couples @ 8-ft)
   Less than or equal to 0.4 lbf desired

* Assumes $I_Y = 1500$ ft-lb-sec^2

C. Communications Requirements

Minimum telemetry and command requirements for the S&C subsystem are shown in Tables 7 and 8 respectively.

V. FLIGHT-READINESS CHECKOUT

Various CSS flight readiness checks will be performed at several points in the launch sequence so various tests can be conducted on the S&C subsystem during any or all of these. The potential test phases are:

A. Pre-Launch

1. In launch site facility
   a. CSS in test fixture
   b. CSS mated to tug
2. In Shuttle Payload bay -- mated

B. Post-shuttle injection - mated

1. In payload bay
2. Deployed outside shuttle
3. Separated from shuttle

C. Post-tug injection into syn-eq orbit

1. Mated to tug
2. Separated from tug with tug nearby

Obviously it is neither necessary nor desirable to test all S&C functions during all of these intervals.

A suggested matrix of S&C subsystem tests is shown in Table 9. Complete end-to-end check of the various S&C subsystem modes is performed in the hangar before mating CSS to the tug. This scope of test is not repeated again until the CSS is once again free of the tug, after injection into near-synchronous equatorial orbit. A velocity capability equivalent to arresting a 3 deg per day drift to station longitude was provided. During the drift period the CSS S&C subsystem could be exercised with the tug standing by.

VI. S&C COMPONENT DESCRIPTION

A. Horizon Sensor

The horizon sensor provides signals proportional to spacecraft attitude deviations in pitch (East-West pointing) and roll (North-South pointing). Its

Table 7 COMSAT Standard Spacecraft Minimum Telemetry List

NORTH WHEEL SPEED
SOUTH WHEEL SPEED
NORTH WHEEL TEMPERATURE
SOUTH WHEEL TEMPERATURE
ACTIVE HORIZON SENSOR TEMPERATURE
INACTIVE HORIZON SENSOR TEMPERATURE
WHEEL DRIVE ELECTRONICS TEMPERATURE (2)
SUN ASPECT SENSOR TEMPERATURE (5)
SUN ASPECT SENSOR ELECTRONICS TEMPERATURE (5)
RATE GYRO PACKAGE TEMPERATURE
ACP THRUSTER DRIVE ELECTRONICS TEMPERATURE (4)
ROLL GIMBAL ANGLE
YAW GIMBAL ANGLE
ROLL ATTITUDE ANGLE (HORIZON SENSOR)
PITCH ATTITUDE ANGLE (HORIZON SENSOR)
SUN ASPECT SENSOR OUTPUT ANGLE (2)
RATE GYRO OUTPUTS (3)

Table 8 COMSAT Standard Spacecraft Minimum Command List

PLUS PITCH BIAS STEP
MINUS PITCH BIAS STEP
PLUS ROLL BIAS STEP
MINUS ROLL BIAS STEP
GROUND STATION #1 SLANT RANGE
GROUND STATION #1 ELEVATION, AZIMUTH ANGLES
GROUND STATION #2 SLANT RANGE
GROUND STATION #2 ELEVATION, AZIMUTH ANGLES
MOMENTUM WHEEL YAW GIMBAL BIAS STEP - PLUS
" " " " " " - MINUS

INHIBIT UNLOAD - ROLL
INHIBIT UNLOAD - PITCH
PITCH THRUSTER PULSE - PLUS
PITCH THRUSTER PULSE - MINUS
ROLL THRUSTER PULSE - PLUS
ROLL THRUSTER PULSE - MINUS
YAW THRUSTER PULSE - PLUS
YAW THRUSTER PULSE - MINUS
EAST THRUSTER PULSE
WEST THRUSTER PULSE
NORTH THRUSTER PULSE
SOUTH THRUSTER PULSE
UP THRUSTER PULSE
DOWN THRUSTER PULSE
SWITCH TO INACTIVE HORIZON SENSOR
SWITCH TO INACTIVE WHEEL DRIVE ELECTRONICS
SWITCH TO SECONDARY ATTITUDE CONTROL
SWITCH TO PRIMARY ATTITUDE CONTROL
REMOVE POWER - NORTH MOMENTUM WHEEL DRIVE
REMOVE POWER - SOUTH MOMENTUM WHEEL DRIVE
SWITCH PITCH CONTROL SIGNAL TO OPPOSITE WHEEL

Table 9 FUNCTIONAL CHECKOUT FOR STABILIZATION & CONTROL

TYPE OF STABILIZATION & CONTROL SUBSYSTEM TEST / RECOMMENDED TEST PHASE	A. Pre-Launch: 1. In facility (a) CSS alone	A. Pre-Launch: 1. In facility (b) CSS Tug mated	A. Pre-Launch: 2. Mated-in Shuttle bay	B. In Low Orbit: 1. In Shuttle Bay	B. In Low Orbit: 2. Deployed	B. In Low Orbit: 3. Separated from Shuttle	C. In Syn-Eq Orbit: 1. Mated to Tug	C. In Syn-Eq Orbit: 2. Separated from Tug
Component Checks								
Horizon Sensors	x						x	x
Primary Control Software (CDPI)	x	x	x	x	x	x	x	x
Secondary Electronics	x	x	x	x	x	x	x	x
Momentum Wheel	x	x	x	x	x	x	x	x
Rate Gyro Package	x				x	x	x	x
Sun Aspect Sensors	x				x	x	x	x
Function Checks								
Earth pointing loop	x							x
Rate null/attitude hold	x							x
Slew command	x							x
Sun pointing	x							x
Command pointing	x							x
Stationkeeping attitude hold	x							x
Wheel unloading	x							x

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attitude measurement accuracy at synchronous altitude must be about ±0.03 deg RSS (excluding earth radiance variations and horizon uncertainty).

There are three types of synchronous altitude earth sensors capable of meeting the CSS requirements: scanning, balanced radiation, and edge trackers. All have comparable accuracy and development status. The LMSC scanning sensor, representative of the different devices available, is selected for this application for the following reasons:

1. Relatively insensitive to thermal environment
2. Digital output
3. Detailed technical and cost information readily accessible.

Description -- The Lockheed Null Operating Horizon Sensor (NOHS) is entirely contained in a single package (Figure 12). The lens, filter, and detector combination forms two 1.1 degree fields-of-view, six degrees apart which accept energy in the 15-micron CO_2 absorption band. The fields-of-view are scanned back and forth across the earth with a resonant, torsion bar mirror system at 4 cps. The mirror drive contains a counter which puts out a pulse for each 0.005 degree of mirror motion. In addition, there is a center position pulse which is coincident with one of the 0.005 degree pulses.

The sensor scans the fields-of-view of the two detectors across the earth with a single mirror. One field-of-view crosses at 45°N and the other at 45°S latitude. In the direction of scan (pitch) the angle from the space-earth crossing of the detector field-of-view to mirror center is compared digitally to the angle from mirror center to the earth-space crossing. The difference in these angles is proportional to the vehicle attitude error. To minimize error, this computation is made in both directions through one complete cycle (four times) before updating the digital counter. The noise error is minimized by the digital averaging. Roll is computed by comparing the chord length of the north scan to that of the south scan. When these are equal, the vehicle is at roll null. The NOHS specifications are summarized in Table 10.

The earth coverage with the scan pattern is illustrated also, as are the extreme positions in which the earth can still be sensed. The use of two detectors allows full operation during sun or moon presence. For pitch, the sun-free detector is used for computation; for roll, a "standard" chord length is substituted for that of the detector viewing the sun. During sun presence the sensor will operate normally at null, but its off axis performance will be limited.

The signal characteristics of the NOHS are shown in Figure 13.

Optics

The lens is an F/1.0, 1.5-inch aperture, aspheric, germanium singlet. The front surface is coated for peak transmission at 15 microns. The rear surface is coated with a long wavelength pass filter to suppress the short wavelength harmonics of the bandpass filter. The detectors are thermistor bolometers with 0.060-inch radius immersion lenses. These small lenses help minimize the germanium absorption losses. The detectors are located plus and minus six degrees off axis.

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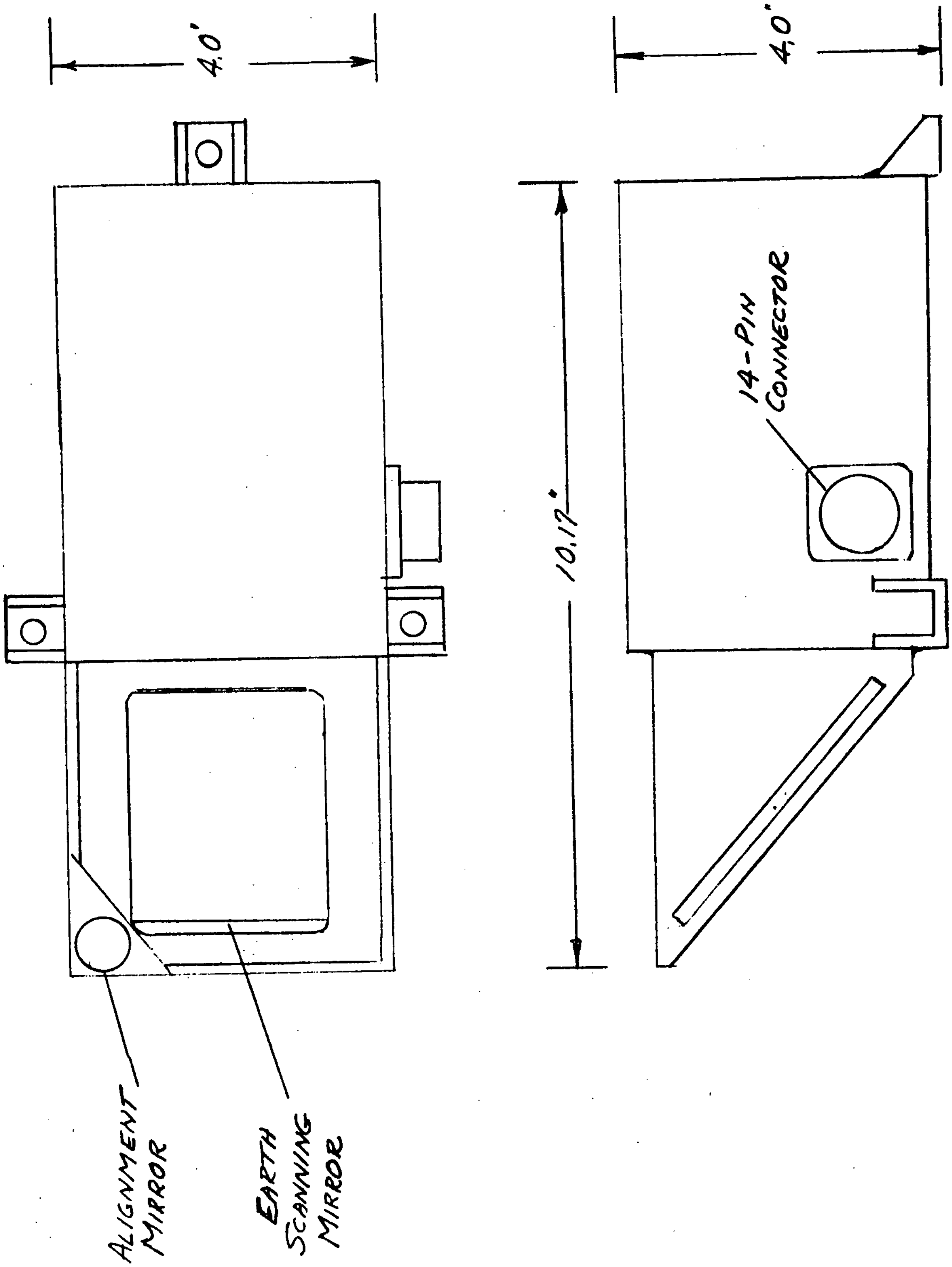
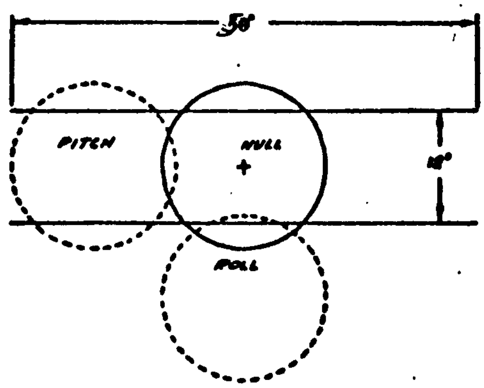


Fig. 12 LMSC Null Operating Horizon Sensor

Table 10

NOHS SPECIFICATIONS



Aperture:		1.5 inches
Spectral Inter.:		14.1 - 15.8 Microns
Inst. F.O.V.		1.1 X 1.1 Degrees
Scan Field:		50° pk-pk Sin @ 4 Hz
Range:	Pitch:	6 Degrees
	Roll:	2.56 Degrees
Null Accuracy:		± 0.03 Degrees (.003° RMS Noise)
Earth Effects:		± 0.03 Degrees
Linearity:		Within 0.1 Degree @ 2 degree roll angle
Output Rate:		Update Once Per Second
Output Signal:		Digital
		Multiplex on Command

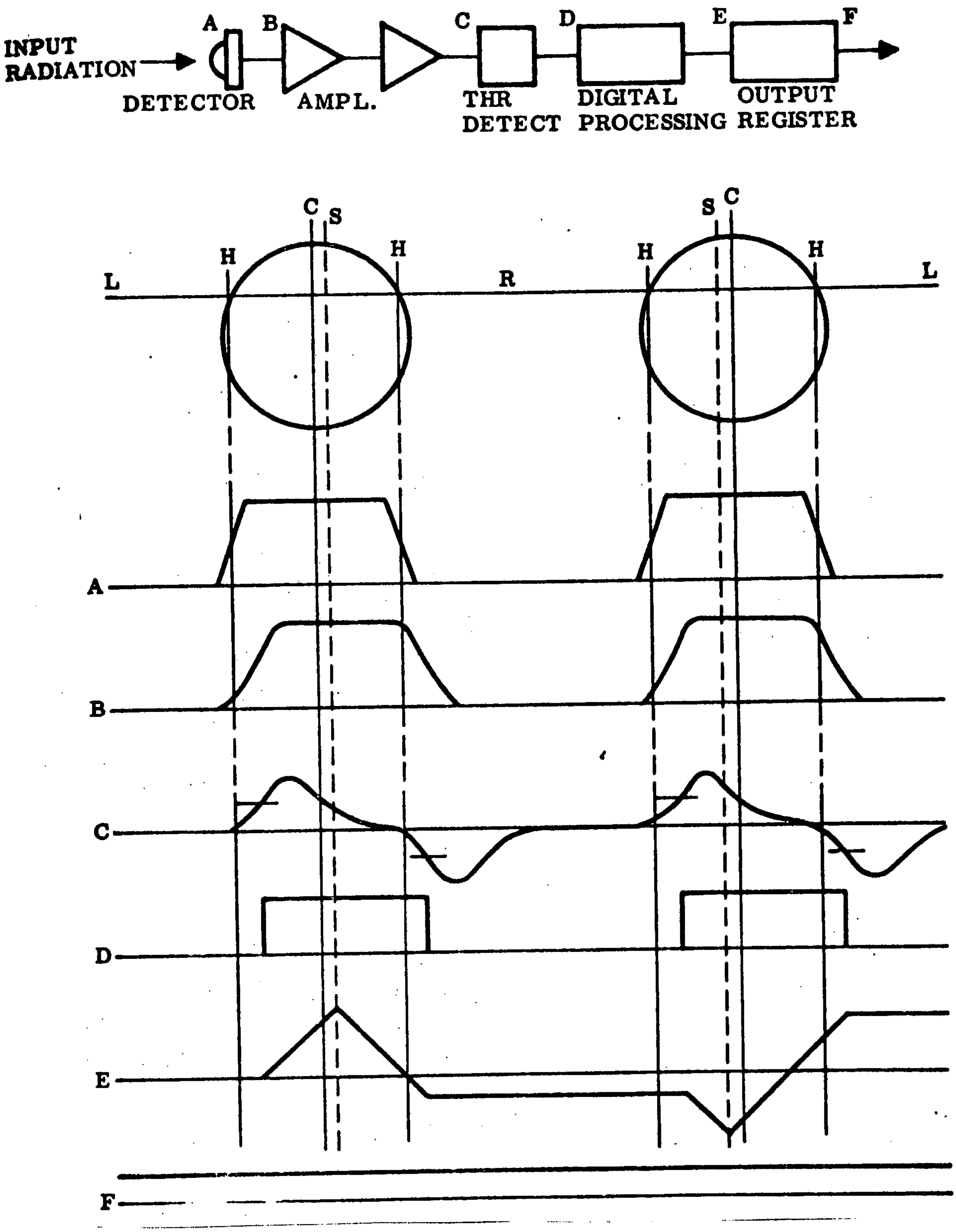


Fig. 13 NOHS Block Diagram and Signal Processing

Scanning Mirror

The scanning mirror is a resonant system with a very high Q. A torsion bar suspension, incorporated for high reliability, is of Elgiloy, a cobalt-nickel alloy with a very high endurance strength. The maximum stresses are only about 50% of the endurance limit giving theoretically infinite life. The mirror has two permanent magnets, one for a pickup coil and one for a drive coil. When the mirror amplitude is 25 degrees pk-pk, the drive voltage is removed. This method allows minimum turn-on time and minimum operating power when at full amplitude (a few milliwatts). The frequency of the motion, 4 cps, is determined by the mechanical frequency of the system.

Amplifiers

The preamplifiers are a high-impedance, low-noise, differential field effect transistor design. The gain of the preamplifier is about 500 while the main amplifier will have a gain of 100. There are antisaturation circuits in the main amplifier to minimize sun recovery time.

Detector Bias Supply

A DC-to-DC converter is used to bias the detectors ± 20 volts. This is necessary to obtain optimum performance from the detector and to keep noise on the detector below one microvolt. Using an active filter on the power lines and applying plus and minus voltages to the detector bridge is the best method to assure a detector-noise-limited system.

Threshold Detectors

A fixed threshold detector is used on both the positive and negative pulses set at the nominal point of maximum slope. This simple operation requires only one hysteresis type threshold detector per channel.

Other processing electronics elements include the detector select logic, error detect logic (e.g., moon in view), mirror clock, mirror scan timing, up-down and counter.

B. Gimballed Momentum Wheel

The dual-gimbal momentum wheel consists of a baseplate and housing assembly, flywheel and bearing assembly and brushless DC motor and optical commutation electronics. The wheel pictured in Fig. 14 is built by Teldix of Germany for the Lockheed Communications Satellite Program. Its size and form are representative of that needed for the CSS. Dimensions shown for the momentum wheel are in millimeters. The cross-section of a wheel is shown in Fig. 15.

The wheel features two housings rigidly linked to each other across a central gimbal system. The actuators required for the attitude control operations about the roll and the yaw axis, and their associated spring dampers, act upon the

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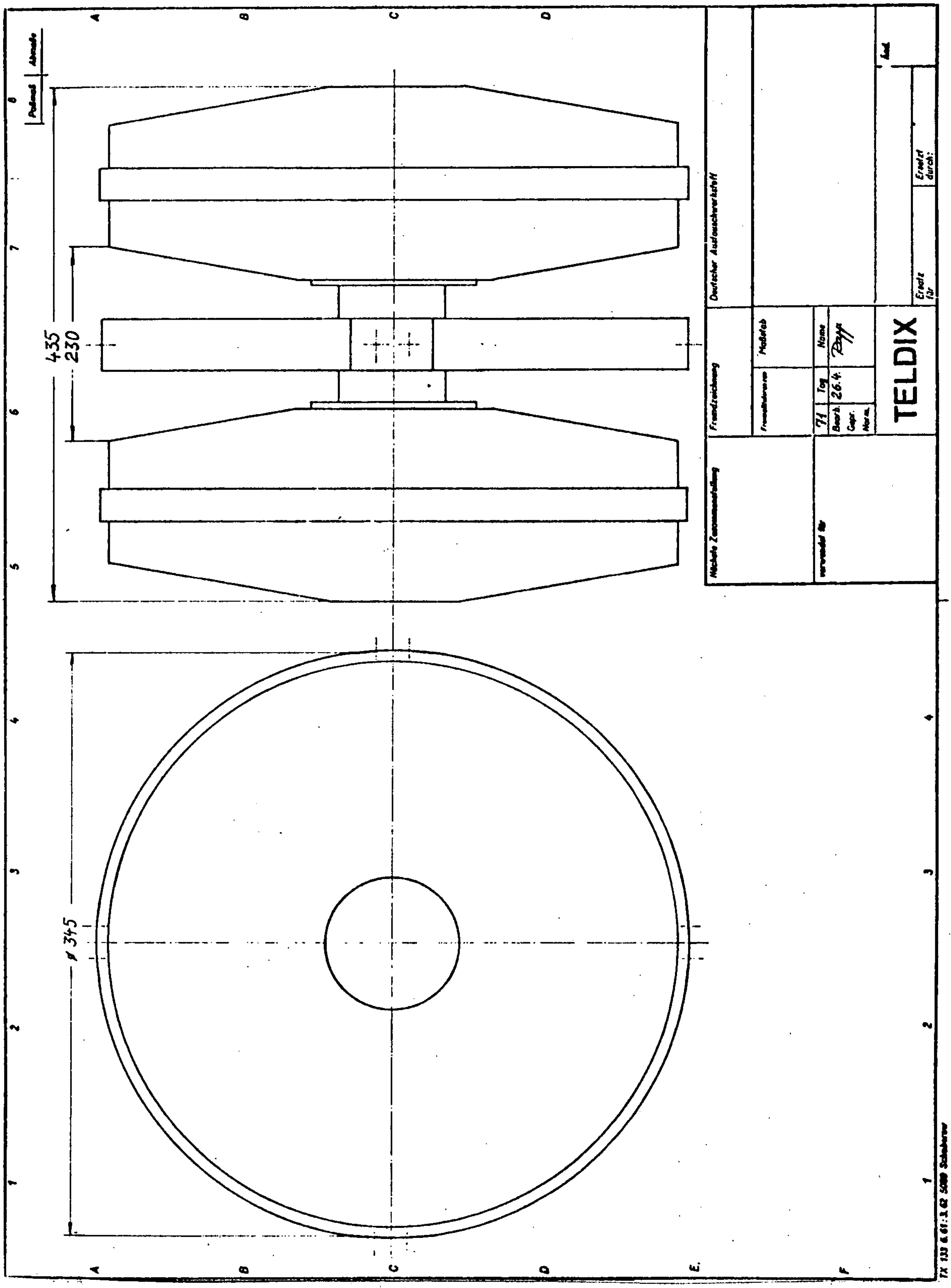

Fig. 14 Momentum Wheel Dimensions

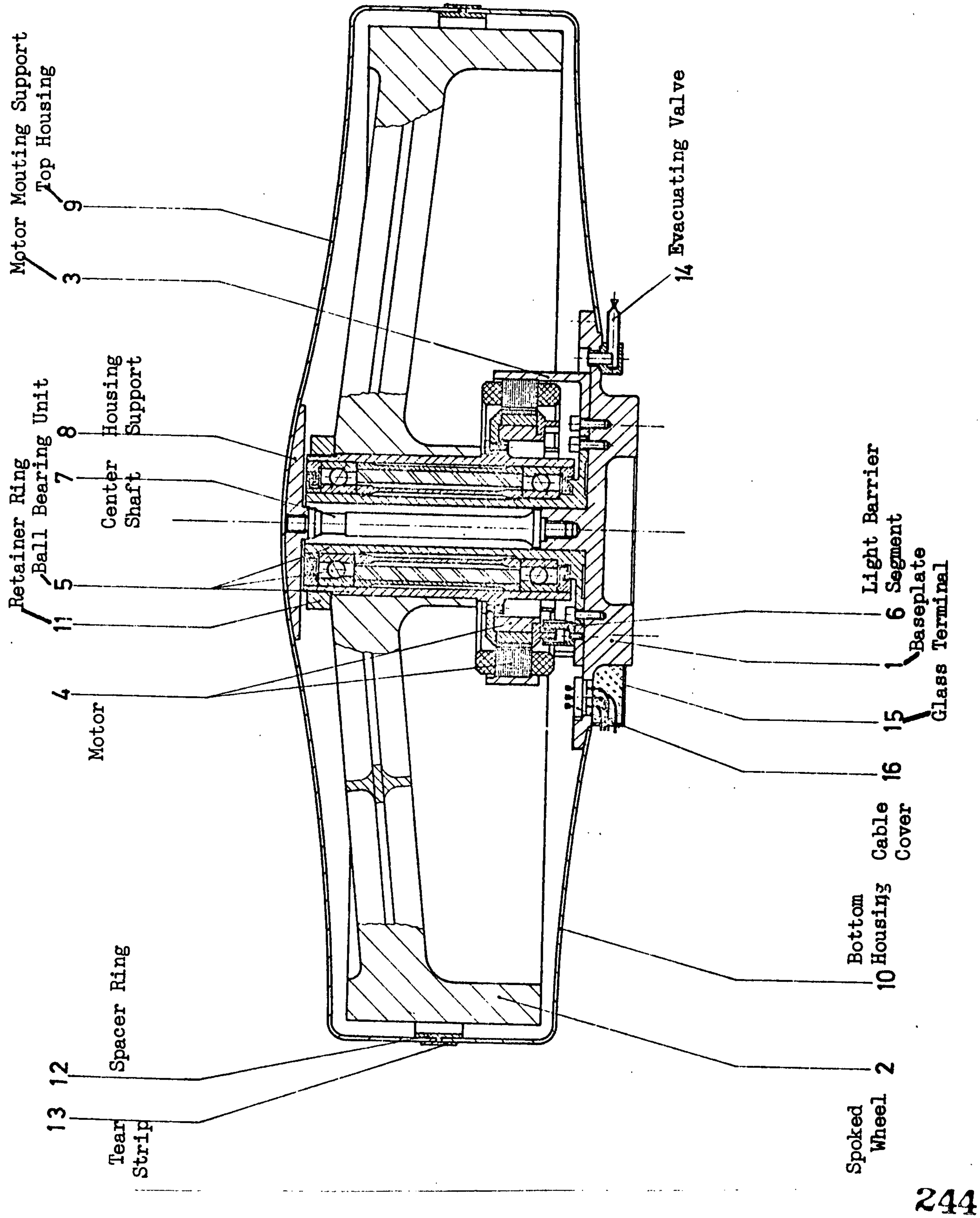

Fig. 15 Cross-Sectional of Momentum Wheel

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inner gimbal from the satellite structure (Fig. 16). The outer ring of the gimbal system is bolted to the satellite structure; through the intermediate ring, it supports the inner flange and the two housings. This twin-wheel configuration, chosen for application in advanced communications satellites having an operational life requirement of 10 years, avoids single-point failures.

The two wheel housings are arranged symmetrically with respect to the gimbal platform and are identical in outward appearance. Mounting elements are provided on the flange near the housings to transfer the thrust forces of the actuators required for angular displacements of the momentum wheel, as well as for the attachment of the spring dampers. The flange is designed to permit as much as ± 25° angular movement about two axes.

Gimbal Bearings

The relatively large angular travel range of the gimbal-mounted momentum wheel under discussion, in combination with the high weight to be supported, precludes the utilization of flexural pivots as gimbal bearings. Hence the gimbal rings are suspended by preloaded, paired and grease lubricated ball bearings protected by side plates on this design(**housing evacuated and sealed after initial assembly to prevent space-dissipation of grease, etc.**
The use of ball bearings offers the advantages of low actuator forces and simultaneously, high accuracy and stability of gimbal axes alignment. The electrical leads from the electronic unit to the momentum wheel have to be laid across this gimbal system. Due to the limited gimbal movement, the use of a flexible and shielded cable is practical and is arranged to avoid high torsional or bending torques and mass unbalance during gimbal displacements.

Gimbal Positioning Actuators

Reaction torques are produced in roll and yaw axes by commanding actuators to tilt the momentum wheel gimbals. Trade studies were conducted to determine the type of gimbal actuator best suited for use on the dual gimballed momentum wheel. Actuators investigated included:

(1) DC or AC motor driven vs stepper motor driven
(2) Open-loop vs closed-loop operation

A linear, open-loop, stepper motor driven actuator appeared best suited in comparison.

The specific linear actuator selected was derived from a Kearfott type C200537600. The linear actuator consists of a stepper motor, a precision gear train, a preloaded ball-nut lead-screw which translates angular motion to linear motion, a position indicator for telemetry, and electromechanical brake.

For the selected actuator, the maximum command pulse rate is 20 steps per second, which yield a maximum gimbal slew rate of 0.4 deg per second.

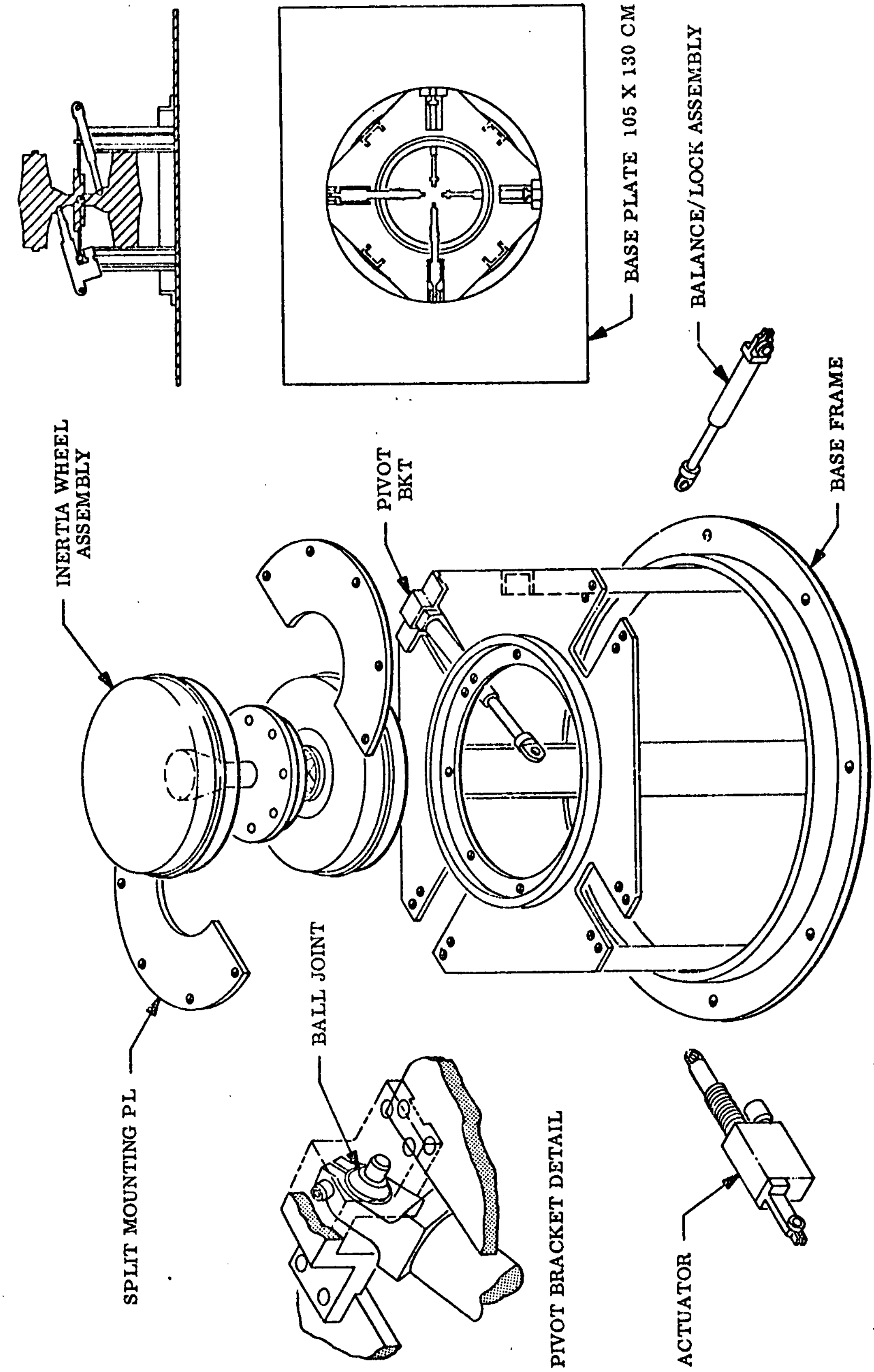

Fig. 16 Double-Gimballed Momentum Wheel Assembly

The salient specification requirements for the gimbal positioning linear actuator are as follows:

Voltage	+ 24 to + 31 vdc
Maximum Pulse Rate	50% at maximum pulse rate
Duty Cycle	20 pulses per second
Operating Power	4.0 watts at 24 vdc 6.0 watts at 31 vdc
Nominal Operating Load	+ 8.9 Newtons to -14 Newtons
Linear Travel	+ 25.4 millimeters
Incremental Travel	0.0254 millimeters/step
Overload Non-operating	980 Newtons
Weight	1.8 kilograms

Wheel Drive Motor

Comprehensive studies have led to the decision to use a brushless DC motor as drive motor for the flywheel. It provides excellent speed controllability and high efficiency. With photo-electric commutation, life restrictions caused by wear are avoided. The effect of electronic component failures on total life is reduced by incorporation of redundant commutation. Another advantage of using a DC motor is that it can be connected directly to the primary 28V DC power source avoiding the necessity of voltage and frequency conversion.

The reliability, efficiency, and total weight of this motor type are superior to those of comparable synchronous or asynchronous motors.

Drive Electronics

The electronics of the momentum wheel consists of the commutation electronics for brushless DC motor and the electronics which control the commutation electronics according to the input signals from the S&C data processing (Figure 17).

The logic assigns to all input possibilities one of the four outputs: Inputs 1 to 6 are plus torque, minus torque, and torque level signals; signals 7 to 10 represent output signals. The speed outputs (tachometer signals) consist of two analog and two digital signal lines each.

The motor commutation electronics for all three phases uses the signal from the phototransistor to trigger the switching transistor via amplifier stages. At the same time, a current limiting circuit maintains the motor current at a defined value independent of input voltage variations and motor speed.

C. Digital Solar Aspect Sensor

The Digital Solar Aspect Sensor (SAS) consists of five two-axis digital solar aspect sensors and is used to determine the two angles which describe the

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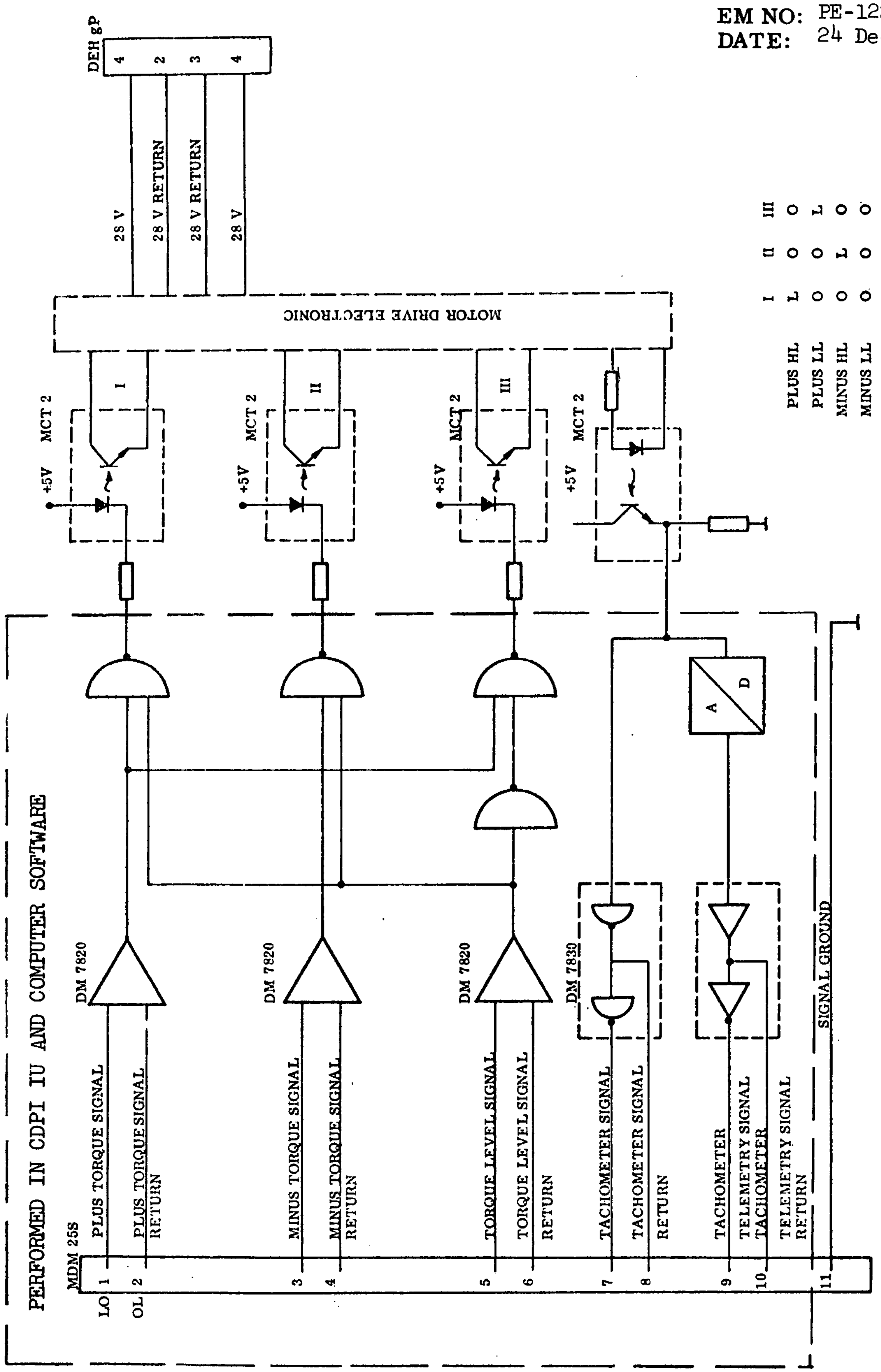

Fig. 17 Logic Circuit Diagram

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position of the sun vector with respect to the spacecraft. Information that uniquely describes the sun vector, plus a binary code to identify the sensor being read, is presented in digital form. Figure 18 shows the basic principle. Light passing through a slit on the top of a quartz block is screened by a Gray-coded pattern on the bottom of the block to either illuminate or not illuminate each of seven photocells. The angle of incidence determines which photocells are illuminated. The outputs from each cell are amplified, and the presence ("one") or absence ("zero") of a signal is stored and processed in the electronics to provide the desired output to telemetry. The pattern and the seven photocells shown provide 2^7 or 128 degrees in one degree increments. The units used for DCS would have 8 photocells and thus provide 0.5-degree resolution.

The signal processor compares the level of the bit outputs to the level of the Automatic Threshold Adjust (ATA) output and presents the appropriate outputs in digital form. The field-of-view indicator monitors the ATA output of the sensor that is connected and indicates to the telemetry whether or not the sun is in the field-of-view of the sensor. Figures 19 and 20 show the Adcole 15671 (SAS Electronics) and 15672 (Sensor Unit) respectively. The field-of-view of each sensor is ±32 deg by ±32 deg.

D. Rate Gyro Package

The rate gyro package measures angular rates about the roll, pitch, and yaw axis. The backup requirements are lax enough to permit the fulfillment by a wide variety of gyros. A typical package uses three Nortronics GR-G5 sub-miniature fluid-filled rate gyros. Table 10 and Figure 21 summarize the rate gyro package characteristics.

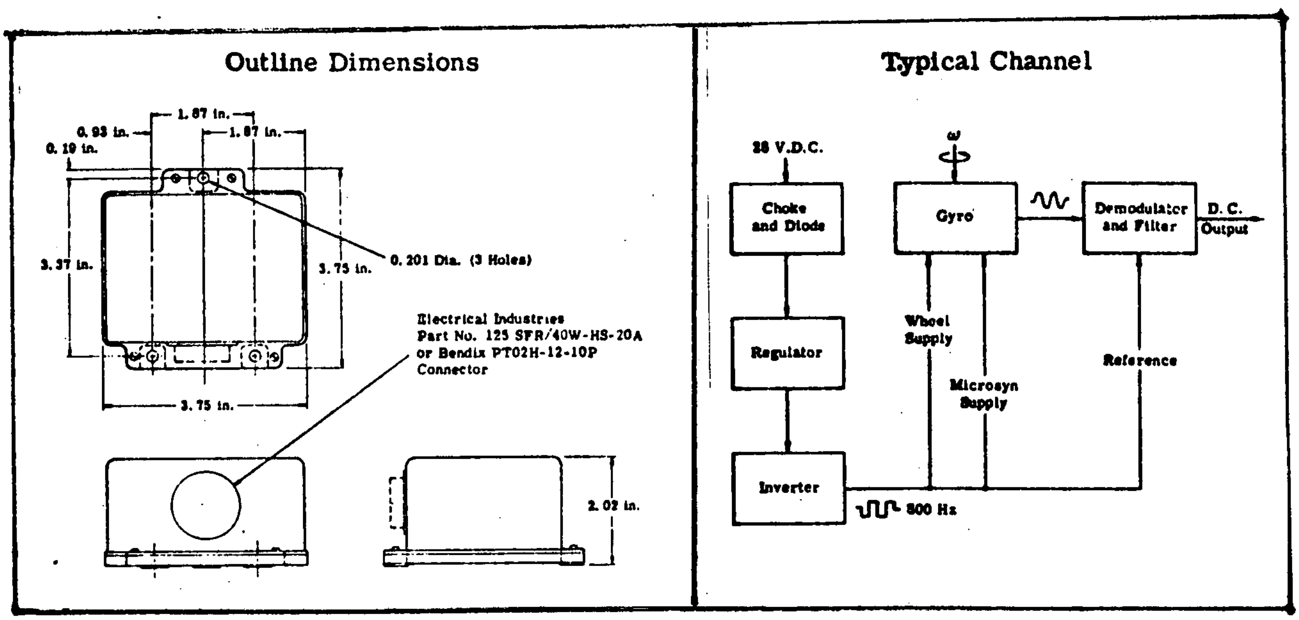

Fig. 21 Rate Gyro Package (Nortronics)

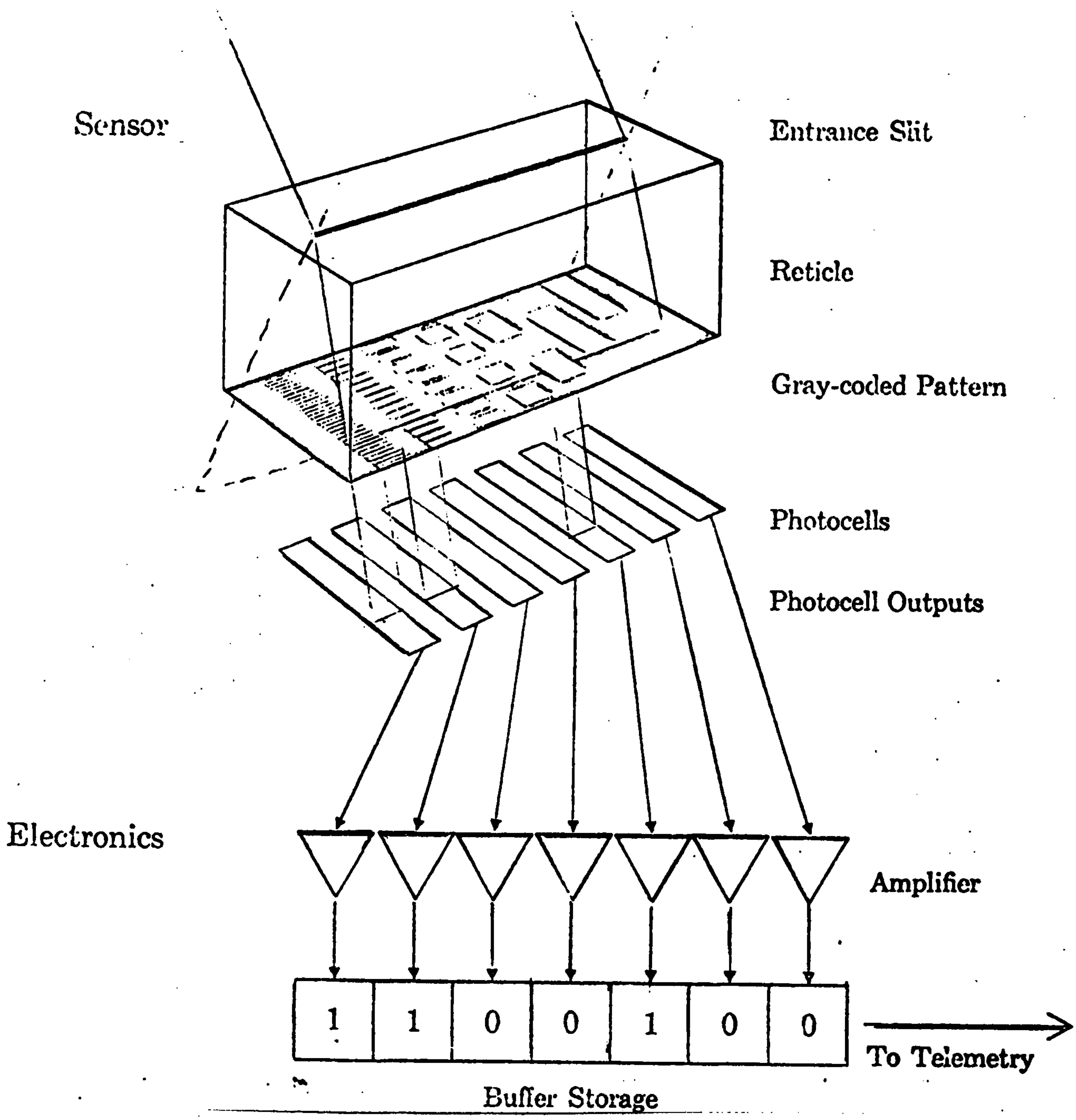

Fig. 18 SAS Schematic

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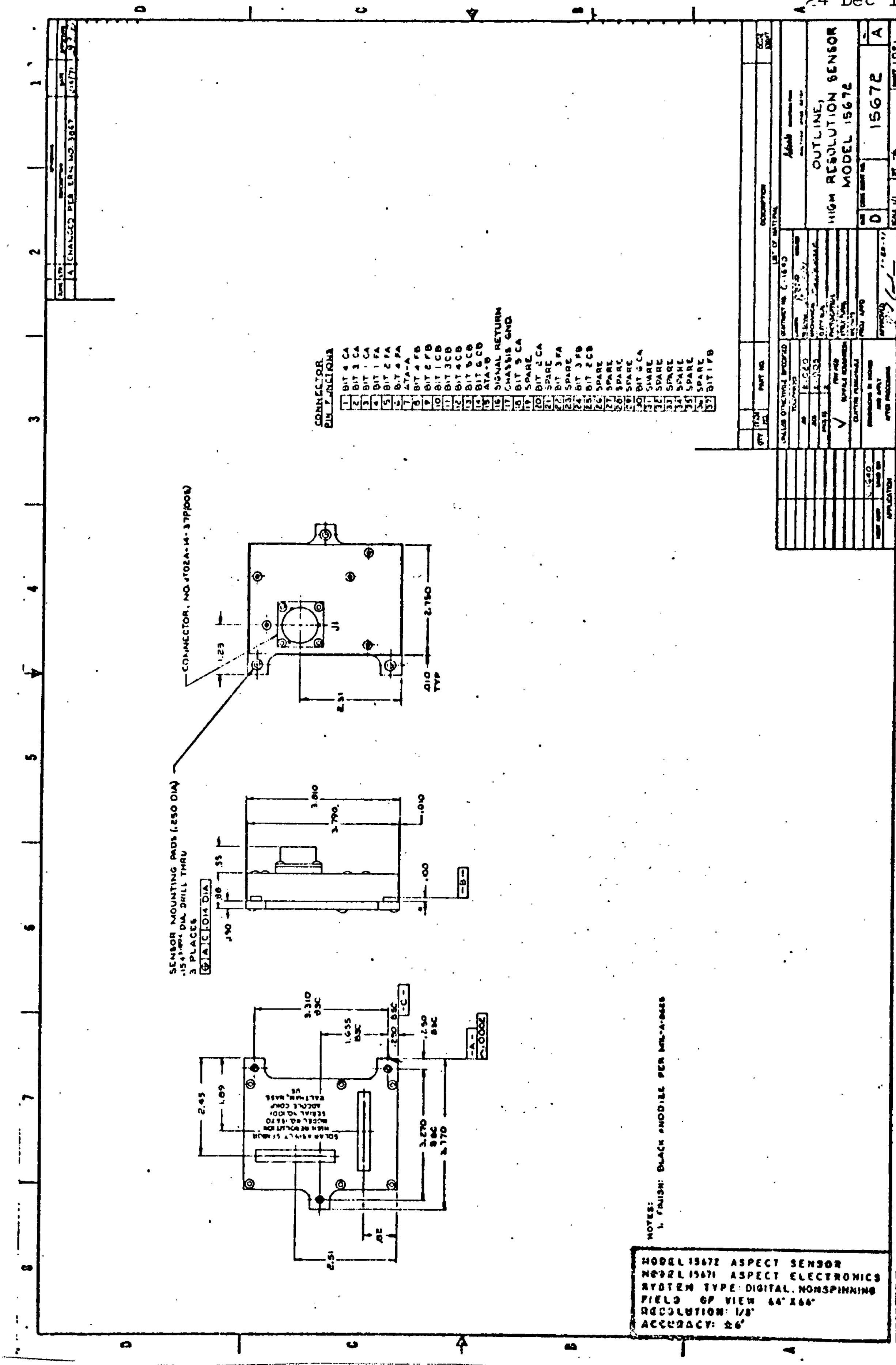

Fig. 19 Sun Aspect Sensor (Adcole)

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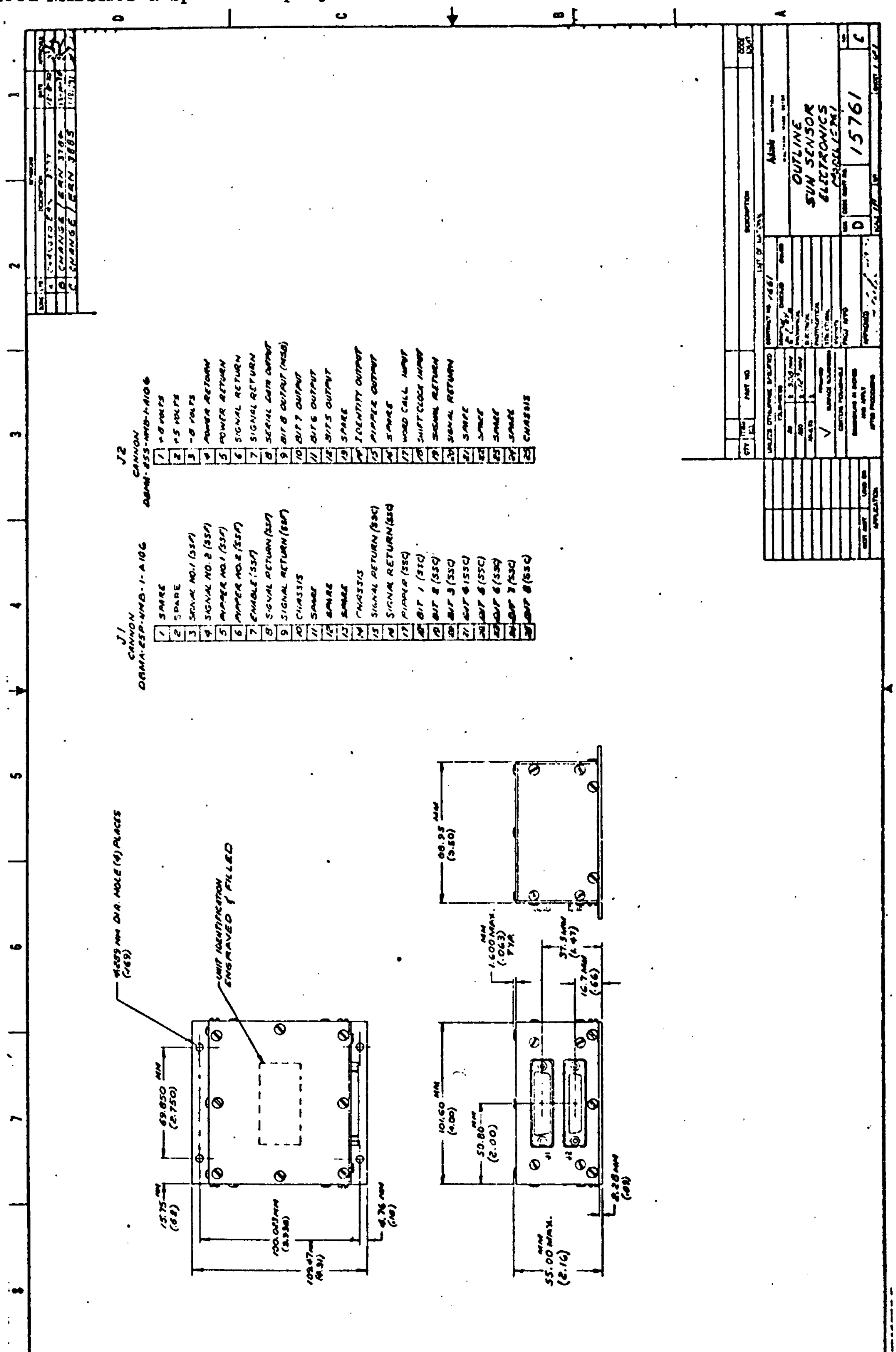

Fig. 20 Sun Aspect Sensor Electronics (Adcole)

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3 AXIS DC/DC STANDARD

RATE SENSOR ASSEMBLY

PART NUMBER 79157-350

NORTHROP
Northrop Corporation Electronics Division
Precision Products Department
100 Morse Street, Norwood, Massachusetts 02062
East Coast Tel.: 617-762-5300
West Coast Tel.: 213/289-9181

Table 10

MEASURED CHARACTERISTICS

Weight	2.0 lb. (max.)
Outline Dimensions	3.75 x 3.75 x 2.0 in. (max.)
Power Input	15 w. (max.) at 31 vdc
Input Voltage Limits	28 ± 3 vdc
• Full-Scale Output	2.5 ± 2.5 vdc
• Output Impedance	5000 ohms (max.)
Output Load Resistance	500K ohms (nominal)
• Ripple	25 mv. peak-to-peak (max.)
• Zero Rate Setting	±1/2 % FS
Input Range (Pitch, Roll, Yaw)*	100°/sec.
Maximum Input Rate	600°/sec.
Output Voltage Overrange Limits	-0.75 vdc, 7.0 vdc
• Output Stability, Input Voltage Variations	1/2% FS
Repeatability	1% FS
• Threshold*	0.01°/sec.
• Resolution*	0.01°/sec.
• Hysteresis*	0.1°/sec.
Operating/Storage Temperature*	-35°F to +160°F/-65°F to +200°F
Temperature Sensitivity	Zero Rate Output 1% FS/100°F (max.) Scale Factor 3%/100°F (max.)
Warm-up Time	10 minutes
• Gyro Spin Motor Acceleration Time	30 sec. (max.)
Gyro Gimbal Deflection Angle	±2° typical
Acceleration Sensitivity	
Linear*	0.05°/sec./g
Angular	0.08°/sec./rad./sec.^{2}
Linearity	1/2% FS, from 0 to half scale 2% FS, half scale to full scale
Service Life	1000 hr. (min.) or one year
• Insulation Resistance	10 megohms (min.) at 50 vdc
• Damping Ratio*	0.5 to 0.9
• Natural Frequency (min.)*	35
Environments	
Shock	250 g peak sawtooth, 5 msec.
Vibration	0.1 g^2/Hz, 20-2000 Hz
Storage Temperature	-65°F to +200°F
Radio Frequency Interference	MIL-I-8161D

*Parameters that are a function of the rate sensor used
• Acceptance tested parameters

NOTES:

1. The three output signals are isolated from input common and from each other.
2. The output signals are protected from any damage occurring as a result of inadvertent shorting.
3. The standard three-axis DC/DC configuration is also available with control output of ±5 vdc.
4. There are several models of the Standard GR-G5 Rate Gyro to choose from, accommodating full-scale rate inputs from 20 to 10,000°/sec. Variable limits for natural frequency, acceleration sensitivity, threshold and resolution as a function of input rate are shown in the G5 parameter table.

E. Docking Reflectors

The Docking Reflectors are the target part of the ITT Scanning Laser Radar System developed for MSFC.

The target equipment consists of a quartet of 4" optical corner cube reflectors. The corner cube reflectors allow the Tug to acquire and track CSS out to a range of 75 nm for rendezvous. The scanning laser radar equipment is Tug-borne; the function of the CSS portion is solely to reflect the laser beam back to the Tug.

It is desirable for the CSS to correct its attitude with respect to the Tug docking LOS; four cubes provide sufficient information to the Tug to enable it to compute the CSS LOS angles and roll angle for RF transmission to CSS.

VII. REFERENCES

1. Proposal for Momentum Wheel and On-Board Processor; LMSC/A989382; June 1971
2. Lebsock, K. and Dougherty, H. - "Final Report - Wheel Attitude Control System Comparison Study;" LMSC/A965660; Feb. 1970
3. Lebsock, K., Lyons, M., and Davis, T. - "3-Axis Stabilization Study for a 500-Pound Geostationary Communications Satellite;" LMSC/A903434; July 1970.
4. Anderson, R. H., "An Advanced Horizon Sensor for Synchronous Altitude 3-Axis Stabilized Satellites;" AIAA 70-476; Apr. 1970
5. Michielsen, H., and Webb, E. - "Stationkeeping of Stationary Satellites Made Simple;" Proceedings of First Western Space Conference-Part II; Vandenberg Scientific & Technical Societies Council; Santa Maria, Calif.; Oct. 1970

APPENDIX A

COMPARISON OF WHEEL STABILIZATION AND CONTROL SYSTEM CONCEPTS

This section compares the competing stabilization and control system concepts for the COMSAT Standard Spacecraft.

THREE-AXIS SYSTEMS

The three-axis systems are essentially three parallel single-axis systems, identical in form. Schematic and block diagrams are shown in Figure A1 for the three reaction-wheel system.

The all-gas system would look similar if the wheels were deleted and hysteresis/ pulse modulator electronics substituted for the "desaturation electronics."

An attitude error sensing source is required for each of the three axes. Roll and pitch error sources include earth horizon and RF sensors; a yaw reference could be obtained from a star sensor, orbital gyrocompass, or, intermittently, from a sun sensor.

For small angle motion, the three wheel or gas channels operate essentially as independent control systems in which attitude errors are shaped to drive a corresponding reaction wheel or to actuate jets. The controller compensation contains an integral term in addition to the conventional proportional plus rate terms to limit spacecraft offset associated with external torques or internal momentum transfer between the roll and yaw wheels.

The three-axis systems are competitive in multi-mode compatibility, cost,* development status, and physical characteristics. The all-gas system will, however, exhibit a high thruster duty-cycle and comparatively poor reliability.

The yaw sensor is critical from the standpoint of life/reliability and consequences of failure. Several firms have Polaris tracker designs but only two, HI and Ball Brothers, have reached the breadboard stage. HI's will fly on NASA's ATS F&G in 1972-73.

Advanced RF polarization techniques are under development and limited flight experience recorded. This technique, however, requires an active ground station.

An active gyrocompass loop using one or two gas-bearing gyros would only be feasible if drift compensation were provided periodically since a gyro drift of as little as 0.5 deg/hr would produce a yaw attitude error of two degrees in a synchronous orbit. To avoid complicated on-board data processing, this necessary calibration could be commanded from the ground based upon readout of the sun sensors. Two gyros would probably be needed: implementation of a single-gyro orbital gyrocompass would require precision on-board attitude control electronics since the attitude control gains must be held to within very tight tolerances.

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* Provided it is feasible to use a single wheel design for all three axes.

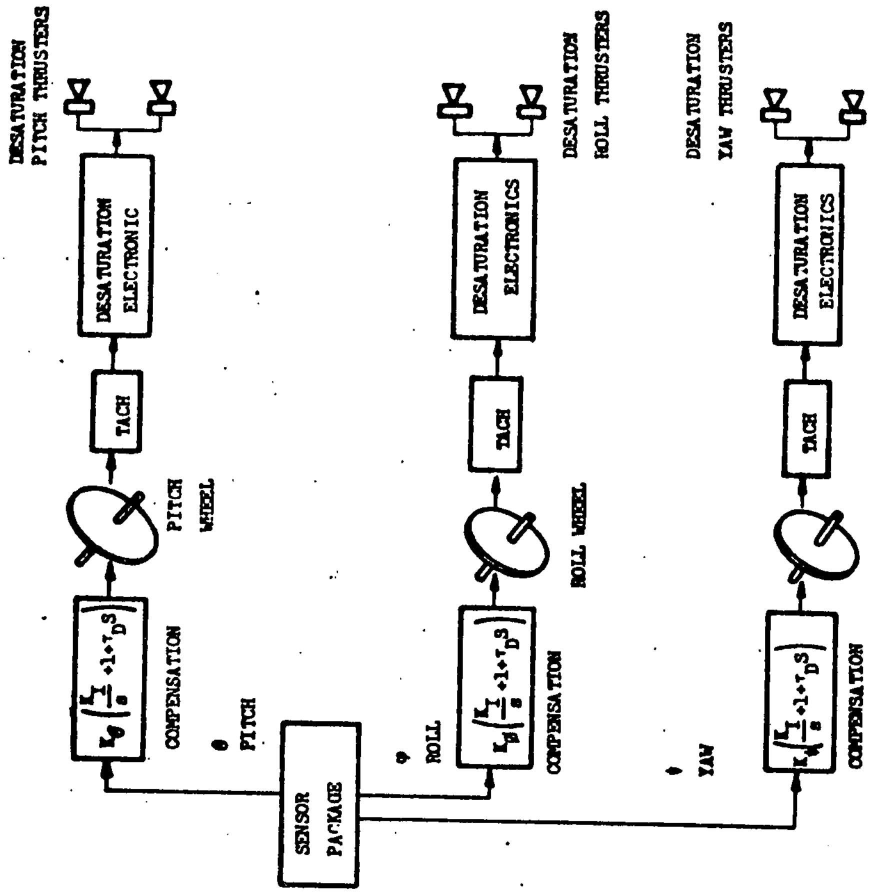

Fig. A1(b)

THREE WHEEL SYSTEM BLOCK DIAGRAM

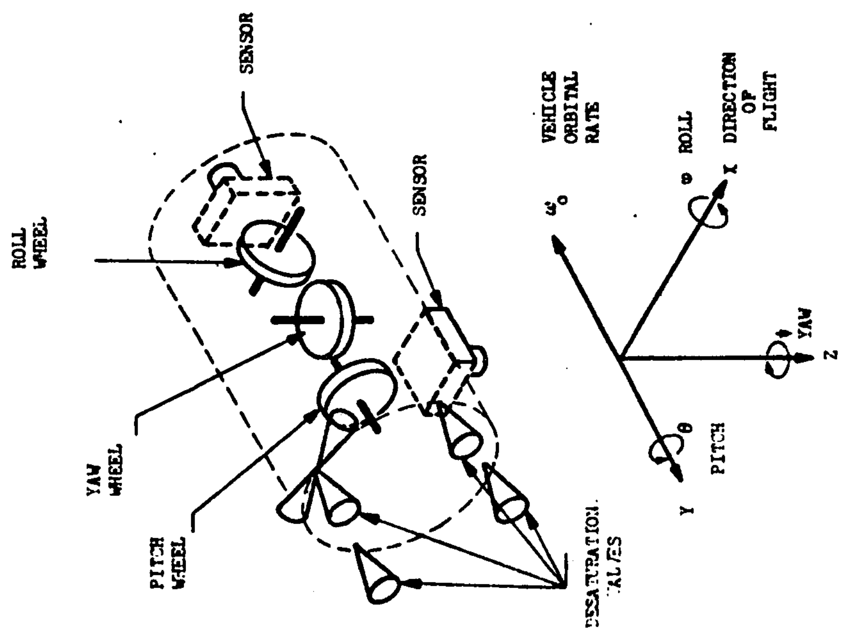

Fig. A1(a)

THREE WHEEL SYSTEM SCHEMATIC DIAGRAM

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TWO-AXIS SYSTEMS

Body-Spin Stabilized

To this date nearly all synchronous satellites have been spin-stabilized, some with de-spun antennas. More recently, the "dual-spin" satellite has appeared. This configuration retains the spinning drum-shaped body but a significant portion of the satellite is de-spun to point at earth, thereby providing a platform for the mounting of payload sensors and antennas. Much of the usual motivation for the spinning approach is lacking in the CSS specifications: in particular, the lack of synchronous equatorial orbit injection burn normally the major source of disturbance torques. The dual-spinner would be costly: for example it needs approximately 50 percent more solar cells than an oriented-array for the same power generation capability and, most important, very precise alignment and balance. Experience has shown that attitude reacquisition of a dual-spin satellite to be difficult, if not impossible, due to the tumbling motion resulting when an adverse moment-of-inertia ratio is used.

Finally, a spinning satellite must be de-spun and stabilized prior to docking at each visit of the space tug for repair or refurbishments and re-spun afterwards. The drawbacks of body spin-stabilized spacecraft can be eliminated by systems which employ momentum and/or reaction wheels in various combinations.

Wheel - Spin Stabilized

Two-axis systems which use a pitch momentum wheel in lieu of a yaw channel have successfully operated in similar orbits to CSS. This family of stabilization systems appear to be the best choice for this application.

The discussion of the different momentum wheel concepts will be restricted to the roll-yaw control channels and will progress from system to system in a building block approach to add continuity to the presentation; however, each system is a competing concept.

a) Fixed Wheel

The fixed wheel system[*] is an active attitude control system incorporating a pitch momentum wheel and mass expulsion jets. The wheel spin axis is nominally aligned normal to the orbit plane. Roll error signals from the attitude sensor are processed by a lead controller, best implemented by a pseudorate circuit, and used to actuate gas valves to supply attitude control torques about the roll and yaw axes. Restraint about the yaw axis is provided by the wheel angular momentum and, due to the gyrocompassing effects of the momentum, yaw errors are transferred into roll at orbit rate and then corrected by the cold gas thrusters. The unique feature of the one-wheel system is the use of an offset, roll-actuated control torque and the momentum wheel to control the yaw axis without direct yaw sensing.

(*) Dougherty, H.J., Scott, E.D., Rodden, J.J. - Analysis and Design of WHECON - An Attitude Control Concept, AIAA paper no. 68-461, AIAA 2nd Communications Satellite Systems Conference, San Francisco, CA, April 8-10, 1968.

The nutation - frequency mode of the momentum coupled roll-yaw dynamics is damped by the roll cold gas thruster action and, therefore, the fixed wheel system will exhibit an undamped nutation oscillation within the thruster controller dead bands. The amplitude of this nutation must be much less than the deadband to preclude violating the maximum allowable rate requirement; as a matter of fact the allowable rate amplitude would fix the minimum bit size. Reducing the bit size increases the number of thruster pulses needed over the mission life. Since the thruster action at the deadbands will continually excite and damp the nutation motion, a high thruster duty cycle can be expected and the number of thruster pulses over the 5-year life is potentially quite large. An attractive alternative to nutation damping by mass expulsion is to gimbal the wheel and torque it with a suitable error signal.

b) Single-Gimballed One-Wheel System (and Two-Wheel System)

The single-gimballed one-wheel system uses the roll error signal to drive the wheel gimbal. The gimbal axis is aligned along the roll axis and its null position is such that the spin axis of the momentum wheel is aligned as in the previous case, along the negative pitch axis. Rotating the gimbal from null produces a component of angular momentum along the yaw axis. To stabilize the orbit rate roots, an angle desaturation mechanism is needed and this mechanism would be logically the mass expulsion system. The control signal to drive thrusters would be derived from the gimbal angle.

An equivalent dynamical mechanization uses two wheels. One wheel would be a fixed pitch momentum wheel, as described before and the other a yaw reaction wheel. The linearized equations for both mechanizations are the same; the roll error signal is similarly shaped and would be used to vary either the gimbal angle or the speed of the yaw reaction wheel to damp transient motions and nutation.

The low frequency damping of this system (at orbit rate) is directly dependent upon the gain of the desaturation loop used to unload the gimbal angle (or yaw wheel). Even moderate damping ratios introduce the requirement for nearly continuous operation of the mass expulsion system; this can be avoided by using a double-gimballed wheel.

c) Double Gimballed One-Wheel System

The components of the double gimballed one-wheel system (Ref. 2) are a roll sensor, a compensation network, and a momentum wheel mounted in two-degree-of-freedom gimbals. Roll and yaw control torques are obtained by driving the corresponding gimbals. A schematic diagram of the system and a block diagram are shown in Fig. A1.

Typically, the motion of each gimbal would be relatively small. Consequently, the actual mechanization of this system permits use of a two-degree-of-freedom tilt-table (flex pivots).

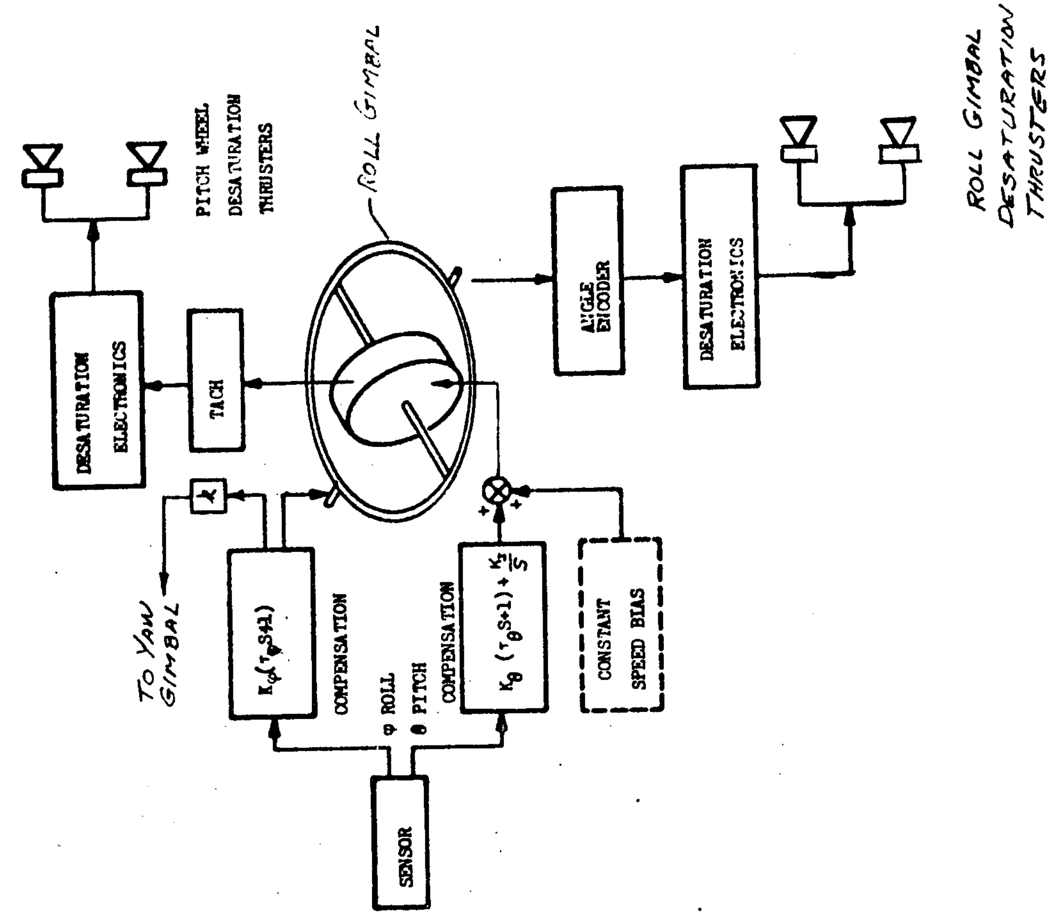

Fig. A2(b)

GIMBALLED WHEEL SYSTEM BLOCK DIAGRAM

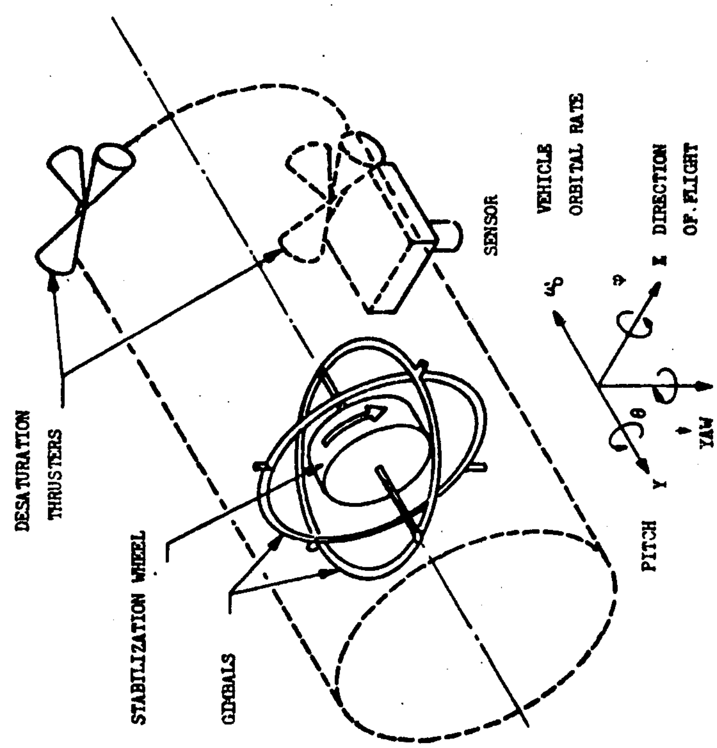

Fig. A2(a)

DOUBLE GIMBALLED WHEEL SYSTEM SCHEMATIC DIAGRAM

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Roll attitude is controlled by applying a torque on the roll gimbal in response to a roll error from the attitude sensor. Yaw is controlled by a gyro-compassing technique similar to that of the one-wheel-with-thruster system. As in that system, yaw restraining force is determined by the amount of gyroscopic coupling (the product $h\omega_o$). The yaw rate gain is produced by commanding a proportional yaw torque whenever a roll torque is commanded, similar to the one-wheel-with-thruster system where the yaw rate gain was produced by offsetting the roll thrusters into yaw by a small angle.

The dynamic behavior of the double gimballed system would, in fact, be very similar to that of the one-wheel-with-thrusters system except that the double-gimbal permits motion decoupling. Nutation motions are restricted to the wheel itself by torquing the roll gimbal; orbit rate motions are decoupled in roll by subtracting the roll error from the roll gimbal drive signal. The former allows the wheel spin axis to nutate while the satellite attitude stays fixed in space while the latter allows near-zero spacecraft roll attitude error while the orientation of the momentum wheel remains fixed, e.g., when the wheel angular velocity vector is not perfectly aligned normal to the orbit plane.

APPENDIX B

Dual-Gimbal Momentum Wheel Sizing (Ref. 2)

In this section criteria and techniques are presented to enable determination of the necessary:

(1) Wheel nominal angular momentum
(2) Angular momentum variation
(3) Gimbal angular freedom
(4) Wheel reaction torque capability
(5) Wheel weight and power

Principal factors are: disturbance torque size and character, allowable attitude deviations, desaturation interval, and runup time.

Disturbance Torques

Wheel size is primarily a function of the magnitude of the environmental disturbance torques. The principal source of disturbance torque at synchronous altitude is solar radiation pressure. Solar torque profiles for a similar (Ref. 3) spacecraft configuration were determined using a digital computer program capable of computing solar torques for a model of the spacecraft made up of plates, cylinders, conic sections, surfaces of revolution and other common geometric shapes. The various surfaces of the model can be solid, porous, or mesh, and surface shading by other parts of the spacecraft is included in the program. When sun tracking arrays are included in the model, their orientations are automatically calculated prior to the solar torque computation.

The solar torque profiles were generated by calculating the solar torque at 15° intervals around the orbit. Roll, pitch, and yaw components of the solar torque are shown on a single plot. Each component of the torque is identified by a letter repeating along the curve, X for roll, Y for pitch, and Z for yaw. The solar torques are plotted against right ascension, which is an inertially oriented orbit angle. For example, at a right ascension of zero the spacecraft is between the Earth and sun at spring equinox but a right ascension of zero at summer solstice means that the Earth-spacecraft line is at 90° to the Earth-sun line.

Minimum solar torque profiles were generated by modeling the spacecraft with perfectly balanced solar arrays. The dominant factor in sizing the amplitude of the pitch torque is the offset of the solar panel centerline and the spacecraft center of mass (approximately 20"). The spacecraft passes through the Earth's shadow in the vicinity of a right ascension of 180° and as a result all solar torques go to zero.

A minimum solar torque profile using the same model was generated at the summer solstice. The most significant difference between this case and the spring equinox case are that the roll torque takes on a constant bias and that the spacecraft does not pass through the Earth's shadow. In this example the roll bias torque is approximately 35 micro ft-lb and is caused by the constant component of solar radiation normal to the orbit plane acting on a vehicle which is asymmetrical about the roll axis.

The solar torque profiles for the preceding two cases are idealized in the sense that the two solar panels are assumed to be perfectly matched. In an actual spacecraft the solar arrays will be unbalanced due to dimensional differences, mechanical misalignments, variations in the surface properties of the solar cells, and imperfect operation of the sun tracking mechanism. It is unwieldly to model all of the array unbalance sources. An estimate of the total unbalance effect may be obtained by using a conservative model of only the most dominant sources of array unbalance. A model of the arrays with ± 10% variations is absorptivity was selected in order to simulate the overall effects of array unbalance. The absorptivities of the arrays are ideally both equal to 0.82. For the unbalanced model the absorptivity of the +Y (South) array was taken as 0.902 and the absorptivity of the -Y (North) array was set equal to 0.772 The profile was computed for the summer solstice in order to maximize the roll bias torque. The resulting solar torque profile, scaled up for CSS dimensions, is shown in Figure B1. These torques represent a conservative practical estimate of the design solar torques acting on the spacecraft and will serve as the basis for sizing the stabilization and control system.

Sizing of Momentum Wheel

The required wheel angular momentum is determined by the peak disturbance torques and by the allowable attitude errors.

Yaw Attitude Error Criterion

The peak yaw angle due to an external torque is given approximately by:

$$\psi \approx \frac{57.3\ T_z}{\omega_o h}$$

where the yaw angle, ψ, is in degrees, the torque, T_z, is in ft-lb, the angular momentum, h, is in ft-lb-sec, and orbit rate at synchronous altitude is $\omega_o \approx 7.29 \times 10^{-5}$ rad/sec. This equation can be used to size the momentum wheel for a given yaw torque and allowable yaw offset, based on the assumption that the external torque is constant in the orbit frame.

Figure B2 is a plot of this equation for synchronous orbit rate. It shows that for the peak yaw torque of 60 $\times 10^{-6}$, the required wheel angular momentum is 100 ft-lb-sec. With this size wheel, the yaw angle would oscillate ± 0.40 deg daily.

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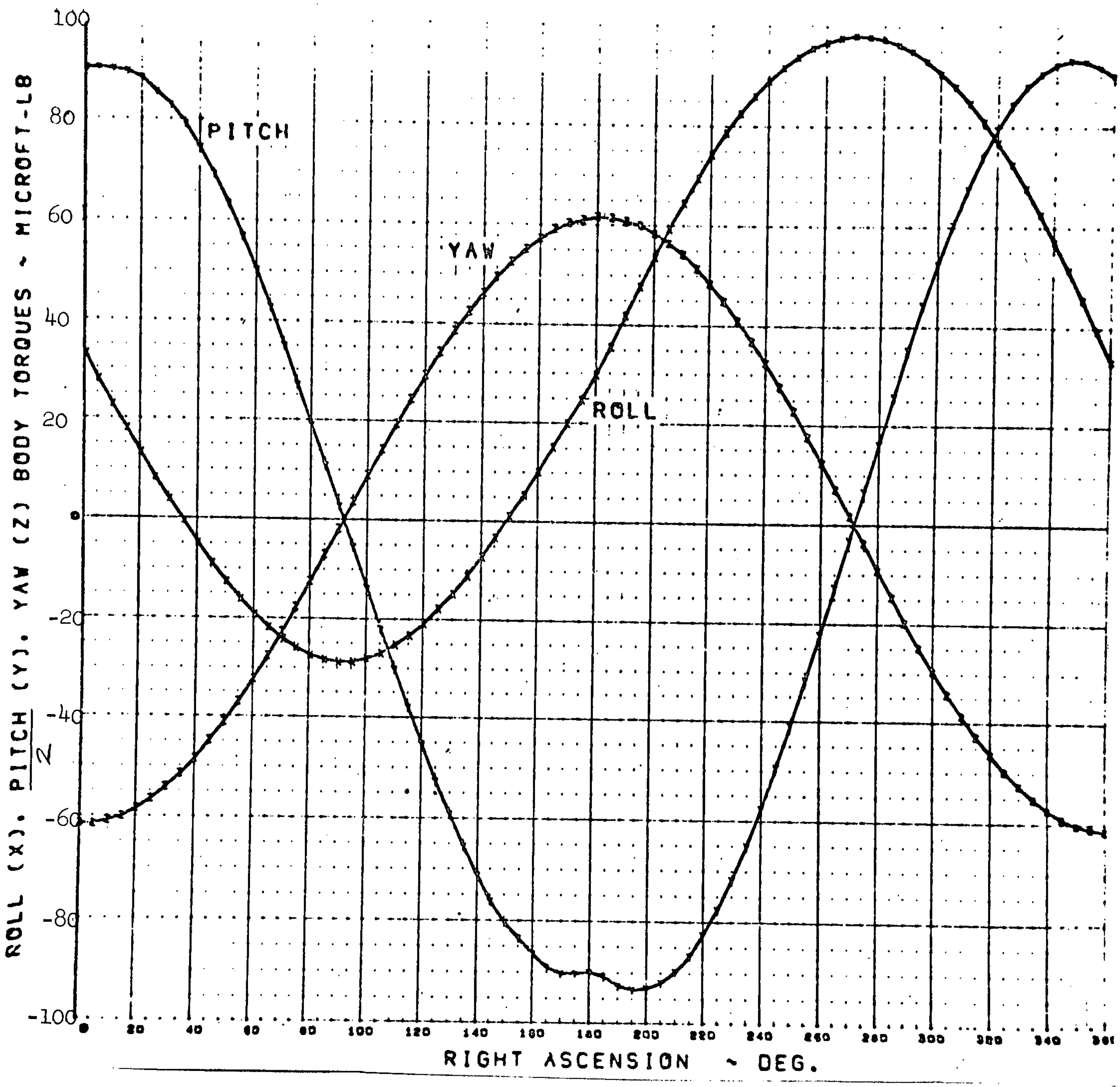

Fig. B1 - Design Solar Torque vs Orbit Angle (Summer Solstice)

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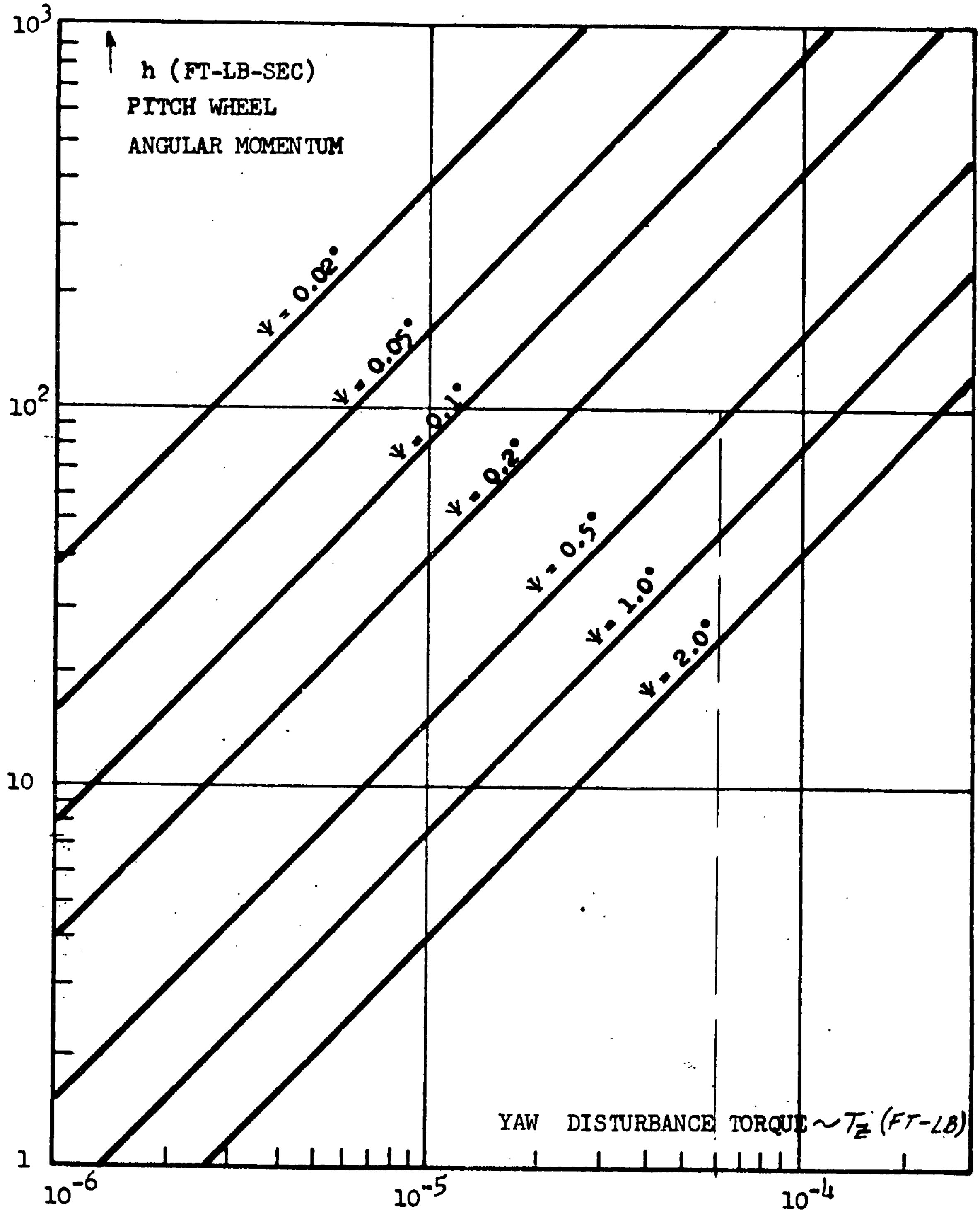

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Fig. B2 - Selection of Momentum Wheel Size

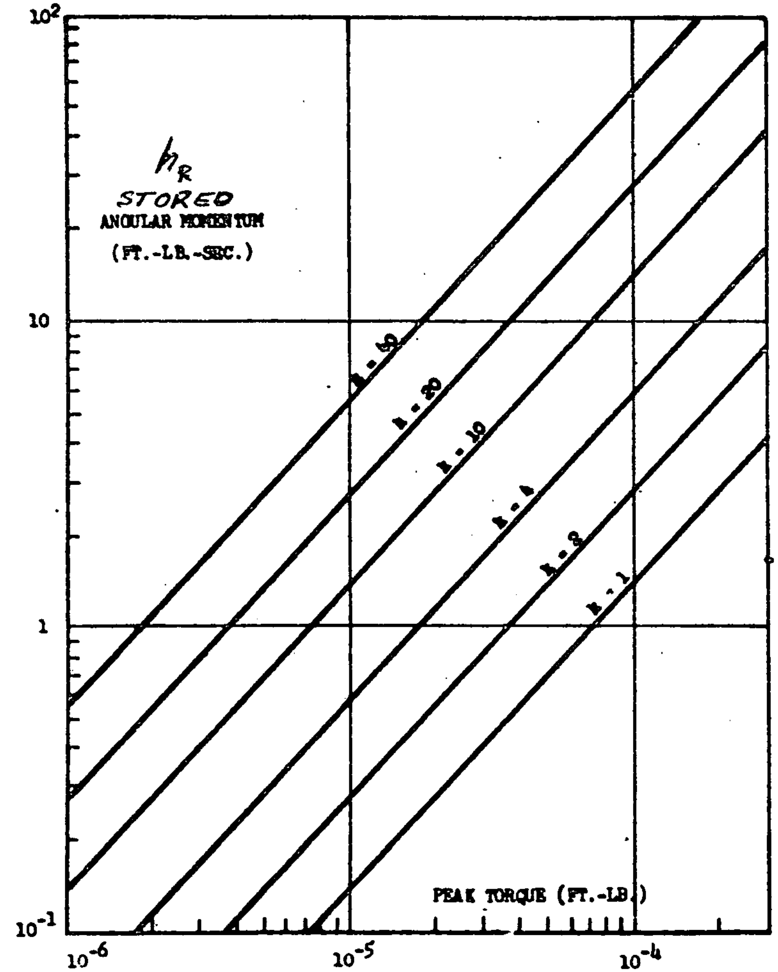

Fig. B3(a) Selection of Wheel Size

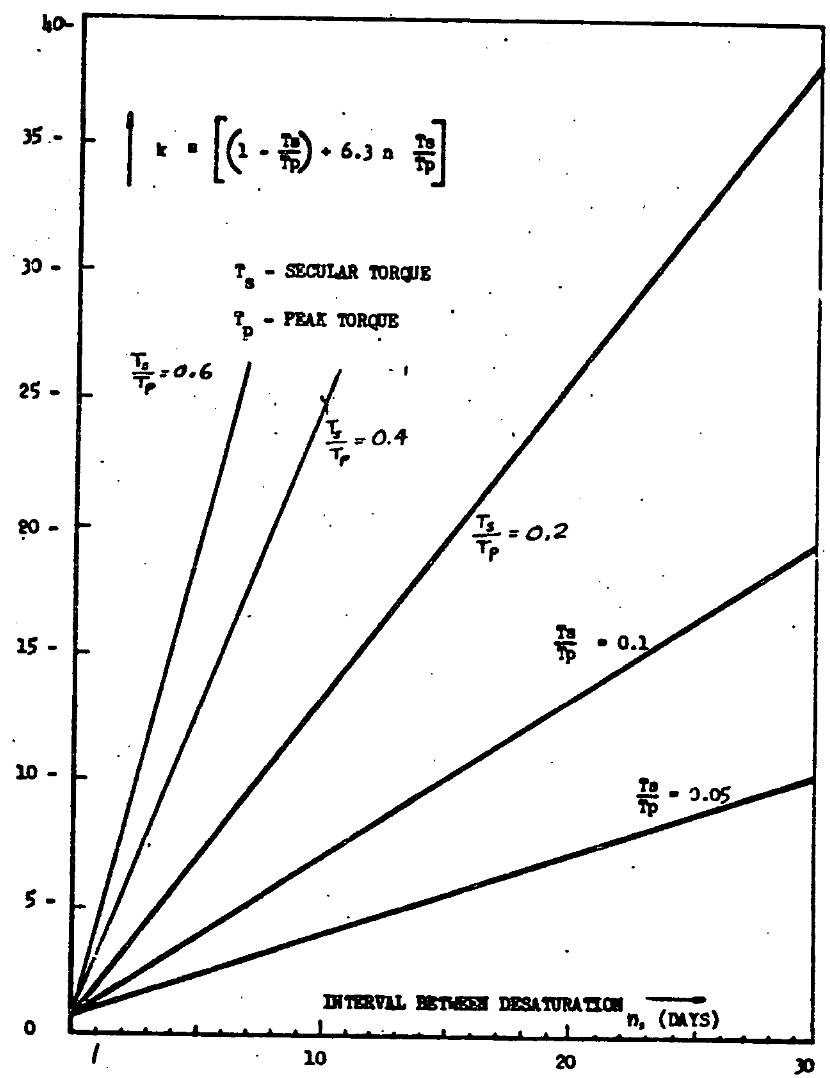

Fig. B3(b) Definition of k Factor

Gimbal Desaturation Interval Criterion

An inertially-fixed roll solar torque will cause the gimbal angle to oscillate at orbit frequency, while a body-fixed torque will cause a gimbal angle offset. When the sum of these cause the wheel gimbal to reach the limit of its angular travel, the mass expulsion system must be used to unload, or desaturate it. The magnitude of the roll angular momentum at this time then is a function of external torques and frequency of momentum dumping. A conservative estimate of the required angular momentum is given by:

$$h_R = \frac{T_c}{\omega_o} + 8.64 \times 10^4 \, T_s n \qquad (1)$$

where T_c and T_s are the magnitudes of the body- and inertially-fixed torques, the frequency of desaturation in days is given, by n, and ω_o is orbit rate.

The peak torque on an axis is given by $T_p = T_c + T_s$. Substituting for T_c, obtain:

$$h_R = \frac{T_p}{\omega_o} \left[1 - \frac{T_s}{T_p} + 6.3n \frac{T_s}{T_p} \right] \qquad (2)$$

or

$$h_R = k \frac{T_p}{\omega_o} \qquad (3)$$

Figure B3, a plot of h_R and k, allows the required roll angular momentum storage to be estimated. For the estimated roll solar torques:

$$T_s = 35 \times 10^{-6} \text{ ft-lb}$$
$$T_p = 98 \times 10^{-6} \text{ ft-lb}$$
$$\frac{T_s}{T_p} = 0.36, \quad \frac{T_p}{\omega_o} = 1.35$$

From Figure B3:

n (Days)	k	h_R (ft-lb-sec)
1	3	4
3	7	10
5	12	16
7	17	23

Roll control torques are obtained by rotating the wheel about the roll gimbal axis. The angular momentum available in roll is the product of the wheel angular momentum and the allowable gimbal travel, $H_N \gamma_R$. The roll gimbal angle required is thus given by: $\gamma_R \,(\mathrm{RAD}) = \frac{h_R}{H_N}$. Rather than calculate the gimbal angle in this way, the preferred approach is to limit the gimbal angle to a small angle to assure a simple design and then compute the required wheel angular momentum, H_N, in ft-lb-sec is:

		$H_N = h_R/\gamma_R$ (ft-lb-sec)	
n (Days)	h_R (ft-lb-sec)	γ_R = 0.1 (RAD)	γ_R = 0.2 (RAD)
1	4	40	20
3	10	100	50
5	16	160	80
7	23	230	115

The table indicates that a 100 ft-lb-sec wheel would be a good choice to assure a low unloading duty cycle (≈ 5-day intervals) for a modest design gimbal travel (≈ 10 deg).

East-West Stationkeeping Criterion

The yaw torque due to thruster misalignment is opposed mainly by the gyroscopic precession torque of the wheel. The yaw angle induced during the maneuver is given by:

$$\Delta\psi_{Sk} = 57.3 \frac{m \Delta V \delta}{H} \text{ in deg,}$$

where:

m = spacecraft mass = 140 slugs

ΔV = velocity imparted during stationkeeping = 0.48 fps daily

δ = thrust vector-to-spacecraft c.g. misalignment = 0.85 inches, conservatively.

H = wheel angular momentum in ft-lb-sec

For a 100 ft-lb-sec wheel momentum:

$$\Delta\psi = 57.3 \frac{140 \times 0.48 \times \frac{0.85}{12}}{100}$$

$$= \underline{\underline{27}} \text{ deg}$$

∴ A relatively crude thrust-to-cg alignment and a daily stationkeeping interval causes a yaw excursion of about 7 times that allowed.

Momentum Variation Capability Required

To absorb the daily pitch solar torque without mass expenditure, the wheel momentum (speed) change is given by (Fig. B4):

$$\Delta H_{max} = \int_0^t Ty \; dt$$

$$= \int_0^t Ty_{max} \cos \omega_o t \; dt$$

$$= \frac{Ty_{max}}{\omega_o} \sin \omega_o t \Big|_0^{\pi/2} \text{, Ft-Lb-Sec}$$

$$= \frac{93 \quad x\ 10^{-6}}{7.3\ x\ 10^{-5}} = \quad 1.27 \text{ Ft-Lb-Sec}$$

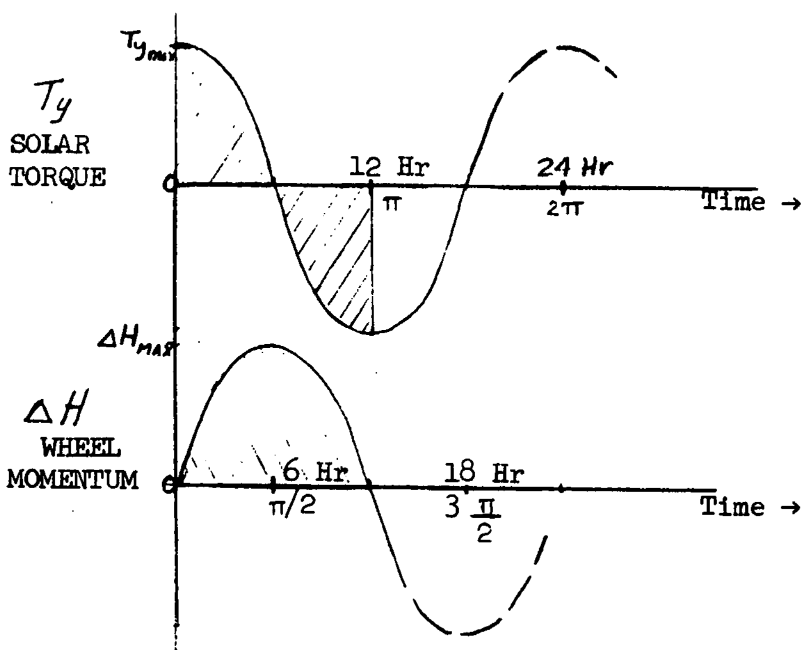

FIGURE B4

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Therefore, the required momentum storage capability of the wheel is ± 1.27 ft-lb-sec, a speed variation easily implemented by simple wheel speed control electronics.

Weight and Power of Momentum Wheels

A rule-of-thumb relationship of wheel, housing, and associated electronics weight to the angular momentum for both reaction wheels and momentum wheels is given by:

$$W = 7\ h^{0.4}$$

where W is the weight in pounds and h is the angular momentum in ft-lb-sec. The relationship is based on a fit to published vendor data. It agrees well with other published results (*). Figure B5 is a plot of this equation. It shows that the weight of a momentum wheel not including the drive electronics, gimbals or gimbal actuation. A 100-ft-lb-sec wheel will thus weigh about:

Wheel Assembly (2)	69 lb
Wheel drive electronics	6 lb
Gimbals (or pivots) & actuation	10 lb
Double-gimbal Wheel-Total	85 lb

Approximations to peak power, P_f, and average power, P_A, for the wheel are also based on consideration of vendor data.

$$P_p = 250\ T_{MP}$$
$$P_A = 1.2\ T_M + 1.5 + 0.08\ h$$

where T_{MP} is the peak motor torque, T_M is the torque supplied by the wheel to balance external torques, and h is the wheel momentum. The units of T_{MP} and T_M are ft-lb, to be consistent with the previous units of environmental torques, although a motor designer would probably use units of oz-in. The equations assume a peak wheel speed in the range of 2000 to 3000 rpm, and a 90 percent motor efficiency. The electronics are assumed 80 percent efficient with a standby power of 0.5 watts. The running power for the wheel is given by the expression: (1 + 0.08 h).

The torque during run-up is:

$$T = \frac{\Delta H}{t_{runup}} = \frac{100-0}{10 \times 60} = 0.17 \text{ ft-lb}$$

for a 10-minute run-up time. Run-up power is then 43 watts and running power should be less than 12 watts.

(*) Auclair, G. F., Advanced Reaction Wheel Controller for Spacecraft Attitude Control, AIAA Guidance, Control and Flight Dynamics Conference, Princeton, New Jersey, 1969

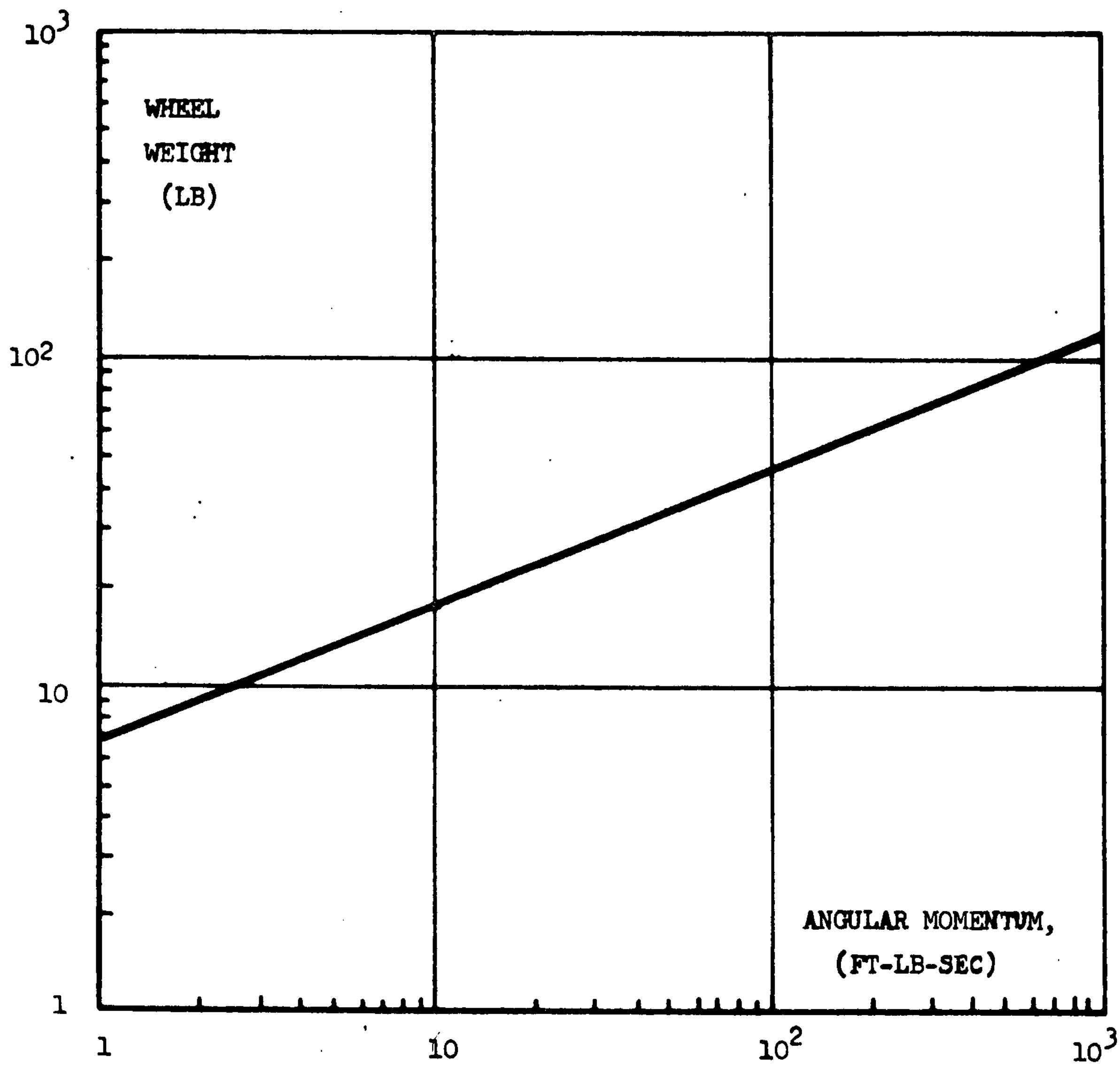

Fig. B5 - Weight of Momentum and Reaction Wheels

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DATE: 24 Dec 1971

APPENDIX C

BREAKDOWN OF ATTITUDE CONTROL REQUIREMENTS

1. Impulse Load

a. Translation (W = 4,500 lb)

SEO ΔV requirements are:

		(FPS)
(i)	Vernier tug injection errors after ground tracking	6
(ii)	Arrest 3 deg/day drift to final station	30
(iii)	East-West (longitudinal) stationkeeping (14 FPS/yr*)	70
(iv)	North-South (latitudinal) stationkeeping (168 fps/yr.)	840
	Total ΔV required	946 FPS

$$I_T = M\Delta V$$

$$= \frac{4500}{32.2} \times 964 = \underline{\underline{132,500 \text{ lb-sec}}}$$

b. Rotation

(i)	Pitch momentum dump .... (200 solar eclipses)	100
(ii)	Roll/yaw momentum dump .... (Roll secular torque = 3×10^{-5} ft-lb)	600
(iii)	Translation thrust misalignment .... (1" CT-CG offset)	2800
(iv)	Docking attitude control .... ($\pm$ 0.5° limit cycle for 1 hour following three one-deg/sec slews)	100
(v)	Reacquisition .... (arrest 3 deg/sec rate and perform 3 one-deg/sec slews)	100
(vi)	Backup attitude hold .... ($\pm$ 10 deg limit cycle for 60 days)	100
	Total	3800 lb-sec

∴ Total impulse required = 136,300 lb-sec

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* Ref. 5

2. Minimum Bit

Roll Criterion - During unloading, one pulse precesses attitude less than the deadband angle:

$$\frac{\Delta H}{H} \le DB$$

$$\ell \Delta I_T \le HDB$$

$$\Delta I_T \le \frac{(100)\left(\frac{0.1}{57.3}\right)}{8} = \underline{\underline{0.022}} \text{ lb-sec}$$

Pitch Criterion - During unloading, one pulse disturbs attitude less than deadband when opposed by constant wheel reaction torque:

$$\frac{\Delta W^2}{2\ddot{\theta}} \le DB \qquad \text{where: } \ddot{\theta} = \frac{T_{MAX}}{I_Y}$$

$$\text{and: } \Delta W = \frac{\ell \Delta I_{TOT}}{I_y}$$

$$\Delta I_T \le \frac{(2\, T_{MAX} I_Y DB)^{\frac{1}{2}}}{\ell}$$

$$\le \left(2 \times \frac{20}{192} \times 1500 \times \frac{0.1}{57.3}\right)^{\frac{1}{2}} \frac{1}{8}$$

$$\le \underline{\underline{0.07}} \text{ lb-sec}$$

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3. Thrust

Criterion - Stop tumbling within 30 sec of power failure and restoration.

$$\Delta W = \frac{\Delta H}{I} = \frac{100 \times 57.3}{1500} \cong \underline{\underline{4 \frac{\text{deg}}{\text{sec}}}}$$

$$F\ell\Delta t = \Delta H$$

$$F = \leq \frac{100}{8 \times 30} = \underline{\underline{0.42}} \text{ lbf}$$

Criterion - If continuous thrust is used to unload, do not exceed wheel torque capabilities:

Pitch: $T_{MAX} = 20 \text{ oz-in} \cong 0.1 \text{ ft-lb}$

Roll/Yaw: $T_{MAX} = \frac{0.3}{57.3} \times 100 \cong 0.5 \text{ ft-lb}$

4. Duty Cycle - Unloading

Pitch: Every 5 days
2 lb-sec @ 8 ft

Roll/Yaw: Every 5 days
2 lb-sec @ 8 ft

If pulsed, average torque must be less than given by (3) above.

APPENDIX D

RELIABILITY

The reliability goal for the Stabilization and Control Subsystem of the CSS is 0.933 for five years of orbit life. This goal has been attained in the designs described as shown in Fig. D1.

Stabilization & Control Subsystem Reliability Goal = 0.933 for five years of orbit life

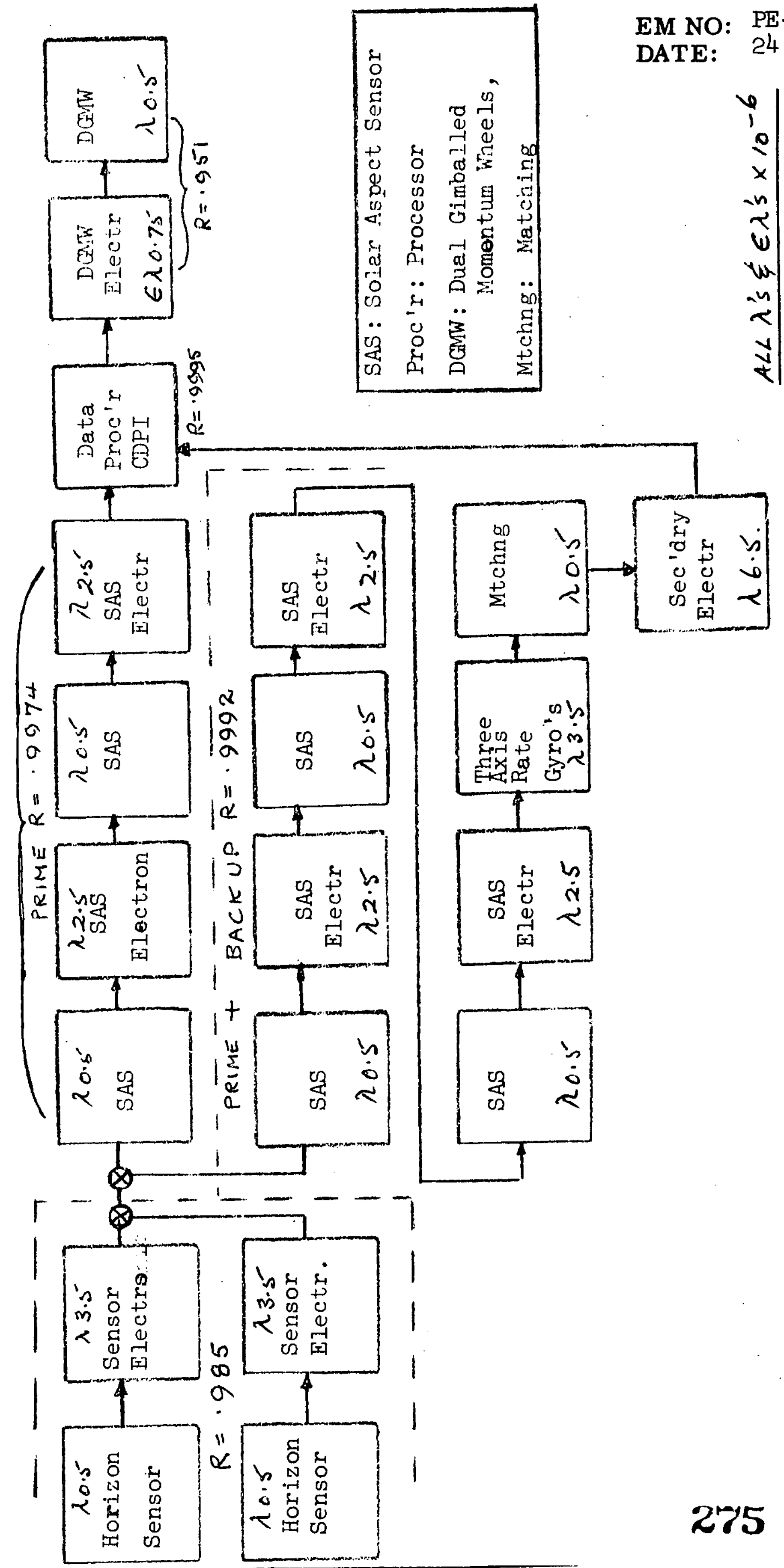

Fig. D1a Reliability Block Diagram - CSS Stabilization & Control Subsystem (Sheet 1 of 2)

• HORIZON SENSOR COMPLEX. $R = \cdot 985$	REDUNDANCY MODE:	STAND-BY. $R = e^{-\lambda t}[1+\lambda t]$ DUTY CYCLE 100% $\Sigma\lambda$ 4.0×10^{-6} $t = 43800$ HRS
• SOLAR ASPECT SENSORS PRIMARY LOOP PLUS SOLAR ASPECT SENSORS & 3 AXIS RATE GYRO SECONDARY LOOP.	REDUNDANCY MODE FUNCTIONAL BACK UP. $R(FBU) = 0.99998$	F.B.U. $R_1 + R_2 - (R_1 R_2)$ DUTY CYCLE PRIMARY LOOP < 1.0% SECONDARY " < 0.10%
• DATA PROCESSOR IS SCORED IN CDPI S/S. 25 COMMANDS OF LOGIC ARE CHARGED AGAINST S/C.	LOGIC BACKED BY REDUNDANCY IN CDPI, ACTIVE & MULTIMODAL.	$R_{LOGIC} = \cdot 9995$
• DUAL GIMBALLED MOMENTUM WHEELS & ELECTRONICS	REDUNDANCY MODE FULL ACTIVE WITHIN EQUIPMENT. DUAL CHANNEL R for LOOP = ·951.	DUTY CYCLE 100% $\Sigma\lambda$ ELECTRONICS: 0.75×10^{-6} λ for WHEELS 0.5×10^{-6} FROM BENDIX, AND G.E.C. $\Sigma\lambda$ for LOOP 1.25×10^{-6}

• R for subsystem (worst case) $R_S = R_1 R_2 R_3 \cdots R_n$

$= (\cdot 985)(\cdot 9998)(\cdot 9995)(\cdot 951)$

$= 0.936$

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• SUBSYSTEM MEETS OBJECTIVE. $R_{CAL} = \cdot 933$ (ALLOCATED) $R_{predicted} = \cdot 936$

Fig. D1b Reliability Block Diagram - CSS Stabilization & Control Subsystem (Sheet 2 of 2)

PE-123

STANDARD U.S. DOMESTIC

COMMUNICATION SATELLITE

COMMUNICATIONS, DATA PROCESSING

AND INSTRUMENTATION SUBSYSTEM

PE-123

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: STANDARD U.S. DOMESTIC COMMUNICATION SATELLITE - COMMUNICATION, DATA PROCESSING AND INSTRUMENTATION SUBSYSTEM	EM NO: PE-123 REF: DATE: 24 Dec 1971
AUTHORS: M. Loeb — Prepared under cognizance of: Payload Integration, Orgn. 69-02 Space Systems Division	APPROVAL: J.E. Bolton ENGINEERING: Wayne Miller

50 pages

PRELIMINARY

Table of Contents

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1.0 Introduction

This report is prepared to present a representative design of a standard Communication, Data Processing, Interface and Instrumentation Subsystem (CDPI), the second in a series of such designs being developed in support of the Payload Effects Follow-On Study. The objective of this study is to establish and demonstrate principles and techniques for realizing low-cost spacecraft configurations, considering Shuttle/Space Tug launch and revisit - an approach which provides capabilities for spacecraft pre- and post-injection checkout and in-orbit part replacement or return to earth for refurbishment. As shown in the previous design report, Reference 1, the principles of standardization and integrated designs, coupled with relaxations in component requirements afforded by Shuttle utilization, are primary approaches to be used in attaining a low-cost standard CDPI.

The CDPI provides the command, ranging and telemetry processes for the spacecraft; it serves as the major interface entity between the various spacecraft sections and subsystems, e.g., Stabilization and Control and Attitude Control. It also contains the computational and on-board decision-making processes for the spacecraft, as well as the status monitoring instrumentation for indicating spacecraft condition or malfunctions. A block diagram of the CDPI is shown in Fig. 1.

The CDPI in this report is for a communication satellite, operating at synchronous altitude. A spacecraft system of this type, and, hence its CDPI, must meet high reliability requirements because of two main reasons:

- Shuttle visits for replacement or refurbishment would be infrequent - five years between visits is assumed.
- Degradation or loss of communication services provided via the spacecraft and its mission equipment have significant cost penalties.

In order to realize the needed high reliability, two somewhat conflicting approaches may be used -- minimizing spacecraft on-board equipment and capabilities through maximum utilization of the communication link and ground control processes, vs. expansion of on-board capabilities to attain maximum spacecraft functional independence. The former approach would have the following advantages and disadvantages:

Minimization of On-Board Capabilities

Advantages	Disadvantages
Lower spacecraft cost for defined mission	Higher cost ground support configuration
Less on-board redundancy required to attain equivalent spacecraft reliability	More extensive ground computations and other processes.
Easier to maintain and repair spacecraft	More susceptible to command link problems and link introduced errors. On-board Shuttle test equipment requirements more extensive.

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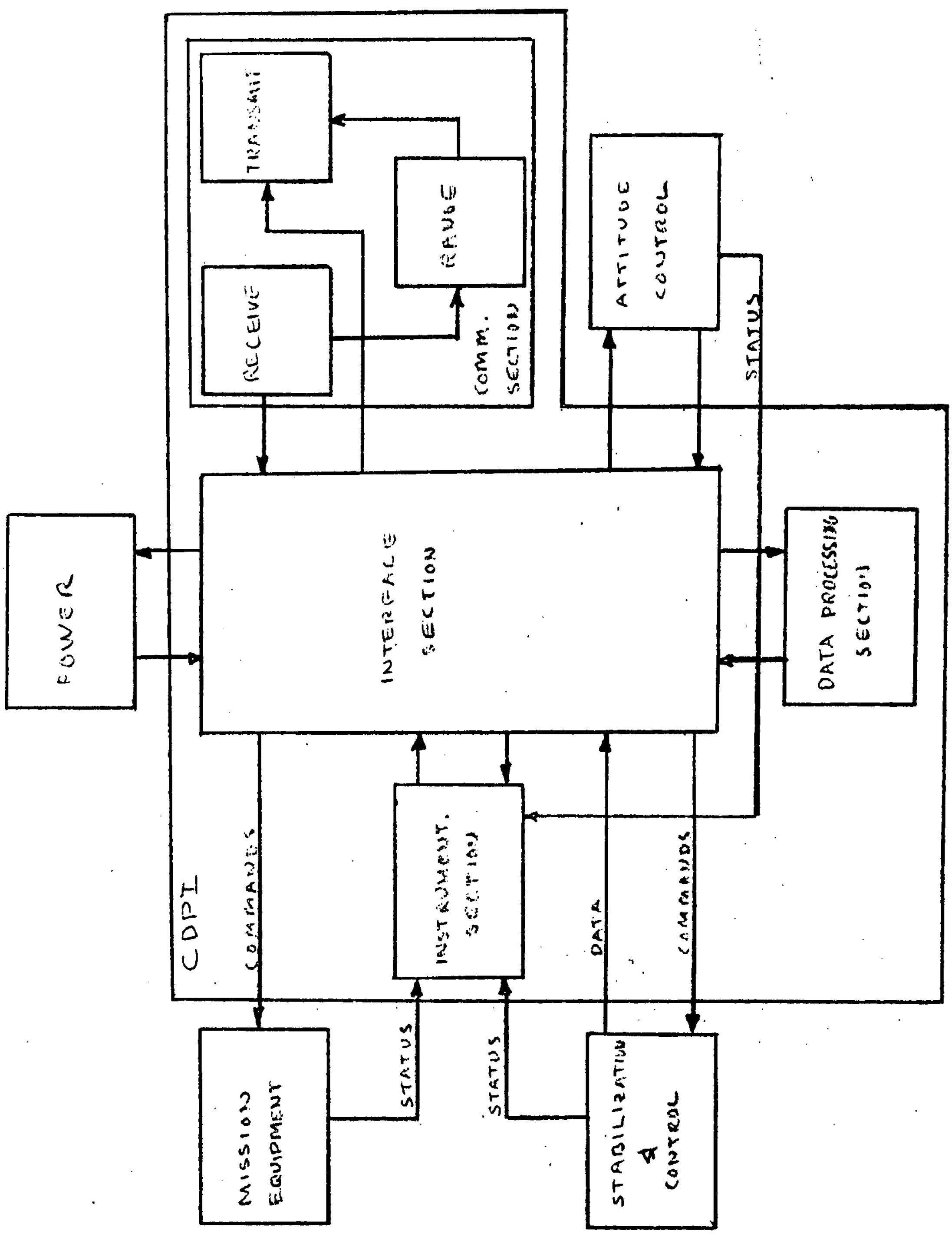

Fig. 1 Standard Communication Satellite - CDPI Block Diagram

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The latter approach would result in a somewhat reversed situation:

Maximization of Spacecraft Functional Independence

Advantages	Disadvantages
Much less susceptible to command link problems	Higher spacecraft cost for defined mission
Less ground processing required	More on-board redundancy required to realize reliability goals
Lower cost ground configuration	
More flexible - more readily adaptable to different configurations or missions	Increased size, weight and power
Simpler Shuttle test equipment configuration	

The necessity to maintain maximally error-free communication services, coupled with the desirability to provide a standard spacecraft which would satisfy other synchronous altitude missions, indicates that the latter approach would be preferred. This choice will be assumed for the point design of the CDPI developed in this report. It should be noted that the relative costs of the two approaches pertain to the communications mission. If the spacecraft is to satisfy other missions as well, the cost advantage would accrue to the more flexible spacecraft configurations.

The reliability requirements imposed on the communication satellite CDPI are much more stringent than those demanded in the first CDPI point design. This results in the following additional functional requirements, compared to the earlier design:

- increase in redundancy required
- broader failure recognition and compensation capability, e.g., inclusion of compensation of multiple failures through on-board detection and control.
- more extensive verification of commands prior to execution.

These additions will necessitate changes in hardware and software design, but the basic design principles of standardization and integrated design still pertain. Since these principles are considered in detail in Reference 1, they will not be discussed here. There is one exception, however, that should be briefly examined in this report, i.e., exploitation of design commonalities.

A basic tenet of integrated design is to gain simplification through utilization of functional commonalities. In the communication satellite, the mission equipment consists of antennas, receivers and transponders - components with considerable commonality with the communication section of the CDPI. This would suggest utilization of mission equipment for CDPI functions; specifically, uplink command and ranging signals and downlink telemetry may be integrated with the data handling channels of the mission equipment. This would result in a reduction in the number of antennas required, and the elimination of CDPI preselector and transmitter circuitry.

The difficulty with this approach is that the components which are eliminated would have to be replaced for spacecraft missions other than ground communication linking, e.g., earth resources investigations. It is possible to employ the mission equipment for CDPI functions as a design variant; however, this would necessitate some changes in the mission equipment configuration, as well as the CDPI itself.

A resolution of these alternatives would require a detailed trade study; however, it is necessary to establish the impact of utilizing a separate communications section for the CDPI in any case. For this reason use of a communication section which is independent of the mission equipment will be assumed in this point design. Resolution of the tradeoff will be reserved for future study.

2.0 CDPI Functional Requirements

2.1 Mission Equipment - Description

The CDPI subsystem is primarily supportive to other subsystems and sections of the spacecraft. Thus the CDPI functional requirements are determined by the demands imposed by the equipment served by the CDPI. In order to establish the requirements on the CDPI fixed by mission equipment needs, a mission equipment subsystem, which is representative of the latest designs, is presented here to serve as a model (Reference 2-4).

The defined mission equipment is based upon the following assumptions:

• Coverage	contiguous U.S.
• Antenna Pointing Accuracy	$\pm 0.16^{\circ}$
• Number of Channels	24
• Channel Bandwidth	36 MHz
• Carrier Separation	40 MHz
• Receive Frequency Range	5.925 - 6.425 GHz
• Transmit Frequency Range	3.7 - 4.2 GHz
• Beam Edge Receive G/T	-4.5 db/$^{\circ}$K (1595°K attainable)
• Beam Center EIRP Xmit	37.5 dbw
• Beam Edge EIRP Xmit	34.5 dbw
• Antennas (shaped beam)	
Receive -	20" x 40" Elliptical 3.5 x 7.0 degree -3 db beamwidth 30.5 db peak gain, relative to isotropic
Transmit	30" x 60" elliptical 3.5 x 7.0 degree -3 db beamwidth 30.5 db peak gain, relative to isotropic

The performance budget summary at edge of coverage area and low earth elevation angle, assuming station EIRP at 85 dbw (1.3 Kw at ground antenna feed - 32 ft. dish assumed) is as follows:

Uplink

EIRP	85 dbw
Satellite GK	4.5 db/^{o}K
Path Loss	200.1 db
C/N in 36 MHz	
Noise Bandwidth	33.5 db

Downlink

EIRP	34.5 dbw
Earth Station G/T	33 db/^{o}K
Path Loss	196.6 db
Receive Equivalent	
Noise Bandwidth	36 db
C/N	24 db
System C/N	23.5 db

Analysis yields the following margins, as seen in Reference 2.

Telephone	13.9 db
TV	13.9 db

A block diagram of the mission equipment is given in Fig. 2. This configuration is based upon the designs of References 3 and 4. In the design single step down conversion is used, rather than conversion and multiplexing at IF frequencies, to eliminate some of the problems the older technique affords. This approach is most modern and within the state-of-the-art; it is therefore used here. A list of components, with estimates of size, weight and power is shown in Fig. 3.

2.1 Mission Equipment Requirements

The mission equipment described in Fig. 2 and 3 imposes the following estimated requirements on the CDPI:

- Control Discretes

RF Switches	- Receiver	-	20
	Transponders	-	96
Applied voltage control			
	- Receiver	-	40
	- Transponders	-	420
			576 total discretes

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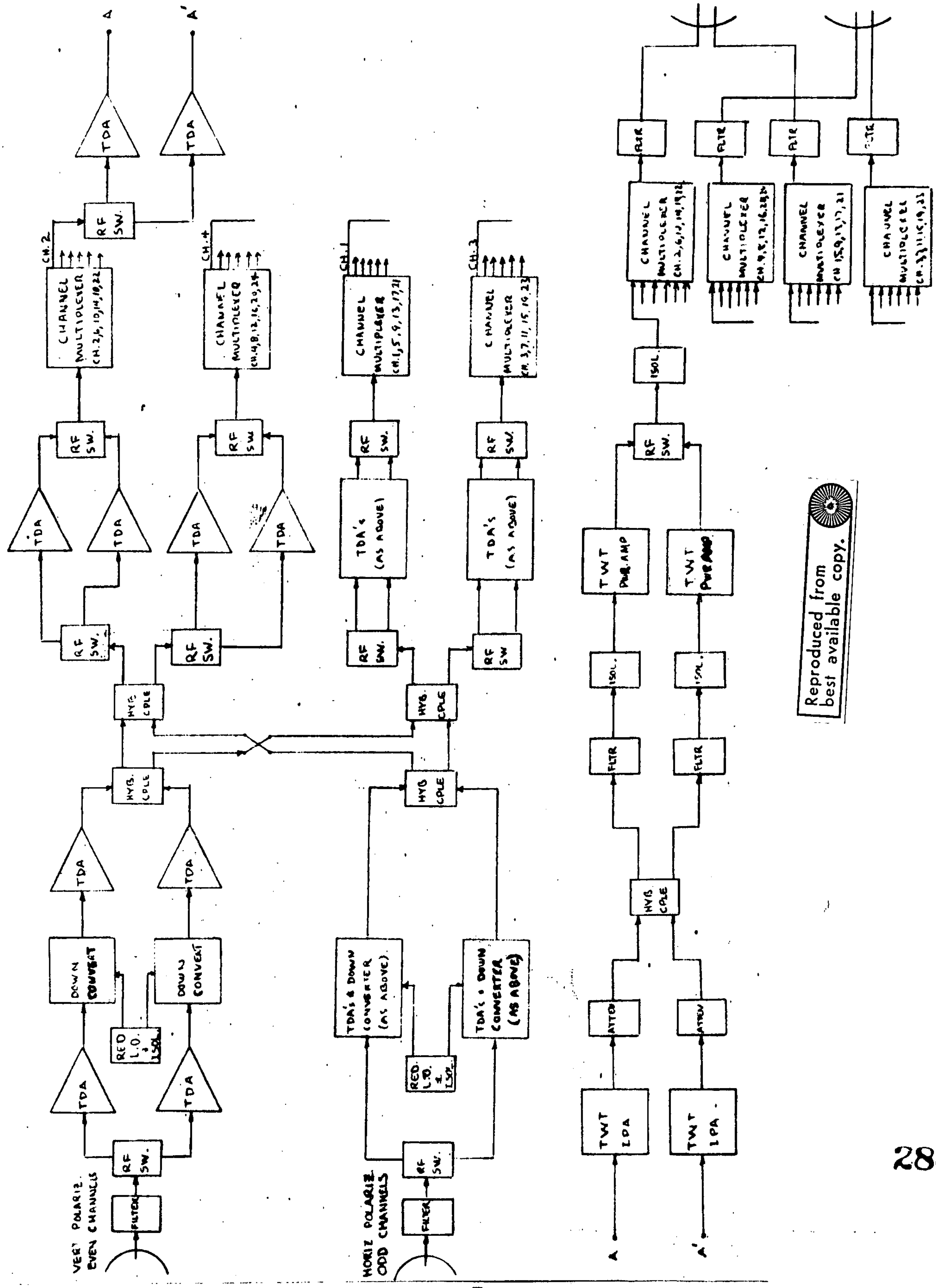

Fig. 2 Standard Communication Satellite Mission Equipment

Item	No.	Basic Size in Inches	Overall Size in in^3	Weight in lbs. Unit	Weight in lbs. Total	Power in Watts Unit	Power in Watts Total
Antenna Assy (Rec)	2	20 x 40 x 22	N/A	8	16	-	-
Antenna Assy (Xmt)	2	30 x 60 x 24	N/A	10	20	-	-
Waveguide	-	1 x $\frac{1}{2}$ x 120	N/A	1	1	-	-
Preamp. Input Filter	2	2.5 x 1 x 0.6	3.0	.2	.4	-	-
RF Switch	2	0.7x1.7x1.8	4.3	.3	.6	28 VDC 60 ma 10 ms	
TDA	8	.75x2.25x2.38	32.0	.33	2.6	.45	3.6
Loc. Osc.	4	2 x 2 x 4.5	72.0	.8	3.2	1.75	7.2
Isolators	4	2 x 1 x 0.5	4.0	.1	.4	-	-
Freq. Chan. Conv.	4	3 x 4 x 0.5	24.0	.2	.8	1.0	-
Hybrid Coupler	4	2 x 1 x 0.6	4.8	.04	.2	-	-
TDA	8	.75x2.25x2.38	37.5	.33	2.4	.45	3.6
RF Switch	8	0.7x1.7x1.8	17.0	.3	2.4	28 VDC 60 ma 10 ms	-
Multiplex (Rec)(1)	4	250.3 in^3	1010.7	4.3	17.2	-	-
		Subtotal	1209.3		67.2		14.4
(1)							
Circulator	24	1.1x1.38x0.71	26.0	.2	4.8		
Filter	24	1.4x1.15x2.78	924.0	.4	9.6		
Equalizer	24	1.1x0.71x3	51.0	.1	2.4		
Dummy Load	4	3x1.15x0.7	9.7	.1	.4		
			1010.7	.8	17.2		

Fig. 3 Standard Communication Satellite Mission Equipment

Item	No.	Basic Size in Inches	Overall Size in in^3	Weight in lbs. Unit	Weight in lbs. Total	Power in Watts Unit	Power in Watts Total
RF Switch	24	0.7x1.7x1.8	51.5	.3	7.2	28 VDC 60 ma 10 ms	
TDA	48	.75x2.25x2.78	801.0	.33	16.0	.45	22
Driver TWT	48	3 x 3 x 12	5200.0	4.1	196.8	3.0	144
Atten.	48	1.15x0.7x2	77.0	.1	4.8	-	-
Hybrid Coupler	24	2.5x1.25x0.5	38.0	.3	7.2	-	-
Filter	48	2.5x1.25x0.5	76.0	.2	9.6	-	-
Isolator	48	1x1.5x0.7	50.4	.1	4.8	-	-
Output TWT	48	3 x 3 x 12	5200.0	5.5	264.0	30 W	1440
RF Switch	24	0.7x1.7x1.8	51.4	.3	7.2	28 VDC 60 ma 10 ms	
Insulator	24	1x1.5x0.7	25.2	.1	2.4	-	-
Multiplexer (Xmt)(2)	4	100 in^3	400.0	1.5	6.0	-	-
Output Filter	4	6 x 2 x 1.5	72.0	.5	2.0	-	-
			12042.5 + 1209.3		528.0 +67.2		1606 14.4(3)
						2	1620.4
		Total	13251.8		595.2		810.2 req'd.

(2)	Filter	24	1.75x2.38x6.75	386	.2	4.8
	Circulator	4	1.1x1.38x0.71	4.3	.2	.8
	Dummy Load	4	3x1.15x0.7	9.7	.1	.4
				400.0	.5	6.0

(3) Not operational power - 810.2 w req'd.

Fig. 3 Standard Communication Satellite Mission Equipment (Cont.)

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- Status Monitoring

RF switches	-	58
Receivers	-	20
Transponders	-	258
278 analog	-	336 bi-level

- * Input rf generator test signal — 24 rf signals
- * Output rf detector — 24 power detectors

In order to provide conservative values the estimates are based upon the assumption that control and monitoring includes all TWTA's, where commands govern "on-off" operation of each TWT regulator and high voltage driver and the applied voltage to the TDA's.

2.2 Command, Ranging & Telemetry Requirements

2.2.1 Command

Command operations of the CDPI are initiated by reception of primary commands from the ground. Computer prestored commands, forming a secondary group may also be activated by ground command or as a function of time or other parameters. The commands must be verified and, if found correct, processed. The command operations are introduced to the spacecraft via a combined command/ranging receiver. Command verification is via the telemetry transmitter. The command section must meet the following requirements:

- Command rf frequency — 5.926 GHz
- Wide deviation FSK of 3 subcarriers modulating a main carrier to form coding. 50-100 bits/second rate.
- Ground Station Power Output - < 2 KW (as per power budget of Fig. 4).
- Ranging and command signals will not be intermixed.
- Command word length - 16 bits nominal

Type	2
Parity	1
Command or Data	10
Verify	2
Synch & Separation	1

- Number of commands — 1024
- Two separate command receivers and antennas required

---

* Optimal for on-board channel tests.

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2.2.2 Ranging

The ranging section employs the same receivers and transmitters as the command and telemetry sections. Range and range rate are determined by taking phase differences between the tone generated and fed via uplink transmission, and the repeated signals obtained from the telemetry downlink signals. Range determination is performed at the ground stations. The ranging section has the following requirements:

- Uplink on command carrier; downlink on telemetry carrier
- Phase modulated sequential tones
- RMS zero mean range error < 30 meters
- Range resolution to 3000 Km
- Acquisition time < 5 seconds

Uplink Ground Transmitter (2 Kw)	+ 63.0 dbm
Uplink Ground Antenna Gain (32 ft. @ 5.926 GHz)	+ 54.0 db
Ground Transmitter EIRP	+117.0 dbm
Atmospheric Loss	- 1.0 db
Space Attenuation Loss	-199.8 db
Polarization Loss	- 3.0 db
Spacecraft Antenna Gain (assumed worst case)	- 3.0 db
Received Signal Strength	- 89.8 dbm
Receiver Sensitivity	- 95.0 dbm
Margin	5.2 db

Fig. 4 Command Power Budget

2.2.3 Telemetry

2.2.3.1 Main Telemetry Section. The telemetry section is provided to enable monitoring spacecraft status and performance at the ground control stations. Two ground control stations are assumed. The section also furnishes the downlink channel for command/data verification and ranging. The requirements assumed are as follows:

• Transmit Frequencies	4.19 to 4.195 GHz
• Transmit power	< 1 watt nominal (as per power budget of Fig. 5)
• Modulation	1024 bits/sec bi-phase on 33 KHz subcarrier
• Data Format	PCM - NRZ, 128 word frame
• Data Types	All data digitized prior to transmission
• Data Frame types	4-6 different frames - selectable repetitiously or in sequence

2.2.3.2 Beacon. In order to enable the narrow beam ground antennas to locate the communication satellite, a small beacon is employed. Beacon requirements assumed are as follows:

• Beacon operating frequency	2.95 - 3.7 GHz range
• Transmit power	< 1 watt nominal (as per power budget of Fig. 5)

- Pulse or tone output with width and rate to be determined.

2.3 Stabilization and Control Requirements

The Stabilization and Control Subsystem (S&C) provides the on-board guidance and navigation functions in addition to supplying attitude and positional control. The S&C for the standard communication satellite is described in Reference 5. Requirements imposed on the CDPI are listed in Fig. 6. Commands and telemetry are identified in Figs. 7 and 8.

2.4 Attitude Control Propulsion Requirements

The Attitude Control Propulsion Subsystem (ACP) is closely aligned with the S&C in that the S&C control functions originate propulsion activation of the gas thrusters. Requirements of the ACP levied against the CDPI are similar to the design of Reference 1. The requirements affecting the CDPI are as follows:

- Four propulsion modules with six thrusters per module
- Selection of redundant couples to compensate for module malfunction
- Response to S&C or ground generated commands
- Telemetry output of driver pulses (from CDPI) - 24 bi-level - 24 analog.

The ACP layout is shown in Fig. 9 and the logic is specified in Fig. 10.

Spacecraft Transmitter (0.75 W assumed)	- 1.2 dbw
Modulation Loss	- 3.0 db
Coupler Loss	- 3.0 db
Spacecraft Antenna Gain (worst case assumed)	- 3.0 db
Space Attenuation (4 GHz)	- 196.8 db
Polarization Loss	- 3.0 db
Atmospheric Loss	- 0.5 db
Ground Antenna Gain (32 ft @ 4 GHz)	+ 50.7 db
Received Signal Strength	- 159.8 dbw
Receiver Noise (Parametric Amp. - 50^{o}K)	- 176.1 dbw
Carrier-to-Noise Ratio (C/N)	+ 16.3 db
Assumed C/N for error rate $< 10^{-5}$	+ 10.0 db
Margin	+ 6.3 db

Fig. 5 Telemetry & Beacon Power Budget

• Horizon Sensor Signal Calibration	10 bits/sec
• Sun Sensor Signal Calibration	10 bits/sec
• Attitude Error Determination (3)	$.03^{\circ}$ resolution
• Control Loop Compensation	
• Pointing	$.05^{\circ}$ uncertainty
• Stationkeeping	0.5° yaw, $.05^{\circ}$ pitch & roll
• Wheel Gimbal Drive Signals (2)	10 update/sec
• Wheel Speed Drive Modulators	1600 pps max. rate
• ACP Thruster Driver Modulators (3)	10/sec max
• Ephemeris Update (sun position)	4/day
• Nominal Sun Aspect Angles (2)	0.25°
• Wheel Unloading Logic	
• Station Determination (lat. & long.)	1/day - 0.02° accuracy
• Stationkeeping (ΔV scheduling)	
• Command & Status Monitoring	As per Fig. 7
• Fault Detection/Isolation	
• Mode Selection	7 Modes
• Earth pointing (normal operational)	
• Stationkeeping (normal operational)	
• Docking (Space Shuttle)	
• Reacquisition	
• Back-up Pointing	
• Anti-tumbling	
• Initialization	

Fig. 6 S&C Requirements Impacting CDPI

Plus Pitch bias Step
Minus Pitch bias step

Plus Roll bias step
Minus Roll bias step

Ground Station #1 Slant Range
Ground Station #1 Elevation, Azimuth Angles
Ground Station #2 Slant Range
Ground Station #2 Elevation, Azimuth Angles

Momentum Wheel Yaw Gimbal Bias Step - Plus
Momentum Wheel Yaw Gimbal Bias Step - Minus

Inhibit Unload - Roll
Inhibit Unload - Pitch

Pitch Thruster Pulse - Plus
Pitch Thruster Pulse - Minus

Roll Thruster Pulse - Plus
Roll Thruster Pulse - Minus

Yaw Thruster Pulse - Plus
Yaw Thruster Pulse- Minus

East Thruster Pulse
West Thruster Pulse
North Thruster Pulse
South Thruster Pulse
Up Thruster Pulse
Down Thruster Pulse

Switch to Inactive Horizon Sensor
Switch to Inactive Wheel Drive Electronics
Switch to Secondary Attitude Control
Switch to Primary Attitude Control
Switch Pitch Control to South Wheel

Remove Power - North Momentum Wheel Drive
Remove Power - South Momentum Wheel Drive

Fig. 7 Communication Standard Spacecraft Minimum Command List

North Wheel Speed

South Wheel Speed

North Wheel Temp

South Wheel Temp

Active Horizon Sensor Temp

Inactive Horizon Sensor Temp

Wheel Drive Electronics Temp (2)

Sun Aspect Sensor Temp (5)

Sun Aspect Sensor Electronics Temp (5)

Rate Gyro Package Temp

ACP Thruster Drive Electronics Temp (4)

Roll Gimbal Angle

Yaw Gimbal Angle

Roll Attitude Angle (Horizon Sensor)

Pitch Attitude Angle (Horizon Sensor)

Sun Aspect Sensor Output Angle (2)

Rate Gyro Outputs (3)

Fig. 8 Standard Communication Spacecraft Minimum Telemetry List

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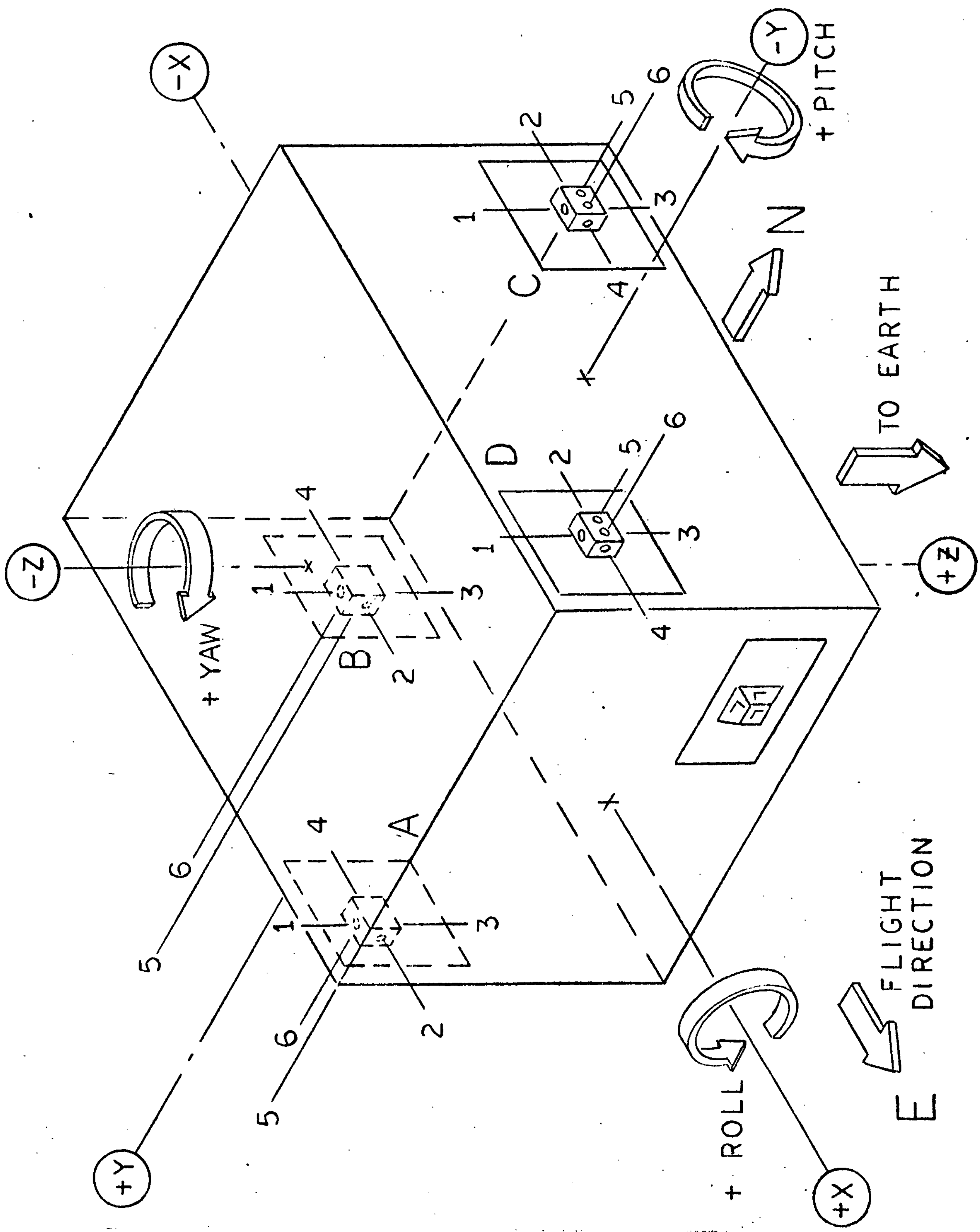

Fig. 9 Attitude Control Thruster Orientation - Standard Communication Satellite

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Vehicle Motion	Active Thrusters
+ Pitch	A3 & B1 or C1 & D3
- Pitch	A1 & B3 or C3 & D1
+ Roll	A1 & D3 or B1 & C3
- Roll	A3 & D1 or B3 & C1
+ Yaw	A2 & D2 or B2 & C2
- Yaw	A4 & D4 or B4 & C4
E Translation	A4 & D2 and B4 & C2
W Translation	A2 & D4 and B2 & C4
N Translation	A5 & B6 or A6 & B5
S Translation	C5 & D6 or C6 & D5

Fig. 10 Attitude Control Thruster Logic Standard Communication Satellite

2.5 Power Subsystem Requirements

Application of power and control of the various spacecraft subsystems are under CDPI/ground command management. This includes the preprogrammed or commanded switching discretes which activate or deactivate components during various operational or failure modes. Status monitoring and telemetry of data include the power supplies and various regulators. Allowance for 16 power supply outputs will be provided. This is in addition to the power level sensing and control at other spacecraft subsystems.

2.6 Instrumentation Requirements

In order to attain and relay downlink the data regarding equipment status and operating conditions of the various spacecraft and mission equipment components, a number of voltage and current sensors, plus pressure and temperature transducers, are distributed throughout the spacecraft. Because the number of sensors and transducers is large, sampling and multiplexing of the outputs from these sources will be necessary. To enable ease of interfacing and standardizing the CDPI sections, signal conditioning of the instrumentation outputs is necessary to adjust signals to desired voltage and impedance levels.

Estimates of instrumentation requirements are as follows:

Pressure Transducers	66
Temperature Transducers	146
Current & Voltage Sensors	390
Bilevel sensors	408

Sensors for mission equipment are provided by the mission equipment suppliers. Thus the requirements identified are for sizing instrumentation data handling and multiplexing only.

2.7 Data Processing Requirements

The subsystem requirements delineated in the previous paragraphs may be organized into operational flow processes, as shown in the Functional Requirements Diagram of Fig. 11. This diagram shows the functional relationships between subsystem elements; it is not meant to portray the actual implementation. Seven main paths are shown in the figure.

- Horizon Sensor — Error signals derived from the sensors are digitized and processed to provide pulse width modulated torquing signals to the momentum wheels.

- Wheel counters & Gimbal Angle Pickoffs — Outputs from the tachometers and gimbal pickoffs on the momentum wheels are digitized and processed to provide wheel speed unloading via the gas jet thrusters. Pulse width modulation is used to drive the wheels and jet thrusters.

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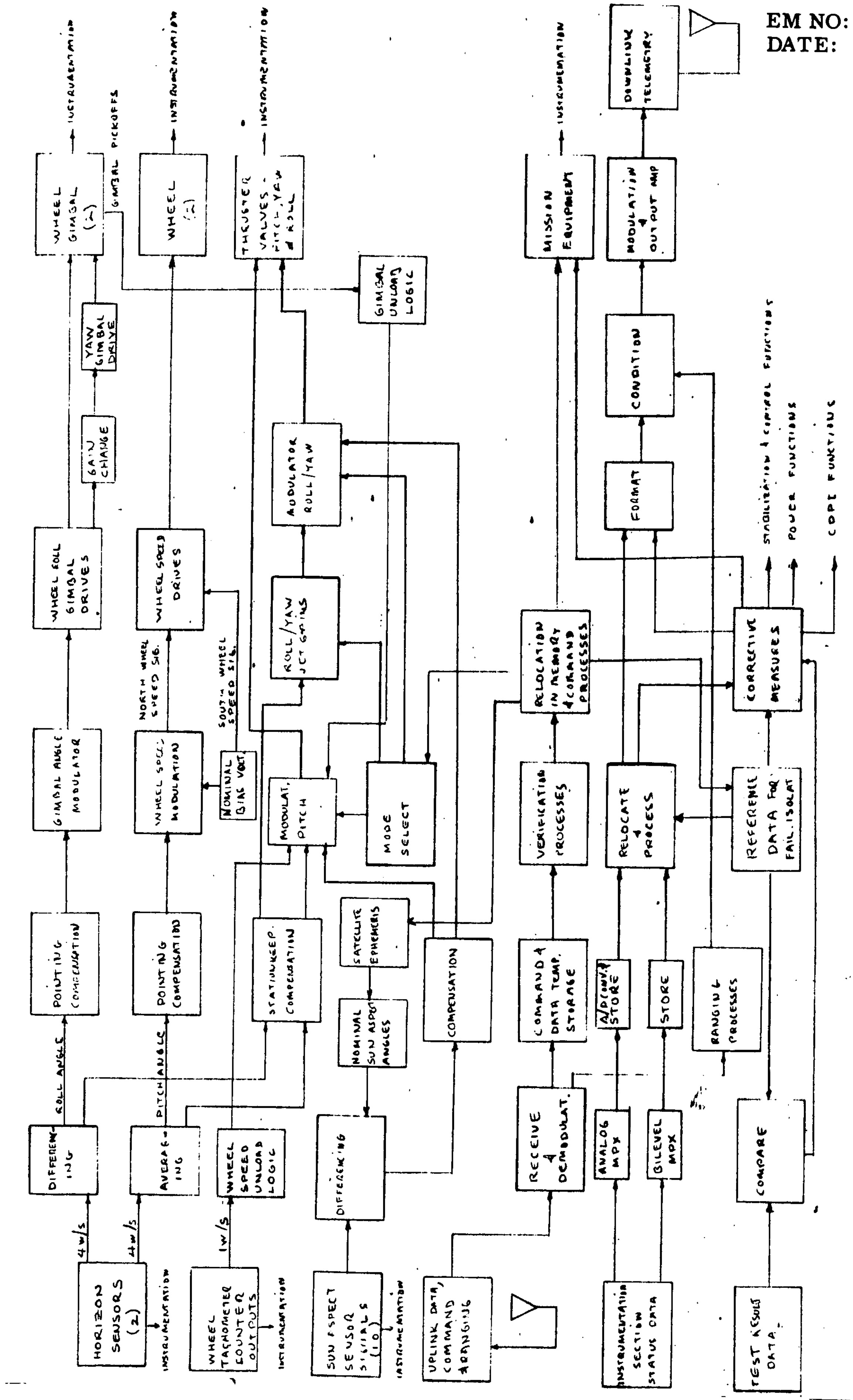

Fig. 11 Functional Requirements Diagram - CDPI Subsystem
Standard Communication Satellite

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- Sun Aspect Sensors — The signals from the sun aspect sensors are digitized and compared with nominal aspect angles. Differences between the two sets of values are processed and the results used as a drive to the pulse modulation logic for the jet thruster valves.

- Uplink Data or Command and Ranging — Uplink command and data bits are received, grouped into words and stored temporarily. A verification process consisting of checking parity bits, establishing coding acceptability, etc., is carried out using multiple transmissions, comparison of word pairs, or similar techniques. If the words are found acceptable they are relocated to the appropriate locations for subsequent telemetry and execution.

  Ranging signals are identified and processed. Ranging tones, as demodulated, are used to modulate the telemetry transmitter modulator. Tone modulation of the beacon transmitter may also be used.

- Downlink Telemetry — Instrumentation, process results, commands and ranging tones are sent downlink via the telemetry transmitters. Processing of these data include formatting and conditioning prior to transmission.

- Test Results Data — Checks are made on processing and other results by comparing with a reference source, e.g., data entered via uplink communication or prestored. Corrective processes, e.g., switching redundant elements, are telemetered to the ground control station.

Implementation of the data handling, storage and computations included in the above processes may be accomplished by hardwired components or through software in a digital computer. Because of the flexibility afforded and relative amenability to change which is obtained by the latter approach, without sacrificing the capability for hardware standardization, utilization of a digital computer in the CDPI is assumed. The computer requirements are as follows:

Word Length - Station and attitude determination, 0.02° and 0.03° error, respectively, indicates that one part in 10^{4} or 14 bits would suffice. Allowing for sign and possible roundoff error a word length of 16 bits appears adequate. The ranging requirement of 30 meters accuracy over a 3000 Km region of ambiguity may be satisfied by double precision. Thus, a 16 bit word length with double precision capability is specified as a requirement.

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Memory Capacity - Memory requirements for both instructions and data are estimated as follows:

Functions	16 bit Word Totals
Stability & Control	400
Commands	1024
Command Execution Routines	300
Telemetry	1082
Verification	350
Fault Isolation Routines	1250
Attitude Control	150
I/O Function Routines	700
Shuttle & Ground Test	350
	5606 words

In order to allow for growth, a minimum memory size of 8,192 words is desirable. This does not include the additional memory capacity needed to meet reliability requirements. The information supplied in Reference 6 indicates a redundant memory is sufficient to meet the reliability goals stipulated for the CDPI computer (see Paragraph 2.8). Thus, a 16,384 16 bit word memory is specified as a requirement.

Memory Type - A random access memory is required; however, the utilization of read-only sections may be desired to protect critical subroutines and stored constants. If read-only memory sections are employed, they should not exceed 10 percent of total memory capacity (1600 words). Redundant read-only memory is probably not necessary due to the high MTBF of such units.

Operating Speed - Computer memory cycle, input/output and processing speeds have not been definitized at this time; however, requirements should be minimal. An add time of 10 microseconds and a multiply time of less than 50 microseconds should be adequate.

Instruction Repertoire - There are no special instructions required for the processes identified thus far. A standard mix of arithmetic, input/output, memory addressing and logical instructions should be adequate. Double precision instructions like add, subtract, read and store would be desirable.

Index Registers - Indexing will be required for telemetry and other routines, thus a minimum of at least one index register is specified as a requirement. Multiple index registers are desired.

Arithmetic Type - The standard fixed point, 2's complement, parallel arithmetic computer is adequate. A double length accumulator is desired.

Input/Output - Input and output should be through computer control. Direct memory addressing (DMA) may be necessary to mechanize redundant memory operations; however, direct computer control of input/output is desired to minimize potentiality for modifying stored instructions or data through externally introduced errors. If DMA is used, protective software and input/output hardware must be provided.

A multiple external interrupt capability is required for power off and power on, notification of data readiness at the input registers and command execute tone reception.

A separate input and output register is the minimum register complement; however, availability of two input and two output registers are desired.

A capability for discrete issuance to provide the pulse width modulation to the momentum wheels and jet thrusters is required. 25 variable discretes are specified. In addition, availability of 10 fixed discretes for activation signals to the logic controlled redundant equipment is desired.

Weight, Size, Power - Specific values for weight, size and power have not been established; however, fourth generation technology is required to satisfy reliability requirements. A fourth generation computer, capable of meeting the requirements specified herein would be as follows:

Weight	< 18 lbs
Size	< 0.35 ft^3
Power	< 70W

2.8 Reliability Requirements

A reliability analysis of the communication satellite standard spacecraft has been performed (Reference 7) and the reliability goals apportioned to the CDPI is as shown in Fig. 12. The reliability figures as given are for a five-year mission lifetime. The mission equipment reliability figures are based upon the apportionment model shown in Fig. 13 and the remaining reliability figures are detailed in the apportionment models of Figs. 14 and 15. Utilization of the redundancy techniques listed in Fig. 16 is incorporated in the designs so that reliability goals will be met.

3.0 CDPI Design

3.1 Communication Section

A block diagram of the CDPI communication section point design for the standardized communication satellite is shown in Fig. 17. The design is relatively straightforward. A common antenna feeds redundant superheterodyne receivers via a filter and hybrid coupler. Input filtering is employed to minimize receiver spurious responses as well as aid in isolation of the mixer input reference frequency. The TDA amplifier reduces the receiver noise figure and contributes to front end skirt selectivity and isolation.

The IF amplifier is nominally two stages, operating at an IF center frequency of approximately 45 MHz. The first stage is 10 MHz wide and the second about 1.0 MHz wide to obtain the necessary receiver selectivity. AGC is incorporated to provide a relatively constant input signal to the demodulator limiter.

The demodulator is in two parts - an amplitude detector for the AGC and a standard Foster-Seeleytype frequency discriminator to detect the command and range signals. The amplitude detector output for AGC is also used for signal presence information

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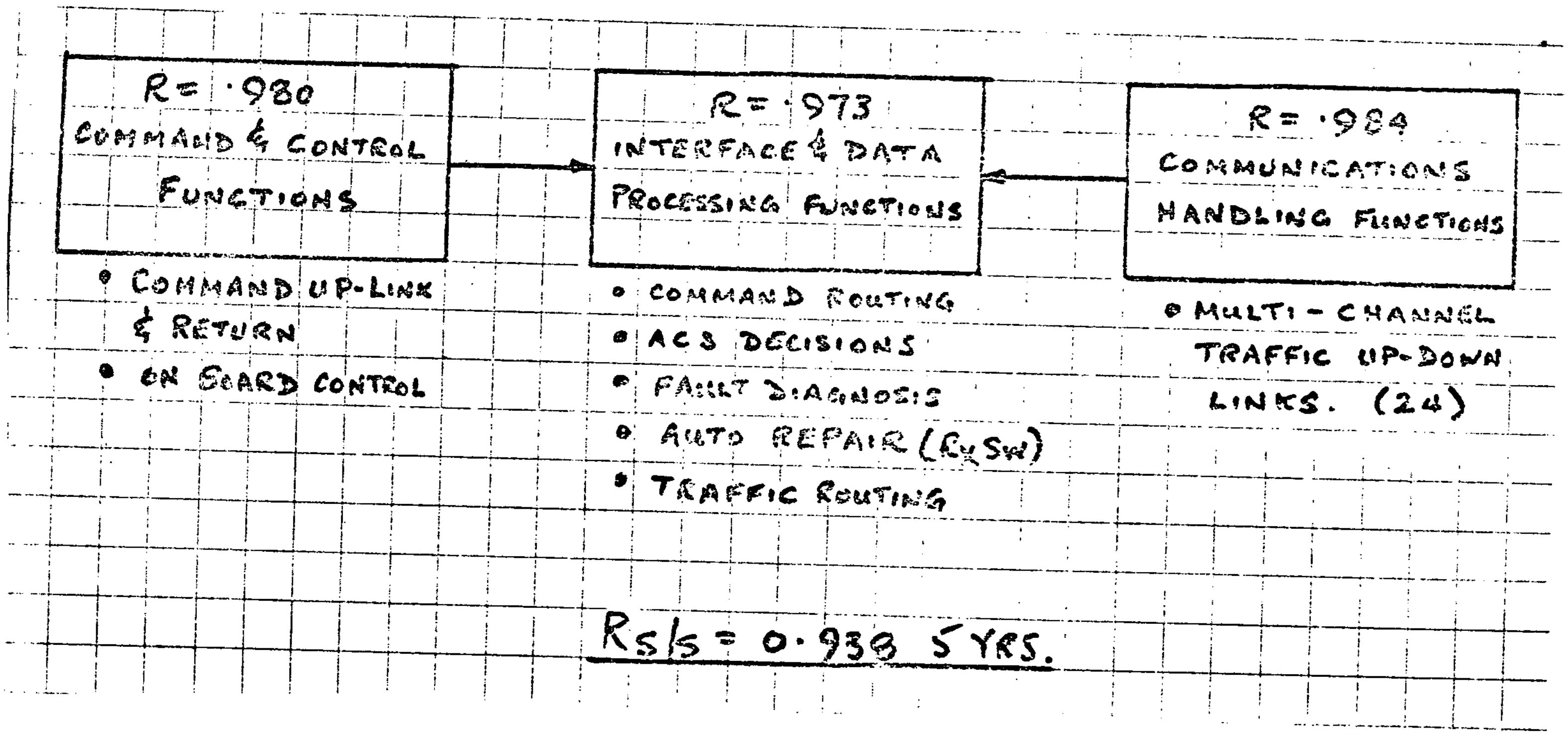

Fig. 12 CDPI Subsystem Reliability Block Diagram (Serial)

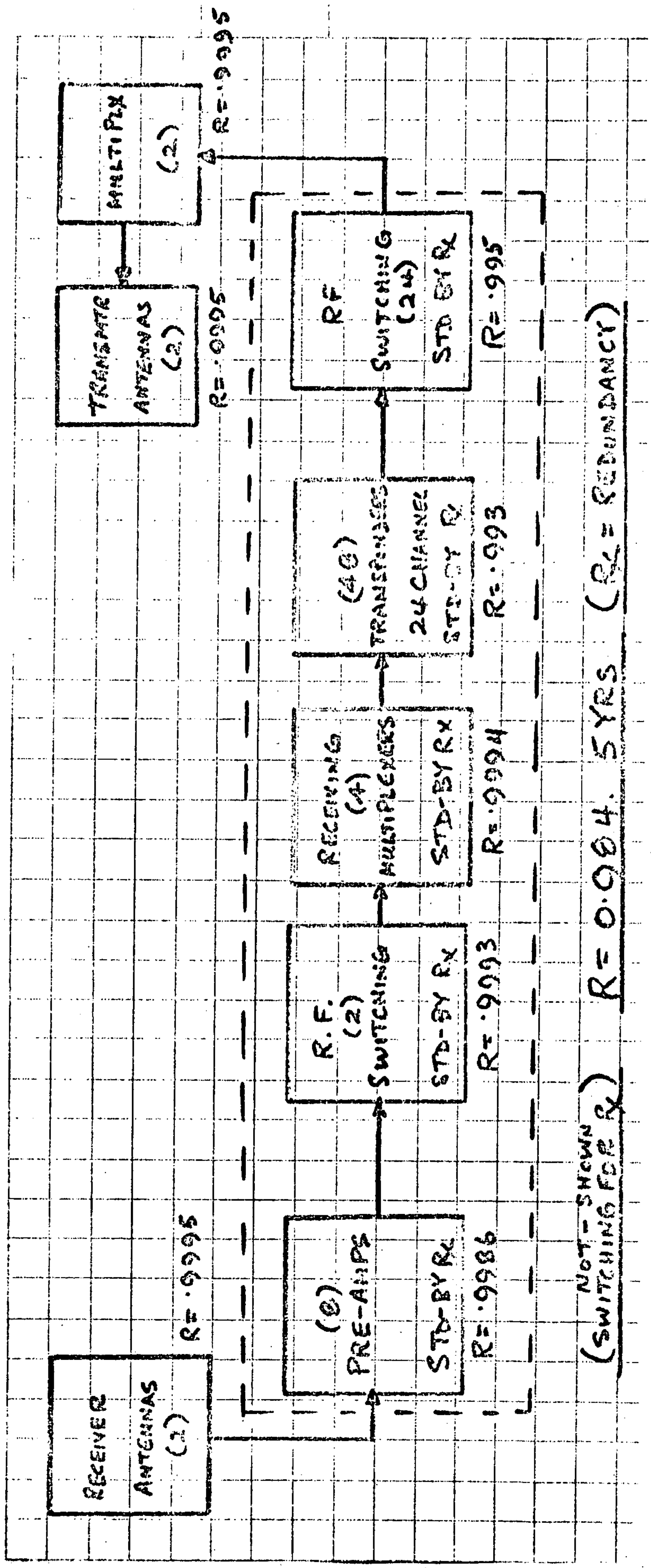

Fig. 13 Communications Handling Loop Reliability Block Diagram (Serial)

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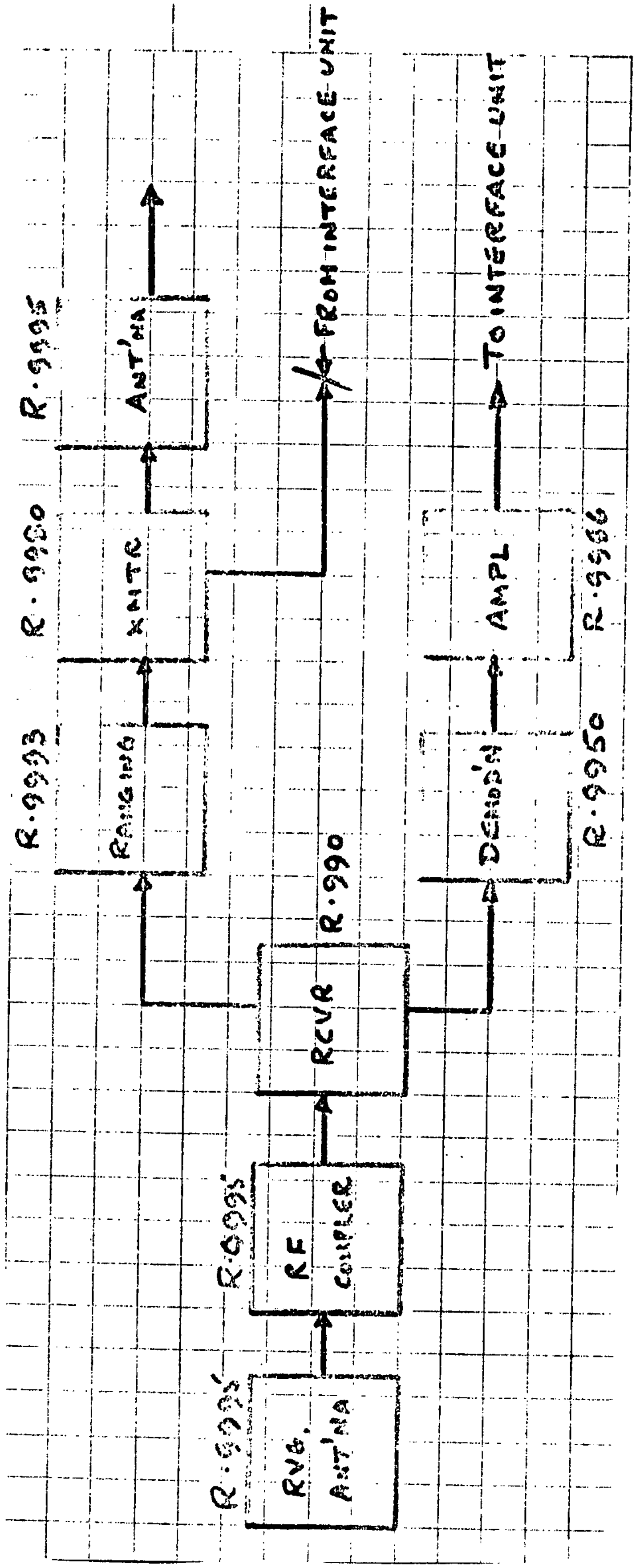

R = .980 5 Yrs.

Fig. 14 CDPI Subsystem Command & Control Loop Reliability Block Diagram (Serial)

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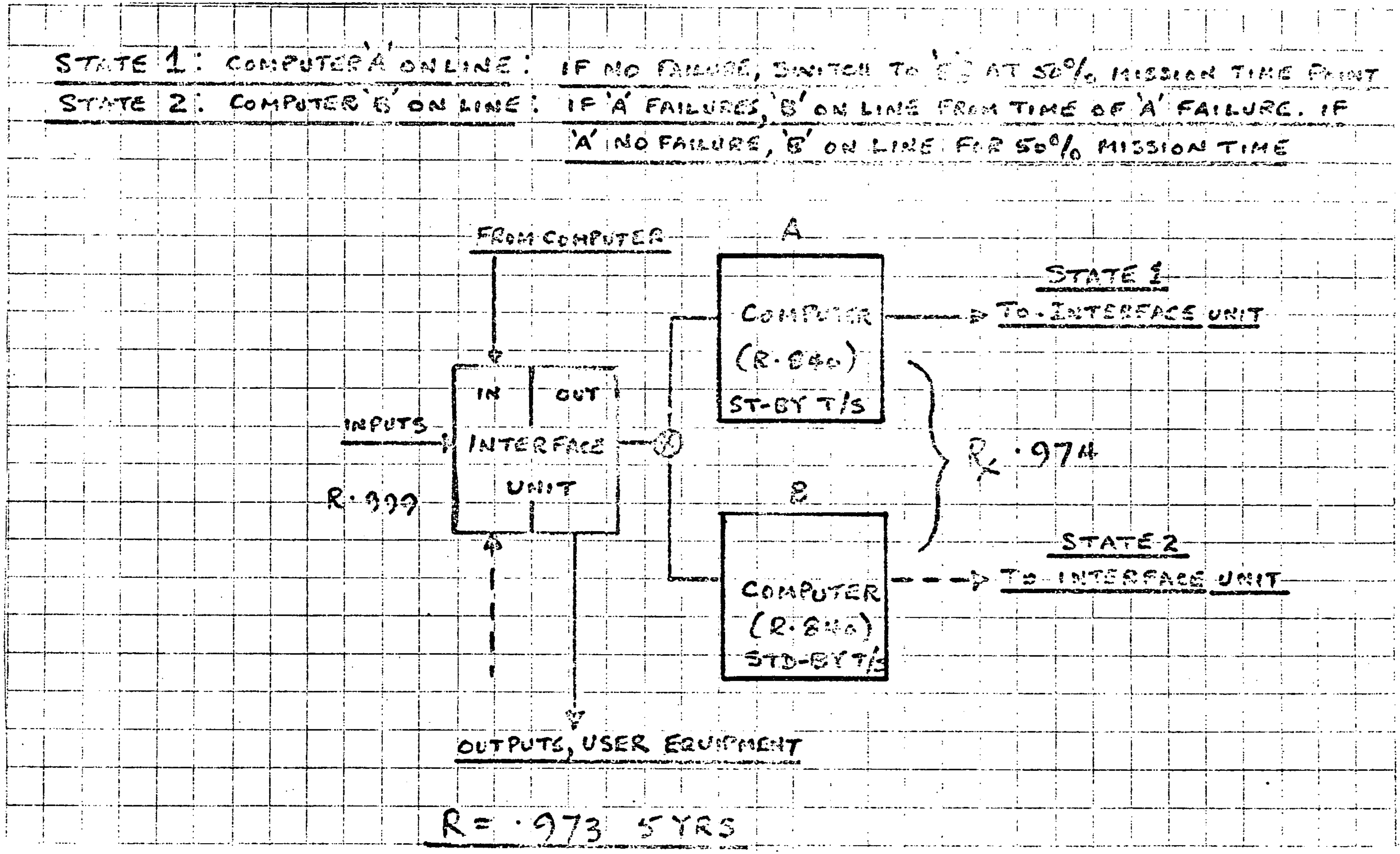

Fig. 15 CDPI Subsystem Interface & Data Processing Loop
Reliability Block Diagram

COMM. TRAFFIC HANDLING LOOP

- All transponders in standby redundancy
- RF switching for active modes only
- Preamplifiers in standby for active modes

COMMAND & CONTROL LOOP

- Receiver full active dual channels
- Transmitter: Active redundancy dual channel
- Ranging: Auto switched standby

INTERFACE & DATA PROCESSING LOOP

- Computers in standby T.S. redundancy
- Interface unit handles all standby redundancy switching command for fault diagnosis & auto repair

GENERAL

- Block diagrams show reliability numerics

Fig. 16 CDPI Subsystem Preliminary Reliability Analysis

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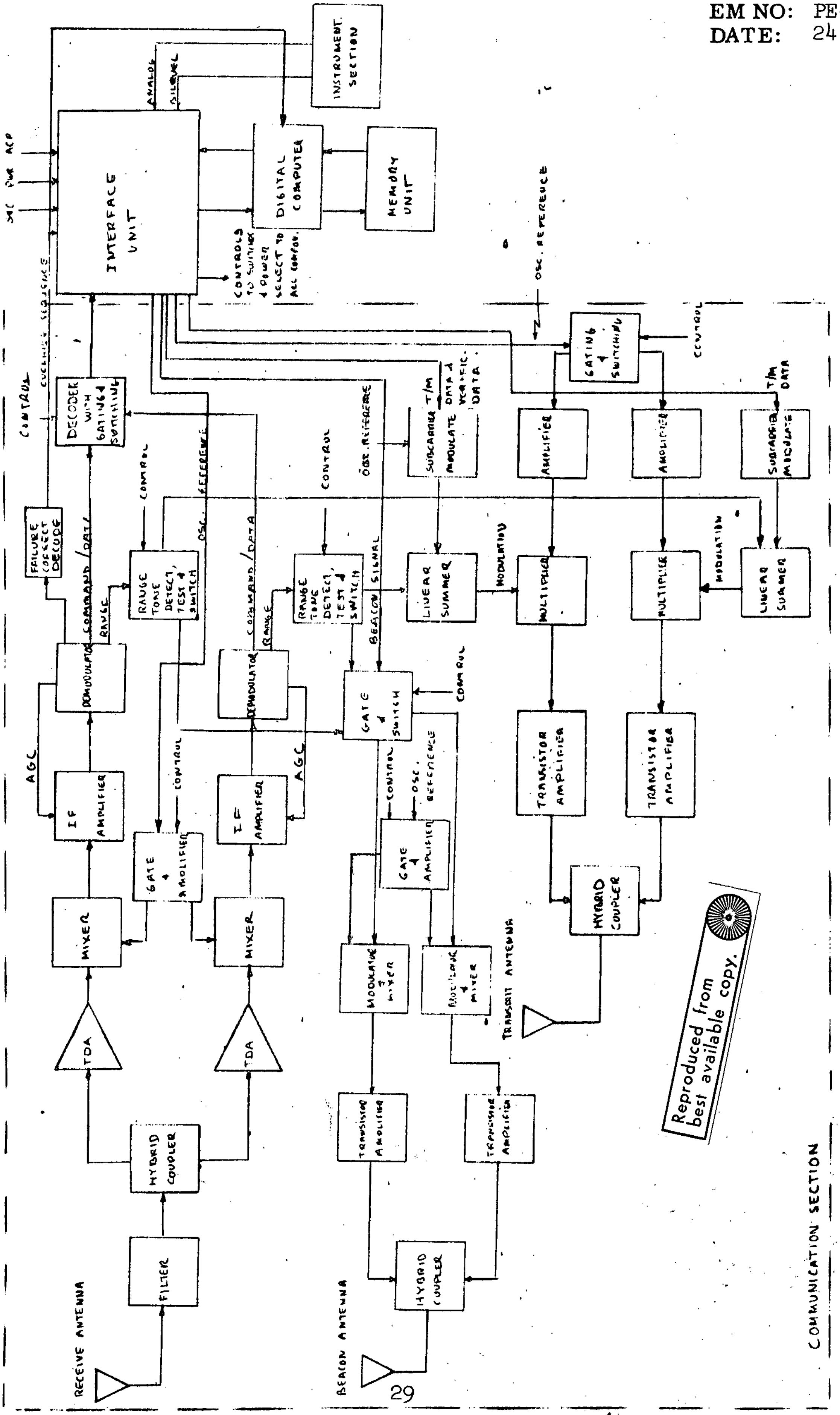

Fig. 17 Communication Section - CDPI Subsystem Standard Communication Satellite

(carrier detection), a signal which may be employed for backup command switching.

Command/data and range information are separated, with the command tones processed in a decoder to form 1-0 bit patterns prior to routing to the computer via the Interface Unit of the CDPI. Range tones are used to modulate a carrier for retransmission to the ground station via a linear summing circuit. The same circuit is used to accept the subcarrier modulated signal which contains the telemetry/command verification information.

The transmitter consists of redundant varactor multipliers and transistor amplifiers which feed a common antenna via a hybrid coupler. The oscillator reference for the transmitter comes from the timing generator circuitry in the Interface Unit. The same circuits are used to provide the receiver mixer oscillator reference frequency and the telemetry subcarrier frequency.

In order to provide greater downlink redundancy, a separate beacon transmitter is provided, feeding a horn antenna. The beacon signal may consist of a particular tone pattern, a ranging signal or, if desired, telemetry or command retransmissions.

Due to the availability of computer control, a 100 percent uplink and downlink duty cycle is not essential and the second receiver and transmitter may be placed in time-shared standby mode. This means that power to the redundant transmitter and receiver components may be switched off until a failure is sensed via the instrumentation or fault testing and isolation hardware and software. If a failure is recognized by interpretation of telemetry data or recognition of improper responses to the uplink commands, a failure override sequence may be activated which controls either power distribution to components or enables the redundant computer memory switchover.

The configuration shown in the figure utilizes one receive and one transmit antenna for the command/telemetry functions; however, a possible, and perhaps more desirable configuration would have redundant combined receive/transmit antennas. The approach depends upon polarization and other requirements and is a subject of future trade study. In this design linear polarization in the direction of earth is assumed for the uplink antenna and circular polarization for downlink telemetry and range.

The major specifications governing each CDPI Communication Section component are given in Figs. 18 and 19, while estimates of size, weight, power and cost are provided in Fig. 20. Instrumentation requirements are nominal and are not separately identified in the data. The Interface Unit design includes commutating multiplexing, etc., the more significant instrumentation cost items.

3.2 Interface Section

A block diagram of the Interface Section for the CDPI point design is shown in Fig. 21. The function of the Interface Unit is to provide data routing and operational control paths between the various spacecraft subsystems and the data processing and communication sections of the CDPI. Data routing consists of the following:

- Accepting, sampling, subcommutating, multiplexing and converting analog instrumentation data for computer input.

Operating Frequency	5.926 GHz
Modulation	Phase angle
Input signal	-89.8 dbm
Input filter insertion loss	2 db
Noise figure	8 db
IF Bandwidth	1 MHz
IF Frequency	45 MHz
Output command	1v. RMS -10 K ohm
Output ranging	1v. RMS 500 ohm
Antenna Beamwidth	360°
Antenna Gain	~ 0 db
Antenna Polarization	Linear

Fig. 18 Command/Data & Ranging Receiver Specifications

Operating Frequency	4.19 + 4.195 GHz
Modulation	Phase
Modulation Linearity	> 1%
Power	> 0.5 watts
Subcarrier Modulation	Bi-phase
Output Impedance	50 ohms
Subcarrier Frequency	33 KHz
Osc. Reference Stability	10^{-5} per day
PCM Modulation	NRZ-PCM @ 1024 bits/sec

Fig. 19 Telemetry/Ranging Transmitter Specifications

Item	No.	Wt. (lbs) Unit	Wt. (lbs) Total	Size Unit in	Size Total in^3	Power (Watts) Unit	Power (Watts) Total	ROM Cost x 10^3 Non-Recurring	ROM Cost x 10^3 Recurring
Waveguide		1.5	1.5						
Rec. Antenna (slot)	1	-	3.0	7x12x½	-	-	-	10	1
Xmt. Antenna (cross slot)	1	-	4.0	5D x 3-3/4	-	-	-	10	2
Bcn Antenna (horn)	1	1	1.0	6"standard	-	-	-	-	-
Receiver	2	3	6.0	3x3x2	36.0	5	10.0	50	10
Decoder Command	1	4.5	4.5	3x3x6	54.0	5.2	5.2	5	2
Decoder Fail	1	0.2	0.2	3x2x2	12.0	1.0	1.0	2	1.2
Range Det, Tst & Sw	2	0.1	0.2	1x1x2	4.0	1.0	2.0	1	.7
Mod. & Summer	2	0.2	0.4	3x4x4½	108.0	1.0	2.0		
Tel Transmitter	2	2.0	4.0	2½x4x7	140.0	10.0	20.0	20	7
Beacon Transmitter	2	2.0	4.0	2½x4x7	140.0	10.0	20.0	10	5
Hybrid Couplers	3	1.0	3.0	2x3½x9	189.0	-	-	8	3
Misc.Amp.,Gates & Sw	5	0.2	1.0	1x1x2	10.0	1.0	5.0	3	1
Input Filter	1	0.2	0.2	2½x1x.6	2.6	-	-	4	1
Totals			33.0		695.6		65.2	$122,000	$32,900

Fig. 20 CDPI Communication Section Equipment List

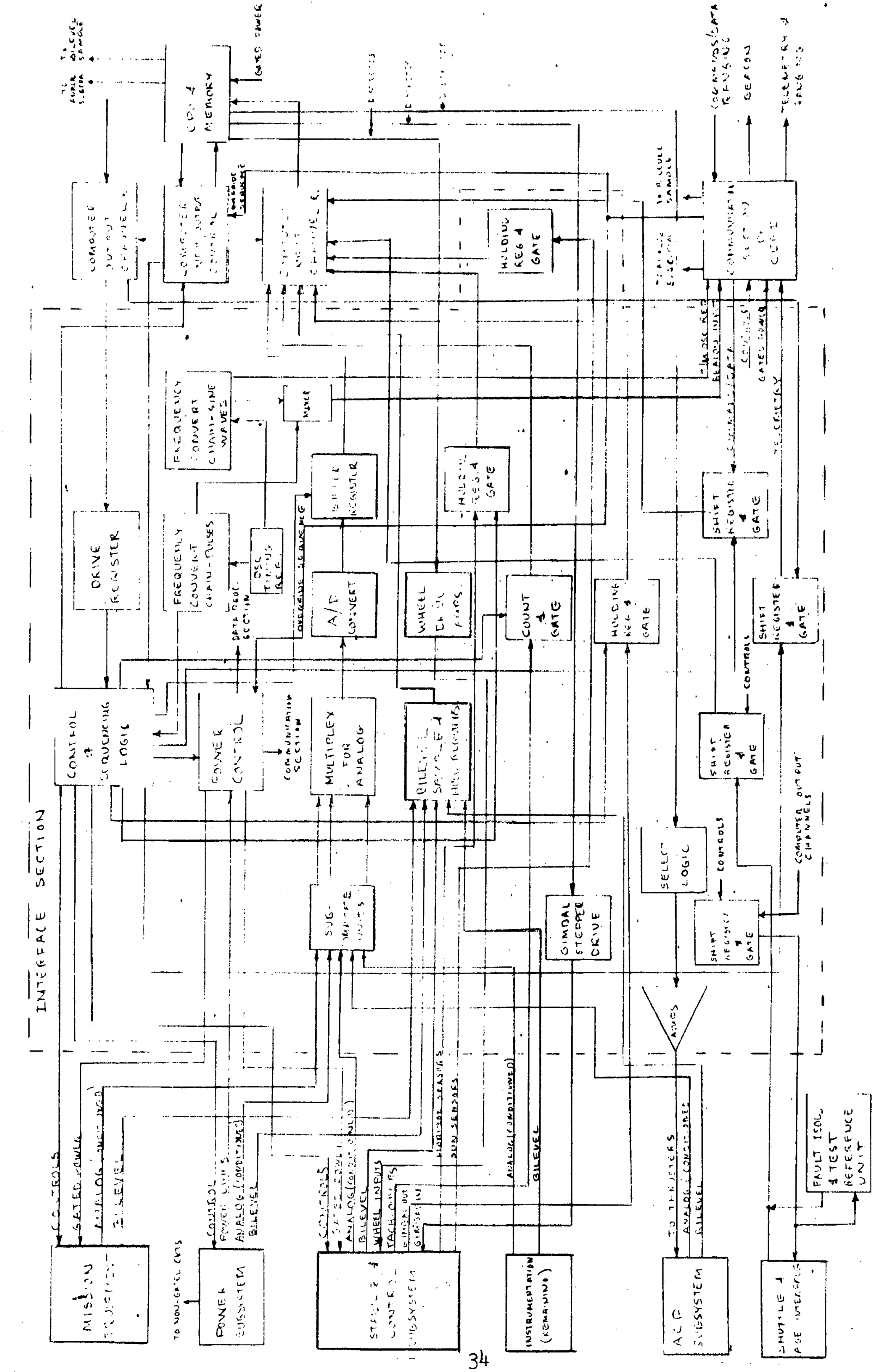

Fig. 21 Interface Section - CDPI Subsystem
Standard Communication Satellite

- Accepting, sampling, formatting and delay of bilevel data for computer input

- Accepting and delivery of S&C horizon and sun sensor output values to the computer

- Relay of information to telemetry, AGE or Shuttle-based equipment

The control processes are as follows:

- Accepts computer stored or verified uplink commands and issues the control discretes via discrete control sequencing logic. This logic operates mission equipment rf and other switches, governs power application via power control logic, operates gating circuits which trigger holding and shift registers in the Interface Unit, etc.

- Provides conditioned drive signals to the S&C momentum wheel torquers and gimbals and ACS thrusters.

The Interface Unit also provides the timing references for synchronizing and operating logic as well as Communication Section functions. This was done to take advantage of system commonalities and improve overall flexibility. The highest frequencies are nominally between 50-100 MHz, since frequency multipliers are used in the communication equipment. Most other frequencies utilized are less than 100 KHz.

Functionally the equipment in the Interface Unit is standard and similar to the Low Data Rate Unit in the Interface Section of Reference 1. The primary deviations from the Reference 1 configuration are the increased number of registers and the addition of fault isolation and test equipment. These changes are made to allow for more positive control of computer input and output, thereby reducing or eliminating computer direct memory accessing (DMA), and to introduce a more positive future correction scheme. Both changes are introduced to aid in realizing the specified reliability goals.

The actual design of the Fault Isolation and Test Reference Unit has not been established as yet and will require further study; however, a mechanization scheme may be suggested at this time. The basic approach is to have a fixed wired memory of n words, with one word for each test performed. A test word is serially sent to the test unit. The word contains address, data and parity check portions. The parity is tested and if found correct, the word in the fixed wire memory corresponding to the specified address is extracted and compared to the input word. The comparison process is repeated three times and majority logic evaluated. If the test indicates a failure, the test unit relays a notification to the computer (and to telemetry) for corrective action. The corrective action may be autonomously operated via the computer or through ground override command. In this design the test unit will be assumed to have a capacity of 128 words. This results in a seven bit address, eight bit data section and one bit for parity, a total of 16 bits.

Descriptions of the major components in the Interface Unit are provided in Fig. 22 while estimates of weight, size, power and cost are given in Fig. 23. The figures include projected descriptions and estimates for the Fault Isolation and Test Reference Unit.

Control & Sequencing Logic	662 Gates - MSI - TTL logic
Power Control	128 Gates
Subcommutations (20)	32 Channels
Multiplexer	20 Channels
A/D Converter	8 bits
Bi-level Sample & Hold Logic	410 Gates - MSI - TTL logic
Registers	3 16 bits length 4 4 bits length 1 8 bits length
Frequency Reference	2 Xtal Osc. + multiply and divide chains. 20 gates in pulse unit - 6 stages in sine wave unit
Fault Isolation & Test Unit	128 8 bit words fixed wired memory - address decode, parity decode - comparator/registers (2)

Fig. 22 Interface Unit Component Descriptions

Item	Wt. (lbs)	Size ft^3	Power (watts)	Cost in Dollars Non-Recurring	Recurring
Frequency Reference Unit	1.0	$.02^3$	2.0	15,000	2,000
(6) Logic Boards (Gates)	6.5	0.35	12.0	20,000	12,000
(3) Logic Boards Registers MPX, Converter, Bilevel, etc.	3.0	0.18	6.0	10,400	6,000
Subtotal	10.5	0.55	20.0	$ 45,400	$ 20,000
Fault Isolation & Test Reference Unit	10	.25	10	100,000	30,000
Total	20.5	0.8	30	$145,400	$ 50,000

*High reliability, 100 percent screened parts, logic boards, etc.

Fig. 23 Interface Unit - Estimates of Size, Weight, Power & Cost*

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3.3 Data Processing Section

If the data processing requirements identified in Paragraph 2.7 are reviewed and compared with those given in Reference 1, it is seen that there are some important differences between the two given sets of data. These differences include increased reliability, decreased speed and changed input/output demands in the case of the communication satellite data processor. It is noted, however, that most other required computer characteristics are very similar, e.g., word length, instruction repertoire, memory type, arithmetic type, etc. In the interests of standardization this would suggest utilization of a common computer type for both systems. It is, however, necessary to account for the differences that do exist in some manner. Each of the changes which may be made are discussed in turn.

3.3.1 Improving Computer Reliability

A number of techniques are available for attaining the needed higher reliability. These include the following:

- Utilization of High Reliability Parts, with more extensive part evaluation and screening
- Utilization of high level redundancy, e.g., use of multiple computers
- Utilization of lower level redundancy, particularly a multiprocessor configuration
- More extensive tests of software processes.

Part screening and other high reliability processes are extremely effective. For example, the Rolm 1601, a computer used as a representative candidate in an early Payload Effects Study (Reference 8), has not had any failures of any of the computers in the field. This result is equivalent to an MTBF of 35,000 hours - a number which continues to grow as experience is obtained. This is far in excess of the predicted MTBF of 11,000 hours. CDC is predicting a minimum of 75,000 hours MTBF for its 469 8K memory computer, assuming utilization of a high reliability program in the design approach, however, a high figure will probably be attained in practice.

The second approach, use of high level redundancy, with the redundant computers switched on by ground command or autonomously by an on-board test process, is utilized in the Reference 1 point design. Except for possible difficulties with initialization procedures, the mechanization of switchover is relatively easy if a power-on interrupt is designed into the computer. Thus, application of power drives the computer to a particular memory address. This address is the start location for the power-on interrupt subroutine, a procedure which initializes registers, input output locations, etc., prior to exiting to the first operational program step.

The third approach, lower level redundancy, is much more effective from a reliability standpoint; however, mechanization is more difficult. One technique is to use a single central processing unit which may address two or more redundant memory modules, as shown in Fig. 24. The Memory Address Register (MAR) has extra bits added on to allow for full addressing of the entire memory configuration. Thus, if the

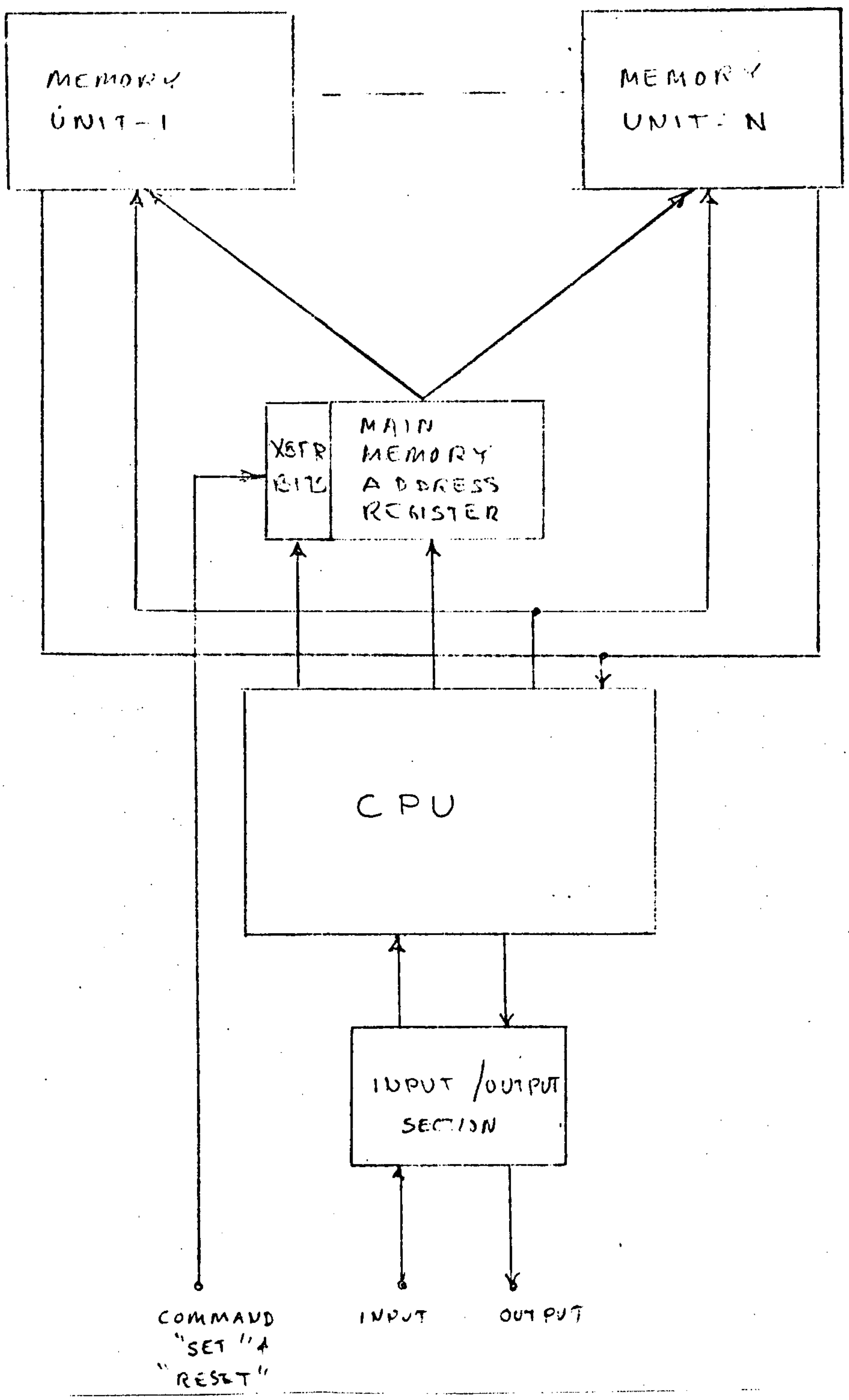

Fig. 24 Multiprocessing - Multiple Memory Unit Addressing for Redundant Operation

memory is made of two 8K modules, the MAR would have 14 bits - 13 bits for addressing one module and the extra bit to substitute the redundant memory unit should the fault identification and isolation processes indicate failure. Setting of the extra bit may be via ground command or through computer processing. It should be noted that if page addressing is employed, a normal technique to conserve word length, it is not necessary to expand the MAR to account for all addresses in memory since the addressing method is not basically changed.

A second low level multiprocessing technique is to employ multiple CPUs and memory units, as shown in Fig. 25. In this configuration, a data bus is employed for input and output to the various units. An address decoder is associated with each module, and the specific module to be addressed is given an assignation number. It is therefore possible to have any unit in the configuration address any other unit. Thus, if a failure is recognized, the assignment numbers are modified by ground command or via the external test unit, to prevent addressing failed components.

It is important to recognize that the operational techniques identified do not include the graceful degradation processes which may be employed with multiprocessors. Module memory contents are generally replicated in each module; no attempt is made to obtain high computation rates via multiprocessing modes.

3.3.2 Decreased Speed Requirements

The relaxation in speed requirements of the communication satellite CDPI data processor may be utilized in three ways - obtaining a slower computer, changing the clock frequency of a specified computer or using the increased available time for performing more detailed testing of computation and decision making processes. Obtaining a slower computer or changing clock rate is attractive from a reliability viewpoint. Decreasing required rate lowers the susceptibility of the circuits to transients and other disturbances, since increased filtering in gating and register circuitry may be used. Memory drive requirements are also reduced and a broader range of component parts may be employed. The main disadvantage is that the computer is more program peculiar, therefore less flexible and amenable to other applications.

Using the increased available computation time to perform more extensive testing does increase the software effort; however, the computer employed is applicable to a wide number of different missions. For this reason, a higher speed machine is used in this point design.

3.3.3 Changed Input/Output Requirements

An important tradeoff which must be made in resolving a CDPI design for a satellite is establishing the optimum functional allocations to the computer input/output unit and the external CDPI Interface Section. Increasing computer input/output circuitry and capability vs expansion of the Interface Section has a number of relative advantages and disadvantages. These include the following:

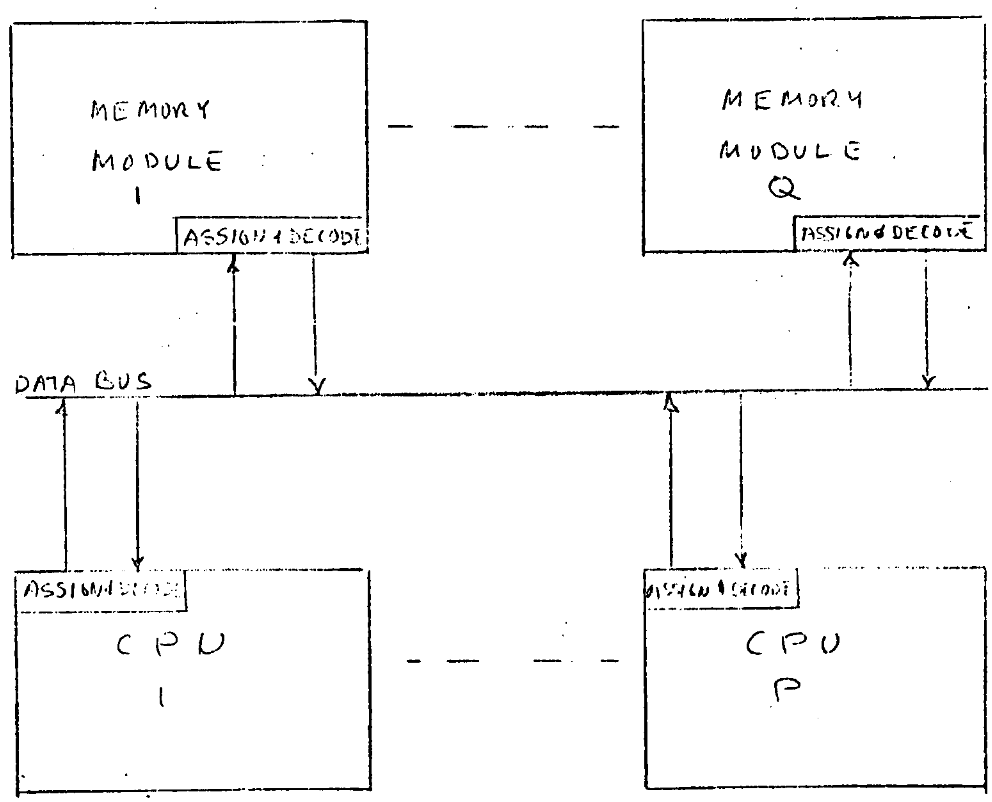

Fig. 25 Multiprocessing - Multiple Memory & CPU Configuration for Redundancy

Advantages	Disadvantages
• Simpler Interface Section	• Tendency toward customization of the computers
• More optimum match to computer characteristics, e.g., synchronization	• Less amenable to modification or change
• Opportunity for improved packaging and reduction in overall size and weight	• Not desirable for high power level functions
• Low power level functions effectively handled	

Considering these results, it is seen that expansion of the inherent computer input/output capability is desirable when tight computer control is required, input/output requirements are fixed, and low power levels are employed. However, a more general solution is obtained by broadening Interface Section capabilities; it affords a greater opportunity for standardization of the computer configuration, a prime objective of this work. Thus, a nominal computer input/output unit is employed in this point design. The input/output of Fig. 26, the input/output section of a CDC 469, typifies such a unit. Details of the data flow and control are shown in Figs. 27 and 28.

3.3.4 Representative Computer Design

By making the recommended adjustments, it is possible to utilize the same type computer for the communication satellite CDPI as was used for the Earth Observatory Satellite (EOS) of Reference 1. An example of a computer which satisfies the stipulated requirements, one that is within present technology capabilities, is the CDC 469. Since this computer is described in Reference 1, it is not necessary to elaborate on the computer here, except to reiterate the overall description, as shown in Fig. 29, for ease of reference. There are, however, the changes needed to reflect the increased reliability requirements imposed on this point design. These changes include the described approaches:

- Utilization of high reliability parts
- Screening
- Use of redundant memories within each computer

In addition, more detailed software verification and validation procedures will be required, compared to the Reference 1 software effort.

Adaptation of these approaches will impact cost and estimated MTBF values. The table in Fig. 30 lists some of the possibilities. Review of reliability requirements indicate that redundant Type 2 configurations will satisfy desired reliability goals. Figure 31 outlines costs which would accrue with the Type 2 configuration. Costs for support equipment and software are included. It is to be noted that these costs do not reflect support software costs, documentation, management, etc., costs which will be added in a practical program.

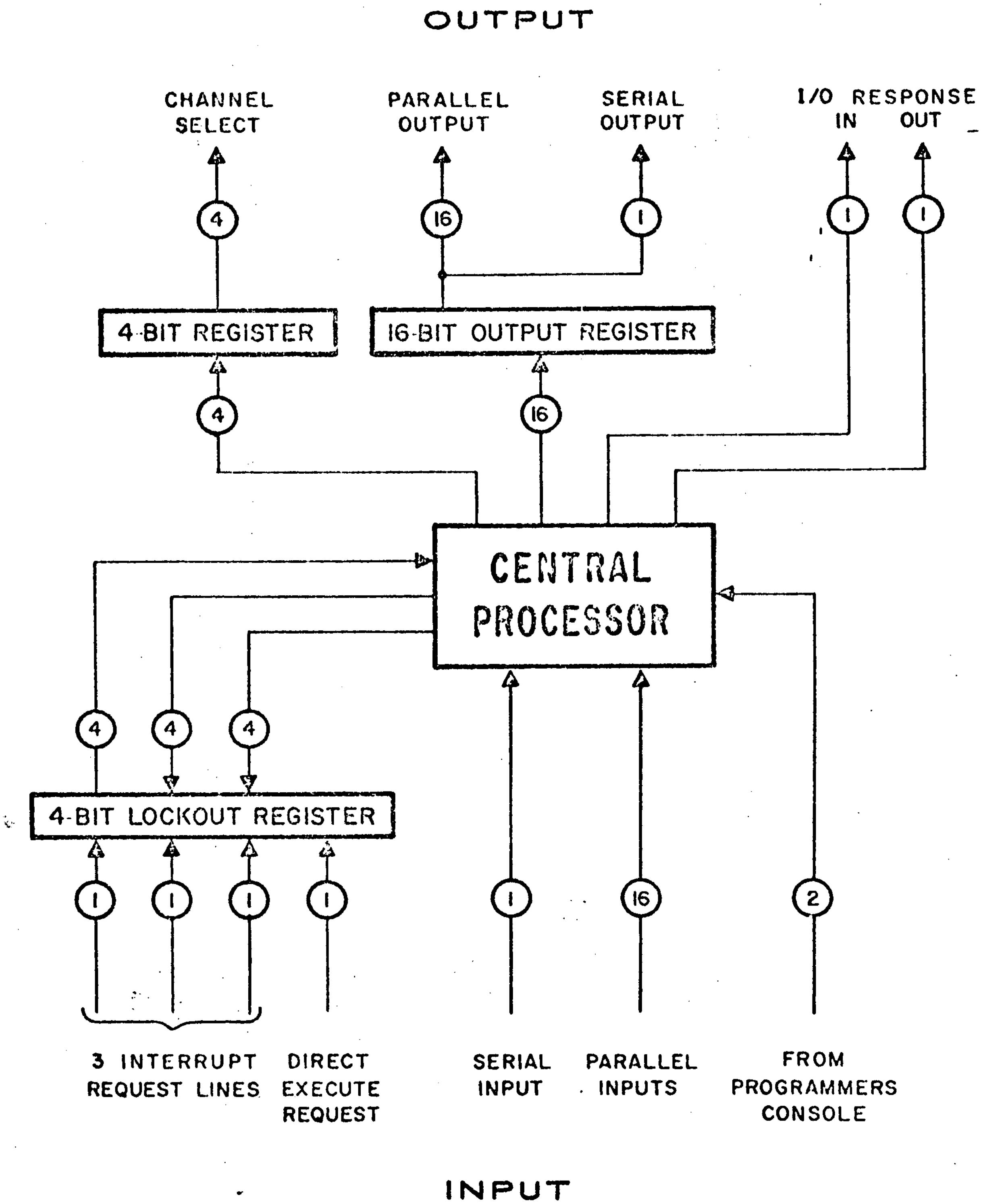

Fig. 26 Data Processor Input/Output Signal Paths

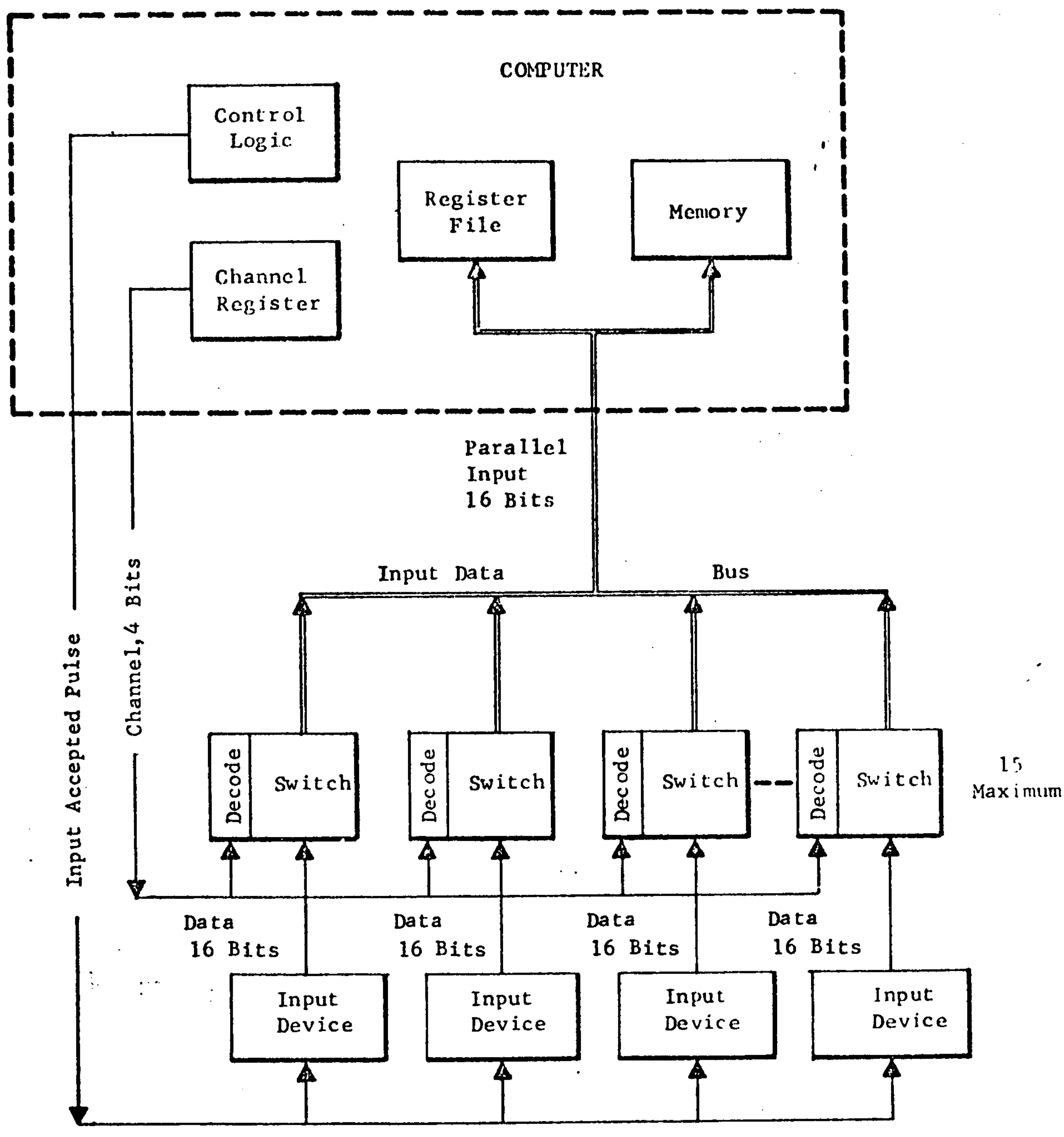

Fig. 27 Data Processor Input Signal Handling & Control

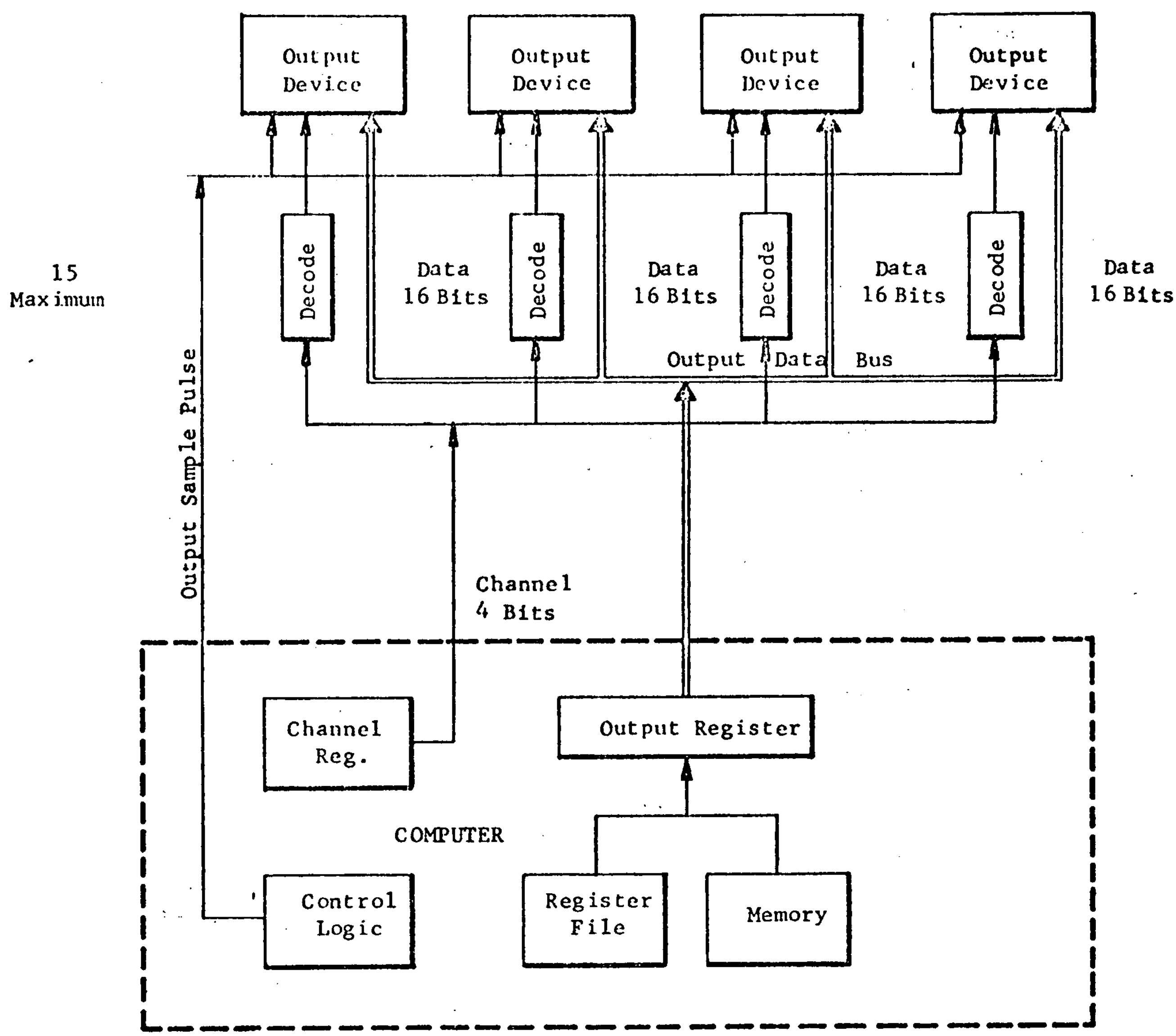

Fig. 28 Data Processor Output Signal Handling & Control

TECHNICAL CHARACTERISTICS

General Data

Weight: 2.5 lbs (8K)
4.0 lbs (16K)

Dimensions: 4.2"h x 4.13"w x 3.0"d (8K)
4.2"h x 4.13"w x 5.0"d (16K)

Power Consumption: ≈12 watts (8K)
≈13 watts (16K)

Input Voltages: ± 15 VDC, + 5 VDC, -3VDC

Circuits: High level PMOS/CMOS/IC devices

Environment: Designed to MIL-E-5400

Cooling: None required ($0^{o}C$ to $+50^{o}C$)

Estimated Reliability:

Central Processor (LSI, 14 devices total)

Type: Binary, parallel, general purpose single address, plus file address

Repertoire: 42 instructions (some double precision)

Word Length: 16 bits

Register Files: 16 Addressable 16-bit word files

Interrupts: 3 external levels, plus 1 direct execute.

Arithmetic: Fractional, fixed point, two's complement. Hardware multiply and divide.
Typical execution times:
Add: 2.4 μsec
Double Precision Add: 3.6 μsec
Multiply: 10.4 μsec
Divide: 30.4 μsec

Memory

Type: Random access, word-organized, NDRO (electrically alterable) Plated Wire Memory (PWM)

Word Length: 16 bits

Capacity: 8K words NDRO expandable to 65K in 8K word increments

Read Cycle Time: 1.6 microseconds

Write Cycle Time: 2.4 microseconds

Access Time: 500 nanoseconds

Input/Output

1-16 bit parallel, party line, buss input
1-16 bit parallel, party line, buss output
4-bit address control lines
1-serial input channel
1-serial output channel
External clock input
69-90 KHz - parallel continuous word rate I/O
400 KHz - parallel burst word rate I/O
130 KHz - serial bit rate I/O

Optional I/O: Solid state keyboard and display suitable for navigational, checkout, and general purpose use can be integral to the computer. Also available are multiplexed A/D input channel(s), buffered and non-buffered peripheral device channel(s) on a Quote Special Equipment (QSE) basis.

Standard Interface: TTL

Software

Assembler, Simulator, Plus Library and Diagnostic Routines

Optional Support Equipment

Programmer's Console with integral CRT display and power supply.

P.W. Memory Loader with integral paper tape reader and power supply.

* Information furnished by Control Data Corp., Minneapolis, Minn.

Fig. 29 CDC-469 Example Computer*

Unit Configuration	MTBS in Hrs x 10^3	Cost in Dollars*
Type 1:		
• One 8K Memory Computer • High Reliability Program	75	65,000
Type 2:		
• One 16K Memory Computer • High Reliability Program • One Redundant 8K Memory included	~ 111	125,000
Type 3:		
• One 24K Memory Computer • High Reliability Program • One triply Redundant 8K Memory included	~ 167	175,000

* Average Cost per Computer - Lots of 8 assumed.

Fig. 30 Configuration & Cost Data on CDPI Computer Representative Example

Configuration	Qty.	Overall Reliability	Overall Cost
Redundant 8K Memory with single CPU	2 Computer	> 0.95	$ 250,000/spacecraft
Programmer's Console (Ground & Shuttle)	2	-	19,600
Plated Wire Memory Loader	2		23,400
			$ 293,400
Basic Software Cost			170,000
Validation Expense (60% basic cost)			102,000
		Overall Total	$ 665,400

Non-Recurring Costs

Console	$ 19,600
Loader	23,400
Software	272,000
	$ 315,000

Recurring Costs

60% Software - $163,000

Fig. 31 CDPI Data Processor Section Hardware and Software Estimated Costs

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4.0 Standardized Design and Packaging

In order to utilize the designs of the CDPI in configuring a standard spacecraft, a modularized packaging structure will be used. The techniques outlined in Reference 1 apply; however, simplifications are introduced in that data rate requirements are not as severe and power levels are much less than those of the earlier design. Thus, a single module containing the entire CDPI will suffice. This is readily accomplished considering the following totals:

	Wt. in Lbs.	Size in Inches^3	Power in Watts
Communication Section	33.0	695.6	65.2
Interface Section	20.5	1300.0	30.0
Data Processing Section	8.0	86.0	13.0
Total	61.5 lbs	2081.6 in^3	108 W

Since the standard module being allocated is 13,824 in^3, there is considerable room available for relaxing packaging, structural and other requirements -- an approach which will lead to a lower cost. This pertains in particular to the low frequency circuits, e.g., the instrumentation outputs to the Interface Section and the control signals to the various spacecraft subsystems.

5.0 Conclusions and Recommendations

A CDPI for a standardized communication satellite has been described to furnish a second point design of such a subsystem. It is seen that functionally the requirements for this spacecraft CDPI differ markedly from the first, a design for an EOS. These differences include the following:

Communication Satellite	EOS
• Direct ground communication	• Relay satellite communication
• Five-year period between visits	• 6-month period or more frequent visits possible
• High reliability requirement	• Reliability requirement relaxed
• High cost penalty from satellite outage	• Satellite outage costs indeterminate at present time
• Mission Equipment is a communication network subsystem. S-Band communication suffices for CDPI functions; may be possible to use VHF	• CDPI must satisfy communication requirements for mission equipment. This leads to handling of higher data rates, broader frequency coverage, etc. K-Band communication necessary; S-Band and VHF also needed.

Despite these differences, a number of similarities are beginning to emerge. For example, the same computer type may be used for both configurations, and the low data rate Interface Unit in the EOS CDPI bears a strong resemblance to the Interface Section of the communication satellite CDPI. Although it is too early to tell, and further study is required, commonalities do appear to exist which may be exploited in realizing a standardized, multi-mission spacecraft.

Some trade studies and other investigations have been suggested in the body of the report or may be suggested at this time and further work in these areas should be carried out. These include incorporating mission equipment communication functions into a standardized CDPI communication section, study and development of low-cost techniques for uplink command verification, fault isolation, detection and correction equipment, and study of optimum spacecraft/Shuttle interfacing.

The reliability goal for the CDPI has been specified at 0.938 for a 5-year period. The design outlined herein meets this reliability requirement.

6.0 List of References

1. Standard Earth Observatory Satellite CDPI Subsystem - EM PE-103; M. Loeb, D. F. Wald - LMSC; dtd 30 Nov 1971

2. LMSC Data on Domestic Communication Satellites - dtd Feb 1971

3. LMSC Data on International Communication Satellites - dtd 8 Apr 1968

4. Domestic Communications Services via Satellites - COMSAT Pilot Program - Robert D. Briskman; AIAA Conference Paper No. 68-412; Dtd 8-10 Apr 1968

5. COMSAT Standard Spacecraft Stabilization and Control - EM PE-122; R. J. Pollak - LMSC; dtd 24 Dec 1971

6. Telephone Conversation with D. Simpson CDC -

7. Analytic Notes - H.K. Burbridge - Undated

8. Low Cost OAO Payload - EM P-3; A.Jeung, M. Loeb; Dtd. 25 Feb. 1971

PE-124

STANDARD U.S. DOMESTIC

COMMUNICATION SATELLITE

ELECTRICAL POWER SUBSYSTEM

PE-124

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: STANDARD U.S. DOMESTIC COMMUNICATION SATELLITE: ELECTRICAL POWER SUBSYSTEM	EM NO: PE-124 REF: DATE: 24 December 1971
AUTHORS: G. A. Deppe — Prepared under cognizance of: Payload Integration, Orgn. 69-02 Space Systems Division	APPROVAL: J. C. Bolton ENGINEERING: Wayne Miller

16 pages

Contents

PRELIMINARY

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1.0 Introduction

The electrical power subsystem described herein is a recommended preliminary design for incorporation into a standard shuttle-launched spacecraft to perform the U. S. Domestic Communications Satellite mission. The electrical load is an average of 1750 watts.

2.0 Requirements and Guidelines

The mission requirements are:

1. Earth synchronous altitude
2. Life of 5 years
3. 1750 watt average load

The main guideline is low overall mission cost. Since the satellite is to be launched by the Space Shuttle, lower costs can be realized due to the reduced satellite weight and volume constraints relative to those imposed by current booster launch systems. Other advantages offered by the Shuttle-launch concept are these: in-orbit checkout, in-orbit maintenance and, return of the spacecraft to ground for refurbishment, if warranted. 1971 state-of-the-art is to be used. Minimizing non-recurring costs (early expenses) to a greater extent than recurring costs (later expenses) is a goal.

Considering these guidelines, the following design approaches are appropriate:

1. Modularize the components into modules which
   a. Are small enough in size for in-orbit handling and replacement
   b. Have efficient interfaces with other modules (minimum interconnects)
   c. Are low enough in cost to be used as on-orbit replacement modules
2. Subsystem modules to be:
   a. Solar Array Module (2 required)
   b. Solar Array Drive Assy
   c. Battery and Battery Charge Control Module (2 required)
   d. Power Distribution Module
   e. Harness Module
3. Little checkout at launch pad; deployments and final in-orbit checkout via an umbilical to the checkout set in the Shuttle before final release.
4. Utilize non-graded solar cells with larger solar array area (cost saving on purchased cells).
5. Minimize redundancy (less material and fabrication costs).
6. "Overdesign" to minimize acceptance and qualification testing (less testing labor).
7. Avoid miniaturization (less fabrication costs).

3. Description of the Electrical Power Subsystem (EPS)

The proposed EPS is sized to deliver 1750 watts of power to the equipment load. It consists of a sun-tracking solar array which is voltage controlled by a shunt regulator, 4 nickel cadmium batteries which are charged by a constant current, and a DC-DC regulator. The system provides a nominal 28 volt DC bus which varies between 25.0 and 28.5 volts. The DC regulator will provide 28.0 VDC $\pm$ 2% for the equipment needing close regulation. A functional block diagram is shown in Fig. 1.

Primary power is supplied by the sun oriented solar cell array and augmented with nickel cadmium batteries during eclipse periods. The solar array incorporates a light weight flexible substrate and the design can be of the retractable or non-retractable type. The system is the direct energy transfer type, that is, the array is fed directly onto the bus. Excess array sections are shorted to ground, with most of the excess energy dissipated as heat out on the solar array. A small amount of heat is dissipated across the shunt power transistors. The solar array regulator unit senses the bus voltage and turns the appropriate number of shunt transistors on or off to maintain bus voltage limits. This unit responds fast enough to control the array cold-to-hot voltage transient. This type of power system is used in at least two other LMSC proposals for communication satellites.

4. Equipment List

An equipment list showing size and weight for each item is shown in Table 1.

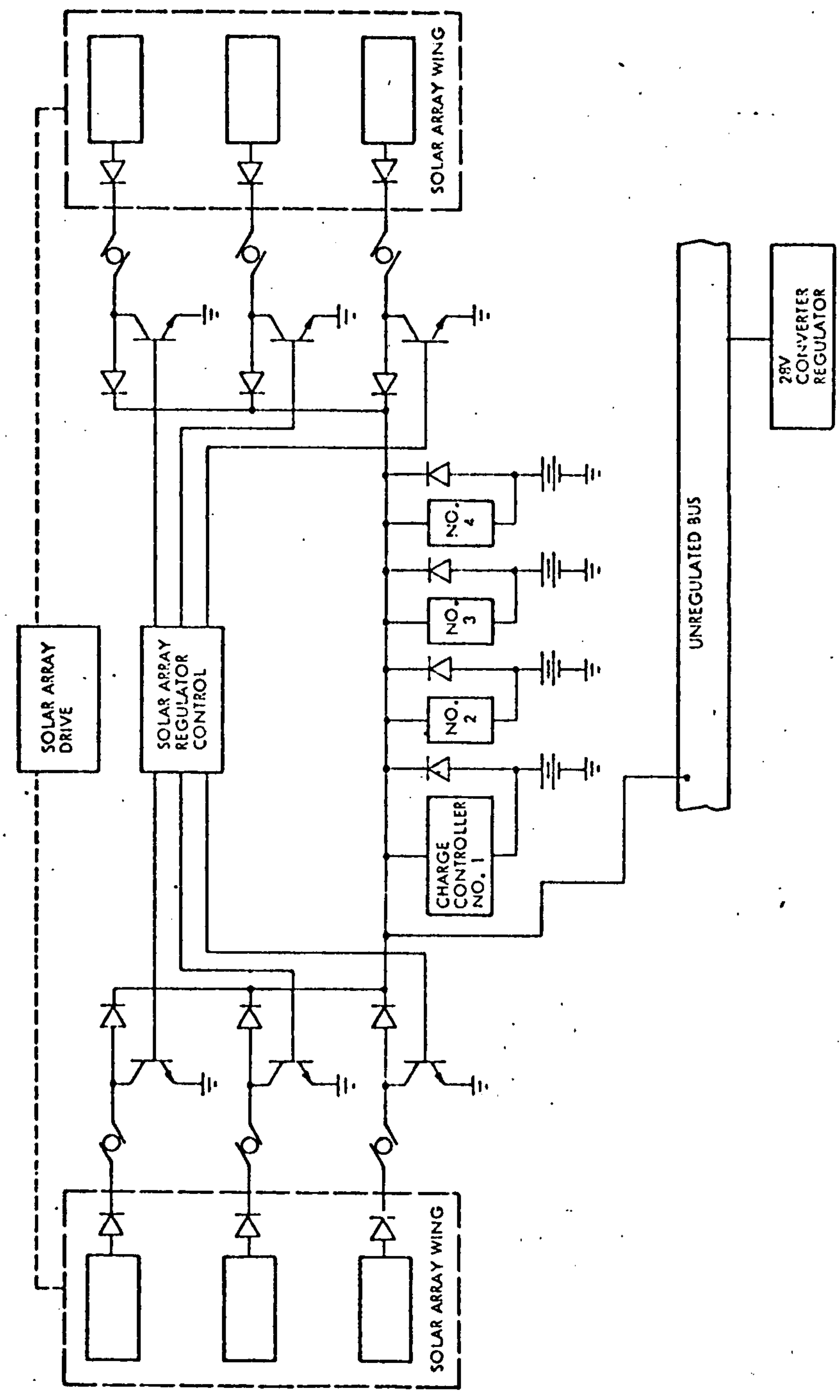


Fig. 1 Standard Communication Satellite - Electrical Power Subsystem

Table 1

EQUIPMENT LIST

Component	Size	Wt. (ea. lb)	Per Vehicle Qty.	Per Vehicle Wt (lbs)
Solar Array Module one module consists of: 1 containment box 132 sq ft .001 Kapton substrate and printed circuit 2 FCC harnesses 13,150 solar cells (2 x 4 CM) 13,150 covers, fused silica	84 x 36 x 6 in.	109	2	218
Extendable Boom Assy.	8 x 14 in.	19	2	38
Solar Array Drive Assy. (drive motors, electronics, slip-rings)	5 in dia. x 10 in length	32	1	32
Solar Array regulator	8 x 11 x 6 in	8	1	8
NiCd Battery (Type VII)	7 x 7 x 21 in	70	4	280
Charge Controller	9 x 10 x 10 in	10	4	28
Power Distribution Unit	20 x 16 x 5 in	34	1	34
DC to DC Regulator	11 x 20 x 8 in	40	1	40
Harness Assy	-	160	1	160

Refer to section 7.0 for total EPS weight.

5.0 Equipment Description

Solar Array Description

The solar array is of the flexible substrate type. Extensive development work has been done at LMSC on this type of array, both as a NASA-funded program for the Large Space Station program and on company sponsored programs. A similar type of array has been flight tested.

The array consists of two wings, one each on opposite sides of the spacecraft as shown in Fig. 2. Each wing is divided into section is controlled by a shunting element for voltage control, as shown in Fig. 1. The power from each section is carried to the base of the array via flat conductor cable (FCC) feeder harnesses. The substrate assembly is the Lockheed-developed Kapton/FEP integrated copper interconnect design. Two parallel contacts on the cell back provide both mechanical fastening and positive power connections; raised series tabs provide negative power connections. Individual modules are joined together mechanically after manufacture in order to form a wing section of the desired length.

Each solar array wing is stowed in a flat pack aluminum box before deployment. This containment forms the structural protection from vibration and acceleration loads and provides support for the solar array wing ends during on-orbit operation. The solar array is folded much like a fire hose and packaged within the flat containment with layers of open cell foam padding between modules. This provides vibration damping to prevent damage before deployment. The folded array is preloaded to assure a tight fit to withstand the vibration environment.

When the array is deployed in orbit, the containment box is positioned 24 inches from the side of the spacecraft to avoid array shadowing and possible thruster deposits on the array surface. The array is then deployed from the box. Both deployments are accomplished by a BI-STEM* extendable boom.

Extendable Boom

The extendable portion of the boom is in two halves. Each half is a flat tape when wound on its storage spool. When it is extended, it assumes a pre-formed curve shape and the two halves combine with an overlap to form a rigid rod. Two reversible DC motors drive the storage spool through a differential, so that either motor is redundant to the other. The operation is reversible.

* BI-STEM is a patended device manufactured by SPAR Aerospace Products, Ltd., Toronto, Canada.

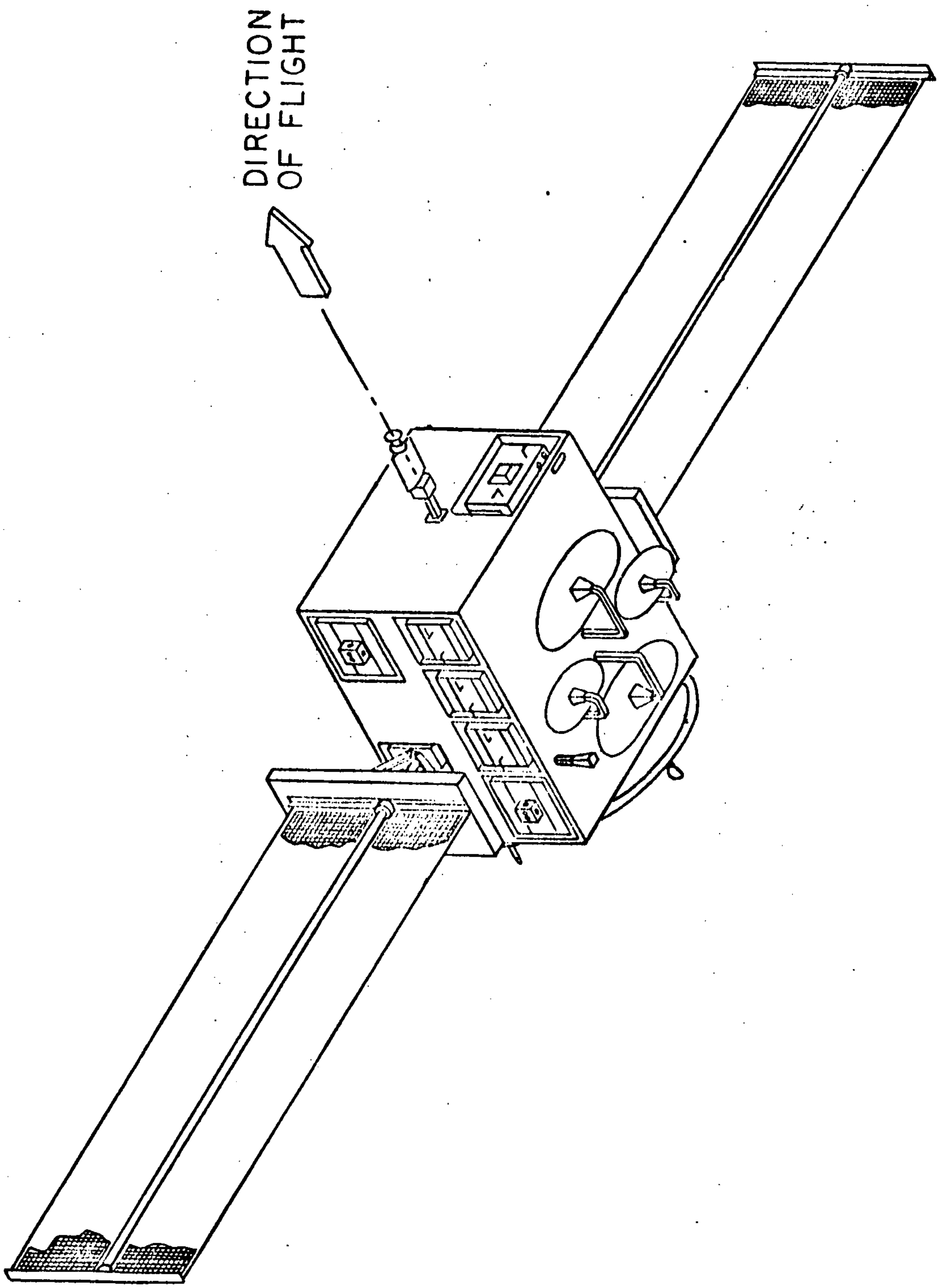


Fig. 2 General Configuration - Standard Communication Satellite

Deployment

In operation, spring-operated releases free the containment box from the spacecraft and open the box lid. The boom starts to deploy, pushing the box out. Spring-loaded extension arms lock into place when the box has reached its proper position. As the boom continues to extend, the folded array is pulled out of the containment box until it is fully extended into a flat sheet. A constant tensioning device is incorporated into the system which assures a uniform preload on the flexible substrate. This preload determines the flatness and natural frequency of the array sheet. After deployment, the boom is locked in place and any dimensional variations in the sheet length due to creep or temperature changes are absorbed in the tensioning device.

Solar Cells and Covers

The cells are 2 by 4 CM, N on P, .012 inches thick, 2 ohm-CM base resistivity, with wrap-around contacts. 97.5% of the production cells (after mechanical screening) will be used instead of the usual 66.7% yield. This reduces cell costs by 30% with a 2% increase in the number of cells needed (2% weight increase). The coverglass is .020 inch fused silica for a maximum radiation protection (less cells) and less handling breakage (higher yield). Solar arrays for conventionally boosted spacecraft systems typically use .012-.014 inch thick coverglass for the weight advantage.

Solar Array Sizing Calculations

The output required from the solar array is the following:

	1750.0	watts	average EOL equipment load
+	50.	"	C/30 battery normal charge rate
+	35.	"	2% wiring losses
+	70.	"	losses across array blocking diodes
+	70.	"	losses across array regulator blocking diodes

1975. watts ≈ 2000 watts needed from array

2 x 4 CM Cell Characteristics:

Cell Output, Normal to Sun, Beginning of Life
Avg. cell power, $28^{\circ}C$, AMO, 66.7 percent yield = 117.5 mW
Avg. cell power, 97.5 percent yield = 117.5 x 0.98 = 115.1 mW
Avg. cell power with 20 mil fused silica coverglass = 115.1 mW x 0.97 = 111.6 mW

$$\text{number of cells} = \frac{\text{power needed from array}}{\text{cell output x angle of incidence x radiation degradation x temperature degradation}}$$

.1116 watts per 2 x 4 cell at 28°C, 20 mil cover glass, and 97.5% yield.

.92 - cosine of 23; (seasonal variation worst case)
.85 - radiation degradation, 5 years
.87 - power decrease from 28°C to 52°C.
(52°C is the array temperature at 23° beta)

$$\text{number of cells} = \frac{2000}{.1116 \times .92 \times .85 \times .87}$$

$$= 26{,}300 \ (2 \times 4 \text{ CM}).$$

With 100 cells per sq. ft. total array = 263 sq ft: (cell area)

Length of array needed for 263 sq. ft. of cell area:

containment box outside dimensions = 84" x 36"

Inside dimensions (area for one fold of flex array) = 83" x 35".

Allowing 3"of non-cell area down the middle of the array for the boom diameter and FCC feeder harnesses, and 1" between cell areas for the folds, the cell area per fold is 80" x 34" = 18.9 sq. ft.

Number of folds needed = $\frac{263}{18.9}$ = 14 folds, or 7 folds per wing.

Length of one wing = 7 x 35" = 245" = 20.4 ft from the containment box.

Width of one wing (box length ID) = 83"

NOTE: The containment box is extended 2 ft from the spacecraft side by the boom, therefore the boom length is 20.4 + 2 = 22.4 ft (one boom per wing)

Solar Array Drive Assembly

This unit performs the sun tracking and power transfer functions. The drive assembly rotates the extension booms and containment boxes to provide sun tracking for the array about a single north-south axis. Both solar array wings are driven simultaneously via a common torque tube. Motive power is from a dc energized stepper drive providing 0.1^{o} incremental steps of rotation. Logic input for the motor is a clock signal which provides orbital rate pulses during both sun and eclipse times. Command override and clock calibration functions are included within the logic.

Power transfer from the rotating solar array to the spacecraft loads is accomplished by multiple slip ring circuits with redundant brushes for each ring.

Solar Array Regulator

The Solar Array Regulator controls the solar array/distribution bus voltage. Essentially, it is a shunt regulator, thus having the desirable feature of no power loss when all of the array capability is needed. The regulator senses the bus voltage, and if the bus tends to rise because of too much array capability, it will short sections of the array to ground successively until the array capability matches the load. The shorting is accomplished through power transistors controlled by the regulator electronics. Large heat dissipating resistors, normally associated with shunt regulators, are not necessary. The only heat to be dissipated is the relatively small amount developed by the one volt drop across the shunting transistor. Most of the excess energy is dissipated as heat out on the large array area. A relay in series with the power circuit of each transistor can be opened in case of failure of the transistor. Ground commands to the regulator can then re-sequence the control program so that the failed transistor(s) is the last in the sequence and thus not needed as long as power is needed.

Batteries

Full load eclipse power is provided by four Ni-Cd secondary batteries. Each battery is connected to the bus through a blocking diode. The batteries are on a low charge rate (C/20 to C/100) when the array is producing power. Heat produced by overcharge (low rate) is dissipated by passive thermal control. The required temperature limits for the batteries is $-10^{o}C$ to $+10^{o}C$. In order to maintain this limit passively, a constant heat dissipation is needed. During solstice periods the heat rate is supplied by the low overcharge. During equinox (eclipse) periods the battery is in discharge or normal charge, replacing the expended energy used during the eclipse. During this time the required constant heat dissipation is supplied by small battery heaters. If the battery goes into overcharge before the eclipse, the heaters will be turned off by battery thermostats.

The batteries are sized at 40 amp-hours each to give a 50% depth of discharge (DOD). The full 50% is reached only twice a year at maximum eclipse. During the remaining eclipse times the DOD is proportionally less. In the event of one battery failure, the DOD increases to 67% for the remaining three batteries.

The battery capacity is calculated as follows:

$$\frac{1750 \text{ watts load}}{25 \text{ volts}} = 70 \text{ amps}$$

70 amps x 7/6 hour (full eclipse) ≈ 40 amp hrs.

The Type VII Ni-Cd battery will deliver the required amount of power.

Charge Controllers

Each battery is controlled by a separate charge controller. One of three different charge rates are selected by ground commands. These are:

(1) C/100 (.4 amps) for full sun periods.

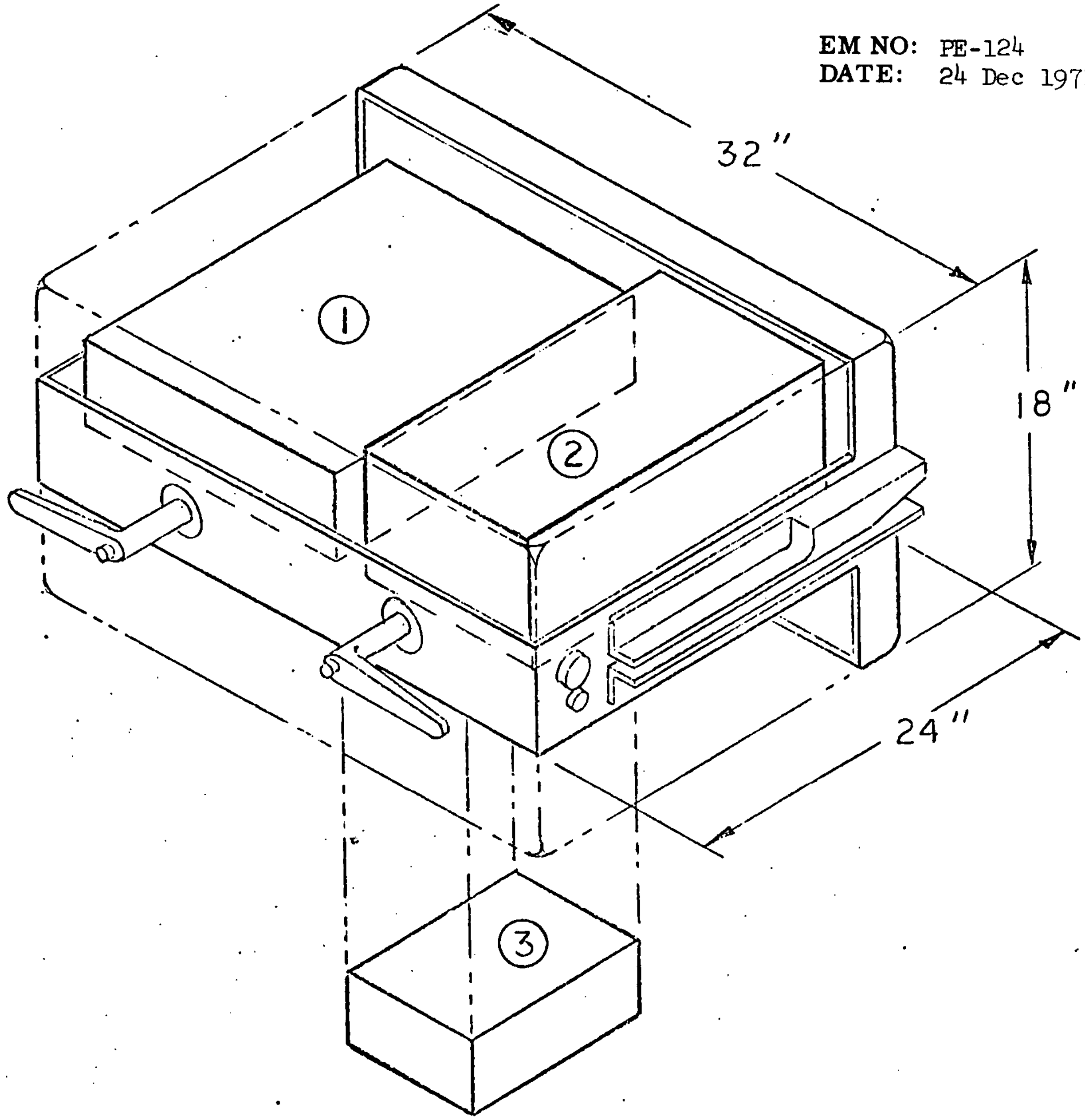
(2) C/30 - C/20 (1.3 - 2.0 amps) for the eclipse periods. This is the normal charge rate for the batteries. The exact charge rate in that range is controlled by the charge controller electronics and is dependent on the battery temperature. It varies linearly with the low rate for the low temperature limit of -10°C and the high rate for the high temperature limit of +10°C.

(3) C/20 (2.0 amps) is a high rate of charge in case the battery is warmer than the design limit of +10°C.

6.0 Modules for the EPS

The modularized concept for the power distribution and regulation is shown in Fig. 3, and the concept for the charge controllers/batteries is shown in Fig. 4.

EM NO: PE-124
DATE: 24 Dec 1971



EQUIPMENT

(1)	1	POWER DISTRIBUTION UNIT	34 lbs
(2)	1	SOLAR ARRAY REGULATOR	8
(3)	1	DC TO DC REGULATOR	40
	1	CABLES AND CONNECTORS (SET)	6
	1	BASE AND COVER	31
		15 % CONTINGENCY	18
			137

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Fig. 3 Power Distribution Module - Standard Communication Satellite EPS

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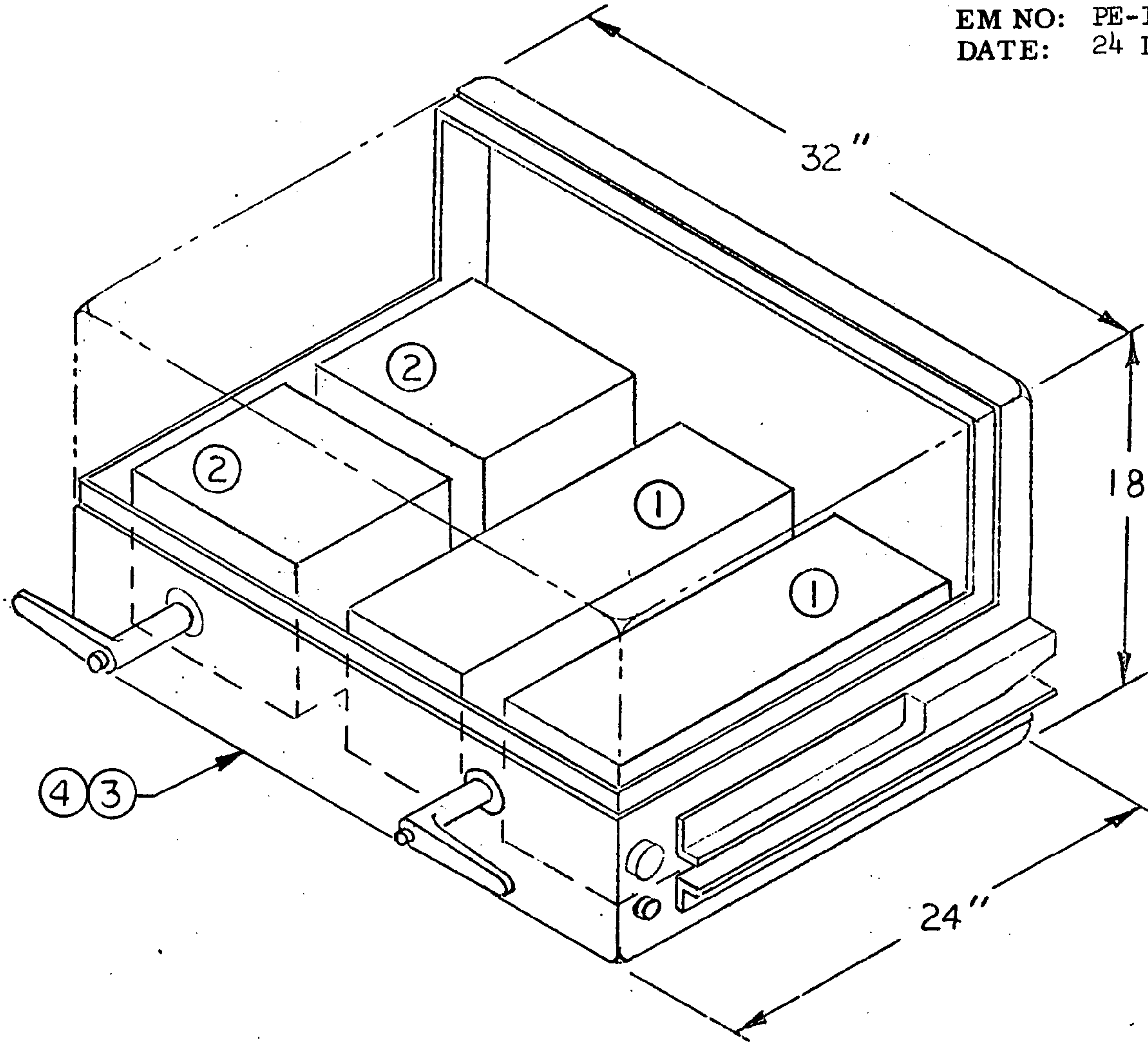
DATE: 24 Dec 1971



EQUIPMENT

(1) TYPE 7 BATTERY - 2 REQD 40 AH NiCd	140 LBS
(2) CHARGE CONTROLLER - 2 REQD	20
(3) BASE & COVER	40
(4) CABLES & CONNECTORS	13
SUBTOTAL	213
10 % CONTINGENCY	21
	234 LBS

Fig. 4 Battery Module - Standard Communication Satellite EPS

7.0 Total EPS Weight

The weights for the EPS are itemized and totaled below:

2 ea. Solar Array Power Module		
Cells, substrate, covers, wires (63.5)		127 lb
Extendable Boom Assy (10.5 lbs)		21
Storage Container (44)		88
Base and Cover		46
		282
15% contingency		42
		324 lb
1 ea. Solar Array Drive Module		
Drive Motor Assy.		20
Electronics		5
Sliprings, Bearings, etc.		27
Base, Cover & Cables		14
		66
15% contingency		10
		76 lb
1 ea. Distribution Module (incl. contingency)		137
2 ea. Battery Modules (234 ea. incl. contingency)		468
1 Spacecraft Harness (set)	160	
50% contingency	80	240
Total Weight, EPS		1245

8.0 Reliability

The reliability goal for the Standard Communication Satellite Electrical Power Subsystem is 0.945 for five years of orbit life. This goal is attained in the designs described in the engineering memorandum as shown in Figs. 5a and 5b.

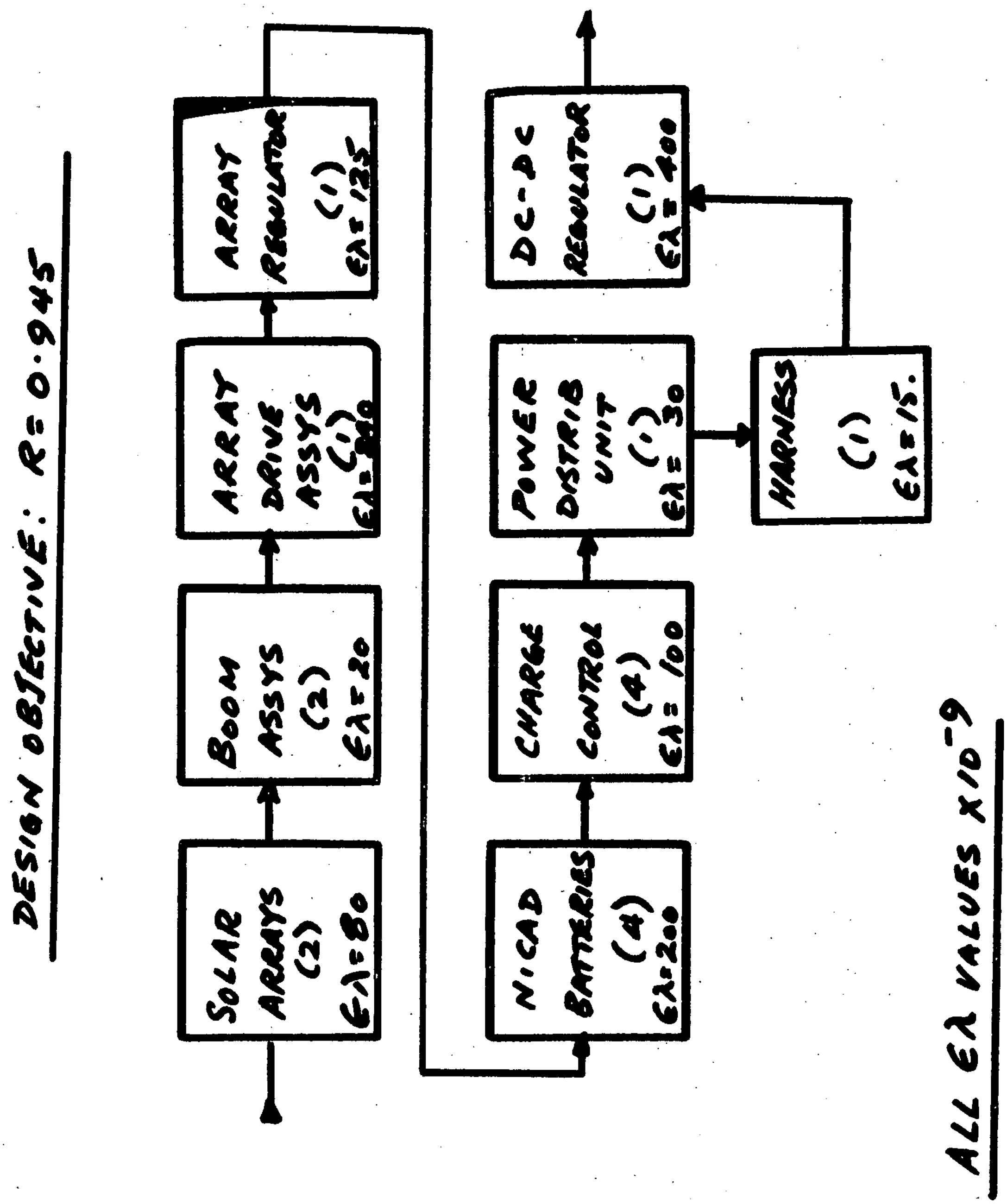


Fig. 5a Reliability Block Diagram - Standard Communication Satellite Electrical Power Subsystem (Sheet 1 of 2)

- Subsystem does not employ classic duplication of equipment with switching to achieve reliability. Uses instead over capacity, low duty cycles, and design safety factors to attain objective of R = .945 for 5 yr. mission. (Exception, DC-DC Regulator)
- Max. permissible S/S λ (from Basic $R = e^{-\lambda t}$) is 1250×10^{-9}

• Solar arrays approx. 30% cells overage (Binomial redundancy)	$\epsilon\lambda = 80 \times 10^{-9}$
• Boom Assys. 1 shot devices, ϵ factor very small	$\epsilon\lambda = 20$
• Charge Controllers (4) ϵ factor $\approx$ 10%	$\epsilon\lambda = 100$
• PDU or Main bus, routing only, passive device	$\epsilon\lambda = 30$
• NiCad batteries (4) Duty cycle 10% Discharge, safety factor 2	$\epsilon\lambda = 200$
• Array regulator. Parts count low. Large derated shunts dynamic operation in high load state $\approx$ 15%	$\epsilon\lambda = 125$
• Harness. Passive device only, all conductors sized so that load vs stress < 10%	$\epsilon\lambda = 15$
• Array Drive Assy. torque load < 15% of rated capability	$\epsilon\lambda = 240$
• DC-DC Regulator. Internal full active redundancy	$\epsilon\lambda = 400$
• $\underline{\Sigma\epsilon\lambda = 1210 \times 10^{-9}\ R = .951 \text{ for 5 yrs.}}$	$\underline{\Sigma\epsilon\lambda = 1210\ (10^{-9})}$

- Method. Failure rate for each device is summed and Rel. is computed taking into account duty cycle, stress level, over capacity, etc. From the resultant R value, an effective λ ($\epsilon\lambda$) is recalculated for the λ budget.

Fig. 5b Reliability Block Diagram - Standard Communication Satellite Electrical Power Subsystem (Sheet 2 of 2)

PE-125

STANDARD U.S. DOMESTIC

COMMUNICATION SATELLITE

ATTITUDE CONTROL SUBSYSTEM

PE-125

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: STANDARD U.S. DOMESTIC COMMUNICATION SATELLITE - ATTITUDE CONTROL SUBSYSTEM	EM NO: PE-125 REF: DATE: 24 December 1971
AUTHORS: J. Cizauskas — Prepared under cognizance of: Payload Integration, Orgn.69-02 Space Systems Division	APPROVAL: J C Bolton ENGINEERING Wayne Miller

15 pages

Table of Contents

PRELIMINARY

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1.0 GENERAL

The Attitude Control Subsystem (ACS) provides thrust for vehicle translation, reaction wheel unloading, and attitude control. Translation maneuvers include docking, both North-South and East-West stationkeeping, correction of injection errors after ground tracking, and the arrest of drift to final station. Attitude hold is provided for translation thrust misalignment, docking maneuvers, reacquisition, and backup attitude hold. The ACS consists of four identical modules installed on the outboard edges of the vehicle as depicted in Fig. 1. Each module contains six 0.5 lbf rated hydrazine thrusters and hydrazine to provide a combined total impulse of 136,300 lb-sec for all four modules, and hence a five year orbital life. Individual thrusters are oriented to provide attitude control in the event of failure of any one thruster and in most cases for failure of more than one thruster, as shown in Fig. 2.

2.0 DESCRIPTION

Each modular propulsion system consists of five major components as depicted in the Fig. 3 schematic. Two independent fill valves are used; one to load 171 lbs of hydrazine propellant through the bottom port of the storage tank and 3.5 lbs of nitrogen pressurant through the top port. After filling and pressurization of the tank, the fill valves are turned closed with a wrench and capped for leakage redundancy. The storage tank delivers hydrazine propellant through a 25μ rated inline filter to the thruster valve inlets over a 230 to 115 psia pressure range. As the propellant is consumed by the thrusters, the nitrogen which initially occupies 50% of the tank volume expands from the initial 230 psia to 115 psia as the tank is emptied. A propellant management baffle in the shape of a cross extends from the upper polar cap of the tank to the lower polar cap where the propellant outlet is located and assures proper propellant orientation during vehicle maneuvers. Figure 4 shows a typical propellant management baffle design for this application.

Each of the six hydrazine thruster assemblies is further protected from contamination with a 25μ rated absolute filter built into the valve inlet. A dual series valve arrangement is used on each thruster assembly to protect against long term leakage and failure of an individual valve. The valves nominally operate over a 25 to 33 VDC range, and have an inlet burst pressure rating of 1900 psi, a factor of 8 over the maximum working pressure. Each thruster is rated at 0.5 lb when supplied with 285 psia hydrazine. However, it is planned to use them over a 0.4 to 0.2 lb range as the storage tank feed pressure decays from 230 to 115 psia. Thus the thrusters meet the maximum 0.4 lb allowable thrust requirement.

Each thruster chamber contains Shell 405 catalyst which decomposes hydrazine into ammonia and nitrogen exhaust products. A large percentage of ammonia disassociates into hydrogen and nitrogen before it leaves the decomposition bed. The resultant hot gases expand out the thruster nozzles thereby producing external control forces on the vehicle.

Thruster valves are driven by the Attitude Control Subsystem Electronics* and provide less than a required .02 lb-sec minimum impulse bit using a typical 20m sec pulse width (.40 lb x .02 sec. = .008 lb-sec). Each thruster incorporates an approximately 1 watt heater to maintain catalyst bed temperature at a 150°F minimum. Testing at the subcontractor and LMSC has shown that catalyst bed life is greatly enhanced when the start temperature is elevated above **the orbital ambient.**

*** Part of the Stabilization & Control Subsystem but mounted in the ACS modules.**

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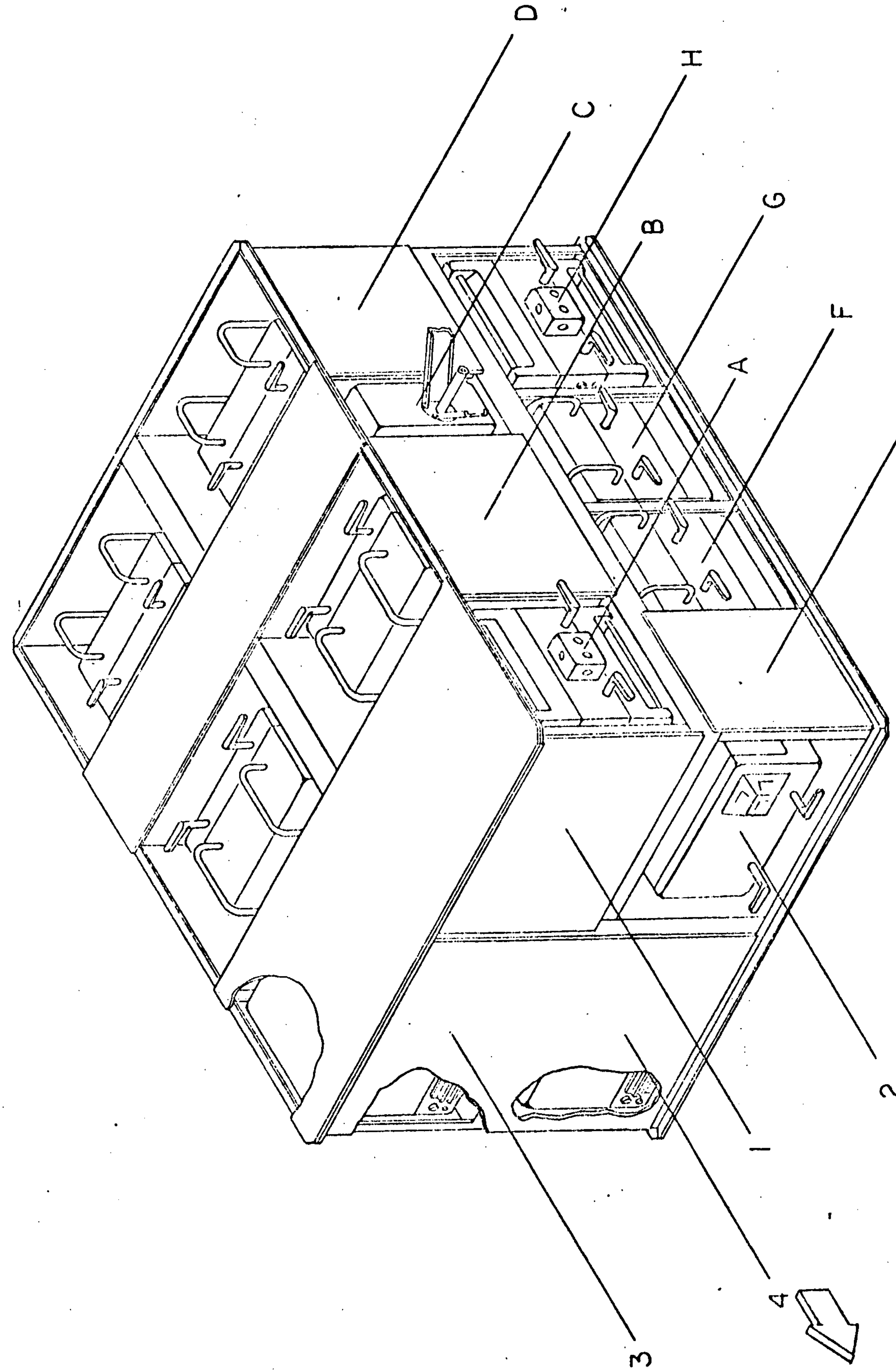

Fig. 1a Equipment Locations - Standard Communication Satellite (Sheet 1 of 2)

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C.7

Spacecraft Subsystem Modules		**Mission Equipment**	
A-2	Attitude Control Module No. 1	F-2	Transponder Module No. 1
A-3	Attitude Control Module No. 2	F-4	Transponder Module No. 2
B-1	Battery Module No. 1	G-2	Transponder Module No. 3
B-2	Battery Module No. 2	G-4	Transponder Module No. 4
C-1	Solar Array Drive Module		
D-1	Power Distribution Module		
D-3	CDPI Module		
E-2	S&C Sensing Module		
E-4	Momentum Wheel Module		
H-2	Attitude Control Module No. 3		
H-4	Attitude Control Module No. 4		

Fig. 1b Equipment Locations - Standard Communication Satellite (Sheet 2 of 2)

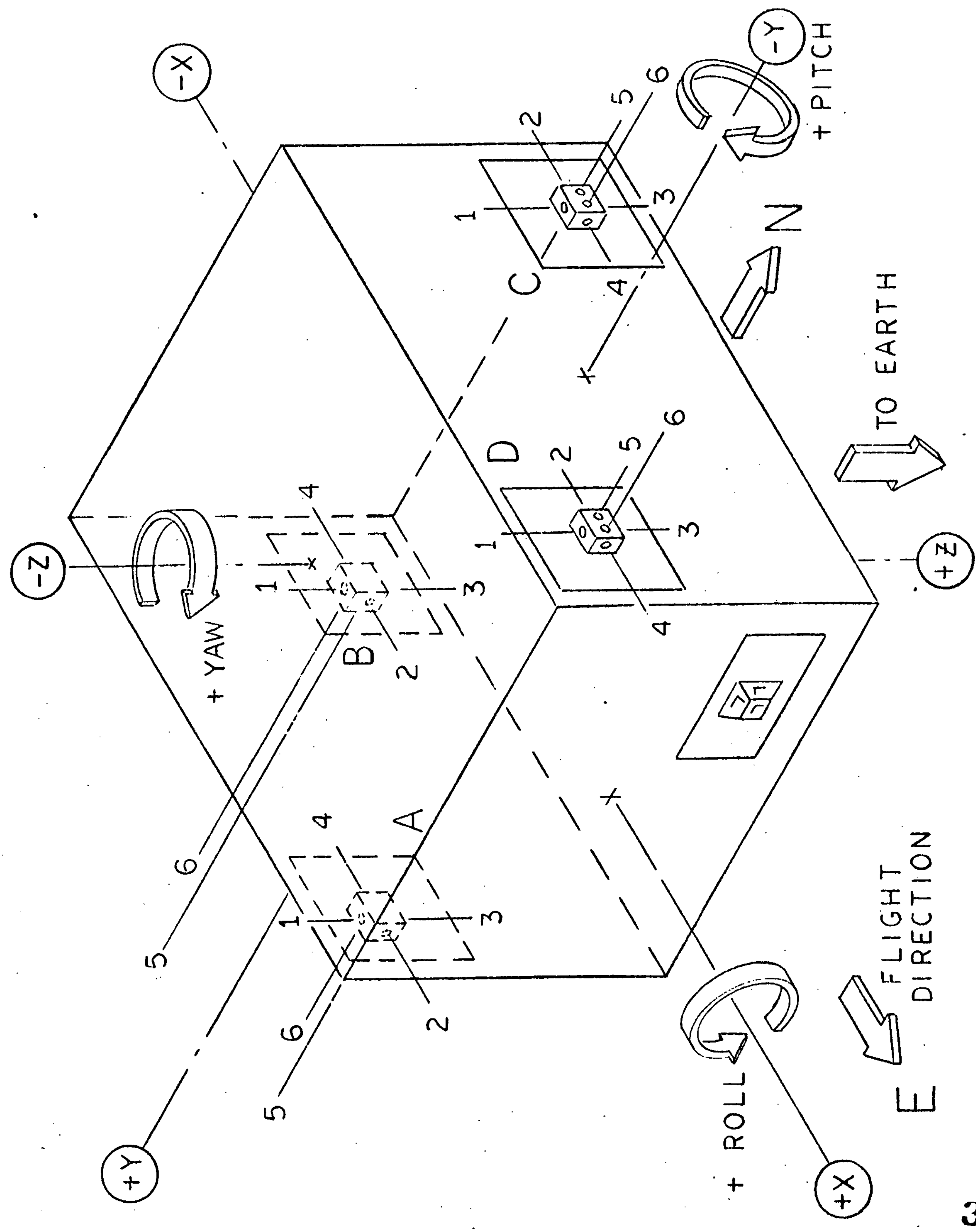

Fig. 2a Attitude Control Thruster Orientation - Standard Communication Satellite (Sheet 1 of 2)

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Vehicle Motion	Active Thrusters
+ Pitch	A3 & B1 or C1 & D3
- Pitch	A1 & B3 or C3 & D1
+ Roll	A1 & D3 or B1 & C3
- Roll	A3 & D1 or B3 & C1
+ Yaw	A2 & D2 or B2 & C2
- Yaw	A4 & D4 or B4 & C4
E Translation	A4 & D2 and B4 & C2
W Translation	A2 & D4 and B2 & C4
N Translation	A5 & B6 or A6 & B5
S Translation	C5 & D6 or C6 & D5

Fig. 2b Attitude Control Thruster Orientation - Standard Communication Satellite (Sheet 2 of 2)

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DATE: 24 Dec 1971

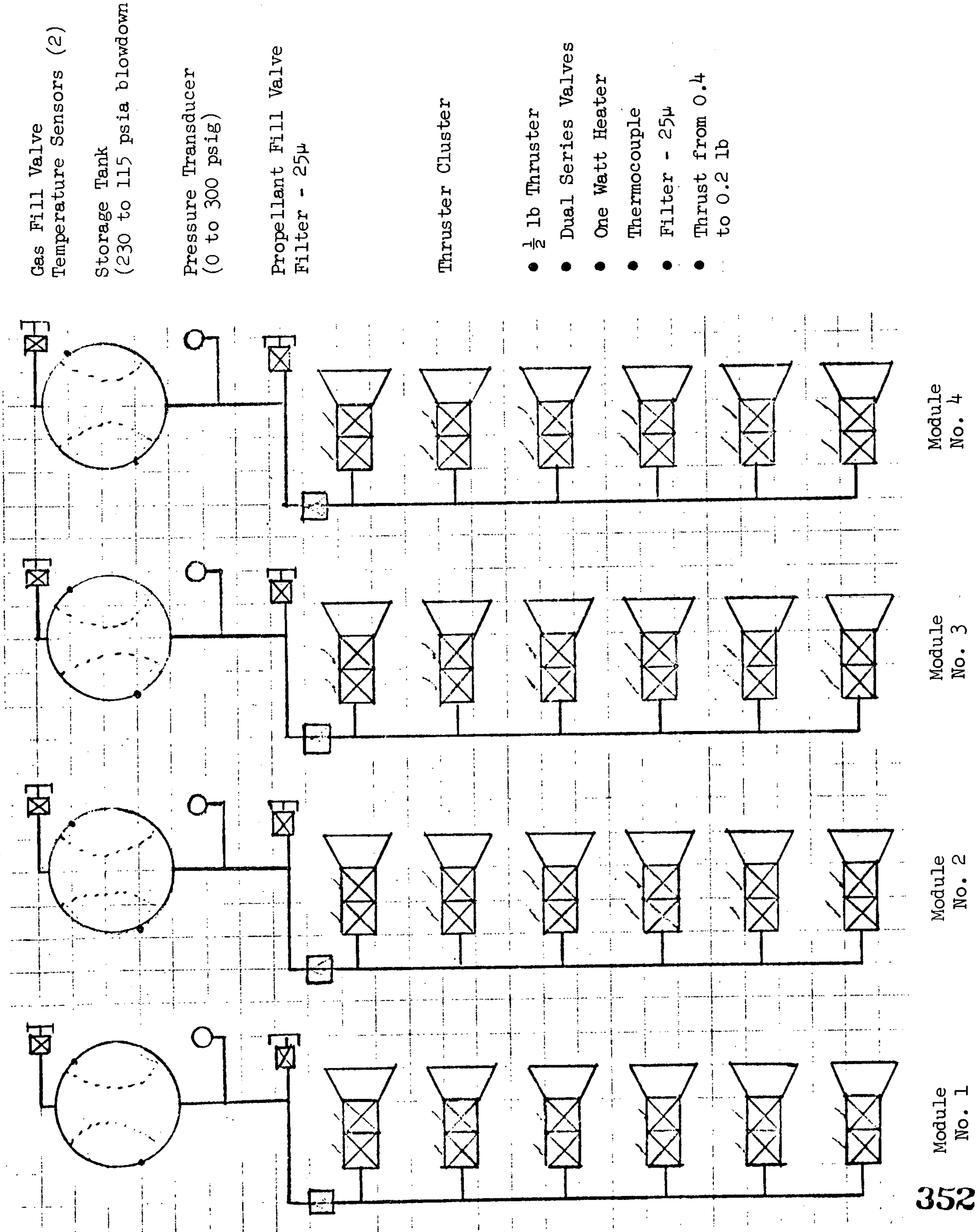

Fig. 3 Attitude Control Subsystem - Standard Communication Satellite

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DATE: 24 Dec 1971

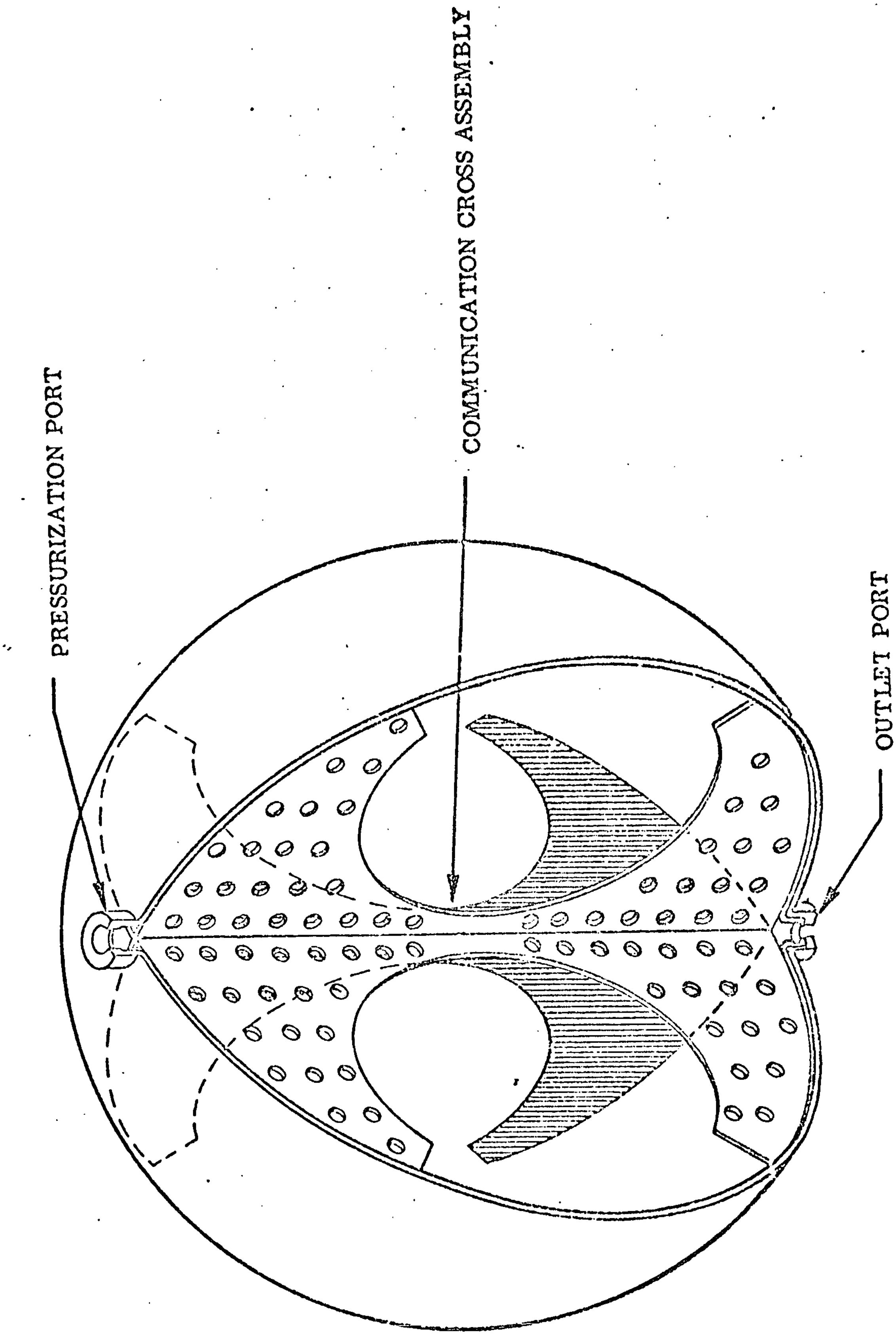

Fig. 4 ACS Storage Tank Propellant Management Device

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3.0 SENSING ELEMENT

Each module contains one pressure transducer, and eight temperature sensors. A 0 to 300 psig pressure transducer is located at the tank outlet and two temperature sensors attached to the external tank skin. These measurements are provided for gas loading, safety minitoring, orbital propellant mass statusing, and leakage detection. A temperature sensor is attached to each thruster to assess performance and leakage. Leakage into hydrazine thruster beds causes significant temperature rise. Thruster valve activity is monitored via the valve driver circuits in the ACS Electronics.

4.0 MODULE

The ACS module is assembled using the major equipment listed in Table I, plus miscellaneous tubing, bracketry, and electrical harnesses. All components are contained in the module (Fig . 5) which can easily be inserted or removed from the vehicle in orbit. Tubing is hard-line stainless steel, brazed to component fittings, and designed to minimize potential leakage points. Each module assembly is locked in place by two cam mechanisms actuated by handles on the outboard face of the module. The installation of the four modules as shown in Fig. 1 permits the use of identical modules and hence minimizes the inventory of modules required to support the program.

5.0 ASSEMBLY & TEST

The module can easily be bench tested due to its low dry weight, size and simplicity. All the previously acceptance tested major components are installed, connected with the appropriately designed tubing and brazed. The wiring harness is connected and instrumentation is connected to a test panel. The system is pressurized to a low value, leak checked, and functionally checked out by providing command signals to individual thruster valves. Upon completion, the module is stored in a slightly pressurized condition to prevent moisture from entering the tank and feed plumbing. Periodic pressure reading checks assures that the system has not developed a leak and is flight ready.

6.0 PRE-FLIGHT SEQUENCE

The module is serviced prior to vehicle installation. Exactly 171 lb of hydrazine is loaded into the storage tank through the bottom fill valve. The fill valve is turned closed, the fill line disconnected, and the fill valve leak checked. A cap is then installed on the fill valve for redundant leakage protection. Exactly 3.24 lb of nitrogen is then loaded through the upper fill valve. A temperature-pressure loading chart is used to assure accurate nitrogen gas loading. Accurate gas loading values are necessary to provide accurate propellant mass statusing. After gas loading, the fill valve is turned closed, fill line disconnected, and valve leak checked. Finally, a cap is installed on the fill valve for redundant leakage protection. The module can now be installed in the vehicle.

7.0 DESIGN OBJECTIVES AND EQUIPMENT SELECTION

Three design objectives were considered and met in designing the modular Attitude Control Subsystem. (1) The subsystem must provide attitude control and translation capability after a single failure, (2) the cost of the module must be minimized, and (3) the subsystem must be simple and safe for ease of ground checkout and man-rated for handling while charged with propellant.

ACS MODULE EQUIPMENT LIST

Equipment	Description	Weight	Cost	Electrical		Remarks
			ROM	Input	Output	
Storage Tank	Made by LMSC 26.4" O.D. sphere 5.41 ft^3 vol. Crossed baffle PM device 1000 psig design burst 2021 aluminum	27 lbs	$115,000 Non-Recurring $3,000 ea			Requires design, development, and qualification
Thruster Cluster Ass'y	RRC P/N MR-6 1400 psi valve burst inlet filter-25µ abs. 1 watt heater per thruster 0.5 lbf rated at 285 psia 6 thrusters per cluster	6.8	$1.6 ± .4 M non-recurring $80K per cluster	25-33V		Qualified individual thrusters Cluster Ass'y not qualified
Fill Valves	LMSC P/N 8106086 Valve Shutoff Plus Cap -3 for Gas Fill -5 for Liquid Fill	 .25 .25	 $400 $400			Qualified
Pressure Transducer	LMSC P/N 8100496-9 0-300 psig	.35	$1600	22-29V	0-5V	Qualified
Temperature Sensors	LMSC P/N 1618702-5 Stick-on (2 req'd)	.07	$100		0-5V	Qualified
Filter	LMSC P/N 8103465-3 25µ abs.	.5	$250			Qualified
Tank Heater	LMSC P/N 1615382 Strip Heater					Qualified

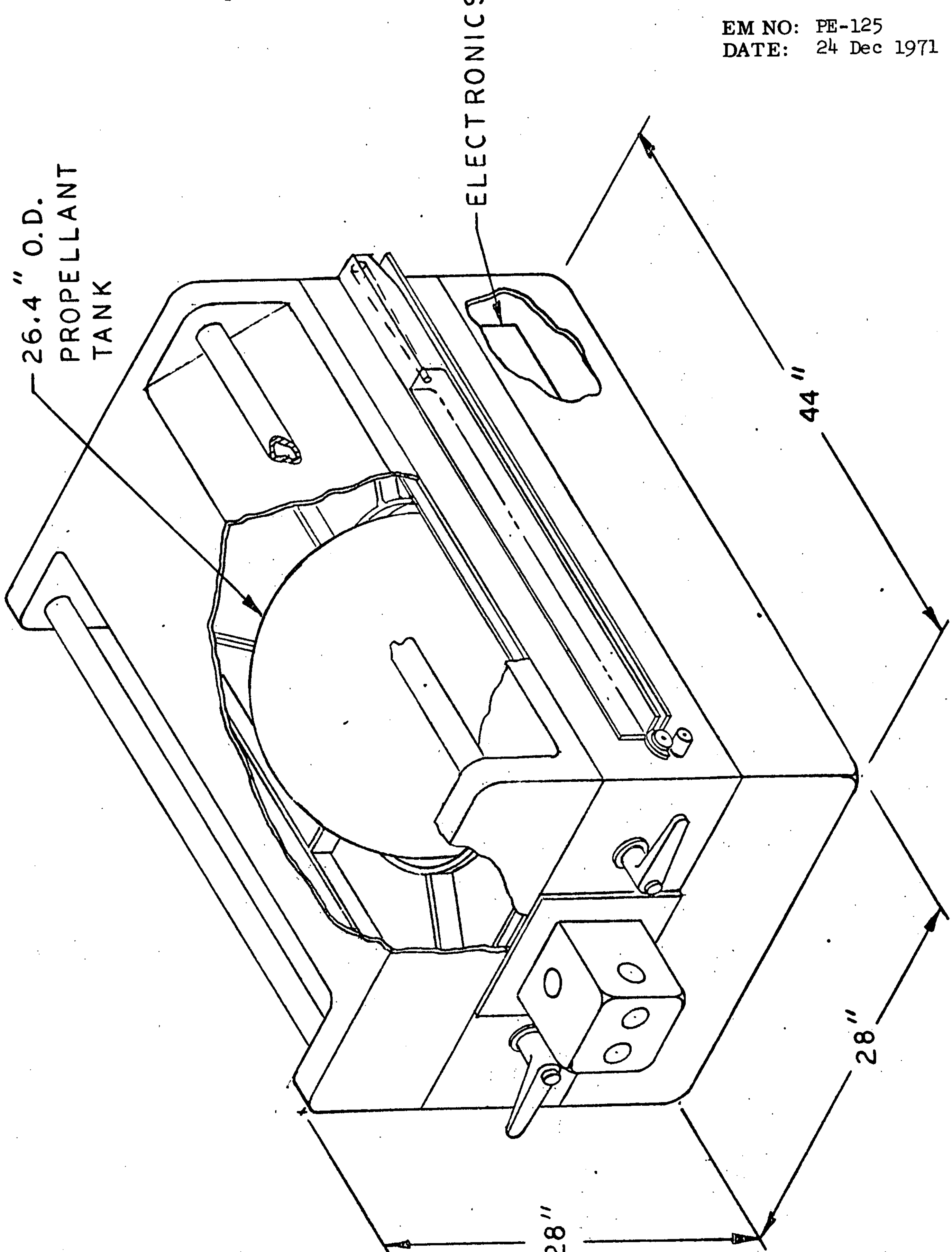


Fig. 5 Attitude Control Subsystem Module Standard Communication Satellite

The first objective was achieved by choosing a simple blowdown feed system using brazed joints and no moving parts, providing redundant thrusters, and the incorporation of dual series propellant valves into each thruster. A failure of one thruster to operate, and in most cases more than one thruster, will not prevent the ACS from performing all its required functions. Failure of one thruster propellant valve to close or not seal will not affect ACS performance, nor allow system leakage. All plumbing joints are brazed to assure no leakage, and redundant seals utilized where it is unfeasible to braze. Hence, the possibility of a propellant supply failure is remote.

The second design objective, low cost, was achieved by specifying existing qualified components to the maximum possible extent, thus minimizing development costs.

Thruster Cluster Assy.

Rocket Research Corporation P/N MR-6 0.5 lb rated thrusters were selected to make up the 6 unit cluster assembly. This thruster is committed to flight on such programs as ERTS, Synchronous Meteorological Satellite and a classified program. As mentioned earlier, this thruster easily meets the .02 lb-sec minimum impulse bit requirement and is below the required 0.4 lb maximum specified thrust level. Three other important parameters evaluated in selecting the MR-6 unit were: (1) number of ambient start pulses, (2) number of hot start pulses, and (3) total impulse.

	Nominally Required	Required in Case of Other Thruster Malfunction	Presently Qualified to	LMSC ID Program
(1) Ambient pulses	1,000	2,000	1800 at 70°F	45,000 at 250° to 125°F
(2) Hot pulses	10,000	20,000	52,000	100,000
(3) Total impulse	15,000 lb-sec	30,000 lb-sec	14,400 lb-sec	-

Based on the above comparison of pulses required to demonstrated performance, there should be no problem in achieving 2000 ambient pulses since it is planned to maintain thruster catalyst bed temperature at 150°F minimum with heaters. Based on experience with other thrusters similar in design to the MR-6, a total impulse life capability of 30,000 lb-sec should be easily achieved. On orbit life of 5 years appears to present no problem; the Synchronous Meteorological Satellite is committed to a 7-year orbital life. In summary, it is planned to use the existing MR-6 design in 6-unit clusters and qualify to the new duty cycle. Past experience has shown that hydrazine thruster life characteristics are duty cycle sensitive. However, based on the MR-6 and similar thruster design test experience there is high confidence that the unit can qualify to the required duty cycle.

Storage Tank

A 26.2 inch diameter aluminum spherical tank incorporating an aluminum propellant management device was selected for storing and supplying gas-free hydrazine propellant to the thrusters. Aluminum 2021 alloy was selected over titanium because of the much lower cost, yet acceptable weight penalty. This alloy is stronger than the 6061 aluminum alloy currently used on LMSC Agena tanks. Stainless steel tanks are competitive from the standpoint of cost, are 25% lighter weight, and could be used alternately. However, LMSC is currently manufacturing and using 60" spherical tanks of 2021 aluminum for a hydrazine flight application. LMSC has recently completed an ID program to develop 2021 aluminum hemispherical shell forming technology using a 22" size tank. Therefore, based on tank shell forming capability and welding procedures developed for 60" tank fabrication, LMSC can successfully produce 26.2" spherical 2021 aluminum tanks. Additionally, the tanks can be produced at a competitive cost due to the large quantity involved.

A propellant management device was selected for this 5-year life application rather than the EPT-10 rubber diaphragm being currently used for some hydrazine applications. Existing data is not sufficient to recommend an EPT-10 diaphragm for more than a two or three year life exposure to hydrazine. The propellant management device consists of a thin symmetrical cross baffle placed between the tank polar caps as shown in Fig. 4. This design has been proposed by LMSC in similar propellant management applications. Should other programs select this design, the non-recurring cost for designing, developing, and qualifying the device can be drastically reduced.

Minor Components

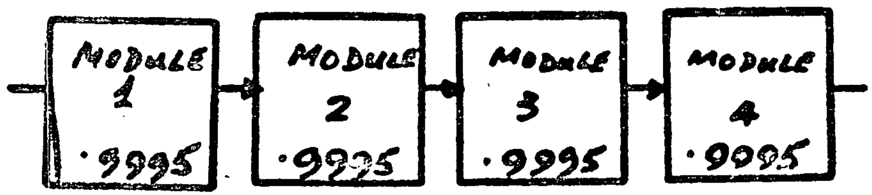
LMSC 8106086 fill valve currently used for hydrazine application was selected for the present application. This valve features a cap for redundant leakage protection after the valve is manually turned closed. LMSC 8100496-9 Whittaker pressure transducer also currently used for hydrazine application was selected for its competitive cost yet high reliability. LMSC P/N 8103465 filter currently used for hydrazine application was selected for the present application. Temperature sensor 1618702-5 and tank heater LMSC P/N 1615382 also currently used by LMSC are specified for this program.

Attainment of the third objective of simplicity and safety is self-evident from the extremely simple design that is presented, hydrazine monopropellant in a simple low pressure blowdown feed system. Pressure vessel design (4 to 1 ratio of burst to working pressure), brazed joints, and redundant sealing, assure safe handling of modules loaded with propellant and pressurized.

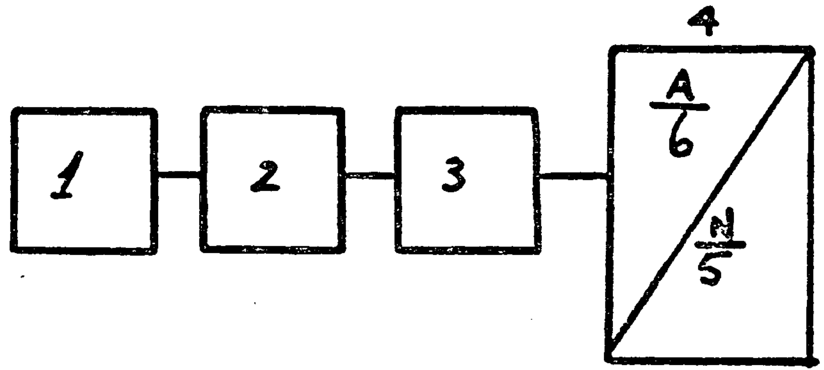
8.0 Reliability

The reliability goal for the Standard Communication Satellite Attitude Control Subsystem is 0.998 for five years of orbit life. This goal is attained in the design presented in this engineering memorandum as shown in Fig. 6.

SYSTEM



- 4 IDENTICAL MODULES
- ALL NEEDED FOR MISSION
- DUTY CYCLE OVER 5 YRS (43800 HRS) LESS THAN 0.1%
- SYSTEM ALLOCATED RELIABILITY R = 0.998.



INDIVIDUAL MODULE

1. TANKAGE & FILL VALVE
2. PROPELLANT FILL VALVE
3. ANCILLARY HARDWARE
4. THRUSTERS. BINOMIALLY REDUNDANT. 6 AVAILABLE, REQUIRED, 5.

Fig. 6a Standard Communication Satellite ACS Reliability Block Diagram (Sheet 1 of 2)

- Allocated Reliability R = .998/5 years
- $R_{sys} = (R_{M1})(R_{M2})(R_{M3})(R_{M4})$. $RM_{1,\ 2,\ 3,\ 4}$ = R per module
- $R_{module} = \left[R \text{ series elements} \right] \left[R \text{ redundant elements} \right]$

  $$\left[R \text{ series} = e^{-\Sigma\lambda t} \right] \left[R_{\text{redundant}} = \sum_{r=m}^{n} \binom{N}{r} p^{r} (1-p)^{n-5} \right]$$

- <u>Failure Rates</u>:

		$\lambda\ (10^{-6})$
	Pressure Xdcr	4.0
	Fill Valve	2.5
	Tankage	1.5
Series	Gas Fill Valve	3.0
Binomial	Series Valves	60.0 (12 @ 5.0 ea)
	Thrusters	8.0 (6 @ 1.33 ea)

- $R_{series} = e^{-\Sigma\lambda t} = e^{-11(10^{-6})\ 43.8} = .999958$

  $R_{red'nt} = .999965.$
- $R_{module} = .99992.$ Goal per module .9995.
- System Goal R = 0.998. $R_{\text{predicted}} = .99968.$

Fig. 6b Standard Communication Satellite ACS Reliability Block Diagram (Sheet 2 of 2)

PE-126

GENERAL DESCRIPTION

STANDARD U.S. DOMESTIC

COMMUNICATION SATELLITE

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: GENERAL DESCRIPTION - STANDARD U.S. DOMESTIC COMMUNICATION SATELLITE	EM NO: PE-126 REF: DATE: 24 December 1971
AUTHORS: F. C. Bolton — Prepared under cognizance of: Payload Integration, Orgn. 69-02 Space Systems Division	APPROVAL: F.C. Bolton ENGINEERING Wayne Miller

39 pages

PRELIMINARY

Table of Contents

Lockheed Missiles & Space Company

EM NO: PE-126
DATE: 24 Dec 1971

1.0 General

The standard United States Domestic Communication Satellite (Standard COMSAT) described in this Engineering Memo is to be placed into geosynchronous orbit by the Space Shuttle and Space Tug. The standard COMSAT and the Space Tug are mated at the launch base and installed in the cargo bay of the Space Shuttle, which is then launched into a low-earth parking orbit. The COMSAT is designed to be checked-out in the Shuttle and repaired, if necessary, by the replacement of equipment modules prior to being transported to geosynchronous orbit by the Space Tug. The COMSAT may be recovered from its operational orbit and returned to earth for repair, refurbishment and reuse.

The system requirements for the standard U.S. Domestic Communication Satellite are as follows:

Mission

To relay between ground stations in the 48 contiguous states the following types of signals:

FMD-FM Voice Transmissions
FM Monochrome or Color TV
Digital Transmissions such as PSK (bi-phase, quadri-phase, etc.)

Number and Location of Satellites **(Coverage Limited to United States, excl. Alaska and Hawaii)**

Primary satellite in geosynchronous orbit at 114° West Longitude
Back-up satellite in geosynchronous orbit at 119° West Longitude
Spare satellite, launch-ready on the ground

Communication System (Mission Equipment)

24 transponders:

Receive frequencies: 5.925 to 6.425 GHz
Transmit frequencies: 3.7 to 4.2 GHz

Antennas:

Separate, redundant receive and transmit antennas providing coverage of the 48 contiguous states.

Reliability:

20 out of 24 transponders to survive 5 years.
R = 0.954 for 5 years.

Stabilization, Attitude Control, & Station Keeping

Attitude Stability:

Approx. ± 0.2 degrees (compatible with antenna pointing accuracy requirements)

Station Keeping:

± 0.10 degree North-South
132 to 168 ft/sec/year

± 0.10 degree East-West
5 ft/sec/year max.

Reliability:

Stabilization & Control Subsystem:
R = 0.933 for 5 years

Attitude Control Subsystem:
R = 0.998 for 5 years

Electrical Power

- Single Axis tracking solar array -
- EOL power approx. 1750 watts
- Reliability: R = 0.945 for 5 years

Communications, Data Processing, and Instrumentation

- Tracking beacon and antenna
- Telemetry - Omni antenna
- Command - Omni antenna
- Data Processing and Computation
- Instrumentation
- Reliability: R = 0.937 for 5 years

Thermal Control

Passive control techniques augmented by heaters.

Overall Satellite Reliability

R = 0.75 for 5 years
No single point failure will disable the satellite.

Space Tug Performance

Reusable Tug:

Payload to Syneq Orbit	7384 lb **(Placement only)**
Payload from Syneq Orbit	4471 lb **(Retrieval only)**

2.0 Standard COMSAT Configuration

The general configuration of the standard COMSAT in flight is shown in Fig. 1. The direction of flight is eastward and the single-axis-tracking solar power panels are extended to the north and south of the spacecraft.

The location of equipment modules is shown in Fig. 2. The modules are designed to be readily removed and replaced in low-earth orbit by a Space Shuttle crewman or teleoperator. Replacement of modules in geosynchronous orbit by teleoperator would also be possible if a Space Tug/Teleoperator system is developed.

The modules in locations B-1, B-3, D-1, and D-3 are protected from solar radiation by hinged doors. The exposed surfaces of the remaining modules will be protected by appropriate surface finishes and insulation as determined by analysis of thermal control requirements.

The designations of the modules and the equipment contained in each module are listed in Fig. 3.

The weight summary for the standard Communication Satellite is presented in Fig. 4. Figure 5 presents the mass properties of the satellite.

3.0 Standard COMSAT/Tug/Shuttle Interfaces and Launch Concept

Figure 6 shows the standard Communication Satellite and Space Tug supported in the Space Shuttle cargo bay and elevated above the bay prior to separation from the Shuttle.

The Tug is provided with four external pins, two on each of the forward and aft rings. These pins engage four mating sockets (cones) which are mounted on a lightweight cradle assembly. The cradle assembly comprises fore and aft partial rings and two fore/aft spreader bars. In the stowed position, the four corners of the cradle assembly are latched down to four independent support fittings attached directly to Shuttle cargo bay bulkheads. The fore-aft loads are sustained by the two forward fittings.

Four synchronized extendable tape booms, one at each corner of the cradle assembly are used to elevate the Tug and the attached COMSAT out of the cargo bay.

Two pins at the forward end of the COMSAT engage hold-down fittings mounted on L/R trusses which in turn are bolted to Shuttle cargo bay bulkheads. All fore-aft loads are sustained by the forward mount. The COMSAT docking ring is engaged with the mating ring (two probes and two drogue funnels) on the forward end of the Tug. To prevent Shuttle loads from being applied to the COMSAT, spring devices have been provided in the four "legs" that connect the docking ring to the COMSAT. These devices will provide load attenuation in both compression and tension. Slots are provided in all pin mountings on one side to compensate for changes in lateral dimension of the Shuttle bulkheads during temperature changes and flight loading.

A single umbilical provides hardline connections between the Shuttle and the COMSAT. In-orbit checkout of the COMSAT is accomplished through the umbilical, which is released by remote control prior to separation of the Tug and COMSAT from the Shuttle.

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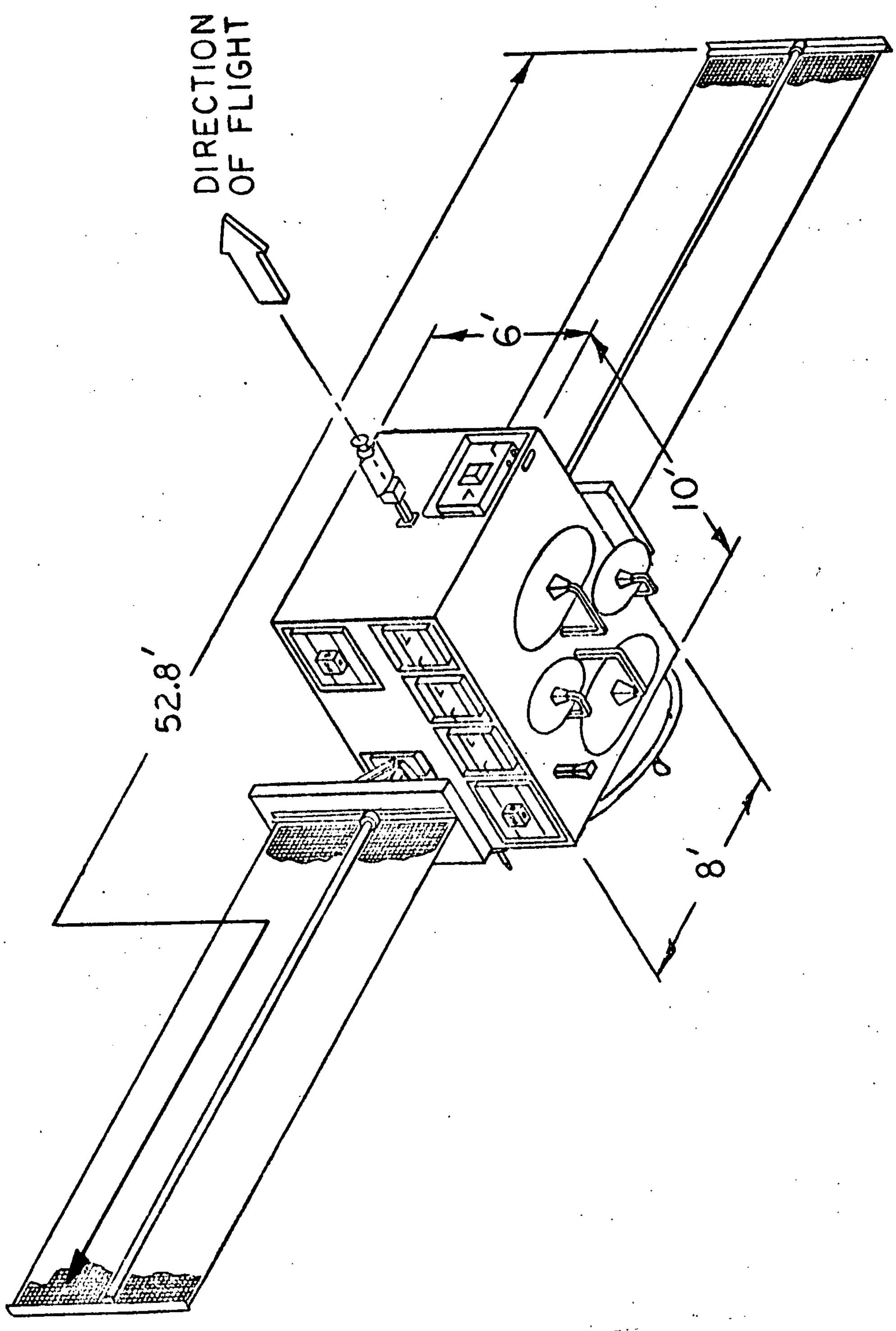

Fig. 1 General Configuration - Standard Communication Satellite

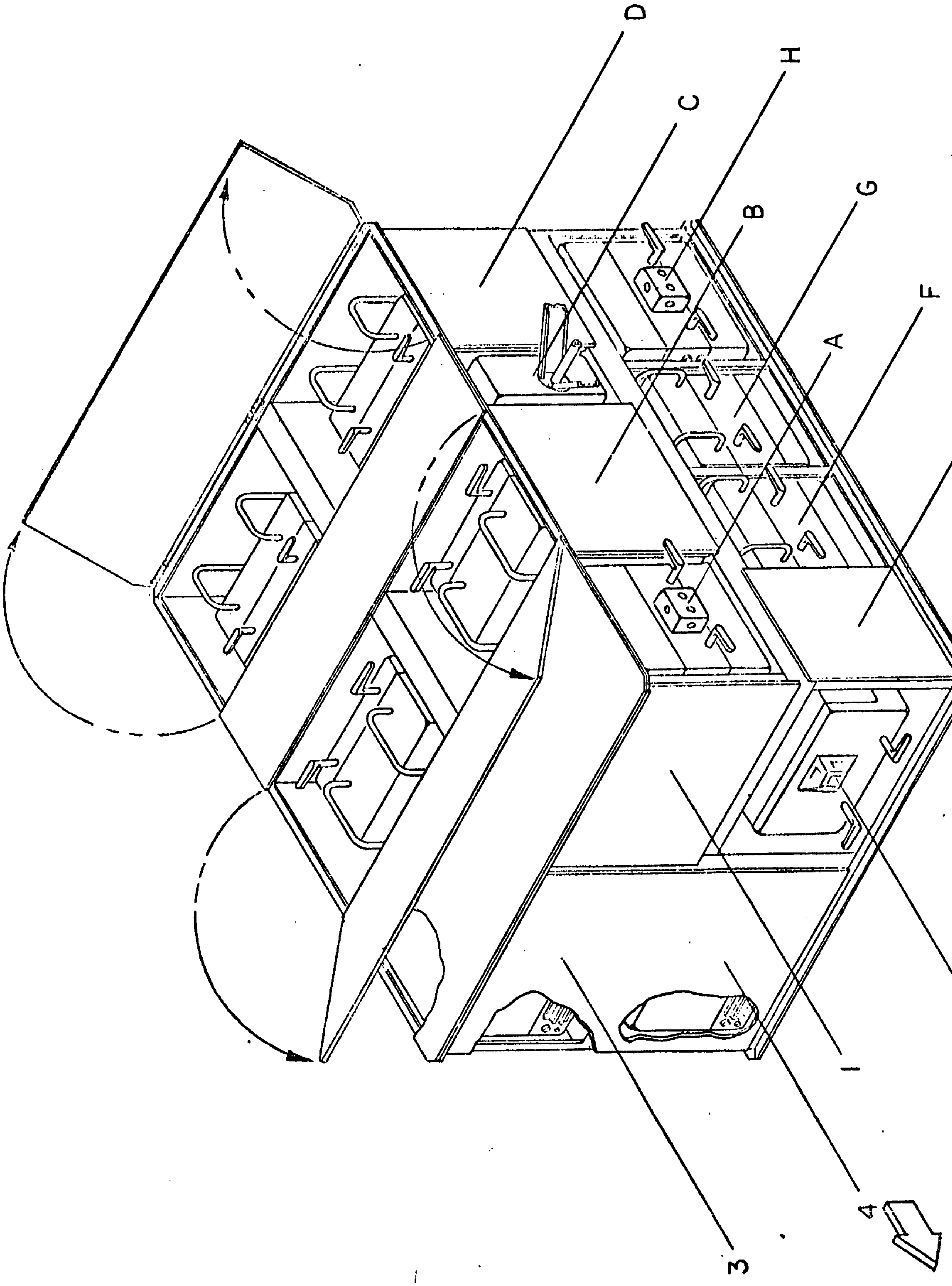

Fig. 2a Equipment Locations - Standard Communication Satellite (Sheet 1 of 2)

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Spacecraft Subsystem Modules	
A-1	Attitude Control Module No. 1
A-3	Attitude Control Module No. 2
B-1	Battery Module No. 1
B-2	Battery Module No. 2
C-1	Solar Array Drive Module
D-1	Power Distribution Module
D-3	CDPI Module
E-2	S&C Sensing Module
E-4	Momentum Wheel Module
H-2	Attitude Control Module No. 3
H-4	Attitude Control Module No. 4

Mission Equipment	
F-2	Transponder Module No. 1
F-4	Transponder Module No. 2
G-2	Transponder Module No. 3
G-4	Transponder Module No. 4

Fig. 2b Equipment Locations-Standard Communication Satellite (2 of 2)

Subsystem	Module	Equipment in Module	Module Weight
Stabilization and Control	Momentum Wheel No. 1	• Momentum Wheel Assembly • Wheel Drive Electronics • Safety Shield • Wire harnesses • Base & cover	Basic 198 lb 15% contingency + 30 Total ea 228 lb
Stabilization and Control	Sensing and Electronics High Altitude No. 1	• Horizon Sensor (2) • Rate Gyro Package • Sun Aspect Sensor (5) • Secondary Control Electronics (5) • Wire harnesses • Base & covers	Basic 58 lb 15% contingency + 9 Total ea 67 lb
Communication, Data Processing, Instrumentation (CDPI)	CDPI No. 1	• Command Receivers (2) • Interface Unit • Digital Computer • Memory Unit • Instrument Section • Decoder, Failure • Decoder, Command • Hybrid Couplers (3) • Input Filter • Transmitter, Telemetry (2) • Transmitter, Beacon (2) • Subcarrier Modulator & Summer (2) • Range Detector, Test & Switch (2) • Base & Covers • Cables & Wiring	Basic 58 lb 15% contingency + 9 Total ea 67 lb
Electrical Power	Battery (2 req'd) No. 1 No. 2	• NiCd Battery (2) - 40 Amp. hr. • Charge Controller (2) • Module Base/Cover • Wire harnesses	Basic 213 lb 10% contingency + 21 Total ea 234 lb

Fig. 3a Standard Communication Satellite Equipment Modules (Sheet 1 of 3)

Subsystem	Module	Equipment in Module	Module Weight
Electrical Power	Power Distribution No. 1	• Pwr Distribution • DC to DC Regulator • Solar Array Regulator • Wire harnesses • Module Base/Cover	Basic 119 lb 15% contingency + 18 Total ea 137 lb
Electrical Power	Solar Array Drive Module No. 1	• Drive Motor Ass'y. • Electronics • Slip Ring Assembly • Module Base/Cover • Wire harnesses	Basic 66.0 lb 15% contingency + 10.0 Total ea 76.0 lb
Electrical Power	Solar Array Power Module (2 req'd) No. 1 No. 2	• Solar Array - flexible fold • Extendable Boom Assembly • Module Base/Cover • Wire harnesses • Array Container	Basic 141.0 lb 15% contingency + 21.0 Total ea 162.0 lb
Attitude Control	Attitude Control Module (4 req'd) No. 1 No. 2 No. 3 No. 4	• Storage Tank • Fill Valves • Thruster Assembly • Pressure Transducers • Temperature Sensors • Filter • Plumbing • Base & Covers • Electronics & Wire • Wire harnesses	Basic 257 lb Contingency (15% of dry weight) + 13 Total ea 270 lb

Fig. 3b Standard Communication Satellite Equipment Modules (Sheet 2 of 3)

Subsystem	Module	Equipment in Module	Module Weight
Mission Equipment	All Channel Receiving and Odd Channel Transmitting (2 req'd) Xmit Channels No. 1 1,5,9,13,17,21 No. 2 3,7,11,15,19, 23 Rcv All Channels	• Filter, Preamplifier Input 1 req'd • RF Switch 15 " • Tunnel Diode Amplifier(TDA) 18 " • Frequency Down Converter 2 " • Redundant Local Oscillator 2 " • Isolator 2 " • Hybrid Coupler 1 " • Multiplexer, Receiver 1 " • TWT, Driver 12 " • Attenuator 12 " • Hybrid Coupler 6 " • Filter 12 " • Isolator 18 " • TWT, Output 12 " • Multiplexer, Transmit 1 " • Filter, Output 1 " • Module Base & Cover 1 " • Wire harnesses	Basic 201 lb 15% contingency + 30 Total ea 231 lb
Mission Equipment	Even Channel Transmitting (2 req'd) Xmit Channels No. 1 2,6,10,14, 18,22 No. 2 4,8,12,16, 20,24	• Hybrid Coupler 1 req'd • RF Switch 14 " • Tunnel Diode Amplifier 14 " • Multiplexer, Receiver 1 " • TWT, Driver 12 " • Attenuator 12 " • Hybrid Coupler 6 " • Filter 12 " • Isolator 18 " • TWT, Output 12 " • Multiplexer, Transmit 1 " • Filter, Output 1 " • Module Base & Cover 1 " • Wire harnesses	Basic 196 lb 15% contingency + 29 Total ea 225 lb

Fig. 3c Standard Communication Satellite Equipment Modules (Sheet 3 of 3)

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Subsystem	Contingency (%)	Weight (lb)
Structure & Mechanisms	15	904
Environmental Control	15	100
Stabilization & Control	15	305
Communications, Data, Proc., Instrumentation	15	67
Electrical Power	15	1,005
Attitude Control	15	388
Mission Equipment	15	912
Vehicle Electrical Harness	15	240
Antennae and Supports	20	54
	Dry Weight	3,975
Propellant - Hydrazine		684
Pressurization Gas - Nitrogen		8
	Total Weight	4,667 lb

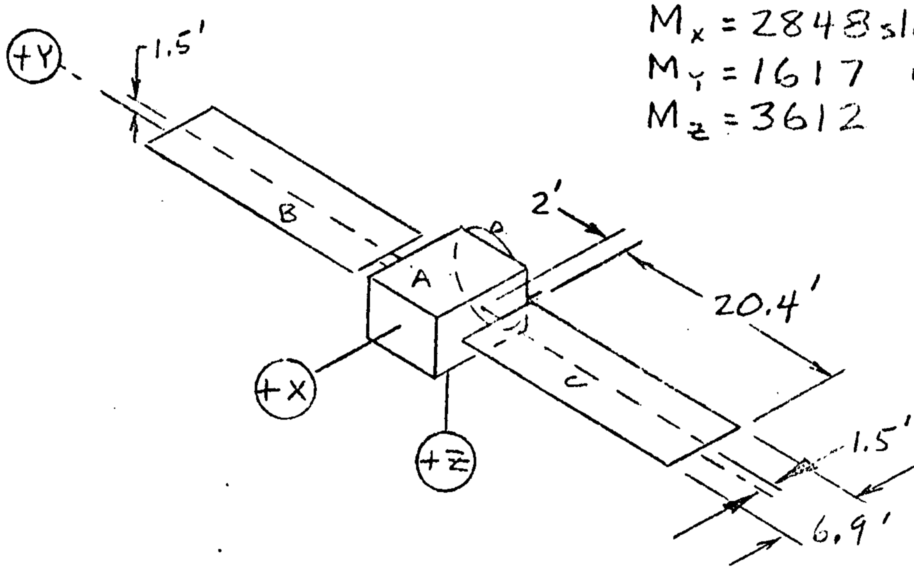
Fig. 4 Weight Summary - Standard Communication Satellite

$W_A = \cancel{4396}$ 4343 lb $W_B = W_C = \cancel{135.5}$ 162.0 lb

$W_{TOTAL} = 4667$ lb

CASE I:

$M_x = 2848$ slug-ft²
$M_y = 1617$ " "
$M_z = 3612$ " "



CASE II:

$M_x = 2879$ slug-ft²
$M_y = 1617$ " "
$M_z = 3581$ " "

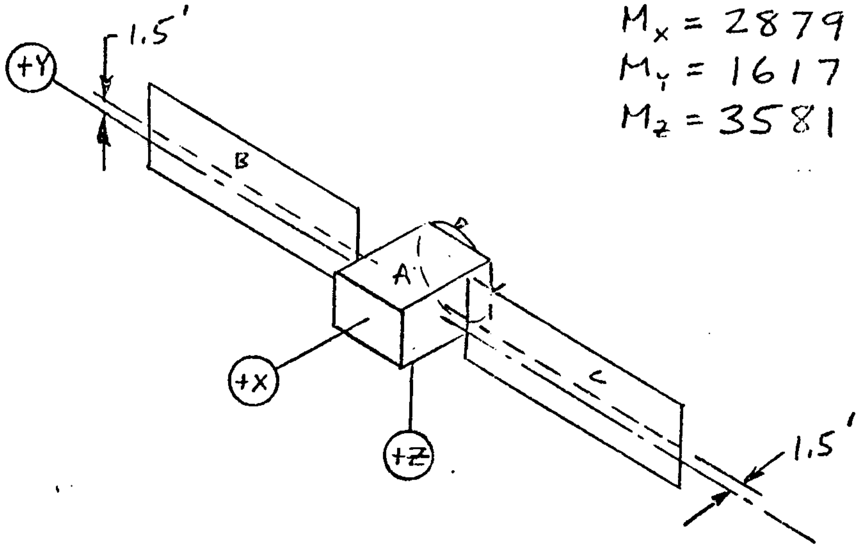


Fig. 5a Mass Properties - Standard Communication Satellite
(Sheet 1 of 2)

Lockheed Missiles & Space Company

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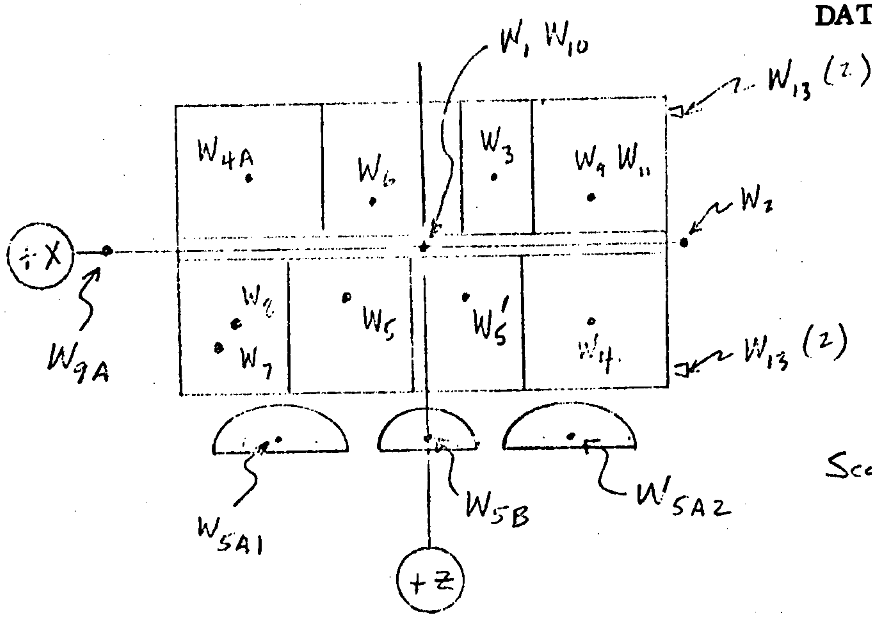

Scale $\frac{1}{4}'' = 1'$

	DESCRIPTION	W	X	Z	Wx	Wz
1	Structure	800	0	0	0	0
2	Docking Ring	104	-5.5	0	-572	0
3	Solar Array	400	-1.5	1.5	-600	600
4	Attitude Control	540	-3.5	1.5	-1890	810
4A	" "	540	3.5	-1.5	1890	-810
5	Mission Equip	462	1.7	1.0	693	462
5'	" "	450	-0.8	1.0	-360	450
5A1	xmt Ant (+20%)	12	3.0	4.0	36	48
5A2	xmt Ant (+20%)	12	-3.0	4.0	-36	48
5B	Rcvg Ant (")	20	0	4.0	0	80
6	Battery	468	1.0	-1.0	468	-468
7	Attitude Sensing	67	4.2	2.0	281.4	134
8	Wheel, Momentum	228	3.9	-1.5	889.2	-342
9	CDPI	67	-3.5	-1.0	-234.5	-67
9A	" Antenna (15%)	10	6.5	0	65	0
10	Environment Cont	100	0	0	0	0
11	Pwr Distr.	137	-3.5	-1.0	-479.5	-137
12	Elec Harness	240	0	0	0	0
13	Docking Refl	10	-5.0	0	-50.0	0
	Totals	4667			100.6	808

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$$\bar{x} = \frac{\Sigma Wx}{\Sigma W} = \frac{100.6}{4667} = .0216' \qquad \bar{z} = \frac{\Sigma Wz}{\Sigma W} = \frac{808}{4667} = .1731'$$

Fig. 5b Mass Properties - Standard Communication Satellite
(Sheet 2 of 2)

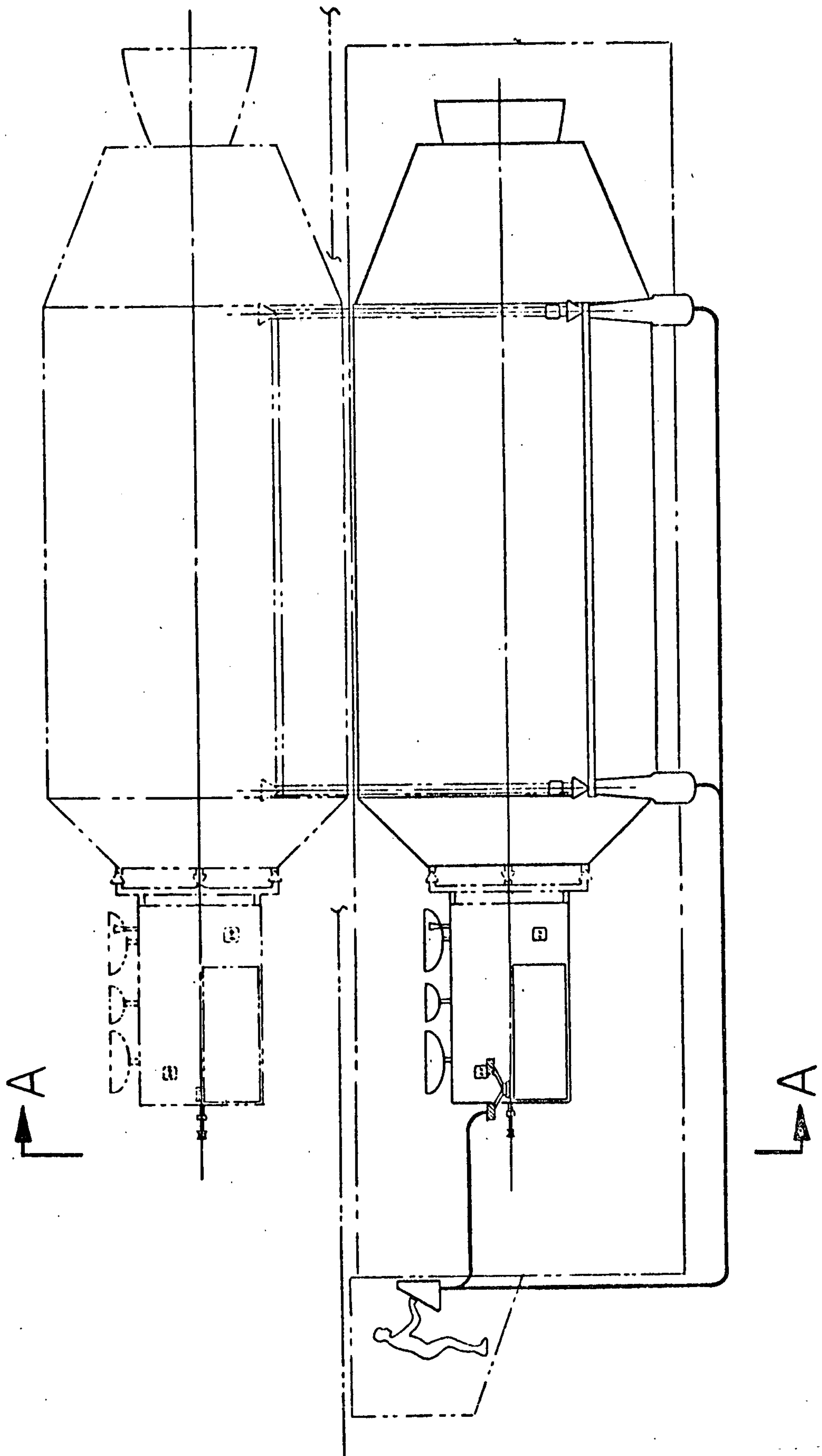

Fig. 6a Standard COMSAT & Space Tug in Shuttle Cargo Bay
(Sheet 1 of 2)

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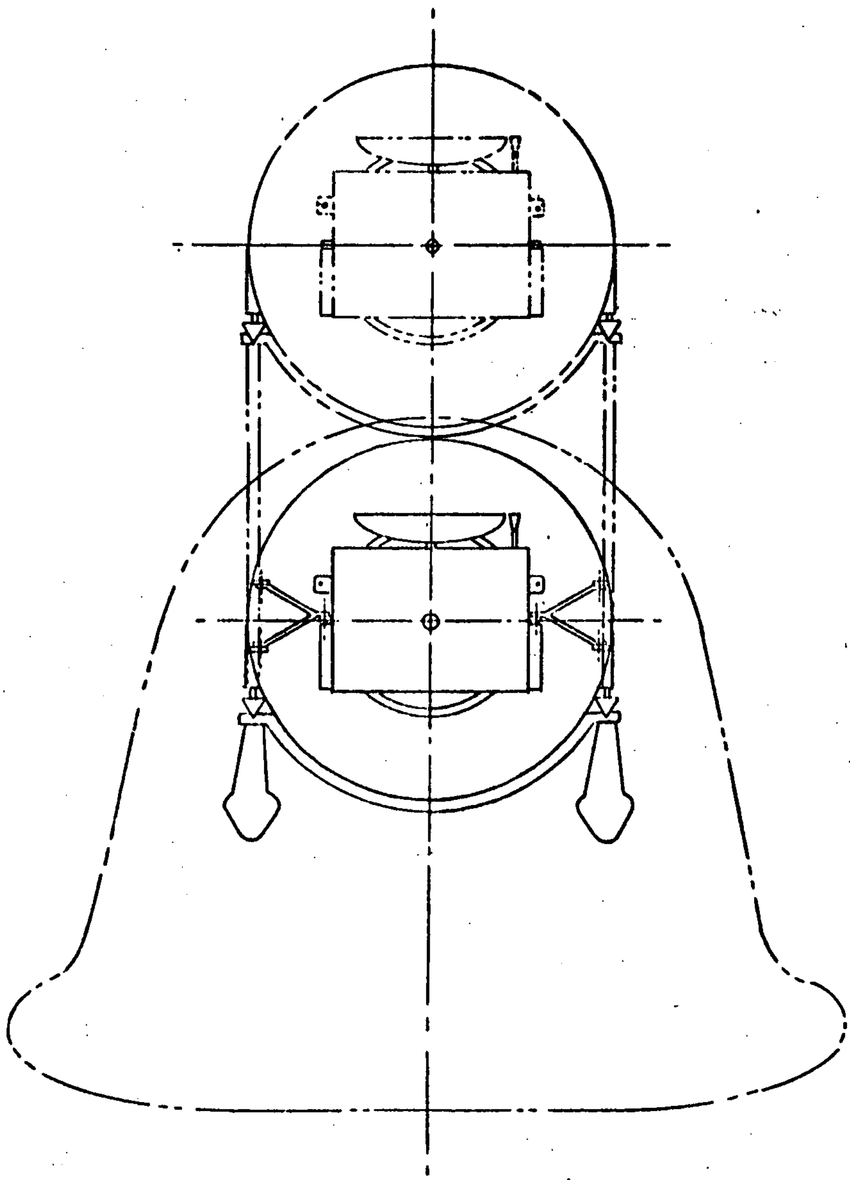

Fig. 6b Standard COMSAT & Space Tug in Shuttle Cargo Bay
(Sheet 2 of 2)

4.0 Description of Standard Communication Satellite

Separate Engineering Memoranda have been written to describe the major subsystems of the standard COMSAT. They are as follows:

PE-122	Standard Communication Satellite Stabilization and Control Subsystem
PE-123	Standard Communication Satellite Communication, Data Processing and Instrumentation Subsystem
PE-124	Standard Communication Satellite Electrical Power Subsystem
PE-125	Standard Communication Satellite Attitude Control Subsystem

These subsystem descriptions will be summarized briefly in subsequent paragraphs.

4.1 Structures Subsystem

The primary structure of the standard COMSAT is shown in Fig. 7. The most significant feature of the structure is its geometric simplicity and symmetry, which minimize the number of different parts required and, consequently, the design and fabrication costs. The entire structure is made up of commercial grade 6061 aluminum sheet and extrusions which are inexpensive and easy to fabricate. 6061 aluminum combines good formability, machinability and weldability with medium strength. All structural elements are sized to have high factors of safety to minimize structural analysis and static load testing.

The primary load bearing structure is the cruciform beam clearly shown in Fig. 7. The beam is built from aluminum sheet and extrusions. The box-like flanges are approximately 6 inches thick; and the electrical harness that interconnects the subsystem and mission equipment modules is largely contained within them. This beam is the baseline for the alignment of equipment.

Two fittings attached to two opposite flanges of the beam as shown support the COMSAT during ground handling and in the Shuttle.

The other elements of the spacecraft structure provide support for the equipment modules. The external skins with appropriate surface finishes and insulation also contribute to the passive thermal control of the COMSAT.

4.2 Mission Equipment

The design of the Mission Equipment (Communication equipment) is not within the scope of this study. However, it was necessary to prepare a brief description of typical Communication Satellite Mission Equipment, and to identify its support requirements before proceeding with the design of the spacecraft subsystems.

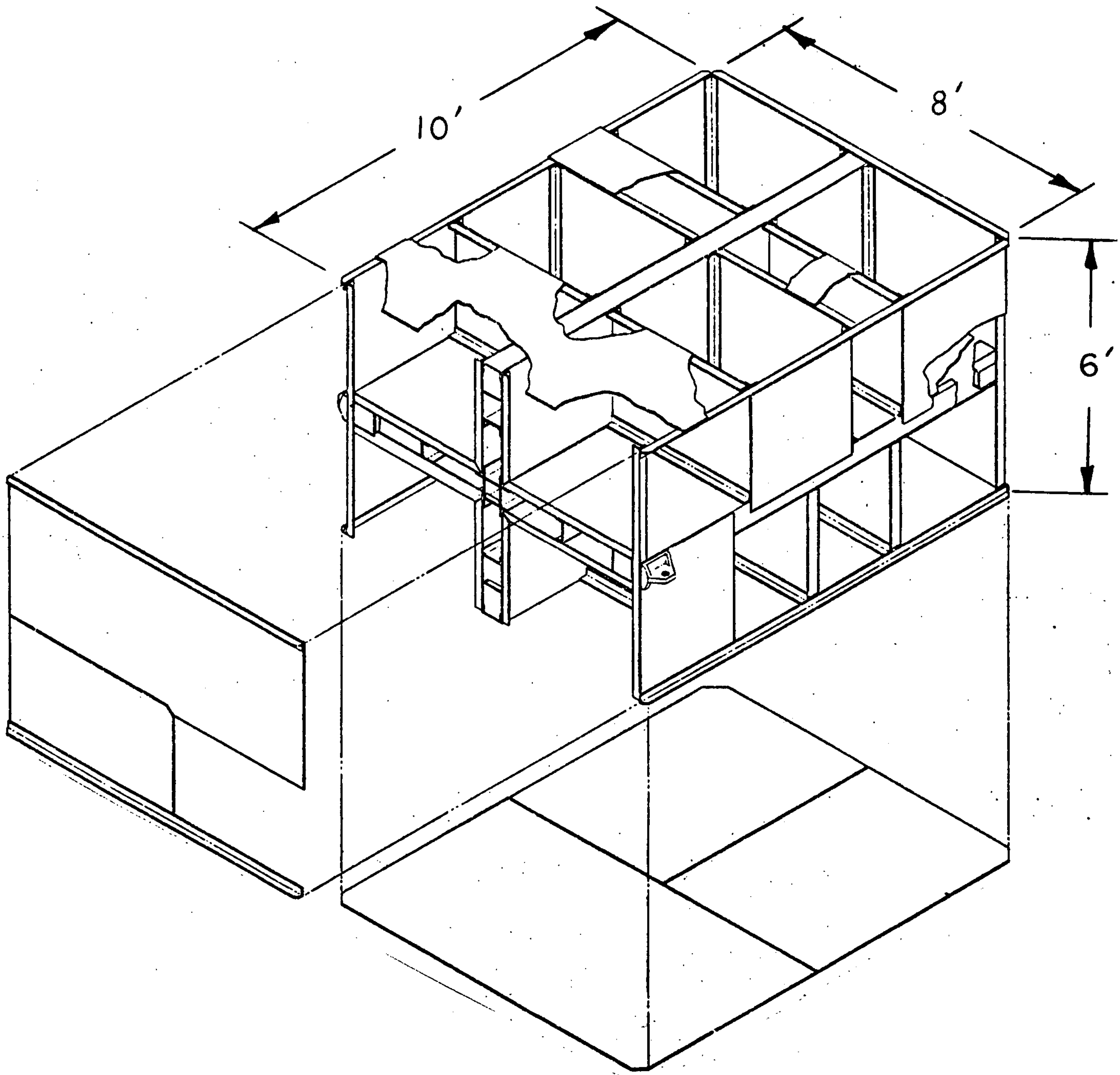

Fig. 7 Primary Structure - Standard Communication Satellite

The defined Mission Equipment is based upon the following assumptions:

• Coverage	contiguous U.S.
• Antenna Pointing Accuracy	$\pm 0.16^{\circ}$
• Number of Channels	24
• Channel Bandwidth	36 MHz
• Carrier Separation	40 MHz
• Receive Frequency Range	5.925 - 6.425 GHz
• Transmit Frequency Range	3.7 - 4.2 GHz
• Beam Edge Receive G/T	-4.5db/$^{\circ}$K (1595°K attainable)
• Beam Center EIRP Xmit	37.5 dbw
• Beam Edge EIRP Xmit	34.5 dbw
• Antennas (shaped beam)	
Receive -	20" x 40" Elliptical 3.5 x 7.0 degree - 3 db beamwidth 30.5 db peak gain, relative to isotropic
Transmit -	30" x 60" elliptical 3.5 x 7.0 degree -3 db beamwidth 30.5 db peak gain, relative to isotropic

The performance budget summary for the Mission Equipment at the edge of the coverage area and low earth elevation angle, assuming station EIRP at 85 dbw (1.3 Kw at ground antenna feed - 32 ft. dish assumed) is as follows:

<u>Uplink</u>

EIRP	85 dbw
Satellite GK	4.5 db/$^{\circ}$K
Path Loss	200.1 db
C/N in 36 MHz	
Noise Bandwidth	33.5 db

<u>Downlink</u>

EIRP	34.5 dbw
Earth Station G/T	33 db/$^{\circ}$K
Path Loss	196.6 db
Receive Equivalent	
Noise Bandwidth	36 db
C/N	24 db
System C/N	23.5 db

Analysis yields the following margins.

Telephone	13.9 db
TV	13.9 db

A block diagram of the Mission Equipment is given in Fig. 8. Single step down conversion is used, rather than conversion and multiplexing at IF frequencies, to eliminate some of the problems that older technique affords. This approach is most modern and within the state-of-the-art; it is therefore used here.

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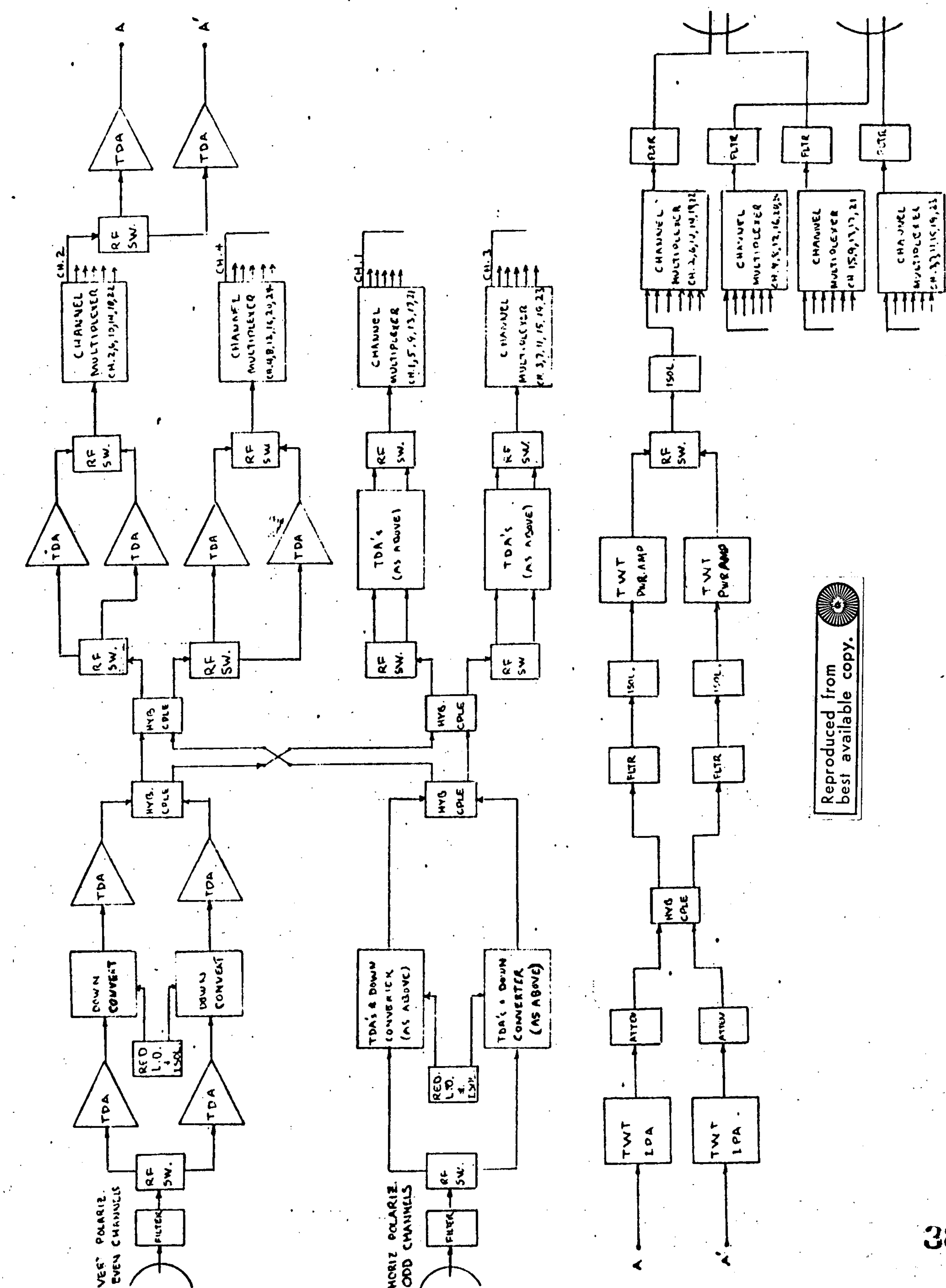

Fig. 8 Standard Communication Satellite Mission Equipment

4.3 Stabilization and Control Subsystem

The Standard COMSAT Stabilization and Control Subsystem has the following functions:

- To control attitude transients due to Space Tug separation
- To meet COMSAT pointing requirements (0.1 deg roll and pitch, 0.5 deg yaw) for five years
- To stabilize and control COMSAT attitude during East-West and North-South station-keeping maneuvers and during space tug docking maneuvers
- To reorient COMSAT to the orbit reference attitude from any attitude following loss of reference for tumbling rates up to 4 deg/sec.
- To point the COMSAT spacecraft to the sun with near-zero rates following primary system failure.

The standard COMSAT Stabilization and Control Subsystem (Fig. 9) 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 gimballed 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. During North-South Stationkeeping, digital solar aspect sensors provide the yaw reference.

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

The General Purpose Digital Computer in the CDPI Subsystem provides timing, sequencing, and logical computations for the S&C subsystem. It accepts commands for real-time execution and for storage from the communications section of the CDPI. The processor section of the CDPI converts sensor signals and command messages into control logic and actuation signals.

The double-gimballed momentum wheel control system is a new concept now being applied to three-axis, long-life satellite attitude control. The system uses a 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 stabilization system requires no yaw sensor to perform its "normal" function, i.e., earth pointing.

The equipment of the Stabilization and Control Subsystem is packaged in two modules: a Sensing and Electronics Module (Fig. 10) and a Momentum Wheel Module (Fig. 11). These modules are designed to be removed and replaced in orbit by a Shuttle crewman or a teleoperator.

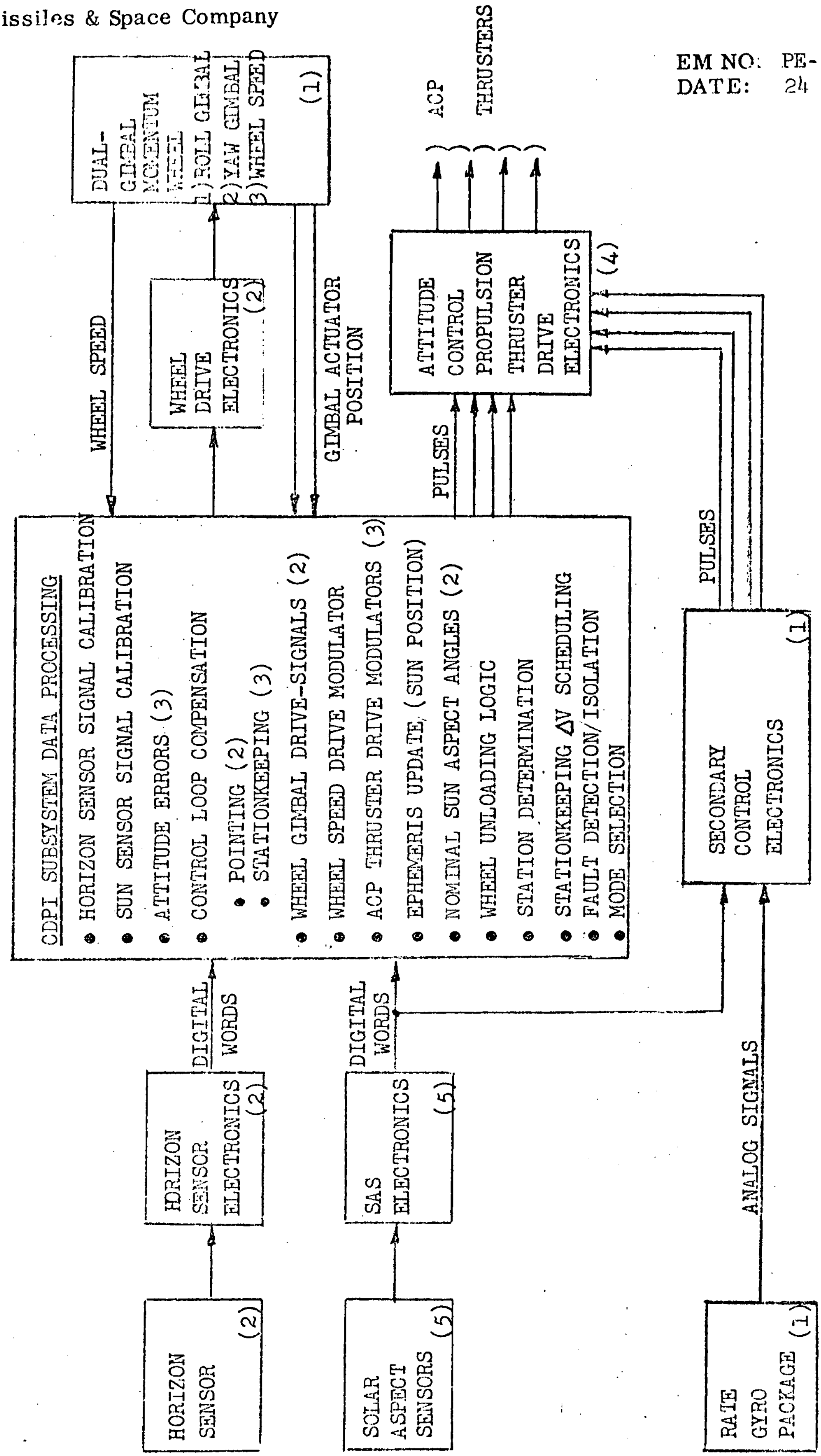

Fig. 9 COMSAT Standard Spacecraft Stabilization & Control Subsystem

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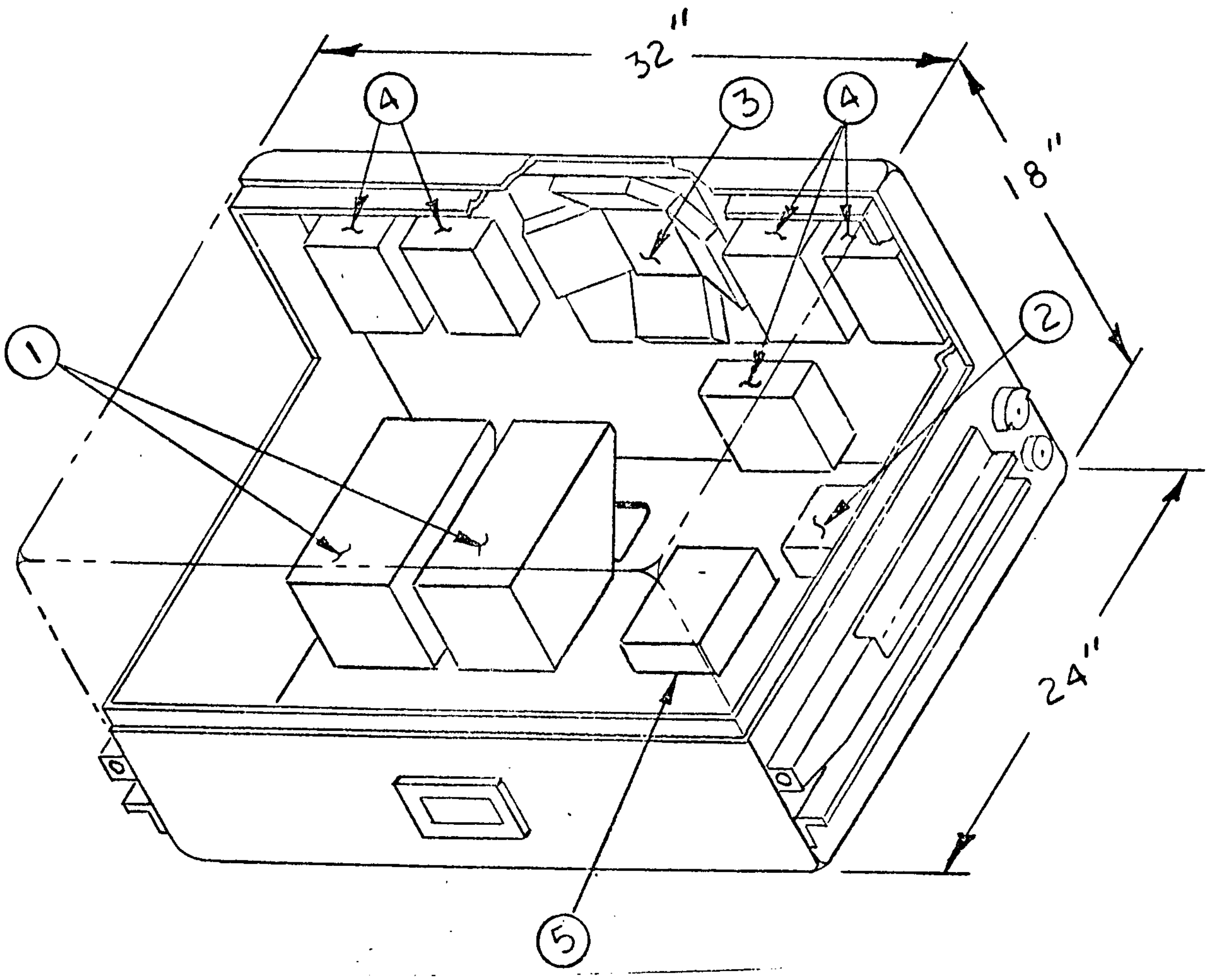

	Equipment	Quantity		Weight (lb)
(1)	Horizon Sensor	2		12
(2)	Rate Gyro Package	1		3
(3)	Sun Aspect Sensor	5		15
(4)	S.A.S. Electronics	5		
(5)	Secondary Control Electronics	1		5
	Internal Wire Harness	-		3
	Base & Cover	-		20
			Subtotal	58
			15% Contingency	8.7
			Total	66.7

Fig. 10 Sensing and Electronics Module - Stabilization & Control Subsystem

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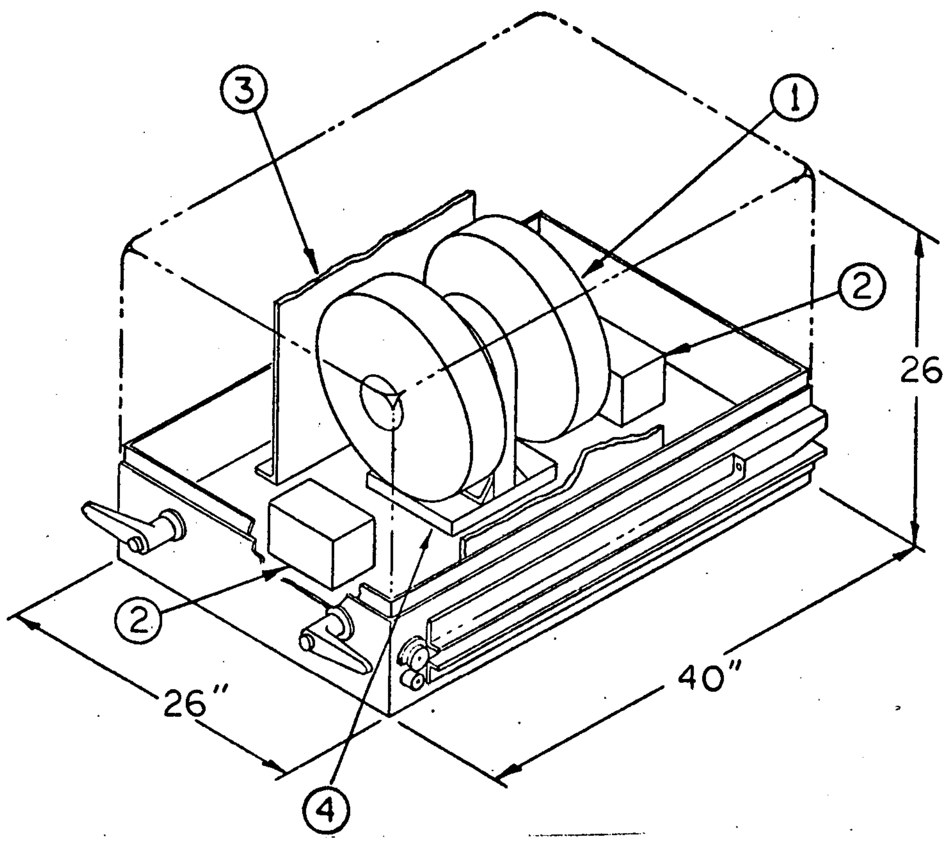

	Equipment	Quantity		Weight (lb)
(1)	Momentum Wheel Assembly	1		79
(2)	Electronics	2		6
(3)	Shielding	1		50
(4)	Mount	1		8
	Base & Cover	1		50
	Internal Wire Harness	Set		5
			Subtotal	198
			15% contingency	30
			Total	228

Fig. 11 Momentum Wheel Module - Stabilization & Control Subsystem

4.4 Communication, Data Processing and Instrumentation Subsystem

The CDPI subsystem provides the command, ranging and telemetry functions for the Standard COMSAT, and serves as the major interface entity between the various spacecraft subsystems, e.g., Stabilization and Control and Attitude Control. It also performs the computational and on-board decision-making processes for the spacecraft and includes the status monitoring instrumentation for indicating spacecraft condition or malfunctions. A block diagram of the CDPI is shown in Fig. 12.

The CDPI must meet high reliability requirements because of two main reasons:

- Shuttle visits for replacement or refurbishment would be infrequent - five years between visits is assumed **for complete refurbishment and additional flights for repair or partial refurb at 2.5 year points.**
- Degradation or loss of the communication services provided by the spacecraft and its mission equipment have significant cost penalties.

A block diagram of the CDPI communication section is shown in Fig. 13. The design is relatively straightforward. A common antenna feeds redundant superheterodyne receivers via a filter and hybrid coupler. Input filtering is employed to minimize receiver spurious responses as well as aid in isolation of the mixer input reference frequency. The TDA amplifier reduces the receiver noise figure and contributes to front end skirt selectivity and isolation.

The IF amplifier is nominally two stages, operating at an IF center frequency of approximately 45 MHz. The first stage is 10 MHz wide and the second about 1.0 MHz wide to obtain the necessary receiver selectivity. AGC is incorporated to provide a relatively constant input signal to the demodulator limiter.

The demodulator is in two parts - an amplitude detector for the AGC and a standard Foster-Seeley type frequency discriminator to detect the command and range signals. The amplitude detector output for AGC is also used for signal presence information (carrier detection), a signal which may be employed for backup command switching.

Command/data and range information are separated, with the command tones processed in a decoder to form 1-0 bit patterns prior to routing to the computer via the Interface Unit of the CDPI. Range tones are used to modulate a carrier for retransmission to the ground station via a linear summing circuit. The same circuit is used to accept the subcarrier modulated signal which contains the telemetry/command verification information.

The transmitter consists of redundant varactor multipliers and transistor amplifiers which feed a common antenna via a hybrid coupler. The oscillator reference for the transmitter comes from the timing generator circuitry in the Interface Unit. The same circuits are used to provide the receiver mixer oscillator reference frequency and the telemetry subcarrier frequency.

In order to provide greater downlink redundancy, a separate beacon transmitter is provided feeding a horn antenna. The beacon signal may consist of a particular tone pattern, a ranging signal or, if desired, telemetry or command retransmissions.

Due to the availability of computer control, a 100 percent uplink and downlink duty cycle is not essential and the second receiver and transmitter may be placed in time-

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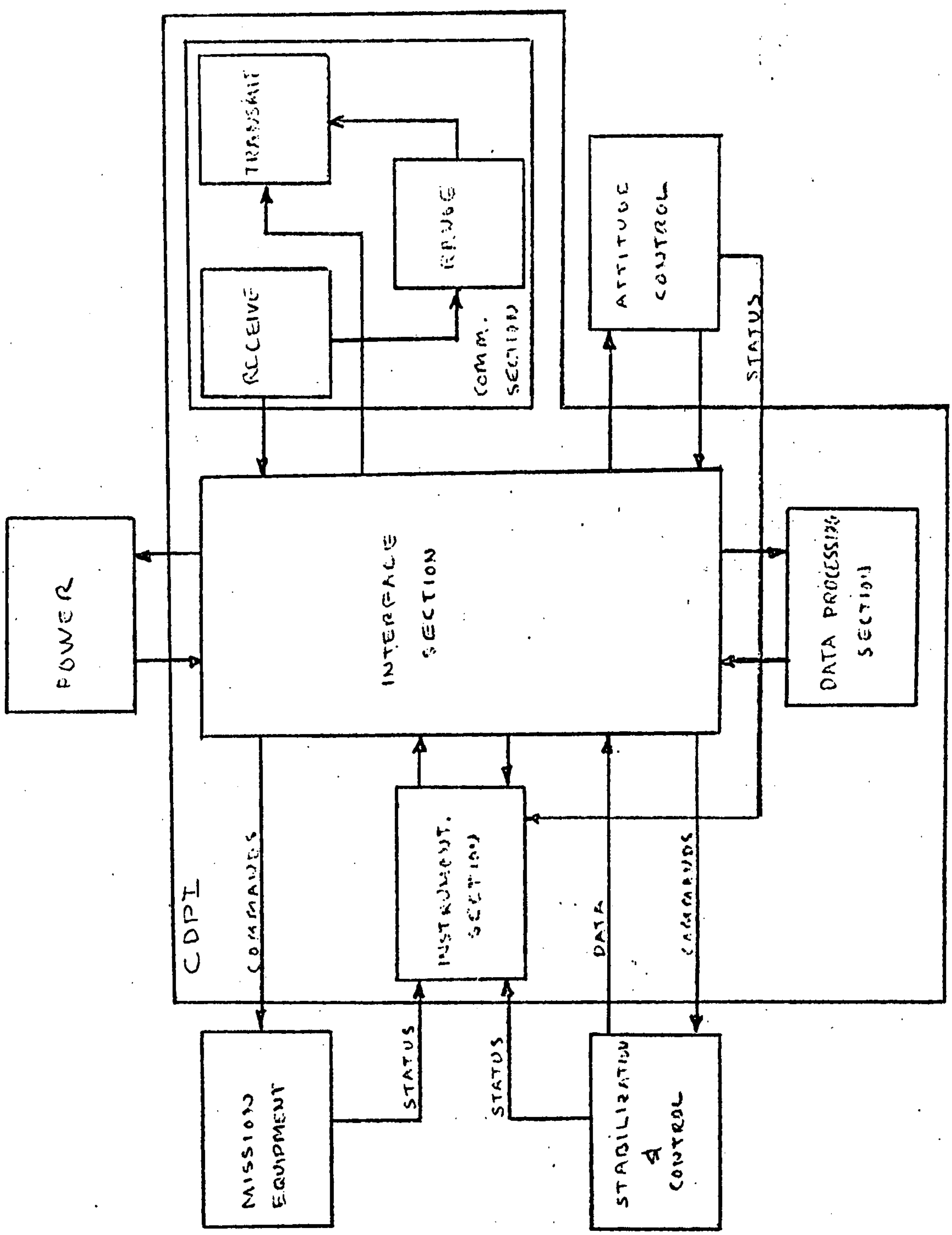

Fig. 12 Standard Communication Satellite - CDPI Block Diagram

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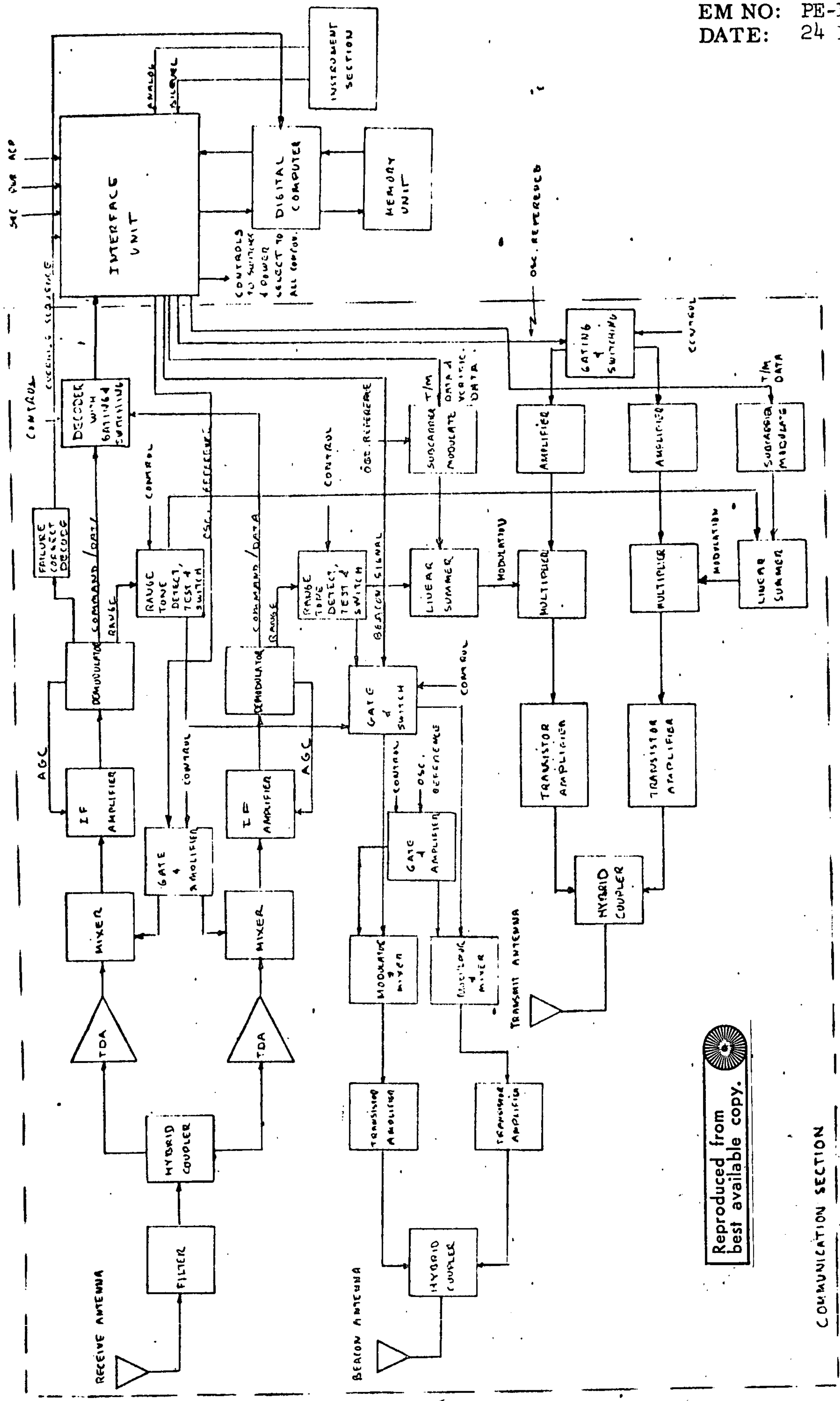

Fig. 13 Communication Section - CDPI Subsystem Standard Communication Satellite

shared standby mode. This means that power to the redundant transmitter and receiver components may be switched off until a failure is sensed via the instrumentation or fault-testing and isolation hardware and software. If a failure is recognized by interpretation of telemetry data or recognition of improper responses to the uplink commands, a failure override sequence may be activated which controls either power distribution to components or enables the redundant computer memory switchover.

A block diagram of the Interface Section for the CDPI is shown in Fig. 14. The function of the Interface Unit is to provide data routing and operational control paths between the various spacecraft subsystems and the data processing and communication sections of the CDPI. Data routing consists of the following:

- Accepting, sampling, subcommutating, multiplexing and converting analog instrumentation data for computer input.

- Accepting, sampling, formatting and delay of bilevel data for computer input.

- Accepting and delivery of S&C horizon and sun sensor output values to the computer.

- Relay of information to telemetry, AGE or Shuttle-based equipment.

The control processes are as follows:

- Accepts computer stored or verified uplink commands and issues the control discretes via discrete control sequencing logic. This logic operates mission equipment rf and other switches, governs power application via power control logic, operates gating circuits which trigger holding and shift registers in the Interface Unit, etc.

- Provides conditioned drive signals to the S&C momentum wheel torquers and gimbals and ACS thrusters.

The Interface Unit also provides the timing references for synchronizing and operating logic as well as Communication Section functions. This was done to take advantage of system commonalities and improve overall flexibility. The highest frequencies are nominally between 50-100 MHz, since frequency multipliers are used in the communication equipment. Most other frequencies utilized are less than 100 KHz.

A computer which meets the functional requirements of the Standard COMSAT is the Control Data Corporation 469 defined by Fig. 15. To insure that the computer has the required reliability for this application the following improvements are necessary:

- Utilization of high reliability parts
- 100% screening of parts
- Redundant memory
- Detailed software verification

The CDPI subsystem equipment is installed in a single module shown in Fig. 16.

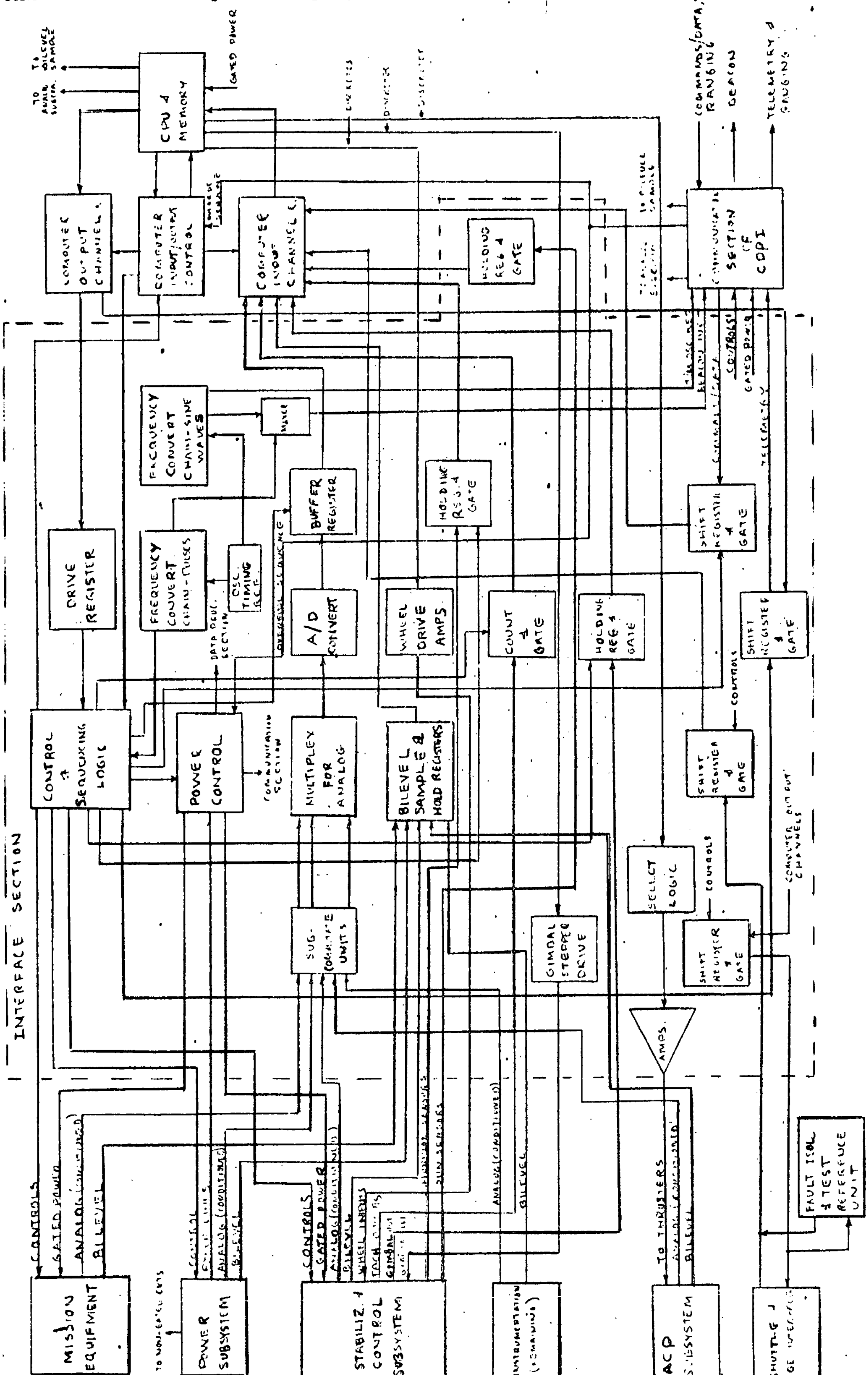

Fig. 14 Interface Section - CDPI Subsystem
Standard Communication Satellite

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TECHNICAL CHARACTERISTICS

eneral Data

eight: 2.5 lbs (8K)
4.0 lbs (16K)

Dimensions: 4.2"h x 4.13"w x 3.0"d (8K)
4.2"h x 4.13"w x 5.0"d (16K)

ower Consumption: ≈12 watts (8K)
≈13 watts (16K)

nput Voltages: ± 15 VDC, + 5 VDC, -3VDC

Circuits: High level PMOS/CMOS/IC devices

nvironment: Designed to MIL-E-5400

Cooling: None required (0°C to +50°C)

stimated Reliability:

entral Processor (LSI, 14 devices total)

ype: Binary, parallel, general purpose single address, plus file address

epertoire: 42 instructions (some double precision)

ord Length: 16 bits

egister Files: 16 Addressable 16-bit word files

nterrupts: 3 external levels, plus 1 direct execute.

rithmetic: Fractional, fixed point, two's complement. Hardware multiply and divide.
Typical execution times:
Add: 2.4 μsec
Double Precision Add: 3.6 μsec
Multiply: 10.4 μsec
Divide: 30.4 μsec

Memory

Type: Random access, word-organized, NDRO (electrically alterable) Plated Wire Memory (PWM)

Word Length: 16 bits

Capacity: 8K words NDRO expandable to 65K in 8K word increments

Read Cycle Time: 1.6 microseconds

Write Cycle Time: 2.4 microseconds

Access Time: 500 nanoseconds

Input/Output

1-16 bit parallel, party line, buss input
1-16 bit parallel, party line, buss output
4-bit address control lines
1-serial input channel
1-serial output channel
External clock input
69-90 KHz - parallel continuous word rate I/O
400 KHz - parallel burst word rate I/O
130 KHz - serial bit rate I/O

Optional I/O: Solid state keyboard and display suitable for navigational, checkout, and general purpose use can be integral to the computer. Also available are multiplexed A/D input channel(s), buffered and non-buffered peripheral device channel(s) on a Quote Special Equipment (QSE) basis.

Standard Interface: TTL

Software

Assembler, Simulator, Plus Library and Diagnostic Routines

Optional Support Equipment

Programmer's Console with integral CRT display and power supply.

P.W. Memory Loader with integral paper tape reader and power supply.

* Information furnished by Control Data Corp., Minneapolis, Minn.

Fig. 15 CDC-469 Computer*

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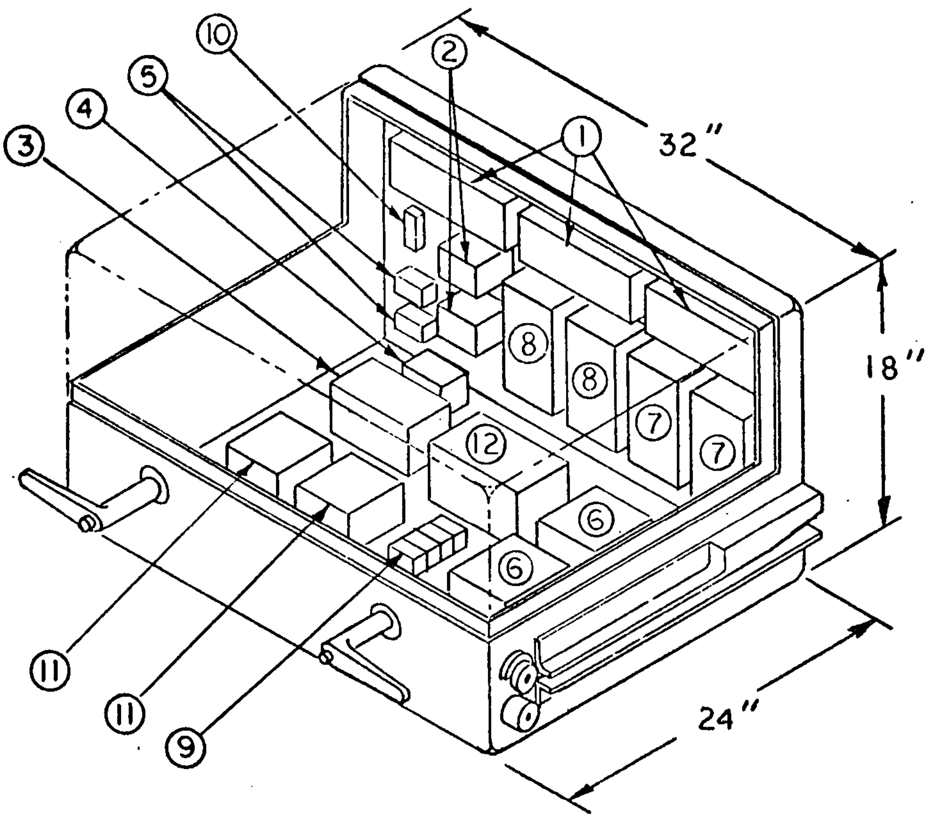
EM NO: PE-126
DATE: 24 Dec 1971



	Equipment	Quantity	Weight (lbs)
(1)	Hybrid Coupler	3	3.0
(2)	Receiver	2	6.0
(3)	Decoder, Command	1	4.5
(4)	Decoder, Failure Correct	1	0.2
(5)	Range Detect, Test & Switch	2	0.2
(6)	Modulator & Summer	2	0.4
(7)	Telemetry Transmitter	2	4.0
(8)	Beacon Transmitter	2	4.0
(9)	Amplifier, Gate & Switch	5	1.0
(10)	Filter, Input	1	0.2
(11)	Computer	2	4.0
(12)	Interface Unit	1	6.0
	Internal Wire Harness	AR	5.6
	Base and Cover	1	19.0
		Subtotal	58.1
		15% Contingency	8.7
		Total	66.8

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Fig. 16 CDPI Subsystem Module Standard Communication Satellite

4.5 Electrical Power Subsystem

The EPS is sized to deliver 1750 watts of power to the equipment load at the end of 5 years of orbit life. It consists of a sun-tracking solar array which is voltage controlled by a shunt regulator, 4 nickel cadmium batteries which are charged by a constant current, and a DC-DC regulator. The system provides a nominal 28 volt DC bus which varies between 25.0 and 28.5 volts. The DC regulator will provide 28.0 VDC ± 2% for the equipment needing close regulation. A functional block diagram is shown in Fig. 17.

Primary power is supplied by the sun oriented solar cell array and augmented with nickel cadmium batteries during eclipse periods. The solar array incorporates a light weight flexible substrate and the design can be of the retractable or non-retractable type. The system is the direct energy transfer type, that is, the array is fed directly onto the bus. Excess array sections are shorted to ground, with most of the excess energy dissipated as heat out on the solar array. A small amount of heat is dissipated across the shunt power transistors. The solar array regulator unit senses the bus voltage and turns the appropriate number of shunt transistors on or off to maintain bus voltage limits. This unit responds fast enough to control the array cold-to-hot voltage transient.

The EPS equipment is modularized for in-orbit maintenance as defined by Fig. 3. The physical configuration of the Battery Module and that of the Power Distribution Module are shown in Figs. 18 and 19.

4.6 Attitude Control Subsystem

The Attitude Control Subsystem (ACS) provides thrust for vehicle translation, reaction wheel unloading, and attitude control. Translation maneuvers include docking and both North-South and East-West stationkeeping. Attitude hold is provided for translation thrust misalignment, docking maneuvers, reacquisition, and backup attitude hold. The ACS consists of four identical modules installed on the outboard edges of the vehicle as depicted in Fig. 2. Each module contains six 0.5 lbf rated hydrazine thrusters and hydrazine to provide a combined total impulse of 136,300 lb-sec for all four modules, and hence a five year orbital life. Individual thrusters are oriented to provide attitude control in the event of failure of any one thruster and in most cases for failure of more than one thruster, as shown in Fig. 20.

Each ACS module consists of five major components as depicted in the Fig 21 schematic. Two independent fill valves are used; one to load 171 lbs of hydrazine propellant through the bottom port of the storage tank and 3.5 lbs of nitrogen pressurant through the top port. After filling and pressurization of the tank, the fill valves are turned closed with a wrench and capped for leakage redundancy. The storage tank delivers hydrazine propellant through a 25μ rated inline filter to the thruster valve inlets over a 230 to 115 psia pressure range. As the propellant is consumed by the thrusters, the nitrogen which initially occupies 50 percent of the tank volume expands from the initial 230 psia to 115 psia as the tank is emptied. A propellant management baffle in the shape of a cross extends from the upper polar cap of the tank to the lower polar cap where the propellant outlet is located and assures proper propellant orientation during vehicle maneuvers.

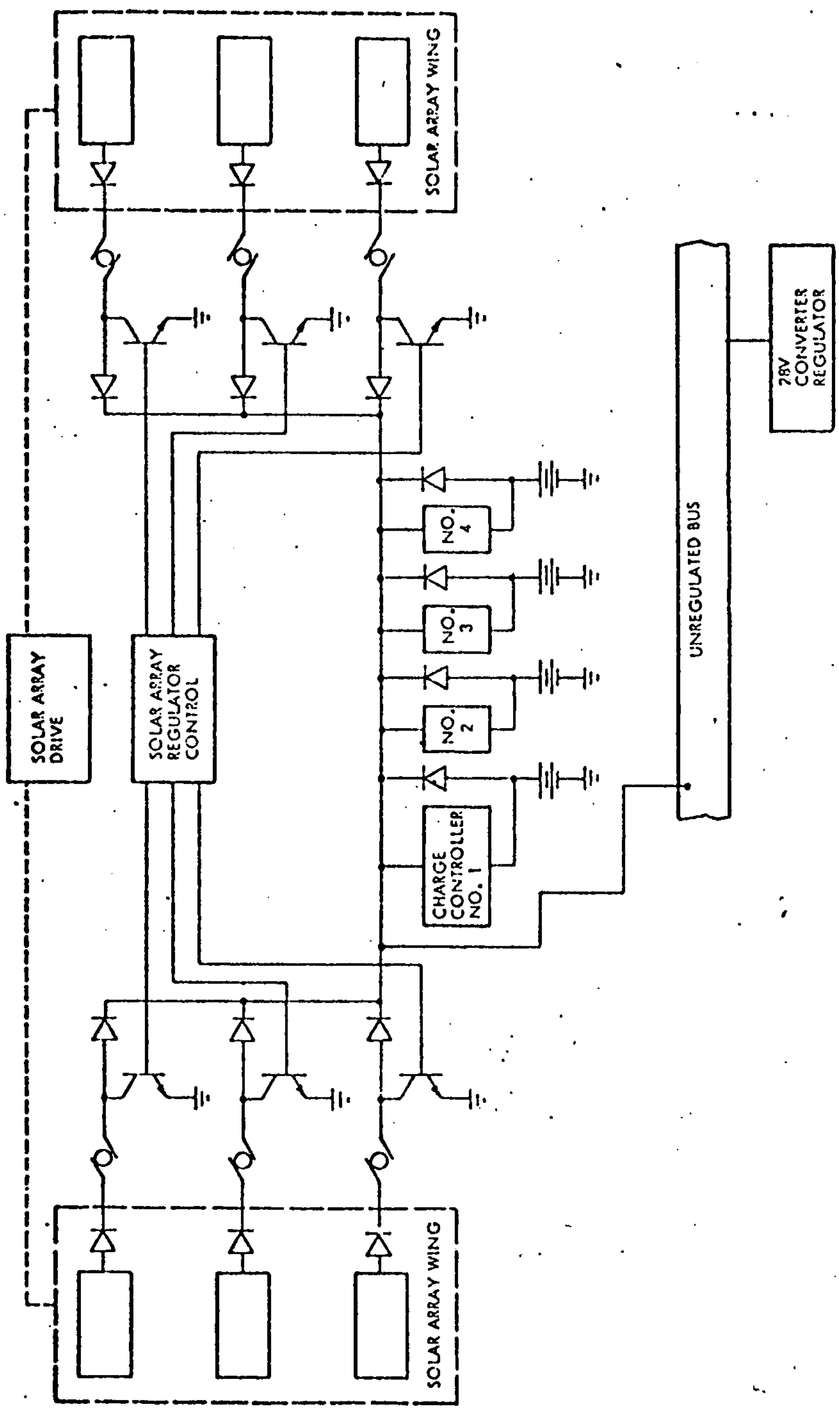

Fig. 17 Standard Communication Satellite - Electrical Power Subsystem

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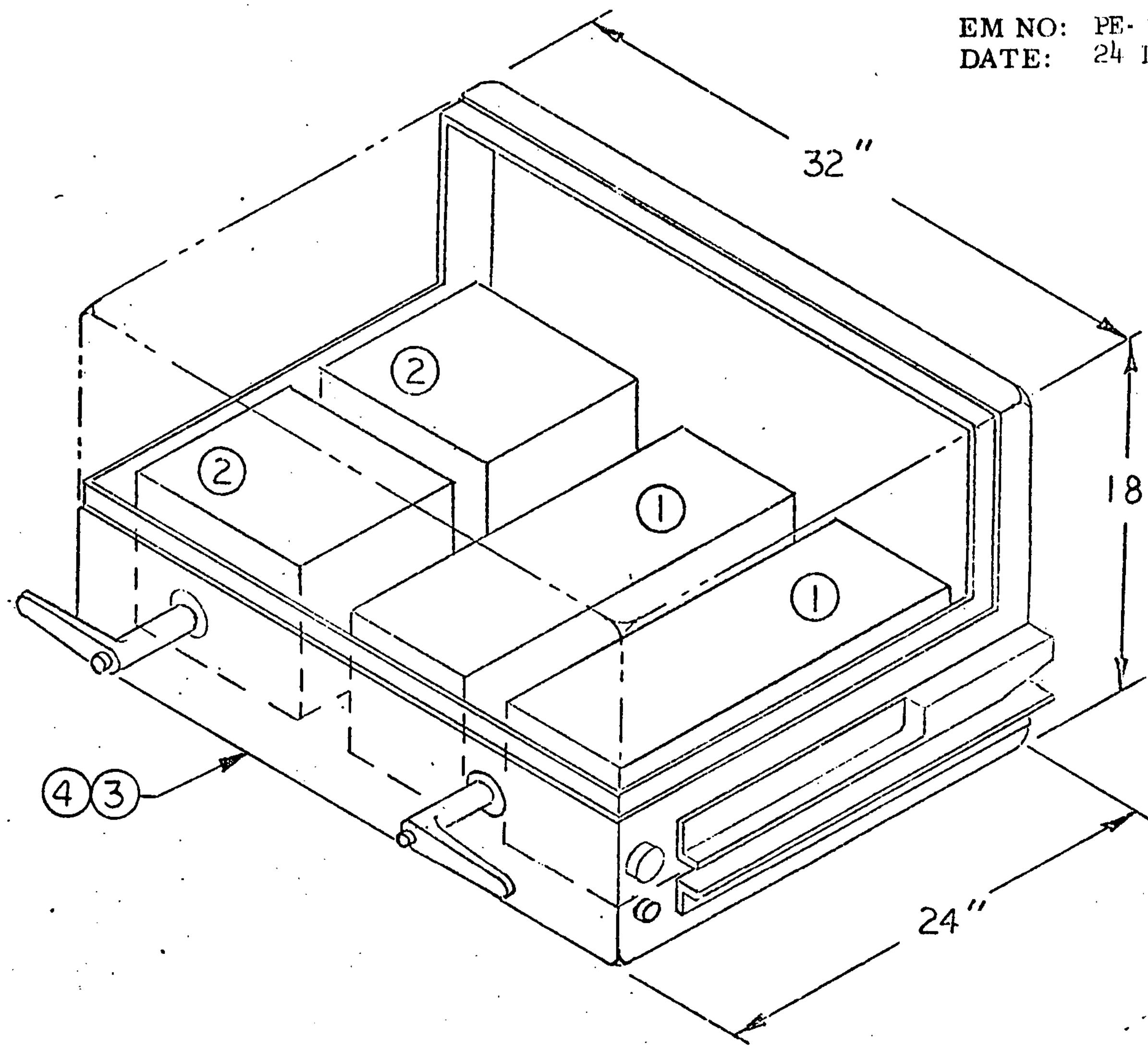
EM NO: PE-126
DATE: 24 Dec 1971

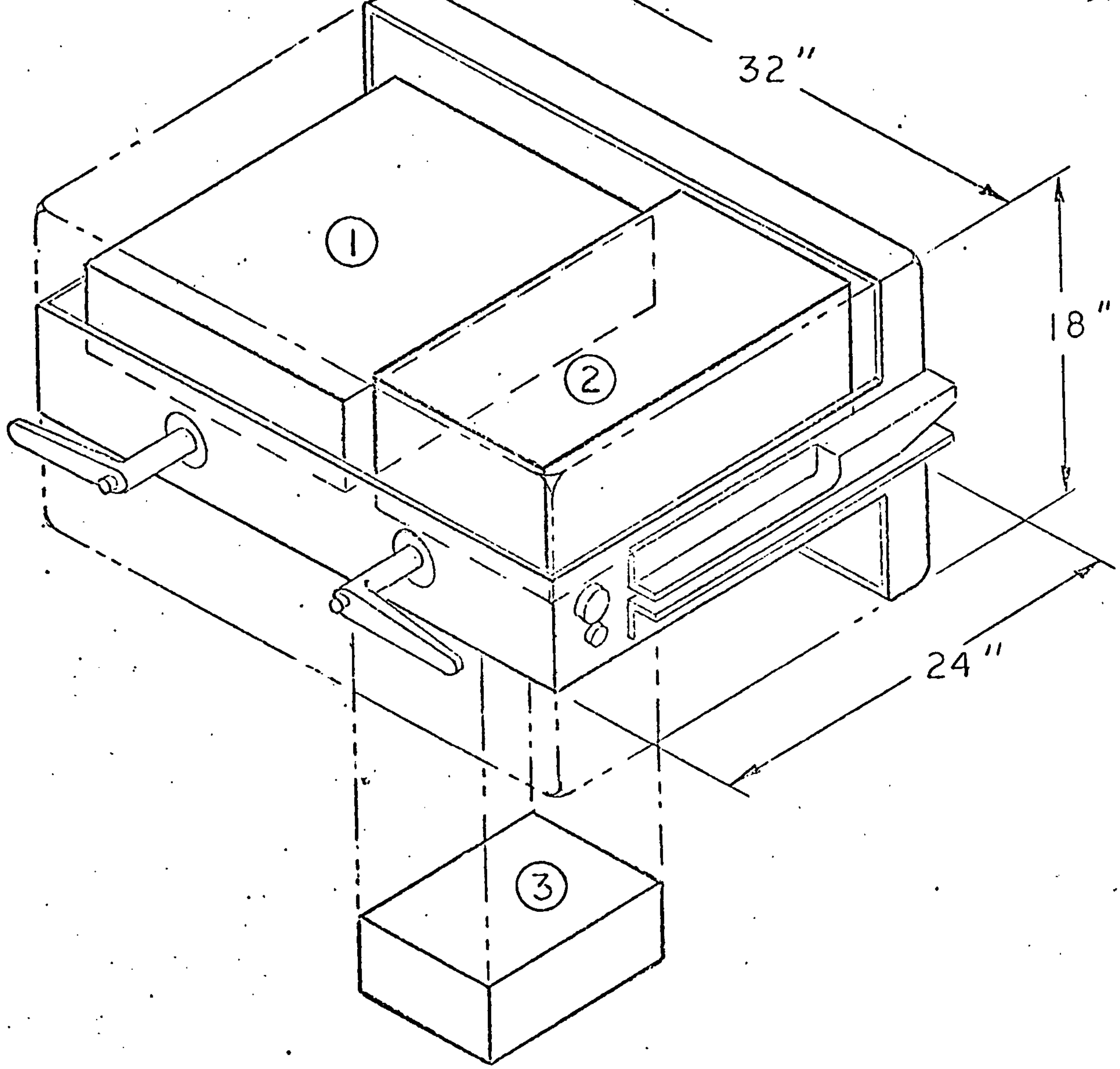


EQUIPMENT

1 TYPE 7 BATTERY - 2 REQD 40 AH NiCd	140 LBS
2 CHARGE CONTROLLER - 2 REQD	20
3 BASE & COVER	40
4 CABLES & CONNECTORS	13
SUBTOTAL	213
10 % CONTINGENCY	21
	234 LBS

Fig. 18 Battery Module - Standard Communication Satellite EPS

EM NO: PE-126
DATE: 24 Dec 1971



EQUIPMENT

1	1	POWER DISTRIBUTION UNIT	34 lbs
2	1	SOLAR ARRAY REGULATOR	8
3	1	DC TO DC REGULATOR	40
	1	CABLES AND CONNECTORS (SET)	6
	1	BASE AND COVER	31
		15 % CONTINGENCY	18
			137

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Fig. 19 Power Distribution Module - Standard Communication Satellite EPS

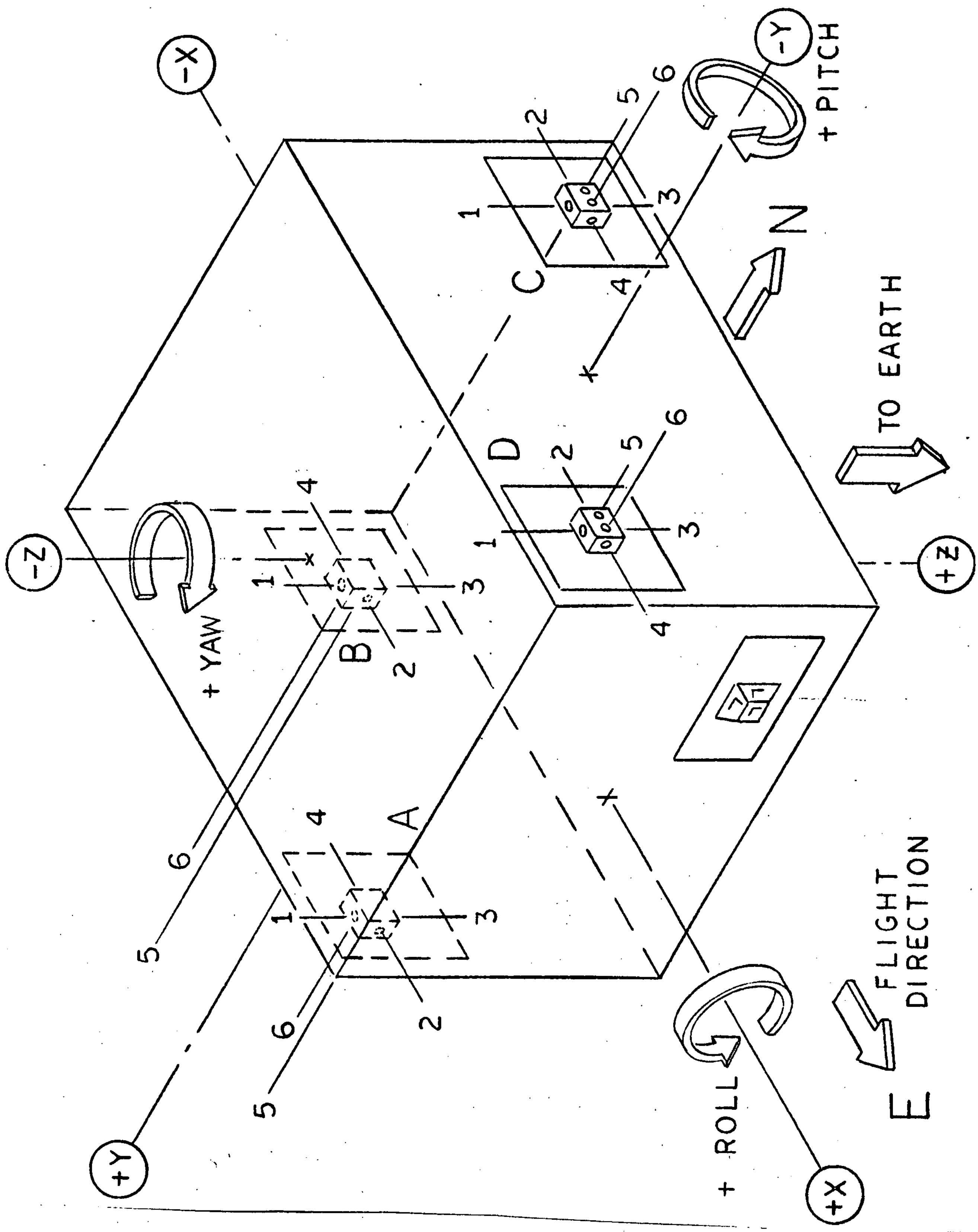


Fig. 20a Attitude Control Thruster Orientation - Standard Communication Satellite (Sheet 1 of 2)

Vehicle Motion	Active Thrusters
+ Pitch	A3 & B1 or C1 & D3
- Pitch	A1 & B3 or C3 & D1
+ Roll	A1 & D3 or B1 & C3
- Roll	A3 & D1 or B3 & C1
+ Yaw	A2 & D2 or B2 & C2
- Yaw	A4 & D4 or B4 & C4
E Translation	A4 & D2 and B4 & C2
W Translation	A2 & D4 and B2 & C4
N Translation	A5 & B6 or A6 & B5
S Translation	C5 & D6 or C6 & D5

Fig. 20b Attitude Control Thruster Orientation - Standard Communication Satellite (Sheet 2 of 2)

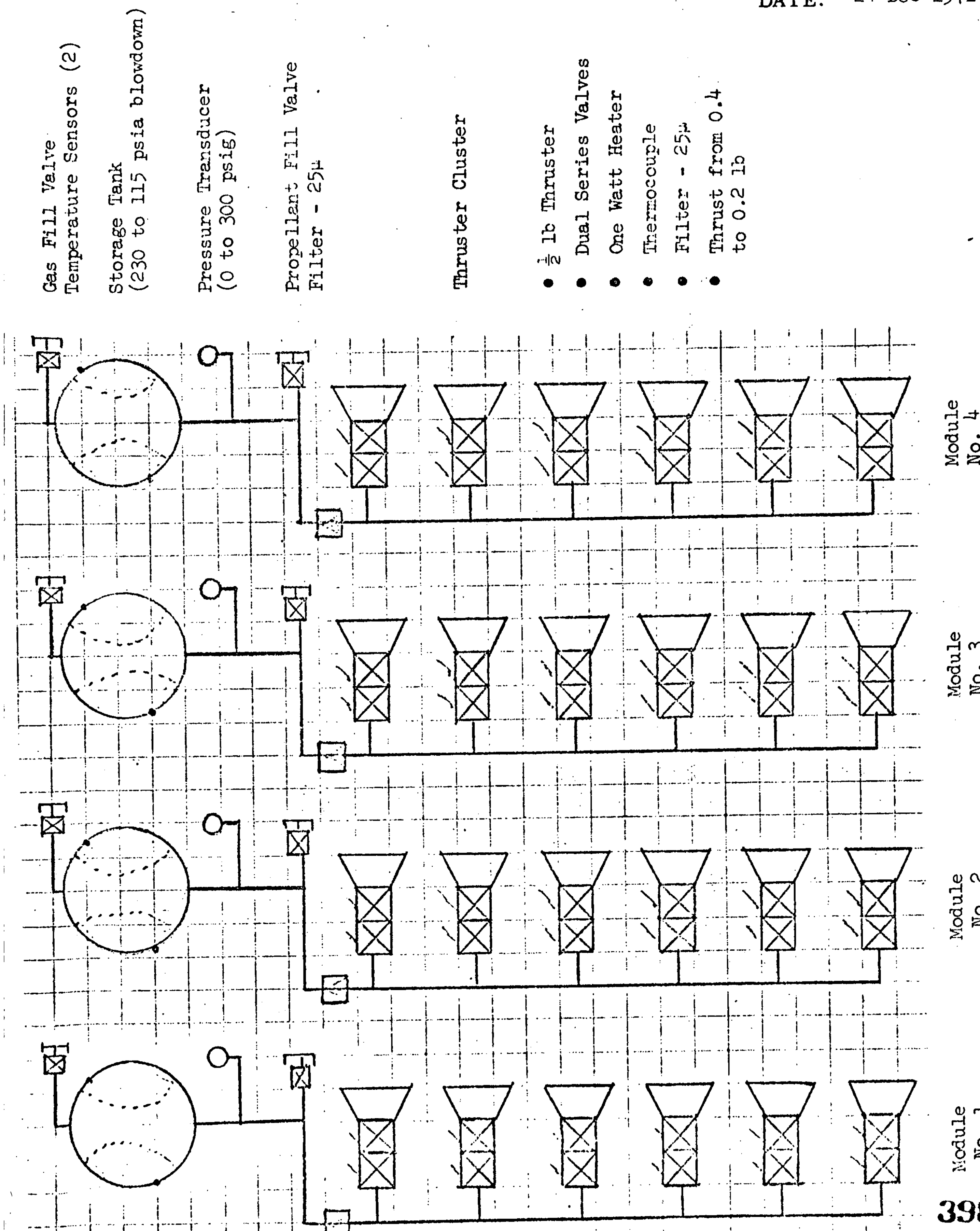


Fig. 21 Attitude Control Subsystem - Standard Communication Satellite

Each of the six hydrazine thruster assemblies is further protected from contamination with a 25μ rated absolute filter built into the valve inlet. A dual series valve arrangement is used on each thruster assembly to protect against long term leakage and failure of an individual valve. The valves nominally operate over a 25 to 33 VDC range, and have an inlet burst pressure rating of 1900 psi, a factor of 8 over the maximum working pressure. Each thruster is rated at 0.5 lb when supplied with 285 psia hydrazine. However, it is planned to use them over a 0.4 to 0.2 lb range as the storage tank feed pressure decays from 230 to 115 psia. Thus the thrusters meet the maximum 0.4 lb allowable thrust requirement.

Each thruster chamber contains Shell 405 catalyst which decomposes hydrazine into ammonia and nitrogen exhaust products. A large percentage of ammonia disassociates into hydrogen and nitrogen before it leaves the decomposition bed. The resultant hot gases expand out the thruster nozzles thereby producing external control forces on the vehicle.

The ACS module is shown in Fig. 22.

4.7 Thermal Control

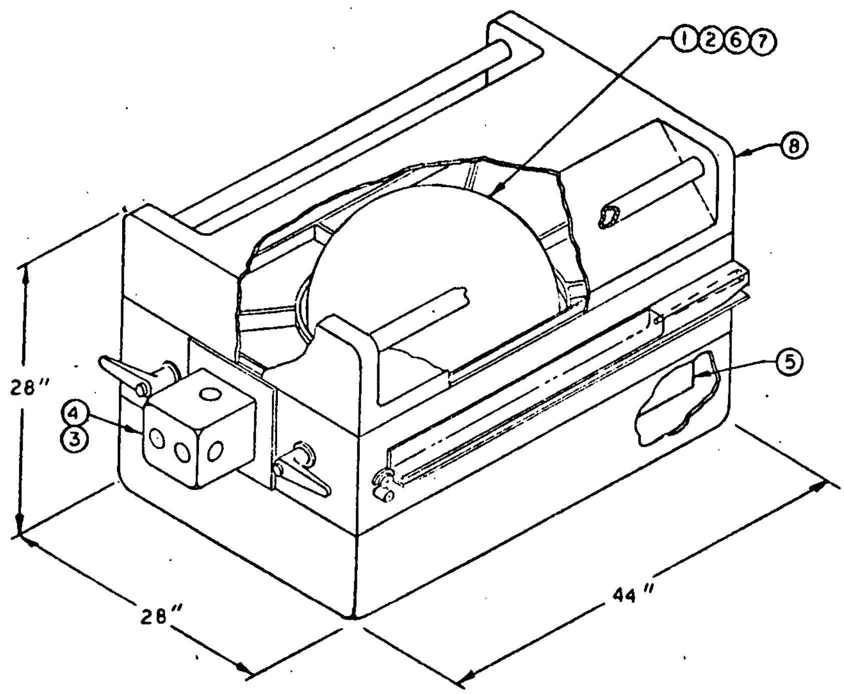
The thermal control of the Standard COMSAT will be accomplished primarily by passive thermal control techniques. To the extent possible thermal control of the spacecraft will be accomplished by the application of appropriate internal and external surface finishes and multilayer insulation. Equipment requiring temperature control within relatively narrow limits may require supplementary methods such as thermostatically controlled heaters.

4.8 Reliability

The reliability goal for the Standard COMSAT is 0.750 for five years of orbit life allocated as follows:

COMSAT	0.750
Mission Equipment	0.9540
Spacecraft	0.7860
Structure	0.9996
Environmental Control	0.9996
CDPI	0.9370
Attitude Control	0.9980
Stabilization & Control	0.9330
Electrical Power	0.9450

The spacecraft reliability goal has been met by the designs described. More details of the designs and of the reliability analyses are contained in the referenced LMSC Engineering Memos.



Equipment	Quantity	Weight (lb)
(1) Tank, 26.8" dia.	1	25.0
(3) Plumbing, Cres	3/8d x .028w x 48" 1	0.5
(4) Valve Cluster - 6 nozzle	1	6.0
(5) Electronics	1	7.0
(7) Fill Valve (S)	-	1.0
(8) Base and Covers	1	42.0
(9) Internal wire harness	AR	3.0
	Subtotal	84.5
	15% contingency	12.7
(2) Propellant - Hydrazine		171.0
(6) Pressurization gas - Nitrogen		2.0
	Total	270.1

Fig. 22 Attitude Control Subsystem Module - Standard Communication Satellite

PE-133

PE-133

STANDARD PLANETARY SPACECRAFT

COMMUNICATION, DATA PROCESSING,

AND INSTRUMENTATION SUBSYSTEM

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE:	STANDARD PLANETARY SPACECRAFT - COMMUNICATION, DATA PROCESSING, AND INSTRUMENTATION SUBSYSTEM	EM NO: PE-133 REF: DATE: 31 March 1972
AUTHORS: M. Loeb	Prepared under cognizance of: Advanced Payload Systems, Orgn 69-02 Space System Division	APPROVAL: J.C. Bolton Wayne F. Miller

33 pages

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PRELIMINARY

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1.0 Introduction

A series of reports have been prepared to provide background rationale and design data on standard spacecraft subsystems, as part of the Payload Effects Follow-On Study. This report presents the requirements and a point design of a standard Communication, Data Processing, Interface and Instrumentation Subsystem (CDPI) for a typical planetary spacecraft. In order to derive the requirements the following general assumptions regarding the mission and the on-board, earth and planet based equipment are made:

- The missions performed by the planetary spacecraft will be similar to those carried out with the Viking Orbiter and Mariner 71. The mission equipment will also be approximately the same as that projected for the Viking Orbiter and Mariner, as per Reference 1 and 2.
- The DSN will be used, with the 210 ft antenna for Earth/spacecraft communications. The DSN front-end will include a maser with an input noise temperature of 25°K. Earth/spacecraft communication is via S-Band.
- The maximum Earth/spacecraft communication range is 2.5 AU.
- A link with a planet-based system will also be available for planet/spacecraft communications. This link will operate at UHF.
- The maximum planet/spacecraft communication range is approximately 2000 Km.
- The planet-based UHF transmitter output is 30W; the antenna is omnidirectional. The UHF antenna on the spacecraft is also omnidirectional.
- Concurrent operation will be possible with the spacecraft UHF and S-Band equipment.
- Concurrent signal and data handling will be possible for on-board mission equipment and the planet-to-spacecraft communicated information.
- The planetary orbits for area survey and planet-spacecraft-Earth data relay will be the same as with the Viking Orbiter.
- The nominal mission lifetime is one year. Growth capability for a two-year lifetime will be provided.

The features of the mission and spacecraft equipment listed above must be augmented by other items resulting from Space Shuttle/Space Tug operational aspects, where they pertain. For example, it is assumed that the planetary spacecraft will be checked out and launched via Shuttle; however, configuring the spacecraft for future recovery and repair is not a requirement. Similarly, since the spacecraft will not be recoverable after launch with the Shuttle, the fail-safe backup mode used in prior point designs no longer applies. A fail operational mode must be used, when reliability requirements on specific equipment so dictate.

2.0 CDPI Functional Requirements

2.1 Mission Sequence and Functions

The CDPI is supportive to or is the major system entity in providing the following functions during a typical mission:

- **Checkout** — Interface with checkout equipment, both ground and Shuttle based. Provide the outputs for controlling automated checkout sequences assigned to the spacecraft and accept and evaluate monitoring data from spacecraft instrumentation.

- **Launch** — Accept and process launch signals from the Shuttle, initialize the spacecraft subsystems at launch.

- **Transfer Orbit** — Provide transponder functions for position/orbit determination. Accept, verify and process commands for orbital corrective maneuvers. Telemeter status monitoring data to the DSN. Provide the interfaces and computational processes for the Stabilization & Control Subsystem (S&C) and Attitude Control Subsystem (ACS).

- **Planetary Orbit Injection** — Accept, verify and process commands for injecting into the planetary orbit. Provide output signals for retrorocket firing. Supply S&C and ACS signal handling and data processing functions. Telemeter spacecraft status and conditions.

- **Planet Survey** — Accept, verify and process commands for operating mission equipment and issue prestored commands for operating the mission equipment sequences. Accept, process and store, if required, mission equipment output data. Telemeter the output data upon command or under predetermined conditions. Serve the needs of the S&C and other spacecraft subsystems. Provide telemetry of spacecraft status information.

- **Orbit Readjustment** — Accept, verify and process commands for adjusting the orbit for synchronization with respect to a particular planetary location, if required. Perform the necessary S&C, ACS and mission equipment functions. Telemeter operational and status data.

- **Planetary Lander Injection** — Accept, verify and process commands for injecting the Lander, if required. Perform the necessary supportive S&C, ACS and other system functions. Supply initializing discretes for turning on Lander equipment, if the systems are designed for this approach, prior to separation. Accept UHF transmitted data from the Lander, process and retransmit to Earth, after separation. Telemeter spacecraft status data.

- Post Planet Landing — Accept, verify and process commands for operating spacecraft subsystems and mission equipment. Accept, process and relay data from the Lander to the DSN. Perform the necessary S&C and ACS support functions. Telemeter spacecraft mission equipment and status data.

2.2 Summary of CDPI Functions

There are many iterations of function during the typical mission sequence. These may be reduced to a basic functional set, as follows:

- Interface with checkout equipment, mission equipment, S&C, ACS, other spacecraft subsystems and Lander.
- Provide control sequences of automated processes.
- Monitor and evaluate instrumentation data.
- Accept initializing and launch information.
- Provide tracking information to DSN.
- Accept, verify and process commands.
- Telemeter status monitoring and experimental or survey data.
- Provide computational processes for S&C, ACS and the mission equipment.
- Store mission equipment output, Lander uplink and other data.
- Accept, process and relay Lander communicated data.

These functional requirements placed on the CDPI are interrelated as shown in Fig. 1. The requirements on the CDPI which will satisfy the functional set may be quantified by examining the data handling and control demands levied by the mission and other equipment the CDPI supports. These quantified requirements will be identified in the following paragraphs.

2.3 Mission Equipment Requirements

The mission equipment consists of the sensors and their support electronics for examining planetary environment, surface and other characteristics from the spacecraft in orbit. A second set of mission equipment is on a planetary Lander, if utilization of a Lander is included in the mission. The only impact Lander mission equipment has on the CDPI is the data rate requirements. A maximum rate of 16.2 Kbps will be assumed, as per References 1 & 2.

In order to establish mission equipment requirements a composite list of mission equipment, as identified in Reference 1 and 2 will be assumed. The composite list is given in Fig. 2. Instrumentation and command requirements will be delineated by the supplier; however, in order to provide working data for CDPI design, some assumptions or requirements are included in the figure.

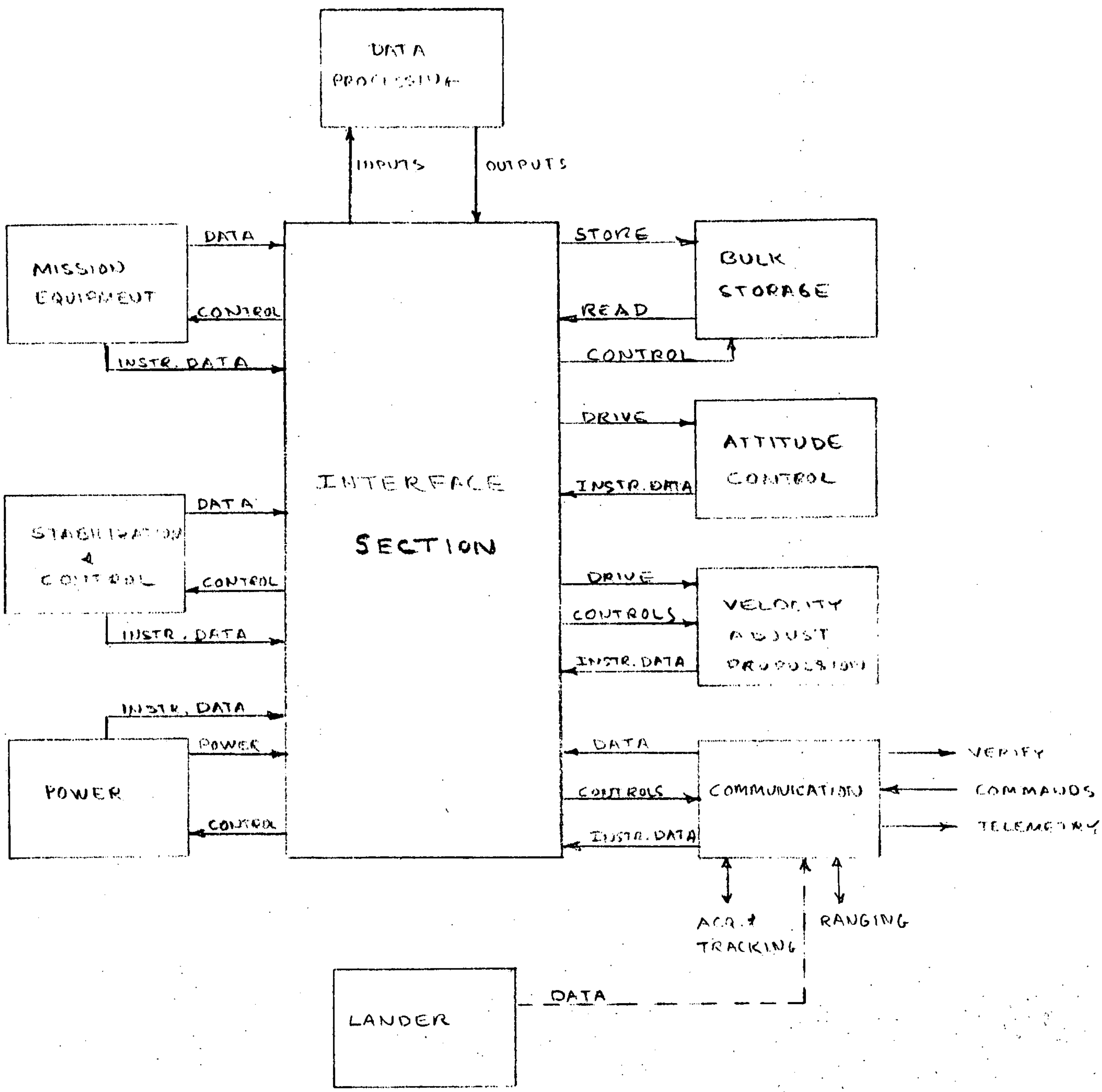

Fig. 1 CDPI Functional Interrelationships

Item	Data Rate	Data Quantity	Data Type	Control Reqmts	Assumed Instrumentation	
					Analog	Bi-Level
Lander	16 Kbps-uplink	14.4×10^6 bits (15 min of data)	Digital	2 Command words	-	-
Imaging Instrument	4 Kbps-12 12 hrs. output 1.45 Mbps input	1.75×10^8/image (150 Km sq. 30 m/line)	Digital	16 discretes	10	10
Water Vapor Detector	1-5 Kbps assumed input	4.5×10^6 bits (15 min of data) max	Digital	6 discretes	8	6
IR Detector	< 10 Kbps assumed input	5 detectors/ channel-digitized (nom. scan)	Digital	7 discretes	10	10
TV Camera	124 Kbps assumed input	1.8×10^8 bits total	Digital	2 discretes	6	2
UV Spectro-meter	1-5 Kbps assumed input	(1)*	Digital	6 discretes	10	10
IR Interfer.	1-5 Kbps assumed input	(1)	Digital	12 discretes	10	6
			Totals	49 discretes	54	44

*(1) = Storage, specified as per Reference 2 for TV, UV Spectrometer and IR Interferometer

Fig. 2 Assumed Mission Equipment in Standard Planetary Spacecraft

Review of the information in Fig. 2 indicates that high data rates are realized with the TV camera and other imaging instruments; however, the majority of the instruments included in the mission equipment have low data rates. Because of space attenuation and other factors, direct communication of the high data rates is not feasible with present technology. Thus, it will be necessary to utilize bulk storage to buffer the data between the data acquisition and data communication periods. A bulk storage requirement of 2.5×10^8 bits will be assumed.

2.4 Stabilization and Control Requirements

The requirements of the S&C Subsystem are similar to those established for the Standard Large Astronomical Observatory (LAOS) as described in Ref. 3; however, accuracy requirements and operating sequences are more constrained in the Standard Planetary Spacecraft. General operating requirements are identified in the following list. A Viking-type mission is assumed.

Operational Functions

(1) Stabilize after Space Tug separation and establish Sun-Canopus attitude with a precision of ± 0.5 deg.

(2) Cruise for 7 months with this precision.

(3) Re-orient to attitude for midcourse trajectory corrections and return to cruise orientation afterwards.

(4) Stabilize during midcourse ΔV maneuvers.

(5) Orient to attitude for planetary orbit insertion and stabilize during 25-min. velocity adjust burn.

(6) Return to Sun-Canopus (cruise) reference attitude.

(7) For payload data acquisition:

  (a) Reorient to initial attitude
  (b) Slew to maintain sensor axis in programmed line-of-sight direction
  (c) At completion of data pass reorient to cruise attitude

(8) Orient to attitude for lander retro and return to cruise attitude after Lander separation.

(9) Orient to attitude for planetary orbit adjust, stabilize during orbit velocity adjust, and return to cruise orientation.

Consideration of the operational requirements listed leads to an S&C Subsystem configuration as shown in Fig. 3. The CDPI is required to interface, route and process S&C sensor data and provide the output signals controlling the momentum wheels, jet thrusters in the Attitude Control Subsystem (ACS) and the engine actuators and controls in the Velocity Adjust Propulsion (VAP) Subsystem. The required operating modes of the S&C Subsystem are shown in Fig. 4 and the computational requirements are given in Fig. 5. S&C instrumentation required is not defined at the present time; however, the following are assumed:

Analog	110
Bilevel	32
Commands	128

2.5 Attitude Control Subsystem Requirements

The Attitude Control Subsystem (ACS) is similar to the Standard LAOS, as described in Ref. 3. These requirements include the following:

- Four propulsion modules with four thrusters per module
- Selection of redundant couples to compensate for module malfunction
- Response to S&C or ground generated commands
- Telemetry output of driver pulses -
  25 quantities - 24 bilevel

2.6 Velocity Adjust Propulsion Requirements

In order to provide the various midcourse velocity corrections and perform orbit adjustments during mission operations in the planetary regions, a Velocity Adjust Propulsion Subsystem (VAP) is required in addition to the ACS. The main engine of the VAP has approximately 2500 lbs thrust and is doubly gimballed $\pm 7^o$ in both axes. CDPI requirements assumed for the VAP are as follows:

Analog measurements	24
Bilevel measurements	16
Output pulse rate	50 per second max
Commands	20

2.7 Power Subsystem Requirements

The Power Subsystem will need CDPI support to provide control switching of power lines and regulators in accordance with commands and to activate redundant Power System components. It will be assumed that 50 power switching controllers will be used to augment, command/control of the components of the spacecraft serviced by the Power Subsystem. An estimate of monitoring requirements of the Power Subsystem will be as follows:

Analog	25
Bilevel	15
Commands	6

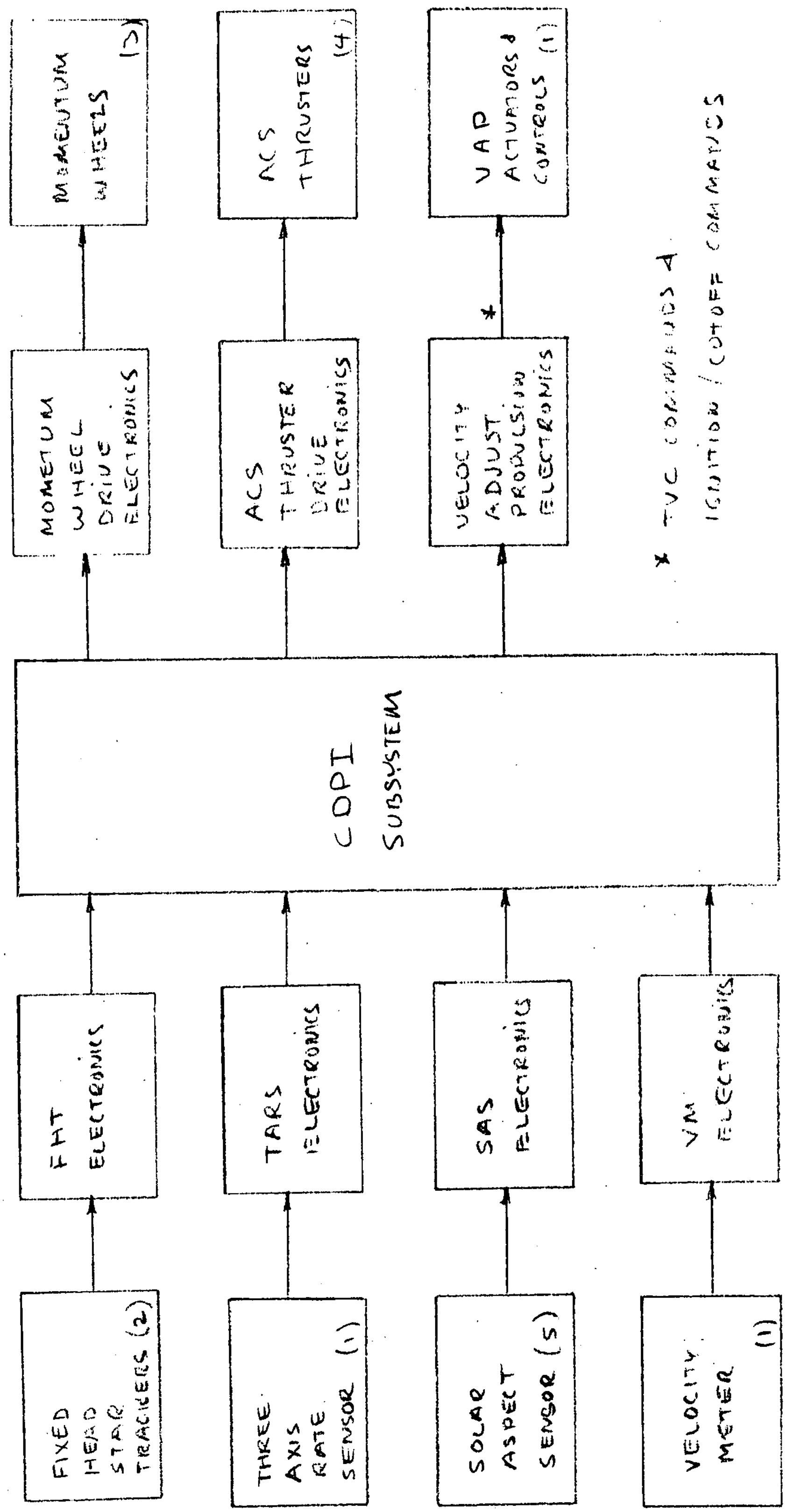

Fig. 3 Standard Planetary Spacecraft - GNS&C Subsystem

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Function	Sensors	Actuators
Space Tug Separation	TARS*	ACS
Midcourse Velocity Adjust	"	TVC, ACS
Planetary Retro	"	" "
Planetary Orbit Adjust	"	" "
Cruise Attitude Hold	SAS, FHT	Momentum Wheel
Payload Sensor Pointing	TARS	" "
Attitude Reorientation (Slew)	"	" "
Fast Attitude Reorientation	"	ACS

Abbreviations

TARS - Three Axis Rate Sensor
SAS - Sun Aspect Sensor
FHT - Fixed-Head Tracker
ACS - Attitude Control Subsystem
TVC - Thrust Vector Control

Fig. 4 S&C Operating Modes

Guide Star Reference Angles	Velocity Adjust Maneuver TVC and ACS
Measured Star Angles	Attitude Error
Inertial Attitude Update	Momentum Wheel Drive Commands (Cruise, Slew, Pitch)
Tars Attitude Sums	ACS Drive Commands (Separation, ΔV, Fast Slew)
Tars Attitude Rates	Momentum Wheel Speed Count/ Gimbal Angle Count
Gyro Drift, Scale Factor, Estimation	Ephemeris Extrapolation
Star Tracker Null Bias, Alignment Error Estimation	High Gain Antenna Pointing Vector
Attitude Program (Midcourse, Retro, Payload Data)	Mode Selection
Velocity Adjust Maneuver ΔV Sum	

Fig. 5 Computational Requirements of the S&C Subsystem

2.8 CDPI Sectional Requirements

2.8.1 Communication Section

The descriptions of the typical mission equipment given in par. 2.3 show that the majority of experiments have relatively low output data rates. The only exceptions are the imaging instruments, e.g., the vidicon camera. However, even with this equipment ultra-high data rate handling, in excess of 10^7 bps, is not required. This represents a significant departure from previous point design requirements.

Another variation in requirements is introduced by the large communication distances between the spacecraft and Earth. Presuming a transmission frequency of 2.3 GHz the free space attenuation is approximately 278 db. This places severe restrictions on data rate, antenna and other requirements. Figure 6 portrays the significant power/rate relationships obtained with the 210 ft DSN antenna as a function of distance. With a 6 ft spacecraft antenna and 50 watt transmitter, similar to the standard EOS point design, and a distance of 2.5 AU, the data rate is limited to approximately .6 KBPS. Increasing the spacecraft antenna to 12 ft would allow for a transmitted data rate of 16 Kbps, the amount specified for the Viking Orbiter and Mariner 71. If the antenna diameter is increased to 30 ft, the data rate allowed would climb to 0.1 Mbps. The higher allowable rates are desirable, particularly since they reduce the total time required to transmit a body of data to the DSN. Such a reduction is important if multiple spacecraft must be serviced by the DSN within a short time frame. A detailed evaluation of the total Earth-based/planet-based/spacecraft complex is beyond the scope of this study; however, the data rate requirements given for Viking may be assumed as a baseline. Thus, for the purposes of this paper, a 12 ft diameter antenna will be assumed as an implementation requirement. Dimensional tolerances on antenna structure must be kept within 0.2" to keep the losses to less than 1 db.

Other Communication Section requirements, as established by spacecraft and link demands are listed in Fig. 7. It should be noted that communication to Earth is via the DSN, not a relay satellite as in previous point designs. This is due to antenna limitations on projected relay satellites, e.g., TDRS.

The Fig. 7 requirements have many items which are similar to prior point designs; however, two major differences should be noted - the need to include bulk storage in the CDPI and changes in ranging technique. The bulk storage is needed to enable storing Lander data for relay to Earth when the Lander is occulted by the planet or other body, as well as for temporary storage of the planetary spacecraft experimental or survey data - particularly the high rate imaging instrument data. The bulk storage capacity should be equivalent to approximately 1.8×10^8 bits, to enable storing the digitized data equivalent to the total data of one pass. According to Ref. 2, the Data Storage System (DDS) of Mariner 71 has this capacity. The same unit will be assumed for the Standard Planetary Spacecraft.

Changes in ranging technique may be necessary because of the high rate of frequency shift which can occur in certain missions, e.g., during periapsis of the Viking mission. This could result in loss of lock-on of the DSN on the spacecraft, as noted in Ref. 3. Digitizing and using a pseudo noise (PN) range code, the method of the standard EOS point design is a good possibility and is used for this point design. Further study would be required to establish appropriate clock rate, coding, etc.

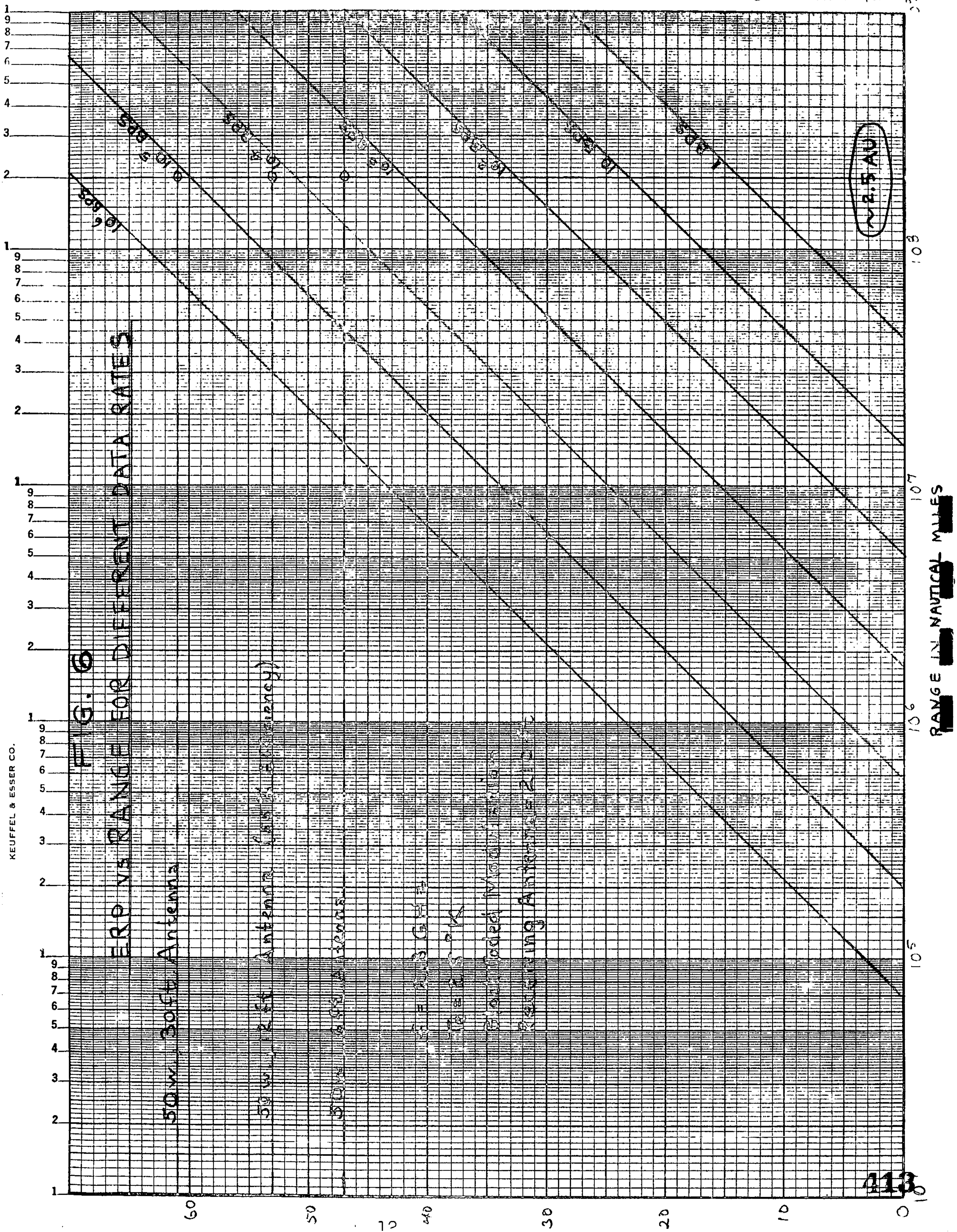
FIG. 6
ERP VS RANGE FOR DIFFERENT DATA RATES
50W, 30ft Antenna
50 W, 12 ft Antenna (55% efficiency)
50W, 4 ft Antenna
f = 2.3 GHz
T = 25°K
Block coded Modulation
Receiving Antenna 210 ft
10^6 BPS
10^5 BPS
10^4 BPS
10^3 BPS
10^2 BPS
10 BPS
1 BPS
~2.5 AU
RANGE [redacted] NAUTICAL MILES [redacted]
KEUFFEL & ESSER CO.
10^5
10^6
10^7
10^8
60
50
40
30
20
10
0

Function	Implementation Requirements
Command and Data Update	• Digital - Data Rate at .1-10 Kbps • Frequency - S-Band uplink - approx. 2.1 GHz • Modulation - SIK or range code or Mariner-type (Ref. 2-3) • Output - Serial PCM to computer for command decoding or computer update
Telemetry	• Data Rate - Variable - 1 to 16 Kbps • Frequency - S-Band downlink - approx. 2.3 GHz • Modulation - SIK or phase-modulated output • Input - Serial PCM
Ranging	• PN Range code (clock TBD) or Mariner type (Ref. 2-3) • 50 ft (1σ) bias error and 60 ft rms noise error • Determination of range rate 0.2 ft/sec (1σ) with 1 sec smoothing
Planet Uplink	• Data Rate - 16.2 Kbps Frequency - UHF 400-800 MHz • Modulation - STK or PCM
Acquisition & Tracking	• Rough pointing from computer • Coarse and fine pointing at S-Band
Bulk Storage (Tape Recorder)	• Record rate - 12 Mbps • Playback - 1-16 Kbps adjustable • Capacity - 500-1000 ft tape 8 discretes

Fig. 7 Communication Section Requirements

2.8.2 Instrumentation Requirements

The instrumentation utilized in the spacecraft for status monitoring of the on-board components is divided into two main groups - mission equipment and spacecraft subsystems. With the modularized approach in standard spacecraft design, the actual sensors and transducers, plus possibly signal conditioning and digitizing circuitry, are included in the modules themselves. Thus, the instrumentation requirements served by the CDPI are commutation and multiplexing of the instrumentation output signals, providing A/D conversion as necessary, and performing the required on-board processing and telemetry of the instrumentation data. Since the experimental units may be changed for different missions, it would be desirable to separate instrumentation interfacing of the two groups. This technique was utilized in prior point designs and is also needed in this CDPI.

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In order to establish the CDPI requirements resulting from instrumentation demands, the following estimates are assumed:

Source	Analog	Bilevel
Mission Equipment	54	44
S&C	110	32
ACS	25	24
VAP	24	16
Power	25	15
CDPI Subsystem	40	24
Totals	278	155

The transducer outputs are standardized at 0-5 vdc or 0-50 mv as full scale output ranges, while the bilevel ranges are nominally 0 and 5 vdc for a "0" or "1" indication. Analog-to-digital conversion to an 8-bit output word is specified as a minimum requirement.

The large number of analog outputs will require subcommutation prior to multiplexing and A/D conversion for computer input. Similarly, sample and hold registers will be required to handle the bilevel data. Sample rate requirements on both the analog and bilevel outputs are nominal. Computer input of 16 A/D converted words per 0.1 second and one 16 bilevel data word per 0.1 sec are representative values.

2.8.3 Data Processing Requirements

As noted in the Introduction to this memo, the orbits of the Standard Planetary Spacecraft will be such that Space Shuttle/Space Tug revisit for maintenance or refurbishment will not be available. Thus "fail-operational" is the defined failure mode. These conditions are very different from those specified for the spacecraft considered earlier in this study and will affect the Data Processing Section of the CDPI, particularly the Input/Output Unit.

In prior point designs the requirements for ease of maintenance and flexibility suggested expanding CDPI Interface Section capability and thereby minimizing the computer Input/Output configuration. This approach was also advantageous in simplifying switchover to a standby computer should the operating computer fail. In all prior point designs reliability goals dictated redundant computers; therefore, reducing the difficulty of switchover was important.

In this point design, the requirement for fail-operational and elimination of Shuttle or Tug revisit overrides the requirements for flexibility and ease of maintenance. The question which needs to be resolved is - can a computer Input/Output Unit be provided which may be standardized, have sufficient capability to satisfy multiple and diverse mission requirements, yet satisfy stringent reliability goals?

The answer to the question probably lies in first defining the most cost-effective Interface configuration which can satisfy the fail-operational requirement. All the residual functions are then assigned to an expanded computer Input/Output Unit. This approach would result in an overdesigned Input/Output configuration for most applications; however, the advantages of standardization would be obtained to offset the overdesign.

In order to resolve the computer Input/Output Unit vs Interface Section tradeoffs, a detailed study is required. Such a study is beyond the scope of this point design; however, some possibilities may be suggested. These include the following:

- subcommutated multiplexing assigned to the Interface Unit; primary multiplexing in the computer I/O
- A/D conversion in the computer I/O
- multiple I/O registers in the computer
- discrete issuance directly from the computer I/O rather than via a discrete sequencing register

These functional requirements will be assumed for this representative point design.

The requirements of other units in the computer-memory, arithmetic and control are similar to prior point design requirements. To eliminate cross referencing the data processor requirements are given here.

Word Length. The accuracy requirements for S&C functions of ±0.5 deg for attitude control with respect to the Sun and Canopus references and 0.03°/sec on line-of-sight rate are very nominal compared to prior standard spacecraft. A word length of 10-12 bits would be sufficient; however, to be consistent with prior designs 16 bits will be specified.

Memory Capacity. The memory capacity required is based upon the following word count estimated.

Function	16 Bit Word Totals
Stability & Control	700
Commands	250
Sequencing & Command Processing	1250
Telemetry Formatting	350
Received Data & Command Test	50
I/O function routines	250
AGE/Tug routines	250
Attitude control	250
	3,350 words

This indicates a 4K 16 bit memory would meet requirement; however, an 8K 16 bit memory is specified to allow for growth items. This does not include any redundancy of memory which may be needed to meet reliability goals.

Memory Type. A random access memory is required with non-destructive readout (NDRO) desired. Read-only memory sections may be needed for the more critical subroutines; however, these needs are not determined at this time.

Operating Speed. Speed requirements are not definite at this time; however, nominal values of 15 microseconds for an add and 85 microseconds for a multiply should suffice. Speeds in excess of these figures would be desirable.

Instruction Repertoire. A standard mix of input/output arithmetic, logical, memory addressing and indexing should suffice. Double precision instructions do not appear a requirement as they did for LAOS; the few double precision computations which may occur may be realized through software routines.

Input/Output. Input/output should nominally be through direct computer control, although direct memory accessing (DMA) may be used if proper precautions are observed to protect memory contents. The equipment complement should nominally be as follows:

Component	Number	Characteristics
Input registers	2	12 - 16 bits word length
Output registers	2	12 - 16 bits word length
Multiplexers	1 or 2	16 channel input
A/D converter	1	8 - 10 bits
Interrupts, External	3	Power-on built-in
Discretes, variable	16 - 32	Direct Logic drivers
fixed	TBD	Duration TBD

Weight, Size and Power. Although specific weight, size and power requirements are not established, a fourth generation technology is specified to enable satisfying reliability goals. This would result in a computer with the following characteristics.

Weight	15 lb
Size	0.3 ft^3
Power	50 w. ave.

2.8.4 Interface Section Requirements

The problem of defining the Interface Section was discussed in the previous paragraph. Presuming the allocation of function given in the paragraph to the computer Input/Output Unit, the following functional requirements are defined:

- Accepting, sampling and subcommutating analog instrumentation output data
- Accepting, sampling and delay of bilevel data for computer input
- Accepting and delivery of S&C outputs to the computer
- Relay of information from the computer to the Communication Section, AGE or Tug-based equipment
- Accepting delaying and delivering mission equipment data outputs to the computer or Communication Section

The control processes which must be mechanized in the Interface Unit include the following:

- Accept computer stored or verified uplink commands and provide the logic for delivery of discretes to the appropriate subsystems
- Provide the control logic circuitry for application or removal of power under uplink or computer command
- Condition and deliver drive signals to the S&C momentum wheels, ACS jet thruster valves, VAP actuator electronics and controls and mission equipment controls
- Provide timing references for synchronizing and operating logic. The basic timing reference accuracy is one part in 10^6.

The quantification of these requirements is in accordance with the input/output demands of each of the subsystems serviced by the Interface Unit and the remaining CDPI sections. These quantities are defined in the previous paragraphs.

3.0 CDPI Design

3.1 Communication Section

A block diagram of a representative point design Communication Section which will meet the requirements specified in par. 2.8.1 is shown in Fig. 8. This design contains components similar to those used in prior designs, e.g., the standard EOS and LAOS; particularly the S-Band receiver and transmitter. Components which are different include the UHF receiving and demodulation units, modulators and antennas; however, no new significant development effort seems necessary. A new S-Band antenna design will be required however.

The S-Band antenna is a 12' dia. parabolic unit. Because of the size a furlable unit is possibly needed. This is well within the state-of-the-art since the ATS F&G designs extend to 30'. Provisioning for steering does not appear to present significant problems since the weight of the antenna would be less than 20 lbs, including mountings and feed, but excluding the servo drive and electronics.

Figure 8 does not include the redundant components which will be necessary to meet the specified reliability goals. The redundant components include the receivers, transmitters and the bulk storage tape recorders. The logic control for switchover of redundant components would be through fault isolation and control routines stored in the computer in the Data Processing Section. Uplink command is an alternate mode for switchover.

The general specifications for each major Communication Section component are given in Fig. 9 and a listing of estimated weight, size, power and costs of the components is contained in Fig. 10.

It is noted that instrumentation will be required for status monitoring of the Communication Section components. Interconnection of the instrumentation will be via the Interface Section of the CDPI.

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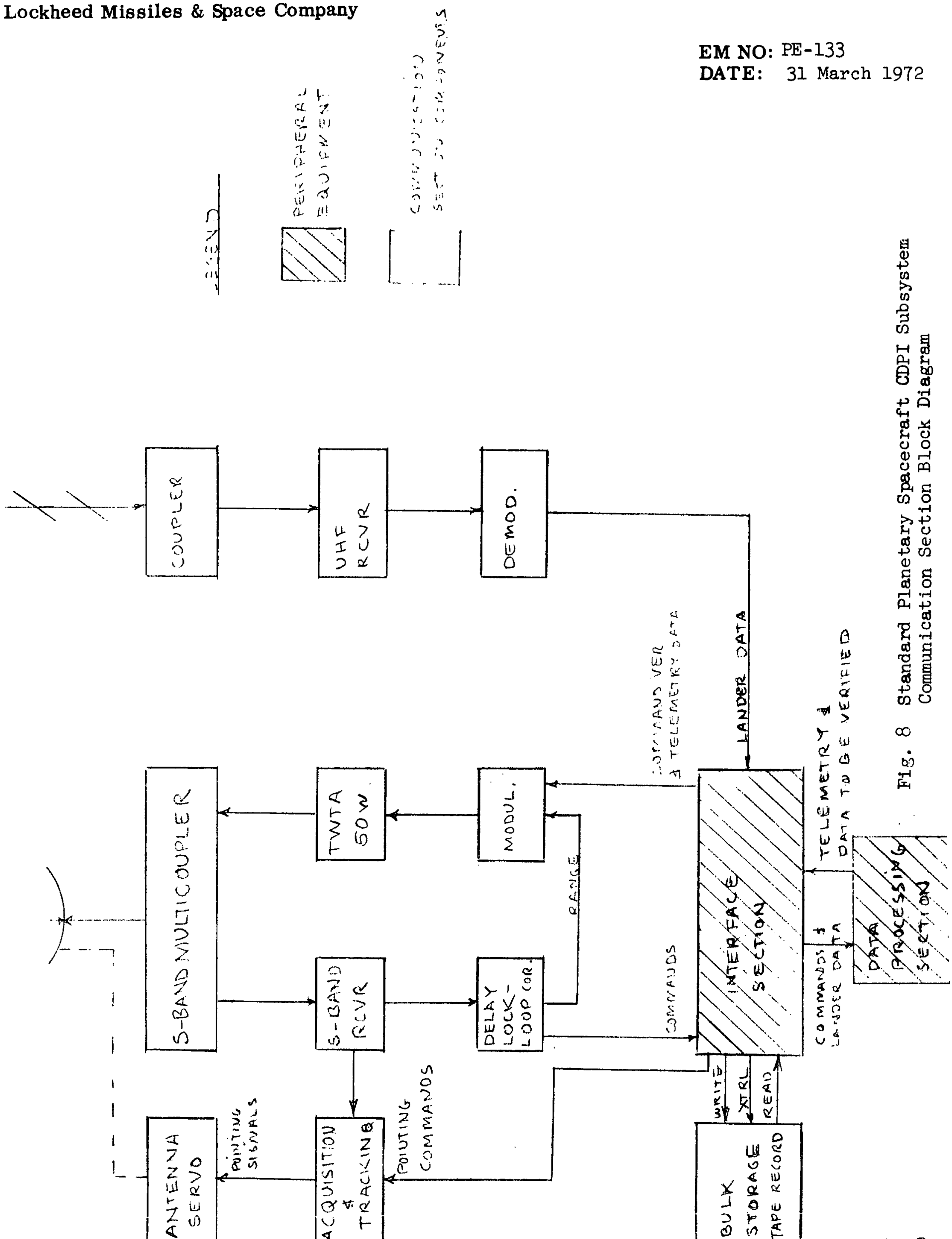

Fig. 8 Standard Planetary Spacecraft CDPI Subsystem Communication Section Block Diagram

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1. S-Band Multicoupler	1 XMTR, 1 RCVR, Power Handling 100 W. AVP. Loss < 1 db
2. S-Band PLL Receiver	$BW_{(RF)}$ = 2 MHz, NF = +6 to +8 db
3. Delay Lock Loop Correlator	TBD
4. S-Band Modulator	SIK or Phased Angle modulation 16 Kbps input. utput ~ 0 dbw
5. Acquisition & Tracking	Rough pointing - computer command - Fine pointing - Phase lock-loop tracking
6. Antenna Servo Electronic	TBD
7. S-Band TWTA	50w. output
8. UHF Receiver & Demod.	BW = 150 KHz, NF = +4 to 6 db
9. S-Band Parabolic Antenna	Diameter, 12 ft., gain at 2.3 GHz 36.4 db at 55% efficiency, furlable, Single S-Band feed, rotary joints (waveguide or coax), total line loss < 2 db
10. UHF Antenna	Omnidirectional, gain 0 to -3 db. Configuration TBD, biconical assumed.
11. Bulk Storage	Tape recorder as per Mariner '71. 8 data tracks + synch. track - Packing density - 3500 bits per inch/ track record 5w, playback 1.5w.

Fig. 9 Communication Section Component Specifications

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Component	Weight (lbs)	Power watts	Size in.	ROM Cost x 10^3	
				Non-Rec.	Recurring
1. S-Band TWTA (50 w. out)	12	20	12" x 4 x 4"	500	50
2. S-Band Mod/Driver	3	10	3" x 3" x 2"	50	10
3. S-Band PLL Receiver Acquisition & Track	3	5	3" x 3" x 2"	100	15
4. S-Band Multicoupler	3	-	4" x 4" x 2"	20	5
5. S-Band Receiver	3	5	3" x 3" x 2"	50	10
6. Delay Lock Loop Correlator	2	4	3" x 3" x 3"	20	5
7. UHF Receiver & Demodulator	3	5	3" x 3" x 2"	10	5
8. Antenna Servo Electronics	8	20	8" x 6" x 6"	30	10
9. Antenna Feed (S-Band	2	-	-	-	(Part of Ant Assy)
10. Antenna Servo motor & Gears	15	20	-	-	(Part of Ant Assy)
11. Antenna (Furlable)	20	-	12' dia.	200	50
12. Antenna (UHF)	5	-	14" D x 14"	20	5
13. UHF coupler	0.5	-	2" x 1" x 1"	-	0.2
14. Rotary Joints	2	-	(Part. of Ant.)	10	5
15. Tape Transport	10 lb, 6 oz	8-10	8" x 6" x 6"	-	Per LEC quote
16. Transport Electronics	10	8-10	8" x 6" x 6"	-	Per Motorola quote

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Fig. 10 Communication Section Component Description

3.2 Interface Section

In order to satisfy interface requirements and still be consistent with prior point designs, the Interface Section of the Standard Planetary CDPI will contain two units. The first unit is primarily devoted to mission equipment interface, particularly for interconnection of the mission component outputs with the tape recorder and the computer. The second unit provides the interfacing functions for the other spacecraft subsystems. This design is similar to that used for the standard LAOS, except that two rather than three units are used in the configuration. Each Interface Unit will be briefly described in turn.

3.2.1 Mission Equipment Interface Unit

A block diagram of the Mission Equipment Interface Unit is shown in Fig. 11. The high rate data derived from the imaging instruments and TV camera are accumulated, encoded and multiplexed prior to storage on the tape recorder in the Communication Section. Normally, only one type imaging is required at a time and the high speed data are not multiplexed; rather, the output is stored on the tape recorder after preliminary encoding and formatting.

The lower speed mission equipment outputs may follow one of two paths - to the high speed multiplexer for intermixing with high speed data or to the computer via the second Interface Unit for subsequent processing, sequencing and formatting.

The remaining mission equipment outputs, inputs and timing are handled in the same manner as prior point designs - analog outputs are subcommutated, bilevel outputs gated and stored in delay registers, while the operating and power controls are realized through gating and computer originating sequence logic. The multipliers and divider circuits needed for mission equipment controls are included in the Interface Unit. The primary frequency reference is included in the other Interface Unit, as in previous designs. The components needed for this Interface Unit are listed in Fig. 12.

Assuming packaging as with prior Interface Units, i.e., plug-in logic boards, except for the components identified in Fig. 12, and utilizing cabling interconnections, the following estimates are obtained:

Logic Boards	Ckts/Board	Total Weight	Total Power	Total Size
2	38 MSI + Amp.	3 lbs	6 W.	< 0.1 ft^3

Item	Base Cost	Final Cost
Art work fabrication	\$1400/board	\$2800
Components (TTL & FET)	300	600
Wiring & Packaging	2500	2500
Estimated Cost for Logic		\$5900

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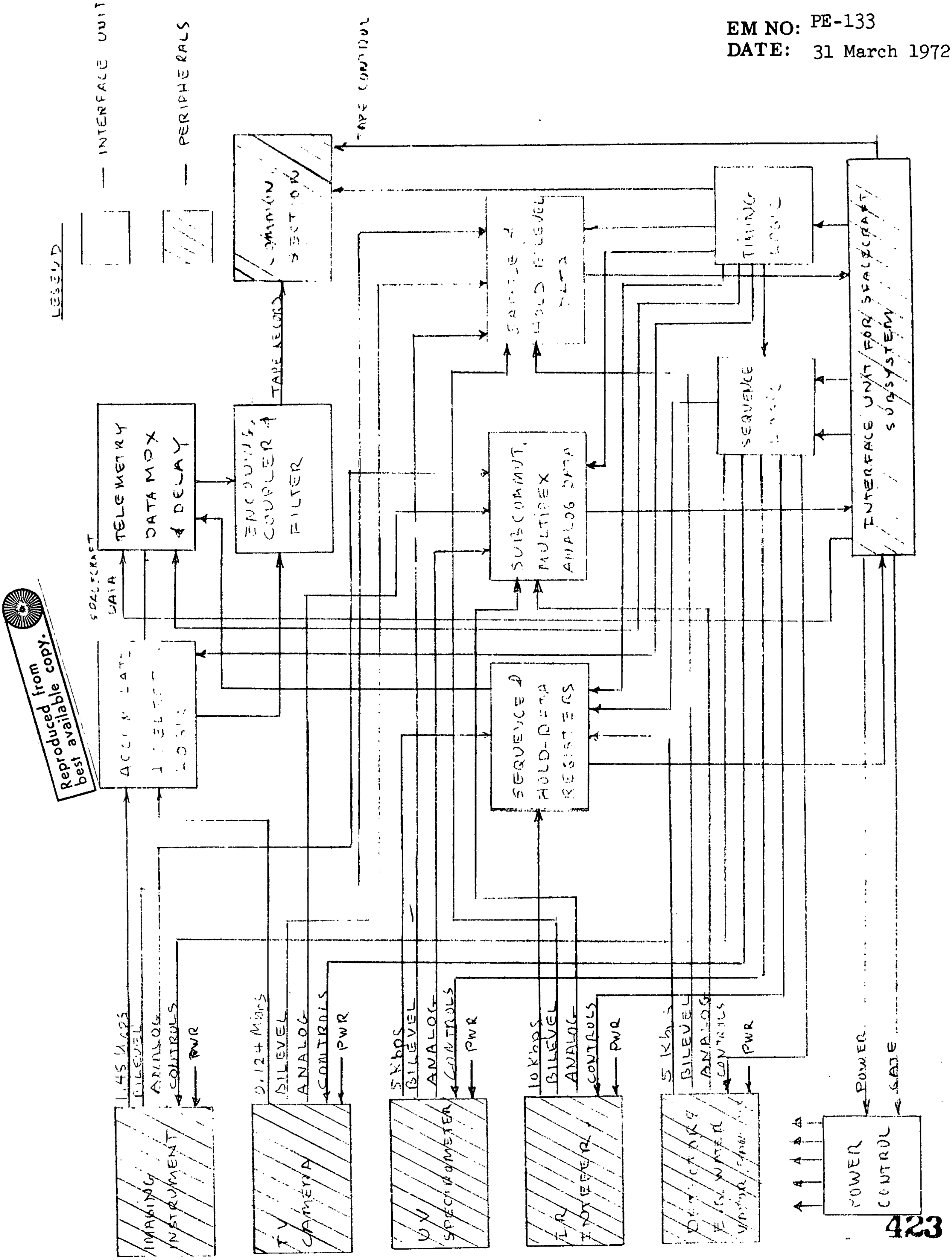

Fig. 11 Standard Planetary Spacecraft - CDPI Mission Equipment Interface Unit Block Diagram

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High Speed Accumulate & Select Logic	2 12 bit registers, 4 gates
High Speed Multiplexer & Delay	4 12 bit shift registers - 48 Impatt diode gates.
Encode, Coupler and Filter	8 bit register, 16 gates for sequence select
Bilevel Sample & Hold Register	44 gates, 1 12 bit register
Sequence & Drive Logic	6 bit register - 49 "and" gates
Power Control	5 gates
Timer	Frequency divider assume 12 bit register and 10 gates
Analog subcommutator	54 gates and one amplifier
Packaging	38 MSI circuits/board + Mother board. Separate packaging required for the select, multiplex and coupling components.
Logic	TTL MSI

Fig. 12 Mission Equipment Interface Unit Components

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Unit	No.	Wt (lb)	Power (w)	Size (ft^3)
Select	1	1	4	0.1
Multiplex	1	1	3	0.1
Coupling	1	< 0.5	2	0.1
		2.5	9	0.3

The costs would be as follows:

	Non-Recurring	Recurring
Select	$ 7,000	$ 2,500
Multiplex	15,000	6,000
Coupling	4,000	1,500
	$26,000	$10,000

Total costs are as follows:

	Non-Recurring	Recurring
	$26,000	$10,000
	2,800	3,100
	$28,800	13,100

3.2.2 Spacecraft Subsystem Interface Unit

A block diagram of the Spacecraft Subsystem Interface Unit is shown in Fig. 13. The unit is much simpler than the equivalent low data rate Interface Unit used in the standard EOS and other point designs. This simplification is obtained through expansion of computer input/output capability as discussed in par. 2.8.3.

The design is relatively straight forward. Each sensor peripheral to the Interface Unit has three outputs - data, analog and bilevel. The data channels are fed to the Data Processing Section via the needed registers, gates or counters for subsequent computation and other processing. Analog data are fed directly to the computer input for multiplex and A/D conversion. Bilevel data are treated in the same manner as in prior point designs, i.e., gated signals are fed to a sample and hold register which is cyclically output to the digital computer. Inputs to the sensors are the discrete sequences for sensor operation and the on-off power controls.

The various peripheral output devices are treated in a manner similar to the inputs, i.e., analog, bilevel, power and control functions are sensed or driven by the computer via Interface Unit components. The drive functions to ACS, VAP and momentum wheels are nominally duration/frequency modulation of computer output variable discretes.

The interface paths between the first Interface Unit and the second are also similar to prior designs, except for the data stored or read out by the tape recorder. Primary data multiplexing is in the first Interface Unit; therefore, a data path must

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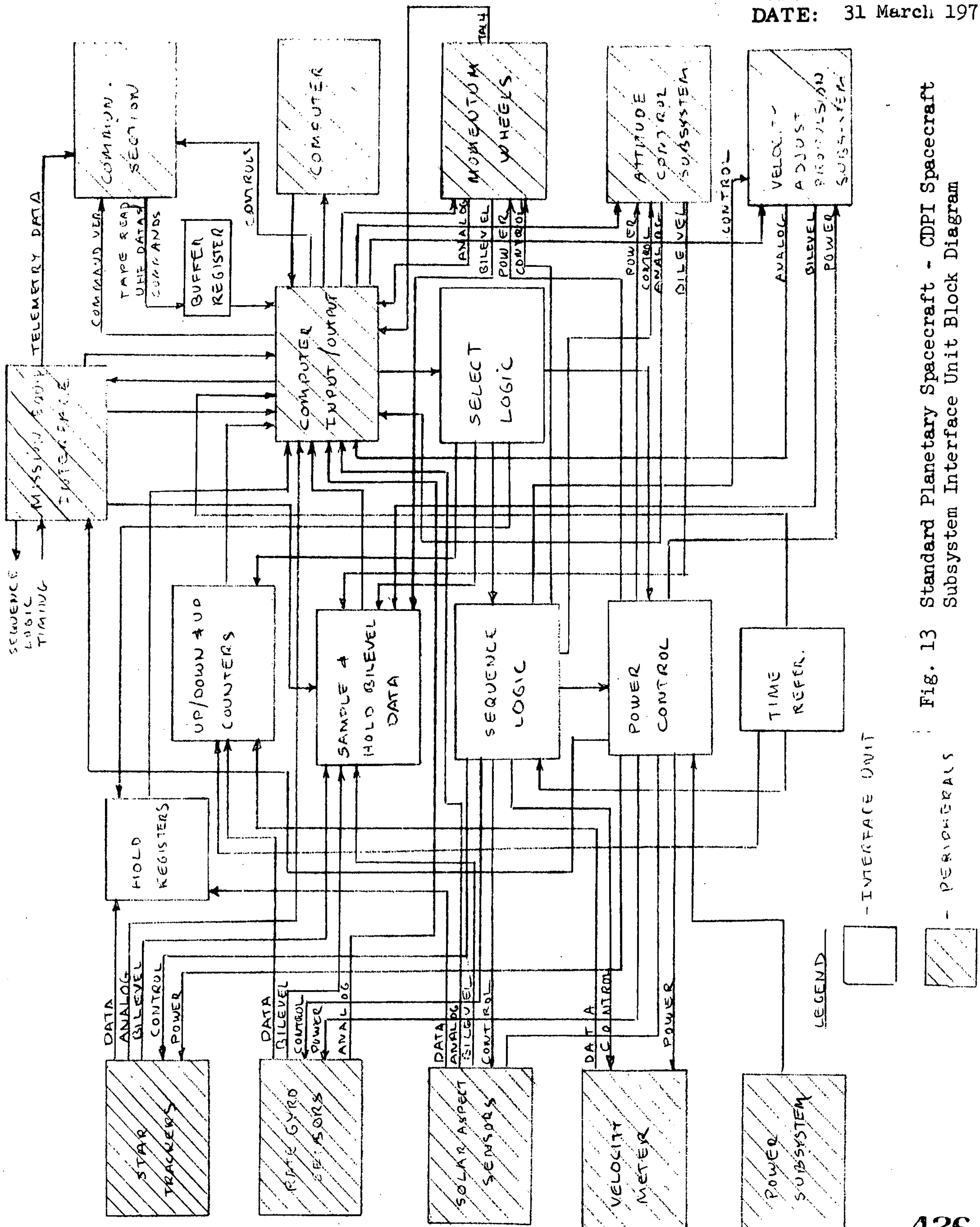

Fig. 13 Standard Planetary Spacecraft - CDPI Spacecraft Subsystem Interface Unit Block Diagram

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be provided to enable multiplexing of the spacecraft subsystem status monitoring data with the mission equipment data.

The primary components used in the Spacecraft Subsystem Interface Unit are listed in Fig. 14. Assuming the same type of packaging as that used for the first Interface Unit, but with addition of a motherboard for interconnection, besides cabling, the following estimates are obtained:

Logic Boards	Ckts/Board	Total Weight	Total Pwr.	Total Size
3	38 MSI + Ampl. + Xtal. Reference	4.5 lbs	9 w.	0.2 ft^3

Item	Base Cost	Final Cost
Artwork Fabrication	$1400/Board	$4200
Components (TTL + FET + Xtal)	300	900
Wiring & Packaging	3500	3500
Total Estimated Cost		$8600

These estimates include the gating circuits for subcommutated multiplexing of analog data.

3.3 Data Processing Section

The main computer requirements specified in par. 2.8.3 may be satisfied by a number of computers which are in production or in advanced stages of development at the present time; however, many of these computers will need additions to the input/output sections to meet the increased capability necessitated by this design. There is one computer, the ADAPT 244, which has an extensive input/output section that meets requirements, yet does not have an overdesigned Central Processing Unit (CPU) with respect to the Standard Planetary Spacecraft demands. This computer will be briefly described.

The ADAPT 244 is a product of AiResearch Manufacturing Company. The antecedent, an air control computer for the Grumman F-14A, is in production. It utilizes MOS-LSI components; however, the MOS chips used are of relatively early design, with comparatively higher power requirements than those needed with newer MOS chips. The basic characteristics are as follows:

Size	(1)	0.24 ft^3	(not including memory)
	(2)	0.5 ft^3	(with 8K memory)
Weight	(1)	13 lb	(not including memory)
	(2)	18 lb	(with 8K memory)
Power	(1)	45 w	(not including memory)
	(2)	65 w ave.	(with 8K memory)

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Hold registers	7 4 bit registers and 7 gates
Up/Down & Up Counters (resettable)	3 12 bit serial registers with add/ subtract logic 1 12 bit serial register with add logic
Sample & Hold Bilevel Data	111 gates & 12 bit register
Sequence Logic	12 bit register 128 gates
Power Control	15 gates
Timing Reference	Master Reference - 1 part in 10^6 - assume 16 bit register + 30 gates
Analog subcommutate	224 gates (may be included in Computer Section)
Select Logic	25 gates
Packaging & Logic	38 MSI ckts/board + mother board, TTL MSI

Fig. 14 Spacecraft Subsystem Interface Unit Components

Architecture	As per Fig.15 (Derived from Ref. 4)
Memory Type	Choice of solid state,ferrite core plated wire or flux ring (core presently being used)
Memory Capacity	8K 20 bit words
Word Structure	Variable instruction word: Nominally 6 bit op code, 2 bit index register, 12 bit address (direct addressing for 4K words) Operand - variable 4 to 20 bits in 4 bit bytes
Clock	500 MHz - May be increased with attendant reduction in memory cycle and instruction execution times
Instructions	52 micro programmable instructions with read only memory. Includes arith., logical, transfer, index, I/O and special.
Speed	12 microseconds add; 114 microseconds multiply with 500 KHz clock. 6.7 microseconds add; 89 microseconds multiply with 750 KHz clock.
Index Registers	4
Addressing	Direct, Relative and indirect
Data Bus	Parallel - accommodate 20-bit word. May be used for DMA.
Input/Output	As per Fig.16 (derived from Ref. 4)

The computer architecture, as shown in Fig. 15 uses a parallel data bus in the main inter-computer communication link. This affords considerable flexibility and should permit utilization of low-level redundancy in future design. The same data bus serves the input/output section, thereby enabling the utilization of DMA.

The Input/Output Unit shown in Fig. 16 is configured to serve as a standard interface entity for the computer; it is very well suited for the Standard Planetary Spacecraft in that it includes the multiple input/output registers, multiplexers, A/D converters, etc., needed by the system. In addition, clock references and other capabilities are available; these may allow for greater simplification of the Interface Section than that provided in the design discussed in par. 3.2.

Cost estimates are indefinite at the present time; however, some ROM values have been provided by Mr. H. Bellamy of the Garrett Corp., Ref. 5. The costs indicated are as follows:

ADAPT 244	$ 20,000
Memory	10,000
Total	$ 30,000 per computer

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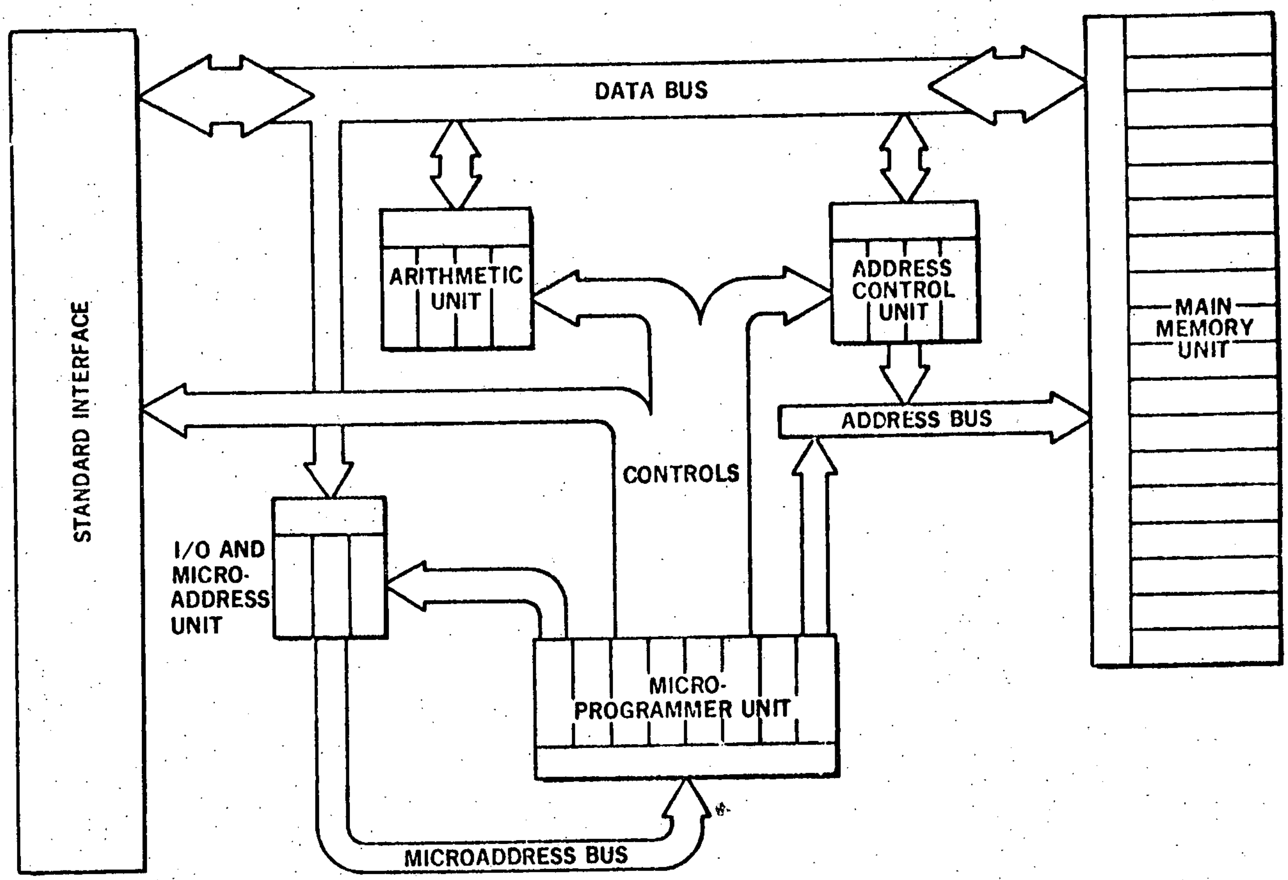
ARITHMETIC UNIT modules manipulate the data associated with the program instructions, with the ability to solve arithmetic and logic equations. Each module contains a four-bit slice of an arithmetic unit, and contains three registers, a fast carry full-adder, and a logic network.

ADDRESS CONTROL UNIT modules interpret the direct, indirect, relative addresses contained in the program instructions, and coordinate the exchange of data between memory, arithmetic, and data interface units. Each module is a three bit slice of the address control unit, and contains an instruction register, a fast carry full-adder, four index registers, and an instruction counter.

MICROPROGRAMMER, I/O, and MICRO ADDRESS units use read only memory modules to provide the elemental control signals for the entire computer. This technique dispenses with the traditional approach to computer design that requires the use of complex, random, combinatorial logic networks. Microprograms can be custom tailored.

MAIN MEMORY UNIT can be integral or remote. In either case the memory subsystem can employ any proven fast access memory technique, including ferrite core, plated wire, flux ring, and solid-state bipolar or MOS devices, in both read-write and read-only modes.



The DATA BUS provides a bidirectional communications path between the memory, arithmetic, address control, and data interface units. The bus is sufficiently wide to allow the transfer of instructions and operands in parallel form, thus minimizing transfer time and optimizing computer speed. Use of the bus is under control of the computer programs, and information is exchanged between units in as little as 200 nanoseconds.

The MICROADDRESS BUS provides closed-loop control to maintain the operation of the self-addressed microprogrammer. The loop is opened in the microaddress unit by each new program instruction, to define the starting address for the next series of microaddresses required for instruction execution. Each microaddress defines a specific storage location in the microprogrammer, which will issue control signals and provide the next microaddress to complete the feedback loop.

The ADDRESS BUS carries information that directs memory unit operation. Memory addresses are generated by the address control unit during normal program execution, and by the microprogrammer under interrupt conditions. Each memory address defines a specific storage location that will furnish or accept information words during the current instruction. Addressing technique allows completely random word selection.

The CONTROLS generated by the microprogrammer are distributed throughout the computer to coordinate the simultaneous operation of the modules. The control signals are simple digital bit patterns that either enable or disable their assigned function. The control signals are standardized to simplify and optimize the module designs, and the microprogrammer bit patterns are designed to create the desired machine instruction repertoire.

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Fig. 15 The ADAPT Architecture

(from Reference 4)

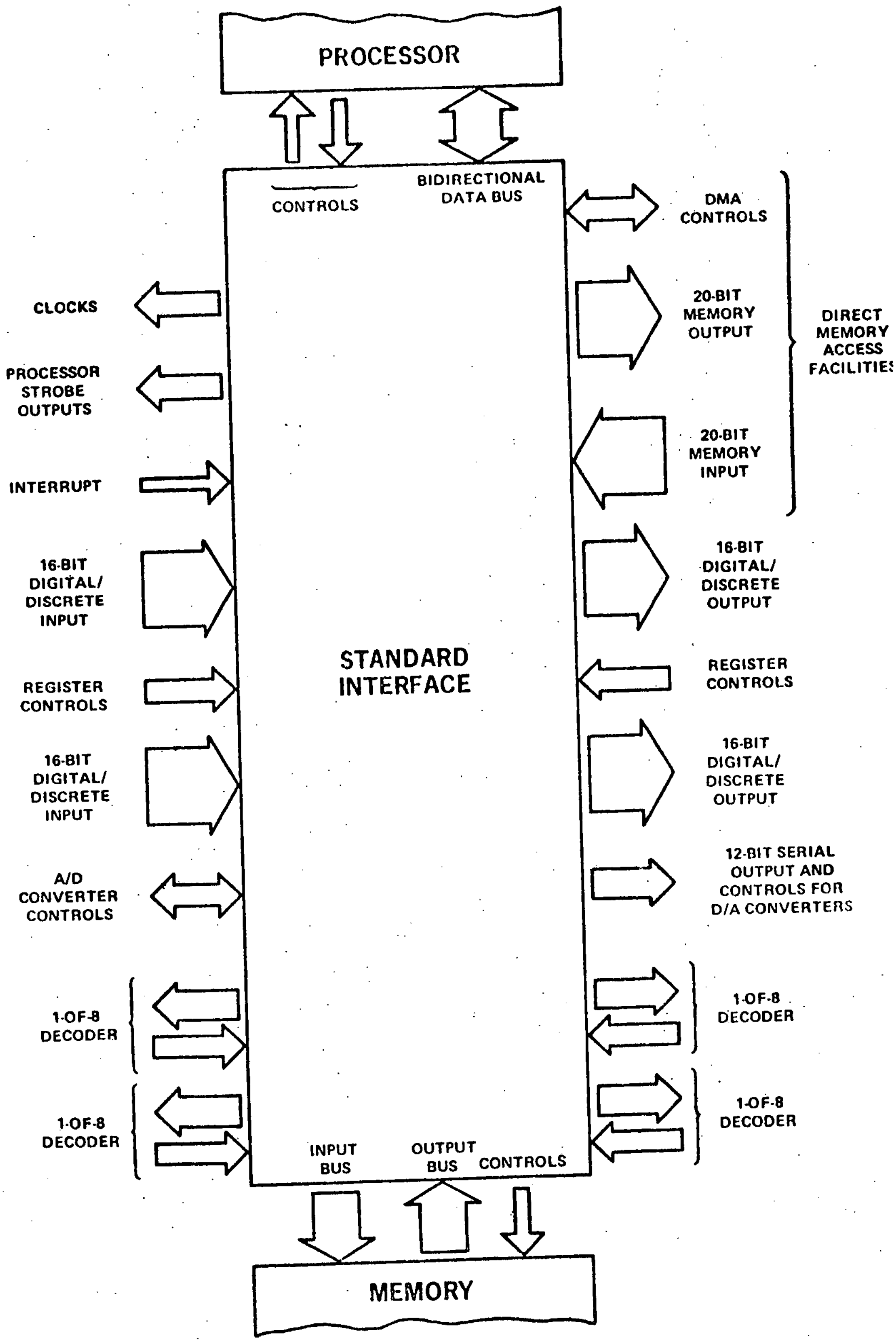

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Fig. 16 Input/Output Reference

(from Reference 4)

Costs for peripheral support equipment have not been identified, however, estimates would be as follows:

Programmer Console	$10,000
Program Loader	12,000
	$22,000

Software mechanization has not been determined; however, extrapolating on the basis of memory capacity, then

basic software cost	$170,000
validation expense (60% basic cost)	102,000
	$272,000

3.4 CDPI Subsystem Module

All of the CDPI subsystem equipment except the antennas and associated components are packaged in a single standard module as shown in Fig. 17. The module is designed to facilitate its removal and replacement, if required, during checkout of the spacecraft in low-earth orbit prior to dispatch of the spacecraft to its planetary destination. The module is guided into its location on the spacecraft by rails, and aligned and supported by two inboard pins and two outboard cams that engage machined grooves in the rails. The cams also transmit force from the cam actuator handles on the outboard face of the module, to accomplish the controlled engagement and disengagement of the bulkhead-type electrical connectors on the in-board face of the module.

4.0 Conclusions and Recommendations

The requirements, design rationale and representative design for a Standard Planetary Spacecraft CDPI have been presented. The major change in this spacecraft compared to others considered in the Payload Effects Follow-On Study, is that Space Shuttle/Space Tug revisit capability would not be available for maintenance services or refurbishment. This changed the failure correction mode from fail-safe to fail operational. Thus, reliability requirements are increased; particularly with respect to the mission definition. This change had important influences on the CDPI, including the following:

- Simpler Interface Section
- Expanded Data Processing capability, particularly the input/output unit
- More extensive software for pre-injection checkout and data monitoring evaluation

The mission requirements also differ from prior designs, e.g., communication with a planet-based Lander, as well as the DSN, data storage of mission equipment and Lander data for delayed communication to the DSN, reduced transmitted data rates, etc. These changes have resulted in significant alterations in the design details of the CDPI compared to earlier subsystem CDPI configurations, tending to necessitate more variants in provisioning to meet standardization requirements.

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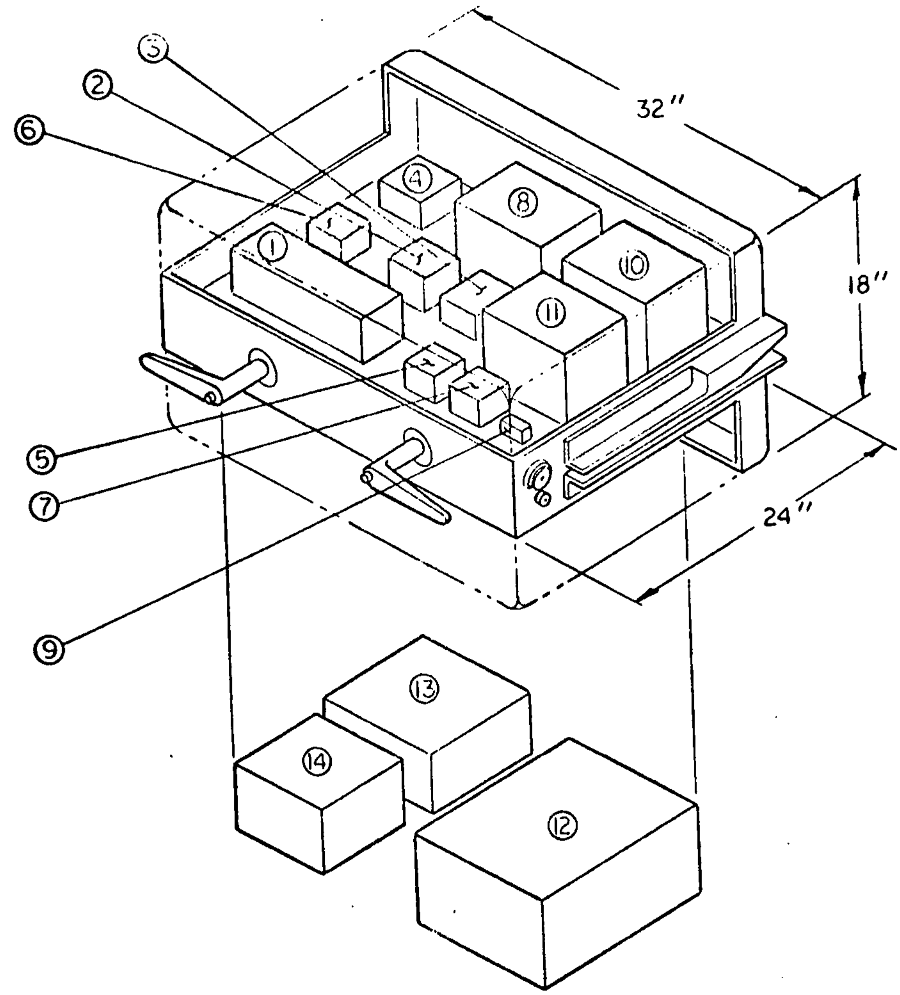

Equipment	Weight (lbs)
1. S-Band TWTA	12
2. S-Band Mod/Driver	3
3. S-Band PLL Receiver	3
4. S-Band Multicoupler	3
5. S-Band Receiver	3
6. Delay Lock Loop Correlator	2
7. UHF Receiver/Demodulator	3
8. Antenna Servo Electronics	8
9. UHF Coupler	0.5

Equipment	Weight (lbs)
10. Tape Transport	10.5
11. Tape Transport Electronics	10
12. Digital Computer	18
13. Mission Equipment Interface Unit	5.5
14. Spacecraft Interface Unit	4.5
Cables and Connectors	7
Module Base & Covers	36
Subtotal	129
15% contingency	20
Total	149 lbs

Fig. 17 CDPI Subsystem Module - Standard Planetary Spacecraft

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Examination of requirements and design mechanization have indicated a need for further study. This included the following:

- Investigation of ranging techniques, e.g., pseudo noise codes
- Tradeoff study between Interface Section and Computer Input/Output
- Fault isolation hardware and software

It is recommended that these studies be carried out.

The design developed in this memo represents a first cut at the CDPI configuration. A more detailed examination, particularly of the Communication Section is suggested.

5.0 References

1. "Viking On Mars", Astronautics & Aeronautics, AIAA, dtd November 1969.

2. "Mariner Mars 1971"; JPL Report, NASA; dtd January 1970.

3. "Standard Large Astronomical Observatory Satellite", EM PE-146, F. Bolton, M. Loeb, R. Pollak, et al; LMSC, dtd 31 March 1972.

4. "MOS-LSI General Purpose Computers" - The Adapt Series"; Electronic Systems - AiResearch Manufacturing Company, a Division of the Garrett Corp., undated

5. Telephone Conversation with Howard Bellamy - Garrett Corporation, Dec. 1971.

PE-137

LOW-COST VIKING

VELOCITY ADJUST PROPULSION SYSTEM

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: LOW COST VIKING VELOCITY ADJUST PROPULSION SYSTEM	EM NO: PE-137 REF: DATE: 31 March 1972
AUTHORS: Dave Burkholder, 62-13 R. J. Pollak, 62-11 Prepared Under Cogn. of Orgn 69-02, Advanced Payload Systems	APPROVAL: J C Bolton Wayne J. Muller

20 pages

PRELIMINARY

Summary

The Viking Propulsion system performs the velocity adjust functions from the time of Space Tug Separation, after TMI, to completion of the Mars Orbiter Mission. Using primarily existing parts, a pressure-fed, earth-storable system has been selected. It has a 3500-lbf thrust engine and 15,500 lb of 309 sec I_{sp} propellant in four identical spherical titanium tanks pressurized to 184 ± 4 psia. The system is described in Table 3.

Requirements

The velocity-adjust propulsion system provides the impulse for retro from the earth-Mars transfer trajectory to the 1400 x 33000 Km elliptical Mars operating-orbit and up to three short midcourse and four orbit trim/synchronization burns.

The VA subsystem is required to produce a total of 2750 meters per second velocity change along the thrust (X) axis of the orbiter. The velocity breakdown is as follows:

Function	ΔV m/sec	ΔV ft/sec
Earth-Mars Midcourse Maneuvers (3)	25	82
[1]Mars Orbit Injection (1)	2500	8200
Mars Orbit Adjust (4)	225	738
Total	2750	9020

Appendix B shows, in detail, the required propellant loadings based on engine delivered specific impulse (I_{sp}) of 309 sec and a vehicle weight (exclusive of propellant) of 10,000 lb. The total propellant requirement is 15,500 lb which includes a 5% contingency factor. This propellant load corresponds to a total impulse of 4.7895×10^6 lb-sec.

For an ignition weight of 25,500 lb, it is desirable to have a velocity adjust engine thrust of 2500 lb, or more, to minimize velocity losses due to Mars gravity and, at the same time, avoid thrust attitude programming.

(1) See Appendix A for derivation.

The total impulse requirements and thrust requirements of the velocity adjust system for this vehicle happen to be quite similar to those of one Orbit Maneuvering System (OMS) module on the proposed Space Shuttle orbiter. The total impulse of one OMS module is approximately 3.3×10^6 lb-sec (~ 30% less than VA system) and the thrust level requirement is 3500 lb (or more), therefore much of the OMS hardware can be utilized for the VA system.

System Description

The candidate concepts considered for Shuttle OMS include earth-storable propellant, pressure-fed engines using Nitrogen Tetroxide (NTO) oxidizer and Hydrazine blends as the fuel; pump-fed systems (such as a Modified Agena Engine, the Bell 8096B) using NTO and MMH with Silicone Oil added; as well as various earth/space storable propellant concepts such as LOX/Propane, LOX/RP-1, LOX/LH_2, and Monopropellant Hydrazine. The only concepts that will be considered for the VA system will be the earth storable propellant concepts.

A tabulation of engine characteristics is shown in Table 1 for the candidates for this application. Two of these engine concepts are described in more detail in this report. These engines are the modified LMA engine used in the ascent stage of the Lunar Module (example of a pressure-fed system) and a modification of the Agena engine (example of a pump-fed system). The LMA engine provides a good thrust level and a low system weight. The Agena engine is higher in thrust but still produces the lightest overall system weight. The burn life of the LMA (RS1801) must be increased from the 935 sec demonstrated to the 1368 sec required life for this application. Most of the other engines would meet the life requirements however.

System flow schematics for both the pressure fed and pump-fed systems are shown in Figs. 1 and 2. Two pressurization concepts are shown, one uses regulators to control the propellant tank pressure (shown in Fig. 1) and the other uses a bang-bang valve (with control orifices) controlled by a control unit (shown in Fig. 2). Either of these pressurization concepts could be utilized with either the pressure-fed or pump-fed engines.

Redundancy (except for thrust chamber and tankage) is included to provide a fail-operational system. Parallel propellant tankage was selected to reduce the overall length of the VA system and still stay within the 14-ft diameter envelope. Propellant acquisition devices are located in each propellant tank and are sized to provide sufficient gas free propellant to the engine until the bulk of the propellant is settled at the tank outlets. Isolation valves are located at each tank outlet so that if the two tanks in parallel do not drain simultaneously, that tank which would deplete first is isolated from the engine just prior to depletion, thus preventing a gas bubble from entering the propellant feed system. Separate pressurization systems were selected for each of the propellants to preclude the possibility of the two propellants from coming in contact with each other via the pressurization system during the long mission duration.

Table 2 shows the system weight breakdown and tankage sizes for the two concepts considered. Both concepts weigh about the same.

General Comments

The thrust level of most pump-fed engines is greater than 10,000 lb_f and probably are too large for this application, since the VA system must perform midcourse corrections which could be small ΔV changes. A pressure-fed engine with a thrust level considerably smaller than 10,000 lb_f is more likely to be able to provide these small ΔV changes. If not, then a small thrust (100 lb_f or 300 lb_f) engine could be installed in the VA to fulfill this requirement.

Attitude control in pitch and yaw during main engine burn can be accomplished by gimballing the engine. All engines considered in Table 1 either have gimbal rings or can be modified to incorporate them. The chosen VA system's equipment is listed in Table 3. The cost of the VA system is minimized by usage of the Shuttle OMS module hardware.

Table 1

CANDIDATE ROCKET ENGINES FOR VELOCITY ADJUST SYSTEM

		Pump Fed	Pressure Fed					
Company		Bell Aero.	Aerojet	Aerojet	Aerojet	Rocketdyne	TRW	Aerojet
Model		8096B	AJ10-137	AJ10-138	AJ10-118F	RS1801F	MIRA-10K	AJ10-131
Application		Improved Agena	Apollo SPS	Transtage	Delta F	LEMA	LEMDE	Apollo Sub-Scale
Ox/Fuel		N_2O_4/MMH+SO	N_2O_4/A-50	N_2O_4/A-50	N_2O_4/A-50	N_2O_4/A-50	N_2O_4/A-50	
O/F Ratio		1.82	1.6	2.0	1.9	1.6	1.6	2.0
F_N vac	lbf	16,000	20,500	8,000	9,600	3,500	9,850	2,200
I_s vac	sec	310.8	311	305	306	309	304	309
ϵ		45	62.5	40	40	45.6	47.5	60
P_c	psi	500	99	105	125	119	104	100
Supply Press.	psi	26.0/14.5	160/169	155/155	225/225	170/170	223/223	
Length	in	80	160	81.6	82.3	57.8	90.5	
Diameter	in	32.5	100	48.2	47.5	31.0	59.0	
Weight	lbm	305	823	232	184	180	326	98
*Mission Req.	sec	299	234	599	499	1368	486	2177
Demo Life	sec	3600	750	500	500	935	1000	1024

* Based on 9020 ft/sec or 4.7895×10^6 lb-sec total impulse.

Table 2

WEIGHT BREAKDOWN & TANK SIZE

Component	LEMA Engine	8096B Engine
Engine	180	305
Oxidizer Tanks (including Acquisition Device)	190	92
Fuel Tanks (including Acquisition Device)	190	80
He Tanks (2)	376	74
Lines, valves, etc.	135	175
Structure	100	115
Contingency (10% except engines)	100	54
Total Dry Weight	1271	895
Loaded Oxidizer	9634	10105
Loaded Fuel	6022	5552
Loaded He	37	8
Total Loaded Wt.	16964	16560
Orbiter Dry Weight	6400	6400
Propulsion inert weight	- 1464	- 1060
Orbiter Dry Weight (excluding Propulsion)	4936	5340
Oxidizer Tank Size	2@ 57" Dia	2@ 56.2" Dia x 60.1" Long
Fuel Tank Size	2@ 57" Dia	2@ 56.2" Dia
He sphere for Oxidizer	1@ 29.5" Dia	1@ 19.4" Dia
He sphere for fuel	1@ 29.5" Dia	1@ 14.9" Dia

Notes: 1) See Appendix C for determination of tank size and weight

2) See Appendix B for determination of Propellant requirements

Table 3

VIKING VELOCITY ADJUST PROPULSION SYSTEM EQUIPMENT LIST

Name	Description	Unit Weight (lb)	Envelope (in)	Unit ROM Cost ($K)
Velocity Adjust Engine (F = 3500 LBF)	1 - Rocketdyne RS 1801E (LEM Ascent)	180	58L x 31D	300 (mod)
Fuel Tank (6022 lb A-50)	2 - 57" I.D., 0.053" t, 6 AL 4V Ti Sphere	85	58" D	55 (new)
Oxidizer Tank (9634 lb NTO)	2 - 57" I.D., 0.053" t, 6 AL 4V Ti Sphere	85	58" D	55 (new)
Propellant Acquisition Device	4 - Internal, Surface Tension screen assemblies	10	-	negl. (new)
Helium Tank, Fuel Tank Pressurization (18.7 lb He @ 4000 psia)	2 - 29.5" I.D., 6 AL 4V Ti Sphere	188	30" D	8.5 (new)
Structural Supports and Brackets	-	100	-	- (new)
Fill Valve	4		← 3 x 3 x 4 typical →	.50 (mod)
Filter	4			negl. (mod)
Shut-off Valve	8			1.2 (mod)
Pressure Regulator	4			2.5 (mod)
Check Valve	8			.15 (mod)
Vent Valve	2			.50 (mod)

Table 3 (Cont.)

VIKING VELOCITY ADJUST PROPULSION SYSTEM EQUIPMENT LIST

Name	Description	Unit Weight (lb)	Envelope (in)	Unit ROM Cost ($K)
Relief Valve	2	135 total	3 x 3 x 4 Typical	Negl. (mod)
Burst Disc	2			1.2 (mod)
Isolation Valve	12			
Pressure Transducer	4			Negl. (mod)
Temperature Transducer	6			Negl. (mod)
Pressure Switch	4			Negl. (new)
Gas/Liquid Lines	-			Negl. (new)

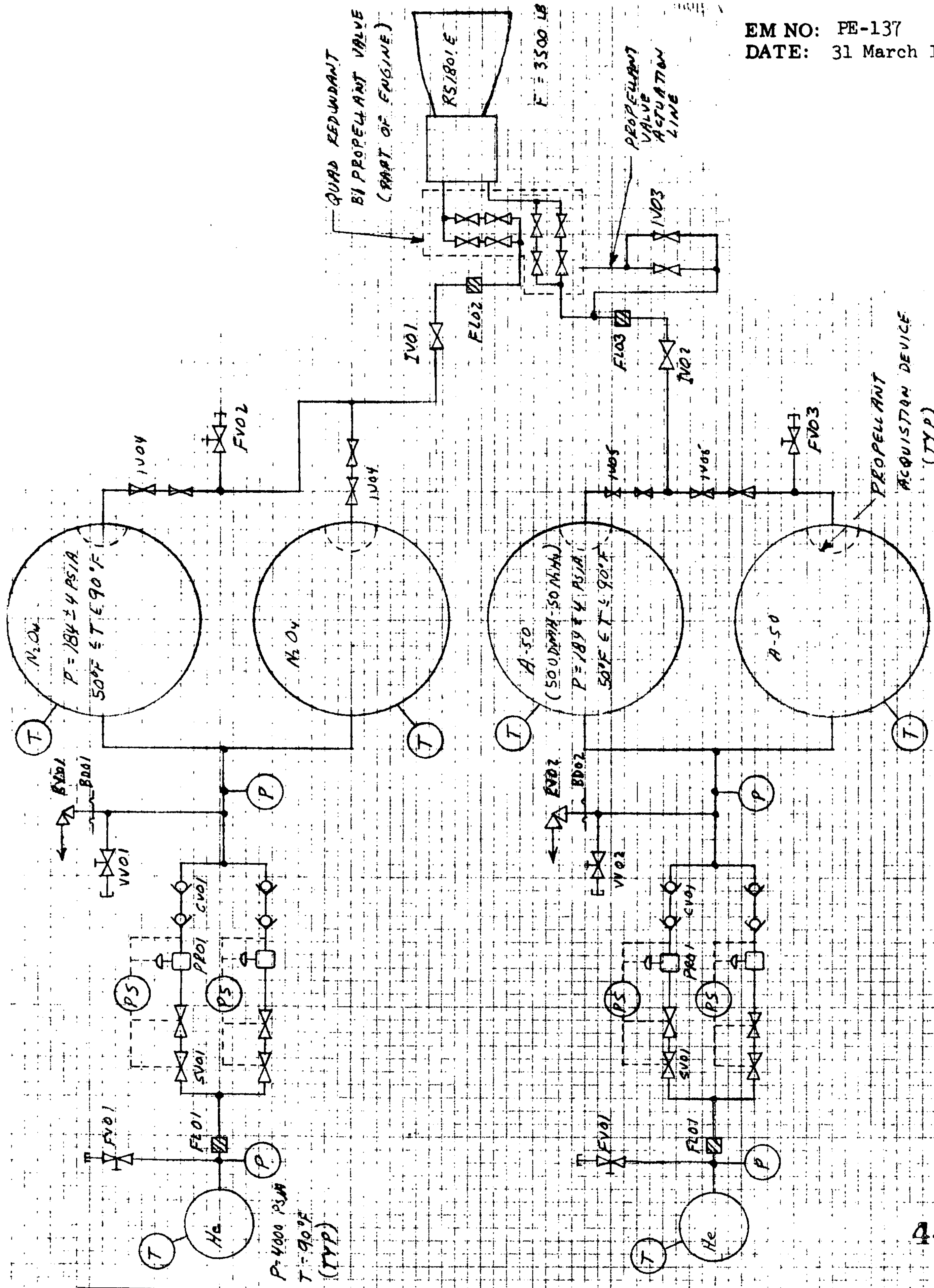

Fig. 1 Velocity Adjust Propulsion System Flow Schematic (Using a LEMA(RS1801E) Engine)
(Typical of a Pressure Fed System)

EM NO: PE-137
DATE: 31 March 1972

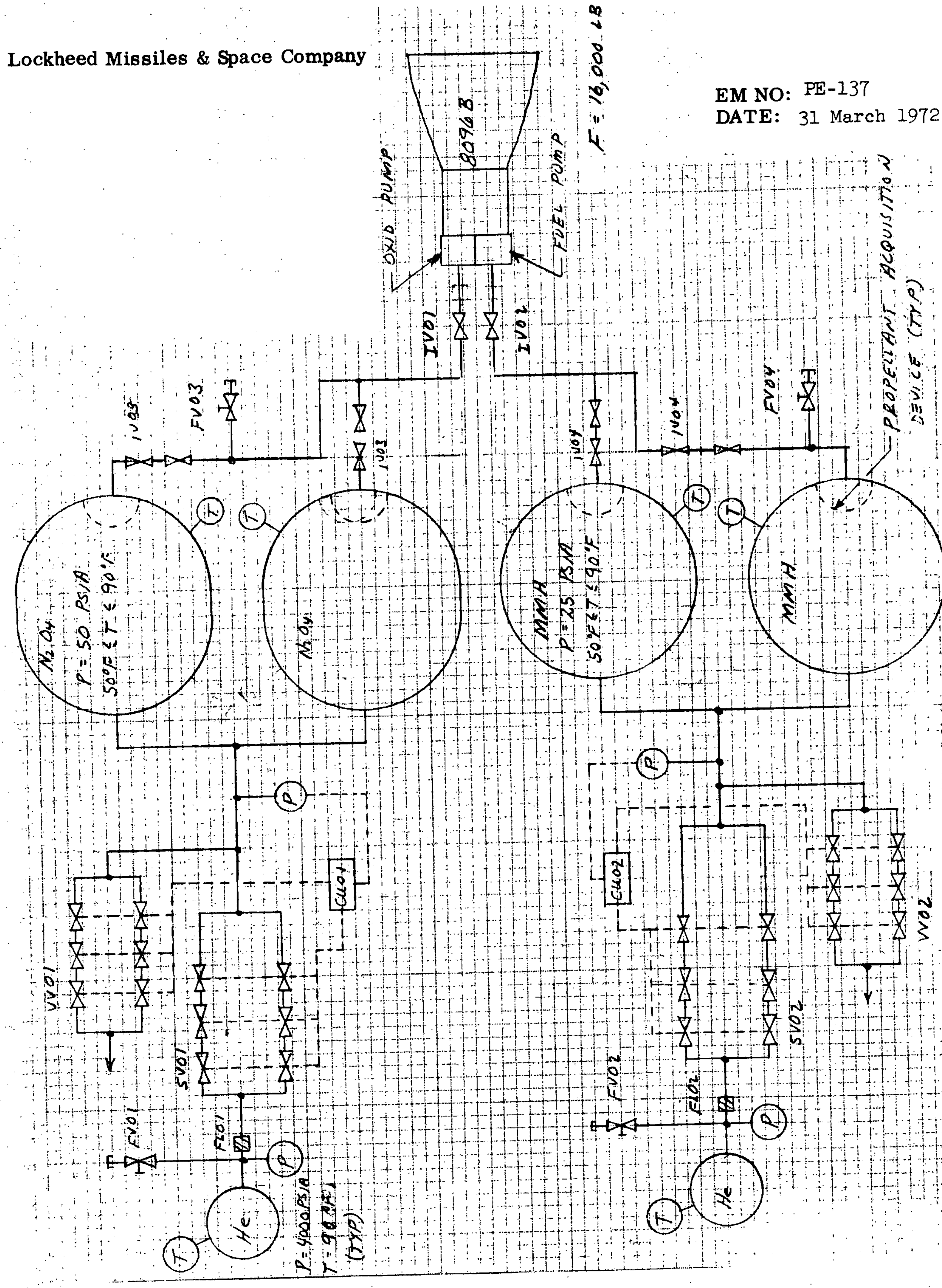

Fig. 2 Velocity Adjust Propulsion System Flow Schematic (Using a Bell 8096 Engine)
(Typical of a Pump Fed System)

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Nomenclature Used on Schematic:

P	Pressure Transducer
T	Temperature Transducer
PS	Pressure Switch
	Filter
FVXX	Fill Valve
FLXX	Filter
SVXX	Shut-off valve
PRXX	Pressure regulator
CVXX	Check valve
VVXX	Vent Valve
RVXX	Relief Valve
BDXX	Burst Disc
IVXX	Isolation Valve
CUXX	Control Unit

APPENDIX A

DERIVATION OF MARS RETRO ΔV REQUIREMENT

For a 30-day launch window in 1984:

$V_{approach}$ = 14 to 18 KFPS, from Page II-34 (Ref 1)

R_{MARS} = 1830 nm

H_{PER} = 1400 KM = 750 nm = 0.41 R_{MARS}

H_{AP} = 33000 KM = 17,800 nm

V_{ESC} @1.41R = 13800 FPS from Page II - 39, (Ref 1)

$$\frac{V_{APPR}}{V_{ESC}} = \frac{16,000}{13,800} = 1.16$$

$$\frac{R_{AP}}{R_{PER}} = \frac{17,800 + 1830}{750 + 1830} = 7.6$$

From P II - 40, (Ref 1) $\dfrac{\Delta V_{RETRO}}{V_{ESC}} = 0.69$

$\Rightarrow \Delta V_{RETRO} = \underline{\underline{8150}}$ FPS = $\underline{\underline{2500}}$ m/s

References:

1. Launch Vehicle Estimating Factors; NASA NHB 7100.5; Jan, 71.

2. Aeronautics & Astronautics; Nov 1969, pp 30-59.

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APPENDIX B

Derivation of Viking Orbiter Propulsion Impulse Requirements:

ΔV Requirement	ΔV m/sec	ΔV ft/sec
• Launch date 1980		
• Injection into TransMars Trajectory from earth parking orbit: By Space Tug		
• Earth Mars Transit Time: 190-220 days		
1) 3 Mid-course Maneuvers:	25	82
2) Mars Orbit Injection	2500	8200
3) Mars orbit correction	225	738
Total ΔV	2750	9020

Vehicle Weights

Total lander wt (loaded)	3600 lb
Dry orbiter wt	6400 lb
Viking Total (Orbiter Dry)	10,000 lb

It is assumed that the orbiter dry wt. of 6400 lb includes the inert wt of the orbiter propulsion system such as dry wt., helium pressurant, and trapped propellant.

An I_{sp} = 309 sec will be used as a first estimate for propellant requirements (applicable to the Rocketdyne RS 1801F)

APPENDIX B (CONT.)

$$\Delta V = I_{sp}\, g_c \ln \frac{W_o}{W_o - W_p}$$

$\Delta V = 9020$ ft/sec

$I_{sp} = 309$ sec

$g_c = 32.2$ ft/sec^2

$W_o - W_p = 10{,}000$ lb

$$\frac{W_o}{W_o - W_p} = e^{\Delta V / I_{sp} g_c}$$

$$W_o = (W_o - W_p)\, e^{\Delta V / I_{sp} g_c} = (10{,}000)\, e^{(9020)/(309)(32.2)} = 10{,}000\, e^{.90655}$$

$W_o = (10{,}000)(2.475) = 24{,}750$

$W_p = W_o - (W_o - W_p) = 24{,}750 - 10{,}000 = 14{,}750$ lb. For a 5% contingency.

$W_p = (1.05)(14750) = 15487$ Use 15,500 lb.

$I_t = (309)(15{,}550) = 4.7895 \times 10^6$ lb sec for a single engine system

required burn time $t_B = \dfrac{4.7895 \times 10^6}{F}$

for the RS1801F: $F = 3500$ lb

$$t_B = \frac{4.7895 \times 10^6}{3.5 \times 10^3} = 1368 \text{ sec}$$

See Table 1 for required burn times for other candidate engines.

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APPENDIX C

TANK SIZING & WT ESTIMATES

Pressure-Fed System

From Appendix B

W_p = 15,500 lb

Use the following assumptions and Design Criteria

1) Oxidizer: N_2O_4 - Nitrogen Tetroxide (NTO)
2) Fuel: A-50
3) Engine Mixture ratio = 1.6
4) Load the tanks so that there is a 3% ullage at maximum expected temp
5) Max temperature = $90^\circ F$
6) Min temperature = $50^\circ F$
7) Expulsion efficiency = 99%
8) Safety Factor = 2.0

Wt of required NTO = $\left(\frac{1.6}{2.6}\right)(15500) = 9538$

Wt of required A-50 = $\left(\frac{1}{2.6}\right)(15500) = 5962$

ρ_{NTO} @ $90^\circ F = 88.5\ lb/ft^3$

$\rho_{A\text{-}50}$ @ $90^\circ F = 55.7\ lb/ft^3$

Vol of NTO Tank(s) = $\frac{(9538)}{(.99)(.97)} \frac{1}{(88.5)} = 112.2\ ft^3$ Use 112 ft^3

Vol of A-50 Tank(s) = $\frac{(5962)}{(.99)(.97)(55.7)} = 111.4\ ft^3$ Use 112 ft^3

NTO & A-50 tanks will be equal size

Wt of NTO loaded = $\frac{9538}{.99} = 9634$ i.e. 96 lb trapped

Wt of A-50 loaded = $\frac{5962}{.99} = 6022$ i.e. 60 lb trapped

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APPENDIX C (CONT.)

for 2 tanks; 1 NTO & 1 A-50

$$D = \frac{(6)(112)}{\pi}^{1/3} = (2139)^{1/3} = 5.97 \text{ ft} \simeq 72'' \text{ in dia}$$

If 4 tanks; 2 NTO + 2 A-50

$$D = \frac{72}{(2)^{1/3}} = 57'' \text{ dia}$$

Use 4 - 57" dia tanks   2 NTO & 2 A-50

For thrust levels greater than 2500 lb but less than 10,000 lb most, if not all, existing engines are pressure-fed with the associated high tank pressure. However, if larger thrust levels can be tolerated, pump-fed engines permit lower tank pressure with the leading candidate being the Bell 8096B (Modified Agena engine).

These two concepts will be considered:

For the LEMA Engine - P_e = 119 psia
P_{inlet} = 170 psia
$P_{TK\ DES}$ = 250 psia

$$\text{Wt of Tank} = \frac{(NOF)(\pi)(S)(\rho/_{Ftu})D^3\ P_{OES}}{4} \quad \text{for spherical Tank}$$

Where NOF = non optimum factor = 1.2
S = Safety factor = 2.0
ρ = density = 0.16 lb/in^2 for 6 Al 4V Ti
F_{tu} = Ultimate strength = 165,000 lb/in^2 for 6 Al 4V Ti
D = diameter = 57 in
P_{DES} = design pressure = 250 lb/in^2

$$Wt = \frac{(1.2)(\pi)(2.0)(.16)(57)^3(250)}{(4)(165000)} = 85 \text{ lb ea.}$$

Add 10 lb per tank for acquisition device

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APPENDIX C (CONT.)

He and He tankage

For the LEMA engine the tank regulated pressure will be 184 ± 4 lb/in^2

Added assumptions: 10% margin

Minimum temperature of propellant tank = $50^{\circ}F$

$$\therefore \text{density of helium at end of mission @ } \left.\begin{array}{l} T = 50^{\circ}F = 513^{\circ}R \\ P = 184\ lb/in^2 \end{array}\right\} \rho_{f_{TK}} = 0.137\ lb/ft^3$$

Min temperature of He in sphere = $0^{\circ}F$

$\therefore$ density of helium in press. tank @ end of mission

@ $T = 0^{\circ}F = 460^{\circ}R$

$P = 300$ psia (P_{TF} + 116 psia ΔP)

$$\rho_{fsp} = 0.2457\ lb/ft^3$$

He loading condition $\left.\begin{array}{l} T = 70^{\circ}F \\ P = 4000\ lb/in^2 \end{array}\right)$ $\rho_i = 2.41\ lb/ft^3$ for the NTO side

$$V_{Hesp} = \frac{(V_{NTO\ TK})(1.10)(\rho_{fTK})}{(\rho_i - \rho_{fSP})}$$

$$V_{NTOTK} = 112\ ft^3$$

$$V_{HeSP} = \frac{(112)(1.10)(0.137)}{(2.41 - .2457)} = 7.80\ ft^3;\ wt_{H_e} = (2.41)(7.8) = 18.7\ lb$$

$$D = \left(\frac{(6)(7.8)}{\pi}\right)^{1/3} = (14.90)^{1/3} = 2.46\ ft = 29.5\ in$$

for the A-50 side - Tank size & pressures are the same

1 He TK @ 29.5 in
Wt of He for A-50 = 18.7 lb

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APPENDIX C (CONT.)

He sphere wts.

$$wt = \frac{(NOF)(\pi)(S)(\rho/Ftu)(D^3)(P_{DES})}{4}$$

where $P_{DES} = 4000$ lb/m^2
$D = 27.5$ in

$$Wt = \frac{(1.2)(\pi)(2.0)(.16)(29.5)^3(4000)}{(4)(165,000)} = 188 \text{ lb ea}$$

PUMP-FED SYSTEM

If the Agena Engine were used (Bell 8096B):

$I_{sp} = 310.8$ sec

Since this is close enough to 309 sec used for Wp calculations

$W_{P_R} = 15,500$ lb (same as before)

MR = 1.82

ρ_{NTO} @ 90°F = 88.5 lb/ft^3

ρ_{MMH} @ 90°F = (.86)(62.4) = 53.7 lb/ft^3

Wt of NTO req'd = $(\frac{1.82}{2.82})(15500) = 10,004$; Wt_{NTO} loaded = $\frac{10004}{.99} = 10105$
i.e. 101 lb trapped

Wt of MMH req'd = $(\frac{1}{2.82})(15500) = 5496$; Wt MMH loaded = $\frac{5496}{.99} = 5552$
i.e. 56 lb trapped

$$\text{Vol of NTO Tank(s)} = \frac{(10,004)}{(.99)(.97)(88.5)} = 117.7 \text{ ft}^3$$

$$\text{Vol of MMH Tank(s)} = \frac{(5496)}{(.99)(.97)(53.7)} = 106.6 \text{ ft}^3$$

APPENDIX C (CONT.)

These tanks could be equal volume (117.7 ft^3) or the MMH could use 2 spherical tanks (V_{TOT} = 106.6 ft^3) and use 2 NTO tanks at this diameter and add a cylindrical section (V_{TOT} = 117.7 ft^3), the latter approach will be used for MMH tanks

$$\text{Vol per tank} = \frac{106.6}{2} = 53.3 \text{ ft}^3 \text{ ea.}$$

$$D = \left(\frac{(6)(53.3)}{\pi}\right)^{1/3} = (102)^{1/3} = 4.68 \text{ ft} = 56.2''$$

for NTO

$$\text{Vol per tank} = 117.7 \text{ ft}^3/2 = 58.85 \text{ ft}^3$$

$$V_{cyl\ sect} = 58.85 - 53.30 = 5.55 \text{ ft}^3$$

$$L_c = \frac{4Vc}{\pi D^2} = \frac{(4)(5.55)}{(\pi)(4.68)^2} = 0.323 \text{ ft} = 3.9 \text{ in}$$

$$L_{overall} = 56.2 + 3.9 = 60.1 \text{ in}$$

Propellant Tank Wt for 8096B system

P_{DES} for NTO tank = 60 lb/in^2

P_{DES} for MMH Tank = 35 lb/in^2

for MMH Tank

using Al 2219 - T87 $\qquad \rho/F_{tu} = 1.65 \times 10^{-6}\ in^{-1}$

$$wt = (1.885)(\rho/F_{tu})\ D^3\ P_{DES} = (1.885)(1.65 \times 10^{-6})(56.2)^3(35) = 19 \text{ lb}$$

If min gauge (.025") ρ = 0.10 lb/in^3 $\qquad$ Add 10 lb for acquisition device

$$wt = (NOF)\pi D^2\ \rho t = (1.2)(\pi)(56.2)^2(.10)(.025) = 29.8 \text{ lb ea for MMH}$$
if min gauge Al

if min gauge Ti @ t_{min} = .020" $\quad wt = (29.8)(\frac{.16}{.10})(\frac{.020}{.025}) = 38.1$ lb, use Al

APPENDIX C (CONT.)

for NTO Tank

$D = 56.2''$

$L_c = 3.9''$

$P_{DES} = 60 \text{ lb/in}^2$

for Al $\quad \rho/F_{tu} = 1.65 \times 10^{-6} \text{ in}^{-1}$

$$wt = (1.885)(\rho/F_{tu})(D^3)(P_{DES}) + (3.770)(\rho/F_{tu})\, D^2\, L_c\, P_{DES}$$

$$wt = (1.885)(1.65 \times 10^{-6})(56.2)^3(60) + (3.77)(1.65 \times 10^{-6})(56.2)^2(3.9)(60)$$

$$wt = 33.1 + 2.6 = 35.7 \text{ lb}$$

This is above min gauge since sphere weighs 33.1 lb and a min gauge sphere would weigh 29.8 lb

Use Al, wt = 35.7 lb ea for NTO
Add 10 lb for acquisition device

He & He Tankage for Bell 8096B Engine

for Bell 8096B engine regulated pressurization :
for NTO tank Preg = 50 lb/in^2
for MMH tank Preg = 25 lb/in^2

for NTO side He densities

@ $P = 50 \text{ lb/in}^2$ $\quad T = 50^\circ F$ $\quad \rho_{fTK} = .03724 \text{ lb/ft}^3$

@ $P = 50 + 116 = 166 \text{ lb/in}^2$ $\quad T = 0^\circ F$ $\quad \rho_{fSP} \cong .1236 \text{ lb/ft}^3$

for MMH side

@ $P = 25 \text{ lb/in}^2$ $\quad T = 50^\circ F$ $\quad \rho_{fTK} = .01938 \text{ lb/ft}^3$

@ $P = 25 + 116 = 141 \text{ lb/in}^2$ $\quad T = 0^\circ F$ $\quad \rho_{fSP} \cong .1236 \text{ lb/ft}^3$

$\rho_i = 2.41 \text{ lb/ft}^3$ all cases (4000 lb/in^2) & $90^\circ F$

APPENDIX C (CONT.)

$$V_{HeS\,P} = \frac{(V_{prop\ TK})(1.1)(\rho_{fTK})}{(\rho_i - \rho_{fSP})}$$

for NTO side

V_{NTO} TKS $= 117.7\ ft^3$

$\rho fTK = .03724\ lb/ft^3$

$\rho fSP = .1236\ lb/ft^3$

$\rho i = 2.41\ lb/ft^3$

$$V_{HeSP} = \frac{(117.7)(1.1)(.03724)}{(2.41 - .1236)} = 2.1\ ft^3$$

Wt of He loaded = (2.41)(2.1) = 5.1 lb

Wt of He sphere = (23.74)(2.1) = 50.0 lb

$$\text{Dia. of He sphere} = \left(\frac{(6)(2.1)}{\pi}\right)^{1/3} = (4.01)^{1/3} = 1.59\ ft = 19.1\ in$$

for MMH side

$V_{TKS} = 106.6\ ft^3$

$\rho_{fTK} = .01938\ lb/ft^3$

$\rho fSP = .1236\ lb/ft^3$

$\rho i = 2.41\ lb/ft^3$

$$V_{HeSP} = \frac{(106.6)(1.1)(.01938)}{(2.41 - .1236)} = 1.0\ ft^3$$

Wt of loaded He = 2.4 lb

Wt of He sphere = (23.74)(1.0) = 24 lb

$$\text{Dia of He sphere} = \left(\frac{(6)(1)}{\pi}\right)^{1/3} = (1.9099)^{1/3} = 1.24\ ft = 14.9\ in$$

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PE-146

STANDARD

LARGE ASTRONOMICAL OBSERVATORY

(General Description)

PE-146

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: STANDARD LARGE ASTRONOMICAL OBSERVATORY SPACECRAFT (LAOS)	EM NO: PE-146 REF: DATE: 31 March 1972
AUTHORS: F. C. Bolton, R. J. Pollak, M. Loeb	APPROVAL: F.C. Bolton ENGINEERING Wayne F. Miller

PRELIMINARY

93 pages

Prepared under cognizance of:
Advanced Payload Systems, 69-02, SSD.

1.0 General

The NASA Mission Model (1979-1990) includes four major astronomical observatory missions as follows:

High Energy Astronomical Observatory (HEAO)

a. To measure the location, angular dimensions, and intensity of selected X-ray and gamma ray sources.

b. To measure the flux, direction, and energy of high-energy radiation from 0.1 keV to the GeV range simultaneously for each source investigated.

Large Stellar Telescope (LST)

Diffraction-limited images from a large telescope in space can provide a significant increase in our knowledge of the spatial structure of astronomical objects and permit the detection of fainter objects than is presently possible from the ground, because of increased angular resolution. It will also allow higher spectral resolution to be achieved more efficiently by instruments employing dispersive optical systems. The advanced stellar astronomy goals in space are:

a. Improved observation of stellar objects in the 10^{-5} to 9×10^{-7} m (10,000 Å to 900 Å) spectral region by imaging, spectrometery, photometry, and polarization measurement.

b. High resolution spectrometry and imaging of planetary bodies.

c. The long term goal of observational astronomy in the short wavelength IR, visible light, and UV portions of the spectrum is to obtain an operational high-resolution large (2.54 to 3.05 m, 100 to 120 in.) diameter telescope in space by the early 1980's. Such a telescope would enable maximum state-of-art observational capabilities with minimum limitation from Earth atmosphere, sky brightness, cloud obscuration, and atmospheric varying refraction effects.

Large Solar Observatory (LSO)

The dramatic influence which the Sun has on the Earth, its atmosphere and magnetic field, as well as the improved knowledge of stars which will result from solar studies, is justification for intensive solar studies. Detailed image, velocity, and spectral information from 1.1×10^{-6} m (11,000 Å) to 2×10^{-10} m (2 Å) wavelength are required.

The goals and objectives of advanced solar astronomy are:

a. Extremely high resolution visible and UV studies of the solar granular structure and areas of high solar activity.

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b. Correlated XUV and X-ray observations with higher spatial and spectral resolution using larger apertures, more efficient reflective surfaces, and improved instrumentation.

Large Radio Observatory (LRO)

The continuing search for interstellar radio spectral line signals will require the development and effective utilization of more sensitive and sophisticated spectral line receiving systems. The scientific value of radio spectral lines in exploring the structure of our galaxy is tremendous; each new discovery adds vital information to the complete picture.

The objectives of the LRO missions are to observe and make measurements of radio spectral lines at both millimeter wavelengths, 50μ to 10 mm and long wavelengths, 0.5 MHz to 10 MHz.

The nominal (NASA Mission Model) size and weight of the major space astronomical observatories are as shown in the following table:

Observatory	Size	Weight
HEAO	14 ft. dia. x 46 ft. long	21,000 lbs
LST	13 ft. dia. x 45 ft. long	21,300 lbs
LSO	14 ft. dia. x 54 ft. long	27,000 lbs
LRO	14 ft. dia. x 30 ft. long	19,300 lbs

The nominal (NASA Mission Model) orbits for the major astronomical observatory missions are as follows:

HEAO: h_a = 230 nm, hp = 230 nm, I = 30°
LST: h_a = 350 nm, hp = 350 nm, I = 28.5°
LSO: same
LRO: same

The requirements imposed on the spacecraft subsystems of the four major astronomical observatories are essentially the same for all four observatories (more discussion of subsystem requirements is contained in subsequent sections). It is, therefore, possible to design a single standard spacecraft to perform all the major observatory missions.

2.0 Standard LAOS Configuration

Figure 1 shows the general configuration of the standard LAOS with an LST payload shown in phantom. In flight the +X axis is pointed at a target star and the satellite is rolled about the X axis until the center line of the solar array is normal to the satellite/sun line. The solar array is then rotated about its center line until the surface of the array is normal to the solar radiation. For every pointing direction of the LST an appropriate combination of satellite roll and solar array rotation maintains the array surface normal to the solar radiation.

The gimballed antenna provides communication with the Tracking and Data Relay Satellite (TDRS) network. Rotation of the antenna about the X axis and about an axis parallel to the Y axis ensure access to one TDRS for every pointing direction of the LST, thus providing real-time communication and eliminating any requirement for bulk data

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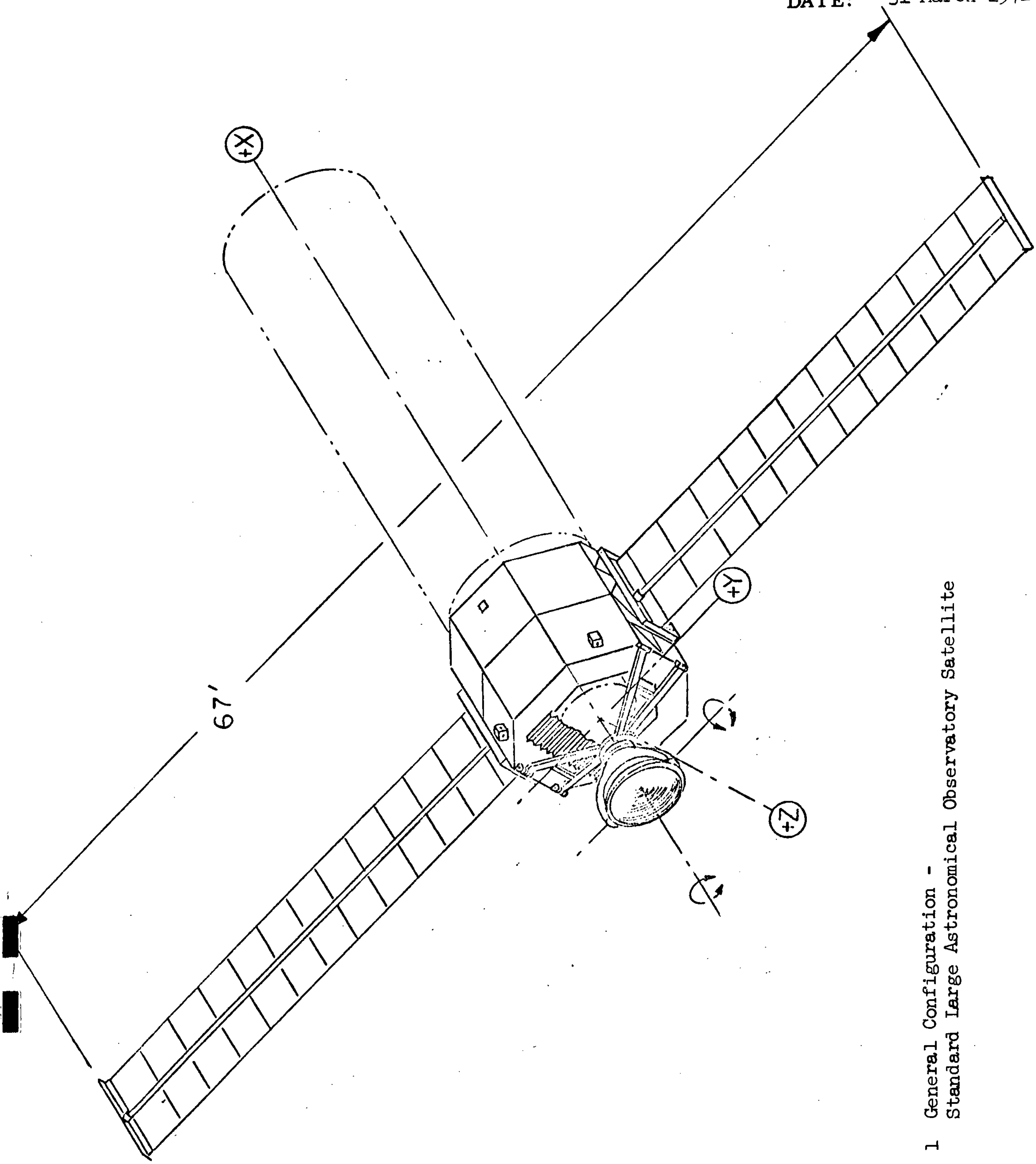

Fig. 1 General Configuration -
Standard Large Astronomical Observatory Satellite

storage on-board the observatory spacecraft. The antenna mount is hinged to permit access to the large central payload compartment.

The spacecraft structure and the payload adapter are shown in Fig. 2. The structure, designed in accordance with low-cost design guidelines, is made up of commercially available aluminum sheet and extrusions. Twenty-four compartments for standard subsystem modules are provided. The modules, designed for in-orbit removal and replacement, are installed, and the compartments are closed by non-structural doors, which provide protection against contamination and contribute to thermal control. Four machined fittings are provided for handling the spacecraft and for supporting it and the attached experiment package in the Shuttle cargo bay. The experiment package, HEAO, LST, LSO, or LRO is assumed to be designed to mate with the forward bolt circle of the spacecraft payload adapter. The internal cavity provides for major experiment modules such as the LST on-axis instrument modules, which may be serviced through the large accordion type door on the aft end of the spacecraft.

The identity and location of the standard subsystem modules of the spacecraft are given in Fig. 3. Not all of the compartments of the structure are required for the full complement of subsystem modules. There are five fully accessible compartments (A-2, B-1, B-2, E-1, E-2) that may be used for auxiliary payloads, such as prototypes of experiment equipment or spacecraft support equipment placed aboard the observatory spacecraft for on-orbit functional testing. There are also four empty compartments, adjacent to the Solar Array Drive modules, that may be used for fixed installations of auxiliary equipment or for small experiment modules.

The complete complement of standard subsystem modules required for the standard LAOS are listed in Fig. 4a through 4b. There are eleven different subsystem equipment modules plus the solar power module; the total complement of modules consists of 21 equipment modules and 2 solar power modules.

Weight Summary

The weight summary for the standard LAOS is presented in Fig. 5. The dry weight of the spacecraft is 6835 lbs including 20% contingency on the weight of the Electrical Power subsystem and 15% contingency on all other weights. Such contingencies, through conservative, are considered appropriate for preliminary low-cost designs.

Figure 6 presents estimates of the on-orbit weights of the four major astronomical observatories incorporating the standard LAOS.

Thermal Control

The thermal control of the standard LAOS will be accomplished primarily by passive thermal control techniques. To the extent possible thermal control will be accomplished by the application of appropriate internal and external surface finishes and multilayer insulation. Equipment requiring temperature control within relatively narrow limits may require supplementary methods such as thermostatically controlled heaters. The experiment package will be isolated thermally from the spacecraft to minimize the complexity of the spacecraft thermal control design. In general the thermal control of the experiment packages will also be simplified by thermal isolation from the spacecraft.

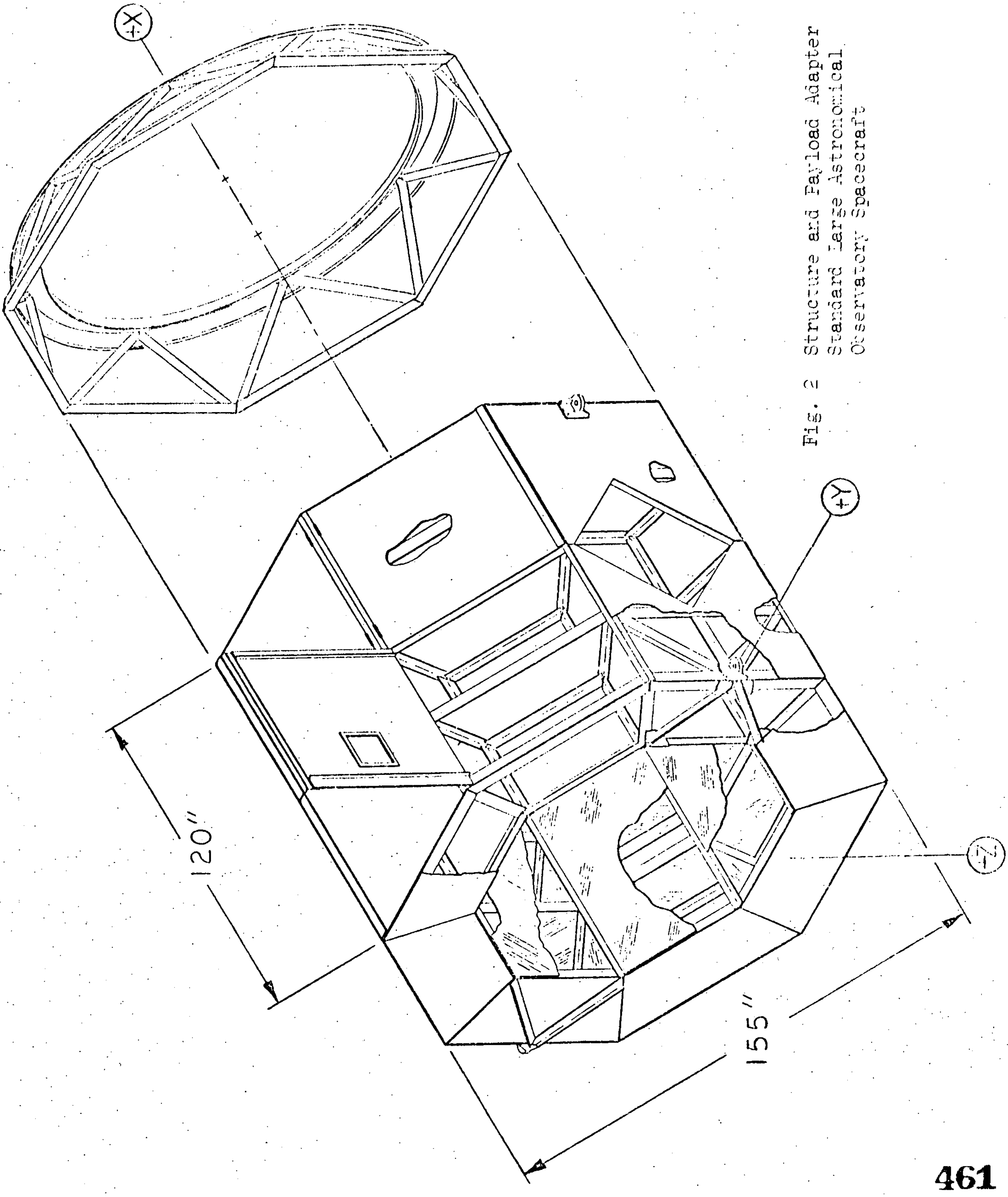

Fig. 2 Structure and Payload Adapter Standard Large Astronomical Observatory Spacecraft

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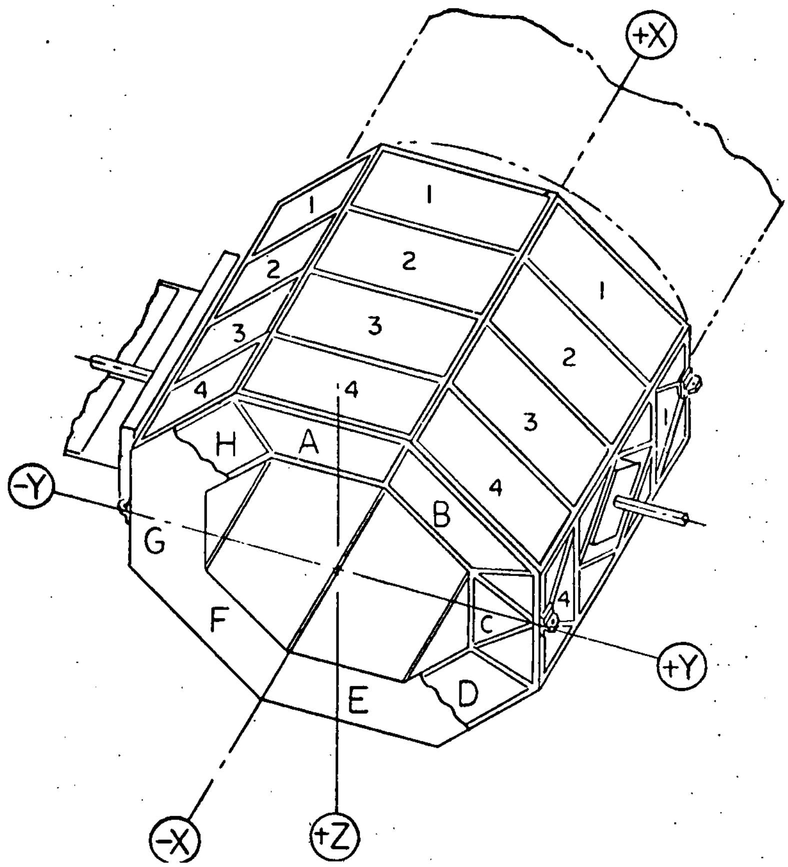

Location	Module	Location	Module
A-1	Secondary Sensing	E-1	Empty
A-2	Empty	E-2	Empty
A-3	S-Band/VHF Communication	E-3	Power Distribution
A-4	K_u Band Communication	E-4	Battery Power
B-1	Empty	F-1	Precision Sensing
B-2	Empty	F-2	Battery Power
B-3	Reaction Torque	F-3	Battery Power
B-4	Attitude Control	F-4	Attitude Control
C-1	Empty	G-1	Empty
C-2	Solar Array Drive	G-2	Solar Array Drive
C-3	Solar Array Drive	G-3	Solar Array Drive
C-4	Empty	G-4	Empty
D-1	Precision Sensing	H-1	Empty
D-2	Battery Power	H-2	Data Processing
D-3	Battery Power	H-3	Reaction Torque
D-4	Attitude Control	H-4	Attitude Control

Fig. 3 Subsystem Module Locations - Standard Large Astronomical Observatory Spacecraft

Subsystem	Module	Equipment in Module	Module Weight (lb)
Stabilization & Control	Primary Sensing Module (2 Req'd.) No. 1 No. 2	• Fixed Head Star Trackers (2) • FHST Electronics (2) • Three-Axis Rate Sensor • Precision Equipment Mount • Module Base • Module Cover • Cables and Connectors	Basic 91 lb 15% contingency 14 Total 105 lb
Stabilization & Control	Secondary Sensing Module No. 1	• Sun Aspect Sensor (5) • Sun Aspect Sensor Electronics • Rate Gyro Package • Secondary Stabilization & Control Electronics • Module Base • Module Cover • Cables & Connectors	Basic 56 lbs 15% contingency 8 Total 64 lbs
Stabilization & Control	Reaction Torque Module (2 req'd) No. 1 No. 2	• Reaction Wheel (3) • Wheel Support and Safety Shield • Wheel Drive Electronics • Magnetic Torquer (3) • Mag. Torquer Electronics (3) • Module Base • Module Cover • Cables & Connectors	Basic 226 lbs 15% contingency 34 Total 260 lb
Communication Data Processing & Instrumentation	K_u-Band Communication Module No. 1	• K-band TWTA (50 watts out) (2) • K-band PLL Receiver • K-band QPSK Modulator/Driver • K-band Multicoupler • Interface Unit (High Rate) • Module Base • Module Cover • Waveguide, Cables, Connectors	Basic 74 lbs 15% contingency 11 Total 85 lbs

Fig. 4a Standard LAOS Subsystem Modules (1 of 3)

Subsystem	Module	Equipment in Module	Module Weight (lb)
Communication Data Processing & Instrumentation	S-Band/VHF Communication Module No. 1	S-Band Transmitter (10 watts out) S-Band Receiver S-Band QPSK Modulator/Driver S-Band Multicoupler Delay Lock Loop Correlator (2) VHF Transmitter (5 watts out) VHF Receiver/Demodulator VHF Multicoupler Module Base Module Cover Cables and Connectors	Basic 68 lb 15% contingency 10 Total 78 lbs
Communication Data Processing & Instrumentation	Data Processing Module No. 1	Digital Computer Interface Unit (Med. & Low Rate) Timer Module Base Module Cover Cables & Connectors	Basic 79 lb 15% contingency 12 Total 91 lbs
Communication Data Processing & Instrumentation	Antenna Module No. 1	Antenna (6 ft. dish) Antenna Gimbal and Base Antenna Feed (S&K Bands) Rotary Joint (K-Band) Rotary Joint (S-Band) Antenna Servo Motors & Gears Antenna Servo Electronics Waveguide, Cables, Connectors	Basic 65 lb 15% contingency 10 Total 75 lbs
Electrical Power	Battery Power Mod. (5 req'd). No. 1 No. 4 No. 2 No. 5 No. 3	NiCd Battery, Type 7, 40AH (2) Charge Controller (2) Module Base Module Cover Cables & Connectors	Basic 187 lb 10% contingency 19 Total 206 lbs

Fig. 4b Standard LAOS Subsystem Modules (2 of 3)

Subsystem	Module	Equipment in Module	Module Weight (lb)
Electrical Power	Power Distribution Module No. 1	Power Distribution Unit Regulator Converter Module Base Module Cover Cables & Connectors	Basic 105 lbs 15% contingency 15 Total 120 lbs
Electrical Power	Solar Power Module (2 req'd) No. 1 No. 2	Flexible fold Solar Array Extendable Boom Assy Array container Array tension member Bearing Assembly Extension Strut Module Base & Cover Cables and connectors	Basic 191 lbs 15% contingency 29 Total 220 lbs
Electrical Power	Solar Array Drive Module (2 req'd) No. 1 No. 2	Drive Motor Assembly Electronics Slip rings and bearings Module base and cover Cables and connectors	Basic 56 lbs 15% contingency 8 Total 64 lbs
Attitude Control	Attitude Control Control Module (4 req'd) No. 1 No. 2 No. 3 No. 4	Gas Storage Tank, 22" O.D. Regulator Valve Assembly Fill Valve Thruster Cluster Transducers (set) ACS Electronics Plumbing Module Base Module Cover Cables & Connectors	Basic 118 lbs 15% contingency 18 Total 136 lbs

Fig. 4c Standard LAOS Subsystem Modules (3 of 3)

Subsystem	Contingency (%)	Weight (lb)
Structure & Mechanisms	15	2800
Environmental Control	15	150
Stabilization & Control	15	794
Communication, Data Processing, Instrumentation	15	329
Electrical Power	20	2218
Attitude Control	15	544
LAOS Dry Weight		6835
Propellant, Freon 14		400
LAOS Wet Weight		7235
Adapter, Mission Payload	15	491
		7726

Fig. 5 Standard LAOS Weight Summary

Mission	LAOS (Wet) Weight	Adapter Weight	Mission Pay-load Weight	Total Weight
HEAO	7235	491	18,000	26,126
LST	7235	491	14,000	22,126
LSO	7235	491	12,000	20,126
LRO	7235	491	13,000	21,126

Fig. 6 On-Orbit Weight (lbs) - Standard LAOS & Mission Payload

3.0 Major Subsystems of the LAOS

The major functional subsystems of the LAOS are as follows:

Stabilization and Control	(S&C)
Communication, Data Processing, and Instrumentation	(CDPI)
Electrical Power	(EPS)
Attitude Control	(ACS)

Each of these subsystems is discussed in detail in subsequent sections.

3.1 Stabilization & Control Subsystem

Summary

The Stabilization & Control (S&C) Subsystem requirements for four large spacecraft observatories (LST, HEAO, LRO and LSO) have been analyzed to determine the applicability of the Standard EOS S&C equipment and to identify necessary changes or add-ons.

The EOS S&C Subsystem design is a prototype Standard S&C Subsystem. A summary description of this concept, extracted from Ref. 1, is provided in Appendix A.

The pertinent S&C design considerations used to investigate applicability of the EOS S&C subsystem are listed in Table 1 and discussed in the pages following.

It has been concluded that the Large Astronomical Observatory Spacecraft (LAOS) S&C Subsystem can be functionally identical to the EOS S&C Subsystem, based upon a review of these design factors.

In fact, much of the Standard EOS S&C Subsystem equipment is directly applicable to LAOS. However, at least two more fixed head star trackers are needed and the reaction wheels and magnetic torquers should have five and ten times respectively as much torque capability. The addition of two star trackers for increased coverage and the recommended division of the large wheel momentum and the magnetic torque capabilities into two parts leads to the conclusion that the LAOS S&C Subsystem can continue to operate with degraded capability after one or more failures of star trackers, wheels, and magnetic torquers.

The recommended changes to the Standard EOS S&C Subsystem were necessitated principally by the expected large size and relatively slender shape of LAOS (plus its experiments) in conjunction with the need to hold inertially-fixed attitudes accurately for extended periods of time.

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Table 1

S&C DESIGN CONSIDERATIONS

A. Orbit
   Altitude, inclination, right ascension

B. Functional Requirements

C. Lifetime/Duty Cycle/Reliability

D. Operating Modes

E. Pointing Directions

F. Pointing Accuracies

G. Pointing Stabilities (Mode, Axis)

H. Determination Accuracies (Mode, Axis)

I. Maximum Attitude Rates (Mode, Axis)

J. Attitude Rate Stabilities (Mode, Axis)

K. Momentum Storage (Pointing, Slewing)

L. Maximum Torques (Mode, Axis)

M. Interfaces
   CDPI (Processing, Communications)
   ACS (Thrust, Bit, Duty Cycle, Total Impulse)
   Structural/Thermal

N. Constraints
   Weight
   Size
   Power
   Cost
   Technology
   Operational

Discussion

The Standard EOS S&C Subsystem Attitude Determination Concept, being stellar-aided inertial, is essentially universally applicable. The equipment operating characteristics are, for the most part, unaffected by the particular orbit (altitude, inclination, right ascension) or by spacecraft attitude. A similar insensitivity to mission particulars can be ascribed to the reaction wheels used for attitude control torques. Each of the pieces of S&C equipment has, furthermore, another important characteristic which enhances its versatility: a wide operating range, that is, the ratio of a part's maximum (or saturated) capability to its minimum (or threshold) value. Figure 7 shows typical values for present-day hardware. An inertial-grade gyro, for example, can measure from below 0.0005 deg per sec to over 5 deg per sec. It is not surprising, therefore, that this S&C implementation is adaptable to a wide variety of spacecraft and missions. There are, however, some subsystem design characteristics which are dependent upon the spacecraft/mission parameters. The more important ones are those shown in Table 2.

	Operating Range
• Fixed Head Star Trackers	$\approx 10^3 : 1$
• Three-Axis Rate Sensors	$10^4 : 1$
• Digital Computers	10^4 to $10^5 : 1$
• Solar Aspect Sensors	10^3 to $10^4 : 1$
• Reaction Wheels	$> 10^4 : 1$

Fig. 7 Versatile Stabilization and Control Equipment - 1971

The equipment necessary, then, to implement the LAOS S&C Subsystem is compared to that from the standard EOS S&C Subsystem in Table 3.

Dividing the momentum/reaction torque producers into twin half-size units facilitates packaging a "complete set" (three orthogonal wheels, three orthogonal magnetic torquers and associated electronics) into each of two identical modules and enhances subsystem reliability.

The other changes to the EOS S&C Subsystem indicated in Tables 2 and 3 are as follows:

1. Increased size of reaction wheels and magnetic torquers. Permits extended periods of inertially-fixed pointing as well as wheel-driven slewing between targets without momentum dumping. Compatible with the greatly increased inertia of LAOS.

Table 2

SPACECRAFT/MISSION EFFECTS ON S&C SUBSYSTEM EQUIPMENT PARAMETERS

Spacecraft/Mission Parameter	S&C Subsystem Elements Affected
Orbit Altitude/Inclination Spacecraft Residual Magnetism Spacecraft Operating Attitudes Spacecraft Inertial Diad* Undisturbed Observation Time	Magnetic Torquer Strength Reaction Wheel Momentum Storage
Orbit Altitude** Spacecraft Pointing Direction Pointing Accuracy/Stability Observation Time	TARS Drift Allowable Number of Star Trackers Star Tracker FOV, Sensitivity Computer Software Complexity
Mission Duration Subsystem Duty Cycle Reliability Allocation Consequences of Outages	Numbers of Components Types of Redundancies Backup Modes
Number of Spacecraft Slews Size of Slews Time Allotted to Slew Spacecraft Inertias	Reaction Wheel Torque Reaction Wheel Momentum Storage Attitude Control Propulsion Thrust Attitude Control Propulsion Total Impulse

* Sizes of moment of inertia differences and of products of inertia.
** Duration of earth occultation.

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Table 3

COMPARISON OF EOS AND LAOS S&C SUBSYSTEMS

Item	EOS	LAOS
	Number and Characteristic	
Three-Axis Rate Sensor	One/0.01 deg/hr	Two/0.01 deg/hr
Fixed Head Star Tracker (plus Electronics)	Two/5 deg FOV +4 magnitude-sensitivity	Four/6 deg FOV +4 magnitude sensitivity
Computer Software	3200 16-bit words	4000 16-bit words
Reaction Wheels (plus Electronics)	Three/10 ft-lb-sec 10 oz-in torque each	Six/50 ft-lb-sec 25 oz-in torque each
Magnetic Torquers (plus Electronics)	Three/10^5 UPC	Six/10^6 UPC
Sun Aspect Sensors (plus Electronics)	Five/$64^\circ \times 64^\circ$	Same as EOS
Secondary Electronics, Rate Gyro Package	One each	Same as EOS
ACS Drive Electronics, Laser Corner Cubes	Four each	Same as EOS

2. Increase in the number of fixed head star trackers from two to four. Compensates for reduced number of star targets due to: (a) inertial pointing as opposed to orbital-rate motion-produced scan; (b) arbitrary celestial pointing direction requirement; and (c) earth occultation. The general need for continuous three-axis attitude information, as opposed to discrete two-axis updates, leads to the requirement for two stars to be in view at all times.

Present-day star trackers, with a 1000-to-1 operating range, are limited to at most about a six-degree square field-of-view in order to provide the desired accuracy (about 20 sec). Figure 8 from Bendix Corp. shows that four trackers (144 sq deg) will have a lowered probability of having a star in view if one or more of the trackers is pointing in the vicinity of the galactic pole. If, however, the spacecraft were permitted to roll ± 6 deg from the optimum sun-line there is virtually 100% probability of one or more stars being in the combined field-of-view which is, in effect 432 deg^2.

To achieve a near 100% probability of having two or more stars in the field-of-view, a rotation of as much as ± 18 deg (total viewed area ≤ 1008 deg^2) is required. This roll angle will degrade the solar power generation capability by, at most, 5%.

The physical characteristics of the LAOS S&C Subsystem equipment are summarized in Table 4.

This equipment will be grouped into modules similar to those of EOS. These are as follows:

(1) Primary Sensing Module (2 req'd)

 2 Fixed Head Star Trackers/Electronics
 1 Three Axis Rate Sensor/Electronics

(2) Secondary Sensing Module (1 req'd)

 1 Secondary Control Electronics
 1 Rate Gyro Package
 5 Sun Aspect Sensors/Electronics

(3) Reaction Torque Module (2 req'd)

 3 50 ft-lb-sec Reaction Wheels
 1 Reaction Wheel Electronics
 3 10^6 UPC Magnetic Torquers
 1 Magnetic Torquer Electronics

Finally, one ACS thruster drive electronics package is located in each of four ACS modules.

The S&C Module locations in the LAOS are chosen to:

(1) Place the two primary sensing modules on the dark side, adjacent to the experiment mounting/alignment plane, and located one to each sloping face.

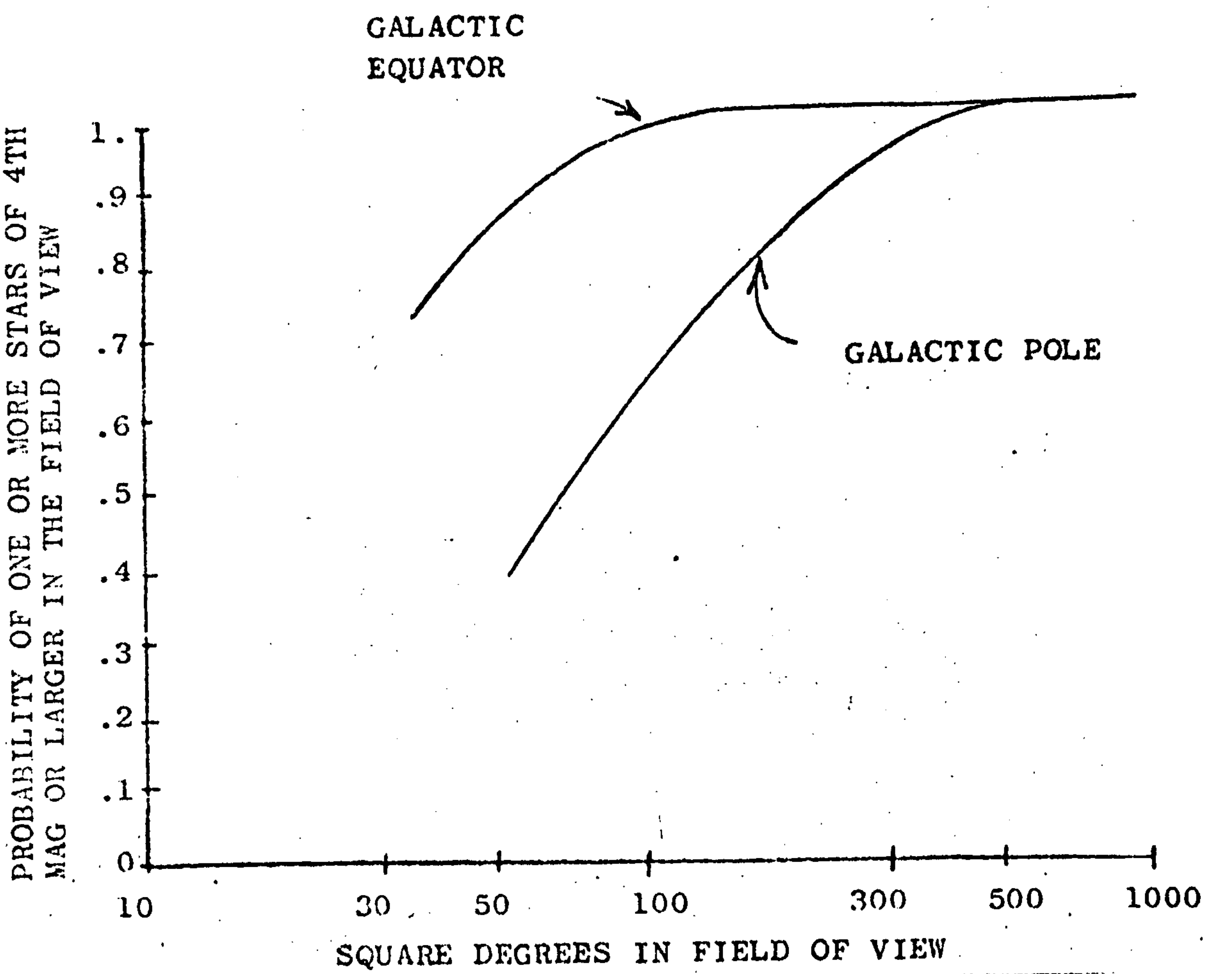

Fig. 8 Star Availability as a Function of Field of View

Table 4

LAOS S&C SUBSYSTEM EQUIPMENT LIST

Item	Qty	Unit Weight (lb)	Unit Power (w)	Est. Failure-Rate per 10^6 hr & Duty Cycle (%)	Total Weight (lb)	Total Avg. Power(w)	Unit Size
Three Axis Rate Sensor/ Electronics	2/2	15	30	10(100)	30	30	9x9x7
Fixed Head Star Tracker/ Electronics	4/4	13	11	5(50)	52	22	6x5x12
Sun Aspect Sensor/ Electronics	5/5	1/2	0/1	3(20)	15	5	3x4x1/ 2x5x4
Reaction Wheel/ Electronics	6/2	35/10	9/10	2/6(100)	230	74	16Dx8/ 5x5x7
Magnetic Torquer/ Electronics	6/2	15/4	30(pulse)/ 3	1(100/1)	98	8	2Dx16/ 3x5x7
Laser Corner Cubes	4	2	0	0	8	0	4x4x2
ACS Drive Electronics	4	7	13	3(10)	28	12	6x4x3
Rate Gyro Package	1	3	15	10(1)	3	Backup	4x4x2
Secondary S&C Electronics	1	5	5	10(1)	5	Backup	2x4x6
Totals					469	151	

The octagonal shape of LAOS thus causes each star tracker pair's line-of-sight to be directed 45 deg to the left or right of the anti-sun line, and, therefore, be 90° apart. Furthermore, the two star trackers within each module are mounted such that their lines-of-sight are 90° apart in the LAOS fore and aft direction.

(2) Place the sensing and electronics (or secondary sensing) module on the sun-lit side of the spacecraft centered on the sunline.

(3) Place the reaction torque modules as remote as is practical from the primary sensing modules.

This arrangement is pictured in Fig. 9 .

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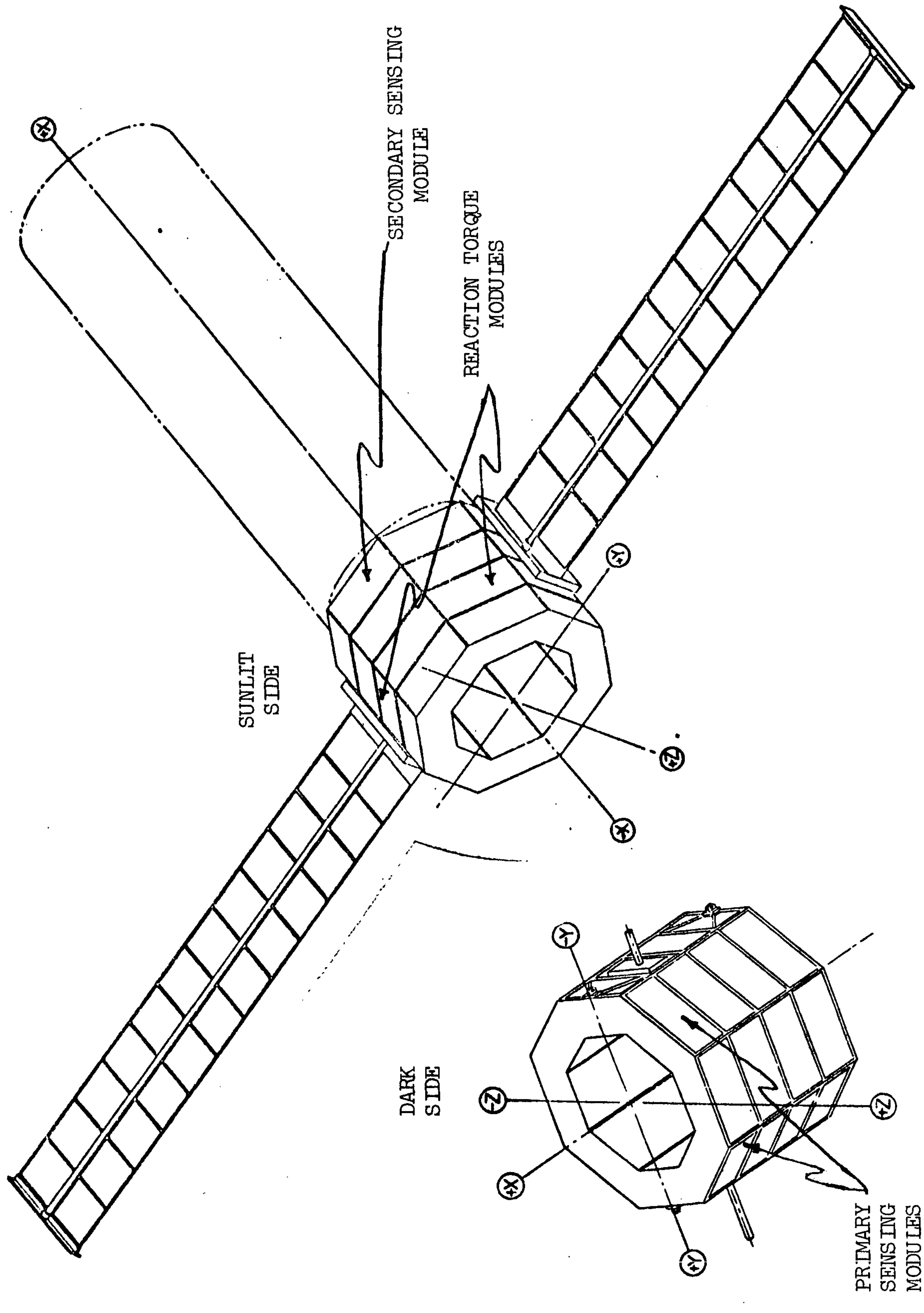

Fig. 9 LAOS S&C Module Arrangement

MATCH-UP OF LAOS S&C REQUIREMENTS WITH STANDARD EOS S&C CAPABILITIES

A. OPERATING ORBIT

(1) EOS (#21, #26, #77)

500 nm circular
99.2° inclination (Sun-Synchronous)
Noon

(2) LST (#15)

350 nm circular
28.5 deg inclination

(3) HEAO (#13)

230 nm circular
30 deg inclination

(4) LSO (#17)

270 nm circular preferred
Sun-Synchronous preferred, 0 to 55° acceptable

(5) LRO (#19)

350 nm circular
30 deg inclination

Conclusion

The sensing and control equipment chosen to implement the Standard EOS S&C Subsystem have functional characteristics which are effectively independent of the orbit parameters listed.

B. FUNCTIONAL REQUIREMENTS

The combined functional requirements of LST and HEAO are considered to be representative of the class of Large Observatory Spacecraft.

(1) LST

The functions to be performed by LST are divided between a mass expulsion system using dedicated electronics (ACS), and conservative reaction torque systems under control of a programmable computer.

The functions of the ACS are:

(a) Post Booster Vehicle/Large Space Telescope separation stabilization.
(b) Initial maneuver to favorable position for Maneuvering and Coarse Control System acquisition.
(c) Inertial hold.
(d) Emergency momentum dumping.
(e) Tumbling captures for Space Shuttle visit and reacquisition.

Computer Associated Control Systems. There are three control systems which depend upon major calculations within the CDPI Subsystem.

(1) MACCS - Maneuvering and Coarse Control System
(2) FACS - Fine Attitude Control System
(3) MTCS - Magnetic Torquing Control System

The fundamentals of each of the above control systems are shown in Fig. 10. More details for each system are discussed in the following sections.

Maneuvering and Coarse Control System (MACCS). The Maneuvering and Coarse Control System (MACCS) is composed of a collection of pulse rebalance gyros (may be the same as those used for ACS) fixed head star sensors, reaction wheels and a control law to tie the above sensors and actuators together.

Its functions are:

(1) To perform coarse acquisition along any inertial direction.

(2) To hold some specified inertial position during the time the telescope line of sight is occulted by the earth.

(3) To perform maneuvers which will allow magnetic torques and/or gravity gradient torques to dump momentum storage.

(4) To perform target to target maneuvering.

(5) To perform a two-axis search if coarse acquisition is not immediately successful.

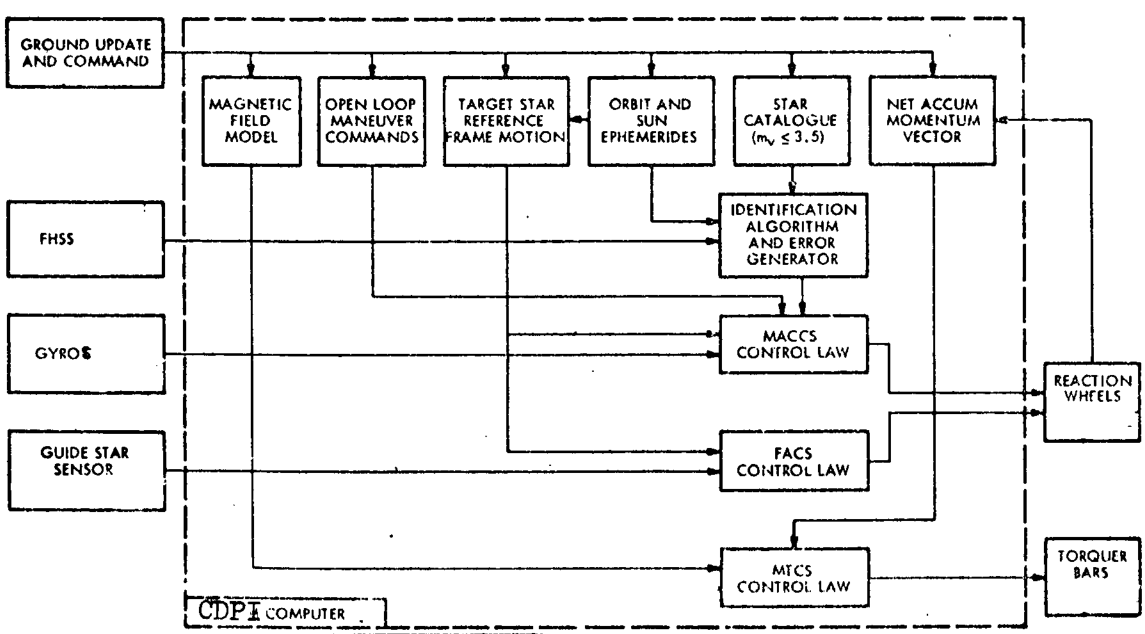

Fig. 10 Functional Diagram for Computer-Associated Control Systems 478

Fine Attitude Control System (FACS). Using error signals obtained from the payload experiment sensor, the Fine Attitude Control System (FACS) must provide spacecraft pointing for payload operation.

The basic block diagram of the Fine Attitude Control System is shown in Fig. 11. Error signals for each of the three body geometric axes are obtained from the fine error sensor and are processed in the computer to produce torque commands to the control torque actuators.

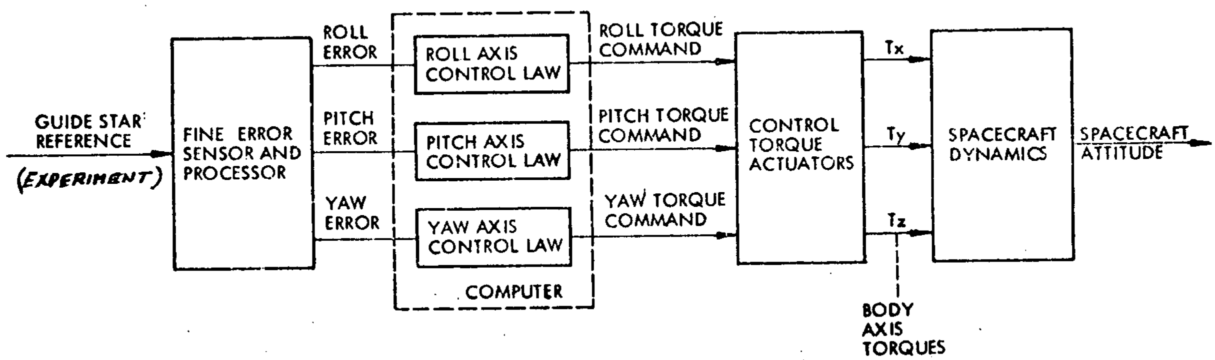

Fig. 11 FACS Functional Block Diagram

Magnetic Torquing Control System (MTCS). The Magnetic Torquing Control System (MTCS) functions are:

(1) To perform complete momentum dumping when maneuvered to an attitude where secular gravity gradient torques are insufficient.

(2) To perform continuous (although not complete) momentum dumping while the telescope is carrying out observations.

(3) To compensate for the residual magnetic dipole moment of the telescope.

(2) HEAO

The "Initialization" (ACS) functions are:

(a) Post-Booster Separation Stabilization
(b) Control during Deployment
(c) Control during Orbit Adjust
(d) Sun Acquisition
(e) Tumbling Capture/Sun Reacquisition
(f) Limit Cycle (ACS Backup)

"Scan" functions are:

(a) Rotate about sun line at 0.1 RPM (Celestial Scan).
(b) Rotate about a line offset from sunline within 30 deg cone.

"Celestial point" and "Hold" functions are:

(a) Direct the viewing axis of any experiment to any point on the celestial sphere.

(b) Hold this pointing direction for up to 3 days.

C. LIFETIME/DUTY CYCLE/RELIABILITY

All four of the large spacecraft will be designed for one year of continuous primary subsystem operation. This includes the star trackers, three-axis rate sensor, CDPI computer, a sun aspect sensor, and three-axis reaction wheel/magnetic torquer control.

The Standard EOS S&C Subsystem is required to provide a reliability of 0.850 for a one-year operating life. The reliability associated with the equipment and modes actually chosen is higher, being 0.869 for one-year. This value could be increased further by adding one or more redundant modules.

Conclusion: The S&C Subsystem as configured can have adequate reliability for the expected one year of continuous operation.

D. OPERATING MODES

Standard EOS S&C utilizes the following operating modes:

- Primary
  - Earth Pointing
  - Momentum Dumping (Magnetic)
- Secondary
  - Rate Nulling
  - Roll Search
  - Sun Pointing
  - Earth Acquisition
  - Momentum Dumping (Gas)
- Docking
  - Single Axis Slewing
  - Rate Nulling

The Large Observatory Spacecraft require the following modes:

- Primary
  - Celestial Pointing/Scanning
  - Momentum Dumping (Magnetic/Gravity)
  - Inertial Hold
  - Single-Axis Reorientation
- Secondary
  - Rate Nulling
  - Momentum Dumping (Gas)
  - Earth Pointing/Acquisition

Conclusion

There is no essential difference between the two sets of modes nor any fundamental limitation on the Standard EOS S&C Subsystem which precludes it from operating in the LAOS modes. The reason for this flexibility is the "celestial-inertial" implementation of EOS which computes the earth vertical as opposed to a system which relies on earth sensing.

E. POINTING DIRECTIONS

In contrast to EOS, the large observatories are required to point in celestial (fixed) directions rather than along orbital (rotating) axes (velocity, orbit normal, local vertical) and to maintain direction precision ("pointing stability") while so doing for extended periods of time. During (or immediately after) these times, angular momentum due to gravity gradient disturbance torques must be continuously and totally absorbed.

Expected duration of celestial observations are:

EOS	NA
LST	4 hrs
HEAO	3 days (4.5 hrs without gas unloading)
LSO	0.75 hrs
LRO	?

Conclusion

There are no explicit restrictions on spacecraft attitude imposed by the Standard EOS S&C Subsystem. Accuracy will depend on star availability, earth/sun interference schedule, and time in a given attitude.

The amount of momentum storage is, however, affected by the pointing requirement. (See K).

F. POINTING ACCURACY

EOS: ± 0.5 deg required (Roll, Pitch)
± 25 sec capability* (Roll, Pitch, Yaw)

LST:

Mode	Requirement
ACS	± 0.4 deg (Roll), ± 0.5 Deg (Pitch), ± 0.6 Deg (Yaw)
MACCS	± 30 sec (Pitch, Yaw), ± 6 min (Roll)
FACS	± 1 sec (Roll, Pitch, Yaw)

HEAOS:

Mode	Requirement
Celestial Scan (Sun Pointing)	± 1.0 deg (Pitch, Roll)
Galactic Scan (Sun Pointing)	± 1.0 deg (Pitch, Roll)
Celestial Pointing	± 1.0 deg (Pitch, Yaw)
Inertial Hold (Occultation)	± 0.05 deg (Roll, Pitch, Yaw)
Terminal Fine	± 2 min(Pitch, Yaw)

LSO:

Mode	Requirement
Acceptable	± 1.0 sec
Desired	± 0.1 sec

LRO:

General	± 1.0 sec

---

* With ephemeris accurate to 60 m., and including alignment tolerances.

Conclusion

The EOS capabilities are compatible with LAOS requirements. A fine error signal is necessary whenever pointing to better than about ± 20 sec is required, but the assumption is made that this signal is derived from sensors contained within the experiment/payload itself.

G. Pointing Stability
H. Attitude Determination Accuracy
I. Maximum Attitude Rates
J. Attitude Rate Stabilization

	(G.) Pointing Stability	(H.) Attitude Determination Accuracy	(I.) Maximum Attitude Rates	(J.) Attitude Rate Stability
EOS	NA	±0.002 deg (1σ)	±4 deg/sec	±0.005 deg/sec
LST	±0.011 deg/hr	± 20 sec	±0.024 deg/sec	±0.016 sec/sec
HEAO	±0.1 deg/hr	±0.1 deg	±0.1 RPM	± 0.01 RPM ± 5 sec/sec (LC)
LSO	±0.016 sec/hr	NA	±10 sec/sec	± 0.1 sec/sec
LRO	± 0.5 sec	±1.0 sec	?	?

Conclusion

The ability of the S&C Subsystem to meet the attitude determination stability requirements is largely dependent upon the quality of error signal supplied by the experiment in each case.

The Attitude Rate Requirements will impose design requirements on the reaction wheel size and "smoothness". For example, to achieve the LST maximum rate (slew) requires over 50-ft-lb-sec of angular momentum storage, and the rate stability "granularity" of less than 0.01 ft-lb-sec.

K. MOMENTUM STORAGE

The Standard EOS Reaction Wheel triad was sized to provide somewhat less than ± 10 ft-lb-sec momentum storage about each axis.

For LST, a storage capability of ± 100 ft-lb-sec along each axis would provide the following viewing capability:

LOS/Orbit Angle	Viewing Time Before Unloading
45°	50 min
15°	110 min
5°	Unlimited

This assumes the following:

LST Residual Magnetism:	3×10^5 UPC
Magnetic Torquer Capability:	0.02 ft-lb (any axis)
Gravity Gradient Torque (avg):	0.09 ft-lb

For HEAO, the viewing time before unloading must be at least three orbits (about 5 hours) in the worst case.

Conclusion

Larger momentum storage sources than those provided for Standard EOS are required to cope with the very large moment of inertia differences of the LAOS in conjunction with the requirement for extended periods of observation in attitudes skewed with respect to local orbit coordinates.

L. MAXIMUM TORQUES

Standard EOS is configured to provide the following reaction control torques along any axis:

Magnetic	0.001 ft-lb (min)
Wheel	0.05 ft-lb (min)

Expected disturbance torques for LST are:

Gravity Gradient	0.18 ft-lb (peak)
Magnetic	0.007 " "
Solar	0.0005 " "
Aerodynamic	0.0015 " "

A reasonable control torque capability is:

Magnetic	0.020 ft-lb (min)
Reaction Wheel	0.25 ft-lb (min)

along any axis.

Conclusion

Larger control torque sources are required to maintain the wheel momentum at a suitable low level.

M. INTERFACES

(1) Data Processing

Beyond the routines anticipated for EOS, the LAOS will require:

(a) Attitude Control during Orbit Adjust
(b) Attitude Control during Fine Pointing
(c) Augmented Optimal Estimator during Fine Pointing
(d) Redundant Wheel/Magnetic Control Logic
(e) Redundant Star Tracker Sensing Logic
(f) Augmented Star Catalog

(2) Attitude Control Subsystem (ACS)

The LAOS AC Subsystem requirements will differ considerably from those of EOS and, in turn, influence the S&C electronic design.

For gross Attitude Control subsystem sizing, the following specifications can be employed:

LAOS ACS Requirements

		Lb-Sec
Impulse:	Separation Stabilization	4,000
	Orbit Adjust	4,000
	Initial Attitude Maneuvers	1,000
	Mass Expulsion Momentum Dumps	4,800
	Reacquisition after Tumbling	500
	30-Days Attitude Hold	1,000
	Shuttle Docking Attitude Control	400
	Total	15,700 lb-sec

Thrust:	Minimum:	0.2	lbf preferred
		2	lbf acceptable
	Maximum:	10	lbf preferred
		1	lbf acceptable

N. CONSTRAINTS

There are no known weight, size, power, cost, technology or operational constraints associated with LAOS which do not apply equally to EOS.

REFERENCES

1. Application of the SCS to the LST Mission Support System Module; LMSC-D152631; Dec, 1971.

2. HEAO Phase C/D Statement of Work for Missions A & B;

3. Preliminary Edition of Reference Earth Orbital Research and Applications Investigations, Vol II - Astronomy; NASA MSFC; Jan, 1971.

4. Conceptual Design of a High Energy Astronomy Observatory, Vol. I; NASA MSFC; Feb, 1970.

5. Fleming Mission Model (Aerospace Corp.)

APPENDIX A

STANDARD EOS S&C SUBSYSTEM

SUMMARY

The standard EOS Stabilization & Control Subsystem is required to maintain the spacecraft in an earth-oriented attitude for up to one-year and provide readout for ground-based precision attitude determination. The principle of operation of the primary attitude determination function is to obtain accurate long-term attitude information from a star sensor and a computer-stored star catalog, while three-axis rate sensor gyros provide a precise short-term reference. A computer processes and mixes the information from both sources. A check on the vehicle orientation in space is available to ground stations by readout of solar aspect sensors. These sensors would also be used for attitude reacquisition should a catastrophic event cause the spacecraft to tumble.

Attitude control torques are obtained by varying the speed of reaction wheels or pulsing attitude control thrusters. Momentum absorbed by the wheels is continuously reduced as the magnetic torquers interact with the ambient earth field under control of the CDPI computer. Should an unexpectedly large disturbance torque cause the wheels to saturate (reach maximum speed); desaturation is accomplished by torquing the spacecraft with the attitude control jets.

Reorientation from one attitude reference to another, if required, is accomplished by a series of slews using the wheels and/or gas jets which also control TARS-detected attitude errors about the non-slew axes.

The standard EOS S&C Subsystem is implemented entirely with equipment planned for the Standard Spacecraft S&C inventory.

The factors leading to the decision to utilize star sensing/tracking for the primary attitude reference in lieu of earth sensing are: It can be shown that the need for high precision spacecraft attitude determination makes a less accurate (e.g., horizon) sensor redundant; the ready availability of an up-to-date ephemeris to the spacecraft via the TDRS; the low-cost of on-board attitude calculation offered by a fourth-generation, general-purpose aerospace computer, the existence of a number of fixed head (gimballess) star trackers, and the benefits of development cost-sharing.

The standard EOS S&C Subsystem utilizes two fixed-head star trackers (FHT) with narrow angle (~5 deg) optics for both on-board (coarse) attitude determination and for after-the-fact ground (precision) attitude determination. Since, after the first orbit or two, all accuracy requirements can be met with but one FHT, the second unit provides essentially full mission redundancy.

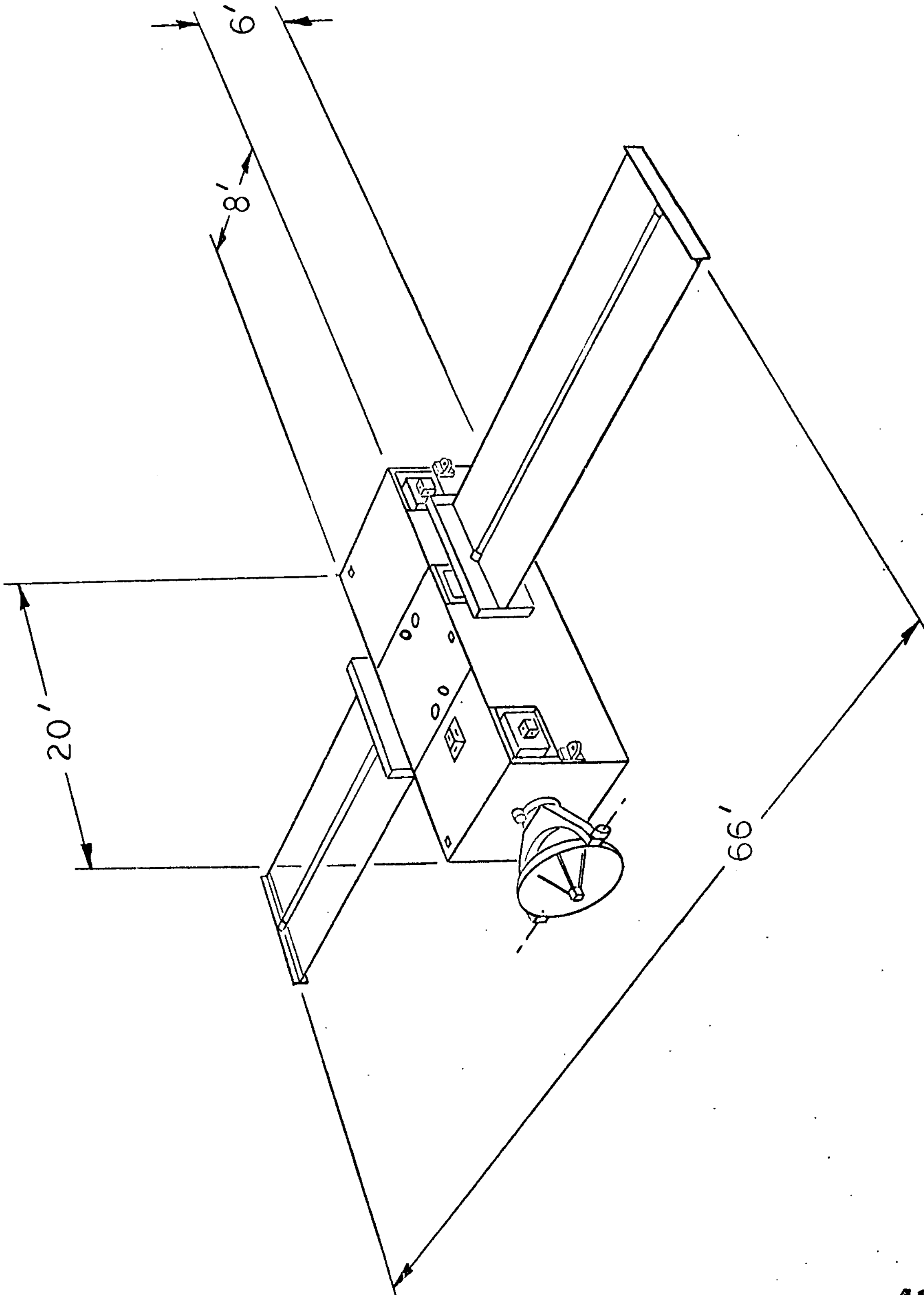

Fig. A-1 Standard Earth Observatory Satellite

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I. PROBLEM STATEMENT

A. Ground Rules and Constraints

The Stabilization & Control (S&C) Subsystem will be designed under the following ground rules and constraints:

1. Space Shuttle/Space Tug-launched
2. 1971-72 technology (concepts reduced to practice)
3. One-year life without refurbishment; up to three reuses
4. Subsystem reliability goal: 0.85
5. Low-cost: Trade weight increase for cost reduction
6. Minimize number of variants: Favor over-design in preference to a multiplicity of mission-peculiar parts
7. Subsystem is checked out and initialized at shuttle separation

Figure A1 shows the standard EOS spacecraft arrangement.

B. Functional Requirements

The Standard EOS Spacecraft Stabilization & Control Subsystem has the following functions:

(1) To stabilize the EOS spacecraft following shuttle separation, and establish the preselected attitude with a precision of ± 0.5 deg.

(2) To hold the spacecraft in the reference attitude with the required accuracy over periods of time up to 1 year with growth capability to 2 years.

(3) To provide a measure of instantaneous spacecraft attitude of sufficient accuracy to permit ground-based after-the-fact attitude determination within ± 0.002 deg (1σ).

(4) To hold the spacecraft attitude rates to less than ± 0.005 deg/sec, all axes.

(5) To reorient the spacecraft to the reference attitude from any attitude, following loss of reference due to reversible system failures, for tumbling rates up to 10 deg/sec.

(6) To point the spacecraft at the sun with near zero attitude rates following primary subsystem failure.

II. DESCRIPTION OF STANDARD EOS S&C SUBSYSTEM

A. Subsystem Operation

(1) Primary Mode

Figure A2 shows the functional and equipment relationships. The three-axis rate sensors (TARS) and one or two fixed-head star trackers (FHT) 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 or gas jets to supply three-axis stabilization and control.

The S&C subsystem functional flow (Fig. A3) uses the on-board CDPI computer for attitude determination. The initial attitude at shuttle separation has been stored in the computer. The TARS continuously measures attitude changes which are processed in the computer to update the stored value. Thus, a numerical record exists of the spacecraft current attitude in an inertial coordinate frame. This attitude is compared to a stored reference attitude and the errors used to drive the wheels or jets. Note that this entire operation is proceeding independently of the star tracker references, which are, in effect, "off-line". The attitude reference sensor outputs are periodically sampled and optimally blended with the gyro-determined attitude to yield a best-estimate update to the attitude in storage at that instant but the sensor operation is decoupled from spacecraft attitude control (Fig. A4).

The subsystem design approach takes advantage of the repeatability and stability of inertial-grade gyros for measurements of high data-rate attitude changes. Because random variations in gyro parameters are very small (better than 0.01 deg/hr) only discrete updating is necessary. The periodic star fixes, via the filter in the computer, bound long-term attitude errors and, at the same time, update the estimated random gyro drifts, scale factor errors, and alignment biases.

In effect, the gyros provide high bandwidth attitude data with unbounded errors (drift) whereas the star sensor provides low bandwidth data with bounded errors. After combination, these data yield high bandwidth attitude knowledge with bounded errors (Ref. 4).

(2) Secondary (Anti-Tumbling) Mode

A significant portion of the anticipated cost savings associated with the Shuttle derives from the planned return to the payload for maintenance, resupply, or recovery for refurbishment/repair.

It is axiomatic that it will be necessary to approach and grasp the satellite prior to performing any of these activities. While various special techniques and mechanisms can no doubt be conceived which will permit this if the satellite is tumbling randomly about any or all of its axes, it seems clear that it would be simpler, safer, and cheaper to assure that the satellite will be stabilized at Shuttle arrival. This is an easy requirement to satisfy: the main decision required is to what degree will an independent capability be provided. The equipment required is: 490

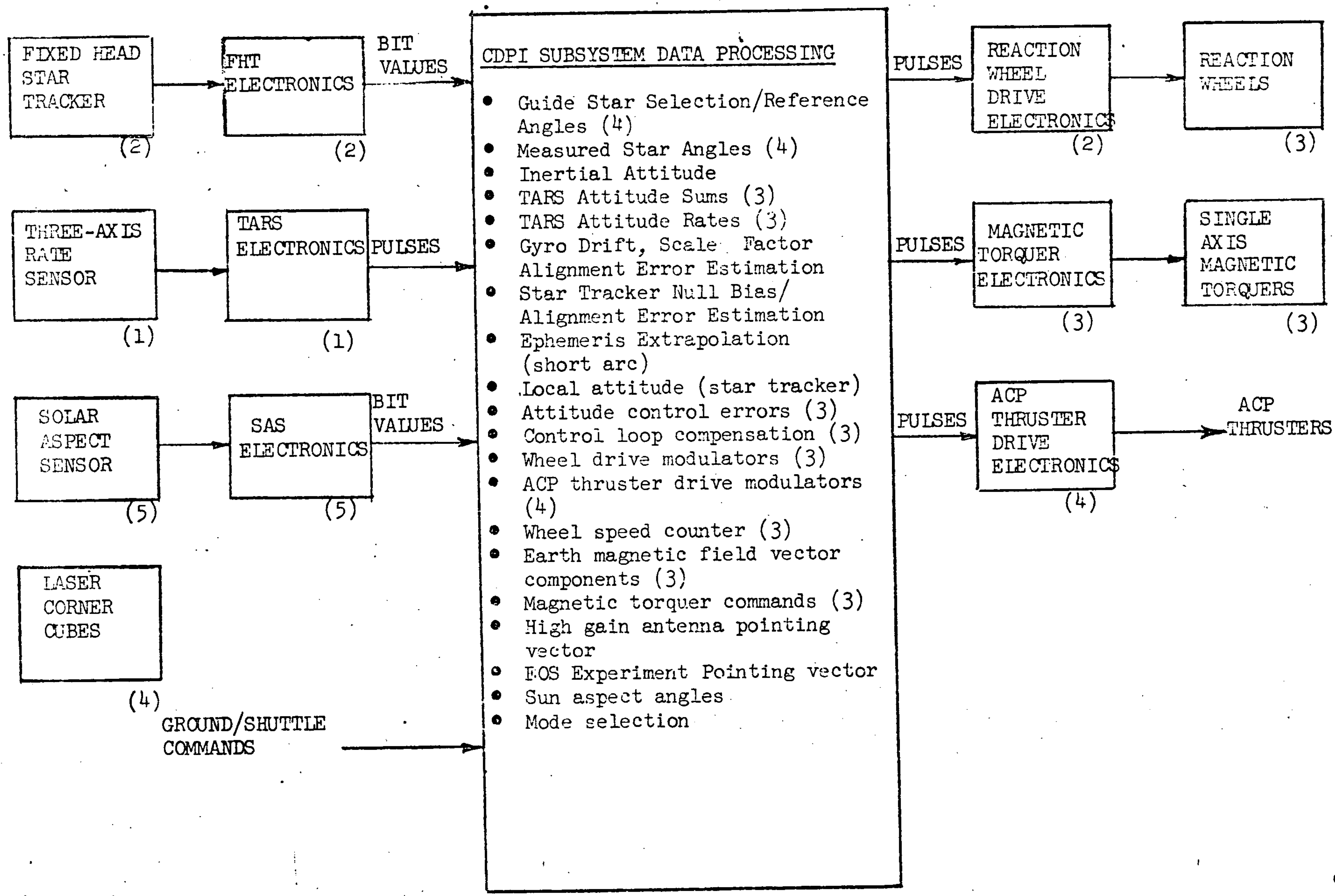

Fig. A2 Standard EOS Stabilization & Control Subsystem - Primary Mode (including Redundant Equipment)

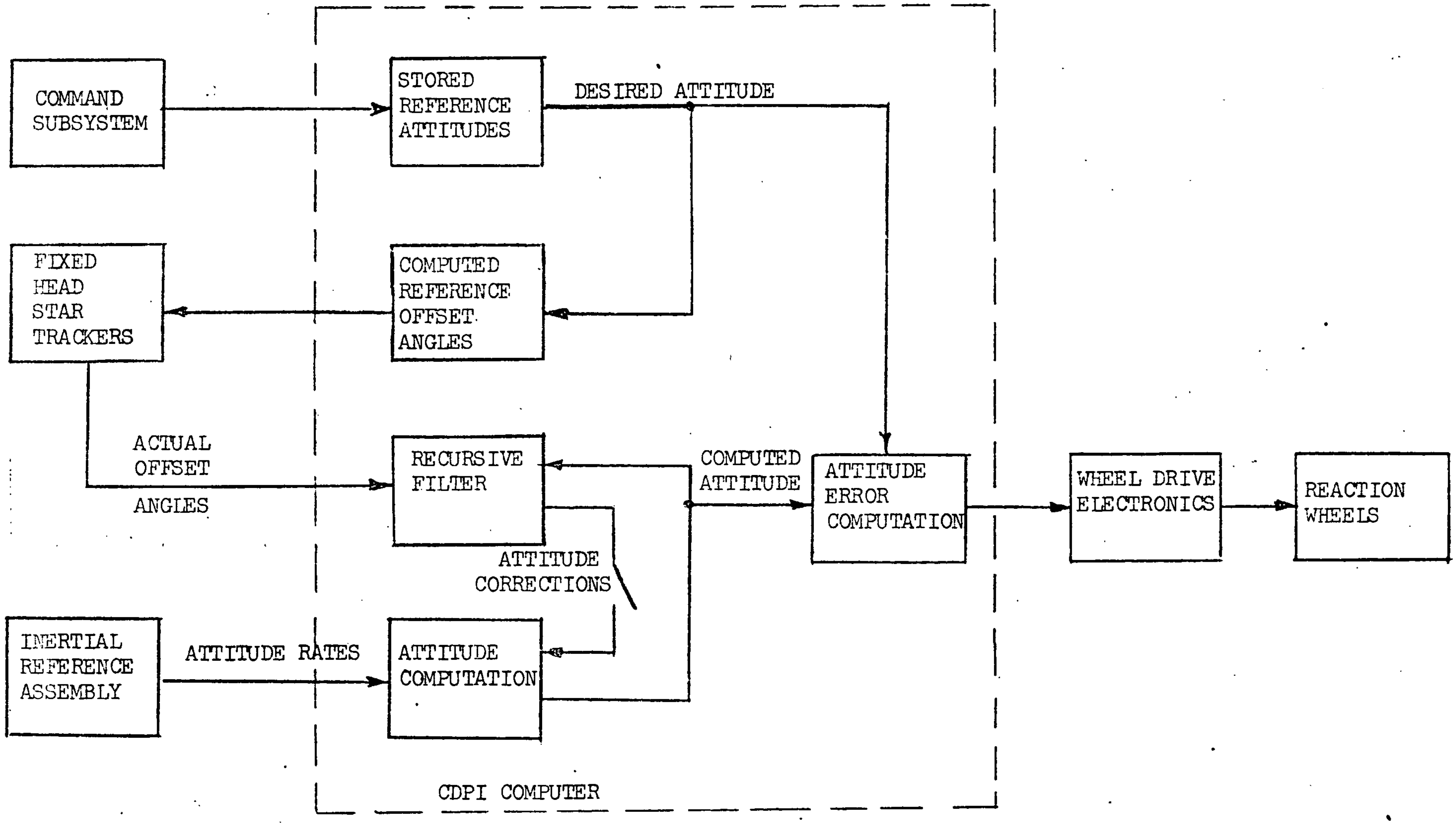

Fig. A3 Stabilization and Control Functional Flow

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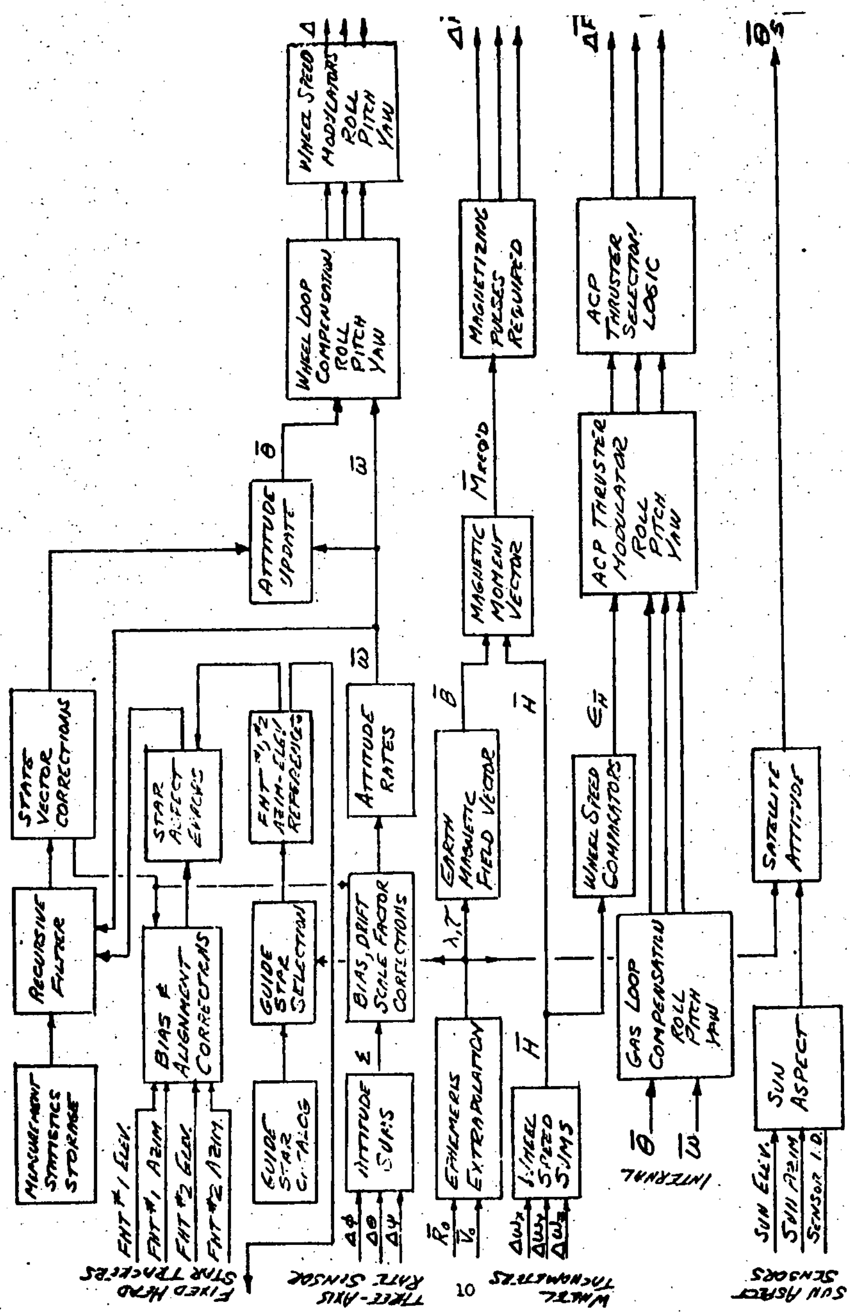

Fig. A4 S&C Subsystem On-Board Data Processing
(CDPI Subsystem)

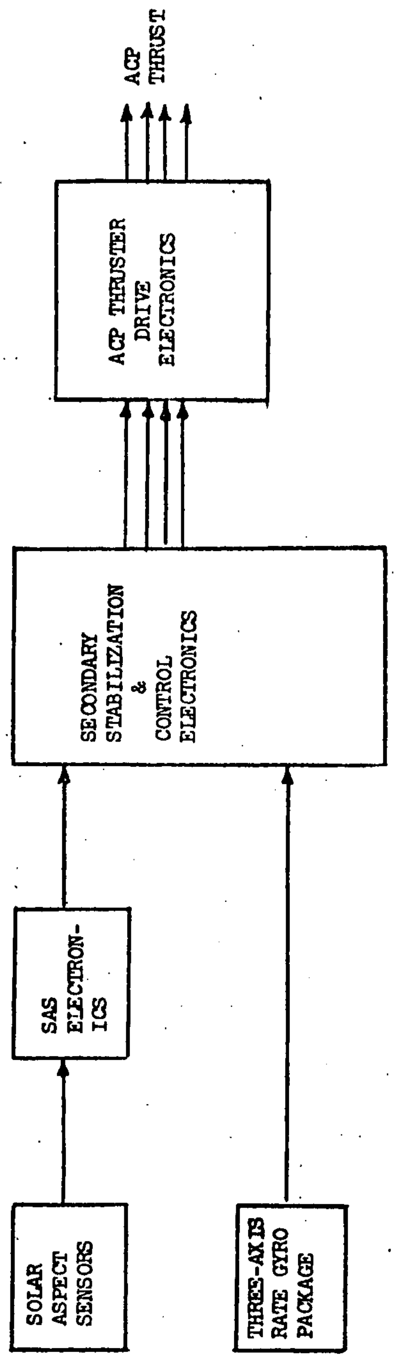

Fig. A5 Standard EOS Stabilization & Control Subsystem - Secondary

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(1) 3-axis rate gyro package (S&C)
(2) Coarse sun sensor (S&C)
(3) Electronics package (S&C)
(4) Cold-gas jets (3-axis)
(5) Cold Gas Tank
(6) Command Receiver/Decoder
(7) Battery
(8) Payload Shutters

Ideally, these parts would be packaged as an entity, independent of the primary S&C subsystem. At the other extreme, no special parts would be added, the functions all provided by the primary equipment. In this case there would obviously be much less assurance of having a quiescent satellite since the electronics and gas jets are tied to the normal system operation.

Figure A5 shows the functional and equipment relationships of the secondary operating mode, used only after an irreversible failure of a piece of gear of the primary mode, including the CDPI computer.

The backup or anti-tumbling system would either be commanded "on" or automatic under some set of on-board logic. The sequence of events would be:

(1) Rate Stabilization - Rate gyros null all rates

(2) Roll Search - Bias signal starts roll rate using cold gas

(3) Sun Acquisition - Sun in sensor FOV removes rate bias signal; sensor and rate gyros drive attitude to lock on sun at desired solar aspect.

(4) Attitude Hold - Sun sensor and rate gyros hold satellite to sunline. (Very slow rotation about sunline due to gyro threshold possible.)

B. Subsystem Implementation

Equipment Complement

The S&C subsystem implementation represents one variant of the Standard Spacecraft S&C Subsystem.

Table A1 shows the S&C subsystem equipment list with estimated weights and power.

Associated with each star tracker, the reaction wheels set, each solar aspect sensor, the three-axis rate sensor, and each magnetic torquer is an electronics package. Its functions will be tailored to produce a low bit rate digital electronic interface with the computer. Figures A6, A7, and A8 show the general arrangement of equipment in the three modules of the EOS S&C subsystem. Figure A9 shows the interconnection of the S&C subsystem modules with the CDPI subsystem and the ACS subsystem.

Table A1

LOW-COST EOS STABILIZATION & CONTROL SUBSYSTEM**

Item	Qty.	Unit Weight (lb)	Unit Power (watts)	Est. Failure Rates (hr x 10^6) & Duty Cycles (in %)	Total Weight (lb)	Total Avg. Power (watts)
Three Axis Rate Sensor/ Electronics	1/1	15	30	10 (100)	15	30
Fixed Head Star Tracker/ Electronics	2*/2*	13	11	5 (100)	26	22*
Sun Aspect Sensor/ Electronics	5/5	1/2	0/1	3 (1)	15	5
Reaction Wheels/ Electronics	3 /2*	18/9	4/9	2/6 (100)	72	21
Magnetic Torquers/ Electronics	3/3	2/1	1/30 (Pulse)	1(100/1)	9	4
Laser Corner Cubes	4	2	Passive	Passive	8	Passive
ACP Drive Electronics	4	7	12	3(25)	28	12
Rate Gyro Package (Backup)	1	2	15	10(1)	2	15
Secondary S&C Electronics (Backup)	1	5	5	10(1)	5	5
					180 lb	114 W

* Including redundancy, if required.
** Not including module weights or component mounting bases, brackets, pads, electrical connectors, cables, etc.

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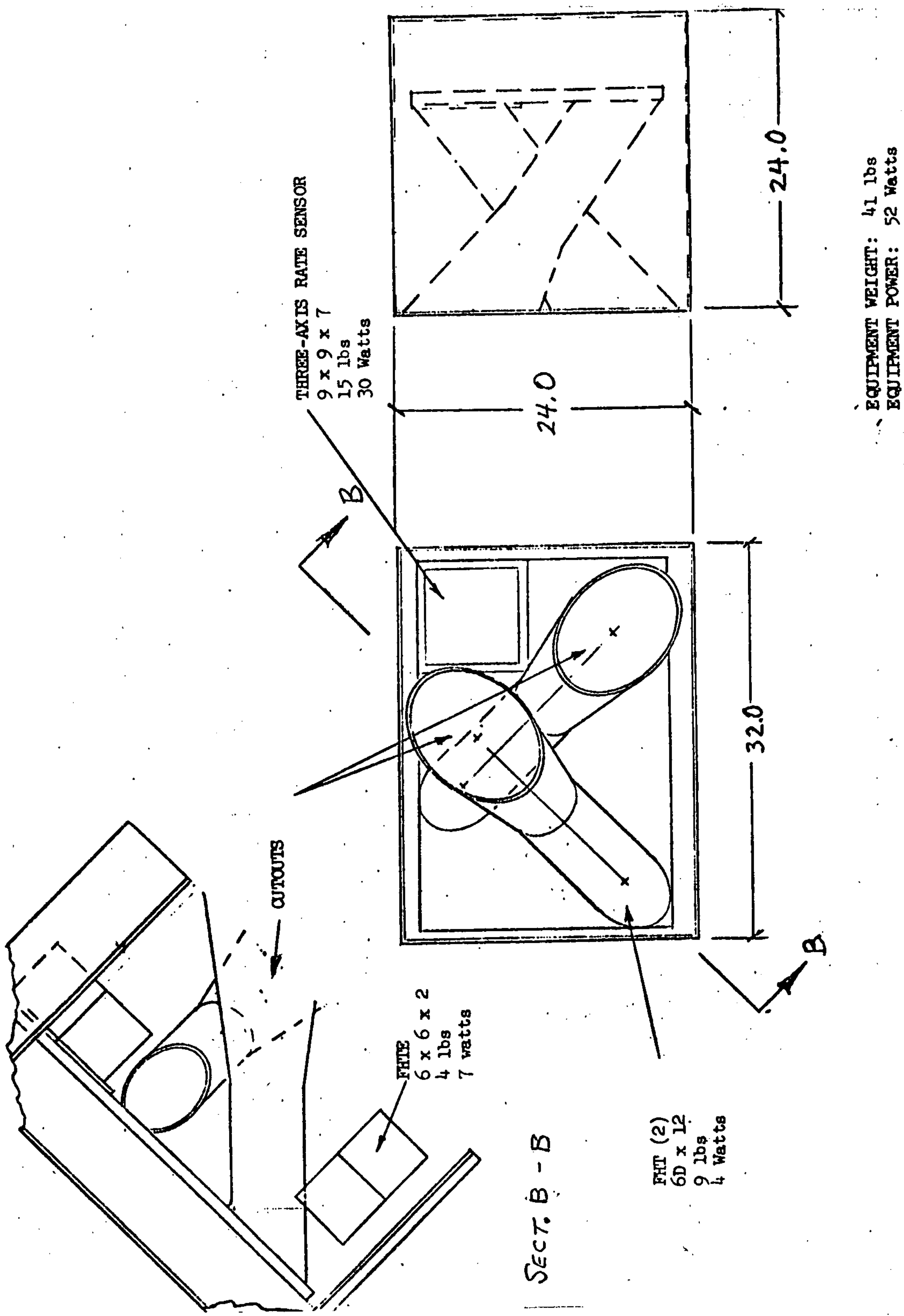

Fig. A6 Primary Sensing Module

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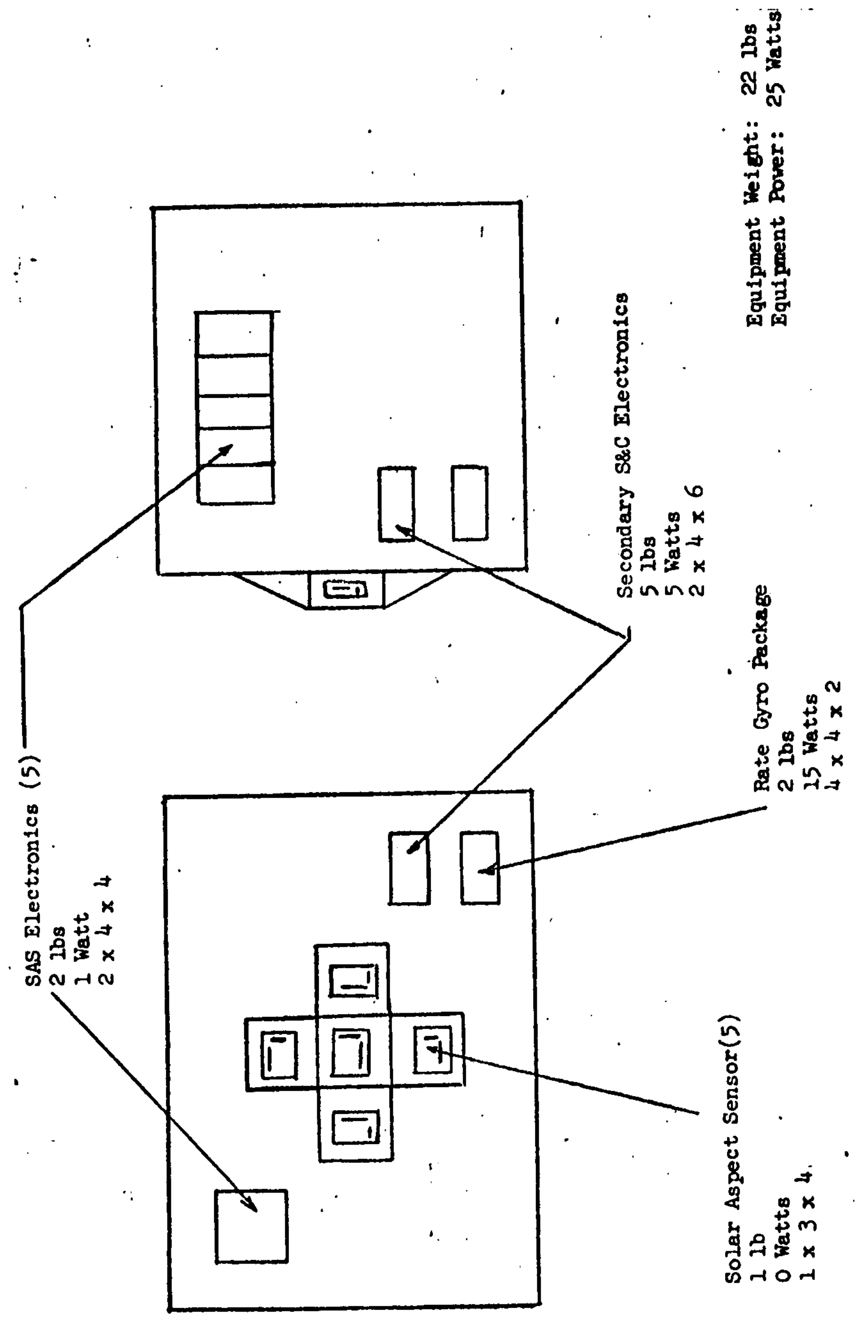

Fig. A7 Secondary Sensing Module

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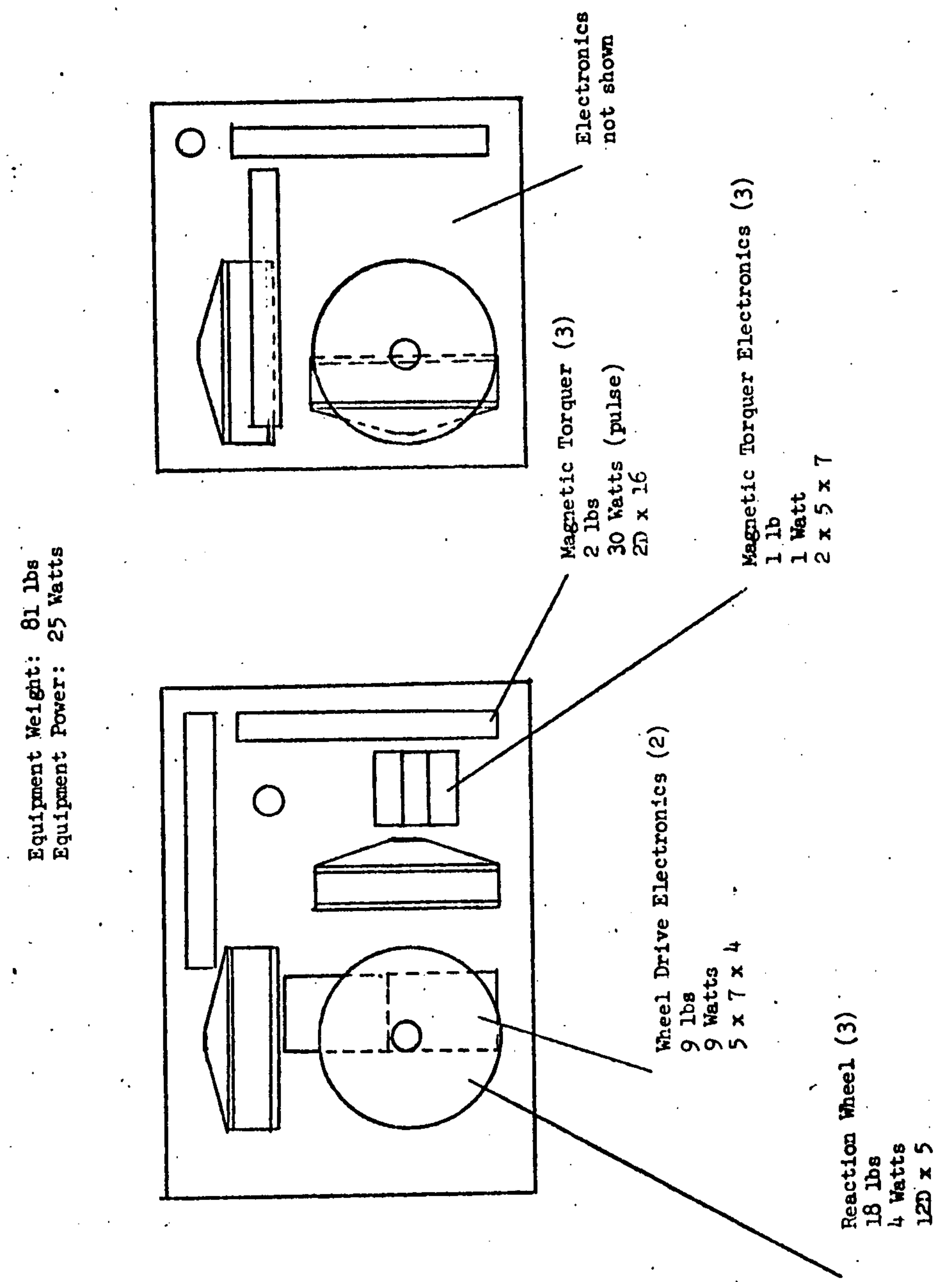

Fig. A8 Reaction Torque Module

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* Note: Each ACP Module contains one thruster drive electronics package

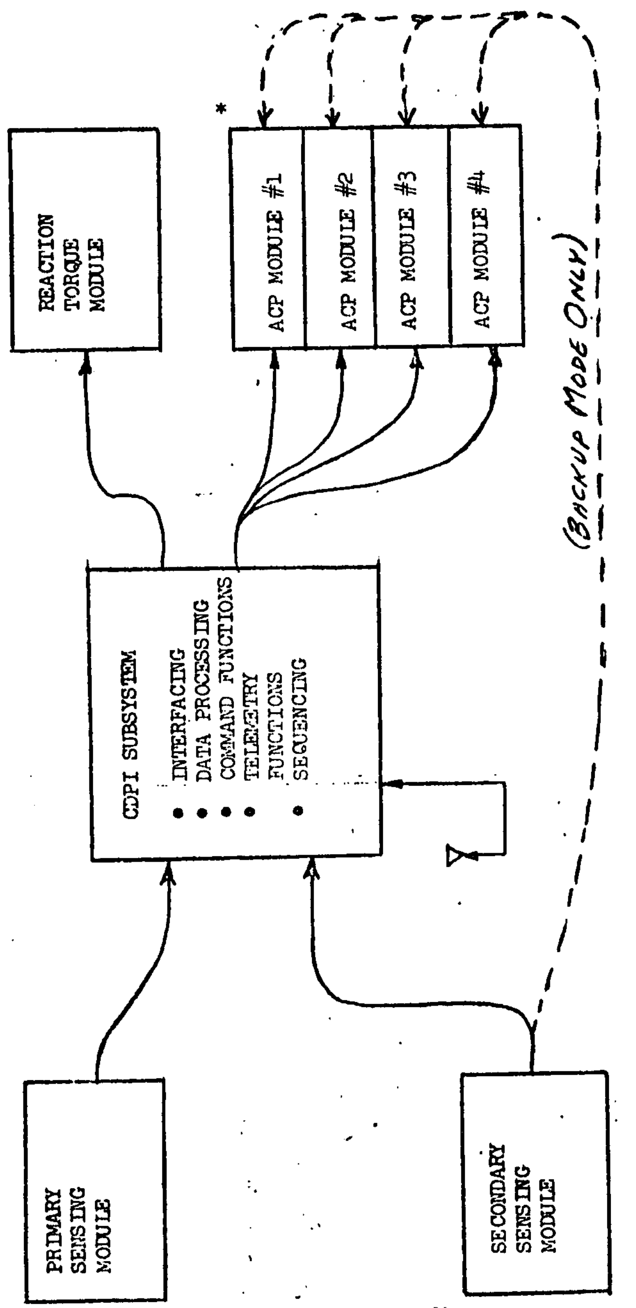

Fig. A9 Low-Cost EOS S&C Module Interconnections

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3.2 CDPI Subsystem

Introduction

This section of the report presents the background rationale and resulting representative design of the Communications, Data Processing, Interface and Instrumentation Subsystem (CDPI) of the standard spacecraft for astronomical missions, identified as the standard LAOS in this report. The predominant mission requirements imposed on the standard LAOS which have a major impact on the CDPI design include the following -

- Orbits - Spacecraft orbits are circular, at low inclination angles and in the range of 200 - 400 nm.
- Flexibility - Minimal changes need be made to the LAOS to account for variations in mission equipment configuration.
- Utilization - The standard spacecraft will fulfill the support functions and accommodate the following mission equipment categories - HEAO, LST, LRO, and LSO.
- Attitude - The spacecraft mission equipment must be capable of being oriented toward and track any designated object or point in the celestial sphere. This will include the capability for assuming and holding any attitude for specified periods of time.
- Accuracy ranges -
  General pointing: seconds of arc
  Locked-on pointing: milliseconds of arc
  Positional fixes: 1 - 10 Km
  Timing reference: 1 part in 10^9

If these requirements imposed on LAOS are compared to those established for the Earth Observatory Satellite (EOS), as described in References 1 and 2, and considered in terms of CDPI functional requirements, the following comparisons may be noted:

Item	Similarities and Effects	Differences and Effects
Orbit	Both are circular and low orbits. This enables utilization of a communication satellite system for command, telemetry and ranging for LAOS, as with EOS.	Orbit of EOS (500 nm) is higher than LAOS. Three communication satellites will be required to support the LAOS system.
Flexibility	Broad range of rates handled in both systems. This permits applying the methodology of CDPI design given in Reference 2 to the LAOS.	Maximum rates handled in LAOS may exceed the 50 Mbps set for EOS. More flexibility in mixes of signal characteristics needed in LAOS.

Utilization	Shuttle visit frequency is approximately the same for both spacecraft. Thus, the CDPI design characteristics, as established by reliability and maintainability requirements, are the same for EOS and LAOS.	Radical changes in mission equipment will be made with LAOS. EOS is a point design with allowance for growth.
Attitude	Orientation with respect to sun must be considered in EOS and LAOS.	EOS oriented with respect to local vertical; LAOS oriented with respect to any point in celestial sphere. Sun is an observation objective in LAOS.
Accuracy	Similar general pointing requirements.	EOS may require more accurate position fixes. Pointing direction sensing will be included in some mission equipment sensors. Pointing in EOS never uses mission equipment outputs for reference; LAOS does use the outputs to attain 10th sec accuracy.

The nature of these similarities and differences, as they influence CDPI design, suggest that a viable strategy in approaching the standard LAOS CDPI is to utilize the EOS version as described in Reference 2 as a baseline concept and introduce modifications to this baseline where required. It will be necessary to delineate the LAOS CDPI detailed functional requirements and definitize representative designs; however, in-depth discussion and analysis, e.g., approaches to standardization, uplink/downlink communication channel characteristics, etc., will not be included, except where differences with Reference 2 data are noted.

CDPI Functional Requirements

Basis for Requirements

A block diagram of the CDPI for the standard LAOS is shown in Fig. 12. Definition of the CDPI functional requirements for each section of the CDPI is dependent upon the input/output demands of the various subsystems interrelated with the CDPI. The primary difference between the Fig. 1 layout and the EOS counterpart is the multiplicity of mission equipment configurations which must mate with a common CDPI.

As pointed out in Reference 2 and Reference 3, a basic approach in reducing costs is to exploit commonalities in the system. Examination of the mission equipment requirements identified in Fig. 13 indicates some duplications in output rates, command requirements, etc. Thus, it would be possible to design to take advantage of these commonalities wherever possible, particularly with respect to the Interface Unit in the CDPI. This approach will, in general be used; however, the emphasis will be on meeting the requirements of each of the four total mission equipment packages.

A second basis of establishing the requirements is that simultaneous operation of arbitrary mixes of mission equipment is not included. This pertains in particular to

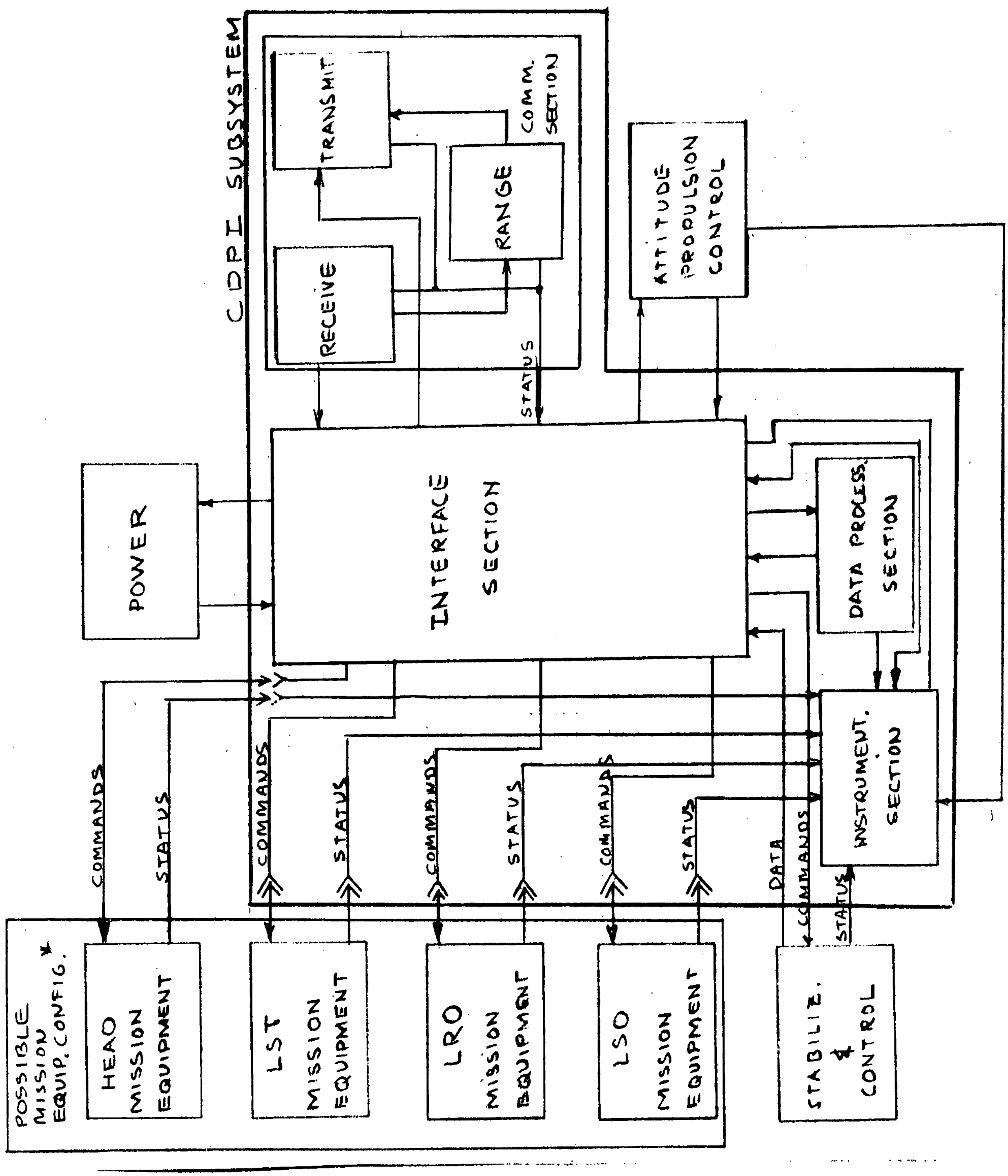

Fig. 12 Standard LAOS - CDPI Subsystem Block Diagram

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Mission	Experiment	Data Rate	Data Qty.	No. of Channels	Comments & Rationale for Rates & Qtys.	Comm. & Cont.	Instrumentation (Assumed)
HEAO-A	Focusing X-Ray Experiment	4.6 Kbps for sci. info. 12.5 Kbps Hi Rate	60×10^6 bits	8-11 bits per channel on proportion counter	Specified - Extrapolation from data on proport. counter array	30 assumed	15 Analog 15 Digital
	Comb. Mod. Collimator	2.7 Kbps	Undet.	6 prop.count arrays 2 scan mod. collim. 4 rotating mod. collim.	Specified	4	12 Analog 4 Digital
	Laxray	4.5 Kbps max rate	Undet.	6 assumed (one per module)	Clock accuracy of one part in 10^9 specified	12 assumed	12 Analog 12 Digital
	Gamma Ray Telescope	1.6 Kbps	Undet.	7 29 photompy. tubes	Scintillation counter assy. - 7 counters in unit	21	14 Analog 14 Digital
	Hexray	.5 Kbps	Undet.	6	Pulse height analyze sampled 10 times per sec	100	16 Analog 4 Digital
	High Energy Cosmic Ray	4 Kbps	Undet.	6 assumed	Pulse height analyze 12 Bit Buffer included in equip.	6	15 Analog 6 Digital Calib. & Count Rate
	Heavy Nuclei	1.2 Kbps	Undet.	6 Modules	Data multiplexer included in Exp. Package	6 assumed	12 Analog 12 Digital

Fig. 13a Mission Equipment Input/Output (Sheet 1 of 4)

Mission	Experiment	Data Rate	Data Qty.	No. of Channels	Comments & Rationale for Rates & Qtys.	Comm. & Cont.	Instrumentation (Assumed)
HEAO-B	Diffuse X-ray meas.	T/M - 2Kbps Sample 300 ms Pulsar ~ 1 ms 4.6 Kb/3 sec 1.5 Kb/60 sec other slow rates	Event Oriented 3 & 5 Analog per event	2	Specified	24 assumed	15 Analog 15 Digital
	Bragg Xtal X-ray Spectr. (3)	7 counters Calib. 5 Drive Modes 3 Det & Count 1.8 Kbps TM	Each anode 16 Ch. pulse Ht. anal. + 1 128 Ch. analyzer	3	Computer stated as part of experimental pkg.	100 assumed	30 Analog 15 Digital
	High Res. Gamma Ray Spectr.	2.5 Kbps	PHA - 13 bits plus time count rates	12 assumed	Clock accurate to 1 part in 10^7	18 + 1 data word	15 Analog 15 Digital
	High Energy Gamma Ray Spark Chamber Telescope	3 Kbps TM	Undet.	4 assumed	9 trigger modes special Pulsar mode; 38 spark chamber modules - scint. counters & shower counter	20 relay 128 data cmds.	40 Analog 40 Digital
	Superconducting Mag. Spectrom.	6.84 Kbps max	Undet.	4 assumed	Number of Scint. counters indef.	9	12 Analog 4 Digital
	High Energy Electron Exp.	1.8 Kbps	Undet.	5 assumed	Plastic Scint. counters; Cerenkov counters logic included	16 assumed	6 Analog 6 Digital
	Isotopic & Charge Compos. of Cosmic Rays	1 MHz ± 10% 10^{-3} stab. 20 KHz-DC-DC conv. 5 KHz for A/D conv. Output undet.	Undet.	19 analog CH. - 0-5V 1% acc.	Counter outputs beside Vidicon view of 2 spark chambers	8 assumed	8 Analog 16 Digital

Fig. 13b Mission Equipment Input/Output (Sheet 2 of 4)

Mission	Experiment	Data Rate	Data Qty.	No. of Channels	Comments & Rationale for Rates & Qtys.	Comm. & Cont.	Instrumentation (Assumed)
LST	Stellar Observ. Electron Imaging	10^3 bps for experiments images/sec-$1.15x10^{-5}$ to 10 $7x10^6$ bps Acc.	1.12 x 10^3 bits permage	3	High resolution Electronic imaging	4 assumed	10 Analog 10 Digital
	Spectroscopy	$1-4.64x10^{-5}$ images/sec	$7x10^5$ bits/image	3 assumed	3 types of spectrometers employed	6 assumed	15 Analog 15 Digital
	Polarimetry	$1-4.64x10^{-5}$ images/sec	10^6 pixels/image	1	Det. of polarity by rotation assumed	4 assumed	4 Analog 4 Digital
LRO - Earlier Experiments Quoted	Radiometer Detector Assys. from UHF through Microwave	90 KHz freq. spread on analog; 90 Kbps digital	Reading of freq. & amplitude at each freq.	7 assumed	Frequency sweep by step increments-compare to noise reference source	14 assumed	12 Analog 12 Digital
	Burst Receiver	Random Bursts < 10 per sec assumed	Undeter.	8	Determine freq. content & ampl. of burst. Noise source included for calibration.	2 assumed	10 Analog 4 Digital
	Supercooled Bolometer Assembly	10 Hz freq. spread	Undeter.	3	Power Level measure	3 assumed	3 Analog 6 Digital
LSO	1.5 Meter Photoheliograph measurements. Assume imagining & TV transmission	Analog 4.2 MHz; Digital $0.4-4x10^9$ bits per image set Exp. Data 2.1 Kbps	90 images/set	4	Simultaneous time correlated measure. of moving solar phenomena	3 drives 2 cameras 2 magnetographs	30 Analog 15 Digital

Fig. 13c Mission Equipment Input/Output (Sheet 3 of 4)

Mission	Experiment	Data Rate	Data Qty.	No. of Channels	Comments & Rationale for Rates & Qtys.	Comm. & Cont.	Instrumentation (Assumed)
	0.25 meter	Analog 2.9 MHz; 1.75×10^{8} bits/image 0.1 to 1 image sets/sec	2.824×10^{7}	3	Simultaneous time correlated data 2.833 min/cycle	20 assumed	45 Analog 30 Digital
	0.5 meter D. X-ray telescope measurements	1.93×10^{-6} bits/sec 10 Kbps exp. data	2.756×10^{5} pixels/image set	6	Prop. counter, spectrum. & imaging system included-other instruments may be added	10 assumed	35 Analog 35 Digital
	Solar Coronograph meas.	4.609×10^{6} pixels/image set 5.5×10^{-3} to 1 image sets/sec	1.2×10^{7}	2	Two instruments utilized	30 assumed	20 Analog 15 Digital
	Magnetograph	2.178 Mbps 104 Mbps raw data with line scan; 10 Mbps with Doppler Zeeman	5.41×10^{7} bits	2	Two imaging instruments	4 assumed	10 Analog 5 Digital

Fig. 13d Mission Equipment Input/Output (Sheet 4 of 4)

the LSO configuration, a spacecraft containing a multiple of experiments which have data rate requirements in excess of 10^7 bits/second. Time sharing of the data channels is of course applicable and will be used in the CDPI design; however, exact detailed mechanization of time shared and multiplex modes should be given further study.

Peripheral Units

Mission Equipment Description

The mission equipment is grouped into the four general spacecraft configurations of HEAO, LST, LRO and LSO. Each configuration has, in turn, a number of versions, e.g., HEAO A through D. At the present time the formulation of the many versions are in a state of flux. Thus a definitized set of mission experiments for all configurations cannot be established at this time. For this reason it will be necessary to assume utilization of typical experimental equipment. The information in Reference 4-6 will be utilized as a basis for the assumptions.

Figure 2 contains a typical listing of the mission equipment presently envisioned for each spacecraft configuration. In some instances the reference sources do not supply complete data. In these cases estimated values are included to serve as a model wherever possible.

Mission Equipment Requirement

Review of the descriptive material given in Fig.13 indicates that ultra high data rates, 10^7 bits per second or higher, are obtained with two types of processes - imaging and graphing, e.g., imaging vidicons and line or Zeeman magnetometers. In all the spacecraft systems only the LSO has a multiplicity of ultra-high data rate outputs. Such rates cannot be simultaneously processed via the CDPI, utilizing present day technology. Similarly, those rates that exceed 60 Mbps will necessitate utilization of new technologies. For example, a value up to 4×10^9 bits/image set is given for the 1.5 meter photoheliograph. If this quantity of data must be transmitted each minute, a laser communication system would be required. Although laser communication systems are not within the scope of the present study, such systems might prove advantageous in future efforts.

Assuming the appropriate restrictions on data rates, i.e., maximum acceptable values and limiting simultaniety of mission equipment outputs, to remain within these rates, the following requirements may be stipulated:

Maximum number of experimental units:	7
Actual data rate ranges*:	
low frequency	< 0.5 - 6.84 Kbps
medium frequency	90 Kbps (may be integrated with low or high frequency range)
high frequency	1 - 10 Mbps
ultra-high frequency	10 - 60 Mbps

* Ranges equiv. to Ref. 2, i.e., Ultra high > 30 Mbps, Med. & High 0.01-1 Mb/sec, Low < 20 Kbps.

Maximum number of digital output channels/unit	12
Maximum number of analog output channels/unit	19
Maximum analog frequency (TV video)	4.2 MHz

Maximum Mission Equipment Status Monitoring and data words as follows:

Analog	130	
Digital (bilevel)	115	
A/D converted Digital (words)	113	(one per channel assumed)
Direct Digital (words)	84	
Maximum commands	316	

Stabilization and Control Requirements

In accordance with Section 3.1 of this report the Stabilization and Control Subsystem (S&C) for the standard LAOS is essentially the same as that used for the Standard EOS, except for four main differences:

- More extensive computational algorithms required plus slew and maneuver logic
- Primary attitude references are also star trackers; however, a larger number is used in LAOS
- Mission Equipment has a functional interface with S&C for precise attitude control
- More complex control selection logic required to match the increased S&C subsystem instrumentation.

These differences will effect the CDPI in the following areas:

- Increased CDPI computer memory - 25% increase over EOS.
- Increased required computer speed - 20% increase over basic EOS requirements
- More S&C interface hardware and logic - 5% increase in EOS circuitry.
- More extensive software - 25% increase over EOS requirements

Since the other S&C requirements are delineated in Section 3.1 they will not be reiterated here except for the telemetry and other data requirements. These include:

- Commands - 50
- Uplink data - up to 300 32 bit words
- Downlink data - 110 quantities - 32 bilevel

Attitude Control Propulsion

The Attitude Control Subsystem (ACS) discussed in Section 3.4 identifies the primary requirements imposed on the CDPI. The ACS layout is shown in Fig. 34. The requirements include the following:

- Four propulsion modules with four thrusters per module
- Selection of redundant couples to compensate for module malfunction
- Response to S&C mission equipment or ground generated commands
- Telemetry output of driven pulses - 25 quantities - 24 bilevel

Power Subsystem

The Power Subsystem will require support from the CDPI to provide control switching functions and engage redundant sections. The majority of the discrete controls are for the mission equipment and the S&C; however, the Communication Section of the CDPI will also require discrete control of the applied power. Approximately 180 power applied controls will be necessary to augment the command controls of the various components. Since mission equipment will be changed for different spacecraft flights, voltage setting controls may prove necessary for gain and bias settings. Such changes may also be needed with S&C components. Fifteen voltages over the range of 0-28 vdc will be assumed sufficient to meet requirements.

CDPI Sectional Requirements

Communication Section

The mission equipment descriptions defined in Fig. 13 indicate that the majority of experiments have low frequency data rates. Although ultra-high data rates must be handled, practically no medium to high data rate equipment appears. The one case of 90 Kbps, associated with microwave radiometry, is an estimated upper bound. If the predicted requirements in Reference 6 hold true then the number of missions using such equipment would be very low. This suggests two possible changes to the standard EOS CDPI communication section implementation requirements to relax demands on the LAOS CDPI - elimination of S-Band and extension of K-Band and VHF utilization. These changes suggest application of a Communication Section equivalent to Option 4, as described in Reference 2. The frequencies and equipment configuration of Option 4 are reiterated in Fig. 14 for ease of reference. This approach, though possessing a number of advantages and adaptable to EOS as well as LAOS, is not recommended since it makes the entire program dependent on K-band developments and does not utilize the projected relay satellite capability (TDRS Mark 1-C assumed) to the fullest extent. This is particularly important when considering requirements for multiple TDRS/User Spacecraft Operations. For these reasons, a configuration requirement equivalent to the standard EOS communication section, as defined in Fig. 15 will be established as a baseline. It should be noted that this approach will lead to overdesign of the CDPI Communication Section for LAOS applications; however, the cost advantages of standardization would offset this disadvantage. The basic functional requirements identified in Fig. 16 would then apply.

Function	Implementation
Command	1. Digital - Data Rate - 10 Kb/s 2. Freq. - Primary K-Band, VHF Backup 3. Modulation - S/K on Range Code on K-Band, PCM/Bi-Phase or FSK on VHF 4. Output - Serial PCM to Computer for Command Verification & Decoding
Telemetry	1. Digital - Data Rate - 1.2 Mb/s 2. Frequency - K-Band, VHF Backup (Low Rate) 3. Modulation - S/K on Returned Range Code 4. Input - Serial PCM from Multiplexed Spacecraft status and Low Rate Experiments
Wide-Band Data	1. Primary - Digital - Data Rate - 30 Mb/s Optional - Analog - Bandwidth - 10 Mb/s 2. Frequency - K-Band 3. Modulation - Digital PCM/QPSK on carrier Analog - FM on carrier 4. Input - Serial PCM directly from experiment
Ranging	1. Primary - PN Range Code 30 Mb/s clock Backup - VHF Minitrack 2. Frequency - Primary K-Band, Backup VHF 3. Modulation - PCM/Bi-Phase on Carrier (primary)
Acquisition & Tracking	1. Rough pointing by computer 2. Coarse pointing cooperatively by de-focused K-Band beam 3. Fine pointing by focused K-band beam

Fig. 14 Frequencies & Equipment Configuration vs Function -- Option #4

Function	Instrumentation Requirements
Command & Computer Update	1. Data Rate - 10 Kb/s 2. Frequency - S-Band (Primary), VHF 3. Modulation - SIK on Range Code at S-Band 4. Output - Serial PCM to Computer for Command Decoding or Computer Update
Telemetry & Medium Rate Data	1. Data Rate - 50 Kb/s on S-Band 10 Kb/s on VHF 2. Frequency - S-Band (primary), VHF 3. Modulation - QPSK on S-Band 4. Input - Serial PCM from VIP or MIRP
Wide Band Data	1. Primary - Data Rate 1 - 60 Mb/s Optional - Analog Bandwidth 5 MHz 2. Frequency - K-Band (15.3 GHz) 3. Modulation - Digital - PCM/QPSK on Carrier Analog - FM on Carrier 4. Input - Serial PCM from MOMS
Ranging	1. Digital - PN Range Code (1 MB/s clock) 2. Frequency - S-Band (both directions) 3. Modulation - Uplink - Range Code SIK'ed by Command Downlink - Range Code and TM in quadrature on carrier
Acquisition & Tracking	1. Rough pointing from computer for TDRS acquisition 2. Coarse pointing at S-Band ($\pm 1^{o}$) 3. Fine Pointing and Tracking at K-Band ($\pm .1^{o}$)

Fig. 15 Frequencies & Equipment Configuration Requirements vs Function - Option #1

	Item	Number
Uplink Communications	Commands Ephemeris data Orbit adjust data Verification information Initialization information Test & diagnostic data Frequency bands	1024 words 7 terms 4 terms 3 words Undetermined Undetermined S & VHF
Downlink Communication	Housekeeping & Status Monitoring	545*
	Mission Equipment Data Verification information On-board data processing results S&C data APC data	84 words 1-3 words Undetermined 24 words 24 words
Ranging	Delay lock-loop Correlation Receive and transmit Range codes	N/A 10-32 bit word at undetermined rate
Power Output (ERP)	VHF S-Band K-Band	17 dBW 40 dBW 62 dBW
Antennas	VHF S, K-Band (gimballed)	Omnidirectional 6' Dish

* Mixture of Low & High Analog values, bilevel and digital quantities.

Fig.16 Standard LAOS CDPI Communication Section Functional Requirements

Instrumentation Section

Instrumentation requirements are subdivided between the spacecraft subsystem and mission equipment needs. The estimated mission equipment instrumentation has been identified in Fig. 13; spacecraft instrumentation is estimated as follows:

Pressure transducers	34
Temperature sensors	101
Current sensors	30
Voltage sensors	25
Total	190

All these instrument outputs will need signal conditioning.

The estimated requirement for the spacecraft subsystems have been given in the previous sections. A summarization of the total estimated requirements are as follows:

	Analog	Bilevel
Mission Equipment	130	115
S&C Subsystem	110	32
AC Subsystem	25	24
Power	15	30
CDPI Subsystem	40	24
Totals	320	225

In order to permit handling this number of signals multiplexing will be required. Two types of multiplexing will be necessary - analog sampling, with A/D conversion, and bilevel sampling and bit assignment. It should be noted that mission equipments, S&C, etc., instrumentation sensors will be supplied as part of the hardware in the various subsystems. Thus the requirements identified here are meant to establish multiplexing and signal conditioning needed in the CDPI Interface Unit.

Data Processing Section

In defining the requirements of the Data Processing Section of the CDPI, the subsystems with major influence are S&C and mission equipment - S&C governing primary computational demands and mission equipment command sequencing. Both types of processing are increased compared to EOS; this will necessitate some additions to the requirements imposed on the CDPI computer. It should be noted, however, that the majority of the requirements established for EOS remain the same for the standard LAOS. Thus to simplify the presentation the requirements given in Reference 2 will be reiterated here; however, those entries which deviate from the reference data will be appropriately modified.

Word Length

The accuracy requirements on high precision pointing are in excess of one part in 10^6. High accuracy will also be needed for the S&C computational processes. These needs

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suggest a word length of nominally 20-24 bits. A shorter word length could be used, e.g., 16 bits; however, the longer word length is preferred.

Memory Capacity

The memory capacity required is based upon the following word count estimates. Double precision computations are included.

Function	24 bit Word Totals
Stability & Control	4200
Sequencing & Command Processing	4000
Telemetry Formatting	350
Ephemeris (short arc)	384
Commands	1024
Star Data	545
Received Data & Command Tests	350
I/O function routines, etc.	700
AGE/Shuttle routines	250
Attitude Control	250
	12053 words

This indicates a 12K 24 bit memory would meet requirements; however, a 16K 24 bit memory is specified to allow for growth items.

Memory Type

A random access memory is required, with non-destructive readout (NDRO) desired. Read-only memory sections may be used for the more critical subroutines; however, selective write circuit lockout under program or command control would also be effective.

The availability of direct memory accessing (DMA) is desired to meet flexibility requirements. DMA allows for entering or extracting data from memory without interrupting the operating program sequences. The DMA may be obtained via cycle stealing, or separate addressing control may be used.

Operating Speed

Speed requirements are not definitive at this time; however, nominal values of 8 microseconds add time and 20 microseconds multiply time appear reasonable. Higher speeds would be most desirable however.

Instruction Repertoire

A standard mix of input/output, arithmetic, logical, memory addressing and indexing should suffice; however, augmentation with external interrupt is required. If the word length is less than 20 bits, double precision add, subtract, store and load instructions are required.

Index Registers

A minimum of one index register is required; however, multiple index registers could provide desired flexibility.

Arithmetic Type

There are no specific performance requirements on arithmetic. The standard 2's complement, fixed point, parallel computer would be adequate. A double length accumulator is necessary with word lengths less than 20 bits.

Input/Output

Input and output should be through two mechanizations - computer control and DMA. Buffer registers are an option since they may be supplied via the external Interface Unit. Both serial and parallel channels are desired.

The computer must handle a minimum of three separately identifiable interrupts - input or register, timing cycle and power fail. Other interrupt handling capability is desired. Resumption of power after a power fail shall cause the computer to start at a specified address, equivalent to a power-on interrupt.

Issuance of fixed and variable discretes under address control are desired. The time duration of the fixed discretes are TBD.

Weight, Size, Power

Although specific weight, size and power requirements are to be determined, a fourth generation technology is assumed. This would result in a computer with the following characteristics:

Weight	< 35 lbs
Size	< 1000 in^3
Power	< 75 W

Interface Section

The primary functional requirements of the Interface Section of the CDPI are to provide data routing and control paths to the various spacecraft subsystems, mission equipment, data processor and communication section Data routing includes the following:

- Accepting, sampling, subcommutating, multiplexing and converting analog instrumentation output for computer input.
- Accepting, sampling and delay of bilevel data for computer input.
- Accepting and delivery of S&C outputs to the computer.
- Relay of information from the computer to telemetry, AGE or Shuttle-based equipment.

- Accepting, delaying and delivering mission equipment data outputs to the computer or communication section.

The control processes required are as follows:

- Accept computer stored or verified uplink commands and deliver discretes to the appropriate subsystem via discrete control sequencing logic. This logic is required to operate the mission equipment through switches and other controls. It also governs power application to the various subsystems and operates gating circuits which trigger the required holding and shift registers in the Interface Section.

- Deliver conditioned drive signals to the S&C momentum wheels and other components, ACS fit thrusters and mission equipment controls.

- Provide timing references for synchronizing and operating logic. Rate conversion to meet subsystem and mission equipment requirements must be provided. The basic timing reference accuracy is one part in 10^9.

Quantification of the various channels has been discussed previously in this section.

CDPI Design

Communications Section

A block diagram of a representative design of a Communication Section for the standard LAOS CDPI is shown in Fig. 17. This design is in accordance with Option 1 of the standard EOS described in Reference 2. As seen in the figure, the Communication Section is in three parts - a K-Band configuration devoted exclusively to ultra-high data rates, an S-Band unit for command input and verification and low data rate telemetry, and a backup VHF unit. Backup consists of enabling communication for activation of redundant components, performing those mission functions, operable after component failure and safing the spacecraft in the event of failure of automatic safing processes. If the failure occurs in the S-Band equipment, ranging data would be lost. A separate K-Band ranger might be included in the design; however, this would overly complicate the equipment. Redundant components are employed in the Communication Section to minimize the probability of loss of communication capability between Shuttle revisits.

The primary specifications for each major Communication Section component is given in Fig. 18, and a listing of estimated weight, size, power and costs of the components is contained in Fig. 19. These data are the same as are contained in Reference 2.

It should be noted that status monitoring and control of the Communication Section components will be required. Interconnection for these functions are provided via the Interface Section units.

Interface Section

In order to satisfy interface requirements and remain consistent with the Standard EOS design, the Interface Section will contain three parts - an ultra-high data rate unit for interfacing the 10^6 and higher output rate mission equipment with the K-Band

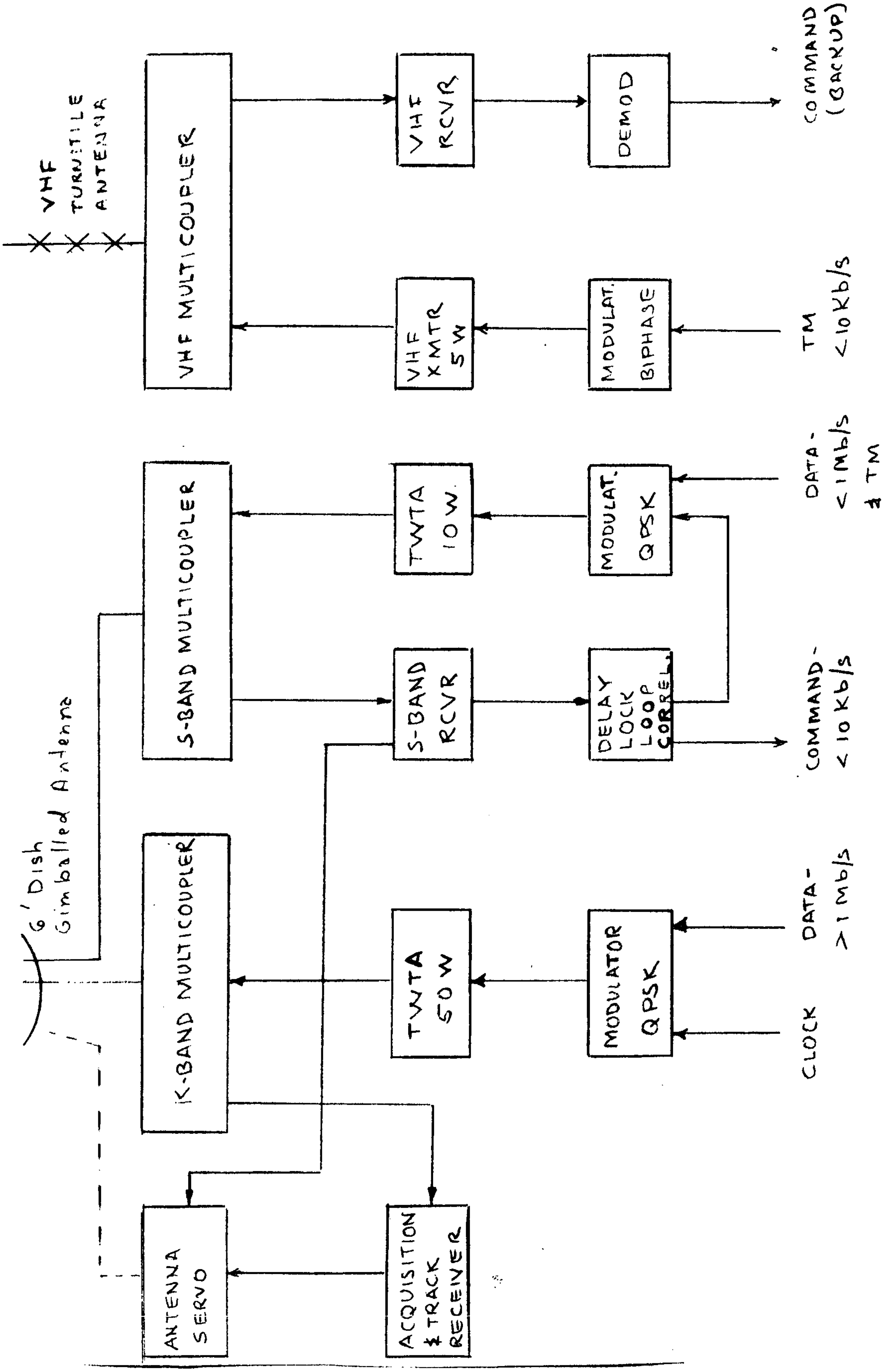

Fig.17 Standard LAOS CDPI Subsystem Communication Section Block Diagram

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1. K-Band TWTA	50 watts out, Gain + 20 dB
2. K-Band QPSK Mod.	60 Mb/s in, Output ~ 0 dBW
3. K-Band PLL Rcvr.	$BW_{(RF)}$ = 200 MHz, NF = +12 to +14 dB
4. K-Band Multicoupler	1 XMTR, 1 RCVR, Power 100 Watts, Loss < 1 dB
5. S-Band XMTR	10 Watts out, input - up to 90 Kb/s, QPSK Mod
6. S-Band PLL RCVR	$BW_{(RF)}$ = 20 MHz, NF = +5 to +7 dB
7. S-Band Multicoupler	1 XMTR, 1 RCVR, Power 30 Watts, Loss < 1 dB
8. Delay Lock Loop Correlator	Cannot define without more study
9. VHF Transmitter	5 Watts out, 10-20 Kb/s Input, Bi-Phase Mod
10. VHF RCVR & Demod.	BW = 50 KHz, NF = +2 dB
11. Antenna Servo Electronics	Cannot define without more study
12. Parabolic Antenna	Diameter 6 ft, gain at S-Band +30 dB, Gain at K-Band +45 dB, Dual S&K-Band Feeds, Rotary Joints (waveguide & coax) Total line loss < 2 dB at both bands
13. VHF Antenna	Turnstile or equiv, gain + 10 dB Total line loss < 1 dB

Fig. 18 Component Specifications

Component	Weight (lbs)	Power (watts)	ROM Cost x 10^3		Size
			Non-Re-curring	Recur-ring	
1. K-Band TWTA (50W out)	12	250	500	50	12"x4"x4"
2. K-Band QPSK Mod/Driver	3	10	100	20	3"x3"x2"
3. K-Band PLL Receiver (Acq. & Track)	3	5	100	15	3"x3"x2"
4. K-Band Multicoupler	3	-	20	5	4"x4"x2"
5. S-Band Transmitter (10W out)	6	80	100	20	6"x6"x6"
6. S-Band Receiver	3	5	50	10	3"x3"x2"
7. S-Band Multicoupler	3	-	-	5	2"x2"x2"
8. Delay Lock Loop Correlator	2	4	20	5	3"x3"x3"
9. VHF Transmitter (5W)	6	40	10	5	4"x4"x4"
10. VHF Receiver & Demod.	3	5	10	5	3"x3"x2"
11. Antenna Servo Electronics	8	20	30	10	8"x6"x6"
12. Antenna Servo Motors & Gears	15	20	-	-	(Part of Antenna Assy)
13. Antenna Feed (S&K Band)	4	-	-	-	(Part of Antenna)
14. Antenna (Dish - 6 ft)	10	-	1000	100	75" dia. x 30"
15. Antenna Mount (6' dish)	10	-	-	-	(Part of Antenna)
16. Antenna (VHF)	5	-	20	5	30"x10"x10"
17. Rotary Joint (K-Band)	3	-	50	5	Part of Antenna
18. Rotary Joint (S-Band)	2	-	10	5	Part of Antenna
19. VHF Multicoupler	1	-	-	1	2"x2"x2"
20. S-Band QPSK Mod/Driver	3	10	100	20	3"x3"x2"

Fig. 19 Communication Section Component Description

transmitter, a low data rate unit for mission equipment outputs to the S-Band Transmitter, and a low data rate unit to handle S&C, Data Processor, ACS and other functions as they relate to S-Band and VHF equipment.

A representative design of the Ultra-High Data Rate Unit is shown in Fig. 20. This unit is used only for LSO missions since the experiment output data rates from the other mission configurations is generally less than 10 Kb/sec. The design is a modified version of the standard EOS unit. The primary changes are as follows:

- Addition of selection and routing of mission equipment data. This enables selective output of specific channels from the mission equipment as well as routing to direct input to the K-Band modulator or a high data rate multiplexer.
- Addition of a multiplexer. This permits multiplexing the outputs from the various channels of a specific experimental unit or intermixing the outputs from multiple mission equipment. The latter application may be used if the sum of the data rates is less than 60 Mb/sec.
- Inclusion of a coupler filter and amplifier which accepts the multiplexer output or the direct output. The analog outputs from the spectroheliograph will bypass the multiplexer.
- Elimination of one wide band amplifier and addition of flip-flop, gating circuits, etc.

These changes will necessitate additions to the number of components used in the EOS unit. Figure 21 contains a listing of the required components.

Assuming packaging with plug-in logic boards, except for the components designated in Fig. 21, and utilizing cabling interconnect circuitry in addition to the mother board, the following estimates are obtained.

Unit	No.	Total Wt. (lbs)	Total Power (Watts)	Total Size (ft^3)
Logic Board	4	7.5	15	0.2
Router	1	2.0	8	0.1
Multiplexer	1	2.0	6	0.1
Coupler	1	0.5	2	0.1
Grand Totals		12.0 lbs	31 W	0.5 ft^3

The cost for the unit would be approximately as follows:

Logic Boards	Base Cost	End Cost
Art work fabrication	$1400/board	$ 5400
Components (TTL + FET)	300	1500
Wiring & Packaging Logic	2500	2500
Estimated Cost for Logic		$ 9400

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DATE: 31 March 1972

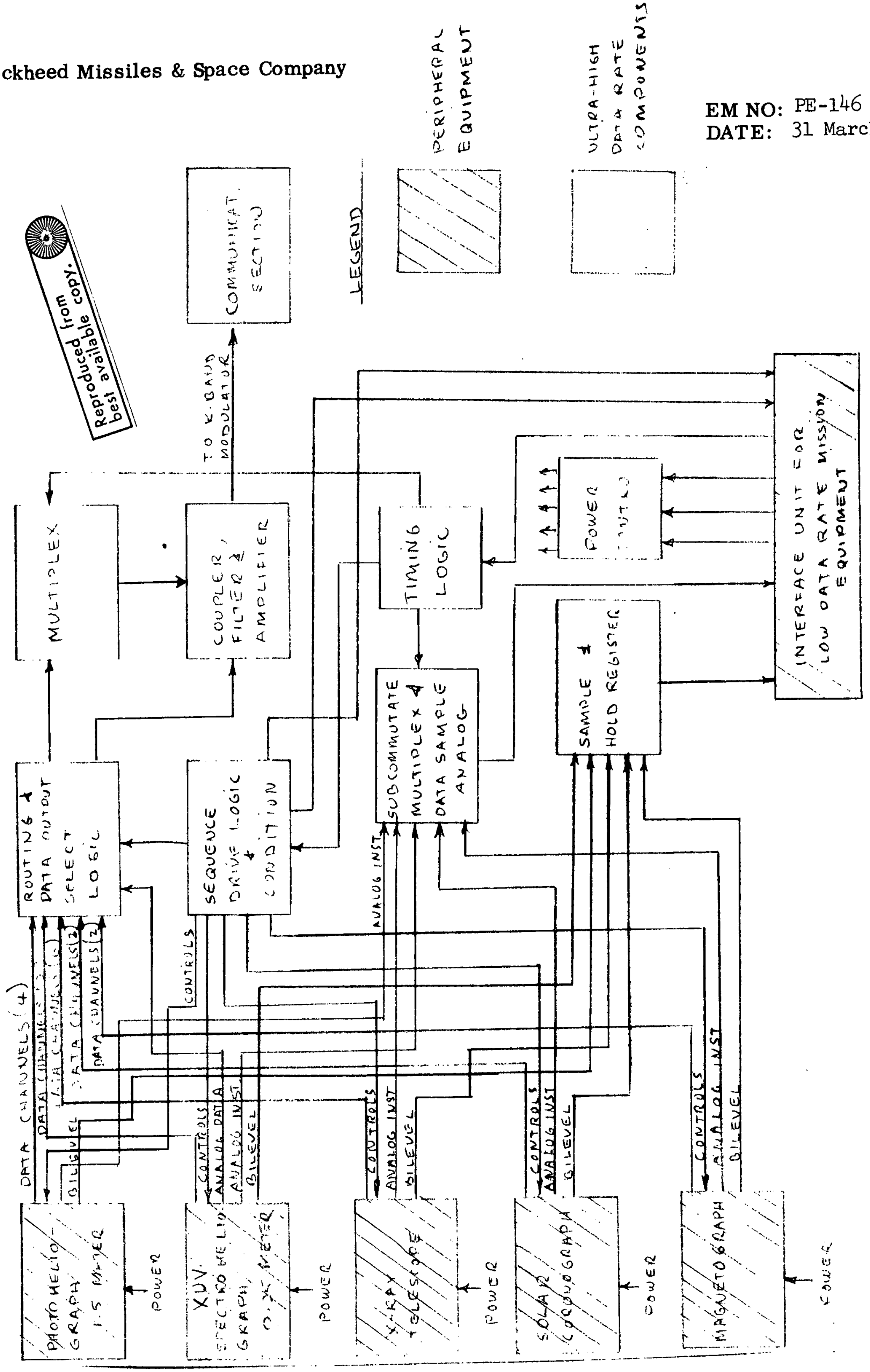

Fig.20 Standard LAOS CDPI Ultra-High Data Rate Interface Unit Block Diagram

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Analog Subcommutation Sample & Multiplex	140 gates and two amplifiers
Bilevel Sample & Hold Register	2 16 bit registers + 100 gates
Sequence & Drive Logic	8 bit register 128 "and" gates
Power Control	5 gates
Timer	Frequency divided assume 16 bit register and 5 gates
Routing & Data	17 Impatt diode amplifiers, 17 gates
Output Select Logic	1 5 bit register
Multiplexer	24 Impatt diode gates, striplines. 2 12 bit shift registers
Coupler, Filter & Amplifier	Stripline coupler & filter - TDA - Gain < 10 db, bandwidth 100 MHz
Logic	TTL MSI
Packaging	38 MSI circuits/board + mother board. Separate packaging required for the routing, multiplex/and coupling components.

Fig.21 Ultra-High Data Rate Interface Unit Components

	Recurring Cost	Non-Recurring
Router	$ 15,000	$ 5,000
Multiplexer	30,000	12,000
Coupler	8,000	3,000
Estimated additional costs	$ 53,000	$ 20,000

Thus, the total costs are as follows:

Recurring Cost	Non-Recurring
$ 4,000	$ 9,400
20,000	53,000
$ 24,000	$ 62,400

Low Data Rate Interface Unit - I

A block diagram of an Interface Unit for interconnecting low data rate mission equipment with other spacecraft and CDPI components is shown in Fig. 22. This unit corresponds to the medium to high data rate unit in the standard EOS CDPI. It differs from the EOS counterpart in the following ways:

- Much lower data rates handled by the unit.
- Interchangeable mission equipment is employed, where up to 7 experimental packages may be accommodated on a specific mission.
- Increased logic and other circuitry.

Interchangeability of mission equipment presents a problem since the interface logic must be modified in accordance with the changes that are made. Thus each general mission would necessitate a somewhat different interface configuration.

In order to accommodate the configuration changes, two approaches may be used: (1) provide a minimal housing and insert only those interface components needed for the particular flight, (2) supply an interface unit containing all the components needed for all missions. Selection switches would then be included to bipass those components not used during the flight regime. The first approach leads to the most compact design; however, the design itself is more difficult to prepare. The second approach is more flexible and simpler than the first; however, the design is wasteful and, more important, more susceptible to error in equipment utilization. Selection of the best approach would require detailed designs and subsequent tradeoff investigations of the designs. This work is outside the scope of the present study. To meet present purposes the utilization of a minimal housing with plug-in of the appropriate components for the specific missions will be assumed.

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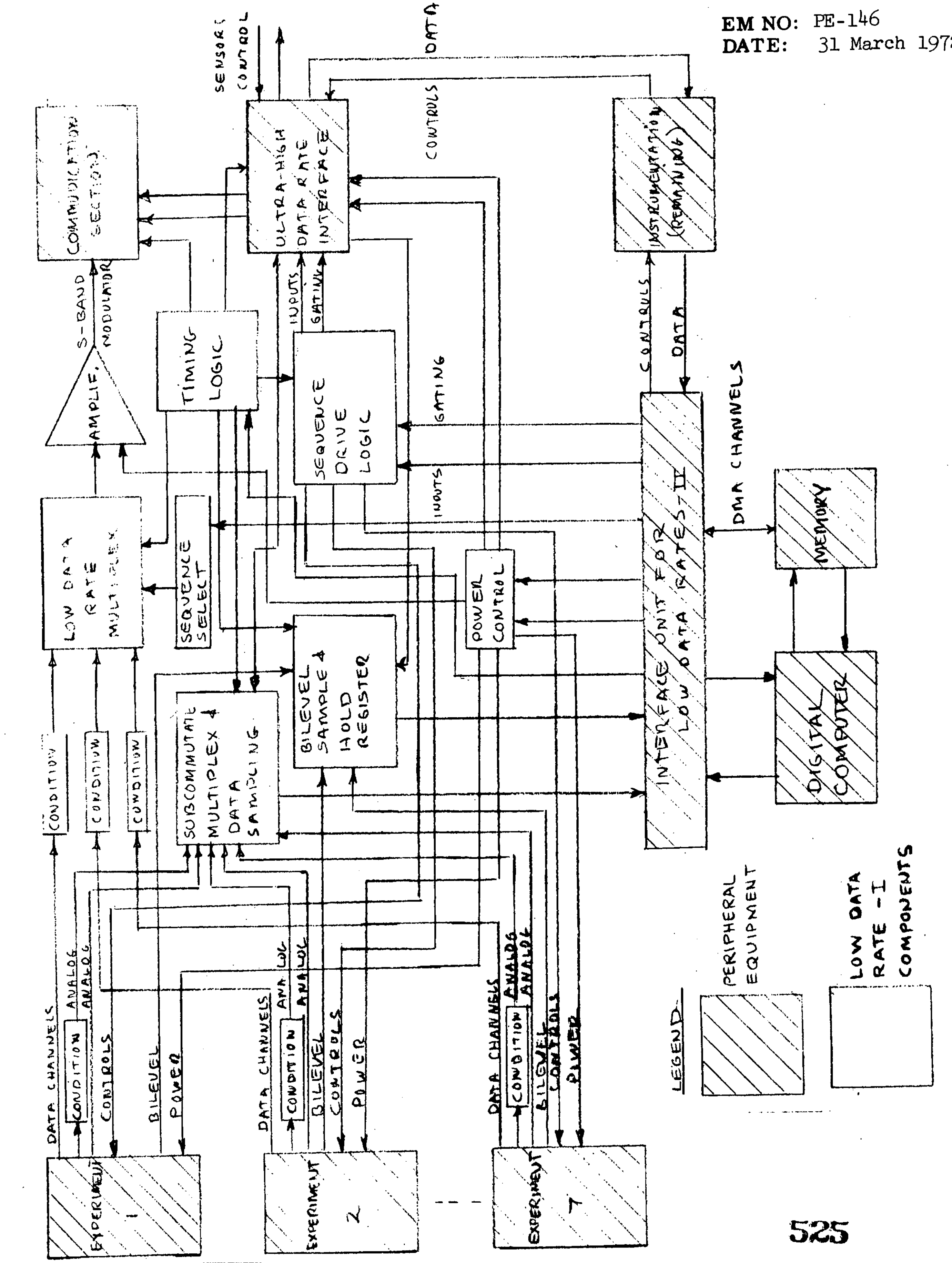

Fig. 22 Standard LAOS CDPI Low Data Rate Interface Unit - I Block Diagram

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C.10

The components needed for this low data rate unit are listed in Fig. 23 . Assuming the same type of electronics and packaging as the Ultra-High Data Rate Unit, the following estimates are obtained:

Logic Boards	Ckts/Board	Total Wt.	Total Power	Total Size
7 per mission	38 MST + Amplifier	9 lbs	17W	0.4 ft^3

The cost for the unit would be approximately as follows:

Item	Base Cost	Final Cost
Art work Fabrication	$ 1,300/board	$ 27,300
Components	250	5,250
Wiring & Packaging	6,000	6,000
Total Estimated Cost		$ 38,550

Low Data Rate Interface Unit - II

A block diagram of the second low data rate Interface Unit is shown in Fig. 24 . The unit is essentially the same as that used in the standard EOS, the primary difference being elimination of direct mission equipment interfacing. Thus the unit is divided into two parts - S&C interfacing functions and computer interfacing functions. The main analog multiplexer, A/D converter, and timing reference are also included in the unit.

The design approach employed in interfacing the digital computer is through use of common data channels for all devices attached to the channel lines. These channels are used with direct memory accessing (DMA). An alternate mode is obtained through input and output registers which operate via direct computer control.

The method of mechanization of the input/output processer is the same as that used in Reference 2. Basically, it consists of an address correlation scheme, where each device on the data channel has a unique identifying code. All information in the channel includes the assigned codes. Before the information is accepted as "true", the codes are checked and compared with prestored information. These precautions are needed to protect computer memory contents by preventing entry of data to the wrong locations.

The primary components of the second low data unit are given in Fig. 25. Compared with the Reference 2 designs, the elimination of direct mission equipment interfacing is more than compensated by the increased S&C functions. In addition, a more stable frequency reference is required - 1 part in 10^9. These changes result in an increase in the number of logic boards from 5 to 7 and incorporation of an oven for the crystal reference. These changes result in the following estimates:

Low Data Rate Multiplexer	Serial bit stream. Maximum input per channel 90 Kb/sec. Maximum output 110 Kb/sec. Order of sampling sequence under control of sequence select logic registers 7 solid state conditioning amplifiers 42 gating and steering circuits. 2 18 bit registers - 1 solid state power amplifier
Analog Subcommutator - Sample & Multiplex	130 gates and 1 amplifier
Bilevel Sample & Hold Register	115 gates, 1 12 bit register
Sequence Drive Logic	6 bit register 96 "and" gates
Power Control	18 gates
Timer	Frequency divider. Assume 12 bit register and 10 gates
Logic	TTL
Packaging	38 MSI circuits/board + amplifier + mother board

Fig. 23 Low Data Rate Interface Unit - I Components

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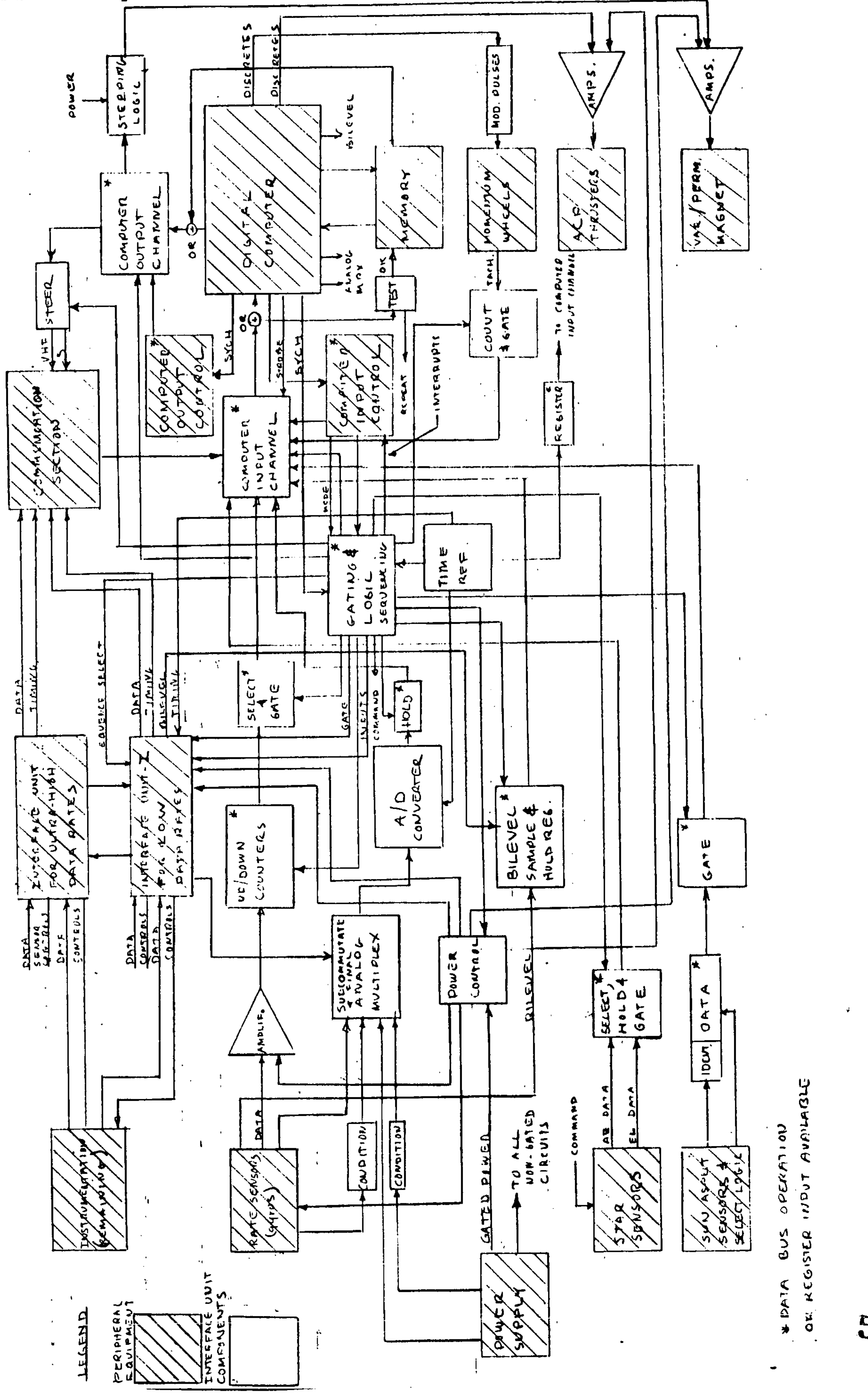

Fig. 24 Standard LAOS CDPI Low Data Rate Interface Unit - II Block Diagram

Analog Sampler + Multiplexer Subcommutator Section	110 gates + 1 amplifier
Final Analog Multiplexer	8 gates + 1 amplifier
Up/Down counters (3) (assume < 10Kbps at sample rate of 50 samples/sec)	3 10 bit registers + logic
Bilevel sample + hold registers	2 16 bit registers + 32 gates
Power Control	15 gates
Timer Xtal Reference 1 part in 10^9 accuracy	Crystal oscillator + oven. Assume 16 bit register and 20 gates
Sequence Driver logic	6 bit register 38 "and" gates
Buffer registers for data channels - 16 assumed	16 16 bit registers 128 "and" gates for divide and logic control
Buffer amplifiers	6 solid state
Logic	TTL

Fig. 25 Component for Low Data Rate Interface Unit II

Logic Boards	Ckts/Board	Total Wt.	Total Power	Total Size
7	38 MSI + A/D Convert	8.4 lbs	14 W	0.75 ft^3

In addition, there is one crystal assembly -

Size	24 in^3 (internal with logic board case)
Wt.	0.3 lb
Power	2 W

The cost for the unit would be approximately

Item	Base Cost	Final Cost
Artwork Fabrication	$ 1,500/board	$ 10,500
Components (TTL)	350	2,450
Wiring & Packaging	5,000	5,000
	Logic	$ 17,950
	Timing Reference	2,000
	Total Estimated Cost	$ 19,950

Data Processing Section

The computer requirements identified on pages 58-60 may be readily met by a number of computers undergoing development at the present time. These include the GE GEMIC, CDC 469, Honeywell 301 and the Autonetics D200 series. All the computers contain MOS LSI circuitry, incorporate plated wire memory and are amenable to input/output customization and memory expansion. They all contain sufficient computational power and speed to meet requirements with adequate safety margins. In order to provide a representative example, highlights of one of the D200 series is presented here. The information is an extrapolation from data contained in Reference 7.

Central Processor	29 MOS devices 14 types plus additional type for discrete outputs
Type	Binary, parallel, general purpose
Repertoire	68 instructions + 17 extensions
Word Length	24 bits - 3 formats
Registers	Accumulator + register(s) (3 index registers used in D200)

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Interrupts	number not specified - 7 extended operations
Arithmetic (assume same timing as double precision in 16 bit version)	3.75 microseconds add; 15 microseconds multiply fractional, fixed point, 2' complement
Memory	Plated wire NDRO
Memory Capacity	2K to 16K within computer package and up to 64K with an external package
Weight	30 lbs
Size	0.7 ft^3
Power (including power-supply and an external package)	75W

This computer is larger and uses more power than the CDC 469; however, it has a much more extensive instruction repertoire, including a greater number of double precision instructions. This added capability would be advantageous, considering the increased computational requirements of the standard LAOS.

Cost data may be extrapolated from that provided in Reference 3. These estimates are based upon using high reliability screening and other procedures to attain the reliability goals of the standard LAOS. The estimated costs are given in Fig. 26.

Conclusions and Recommendations

The rationale and representative design for a standard LAOS CDPI has been presented. The primary point of departure of this spacecraft CDPI from the first point design, the standard EOS spacecraft, is the need to accommodate multiple changes in mission equipment. This has a profound effect on the interface, data processing and communication requirements. The most important changes include the following:

- Redesign of the Interface Units. In the EOS CDPI the various units were configured on the basis of data rate. In the LAOS CDPI the configuration is dependent on both data rate and service objective, i.e., for mission equipment or spacecraft functions.
- More extensive and elaborate software and data processing. Memory capacity increase is required and word length increase desired.
- Communication Section requirements are potentially increased to the point where technology advances may be required; however, by preventing simultaneity of mission equipment operation at ultra high data rates, the Communication Section of the EOS will serve.

Item	Quantity	Overall Cost
16K 24 bit memory CPU	2	$ 374,000
Programmer's Console (Ground & Shuttle)	2	19,600
Memory Loader	2	23,400
		$ 417,000
Basic Software Costs		848,000
Validation Expense (40% basic cost)		329,200
	Overall Total	$ 1,594,200

Non-Recurring Costs		Recurring Costs	
Console	$ 19,600	Computer	$ 187,000
Loader	23,400		
Software	1,177,200	60% Software	$ 706,320
	$1,220,200		$ 893,320

Fig. 26 CDPI Data Processor Hardware and Software Estimated Costs

If the EOS and LAOS CDPI interface circuitry is examined there is a high degree of commonality in the various units. This reinforces the possibility of developing standardized CDPI designs. It is recommended that an effort be undertaken to develop such designs.

References

1. Earth Observatory Satellite (EOS) Definition Phase Report (Preliminary); NASA-GSFC, dtd August 1971

2. Standard Earth Observatory Satellite CDPI Subsystem - EM PE-103; M. Loeb, D. F. Wald; LMSC; dtd 30 November 1971

3. Standard U.S. Domestic Communication Satellite - Communications, Data Processing and Instrumentation Subsystem - EM PE-123; M. Loeb, LMSC; dtd 24 Dec 1971

4. Reference Earth Orbital Research and Applications Investigations (Preliminary) Vol. 2 - Astronomy; NASA, dtd 15 Jan 1971

5. HEAO Phase C/D Statement of Work for Missions A&B - Appendix C (Experiments Data); NASA/MSFC, undated

6. Scientific Satellites, W. R. Corliss, NASA SP-133, dtd 1967

7. "Autonetics Advanced D200 Family of MOS/LSO General-Purpose Computers" - Brochure - Pub. No. P71-828/401; Autonetics"; dtd 9/71

3.3 Electrical Power Subsystem (EPS)

Introduction and Requirements

The Electrical Power subsystem provides electrical power to energy consuming subsystem and experiment equipment; and includes the electrical harness which distributes power and signals throughout the vehicle.

The maximum average power required from the EPS for each of three of the major astronomical observatories, HEAO, LST, or LSO is 1500 watts; that required for the fourth observatory, LRO is 2000 watts.

Description of EPS

The EPS of the standard LAOS has been sized to provide a minimum of 1500 watts average power for an orbital life of two years. It consists of a single-axis-tracking, flexible solar array, 10 nickel-cadmium batteries which are charge-controlled by array switching, and a DC-DC regulator. The system provides a nominal 28 volt bus which varies from 25.0 to 28.0 volts. The regulator will provide 28.0 VDC ± 2% for the equipment needing close regulation. A functional block diagram is shown in Fig. 27.

A single-axis-tracking solar array is appropriate for the standard LAOS. When the vehicle is not in the Earth's shadow, such an array can be maintained normal to the solar radiation at all times while the experiment package mounted on the spacecraft is pointed to any target in the celestial sphere; provided that the roll attitude of the vehicle may be adjusted for every change in pointing direction.

Large battery capacity (more weight) with its attendant low depth-of-discharge, makes elaborate charge-rate controllers unnecessary (at a given temperature, battery life is inversely proportional to depth-of-discharge). Array open-circuit switching conveniently dissipates excess energy as heat out on the large solar array (by not converting solar energy to electrical energy). This eliminates the large bank of heat dissipating resistors associated with shunt regulators. These resistors are traditionally difficult to locate because of the large quantity of heat to be dissipated. The batteries are on the bus full time, establish the array operating point, and serve the same function as a shunt regulator, thus making the system unregulated bus a direct energy transfer system. They also completely absorb the array switching transients and absorb the array cold-to-hot high voltage transients that are characteristic of array-on-bus and battery switching systems.

The major components of the EPS are listed in Fig. 28.

Solar Array Description

The solar array is of the flexible substrate type. Extensive development work has been done at LMSC on this type of array, both for the NASA-funded Large Space Station program and on company-sponsored programs. A similar type of array, developed by Highes, has been flight-tested on an Agena spacecraft by the Air Force.

The array consists of two wings, one on each of the opposite sides of the spacecraft as shown in Fig. 1. Each wing is divided into sections. The power from each section

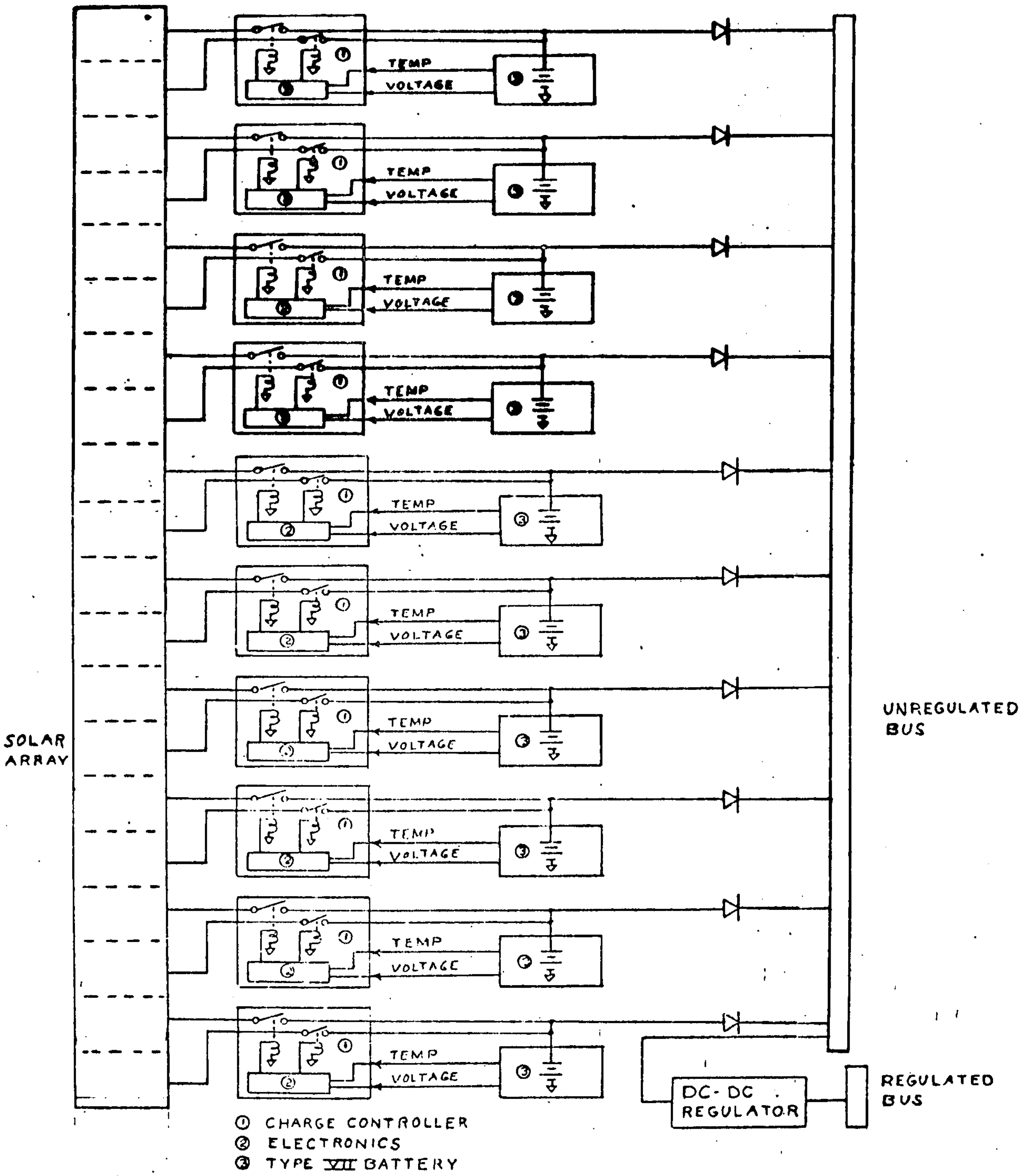

Fig. 27 Standard LAOS Electrical Power Subsystem

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Component	Size	Wt. (ea. lb)	Per Vehicle Qty.	Per Vehicle Wt (lbs)
Solar Array Module One module consists of: 1 containment box 192 sq ft .001 Kapton substrate and printed circuit 2 FCC harnesses 19,200 solar cells (2 x 4 CM) 19,200 covers, fused silica Extendable Boom Assembly	96 x 36 x 6"	220	2	440
Solar Array Drive Assembly (drive motors, electronics, slip-rings)	5" dia x 10" length	32	2	64
NiCd Battery (Type VII)	7 x 7 x 21"	70	10	700
Charge Controller	9 x 10 x 10"	6	10	60
Power Distribution Unit	20 x 16 x 5"	32	1	32
DC to DC Regulator	11 x 20 x 8"	37	1	37
Harness Assy	-	340	1	340

Weights do not include contingency
Module Bases, Covers, Cables and Connectors not included

Fig. 28 Major Components Electrical Power Subsystem - LAOS is carried to the base of the array via flat conductor cable (FCC) feeder harnesses. The substrate assembly is the Lockheed-developed Kapton/FEP/copper-interconnect design. Two parallel contacts on the back of each cell provide both mechanical fastening and positive power connections; raised series tabs provide negative power connections. Individual sections are joined together mechanically after manufacture in order to form a wing of the desired length.

Each solar array wing is stowed in a flat pack aluminum box before deployment. This containment forms the structural protection from vibration and acceleration loads and provides support for the solar array wing ends during on-orbit operation. When the wing is deployed in orbit, the containment box is positioned 24" from the side of the spacecraft to avoid shadowing of and possible thruster deposits on the wing surface. The wing is then deployed from the box. Both deployments are accomplished by a BI-STEM* extendable boom.

The extendable portion of the boom is in two halves. Each half is a flat tape when wound on its storage spool. When it is extended, it assumes a pre-formed curve shape and the two halves combine with an overlap to form a column. Two reversible DC motors drive the storage spool through a differential, so that either motor is redundant to the other. The maximum extension of the boom is 35' to accommodate additional solar array wing length, raising the EPS minimum average power to 2000 watts.

In operation, spring-operated releases free the containment box from the spacecraft and open the box lid. The boom starts to deploy, pushing the box out. Spring-loaded extension arms lock into place when the box has reached its proper position. As the boom continues to extend, the folded wing is pulled out of the containment box until it is fully extended into a flat sheet. A constant tensioning device is incorporated into the system which assures a uniform preload on the flexible substrate. This preload determines the flatness and natural frequency of the sheet. After deployment, the boom is locked in place and any dimensional variations in the sheet length due to creep or temperature changes are absorbed in the tensioning device.

The solar array may be retracted and stowed to facilitate on-orbit servicing of the spacecraft or recovery of the spacecraft by the Shuttle.

The solar cells are 2 by 4 CM, N on P, .012", 2 ohm-CM base resistivity, with wrap-around contacts. 97.5% of the production cells (after mechanical screening) will be used instead of the usual 66.7% yield. This reduces cell costs by 30% with a 2% increase in the number of cells needed (2% weight increase). The coverglass is .020" fused silica for maximum radiation protection (less cells) and less handling breakage (higher yield). Solar arrays for conventionally boosted spacecraft systems typically use .012-.014" thick coverglass for the weight advantage they afford.

<u>2 x 4 CM Solar Cell Characteristics</u>

Cell Output, Normal to Sun, Beginning of Life
Avg. cell power, 28°C, AMO, 66.7 percent yield = 117.5 mW
Avg. cell power. 97.5 percent yield = 117.5 x 0.98 = 115.1 mW
Avg. cell power with 20 mil fused silica coverglass = 115.1 mW x 0.97 = 111.6 mW

Solar Array Sizing

$$\text{Number of cells} = \frac{\text{Power Needed from Array at BOL}}{\text{Cell Output x Temp Factor x Effectivity}}$$

Cell Output = 0.1116 Watts/cell at $28^{\circ}C$

Temperature Degradation
Factor for Operation at $52^{\circ}C$ = 0.87

Effectivity in
Nominal Orbit = 0.6

Power Needed from Array at Beginning of Life

Average Power	1500 watts
Charging Loss	150
Radiation Degradation (7 years at 2.5%)	250
	1900 watts

$$\text{Number of Cells} = \frac{1900}{0.1116 \times 0.87 \times 0.6}$$

Number of Cells = 31,400 2 cm x 4 cm cells

Use standard Solar Power Module EPS-2-4 which has 19,200 cells. Two EPS-2-4 Modules have 38,400 cells and provide 22% power contingency for growth of requirements.

The solar array is sized to provide for up to seven years of orbital life, since in-orbit replacement of solar power modules will be less frequent than for many other spacecraft subsystem modules.

Batteries

Nickel-cadmium batteries were chosen because of the two-year on-orbit life requirements. The 40 ampere-hour capacity is necessary to keep a low depth-of-discharge (DOD) to maximize the battery life. An average 10% DOD has been calculated for ten 40 amp-hr batteries and the 1500 watt load. A NASA EOS report states 25% DOD to be a maximum, therefore 10% represents a good safety margin. Ten batteries in five Battery Power modules will also be adequate for the 2000 watt load of the LRO since the depth of discharge will be less than 15%.

Charge Controllers

Each battery has a charge controller which connects or disconnects a section of the array to the battery. The voltage "tail-up" as the battery nears full charge is used as the signal to cut off charge. This voltage level is modified by the battery temperature which is sensed by transducers in the battery and fed into the charge controller. The control is done in two stages. As the battery proceeds towards full charge, first one-half of the array section is turned off, and then if the battery voltage continues to rise another one-half volt, the other half of the array section is disconnected. In this way the array output can often balance the load with little or no battery cycling, except for the normal night-time cycling. Conversely as the battery voltage decreases, the array section is connected in two stages.

The control levels will be approximately as shown in Fig. 29, with K1 and K2 depicting the two levels of control. The approximately 3 volt dead band between the connect (charge-on) and disconnect (charge-off) levels reflects the three volt difference between the battery charge voltage and its discharge voltage.

A high temperature charge cut-off (both sections) would probably be incorporated into the charge controller, to cut off the charge at about 90°F, regardless of the state of charge. This would interrupt the possible "thermal run-away" mechanism of the battery in some abnormal situation. Additionally, a battery off-line ground command could be incorporated, which would cut a defective battery out of the circuit and connect the associated section of the array directly to the unregulated bus. This one section of the array would "load regulate" itself if its capacity did not exceed the load requirements.

Power Distribution Unit

This unit distributes the power to the various using equipments and also contains fuses, current sensors, and power system telemetry conditioning networks as required.

DC-DC Regulator

A conventional DC-DC regulator will be selected when the regulated power needs are finalized. It will be internally redundant to satisfy design life requirements.

Solar Array Drive Module

One Solar Array Drive is required for each Solar Power Module. This unit performs the sun tracking and power transfer functions. The drive assembly rotates the extension boom and containment box to provide sun tracking about a single north-south axis. Motive power is from a DC energized stepper drive providing 0.1° incremental steps of rotation. Logic input for the motor is a clock signal which provides orbital rate pulses during both sunlit and eclipse times. Command override and clock calibration functions are included within the logic.

Power transfer from the rotating Solar Power Modules to the spacecraft loads is accomplished by multiple slip ring circuits with redundant brushes for each ring.

EPS Equipment Modules

The EPS Battery Power Module (5 required) and Power Distribution Module (1 required) are shown in Figs. 30 and 31. They are typical standard subsystem modules designed for in-orbit removal and replacement by a Shuttle crewman or a teleoperator.

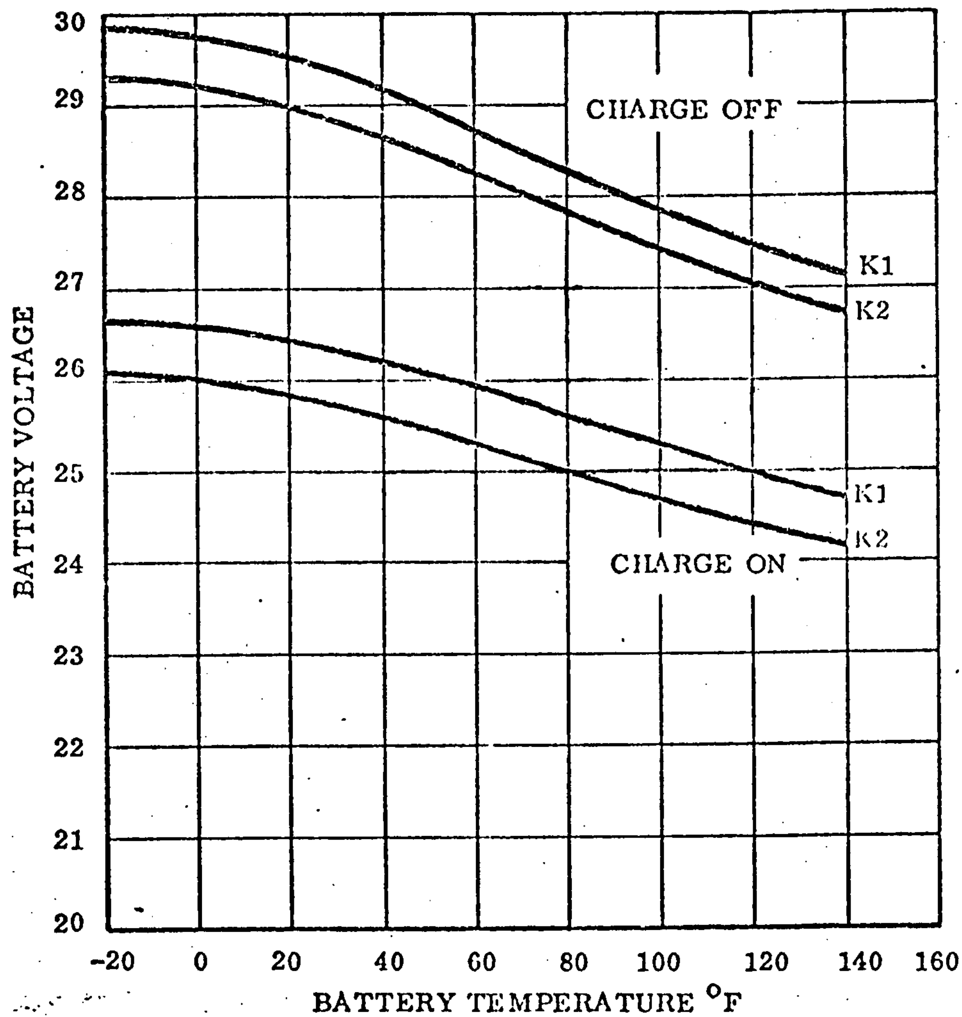
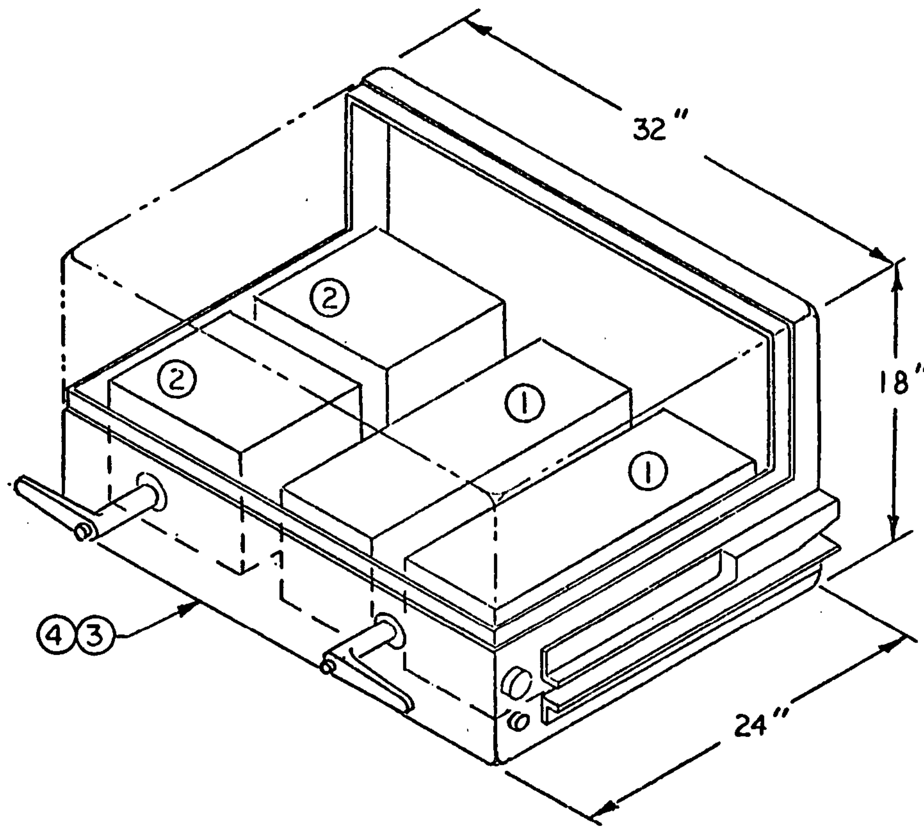


Fig. 29 Charge Controller Operating Characteristics



Equipment

(1)	Type 7 Battery - 2 req'd 40 AH NiCD	140 lbs
(2)	Charge Controller - 2 req'd	12
(3)	Base & Cover	32
(4)	Cables & Connectors	3
	Subtotal	187
	10% contingency	19
		206 lbs

Fig. 30 Battery Power Module (EPS-6)
Electrical Power Subsystem

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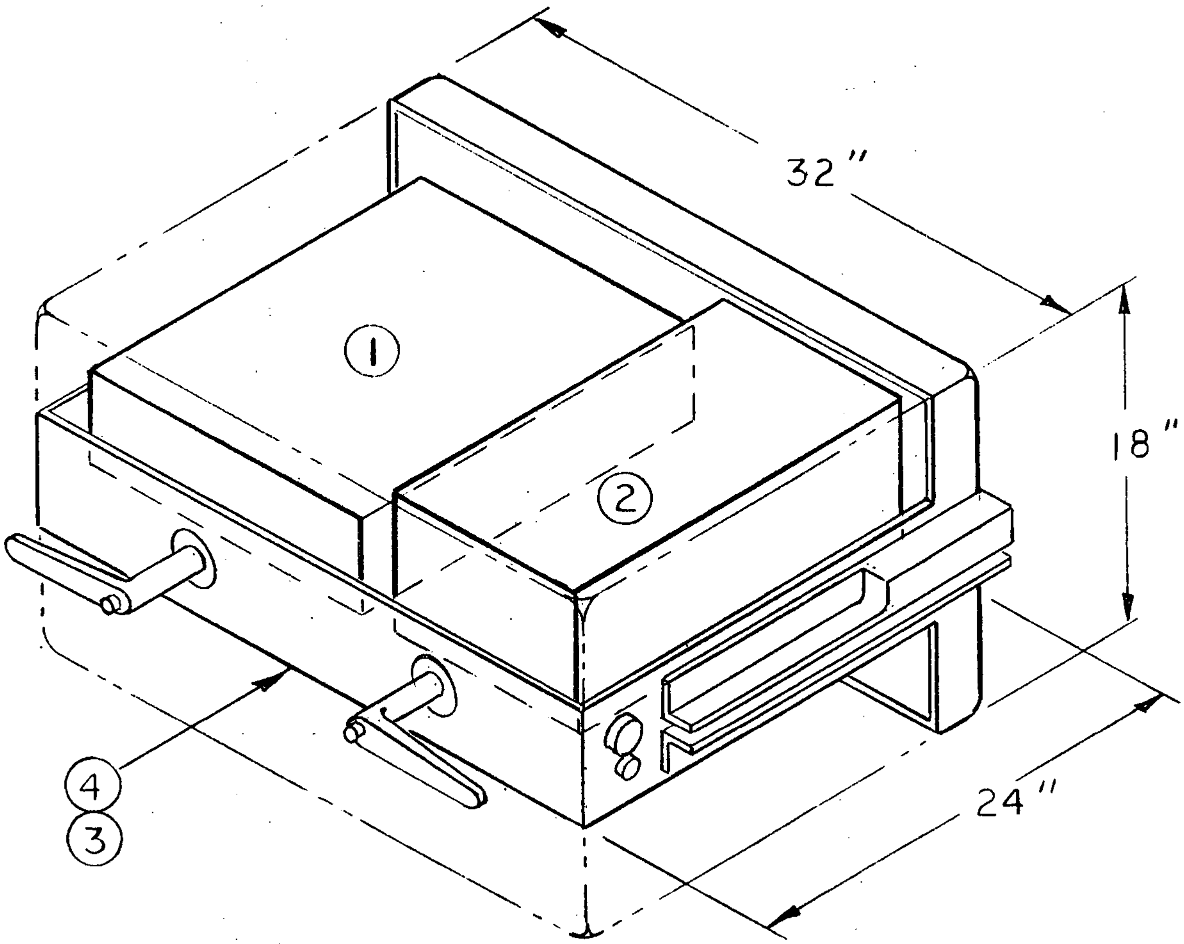

Equipment	Qty.	Weight (lb)
(1) Power Distribution Unit	1	32
(2) DC to DC Regulator	1	37
(3) Internal Wire Harness	-	4
(4) Base & covers	-	32
	Subtotal	105
	15% Contingency	15
	Total	120

Fig. 31 Power Distribution Module (EPS-7) - Electrical Power System

3.4 Attitude Control Subsystem (ACS)

General

The Attitude Control Subsystem (ACS) provides thrust for drag makeup, reaction wheel unloading and backup, and emergency attitude hold for at least one month. The ACS consists of four identical modules installed on the outboard edges of the vehicle as depicted in Fig. 32. Each module contains four 1.75 lb rated thrusters, oriented such that any three modules could provide 3 axis vehicle control. All four modules combined provide a total impulse of 16600 lb-sec thereby providing for at least 2 yrs. of orbital life. The breakdown of ACS impulse requirements is given in Section 3.1. Loss of one module early in life has no appreciable effect, if any, on the 2-year orbit life.

Description

Each ACS module consists of four major components. A fill valve is used to load Freon gas into a high pressure storage tank. The storage tank delivers Freon gas to the inlet of a pressure regulator over a decreasing pressure range from 1750 to 180 psia as the propellant is consumed by the thrusters. The regulator outlet provides 120 ± 10 psia gaseous Freon propellant to the common manifold of the 4 thrusters. As the vehicle stabilization & control system requires external forces, the individual thruster valves are commanded open and closed to allow the 120 ± 10 psia gas to flow into the thruster chambers and expand out the nozzle producing thrust. A flow schematic of the ACS is presented in Fig. 33.

Freon 14 (CF_4), a dry, non-toxic, nonflammable, odorless gas that can be handled like gaseous nitrogen was selected for the ACS propellant. Each tank is initially charged with 100 lbs of Freon at an initial pressure of 1750 psig, which when blowndown to 180 psia minimum regulator inlet pressure, delivers 96.5 lbs of propellant for a total impulse of 4150 lb-sec. A 1750 psig maximum working pressure is selected to take advantage of the low Freon compressibility factor (Z) at that pressure and utilize a low-cost stainless steel pressure vessel. Since Z factor is an inverse function of gas density, $Z = \frac{PV}{RT}$ a low Z factor allows efficient storage tank design when volume is not a primary constraint. Each tank is loaded through a fill valve located at the tank inlet/outlet fitting. This fill valve is opened or closed with a wrench and is capped after fill for redundant protection against leakage. The pressurized Freon gas is supplied to the thruster via a 3600 psig rated inlet Regulator Valve Assembly which contains an inlet filter rated at 40 micron absolute, a solenoid latching valve, a regulator, and a downstream pressure relief valve with thrust nullifier. The regulator is supplied with Freon over a 1750 psig to 180 psig pressure range while supplying regulated 120 ± 10 psia gas to the thruster cluster assembly manifold. A solenoid latching valve controls tank pressure to the regulator dome which in turn positions the regulator main flow poppet. This solenoid latching valve is moved to either open or closed by a 80 millisecond voltage pulse and have a position indicator which is monitored on TM. When the solenoid valve is open, gas pressure to the dome regulates outlet pressure to 120 ± 10 psia. When the solenoid valve is closed, the regulator main valve poppet closes and ceases to supply gas flow to the thrusters. Continued operation of the thruster after the regulator main poppet is closed, results in the downstream line pressure decreasing from 120 psia to 0 psia. The pressure relief valve is located downstream of the main flow poppet and vents should pressure exceed 160 psia due to a regulator malfunction or excessive main flow poppet leakage. This feature assures that the thrust valves will not be subjected to over-pressurization. The thrust valves are rated at

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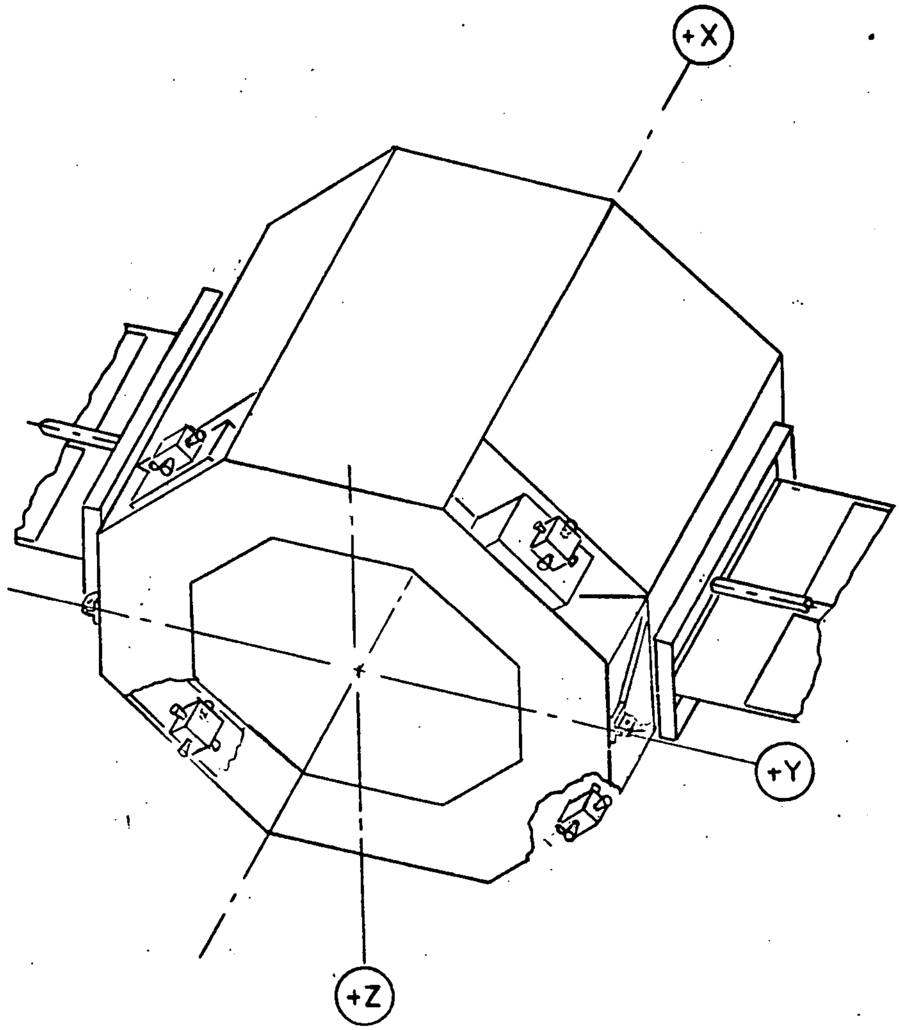

Fig. 32 Attitude Control Subsystem Module Locations - Standard LAOS

Lockheed Missiles & Space Company

EM NO: PE-146
DATE: 31 March 1972

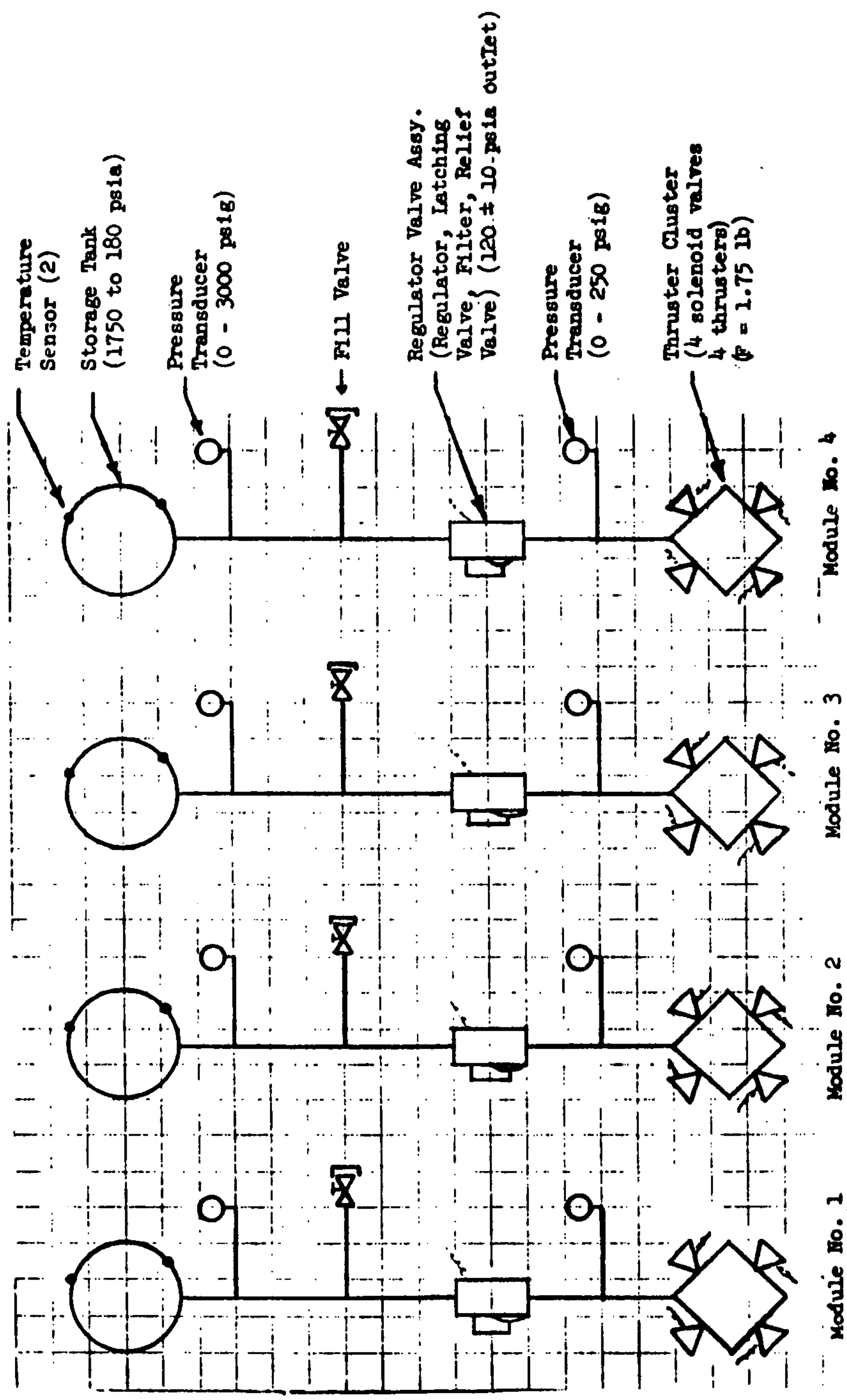

Fig. 33 Standard LAOS Attitude Control Subsystem

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1000 psig burst. A 40 micron absolute rated inlet filter in the Regulator Valve Assy. protects the regulator and thruster valve poppet seals against excessive particle contamination to yield acceptable leakage rates.

Each thruster assembly contains an electrical valve which is driven by the Attitude Control System Electronics. The thruster nozzle is sized to deliver 1.75 lb of thrust at a 120 psia inlet pressure. Application of a 20 millisecond electrical pulse to a pair of thrusters limits the ACS minimum impulse bit to 0.07 lb-sec (2 x 1.75 x .020). Hence, the ACS requirements of limiting maximum thrust to 4 lbf per pair (2 x 1.75 = 3.5), and providing a 0.085 lbf-sec minimum impulse bit are satisfied.

3. Sensing Elements

Each module contains two pressure transducers, two temperature sensors, and a regulator valve indicating switch. A zero to 3000 psig range pressure transducer at the storage tank inlet/outlet and two temperature sensors attached to the external tank skin are provided for gas loading, orbital propellant mass statusing, and leakage detection. The zero to 250 psig range pressure transducer between the Regulator Valve Assy and the thrusters provides a regulator outlet pressure health check and can be used to check thruster valve leakage when the regulator main flow poppet is closed and the thrusters are inactive. Additionally, the pressure safety of the system is verified prior to manned access.

The Regulator Valve Assy latching solenoid valve has a position indicating monitor. Position indicators are used to determine if the valve is responsive to signal input, during operation and checkout. A valve indicating switch can be added to the thrusters if it is desirable to record thruster activity during orbit life.

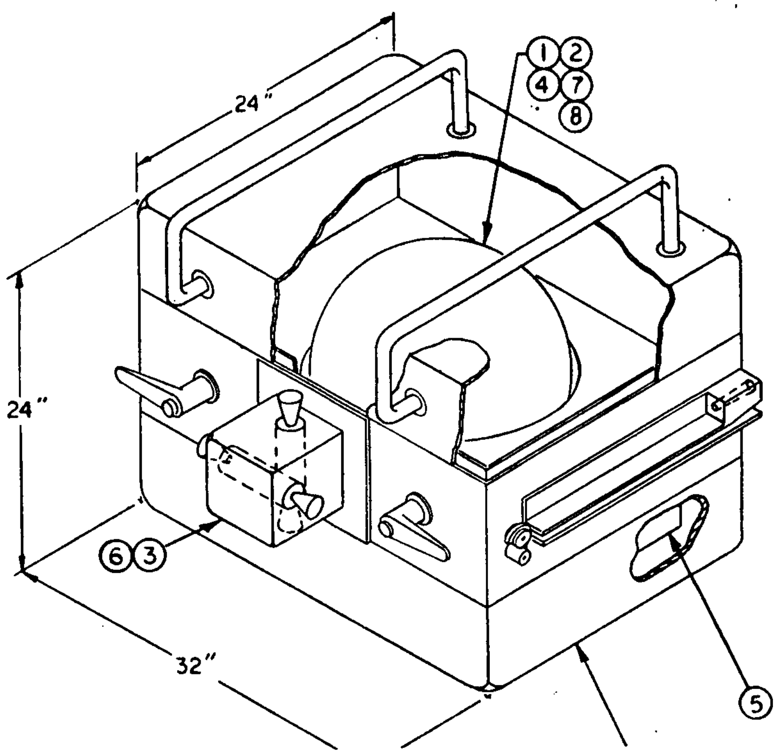
4. Module

The ACS module shown in Fig. 34 is assembled using the major equipment listed in Fig. 35 plus miscellaneous tubing, bracketry, and electrical harnesses. Tubing is hard line stainless steel, brazed to component fittings, and designed to minimize potential leakage at joints. Each module assembly is locked in place in the vehicle by a mechanism designed to permit replacement of modules in orbit. The installation arrangement of the four modules as shown in Fig. 32 allows identical configuring of each module and hence a minimum-stock inventory for module interchangeability.

5. Assembly and Test

Due to the relatively small size, low weight, and simplicity of the module it can readily be assembled and bench tested. All the previously acceptance tested major components are mounted, connected with the appropriately designed tubing, and brazed. The instrumentation is connected to a test panel, the storage tank pressurized, the system leak checked, and the system functionally checked out by providing command signals, After completion, the system is de-pressurized and stored prior to usage. Prior to vehicle installation, the storage tank is loaded with the appropriate quantity Freon 14 gas propellant.

EM NO: PE-146
DATE: 31 March 1972



	Equipment	Quantity	Weight
(1)	Tank, 22" O.D. Cres	1	57.5
(2)	Plumbing, Cves 3/8d x 0.28w	As req'd	.5
(3)	Valve cluster, 4 nozzles	1	5.0
(4)	Fill Valve	1	.3
(5)	Electronics unit	1	7.0
(6)	Regulator valve	1	3.9
(7)	Pressure transducer	2	.7
(8)	Temperature sensor	2	.1
(9)	Internal wire harness	As req'd	3.0
(10)	Base and covers	1	40.0
		Subtotal	118.0
		15% contingency	18.0
		Total (Dry)	136.0
		Freon 14 Propellant	100.0
		Total (Wet)	236.0

Fig. 34 Attitude Control Subsystem Module

Equipment	Description	Weight	Cost ROM	Electrical Input	Electrical Output	Remarks
Storage Tank	ARDE, Inc. 22" O.D. 301 Stainless 5220 in^3 7150 psig design burst	57.5 lbs	$ 4000	-	-	
Regulator Valve • Regulates • Latching solenoid • Filters • Relief & nullifier	LMSC 8100157 Rated at 3600 psig inlet 120 ± 10 psig 80 ms pulse command 40 micron absolute 175 psig crack	3.9 lbs	6000	22-29v	0-5v	Qualified
Fill Valve	LMSC 8106086-1 Valve shutoff plus cap 3000 psig rated	.25 lbs	400	-	-	Qualified
Thruster Cluster Ass'y.	Sterer 24050 1000 psig valve-burst .070 lb-sec MIB at 20 ms PW 4 thrusters per cluster 1.75 lb per thruster	3 lbs	5000	22-29v draws 1 amp at 28v	-	Qualified for 810K cycles vs 90K cycles req.
Pressure Transducers	LMSC 8100496 0-3000 psig - 13 0-250 psig - 11	.35 .35	1600 1600	22-29v 22-29v	0-5v 0-5v	Qualified Qualified
Temperature Sensor	LMSC 1618702-5 stick-on (2 req'd).	.07 lbs	100	22-29v	0-5v	Qualified

Fig. 35 ACS Module Equipment List

PE-156

GENERAL DESCRIPTION

STANDARD EARTH OBSERVATORY

SPACECRAFT

PE-156

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: STANDARD EARTH OBSERVATORY SPACECRAFT	EM NO: PE-156 REF: DATE: 31 March 1972
AUTHORS: F. C. Bolton — Prepared under cognizance of: Advanced Payload Systems - 69-02 Space Systems Division	APPROVAL: ENGINEERING F C Bolton Wayne F. Miller

9 pages

PRELIMINARY

Standard Earth Observatory Spacecraft

1.0 General

Early in the Payload Effects Analysis Follow-On (PEFO) Study, a preliminary design of an Earth Observatory Satellite (EOS) was completed and reported in LMSC Engineering Memos PE-102 through PE-106. Subsequently, the design was reviewed and modified to incorporate concepts that had been developed later in the PEFO Study. This Engineering Memo describes the changes made in the EOS design to comply with the requirements of a Standard Spacecraft which is defined as follows:

> A Standard Spacecraft is one that incorporates standard subsystems and is capable of performing, one at a time, a significant number of the missions defined by the NASA Mission Model.

Only the changes to the initial EOS to convert it into a Standard Earth Observatory Spacecraft (SEOS) are described in this Engineering Memo.

2.0 Standard Earth Observatory Spacecraft (SEOS) Configuration

The general configuration of the SEOS is shown in Fig. 1 and the locations of the standard subsystem modules in Fig. 2. The locations of the Mission Equipment (sensors) are shown in Fig. 2 of PE-106. The subsystem modules are designed to be removed and replaced in orbit by a Space Shuttle crewman, a teleoperator, or manipulators. This makes possible the repair, if necessary, of the SEOS in orbit prior to separation from the Shuttle; and the recovery and repair of the SEOS in orbit if one or more standard modules should fail during the nominal orbital lifetime of the spacecraft. The mission equipment has not been packaged in similar replaceable modules during this limited design study. However, it would be logical to modularize the Mission Equipment to gain the significant cost advantages and operational flexibility that modularization affords.

The SEOS configuration provides spare volume for growth of either spacecraft subsystems or Mission Equipment.

The standard subsystem modules of the SEOS are listed and described in Fig. 3a through 3c.

A typical standard subsystem module is shown in Fig. 4. The module is guided into its location on the spacecraft by rails and aligned and supported by two inboard pins and two outboard cams that engage machined grooves in the rails. The cams also transmit force from the cam actuator handles on the outboard face of the module to accomplish the controlled engagement and disengagement of the bulkhead-type electrical connectors on the in-board face of the module. The two wrap-around handles are designed to facilitate the handling of the module in orbit by a Shuttle crewman.

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EM NO: PE-156
DATE: 31 March 1972

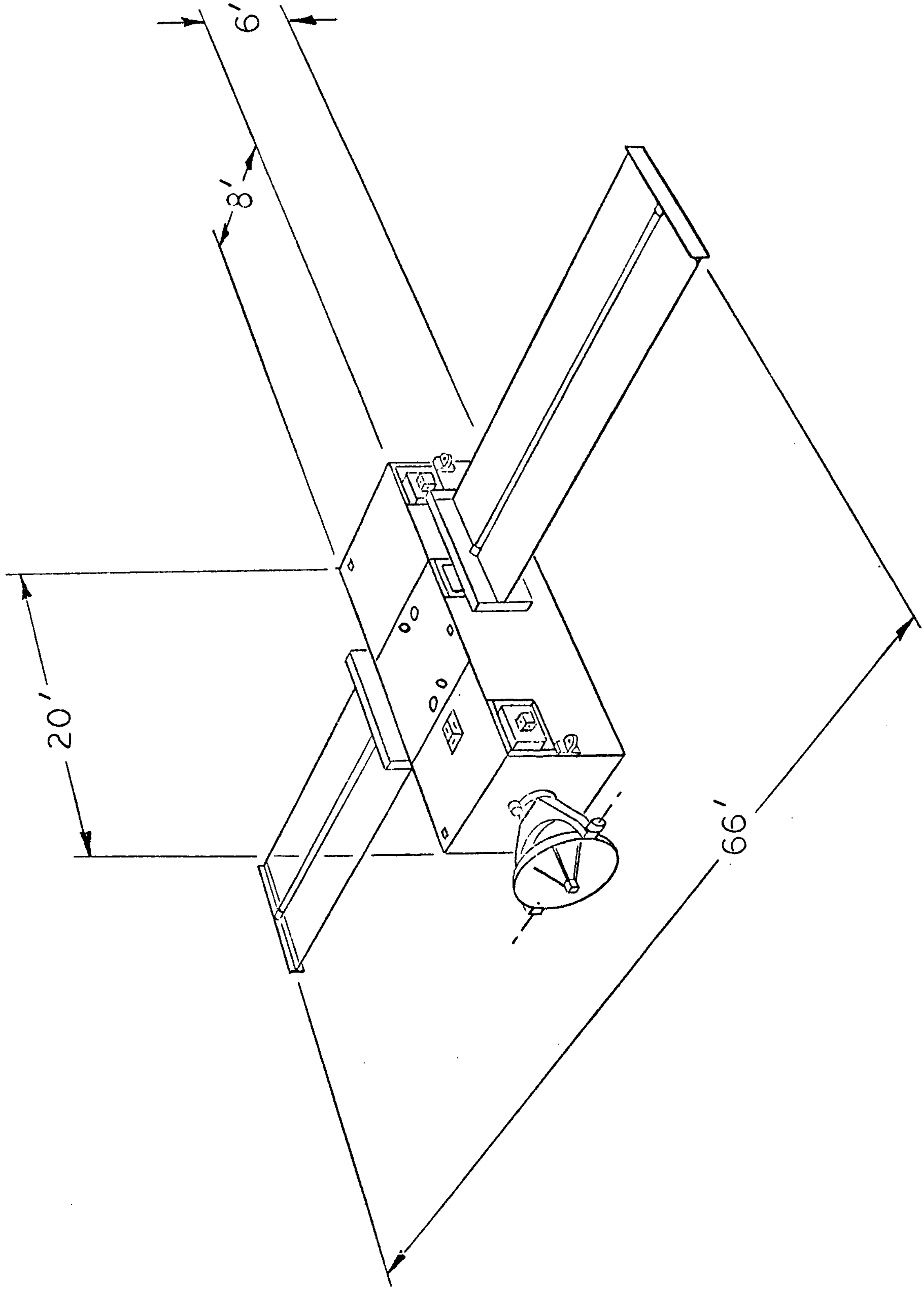
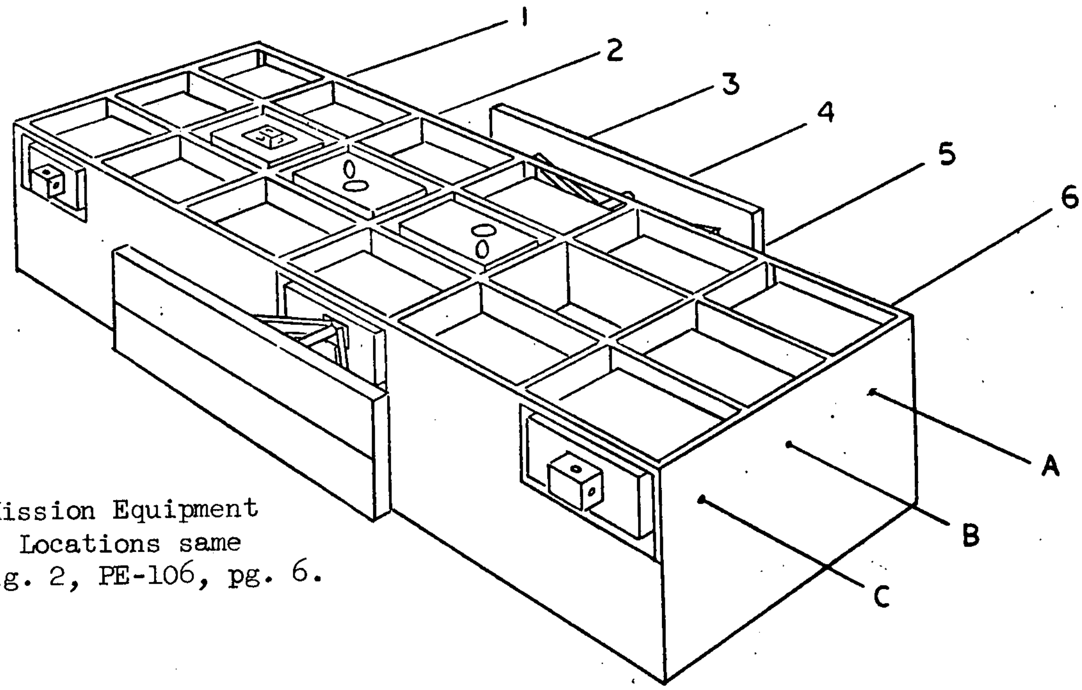


Fig. 1 General Configuration - Standard Earth Observatory Spacecraft



Note: Mission Equipment (Sensor) Locations same as in Fig. 2, PE-106, pg. 6.

Spacecraft Subsystem Modules

A-1	Attitude Control Module No. 1	B-4	S&C Precision Reference Module No. 2
A-2	S & VHF Band Communication Module	B-5	Empty
A-3	Battery Module No. 1	B-6	Reaction Torque Module
A-4	Solar Power Module No. 1	C-1	Attitude Control Module No. 3
A-5	Power Control Module	C-2	Data Processing Module
A-6	Attitude Control Module No. 2	C-3	Battery Module No. 2
B-1	K-Band Communication Module	C-4	Solar Power Module No. 2
B-2	S&C Secondary Reference Module	C-5	Battery Module No. 3
B-3	S&C Precision Reference Module	C-6	Attitude Control Module No. 4

Fig. 2 Subsystem Module Locations
Standard Earth Observatory Spacecraft

Subsystem	Module	Equipment in Module	Module Weight (lb)
Stabilization & Control	Precision Sensing Module (2 req'd) No. 1 No. 2	• Fixed Heat Star Trackers (2) • FHST Electronics (2) • Three-Axis Rate Sensor • Precision Equipment Mount • Module Base • Module Cover • Cables and Connectors	Basic 91 lb 15% contingency 14 Total 105 lb
Stabilization & Control	Secondary Sensing Module No. 1	• Sun Aspect Sensor (5) • Sun Aspect Sensor Electronics • Rate Gyro Package • Secondary Stabilization & Control Electronics • Module Base • Module Cover • Cables & Connectors	Basic 56 lbs 15% contingency 8 Total 64 lbs
Stabilization & Control	Reaction Torque Module No. 1	• Reaction Wheel(3)(10 ft-lb-sec) • Wheel Support and Safety Shield • Wheel Drive Electronics • Magnetic Torquer (3) • Mag. Torquer Electronics (3) • Module Base • Module Cover • Cables and Connectors	Basic 133 lbs 15% contingency 20 Total 153 lb
Communication Data Processing & Instrumentation	K_u-Band Communication Module	• K-Band TWTA (50 watts out)(2) • K-Band PLL Receiver • K-band QPSK Modulator/Driver • K-band Multicoupler • Interface Unit (High Rate) • Module Base • Module Cover • Waveguide, Cables, Connectors	Basicq 74 lbs 15% contingency 11 Total 85 lbs

Fig. 3a SEOS Subsystem Modules (1 of 3)

Subsystem	Module	Equipment in Module	Module Weight (lb)
Communication Data Processing & Instrumentation	S-Band/VHF Communication Module No. 1	S-Band Transmitter (10 watts out) S-Band Receiver S-Band QPSK Modulator/Driver S-Band Multicoupler Delay Lock Loop Correlator (2) Antenna Servo Electronics VHF Transmitter (5 watts out) VHF Receiver/Demodulator VHF Multicoupler Module Base Module Cover Cables and Connectors	Basic 68 lb 15% contingency 10 Total 78 lbs
Communication Data Processing & Instrumentation	Data Processing Module No. 1	Digital Computer (16K memory) Interface Unit (Med. & Low Rate) Timer Module Base Module Cover Cables & Connectors	Basic 79 lb 15% contingency 12 Total 91 lbs
Communication Data Processing & Instrumentation	Antenna Module No. 1	Antenna (6 ft. dish) Antenna Gimbal and Base Antenna Feed (S&K Bands) Rotary Joint (K-Band) Rotary Joint (S-Band) Antenna Servo Motors & Gears Antenna, VHF Waveguide, Cables, Connectors	Basic 65 lb 15% contingency 10 Total 75 lbs
Electrical Power	Battery Module (3 req'd) No. 1 No. 2 No. 3	NiCd Battery, Type 7, 40AH (2) Charge Controller (2) Module Base Module Cover Cables & Connectors	Basic 187 lb 10% contingency 19 Total 206 lbs

Fig. 3b SEOS Subsystem Modules (2 of 3)

Subsystem	Module	Equipment in Module	Module Weight
Electrical Power	Power Distribution Module No. 1	Power Distribution Unit Regulator Converter Module Base Module Cover Cables & Connectors	Basic 105 lbs 15% contingency 15 Total 120 lbs
Electrical Power	Solar Power Module (2 req'd) No. 1 No. 2	Flexible Solar Array Extendable Boom Assembly Array Tension Member Array Container Extension Strut Assembly Module Base and Cover Module Tilt Mechanism Cables & Connectors	Basic 198 lbs 15% contingency 30 Total 228
Attitude Control	Attitude Control Module (4 req'd) Thruster Cluster No. 1 No. 2 No. 3 No. 4	Gas Storage Tank, 22" O.D. Regulator Valve Assembly Fill Valve Thruster Cluster Transducers (set) ACS Electronics Plumbing Module Base Module Cover Cables & Connectors	Basic 118 lbs 15% contingency 18 Total 136 lbs

Fig. 3c SEOS Subsystem Modules (3 of 3)

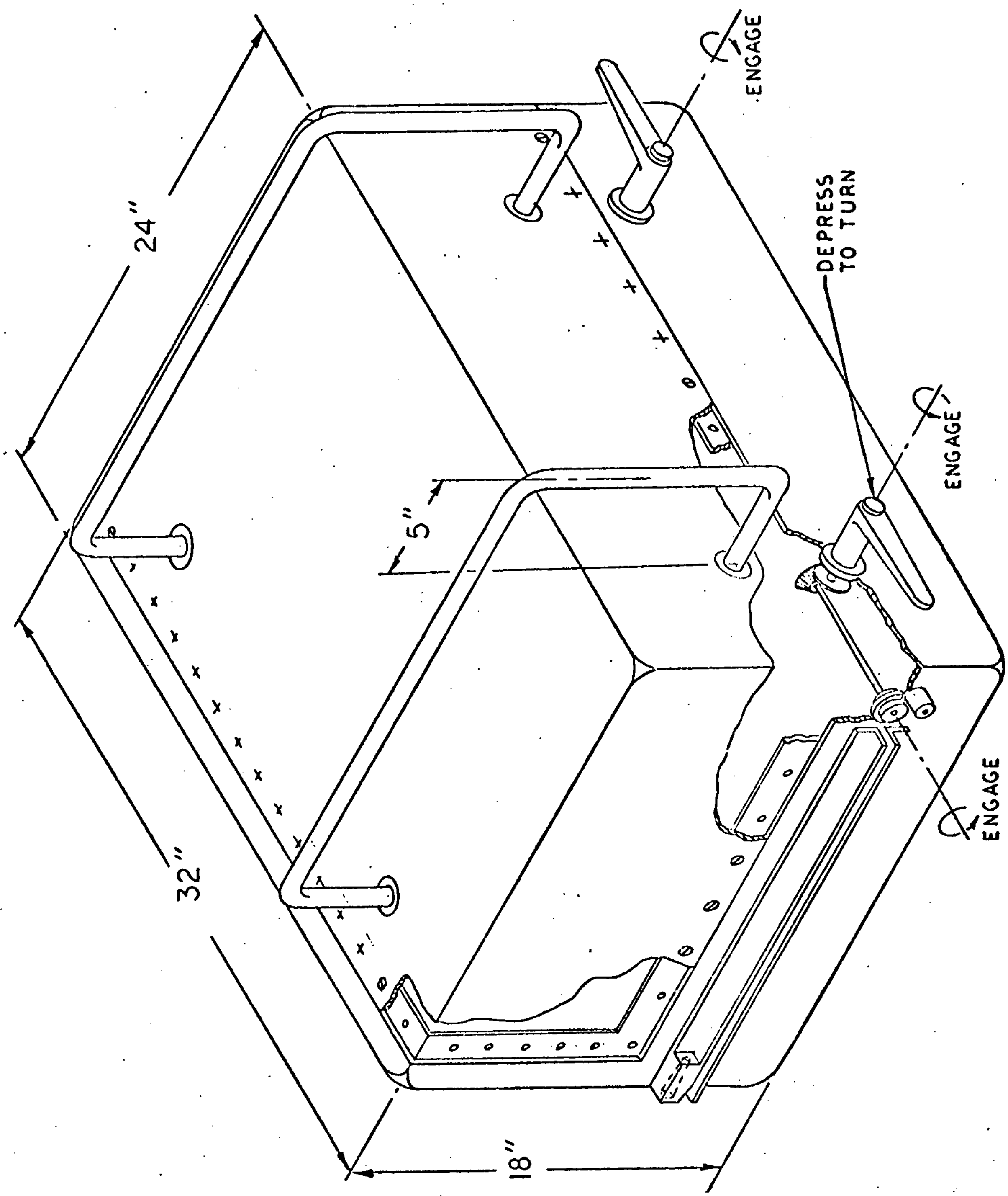


Fig. 4 Typical Standard Subsystem Module

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Weight Summary

The weight summary for the SEOS is shown in Fig. 5. No additional contingency is included for the Mission Equipment. It is assumed that the total weight of the Mission Equipment, 1024 lbs, obtained from the GSFC Earth Observatory Satellite Definition Phase Report already includes appropriate contingency. It is estimated that packaging the Mission Equipment into modules for in-orbit removal and replacement would add approximately 500 lbs to the dry weight of the spacecraft.

Thermal Control

The thermal control of the SEOS will be accomplished primarily by passive thermal control techniques. To the extent possible thermal control provisions will be limited to appropriate internal and external surface finishes and multilayer insulation. Equipment requiring temperature control within relatively narrow limits may require supplementary provisions such as thermostatically controlled heaters. Some Mission Equipment such as the Thematic Mapper will require equipment for the cooling of sensors; however, such equipment is assumed to be part of the Mission Equipment.

Subsystem	Contingency %	Weight lb
Structure & Mechanisms	15	1660
Environmental Control	15	150
Stabilization & Control	15	429
Communication, Data Processing, & Instrumentation	15	329
Electrical Power	20	1674
Attitude Control	15	504
Mission Equipment	Basic	1024
Mission Equipment supports, attachment hardware & electrical cables	15	168
	Dry Weight	5938
Propellant: Freon 14		400
	Total	6338

Fig. 5 SEOS Weight Summary

3.0 Subsystems of the SEOS

The major subsystems of the SEOS are as follows:

Stabilization and Control (S&C)
Communications, Data Processing and Instrumentation (CDPI)
Electrical Power (EPS)
Attitude Control (ACS)

The subsystem descriptions presented in LMSC Engineering Memos PE-102 through PE-105 are generally applicable and only the modifications are described in the following paragraphs.

Stabilization & Control Subsystem

A second Precision (Primary) Sensing module is added to increase the control accuracy and the reliability of the subsystem.

Communication, Data Processing & Instrumentation Subsystem

The computer memory is increased from 12K 24 bit words to 16K 24 bit words to accommodate the increased computational requirements of the Stabilization & Control Subsystem.

Electrical Power Subsystem

The rigid-panel solar array described in PE-104 and PE-106 is removed and replaced by a standardized, retractable, flexible solar array which is a variant of the flexible solar array specified for the Large Astronomical Observatory Spacecraft (LAOS) and described in PE-146. The solar array area is increased for this application by adding 16 ft^2 to each wing. In addition a mechanism is added to the module base to provide for tilting each wing through $\pm 45^{\circ}$ to accommodate adjustment for β-angle variation. No provision is made for single-axis tracking of the SEOS solar array as there is for that of LAOS. Single-axis tracking is not considered advantageous or cost effective for the SEOS application.

Attitude Control Subsystem

The standard Attitude Control subsystem module of the EOS is modified by the substitution of a 22" OD gas storage tank for the basic 16" OD tank, increasing the total-impulse provided by the module to 4500 lb-sec from 1500 lb-sec. The increase in total impulse is required to provide for drag makeup in the lower orbit of the SEOS (270 nm circular).

PE-166

GENERAL DESCRIPTION

CLUSTER EARTH OBSERVATORY

SPACECRAFT

PE-166

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: GENERAL DESCRIPTION CLUSTER EARTH OBSERVATORY SPACECRAFT (CLEOS)	EM NO: PE-166 REF: DATE: 31 March 1972
AUTHORS: F. C. Bolton — Prepared under coginzance of: Advanced Payload Systems, 69-02 Space Systems Division	APPROVAL: F C Bolton ENGINEERING Wayne J. Miller

PRELIMINARY

11 pages

1.0 General

A Cluster Spacecraft is defined as a spacecraft incorporating standard subsystem modules and capable of supporting concurrently the experiment/sensor packages of several of the missions defined in the NASA Mission Model. The Cluster Earth Observatory Spacecraft (CLEOS) is designed to support Earth observation missions including the following in various combinations:

Mission No.	Name
21	Polar Earth Observatory Satellite
25	TIROS
26	Polar Earth Resources Satellite
75	TOS Meteorological Satellite
77	Polar Earth Resources

The CLEOS may also support such missions as the following as auxiliary payloads:

Mission No.	Name
30	Small Applications Satellite
32	Cooperative Applications

2.0 Cluster Earth Observatory Spacecraft Configuration

The CLEOS design is derived from that of the Standard Earth Observatory Satellite (SEOS) described in LMSC Engineering Memos PE-102 to PE-106 and PE-156. The general configuration of the CLEOS is shown in Fig. 1.

The principal physical differences between CLEOS and SEOS are summarized in the following table:

Physical Characteristic	SEOS	CLEOS
Length (ft)	20	30
Cross Section (ft)	6x8	6x8
Subsystem Compartments	18	27
Earth Viewing Surface (ft^2)	160	240
Weight (lbs)	6240	10900
Flexible Solar Array Area (ft^2)	416	1040

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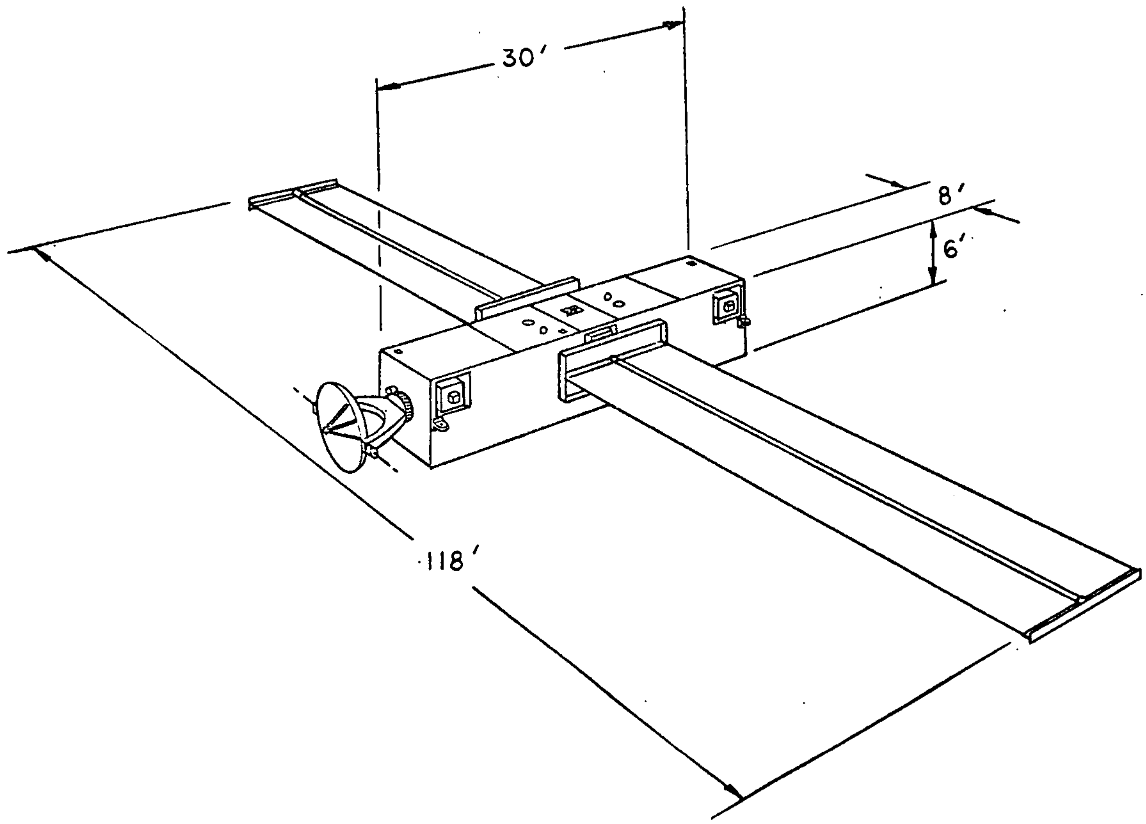

Fig. 1 General Configuration Cluster Earth Observatory Spacecraft

The standard subsystems designed for the SEOS are directly applicable to the CLEOS but require some augmentation to meet the greater requirements of the CLEOS. This augmentation can be accomplished by the addition of standard subsystem modules and is discussed in more detail in Section 3.

The complement of standard subsystem modules installed in the CLEOS is listed in Fig. 2a through 2c.

The locations of standard subsystem modules and of a typical set of mission experiments/sensors are shown in Fig. 3a and 3b. There are four empty subsystem module compartments available to accommodate further growth of the subsystems if necessary; or they may be used to accommodate auxiliary experiments designed to be compatible with the standard interfaces presented in the compartments. In addition, one large compartment on the Earth-viewing side of the spacecraft is available for additional sensors. Sensors may be added, substituted, updated, adjusted, or repaired during the regularly scheduled visits of the Shuttle to the orbiting CLEOS.

A typical standard subsystem module is shown in Fig. 4. It is designed to be guided into its location in the spaceraft by rails and aligned and supported by two inboard pins and two outboard cams that engage machined grooves in the rails. The cams also transmit force from the cam actuator handles on the outboard face of the module to accomplish the controlled engagement and disengagement of the bulkhead-type electrical connectors on the inboard face of the module. The two wraparound handles are designed to facilitate the handling of the module in orbit by a Shuttle crewman. 563

Subsystem	Module	Equipment in Module	Module Weight (lb)
Stabilization & Control	Primary Sensing Module (2 req'd) No. 1 No. 2	• Fixed Head Star Trackers (2) • FHST Electronics (2) • Three-Axis Rate Sensor • Precision Equipment Mount • Module Base • Module Cover • Cables and Connectors	Basic 91 lb 15% contingency 14 Total 105 lb
Stabilization & Control	Secondary Sensing Module No. 1	• Sun Aspect Sensor (5) • Sun Aspect Sensor Electronics • Rate Gyro Package • Secondary Stabilization & Control Electronics • Module Base • Module Cover • Cables & Connectors	Basic 56 lbs 15% contingency 8 Total 64 lbs
Stabilization & Control	Reaction Torque Module No. 1	• Reaction Wheel (3)(50 ft-lb-sec) • Wheel Support and Safety Shield • Wheel Drive Electronics • Magnetic Torquer (3) • Mag. Torquer Electronics (3) • Module Base • Module Cover • Cables & Connectors	Basic 237 lbs 15% contingency 23 Total 260 lbs
Communication Data Processing & Instrumentation	K_u-Band Communication Module (2 required) No. 1 No. 2	• K-band TWTA (50 watts out) (2) • K-band PLL Receiver • K-band QPSK Modulator/Driver • K-band Multicoupler • Interface Unit (High Rate) • Module Base • Module Cover • Waveguide, Cables, Connectors	Basic 74 lbs 15% contingency 11 Total 85 lbs

Fig. 2a CLEOS Subsystem Modules (1 of 3)

Subsystem	Module	Equipment in Module	Module Weight (lb)
Communication Data Processing & Instrumentation	S-Band/VHF Communication Module No. 1	S-Band Transmitter (10 watts out) S-Band Receiver S-Band QPSK Modulator/Driver S-Band Multicoupler Delay Lock Loop Correlator (2) Antenna Servo Electronics VHF Transmitter (5 watts out) VHF Receiver/Demodulator VHF Multicoupler Module Base Module Cover Cables and Connectors	Basic 68 lb 15% contingency 10 Total 78 lbs
Communication Data Processing & Instrumentation	Data Processing Module No. 1	Digital Computer Interface Unit (Med. & Low Rate) Timer Module Base Module Cover Cables & Connectors	Basic 85 lb 15% contingency 13 Total 98 lbs
Communication Data Processing & Instrumentation	Antenna Module No. 1	Antenna (6 ft. dish) Antenna Gimbal and Base Antenna Feed (S&K Bands) Rotary Joint (K-Band) Rotary Joint (S-Band) Antenna Servo Motors & Gears Antenna, VHF Waveguide, Cables, Connectors	Basic 65 lb 15% contingency 10 Total 75 lbs
Electrical Power	Battery Module (8 required) Nos. 1,2,3,4 Nos. 5,6,7,8	NiCd Battery, Type 7, 40AH (2) Charge Controller (2) Module Base Module Cover Cables & Connectors	Basic 187 lb 10% contingency 19 Total 206 lbs

Fig. 2b CLEOS Subsystem Modules (2 of 3)

Subsystem	Module	Equipment in Module	Module Weight
Electrical Power	Power Distribut. Module No. 1	Power Distribution Unit Regulator Converter Module Base Module Cover Cables & Connectors	Basic 105 lbs 15% contingency 15 Total 120 lbs
Electrical Power	Solar Power Module (2 Required) No. 1 No. 2	Flexible-fold Solar Array Extendable Boom Assy; Array Container Array tension member; Bearing Assy Extension strut Module Base & Cover Cables and connectors	Basic 370 lbs 15% contingency 55 Total 425 lbs
Attitude Control	Attitude Control Module (4 Required) No. 1 No. 2 No. 3 No. 4	Gas Storage Tank, 22" O.D. Regulator Valve Assembly Fill Valve Thruster Cluster Transducers (set) ACS Electronics Plumbing Module Base Module Cover Cables & Connectors	Basic 118 lbs 15% contingency 18 Total 136 lbs

Fig. 2c Standard CLEOS Subsystem Modules (3 of 3)

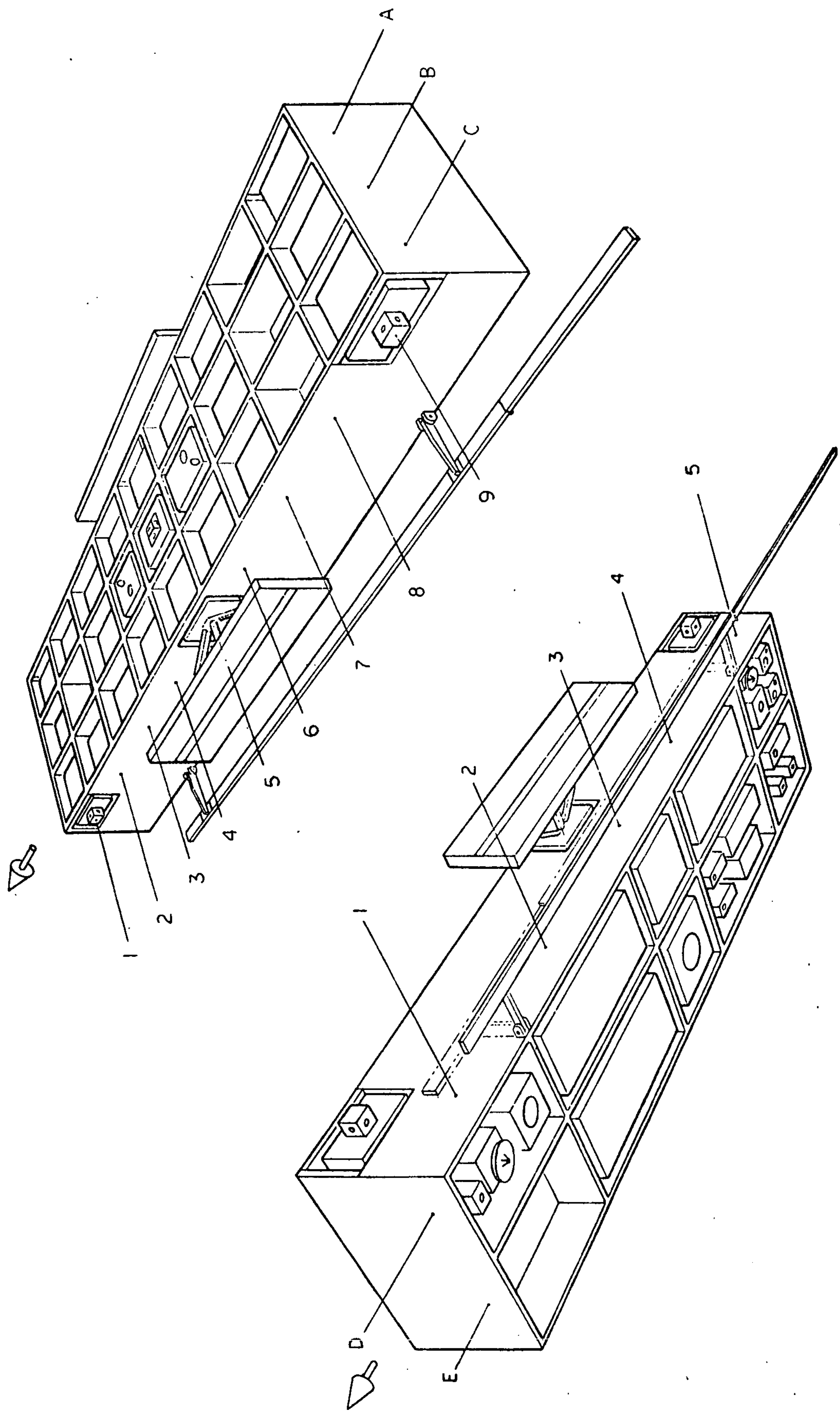

Fig. 3a Subsystem Module and Sensor Locations Cluster Earth Observatory Spacecraft

Location	Subsystem Module	Location	Sensor
A-1	Attitude Control	D-1	Surface Composition Mapping Radiometer
A-2	S-Band/VHF Communications		Imaging Radiometer
A-3	Battery Power		Radar Cloud Top Ranger
A-4	Battery Power	D-2	Passive Microwave Radiometer (λ = 0.81 cm)
A-5	Solar Power	D-3	Synthetic Aperture Radar
A-6	Battery Power	D-4	Passive Microwave Radiometer (λ = 2.81 cm)
A-7	Battery Power	D-5	Radar Altimeter
A-8	Empty		Temperature Profile Radiometer
A-9	Attitude Control		Multispectral TV Camera (2)
B-1	K_u Band Communications	E-1	Empty
B-2	Empty	E-2	Passive Microwave Radiometer (λ = 6.01 cm)
B-3	Data Processing	E-3	Thematic Mapper
B-4	Primary Sensing	E-4	Cloud Physics Radiometer
B-5	Secondary Sensing		Sea Surface Temperature Radiometer
B-6	Primary Sensing		Passive Microwave Radiometer (λ = 1.67 cm)
B-7	Power Distribution		Passive Microwave Radiometer (λ = 1.40 cm)
B-8	Empty	E-5	Ocean Scanning Spectrophotometer
B-9	Reaction Torque		Atmospheric Pollution Sensor
C-1	Attitude Control		Upper Atmosphere Sounder
C-2	K_u-Band Communications		
C-3	Battery Power		
C-4	Battery Power		
C-5	Solar Power		
C-6	Battery Power		
C-7	Battery Power		
C-8	Empty		
C-9	Attitude Control		

Fig. 3b Subsystem Module and Sensor Locations - Cluster Earth Observatory Spacecraft

Lockheed Missiles & Space Company

EM NO: PE-166
DATE: 31 March 1972

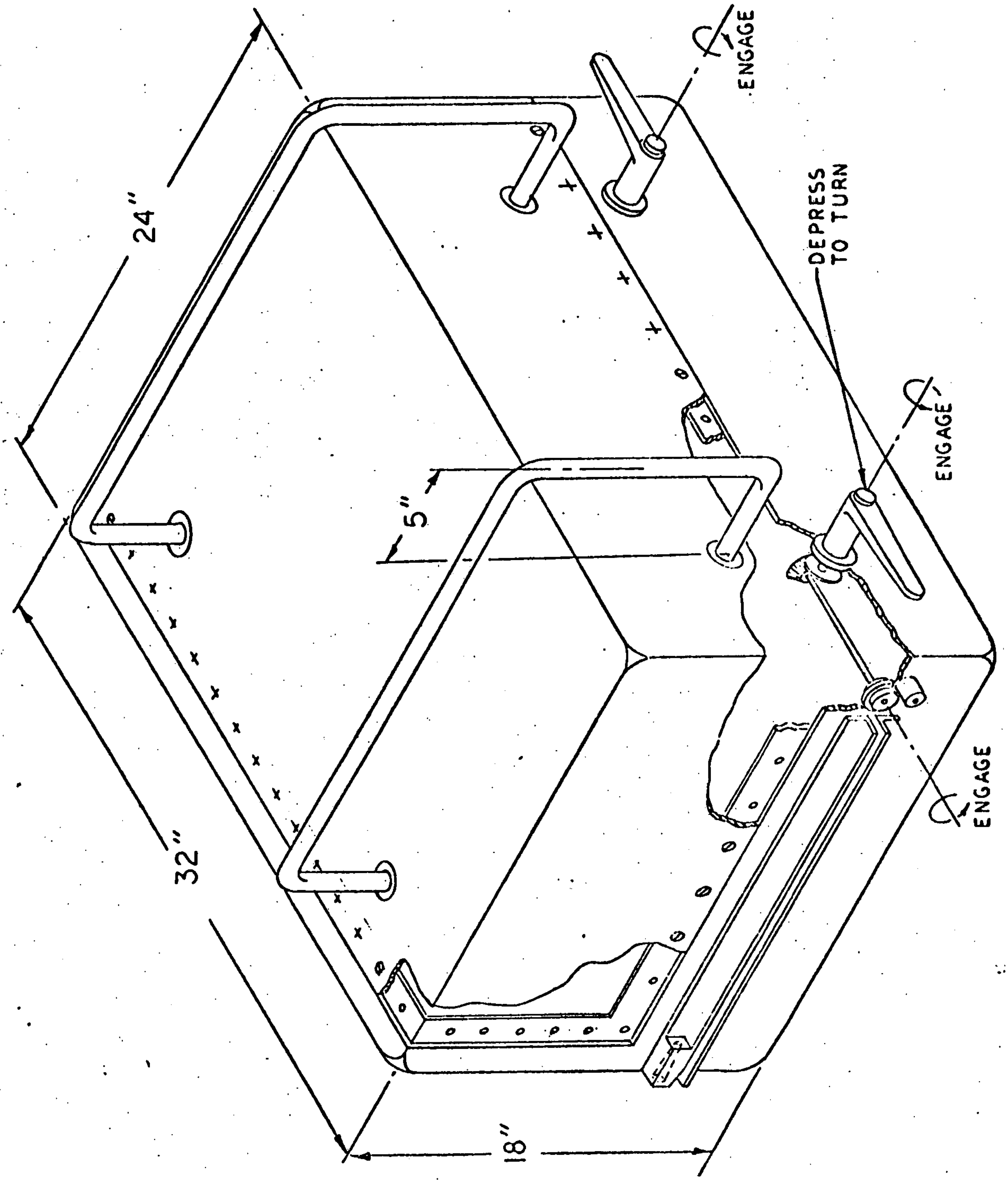

Fig. 1 Typical Standard Subsystem Module

Weight Summary

The weight summary for the CLEOS is presented in Fig. 5. The on-orbit weight of the CLEOS is conservatively estimated to be 10897 lbs. However, the baseline Shuttle can transport 32,000 lbs to the CLEOS orbit, namely 270 nm circular sun synchronous (inclination 97.4°). The performance margin of more than 20000 lbs provides for the carrying of spare modules, checkout equipment, additional propellant for orbital maneuvers, other spacecraft, etc.

Thermal Control

The thermal control of the CLEOS will be accomplished primarily by passive thermal control techniques. To the extent possible thermal control of the spacecraft will be accomplished by the application of appropriate internal and external surface finishes and multilayer insulation. Equipment requiring temperature control within relatively narrow limits may require supplementary methods such as thermostatically controlled heaters. Some mission equipment such as the thematic mapper or the synthetic aperture radar will require special provisions for cooling; however, such provisions are assumed to be part of the mission equipment.

3.0 Major Subsystems of CLEOS

The major subsystem of the CLEOS are as follows:

Stabilization & Control (S&C)
Communication, Data Processing & Instrumentation (CDPI)
Electrical Power (EPS)
Attitude Control (ACS)

The modular subsystems designed for the Standard Earth Observatory Satellite, described in LMSC Engineering Memos PE-102 through PE-106 and PE-156, are applicable to the CLEOS but require some augmentation to meet the requirements of the larger spacecraft. The descriptions of the subsystems are not repeated in this Engineering Memo; only the augmentation of the subsystems is discussed in the following paragraphs.

Stabilization & Control (S&C) Subsystem

The S&C subsystem of the SEOS consists of the following standard modules:

Quantity	Module Name
1	Primary Sensing
1	Secondary Sensing
1	Reaction Torque (10 ft-lb-sec/wheel)

To increase the capability of this subsystem to meet CLEOS requirements a second Primary Sensing module is added, and the Reaction Torque module (10 ft-lb-sec/wheel) is replaced by a standard Reaction Torque module rated at 50 ft-lb-sec/wheel.

Subsystem	Contingency (%)	Weight (lbs)
Structure & Mechanisms	15	2750
Environmental Control	15	250
Stabilization & Control	15	534
Communication, Data Processing & Instrumentation	15	421
Electrical Power	20	3408
Attitude Control	15	544
Mission Equipment	15	2310
Mission Equipment Support and Attachment Hardware	15	280
	Dry Weight	10497
ACS Propellant (Freon 14)		400
	On-Orbit Weight	10897

Fig. 5 Weight Summary - Cluster Earth Observatory Spacecraft

The second Primary Sensing module increases the quantity and quality of star data available for attitude determination and control, and increases the reliability of the subsystem.

The higher rated Reaction Torque module provides the large torques required for control of the CLEOS.

Communication, Data Processing and Instrumentation (CDPI) Subsystem

The CDPI subsystem of the SEOS consists of the following standard modules:

Quantity	Module Name
1	K_u-Band Communication
1	S-Band/VHF Communication
1	Data Processing
1	Antenna

To increase the capability of this subsystem to meet CLEOS requirements a second K_u-Band Communication module is added, doubling the quantity of wide band data that can be transmitted.

Electrical Power Subsystem (EPS)

The Electrical Power subsystem of the SEOS consists of the following standard modules:

Quantity	Module Name
3	Battery Power
1	Power Distribution
2	Solar Power

It is capable of providing an average of 1000 watts. To increase the average power capability to the 2600 watts required by the CLEOS five standard Battery Power modules are added, and the total area of the flexible solar array is increased from 416 ft^2 to 1040 ft^2. The power handling capability of the power distribution unit and of the regulator/converter are increased in proportion to the increase in load.

Attitude Control Subsystem (ACS)

The Attitude Control Subsystem of the SEOS consists of four identical ACS modules, each providing 1500 lb-sec of impulse. To provide the total impulse requirements of CLEOS, approximately 16,000 lb-sec for two years, the 16" OD gas storage tank of the basic module is replaced by a 22" OD tank.

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PE-186

GENERAL DESCRIPTION

CLUSTER ASTRONOMICAL OBSERVATORY

SPACECRAFT

PE-186

Lockheed Missiles & Space Company

ENGINEERING MEMORANDUM

TITLE: GENERAL DESCRIPTION CLUSTER ASTRONOMICAL OBSERVATORY SPACECRAFT	EM NO: PE-186 REF: DATE: 31 March 1972
AUTHORS: F. C. Bolton — Prepared under cognizance of: Advanced Payload Systems, 69-02 Space Systems Division	APPROVAL: F C Bolton Wayne J. Miller

PRELIMINARY

15 pages

1.0 General

A Cluster Spacecraft is defined as a spacecraft, incorporating standard subsystem modules, and capable of supporting concurrently the experiment/sensor packages of several of the missions defined in the NASA mission model. One such spacecraft is the Cluster Astronomical Observatory Spacecraft (CAOS) which can support the High Energy Astronomical Observatory (HEAO) and Large Stellar Telescope (LST) missions concurrently.

It can also support the Large Solar Observatory (LSO) and Large Radio Observatory (LRO) missions simultaneously.

The Cluster Astronomical Observatory Satellites, which include both the spacecraft (CAOS) and the mission experiment packages, are expected to be flown in a standard circular orbit at 324 nm altitude and 30° inclination to facilitate the revisit of two or more such satellites by a single Shuttle flight.

2.0 Description of Cluster Astronomical Observatory Spacecraft

The Cluster Astronomical Spacecraft shown in Fig. 1, with both HEAO and LST experiment packages shown in phantom, is derived from the Standard Large Astronomical Observatory Spacecraft described in LMSC Engineering Memo, PE-146, to which reference should be made.

The length of the LAOS structure is increased by 2.5 ft to provide six more compartments for subsystem modules or for auxiliary experiment sensors. The solar array is larger than that of the LAOS to generate the higher average power required.

In flight the +X axis of the CAOS is pointed at a target star and the satellite is rolled about the X-axis until the center line of the solar array is normal to the satellite/sun line. The solar array is then rotated about its center line until the surface of the array is normal to the solar radiation. For every pointing direction of the LST, an appropriate combination of satellite roll and solar array rotation maintains the array surface normal to the solar radiation.

Two tracking antennas are installed on the CAOS as shown in Fig. 2, rather than one as on the LAOS; one on the sunlite side and one on the side opposite, to provide continuous communication with the TDRS network regardless of satellite attitude. The switching of signals from one antenna to the other is under control of the spacecraft computer.

The HEAO is supported by a two-degree of freedom yoke assembly, which permits the HEAO sensors to be pointed to any point in the celestial hemisphere opposite to the pointing direction of the LST. The LST is pointed to a target star, and then the satellite is rolled until the center line of the solar array is normal to the satellite/sun line. This attitude is held while the HEAO is rotated about the spacecraft X-axis by means of the Axial Rotation Mechanism shown in Fig. 2, and tilted in its

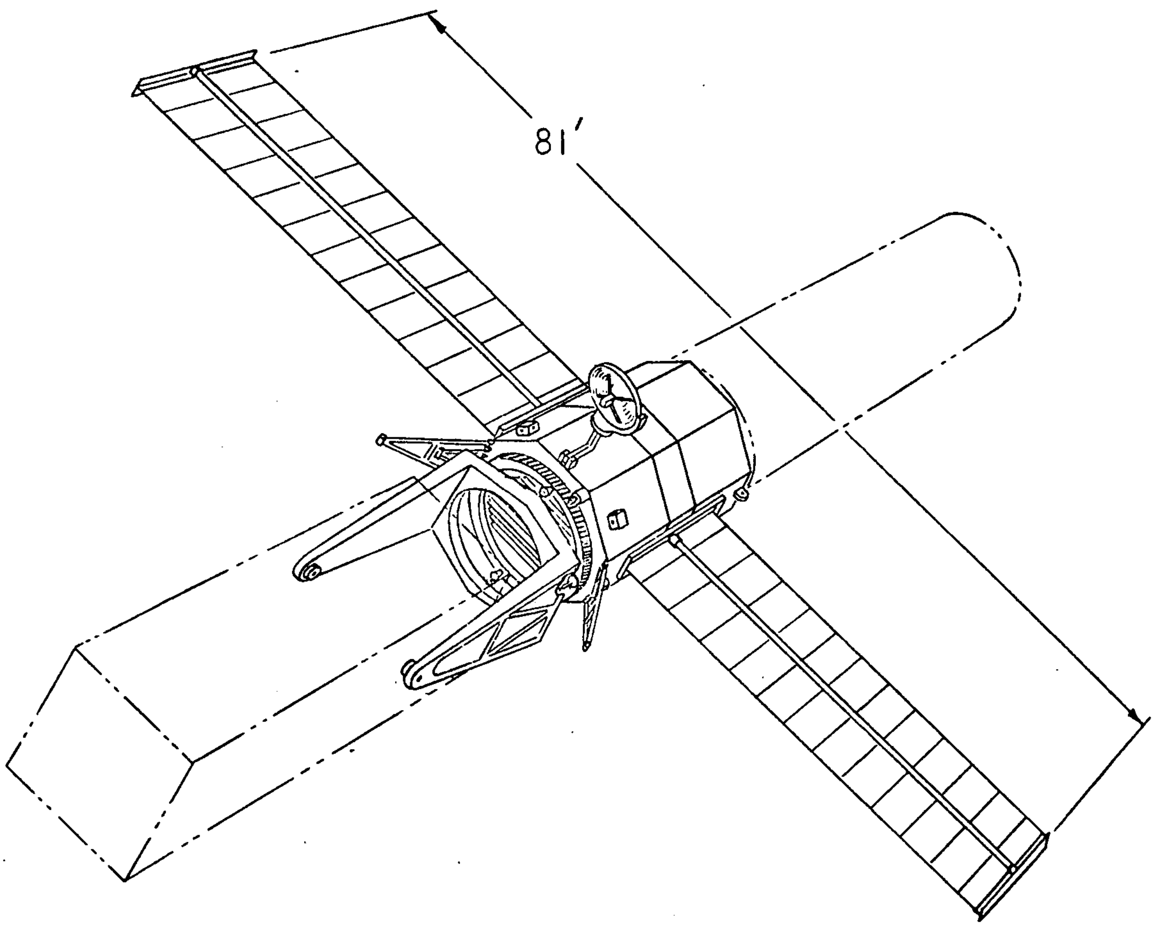


Fig. 1 Cluster Astronomical Observatory Spacecraft with HEAO and LST Experiment Packages

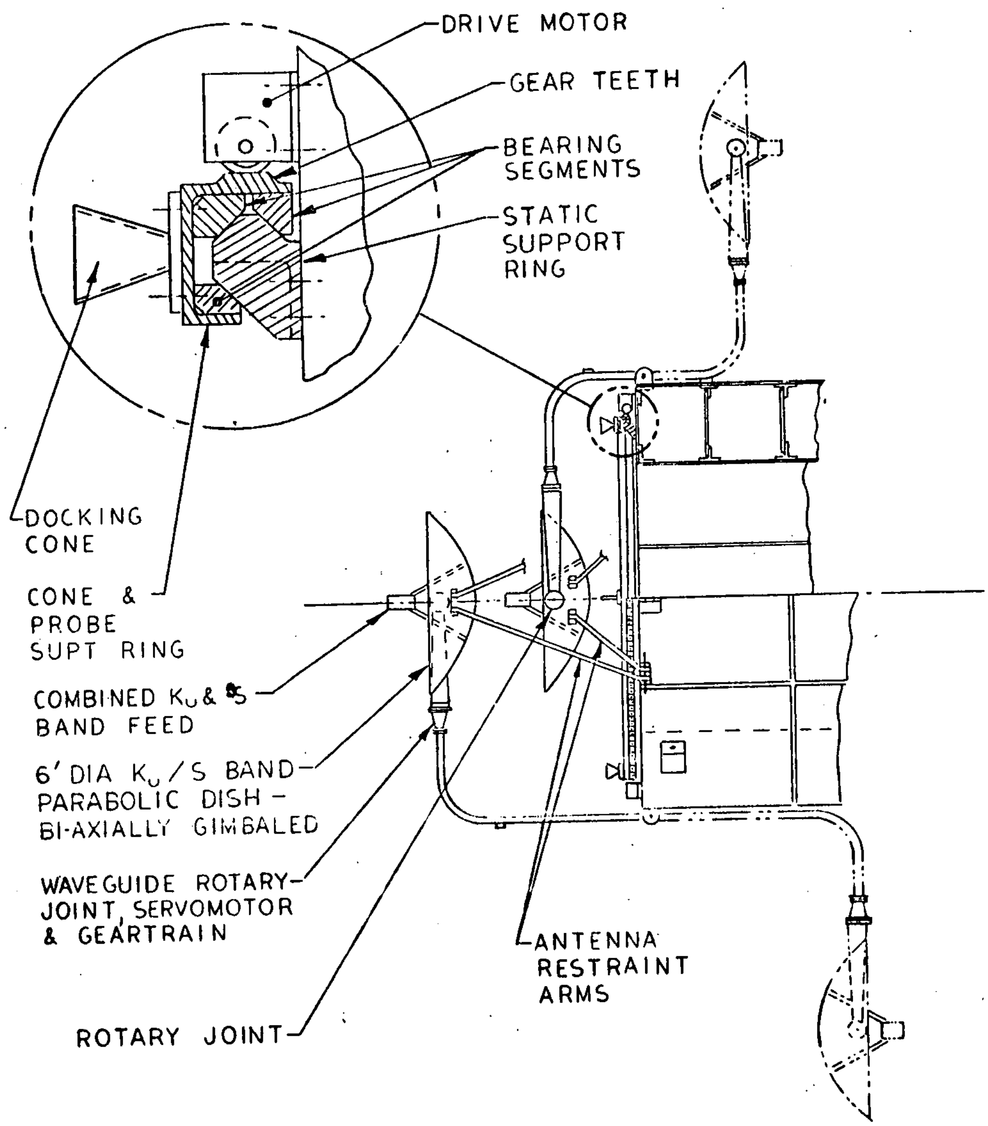


Fig. 2 Antenna Installation and Axial Rotation Mechanism Concept, Cluster Astronomical Observatory Spacecraft

yoke until the desired HEAO pointing direction is attained. The satellite is then held in the fine pointing mode, under the control of the LST fine attitude control system, while LST and HEAO observations proceed simultaneously. Reconciliation of the pointing programs of the two observatories is required, but that is a minor concession to make to attain the cost-savings that clustering affords.

The rotation ring assembly is attached to the spacecraft structure prior to its placement into orbit with the LST. Docking cones and probes are installed on the aft face of the rotation ring to accept mating probes and cones on the yoke assembly. The HEAO is mounted in the yoke assembly on the ground, launched by the Shuttle and docked with the Spacecraft/LST to form the Cluster Spacecraft. An umbilical, engaged at the time of docking provides electrical connection of the HEAO to the Spacecraft subsystems.

The CAOS is also capable of supporting the LSO and LRO payloads simultaneously. The satellite is stabilized with the LSO pointed to the sun. In order that the LRO may point to any point in the celestial sphere the base of the yoke assembly must be modified to incorporate approximately 90° of tilt about an axis parallel to the axis through the LRO support pivots.

The CAOS structure and forward payload adapter are essentially the same as those of the LAOS described in PE-146.

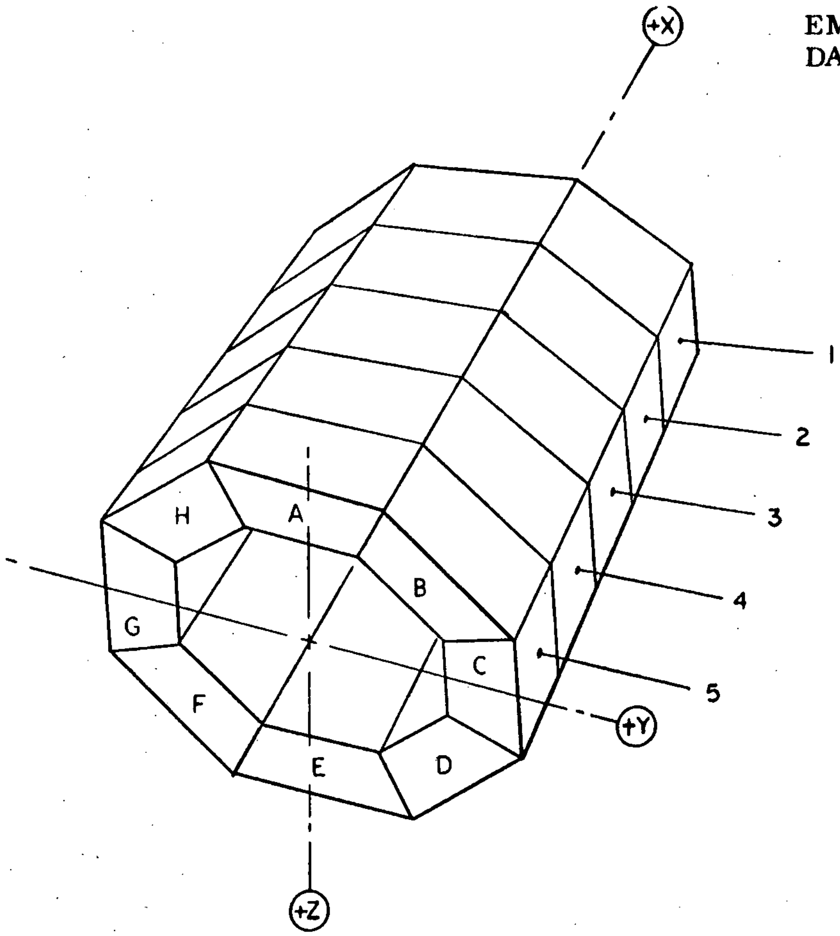
The locations of the standard subsystem modules in the CAOS are shown in Fig. 3. The empty compartments identified may be used for the installation of auxiliary sensor modules, designed to be compatible with the standard mechanical and electrical interface provisions of the compartments.

The standard subsystem modules of the CAOS are listed in Fig. 4a through 4b, together with a list of the major equipment contained in each module and the weight of each module.

A typical standard equipment module is shown in Fig. 5. The module is designed to be guided into its location in the spacecraft by rails and aligned and supported by two inboard pins and two outboard cams that engage machined grooves in the rails. The cams also transmit force from the cam actuator handles on the outboard face of the module to accomplish the controlled engagement and disengagement of the bulkhead-type electrical connectors on the in-board face of the module. The two wrap-around handles are designed to facilitate the handling of the module in orbit by a Space Shuttle crewman.

<u>Weight Summary</u>

The Weight Summary for the CAOS is presented in Fig. 6. The on-orbit weight of the CAOS is 11,818 lbs, including 400 lbs of Freon 14 attitude control propellant. The approximate weight of the CAOS with HEAO and LST experiment packages is 42,000 lbs.



Location	Module	Location	Module
A-1	Secondary Sensing	E-1	Empty
A-2	Empty	E-2	Data Processing
A-3	Empty	E-3	Power Distribution
A-4	Empty	E-4	Empty
A-5	K_u-Band Communication	E-5	S-Band/VHF Communication
B-1	Empty	F-1	Primary Sensing
B-2	Battery Power	F-2	Battery Power
B-3	Reaction Torque	F-3	Reaction Torque
B-4	Battery Power	F-4	Battery Power
B-5	Attitude Control	F-5	Attitude Control
C-1	Empty	G-1	Empty
C-2	Empty	G-2	Empty
C-3	Solar Array Drive	G-3	Solar Array Drive
C-4	Empty	G-4	Empty
C-5	Empty	G-5	Empty
D-1	Primary Sensing	H-1	Empty
D-2	Battery Power	H-2	Battery Power
D-3	Reaction Torque	H-3	Empty
D-4	Battery Power	H-4	Battery Power
D-5	Attitude Control	H-5	Attitude Control

Fig. 3 Subsystem Module Locations - Cluster Astronomical Observatory Spacecraft

Subsystem	Module	Equipment in Module	Module Weight (lb)
Stabilization & Control	Primary Sensing Module (2 Req'd.) No. 1 No. 2	• Fixed Head Star Trackers (2) • FHST Electronics (2) • Three-Axis Rate Sensor • Precision Equipment Mount • Module Base • Module Cover • Cables and Connectors	Basic 91 lb 15% contingency 14 Total 105 lb
Stabilization & Control	Secondary Sensing Module No. 1	• Sun Aspect Sensor (5) • Sun Aspect Sensor Electronics • Rate Gyro Package • Secondary Stabilization & Control Electronics • Module Base • Module Cover • Cables & Connectors	Basic 56 lbs 15% contingency 8 Total 64 lbs
Stabilization & Control	Reaction Torque Module (3 Req'd) No. 1 No. 2 No. 3	• Reaction Wheel (3) • Wheel Support and Safety Shield • Wheel Drive Electronics • Magnetic Torquer (3) • Mag. Torquer Electronics (3) • Module Base • Module Cover • Cables & Connectors	Basic 226 lbs 15% contingency 34 Total 260 lbs
Communication Data Processing & Instrumentation	K_u-Band Communication Module No. 1	• K-band TWTA (50 watts out)(2) • K-band PLL Receiver • K-band QPSK Modulator/Driver • K-band Multicoupler • Interface Unit (High Rate) • Module Base • Module Cover • Waveguide, Cables, Connectors	Basic 74 lbs 15% contingency 11 Total 85 lbs

Fig. 4a Standard CAOS Subsystem Modules (1 of 3)

Subsystem	Module	Equipment in Module	Module Weight (lb)
Communication, Data Processing & Instrumentation	S-Band/VHF Communication Module No. 1	S-Band Transmitter (10 watts out) S-Band Receiver S-Band QPSK Modulator/Driver S-Band Multicoupler Delay Lock Loop Correlator (2) Antenna Servo Electronics VHF Transmitter (5 watts out) VHF Receiver/Demodulator VHF Multicoupler Module Base Module Cover Cables and Connectors	Basic 68 lb 15% contingency 10 Total 78 lbs
Communication, Data Processing & Instrumentation	Data Processing Module No. 1	Digital Computer Interface Unit (Med. & Low Rate) Timer Module Base Module Cover Cables & Connectors	Basic 79 lbs 15% contingency 12 Total 91 lbs
Communication Data Processing & Instrumentation	Antenna Module (2 Req'd) No. 1 No. 2	Antenna (6 ft. dish) Antenna Gimbal and Base Antenna Feed (S&K Bands) Rotary Joint (K-Band) Rotary Joint (S-Band) Antenna Servo Motors & Gears Antenna Servo Electronics Antenna, VHF Waveguide, Cables, Connectors	Basic 65 lbs 15% contingency 10 Total 75 lbs
Electrical Power	Battery Module (8 Req'd) No. 1 No. 5 No. 2 No. 6 No. 3 No. 7 No. 4 No. 8	NiCd Battery, Type 7, 40AH(2) Charge Controller (2) Module Base Module Cover Cables & Connectors	Basic 187 lbs 10% contingency 19 Total 206 lbs

Fig 4b Standard CAOS Subsystem Modules (2 of 3)

Subsystem	Module	Equipment in Module	Module Weight
Electrical Power	Power Distribution Module No. 1	Power Distribution Unit Regulator Converter Module Base Module Cover Cables & Connectors	Basic 105 lbs 15% contingency 15 Total 120 lbs
Electrical Power	Solar Power Module (2 Req'd) No. 1 No. 2	Flexible-fold Solar Array Extendable Boom Assy. Array container Array tension member Bearing Assembly Extension Strut Module Base & Cover Cables and connectors	Basic 278 lbs 15% contingency 42 Total 320 lbs
Electrical Power	Solar Array Drive Module (2 Req'd) No. 1 No. 2	Drive Motor Assembly Electronics Slip ring and bearings Module Base and Cover Cables and connectors	Basic 56 lbs 15% contingency 8 Total 64 lbs
Attitude Control	Attitude Control Module (4 Req'd) No. 1 No. 2 No. 3 No. 4	Gas Storage Tank, 22" O.D. Regulator Valve Assembly Fill Valve Thruster Cluster Transducers ACS Electronics Plumbing Module Base Module Cover Cables & Connectors	Basic 118 lbs 15% contingency 18 Total 136 lbs

Fig. 4c Standard CAOS Subsystem Modules (3 of 3)

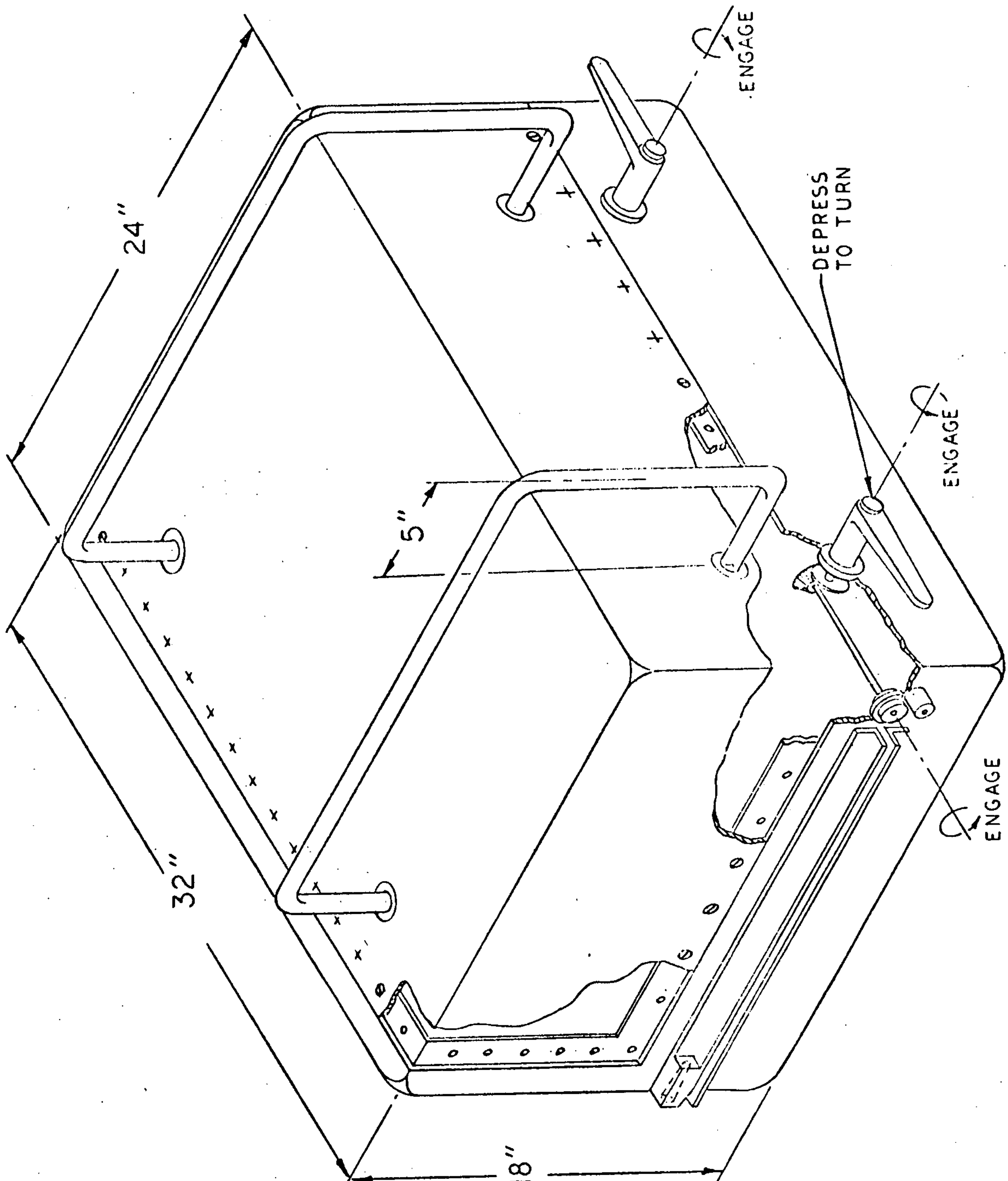


Fig. 5 Typical Standard Equipment Module

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Subsystem	Contingency	Weight
Structure & Mechanisms	15%	4053 lb
Environmental Control	15%	206
Stabilization & Control	15%	1054
Communication, Data Processing, Instrumentation	15%	404
Electrical Power	20%	3224
Attitude Control	15%	544
CAOS Dry Weight		9485 lb
Adapter, Mission Payload (+X)	15%	491
Adapter, Mission Payload (-X)	15%	1242
CAOS + Payload Adapters, Dry Weight		11418 lb
Propellant, Freon 14		400
CAOS + Payload Adapters, Wet weight		11818 lb

Fig. 6 Standard CAOS Weight Summary

3.0 Major Subsystems of CAOS

The major subsystems of the CAOS are as follows:

Stabilization & Control (S&C)
Communication, Data Processing & Instrumentation (CDPI)
Electrical Power (EPS)
Attitude Control (ACS)

They are very similar to the same subsystems of the LAOS described in PE-146. Only the few significant differences will be discussed in the following paragraphs.

3.1 Stabilization and Control Subsystem

Two significant changes in the LAOS S&C Subsystem design/functional requirements are introduced by clustering the LST and HEAO experiments on a single spacecraft. These specification differences are necessitated by the large change in mass properties, and by the general arrangement of CAOS, which places HEAO on a gimbal mount at a significant distance from the spacecraft alignment reference plane.

The S&C subsystem requirements should not, however, be discernibly affected by possible differences in: (1) orbit altitude and inclination, (2) pointing accuracy/stability, (3) operating attitudes, (4) mission duration/duty cycle, or (5) reliability allocations.

Effects of Size and Weight Change

The combined HEAO and LST experiments will weigh over 30,000 lb, effectively doubling the LAOS experiment weight. The estimated effect on the LAOS inertia dyad with the HEAO in the worst orientation is:

$$I_x \text{ (CAOS)} = 1.2 \times \text{LAOS}$$
$$I_{y,z} \text{(CAOS)} = 1.5 \times \text{LAOS}$$

Consequently, at the same operating altitude and with the same observation attitudes as LAOS, the gravity gradient torques will be about 150% higher. The impact of this on the S&C subsystem parameters will be to require greater reaction wheel angular momentum storage and control torque capabilities for CAOS in about the same ratio. It is recommended, therefore, that a third reaction wheel module be added, providing thereby 150 ft-lb-sec of angular momentum storage capability about each spacecraft axis.

Note that adding the third reaction torque module simultaneously increases the control torque capability by 50%. It is desirable that the HEAO experiment be driven from one pointing attitude to another at a slow enough rate not to saturate the momentum storage capability. On the average this capability will be ± 150 ft-lb-sec. The allowable rate is 90 deg of HEAO rotation in 7.5 min, from:

$$\Delta W = \frac{\Delta H}{I} \quad \begin{array}{l} \Delta H = 150 \text{ ft-lb-sec} \\ I_y = I_z = 40{,}000 \text{ slug ft}^2 \end{array}$$

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$$\Delta W = \frac{150}{40,000} \times 57.3 = 0.20 \text{ deg/sec}$$

$$t_{90^\circ} = \frac{90}{0.20 \times 60} = \underline{\underline{7.5 \text{ minutes}}}$$

This is a reasonable time to allot to this function.

Effects of HEAO Mounting Arrangement

The design concept for CAOS places HEAO at the opposite end of the spacecraft from LST, which is fastened directly to the spacecraft alignment reference surface. A chain of alignment error sources between this reference surface and the HEAO boresight is thereby created. Furthermore, HEAO will be gimballed in a two-axis yoke for full hemispheric freedom of target selection, adding several more alignment tolerances to the chain. The combined alignment error cannot exceed ± 118 arc sec*. Nine alignment error contributors have been identified**. These are:

E_1 = Fixed Head Tracker to Module

E_2 = Module to Alignment Reference Plane

E_3 = Spacecraft Structure to Alignment Reference Plane

E_4 = Azimuth Bearing Axis to Spacecraft Structure

E_5 = Yoke Azimuth Bearing Axis to Spacecraft Azimuth Bearing Axis

E_6 = Yoke Structure to Yoke Azimuth Bearing Axis

E_7 = Yoke Elevation Bearing Axis to Yoke Structure

E_8 = HEAO Bearing Axis to Yoke Elevation Bearing Axis

E_9 = HEAO Sensing Axis to HEAO Elevation Bearing Axis

$$E_{total} = \sum_{i=1}^{i=9} E_i^2 \leq 118 \text{ arc sec}$$

The error contributors are located on a simplified sketch of the CAOS, Fig. 7.

* To keep the RSS total pointing error under ±2 arc min including star tracker error (±20 arc sec).

**Two of which also apply to the LST experiment.

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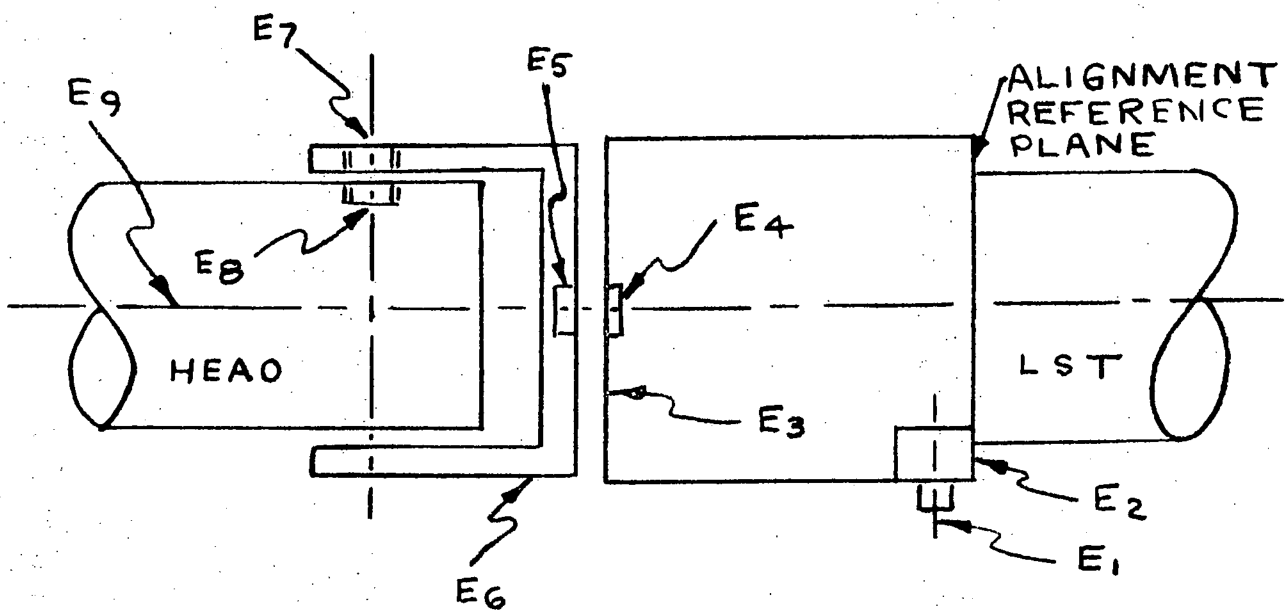

E_1 = Fixed Head Tracker to Module
E_2 = Module to Alignment Reference Plane
E_3 = Spacecraft Structure to Alignment Reference Plane
E_4 = Azimuth Bearing Axis to Spacecraft Structure
E_5 = Yoke Azimuth Bearing Axis to Spacecraft Azimuth Bearing Axis
E_6 = Yoke Structure to Yoke Azimuth Bearing Axis
E_7 = Yoke Elevation Bearing Axis to Yoke Structure
E_8 = HEAO Bearing Axis to Yoke Elevation Bearing Axis
E_9 = HEAO Sensing Axis to HEAO Elevation Bearing Axis

$$E_{total} = \sum_{i=1}^{i=9} E_i^2 \leq 118 \text{ arc sec}$$

Fig. 7 HEAO Alignment Error Contributors
RSS_{Allow} = 118 arc sec

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Of course, an alternative to controlling the spacecraft-experiment alignment through tolerance control and initial alignment is to measure the actual alignment at regular intervals during on-orbit operations. One effective way to implement this is to mount one fixed head star tracker directly on the HEAO experiment. Its measured star angles are a precise indication of the HEAO attitude in inertial space and therefore, of the spacecraft-experiment end-to-end alignment.

This approach to controlling the alignment of the HEAO experiment package is recommended to simplify and reduce the costs of the CAOS.

Another, though less significant, effect of the HEAO mounting is the possibility of its interfering with the field-of-view of one or the other of two of the four S&C subsystem fixed head star trackers. This could only occur, however, if the HEAO is oriented normal to the CAOS X-axis at such times when the yoke is at a particular "roll" angle. This potential occultation of the star tracker field-of-view can be easily circumvented by rotating their mounting orientation away from the spacecraft aft direction. For example, a 15-deg rotation would move the FOV far enough to preclude occultation by a 25-ft long HEAO.

The same complement of standard S&C subsystem modules assigned to the LAOS (PE-146), augmented by a third Reaction Torque module and by a fixed head star tracker in the HEAO experiment package, satisfy the stabilization and control requirements of CAOS.

3.2 Communication, Data Processing, and Instrumentation Subsystem

The CDPI subsystem of the LAOS, described in LMSC PE-146, was designed to meet the maximum requirements of the HEAO, LST, LSO, and LRO missions, one at a time. It is, therefore, capable of meeting most of the requirements of the HEAO and LST together, and those of the LSO and LRO together. Where it fails to meet the requirements of CAOS fully, the CDPI subsystem designed for LAOS must be augmented as follows:

- The computer memory must be increased by the addition of an external package from 16K 24 bit words to 24K 24 bit words.

- The Low Data Rate Interface unit must be expanded to accommodate the greater quantity of data to be sampled, multiplexed, amplified and routed.

These increases in CDPI subsystem capability may be accomplished by the addition of a memory unit to the standard Data Processing module of the LAOS, and by the substitution of a modified Low Data Rate Interface Unit for the existing unit in the same module.

3.3 Electrical Power Subsystem

The Electrical Power subsystem of the LAOS, described in LMSC PE-146, may be augmented by the addition of solar array area and standard battery power modules to meet the requirements of CAOS. The maximum average electrical power load of the CAOS is estimated to be 2600 watts, requiring a 30% increase in the area of each of the two standard flexible solar power modules. This module is designed to accommodate such an increase without any changes in the structure and mechanisms of the module. Also, the 2600 watt average power load requires seven standard battery modules to maintain the depth-of-discharge of the batteries below 15%. To raise the reliability of the 587 electrical power subsystem eight standard battery modules are included, three more than the number of battery modules specified for the LAOS.

The standard Power Distribution module of the LAOS has sufficient power control circuitry and Regulator/Converter capability to meet the requirements of CAOS.

3.4 Attitude Control Subsystem

The Attitude Control subsystem of the LAOS, described in LMSC PE-146, is directly applicable to the CAOS. The four standard ACS modules together have sufficient total impulse to meet the requirements of CAOS; including drag makeup, momentum dumping, and backup stabilization for two years of orbital operations.